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Starship 1® Model 2000

(NC-4 and After)

Maintenance Manual

Volume 2 of 2
Chapter 30 thru Chapter 80

This maual includes the maintenance information required to be available by 14 CFR Part 23.

Note: This manual was formerly called the Beechcraft Starship Maintenance Manual.

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CHAPTER 30 - ICE AND RAIN PROTECTION

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WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The airplane is equipped with a variety of ice and rain protection systems that can be utilized during operation under inclement weather conditions.

AIRFOIL SURFACE DEICING (Effectivity: All)

Pneumatic deicer boots on the wings remove the formation of ice during flight. Regulated bleed air pressure and vacuum are cycled to the pneumatic boots for the inflation-deflation cycle. The system is automatically controlled by an electronic controller but can also be operated with manual control switches. Refer to Chapter 30-10-00 for description and maintenance of the system.

ENGINE AIR INTAKES (Effectivity: All)

Ice protection for the engine air inlet is provided by an inertial separation system utilizing an electrical actuator. Should the main electrical actuator fail, the ENG ANTI-ICE PWR SEL switch is placed in the STBY position to continue operating the vanes to the desired position. The leading edge lip of the engine air inlet is continuously anti-iced by engine P_3 bleed air when the firewall bleed air shutoff valve is open. Refer to Chapter 30-20-00 for description and maintenance of the air intake anti-icing systems.

PROPELLER DEICING (Effectivity: All)

The propellers are protected against icing by the engine exhaust.

WATER LINES (Effectivity: All)

Refer to Chapter 30-70-00 for information regarding the overboard drain anti-ice system. The system consists of a heater element, thermostat and relays which keep the drain mast from freezing. Thermal protection is provided to keep excess heat from being transferred to the fuselage skin.

DETECTION (Effectivity: All)

Chapter 30-80-00 provides information on the detection system that operates the airfoil surface deice system.

SPECIAL TOOLS AND RECOMMENDED MATERIALS (Effectivity: All)

The special tools and recommended materials listed in Charts 1 and 2 as meeting federal, military or supplier specifications are provided for reference only and are not specifically prescribed by Raytheon Aircraft Company. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Raytheon Aircraft Company, by the supplier, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to the usage of the product listed. This can be done by contacting the supplier of the product to be used.

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All)

TOOL NAME	PART NO.	MANUFACTURER	USE
1. Spray Gun	No. 7 with 63 head	Binks Mfg. Co., Chicago, IL	Apply adhesive to deicer boots.
	JGA 75X	DeVilbiss Co., Toledo, OH	Apply adhesive to deicer boots.
2. Carpenter's Chalk Line	N/A	Obtain locally.	To mark deicer boots for installation.
3. Rubber Hand Roller, Two-Inches Wide	P/N 74-451-74	B.F. Goodrich Akron,OH	Remove air bubbles under the deicer boot.
4. Viscosity Meter	No. 4 Ford Cup	Obtain locally.	Check the consistency of thinned adhesive.

CHART 2
RECOMMENDED MATERIALS (Effectivity: All)

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
1. Solvent (MEK)	TT-M-261	Methyl Eythyl Ketone	Obtain locally.
2. Solvent	TT-N-95	Naphtha	Obtain locally.
3. Adhesive	MIL-A-1154	EC1300L	Minnesota Mining Manufacturing Co., 900 Bush Ave., St. Paul. MN 55144
4. Cleaner	O-T-634	1,1,1, Trichlorethane	B.F. Goodrich, 500 S. Main Street, Akron, OH 44318
5. Primer		Bostik 100P	USM Chemical Company, Middleton, Mass. 01949
6. Solvent	TT-T-548C	Toluene	Obtain locally.
7. Silicone Coating		ICEX	B.F.Goodrich Akron, OH 44311
8. Tape		Mylar Y9133	Minnesota Mining and Manufacturing Co., 900 Bush Ave., St. Paul. MN 55144
9. Tape		Masking 200	Minnesota Mining and Manufacturing Co., 900 Bush Ave., St. Paul. MN 55144
10. Sealant		0165TS38 A/B	B.F. Goodrich Akron, OH 44311
11.Adhesive		A1503B/A1343B	B.F.Goodrich Akron, OH 44311
12. Deicer Boot Surface Repair Kit		74-451-S-2	B.F.Goodrich Akron, OH 44311
13. Silver Conductive Edge Sealant		0165RR26	B.F.Goodrich Akron, OH 44311

CHART 2
RECOMMENDED MATERIALS (Effectivity: All) *(Continued)*

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
14. Sealant	MIL-S-8802,Type II	Pro Seal 890	Pro-Seal Corp., Div. of Essex Chemical Corp., Aerospace Products, Compton, CA 90220
15. Deicer Boot Patch Kit		74-451-H	B. F. Goodrich, Akron, OH 44311
16. Sealing Compound		EC801CLA 1/2	Minnesota Mining Manufacturing Co., 900 Bush Ave., St. Paul. MN 55144
17. Edge Sealer Kit		74-451-K	B. F. Goodrich Akron, OH 44311

AIRFOIL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The airfoil surface deice system is automatically controlled by the surface deice controller (A215 PCB) (Figures 1 and 3) and the ice detection system (Chapter 30-80-00). The system is fully operational at the completion of the preflight checkout. The system is self-monitoring and has the capability of displaying system messages on the Engine Instrument, Crew Alerting System (EICAS) display on the pilot's instrument panel. Refer to Chapter 31-50-00 for information on the EICAS system and the messages that will appear on the display. The controls consist of the circuit breakers, test switches and the auto/manual control switches. See Figure 1 for the location of these components.

Pneumatic pressure is used to inflate the deicer boots on the forward and aft wings, thereby preventing the adhesion of ice particles. The surface deice system uses a dull-silver-coated pneumatic Estane deicer boot with faired in edges for improved laminar flow. The forward wing utilizes two deicer boots (one each side) and the aft wing utilizes six deicer boots (inboard, center and outboard for each side). Regulated pneumatic pressure is routed to the deicer system through orifices and check valves from both sides of the pneumatic system. Redundancy of the pneumatic pressure source is assured by crossovers in both the forward and aft wing deicer systems. Refer to Chapter 36-10-00 for detailed information on the pneumatic system.

AIRFOIL SURFACE DEICE SYSTEM (Effectivity: All)

SURFACE DEICE CONTROLLER (A215 PCB) (Effectivity: All) (Figures 1, 3 and 4)

The surface deice controller (A215 PCB) monitors and counts the icing signals received from the ice detection system (Chapter 30-80-00). When eight icing signals have been counted, the surface deice controller then actuates the distributor valves in the sequence shown in Figure 6. During operation of the deicing system, the surface deice controller monitors the signals from the pneumatic pressure switches and provides signals to the data acquisition unit (Chapter 31-40-00) which furnishes discrets to the EICAS for display to the pilots. The surface deice controller has the printed circuit boards that contain the electronic components required for automatic control of the airfoil surface deicer system.

AFT WING PNEUMATIC DEICE EQUIPMENT (Effectivity: All)

The aft wing pneumatic deice system utilizes two check valves, two orifice assemblies, a jet pump vacuum ejector, a separator, one distributor valve, one vacuum switch and two pressure switches on each wing. These components are located between the fuselage and the wing root rib.

FORWARD WING PNEUMATIC DEICE EQUIPMENT (Effectivity: All)

The forward wing utilizes two check valves, two orifice assemblies, two distributor valves, two pressure switches, two vacuum switches, two jet pump vacuum ejectors and a crossfeed valve mounted forward of the forward pressure bulkhead on the lower right side.

DISTRIBUTOR VALVES (Effectivity: All)

The distributor valves are flow control valves and have two solenoid-operated valves each. The solenoid valve alternates the application of pressure or vacuum to the deicer boots. The valves are normally in the vacuum mode (spring-loaded to that position). Any time the engines are running and the firewall bleed air shutoff valves are open, a vacuum is placed on the

boots. The forward wing distributor valves use only one boot port each since each operates only one boot. The boot port A of the aft wing distributor valves operates the center and outboard boots on each wing and port B operates the inboard boot on each wing. When the valve is switched from the pressure mode to the vacuum mode, the boot pressure is vented overboard through an exhaust tube. The vacuum ejector then removes all remaining air from the boot. The distributor valve is equipped with an electric heater circuit to prevent moisture from freezing and hindering operation. See Figure 5 for the heater circuit.

SEPARATORS (Effectivity: All)

The separators in the aft wing system collect water that may enter the system. During the vacuum cycle of the aft wing distributor valves, the water will drain down and go out the vacuum ejector exhaust. At the forward wing deice system, any water in the system will drain downward and go out through the vacuum ejectors exhaust.

VACUUM EJECTORS (Effectivity: All)

Pneumatic pressure passing through the venturi of the vacuum ejector creates the vacuum necessary to remove all of the air from the boots and cause them to lay flat against the leading edge of the wing. The pneumatic pressure passing through the ejector is vented overboard.

SURFACE DEICE CROSSFEED VALVE
(Effectivity: All)

The normally closed crossfeed valve (E247) is installed just forward of the forward pressure bulkhead between the left and right pneumatic system lines to the forward wing distributor valves. This valve is opened when the BLEED AIR CONTROL SWITCH (A175) is in the EMER, L or R position. This valve would be opened in the event that pneumatic pressure is lost on one side.

SURFACE DEICE TEST SWITCHES
(Effectivity: All)
(Figures 1 and 4)

The SURF DEICE test switches (VAC, STBY and MAIN) are the press-to-test type and are pressed momentarily (approximately six seconds) to check that the system is operational. The VAC switch checks the vacuum switches which are used only for test purposes. The MAIN and STBY switches check the main and standby ice detectors. The test switches are normally used during the preflight check with the engines running. Testing the standby detection system first assures that the deice system will be operating on the main detection system at the completion of the preflight checkout. Refer to Section IV Normal Procedures, ICING FLIGHT, BEFORE TAKEOFF (RUNUP) of the Starship 1 FAA Approved Airplane Flight Manual for the messages that will appear on the EICAS display when the test switches are pressed.

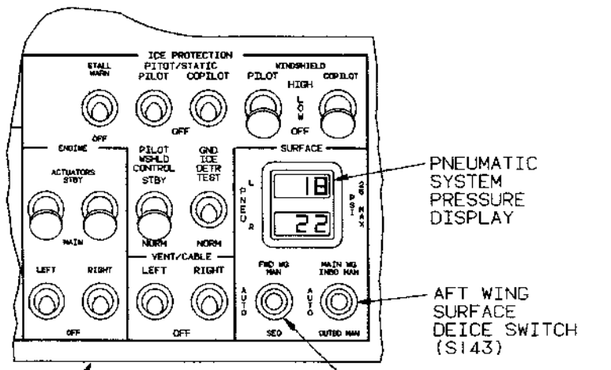
SURFACE DEICE CONTROL SWITCHES
(Effectivity: All)
(Figure 1)

The forward and aft wing surface deice control switches (Detail F) are normally in the AUTO position which permits automatic operation of the deice system by the surface deice controller and the ice detection system (Chapter 30-80-00). When the switches are held in the manual position, the boots will be inflated for as long as the switch is held in that position. The MAIN WG INBD MAN position inflates the inboard boots of both aft wings. The OUTBD MAN position inflates the center and outboard boots of both aft wings. The FWD WG MAN position inflates both boots on the forward wings. The SEQ position will cycle all of the boots as if the ice detector had signaled for a normal cycle shown in Figure 6. The SEQ position does not have to be held in the actuated position to complete the cycle.

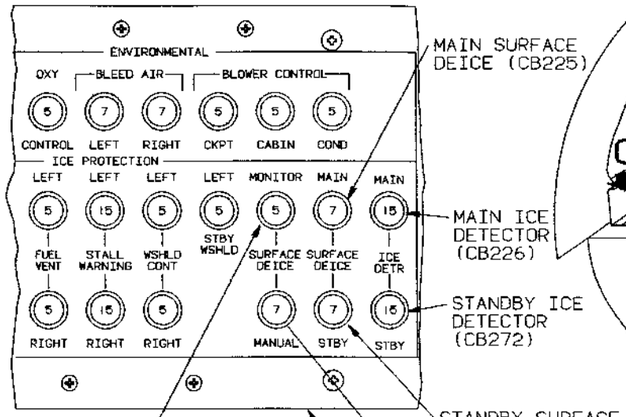
NORMAL DEICING OPERATION
(Effectivity: All)

All deicer boots will be held in a deflated position during normal (non-icing) flight conditions with vacuum provided by the four vacuum ejectors, one each in the left forward wing, right forward wing, left aft wing and the right aft wing.

As a flight proceeds into icing conditions, both ice detectors (Chapter 30-80-00) will accumulate ice on the vibrating probe. With the system operating normally (i.e. no faults detected in the main or standby components), either ice detector(main or standby) will provide the signal for the EICAS display ICING message when a thickness of .020 ± .005 inch of ice has accumulated on the ice detector probe. If all the ice protection systems are on and operating, the ICING display will be white. If any of the ice protection systems are not on, the ICING display will be yellow. The deicing heater signal of the ice detector probe is used by the monitor/logic controller of the surface deice



DETAIL F



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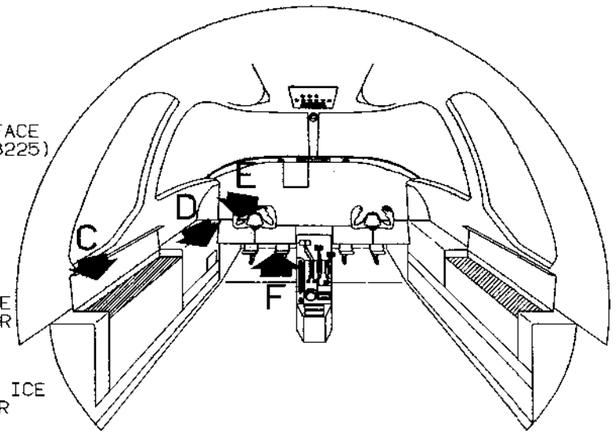
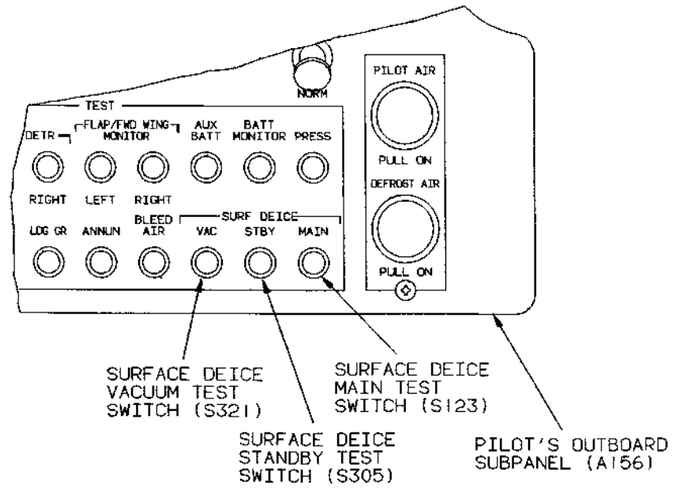


DETAIL A



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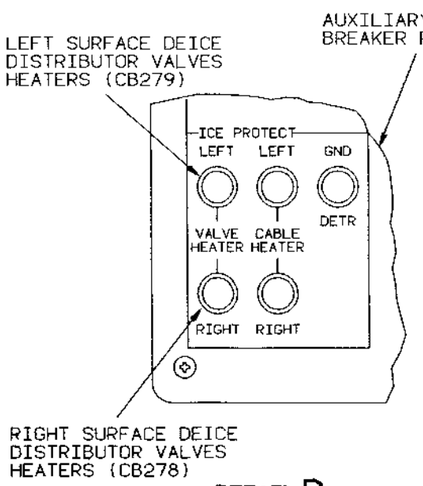

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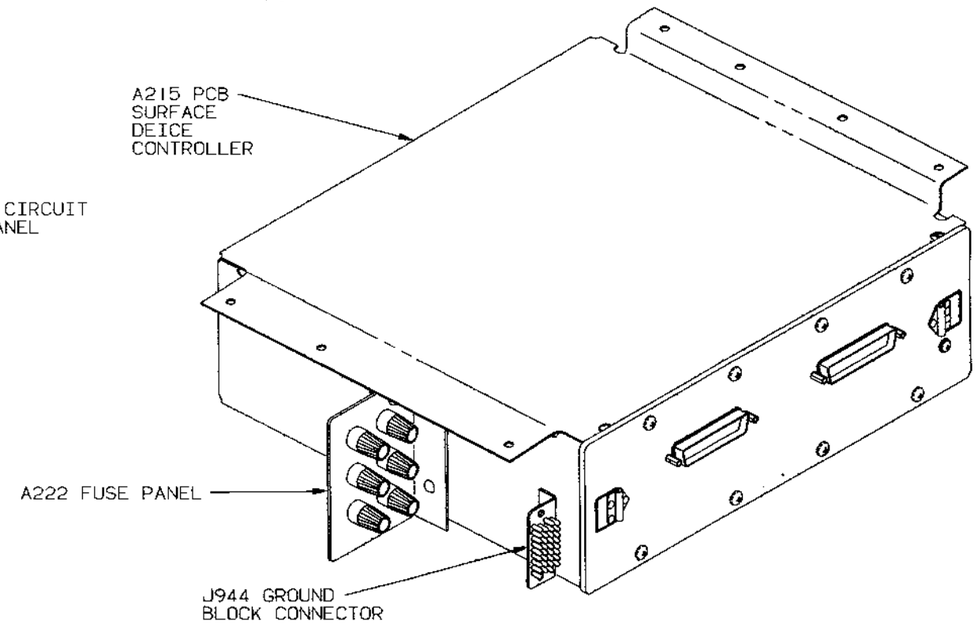
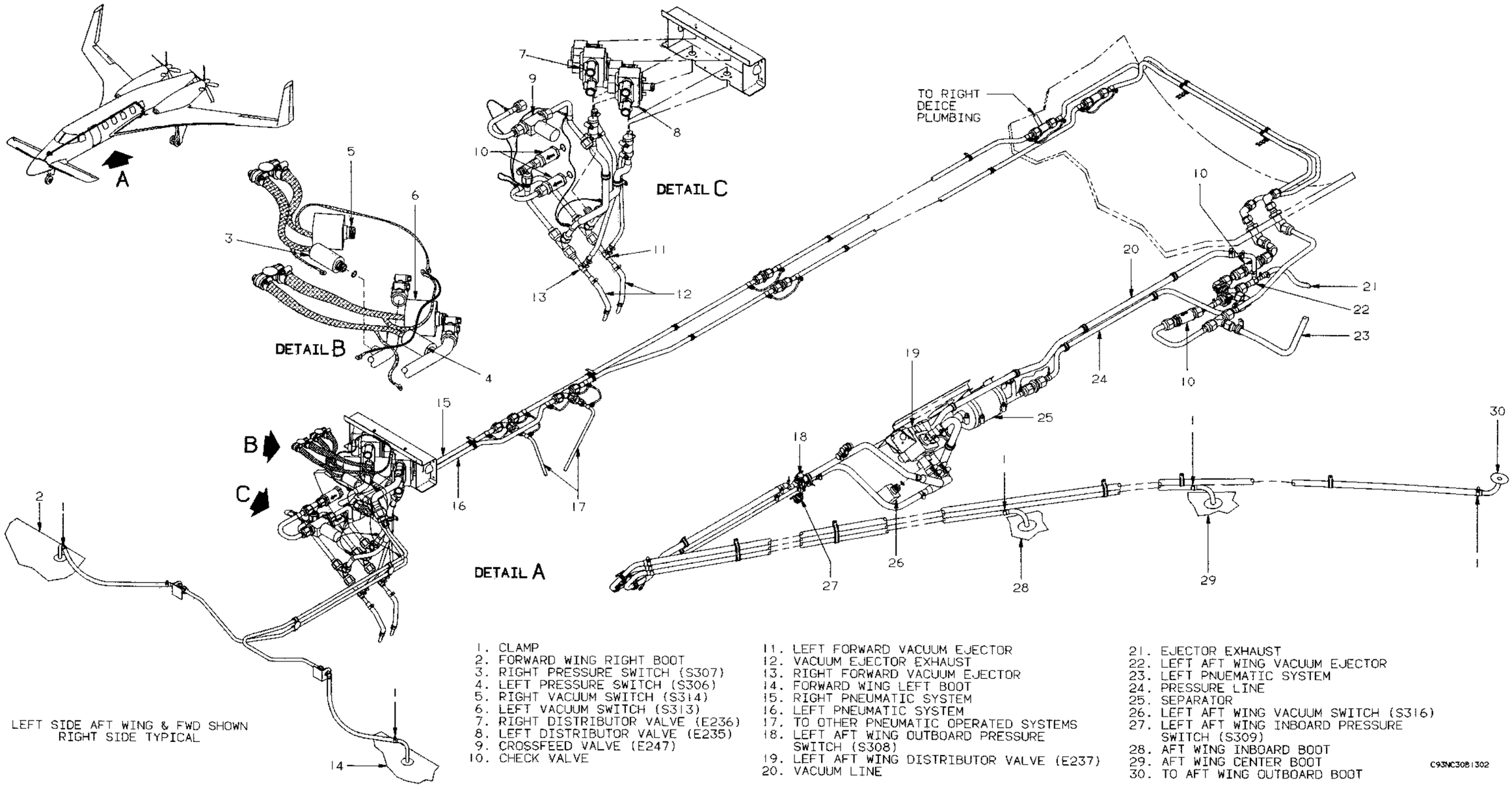


DETAIL B

C93NC30B4454 C

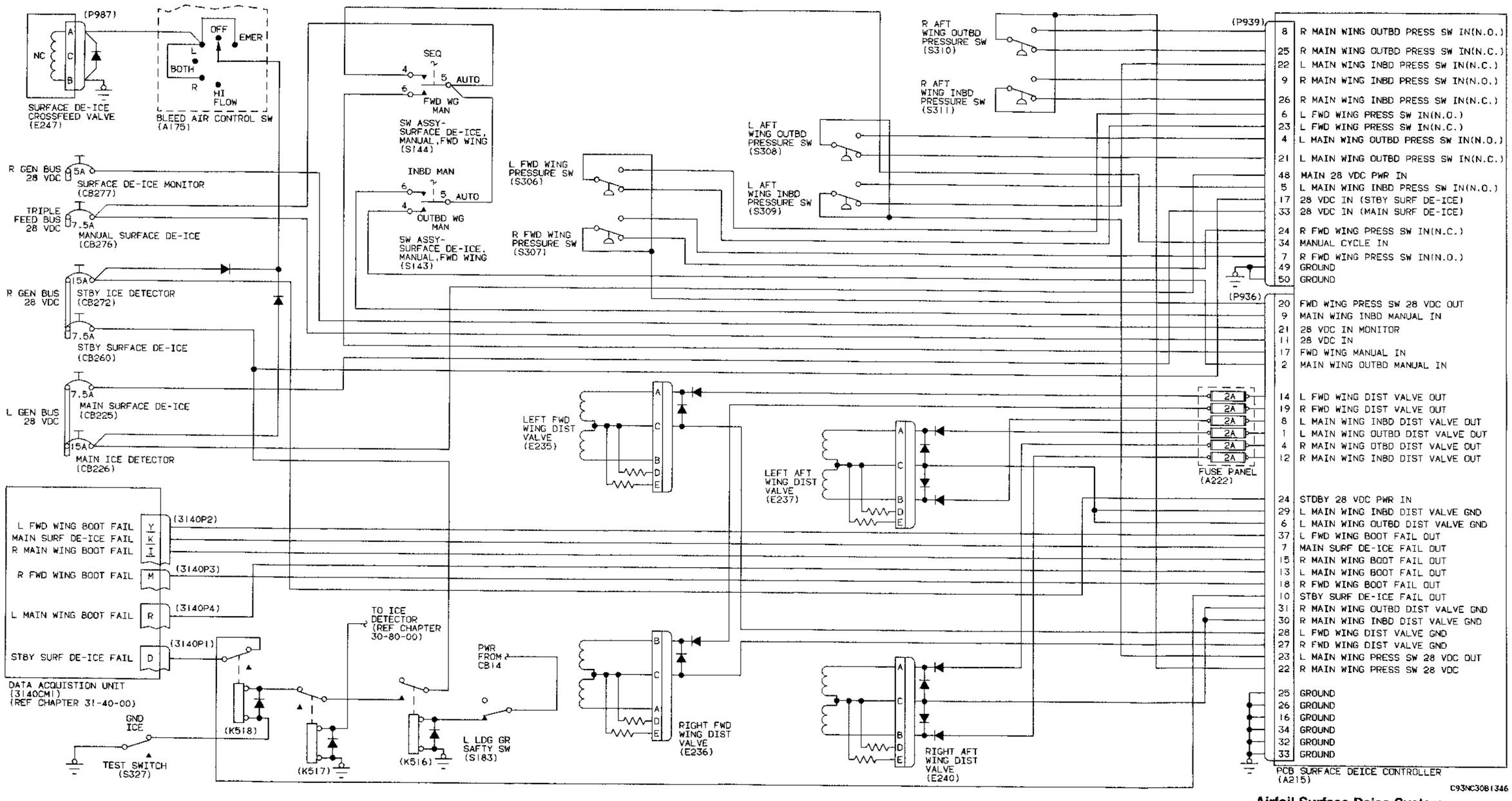
Airfoil Surface Deice System Controls
(Effectivity: All)
Figure 1



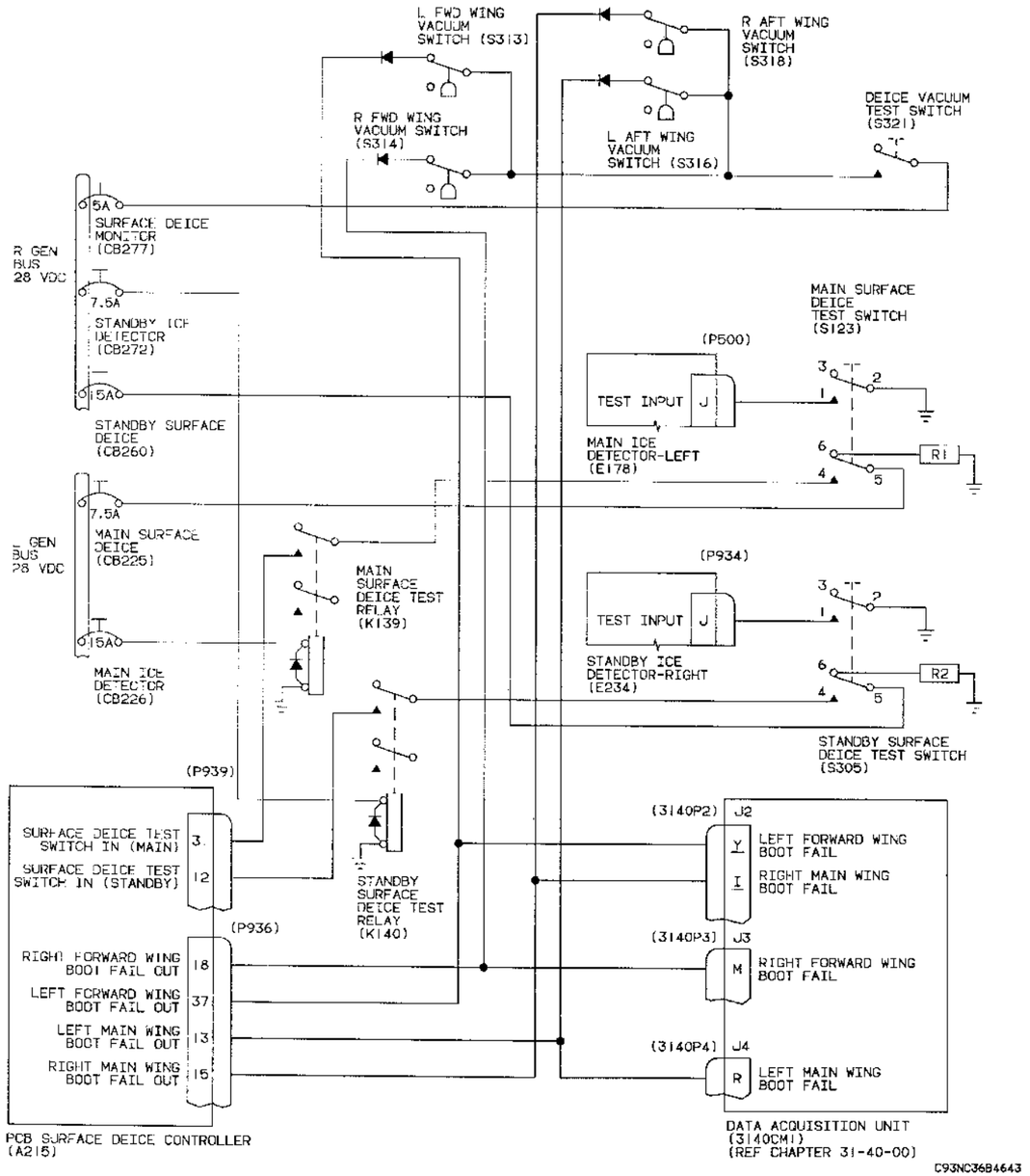
1. CLAMP
2. FORWARD WING RIGHT BOOT
3. RIGHT PRESSURE SWITCH (S307)
4. LEFT PRESSURE SWITCH (S306)
5. RIGHT VACUUM SWITCH (S314)
6. LEFT VACUUM SWITCH (S313)
7. RIGHT DISTRIBUTOR VALVE (E236)
8. LEFT DISTRIBUTOR VALVE (E235)
9. CROSSFEED VALVE (E247)
10. CHECK VALVE
11. LEFT FORWARD VACUUM EJECTOR
12. VACUUM EJECTOR EXHAUST
13. RIGHT FORWARD VACUUM EJECTOR
14. FORWARD WING LEFT BOOT
15. RIGHT PNEUMATIC SYSTEM
16. LEFT PNEUMATIC SYSTEM
17. TO OTHER PNEUMATIC OPERATED SYSTEMS
18. LEFT AFT WING OUTBOARD PRESSURE SWITCH (S308)
19. LEFT AFT WING DISTRIBUTOR VALVE (E237)
20. VACUUM LINE
21. EJECTOR EXHAUST
22. LEFT AFT WING VACUUM EJECTOR
23. LEFT PNUEMATIC SYSTEM
24. PRESSURE LINE
25. SEPARATOR
26. LEFT AFT WING VACUUM SWITCH (S316)
27. LEFT AFT WING INBOARD PRESSURE SWITCH (S309)
28. AFT WING INBOARD BOOT
29. AFT WING CENTER BOOT
30. TO AFT WING OUTBOARD BOOT

C93NC30B1302

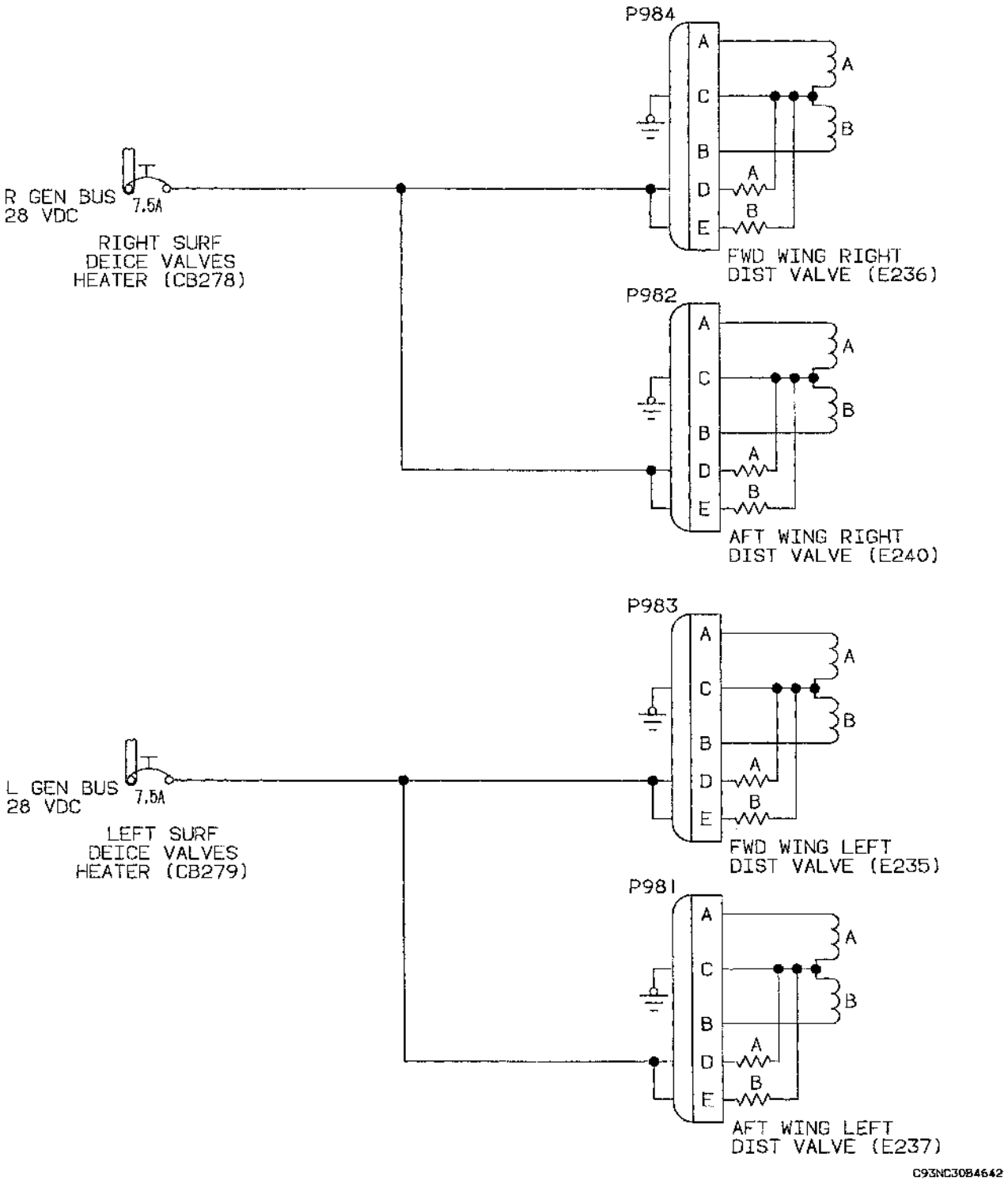
Airfoil Surface Deice System Equipment
(Effectivity: All)
Figure 2



Airfoil Surface Deice System
Electrical Controls Schematic
(Effectivity: All)
Figure 3



Airfoil Surface Deice System Electrical Test Schematic
(Effectivity: All)
Figure 4

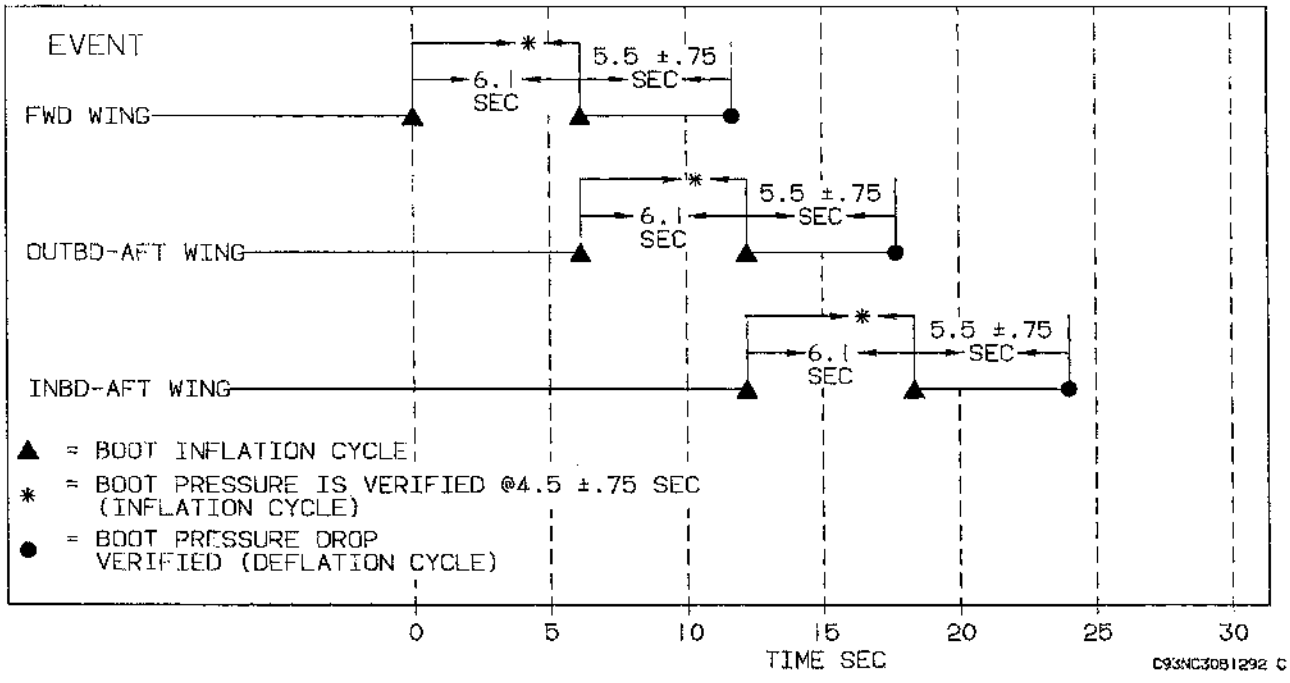


Distributor Valve Heater Circuits Schematic
(Effectivity: All)
Figure 5

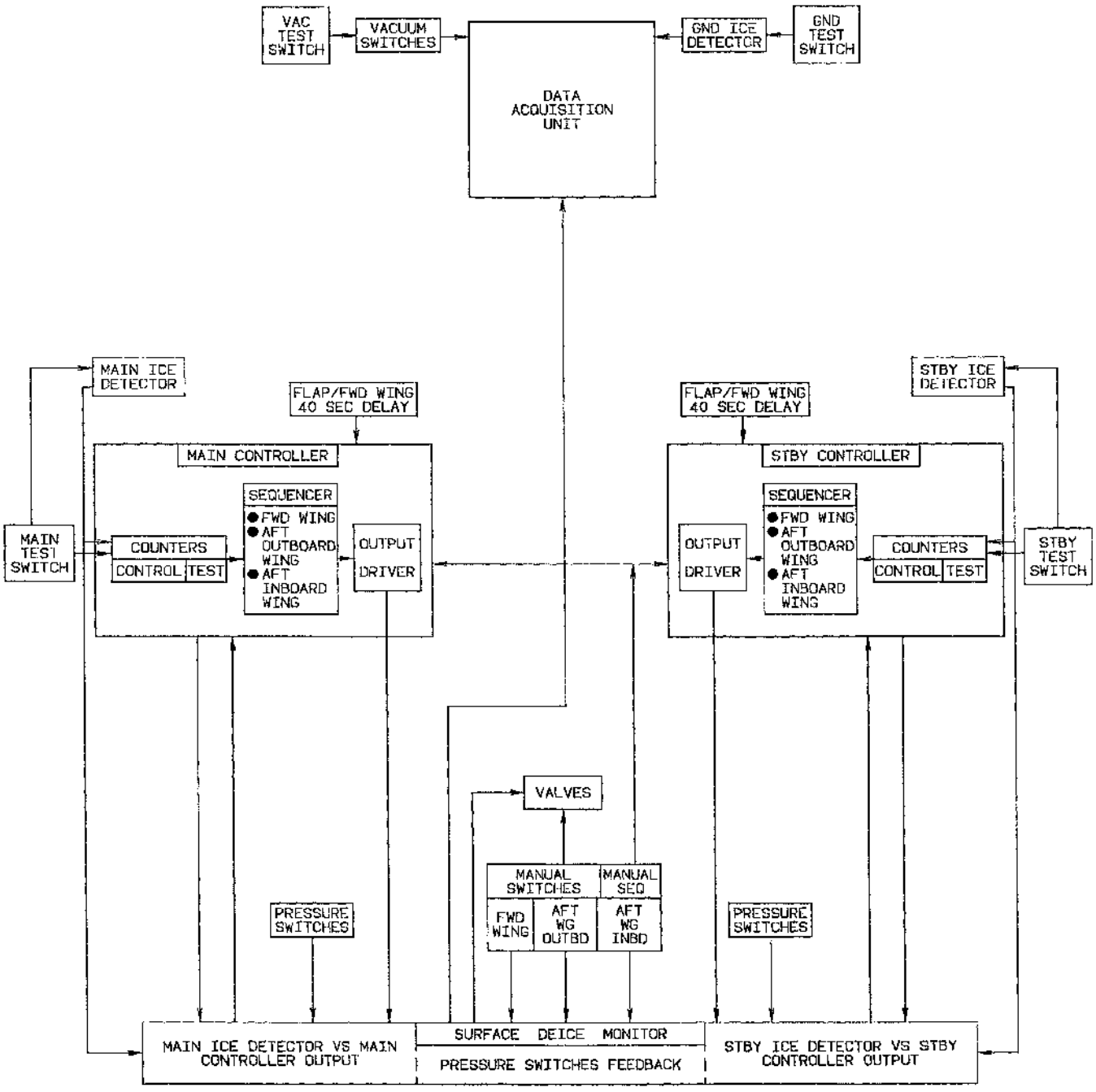
controller (Figure 7) to initiate deicer boot sequencing. The probe deice heater will turn on simultaneously with the output signal for the ICING message and remain on for 5 ± 1 seconds after the ice on the probe has melted. The main monitor/logic controller will note that the deice heater cycle has started and will count one icing cycle. The probe cools and the icing cycles are repeated when sufficient ice builds up. If the subsequent .020-inch buildup occurs within one minute, the EICAS display ICING message will be continuous.

After eight icing cycles of the detector have been counted, the main monitor/logic controller (Figure 7) actuates the forward wing deicer distributor valves to inflate the forward wing deicer boots for six seconds (Figure 6) and resets the counter to zero counts. If the the deicer boots have not reached 18 psi within four to five seconds after valve actuation as sensed by the appropriate pressure switches, an EICAS message of L FWD WG BOOT FAIL or R FWD WG BOOT FAIL will be displayed. After six seconds of elapsed time, the distributor valves are signaled to revert to vacuum on the forward wing deicer boots. If the forward wing boots pressure has not been reduced to 16 psi within six seconds after valve deactuation as sensed by the pressure switches, an EICAS message of L FWD WG BOOT FAIL or R FWD WG BOOT FAIL will be displayed. At the same time that the forward wing distributor valves are signaled to revert to vacuum, the main monitor/logic controller actuates the aft wing deicer distributor valves port A to inflate the outboard and center deicer boots for six seconds. Timing, pressure sensing, EICAS messages (L WG BOOT FAIL or R WG BOOT FAIL) and deflation is the same as for the forward wing boots. Port B of the distributor valves for the aft wing inboard deicers are then actuated to inflate the inboard deice boots for six seconds during the transition from pressure to vacuum by the outboard and center boots of the aft wing. Timing, pressure sensing, EICAS messages (L WG BOOT FAIL or R WG BOOT FAIL) and deflation is the same as for the forward wing boots.

If forward wing movement is started during any deicer boot inflation cycle, the monitor/logic controller will immediately deflate all boots. The system inflation capabilities are stopped for 40 seconds to avoid interference of the deicer boots with the nose structure. At



Deice Boot Inflation/Deflation Cycle Time Intervals
(Effectivity: All)
Figure 6



Controller Logic Circuit Schematic
(Effectivity: All)
Figure 7

the end of the 40-second time delay, the monitor/logic controller will reset the icing cycle counter to zero counts and sequence the boots beginning with the forward wings. The 40-second time delay is initiated with each crew selection of flap/forward wing movement whether in icing conditions or not. Should flap/forward wing movement be in progress when the ice detector registers the eighth icing cycle count, boot inflation will be stopped for the remainder of the 40 seconds at which time normal sequencing will begin.

When in icing conditions, all components of the main deice control system are continuously self-tested (ice detectors, monitor/logic controller, counter, sequencer, etc.). In flight in dry air, only the ice detectors are continuously self-tested since icing cycles do not occur; the pressure switches are also monitored for inadvertent actuation.

Should the flight crew desire to override the automatic main monitor/ logic controller, they can manually inflate all of the boots sequentially with the operation of the SEQ position of the forward wing surface deice and sequence switch or they can selectively inflate any of the three boot groupings with the operation of the FWD WG MAN, MAIN WG INBD MAN or MAIN WG OUTBD MAN position.

DEICER BOOT CONTROL FAILURE MODE OPERATION (Effectivity: All)

During flight operations, display of the EICAS message MAIN DEICE FAIL indicates a fault has been detected in the main ice detector (Chapter 30-80-00) if the flight is in dry air. If the flight is in icing conditions, the MAIN DEICE FAIL message denotes a fault in the main ice detector or in the main monitor/logic controller of the surface deice controller. In the event of this happening, the system will be automatically switched to the standby ice detector and the standby monitor/logic controller to sequence the boots and reset the counter to 0 counts. The integrity of the surface deice system is not compromised when operating in the standby mode as all monitoring, sequencing and override circuits are duplicated.

Display of the EICAS message STBY DEICE FAIL, in conjunction with the MAIN DEICE FAIL message, indicates that faults may have been detected in both ice detectors or monitor/logic controllers. In the event of this happening, the automatic mode of the surface deice system and the SEQ position of the forward wing surface deice switch are inoperative, thus requiring the flight crew to operate the system in the manual control mode. Any of the three boot groupings can be manually inflated by the operation of the manual position on the forward and aft surface deice switches. To insure adequate pressure is being supplied to the boots for ice removal, the momentary manual positions for each of the three boot groupings should be selected individually and operated for approximately six seconds. As each switch is selected, the appropriate EICAS message L FWD BOOT FAIL, R FWD BOOT FAIL, L WG BOOT FAIL or R WG BOOT FAIL will be displayed and then extinguished as each boot grouping inflates to the required deice pressure.

If one of the pneumatic system pressure regulators fails, it is designed to fail in the open position. A relief valve in the pressure regulator will maintain pneumatic outlet pressure at 30 ± 1 psi as the system continues functioning. Pneumatic pressure regulator failure would be noted by the flight crew due to higher pressures displayed on the digital pneumatic pressure display. Air supply line breakage resulting in air pressures below 10 ± 0.5 psi would result in: (1) low pneumatic pressure digital display readings, and (2) displaying of the EICAS message PNEU PRESS LOW. Dual source plumbing allows for the surface deice system to function normally with one pressure source inoperative.

The failure of one boot to inflate will result in the loss of deicing capability only in the failed boot when located on either side of the forward wing or on either inboard side of the aft wing. Should one of the center or outboard boots of the aft wing fail to inflate, both boots on the side of the failed boot will lose deicing capability, but all other boots will continue to operate. If pressure switch feedback to the monitor/logic controller indicates that any of the six pressure switches did not reach the pressure required for boot operation within four to five seconds after the associated distributor valve has been actuated, the appropriate EICAS message (L FWD BOOT FAIL, R FWD BOOT FAIL, L WG BOOT FAIL or R WG BOOT FAIL) will be displayed.

AIRFOIL - TROUBLESHOOTING (Effectivity: All)

Troubleshooting the Starship 1 airfoil surface deicer system consists of performing the BEFORE TAKEOFF (RUNUP) checklist in Section IV of the Starship 1 FAA Approved Airplane Flight Manual.

AIRFOIL - MAINTENANCE PRACTICES (Effectivity: All)

AFT WING DISTRIBUTOR VALVE REMOVAL (Effectivity: All) (Figure 8)

Figure 8 shows the aft wing left side distributor valve. This illustration is typical for the right side valve which is installed in the opposite direction as the left valve.

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallatiom.

a. Remove all electrical power from the airplane.

b. Remove the forward (527BB or 627BB) and center (527CB or 627CB) fairing panels between the fuselage and wing as instructed in Chapter 53-50-00.

c. Disconnect the electrical connector (18) from the distributor valve (2).

d. Remove the vacuum switch as instructed in the AFT WING VACUUM SWITCH REMOVAL procedure in this section.

e. Loosen the clamps (12 and 14) at both ends of the distributor valve boot port tubes (13) and remove the tubes.

f. Remove the screw connecting the clamps (9) on the distributor valve exhaust tube (8) and the inlet pressure tube (4). Loosen the clamp (10) and remove the exhaust tube (8).

g. Loosen the clamps (3 and 11) on the inlet vacuum and pressure tubes (4 and 6).

h. Remove four screws and washers (16) attaching the distributor valve mounting brackets (1) to the fuselage bracket (5).

i. Separate the distributor valve (2) and brackets (1) from the inlet vacuum tube (6) and the inlet pressure tube (4). Remove the distributor valve (2) and brackets (1) from the airplane.

j. Remove two screws (17) to remove each bracket (1) from the distributor valve (2).

k. Cap all tubes and plug all ports to prevent contamination from entering the system.

AFT WING DISTRIBUTOR VALVE INSTALLATION (Effectivity: All) (Figure 8)

CAUTION

The distributor valve must meet electrical bonding requirements specified in Chapter 20-00-03.

a. Attach the mounting brackets (1) to each side of the distributor valve (2) with two screws (17) each.

b. Position the distributor valve (2) and brackets (1) on the fuselage bracket (5) and engage the inlet ports with the inlet vacuum tube (6) and the pressure tube (4). Attach the distributor valve brackets to the fuselage bracket (5) with four screws (16).

c. Tighten the clamps (3 and 11) on the inlet vacuum tube (6) and the pressure tube (4).

d. Install the exhaust tube (8) on the distributor valve (2) with clamp (10). Install the screw connecting the clamps (9) on the exhaust tube (8) and the inlet pressure line (4).

e. Install the tubes (13) to the boot ports of the distributor valve (2) with clamps (12) and to the boot tubes (15) with clamps (14).

f. Install the vacuum switch as instructed in the AFT WING VACUUM SWITCH INSTALLATION procedure in this section.

g. Connect the distributor valve electrical connector (18).

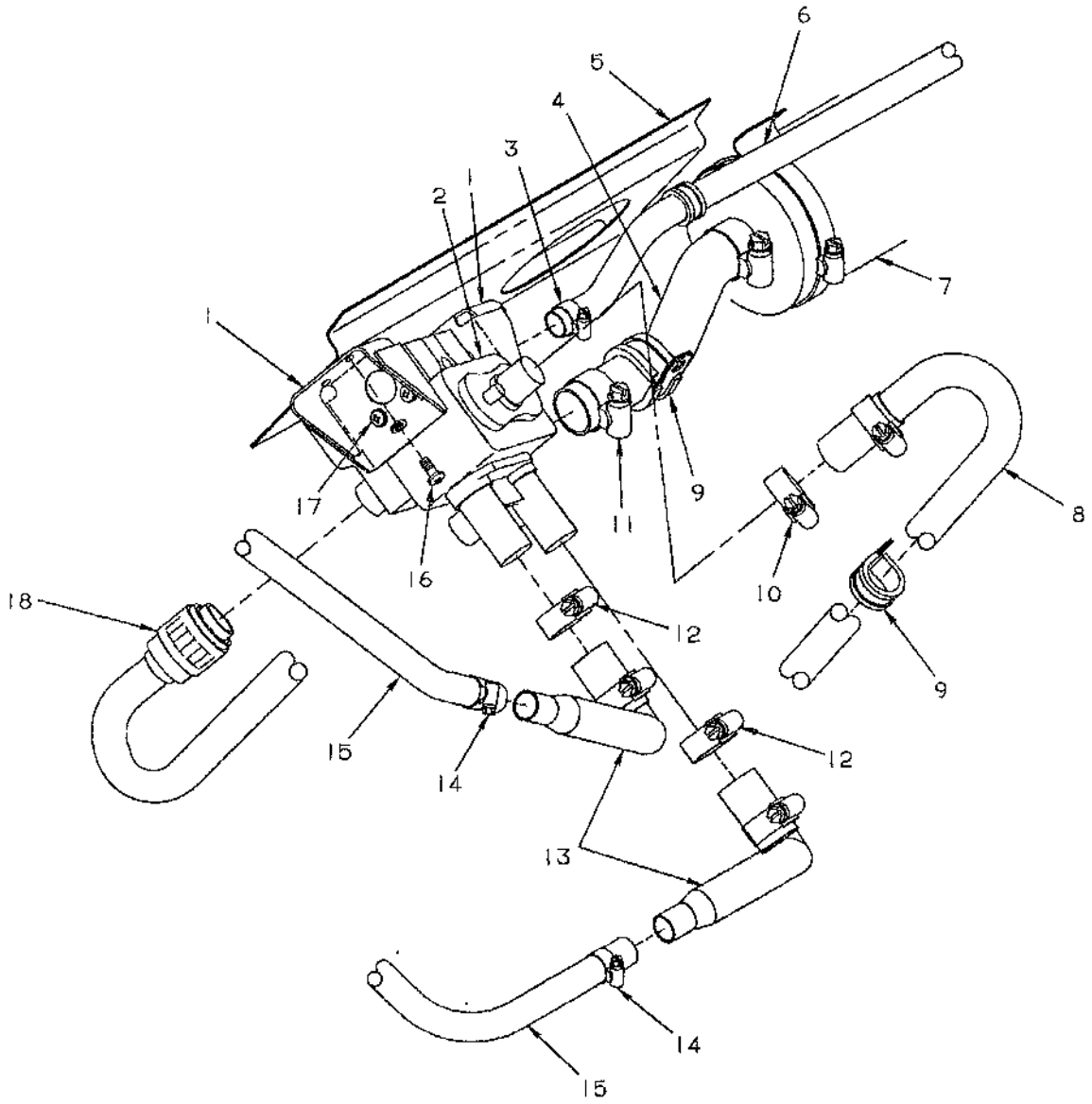
h. Install the forward (527BB or 627BB) and center (527CB or 627CB) fairing panels between the fuselage and the aft wing as instructed in Chapter 53-50-00.

FORWARD WING DISTRIBUTOR VALVE REMOVAL (Effectivity: All) (Figure 9)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallatiom.

a. Remove all electrical power from the airplane.



1. VALVE MOUNTING BRACKET (2)
2. DISTRIBUTOR VALVE
3. CLAMP
4. INLET PRESSURE TUBE
5. FUSLAGE BRACKET
6. INLET VACUUM TUBE
7. SEPARATOR
8. EXHAUST TUBE
9. CLAMPS
10. CLAMP
11. CLAMP
12. CLAMP
13. BOOT PORT TUBES
14. CLAMP
15. VACUUM/PRESSURE TUBES TO BOOTS
16. SCREW AND WASHER (4)
17. SCREW AND WASHER (4)
18. ELECTRICAL CONNECTOR

C93NC30B1291 C

Aft Wing Distributor Valve Installation (Effectivity: All)
Figure 8

b. Remove the right nose fairing (212BR as instructed in Chapter 53-50-00.

c. Remove the vacuum switches as instructed in the VACUUM SWITCH REMOVAL procedure in this section.

d. Remove the pressure switches as instructed in the PRESSURE SWITCH REMOVAL procedure in this section.

e. Remove the A and B boot port tubes (7 and 10).

f. Loosen the inlet pressure and vacuum tubing clamps (8) below the distributor valve bracket (4).

g. Remove the six bracket mounting screws (5) and washers.

h. Remove the valve mounting bracket (4) and distributor valves (1 and 2) by lifting up to disconnect the inlet pressure and vacuum tubes (9).

i. Remove the distributor valves (1 and 2) from the mounting bracket (4) by removing four screws (3) each.

j. Plug or cap all open ports and tubes to prevent the entry of contamination into the system.

FORWARD WING DISTRIBUTOR VALVE INSTALLATION (Effectivity: All) (Figure 9)

CAUTION

The distributor valve must meet electrical bonding requirements specified in Chapter 20-00-03.

a. Install the distributor valves (1 and 2) in the mounting bracket (4) by inserting the inlet vacuum port through the hole in the bracket (4). Secure the distributor valves to the bracket with two screws (3) each top and bottom.

b. Remove all caps and plugs from ports and tubes.

c. Install the bracket (4) with valves (1 and 2) in the airplane, engaging the valve inlet ports with the inlet pressure and vacuum tubes (9). Attach the valve bracket (4) to the bulkhead with six screws (5), lock washers and flat washers.

d. Tighten the clamps (8) on the inlet pressure and vacuum tubes (9).

e. Install the A and B boot port tubes (7 and 10). All unused boot ports must be capped or plugged.

f. Install the pressure switches as instructed in the PRESSURE SWITCH INSTALLATION procedure in this section.

g. Install the vacuum switches as instructed in the VACUUM SWITCH INSTALLATION procedure in this section.

h. Connect all ground jumpers that were disconnected.

i. Install the right nose fairing as instructed in Chapter 53-50-00.

FORWARD WING VACUUM SWITCH REMOVAL (Effectivity: All) (Figure 10)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove all electrical power from the airplane.

b. Remove the right nose fairing (212BR) as instructed in Chapter 53-50-00.

c. Remove clamp (7) and disconnect the electrical connector (8).

d. Disconnect the ground jumper (9).

e. Unscrew the vacuum switch (10 or 11) from the port on the distributor valve and then plug the port on the valve to prevent the entry of contamination into the system.

f. Remove the O-ring (3) and discard it.

FORWARD WING VACUUM SWITCH INSTALLATION (Effectivity: All) (Figure 10)

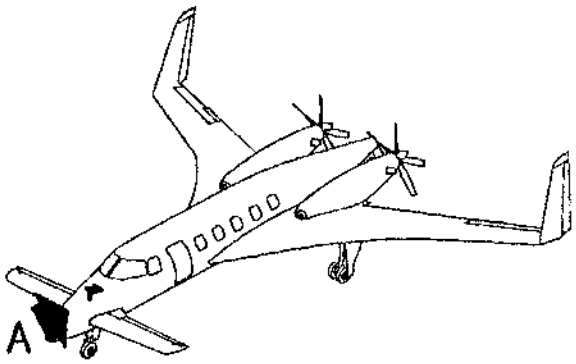
CAUTION

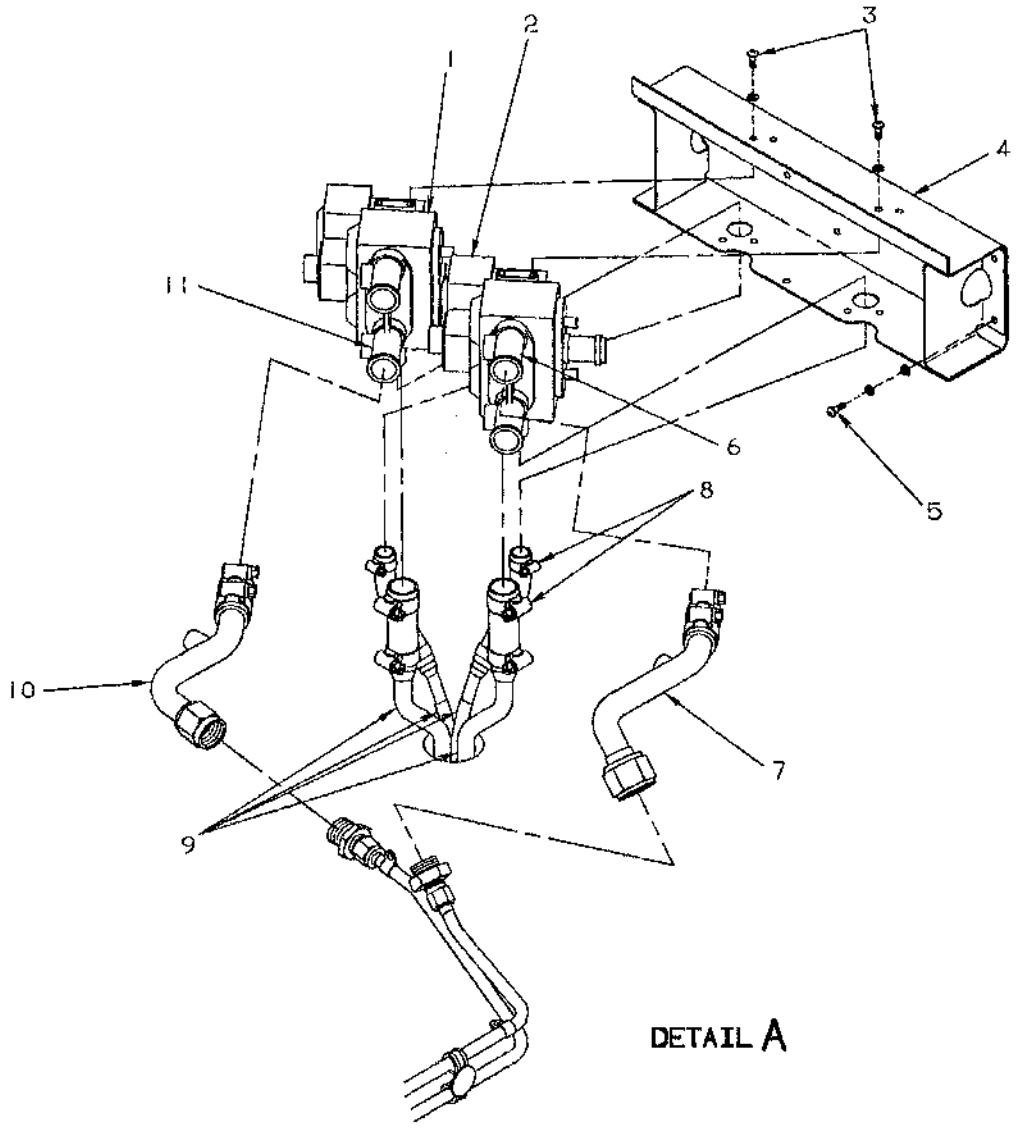
The vacuum switch must meet electrical bonding requirements specified in Chapter 20-00-03.

a. Remove the plug from the port in the distributor valve and install the vacuum switch (10 or 11) with a new O-ring (3).

b. Connect the ground jumper (9) and electrical connector (8).

1. RIGHT SYSTEM DISTRIBUTOR VALVE
2. LEFT SYSTEM DISTRIBUTOR VALVE
3. VALVE MOUNTING SCREWS (4 EACH VALVE)
4. VALVE MOUNTING BRACKET
5. BRACKET MOUNTING SCREWS (6)
6. BOOT PORT A
7. BOOT PORT A TUBE
8. CLAMP
9. INLET PRESSURE AND VACUUM TUBES
10. BOOT PORT B TUBE
11. BOOT PORT B





Forward Wing Distributor Valve Installation
(Effectivity: All)
Figure 9

c. Secure the electrical connector (8) with a clamp (7).

d. Install the right nose fairing (212BR) as instructed in Chapter 53-50-00.

AFT WING VACUUM SWITCH REMOVAL (Effectivity: All) (Figure 10)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallatiom.

a. Remove all electrical power from the airplane.

b. Remove the forward (527BB or 627BB) and center (527CB or 627CB) fairing panels between the fuselage and the aft wing as instructed in Chapter 53-50-00.

c. Remove clamp (1) and disconnect the vacuum switch electrical connector (2) shown in Detail A for the left vacuum switch (4) or Detail B for the right vacuum switch (6).

d. Disconnect the ground jumper (5).

e. Unscrew the vacuum switch (4 or 6) from the port in the distributor valve and then plug the port to prevent the entry of contamination into the system.

f. Remove the O-ring (3) and discard it.

AFT WING VACUUM SWITCH INSTALLATION (Effectivity: All) (Figure 10)

CAUTION

The vacuum switch must meet electrical bonding requirements specified in Chapter 20-00-03.

a. Remove the plug from the port in the distributor valve and install the vacuum switch (4 or 6) with a new O-ring (3).

b. Connect the ground jumper (5) and electrical connector (2) as shown in Detail A for the left vacuum switch (4) or Detail B for the right vacuum switch (6).

c. Secure the electrical connector (2) with a clamp (1).

d. Install the fairing panels between the fuselage and aft wing as instructed in Chapter 53-50-00.

FORWARD WING PRESSURE SWITCH REMOVAL (Effectivity: All) (Figure 11)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallatiom.

a. Remove all electrical power from the airplane.

b. Remove the right nose fairing (212BR) as instructed in Chapter 53-50-00.

c. Remove the clamp (10) and disconnect the pressure switch electrical connector (11) shown in Detail C.

d. Disconnect the ground jumper (14).

e. Unscrew the pressure switch (12 or 13) from the tubing port and remove it from the airplane.

f. Cap the open port to prevent contamination from entering the system.

g. Remove the O-ring (4) and discard it.

FORWARD WING PRESSURE SWITCH INSTALLATION (Effectivity: All) (Figure 11)

CAUTION

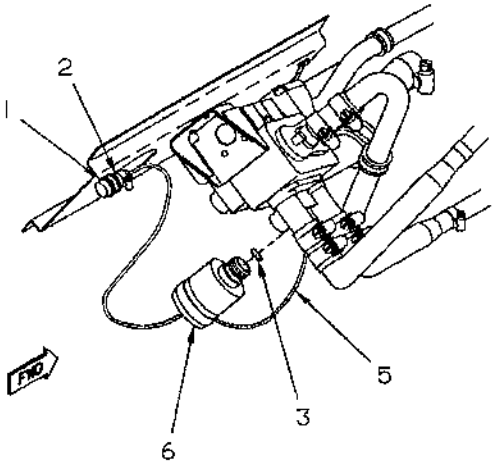
The pressure switch must meet electrical bonding requirements specified in Chapter 20-00-03.

a. Remove the cap from the tubing port and install the pressure switch (12 or 13) with a new O-ring (4).

b. Connect the electrical connector (11) shown in Detail C and secure it with a clamp (10).

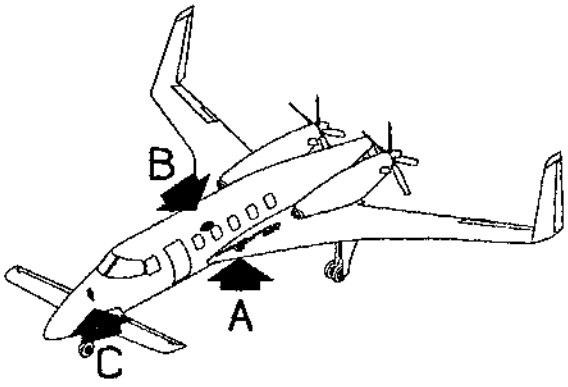
c. Connect all of the ground jumpers (14).

d. Install the right nose fairing as instructed in Chapter 53-50-00.

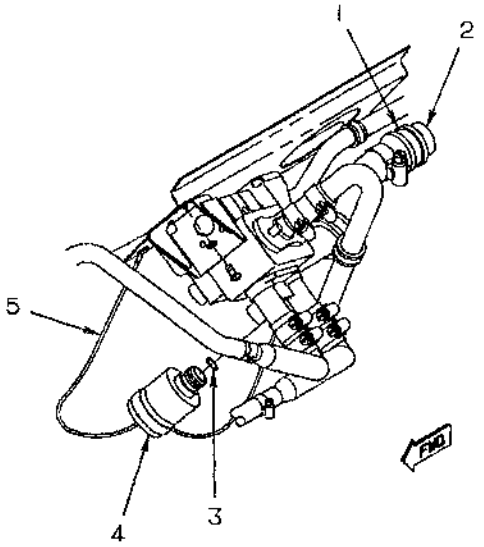


AFT WING, RIGHT SYSTEM

DETAIL B

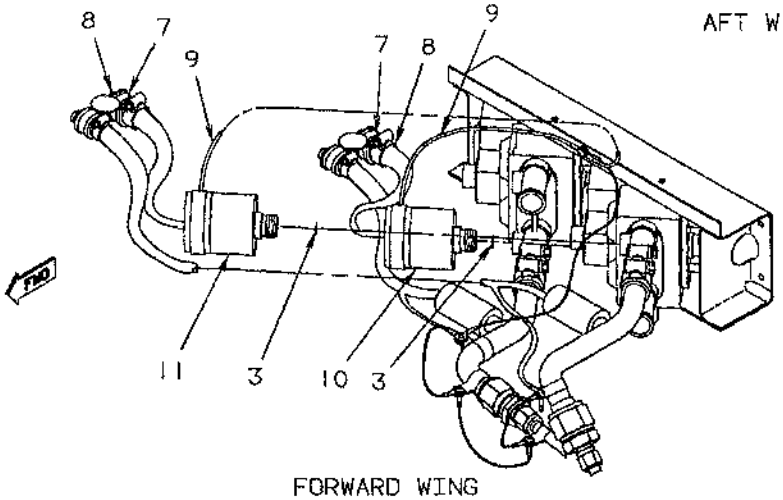


1. CLAMP
2. ELECTRICAL CONNECTOR
3. O-RING
4. LEFT VACUUM SWITCH (S316)
5. GROUND JUMPER
6. RIGHT VACUUM SWITCH (S318)
7. CLAMP
8. ELECTRICAL CONNECTOR
9. GROUND JUMPER
10. RIGHT VACUUM SWITCH (S314)
11. LEFT VACUUM SWITCH (S313)



AFT WING, LEFT SYSTEM

DETAIL A

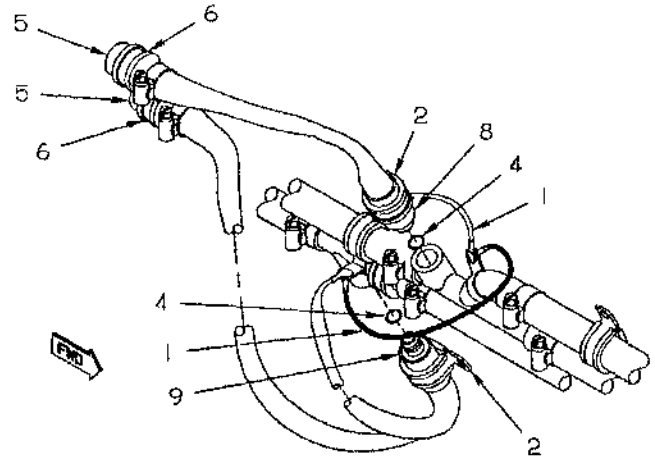


FORWARD WING

DETAIL C

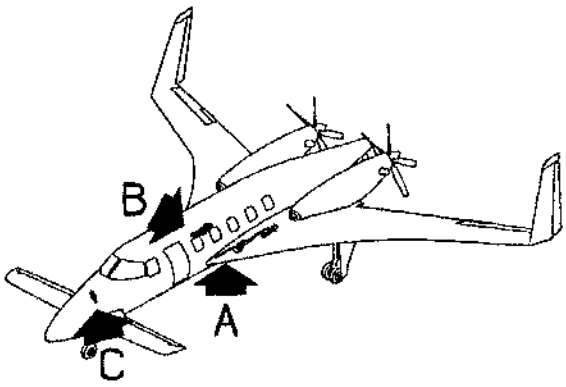
C93NC30B1296 C

Vacuum Switch Installation
(Effectivity: All)
Figure 10

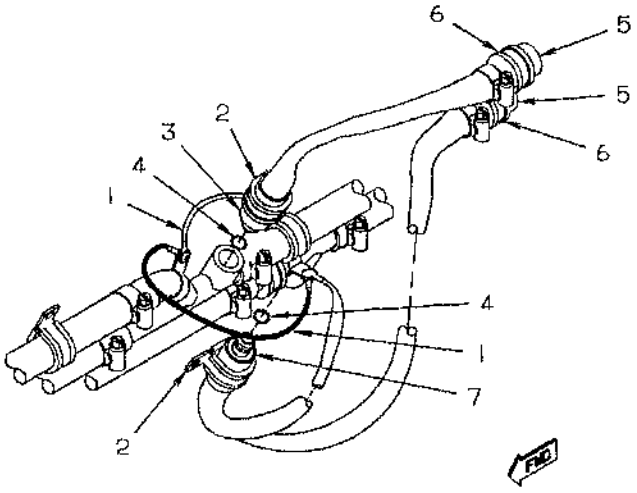


AFT WING, RIGHT SYSTEM

DETAIL B

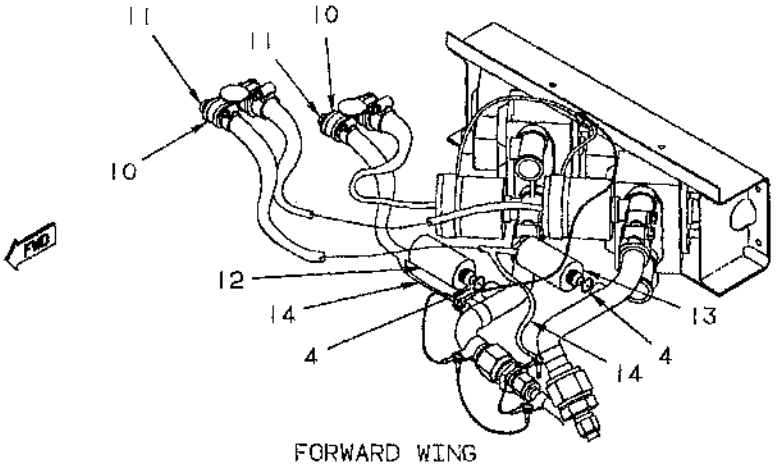


1. GROUND JUMPER
2. CLAMP
3. LEFT PRESSURE SWITCH (S308)
4. O-RING
5. ELECTRICAL CONNECTOR
6. CLAMP
7. LEFT PRESSURE SWITCH (S309)
8. RIGHT PRESSURE SWITCH (S310)
9. RIGHT PRESSURE SWITCH (S311)
10. CLAMP
11. ELECTRICAL CONNECTOR
12. RIGHT SYSTEM PRESSURE SWITCH (S307)
13. LEFT SYSTEM PRESSURE SWITCH (S306)
14. GROUND JUMPER



AFT WING, LEFT SYSTEM

DETAIL A



FORWARD WING

DETAIL C

C93NC30B1297 C

Pressure Switch Installation (Effectivity: All)
Figure 11

AFT WING PRESSURE SWITCH REMOVAL (Effectivity: All) (Figure 11)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallatiom.

a. Remove all elecrical power from the airplane.

b. Remove the forward fairing panel (527BB or 627BB) between the fuselage and aft wing as instructed in Chapter 53-50-00.

c. Remove the clamp (6) and disconnect the pressure switch electrical connector (5) shown in Detail A or B.

d. Disconnect the ground jumper (1).

e. Remove the clamp (2) around the pressure switch (3, 7, 8 or 9).

f. Unscrew the pressure switch (3, 7, 8 or 9) from the tubing port and remove it from the airplane.

g. Cap the open port to prevent the entry of contamination into the system.

h. Remove the O-ring (4) and discard it.

AFT WING PRESSURE SWITCH INSTALLATION (Effectivity: All) (Figure 11)

CAUTION

The pressure switch must meet electrical bonding requirements specified in Chapter 20-00-03.

a. Remove the cap from the tubing ports, place clamp (2) over the pressure switch (3, 7, 8 or 9) and install the pressure switch with a new O-ring (4).

b. Attach the clamp (2) to the stud on the side of the fuselage with a washer and nut.

c. Connect all of the ground jumpers (1).

d. Connect the electrical connector (5) shown in Detail A or B and secure it with a clamp (6).

e. Install the forward fairing panel between the fuselage and the aft wing as instructed in Chapter 53-50-00.

CROSSFEED VALVE REMOVAL (Effectivity: All) (Figure 12)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallatiom.

a. Remove all electrical power from the airplane.

b. Remove the right nose skin panel (132AR) as instructed in Chapter 53-30-00.

c. Disconnect the electical connector (8) from the crossfeed valve (4).

d. Remove the screw and washer (9) from the mounting bracket (5). Remove the clamp (10) from the crossfeed valve (4).

e. Disconnect the inlet tube (1) from the crossfeed valve (4) and the tee (6).

f. Disconnect the crossover tube (7) from the crossfeed valve (4) and remove the valve from the airplane.

g. Remove the unions (2) and the O-rings (3) from the ports of the crossfeed valve (4). Discard the O-rings (3).

h. Cap all open ports and tubes to prevent contamination from entering the system.

CROSSFEED VALVE INSTALLATION (Effectivity: All) (Figure 12)

CAUTION

The crossfeed valve must meet electrical bonding requirements specified in Chapter 20-00-03.

a. Install the unions (2) with new O-rings (3) in the ports of the crossfeed valve (4).

b. Remove the caps from the tubing and connect the crossfeed valve (4) to the crossover tube (7).

c. Connect the inlet tube (1) to the crossfeed valve (4) and tee (6).

d. Place the clamp (10) over the body of the crossfeed valve (4) and attach it to the bracket (5) with a washer and screw (9).

e. Connect the electrical connector (8) to the cross-feed valve (4).

f. Install the right nose skin panel as instructed in Chapter 53-30-00.

SEPARATOR REMOVAL (Effectivity: All) (Figure 13)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove the center fairing panel (527CB or 627CB) between the fuselage and the aft wing as instructed in Chapter 53-50-00.

b. Unscrew the tubing nuts (3) on the pressure inlet tubes (2) to the separator (5).

c. Loosen the clamp (6) on the outlet tube (7) of the separator (5).

d. Unscrew the separator clamps (4) to completely separate them.

e. Loosen or remove the clamps (1) on the pressure tube as necessary to spread the tubing (2) away from the separator (5) and remove the separator.

f. Cap the tubing and ports to prevent entry of contaminants into the system.

SEPARATOR INSTALLATION (Effectivity: All) (Figure 13)

CAUTION

The separator must meet electrical bonding requirements specified in Chapter 20-00-03.

a. Remove the caps from the tubing and ports.

b. Place the separator (5) in position while engaging the separator ports with the pressure tubing (2 and 7).

c. Tighten the pressure inlet tube nuts (3).

d. Tighten the clamp (6) on the outlet tube (7)

e. Tighten the clamps (4) around the separator (5).

f. Install the fairing panel between the fuselage and the aft wing as instructed in Chapter 53-50-00.

SURFACE DEICER BOOT CLEANING (Effectivity: All)

The surface of the deicer boots should be checked for indications of engine oil after servicing and at the end of each flight. Any oil spots that are found should be removed with a mild detergent and water solution. Care should be exercised during cleaning to avoid scrubbing the surface of the deicer boots, as this will tend to remove the special ICEX application. It should also be remembered when servicing the airplane that the Estane deicer boots are made of a soft flexible material, which may be damaged if refueling hoses are dragged over the surface of the deicer boots, or if ladders and platforms are rested against them.

SURFACE DEICER BOOT REMOVAL (Effectivity: All)

CAUTION

Never allow Methyl Ethyl Ketone (MEK) to contact the deicer boots. Estane, used in the construction of the Starship 1 deicer boots, is not compatible with ketone type solvents.

a. When the outboard deicer boot on the aft wing is to be removed, remove the vortilons from the leading edge of the wing as instructed in Chapter 57-00-00. The vortilons must be removed to aid in the removal and installation of the deicer boot.

b. Starting at one corner of the upper trailing edge of the deicer boot, soften the adhesion line with solvent (6, Chart 2, 30-00-00).

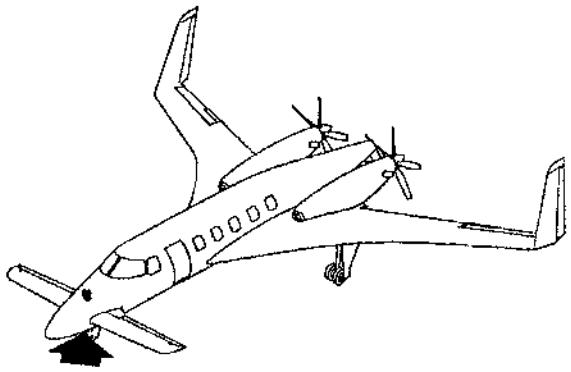
c. Apply a minimum amount of the solvent to the seam line while carefully applying tension to peel back the deicer boot.

CAUTION

Removal should be slow enough to allow the solvent to undercut the boot, thus preventing damage to the deicer boot surface. Avoid excess use of the solvent.

d. Using a brush or trigger-type oil can to apply the solvent, separate the deicer boot from the leading edge for a chordwise distance of four inches across the upper trailing edge.

1. INLET TUBE
2. UNION
3. O-RING
4. CROSSFEED VALVE
5. MOUNTING BRACKET
6. TEE
7. CROSSOVER TUBE
8. ELECTRICAL CONNECTOR
9. SCREW AND WASHER
10. CLAMP



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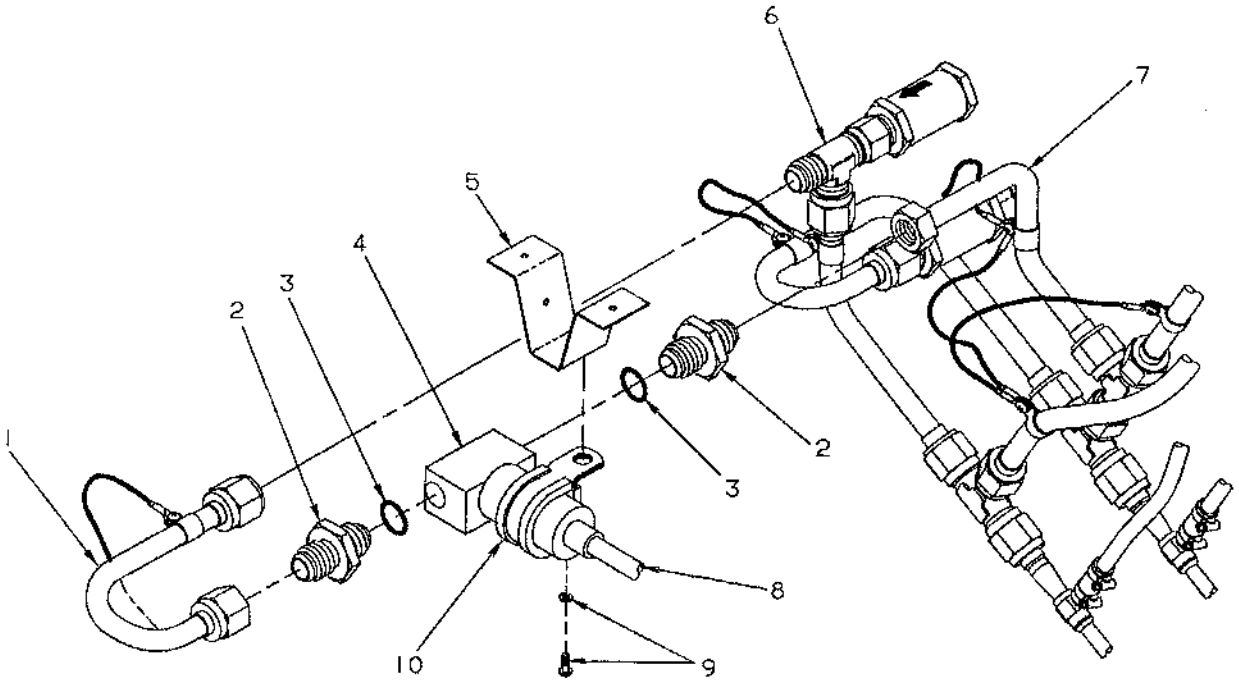
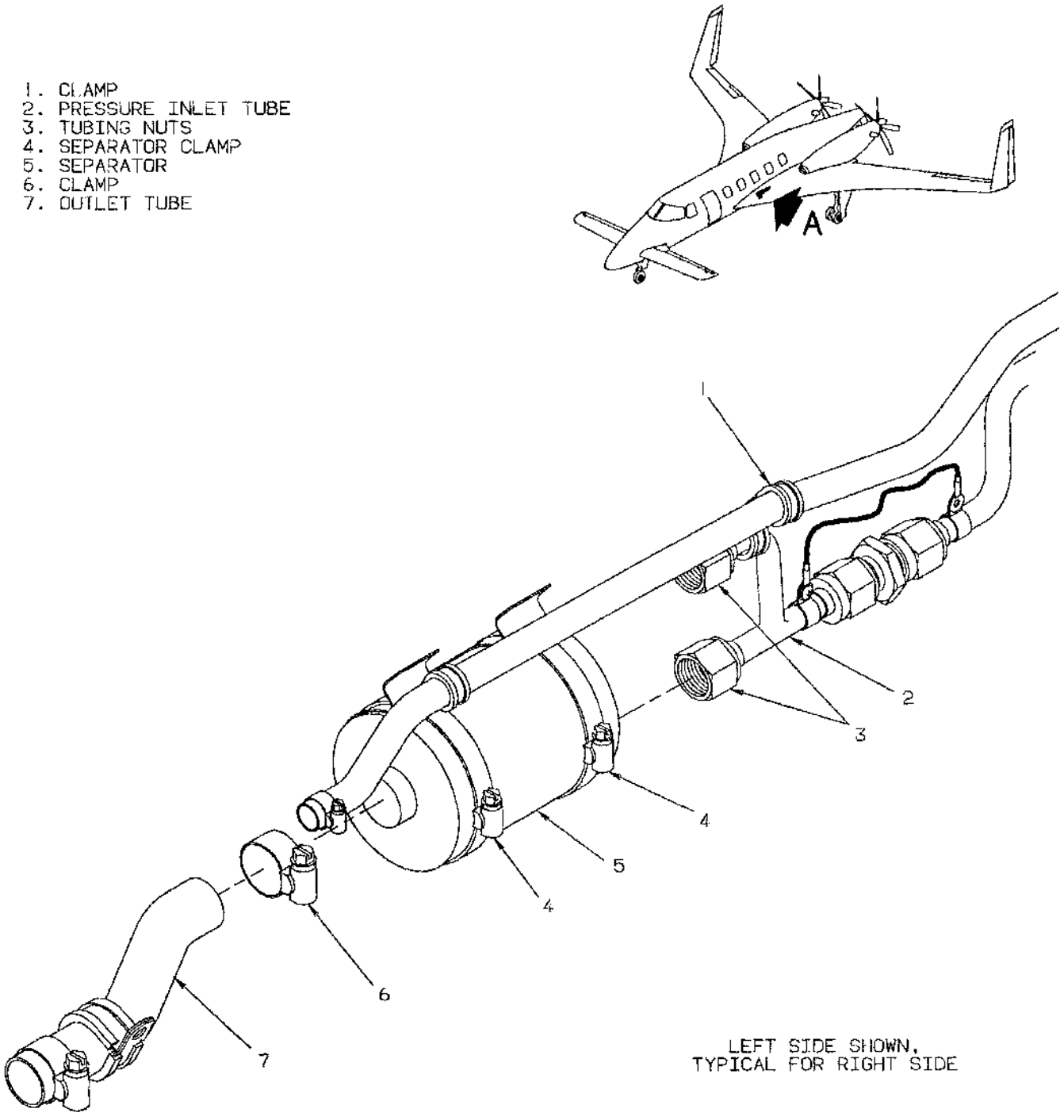


DETAIL A

C93NC30B1298 C

Crossfeed Valve Installation
(Effectivity: All)
Figure 12

1. CLAMP
2. PRESSURE INLET TUBE
3. TUBING NUTS
4. SEPARATOR CLAMP
5. SEPARATOR
6. CLAMP
7. OUTLET TUBE



DETAIL A

Separator Installation
(Effectivity: All)
Figure 13

NOTE

The area between the deicer boot and leading edge, which has been separated, will act as a reservoir for the solvent; consequently, the deicer boot may be pulled forward with uniform tension.

e. From the centerline of the leading edge, continue to apply solvent to undercut the adhesive while applying tension to the deicer boot until the deicer boot is removed.

NOTE

If extreme care is exercised during removal and no adhesive is removed from the deicer boots or leading edges, reinstallation of the deicer boots may be accomplished by reactivating the adhesive and following the instructions outlined under SURFACE DEICER BOOT INSTALLATION in this section. If any adhesive is removed, refer to the SURFACE DEICER BOOT PREPARATION AND INSTALLATION PROCEDURES in this section to remove the remaining adhesive and apply new adhesive before reinstalling the deicer boots.

SURFACE DEICER BOOT PREPARATION AND INSTALLATION PROCEDURES (Effectivity: All)

SURFACE PREPARATION (Effectivity: All)

CAUTION

Never allow Methyl Ethyl Ketone (MEK) to contact the deicer boots. Estane, used in the construction of the Starship 1 deicer boots, is not compatible with ketone type solvents.

a. With 3/4-inch tape (9, Chart 2, 30-00-00), mask off the area to be covered by the deicer boot, allowing a 1/2- to 3/4-inch margin beyond the actual deicer boot bonding area. Mask off accurately to avoid unnecessary clean-up of excess adhesive.

b. Locate the vortilon attachment holes and install a strip of masking tape (9, Chart 2, 30-00-00) on the wing where each of the vortilons were attached. Mark the masking tape so the vortilon attachment holes can be located after installation of the deicer boot. After the deicer boot is installed, holes have to be cut in the boot for installation of the vortilons.

c. Clean the masked area with a clean, white, grease-free cloth dampened with solvent (6, Chart 2, 30-00-00).

NOTE

Change cloths frequently to avoid dragging contamination from one area to another. Do not contaminate the clean supply of solvent by dipping a contaminated cloth in it.

d. Scuff sand the masked area with 320-grit sandpaper. Use care not to expose any fibers when sanding on the composite portion of the leading edge.

e. Reclean the masked area with a clean cloth dampened with solvent (6, Chart 2, 30-00-00), then quickly wipe the surface dry with a clean, grease-free dry cloth. Do not allow the surface to air dry.

f. Cover the cleaned surface with kraft paper and wait an hour before rechecking the surface for cleanliness with a clean, dry, grease-free white cloth. If the clean cloth picks up any dirt, repeat the preceding steps.

g. Apply primer (5, Chart 2, 30-00-00) to the titanium portion of the leading edge for maximum adhesion on airplanes NC-4 thru NC-20.

NOTE

If the finish on the back of the deicer boot is smooth, mechanically roughen it before cleaning.

h. Carefully clean the rough back surface of the deicer boot using a clean, white, grease-free cloth dampened with solvent (6, Chart 2, 30-00-00). Change cloths frequently to avoid recontaminating the cleaned portions of the deicer boot.

i. Perform the previous step again to remove all contaminants. If the surface still shows signs of contamination, repeat the cleaning sequence a third time or more until the deicer boot is free of all contaminants.

j. Prepare and apply two coats of adhesive as instructed in the APPLICATION OF ADHESIVE procedure that follows.

APPLICATION OF ADHESIVE (Effectivity: All)

Open the container and thoroughly stir the adhesive (3, Chart 2, 30-00-00) with a clean stick. The adhesive can be used without thinning if a smooth coat can be

obtained. It may be thinned 10% with a 50-50 blend of solvent (1, Chart 2, 30-00-00) and solvent (6, Chart 2, 30-00-00). If the solids have settled to the bottom of the can and cannot be blended by stirring, a paint shaker can be used. Do not attempt to use adhesive that has jelled and/or separated and cannot be blended together.

NOTE

DO NOT apply adhesive at temperatures below 40° F. or when the relative humidity is above 80 percent. If this requirement cannot be met, allow an extended drying time for the adhesive. DO NOT apply adhesive (3, Chart 2, 30-00-00) when the air is contaminated with dust or other elements.

The following chart will serve as a guide for adhesive application.

CHART 1
ADHESIVE DRYING TIMES
(Effectivity: All)

Temperature in °F.	Drying Time In Minutes
Above 80°	30
60° to 80°	45
40° to 60°	60

Apply the adhesive to the deicer boots and leading edges using one of the following methods:

BRUSH COAT METHOD (Effectivity: All)

a. Apply one fairly even coat of adhesive (3, Chart 2, 30-00-00) to the cleaned back surface of the deicer boot and to the cleaned surface on the leading edge. Allow this coat to dry for at least 30 minutes.

b. Apply a second coat of adhesive to the same surfaces.

NOTE

Care should be taken to apply an even, uniform coating. Brushing the adhesive too long in one area tends to soften the first coat, causing rolling and balling up of the adhesive.

c. Allow the second coat of adhesive to dry for a minimum of 30 minutes (preferably an hour). Remove the masking tape before the adhesive thoroughly dries.

NOTE

Excess open dry time (not to exceed 7 days) is allowable, providing the prepared surfaces are covered and kept clean.

SPRAY COAT METHOD (Effectivity: All)

NOTE

For spraying consistency, dilute the adhesive (3, Chart 2, 30-00-00) with a blend of two parts solvent (2, Chart 2, 30-00-00) and one part solvent (1, Chart 2, 30-00-00). The viscosity of the adhesive shall be equivalent to 20 seconds through a No. 4 Ford cup (4, Chart 1, 30-00-00).

a. Using one of the spray guns (1, Chart 1, 30-00-00), spray an even coat of the diluted adhesive on the cleaned back surface of the deicer boot and on the cleaned surface of the leading edge. Allow this coat to dry for at least 30 minutes.

b. Spray a second coat of adhesive on the same surfaces.

c. Allow the second coat of adhesive to dry for a minimum of 30 minutes (preferably an hour). Remove the masking tape before the adhesive thoroughly dries.

NOTE

Excess open dry time (not to exceed 7 days) is allowable, providing the prepared surfaces are covered and kept clean.

SURFACE DEICER BOOT INSTALLATION
(Effectivity: All)

NOTE

While installing the deicer boots, a rubber roller (3, Chart 1, 30-00-00) may be used to remove air bubbles from under the deicer boots. Use care to avoid entrapping air during the rolling operation. If an air blister does occur, carefully remove the deicer boot by applying solvent (6, Chart 2, 30-00-00) to the adhesive and carefully pulling outward. Allow the air to escape, then press on the deicer boot until the adhesive seals the deicer boot in place. A vacuum may be applied to aid in installation of the deicer boots.

a. For deicer boots and leading edges that do not have a complete coating of adhesive, prepare the boot and the leading edge boot area as instructed in the SURFACE PREPARATION procedure and apply two coats of adhesive as instructed in the APPLICATION OF ADHESIVE procedure.

b. Using a carpenter's chalk line, snap a line centered on the leading edge of the airfoil sections.

c. Snap a corresponding line along the center of the adhesive side of the deicer boot.

NOTE

Ensure that both chalk lines are approximately centered. If the deicer boot has a white center reference line, the additional chalk line is not required.

d. Trace the reference and/or chalk lines with a ballpoint pen.

e. Remove the chalk line from the adhesive with a dry cloth if necessary.

NOTE

Have a sufficient number of personnel present to effectively handle the deicer boot.

f. Attach all pneumatic lines to the deicer boot pneumatic connection tubes with a clamp and use care not to leave fingerprints on the deicer boot adhesive.

g. Position the deicer boot on the leading edge so that the centerlines match.

h. Using a clean, lint-free cloth heavily moistened (but not dripping) with solvent (6, Chart 2, 30-00-00), reactivate the adhesive on the leading edge and deicer boot around the connections and join as the adhesive becomes tacky.

i. Continue to reactivate the adhesive along the center line on both the deicer boot and leading edge in span-wise strips about one or two inches wide, and join together when the adhesive becomes tacky. Avoid excess rubbing of the adhesive during activation.

j. Remove air bubbles by rolling a rubber roller (3, Chart 1, 30-00-00) on the deicer boots parallel with the inflatable tubes. If the deicer boot becomes off-course, carefully peel it back, using a minimum amount of solvent (6, Chart 2, 30-00-00), and reposition it properly.

CAUTION

DO NOT allow an excessive amount of solvent to contact the deicer boot. The solvent is to undercut the adhesive, thus preventing damage to the deicer boot. Should the adhesive be removed from either the deicer boot or leading edge, it must be replaced and allowed to dry for a minimum of 30 minutes before continuing.

k. Continue to activate the top and bottom half of the deicer boot in sequence. Work spanwise over an area 3 to 4 inches wide and 24 inches long while hand rolling the mated surfaces to get good contact between the surfaces and remove any air pockets. The rubber roller (3, Chart 1, 30-00-00) may be used on the deicer boots, providing the direction of the rolling is parallel with the inflatable tubes.

l. Attach the deicer boot in this manner until all but about two inches of the deicer boot is attached.

CAUTION

When trimming the deicer boots, do not trim the deicer boot closer than 0.75 inch from the edge of the inflatable tubes.

m. Using a sharp knife, trim the deicer boot as required for proper fit. On the aft wing outboard boot, locate the vortilon attachment holes and trim a hole to fit around the vortilons within .03 inch. The outboard end of the forward wing boots do not require trimming.

n. After the trimming process has been completed, reroll the entire surface of the deicer boot by applying pressure at the centerline area and working outward.

o. Roll the edges of the deicer boot with a steel stitcher roller

p. Remove any remaining masking and clean any extra adhesive from the deicer boot and airfoil using a clean, white cloth dampened with solvent (6, Chart 2, 30-00-00). Do not allow the solvent to run under the edges of the deicer boot.

q. Install the vortilons on the aft wing as instructed in Chapter 57-00-00.

r. Seal all edges as instructed in the EDGE SEALING procedure.

EDGE SEALING (Effectivity: All)

All cut edges of trim outs, vertical splices and outboard end edges of the deicer boots must be filled and sealed with sealant (10, Chart 2, 30-00-00). The trailing edges of the deicer boots must be sealed with silver conductive sealant (13, Chart 2, 30-00-00). Seal the deicer boots as follows:

a. Ensure that all excess adhesive has been removed from the deicer boots and airfoil. Use a clean white cloth dampened with solvent (6, Chart 2, 30-00-00) to remove the adhesive, but use care not to allow the solvent to run under the edges of the deicer boots.

b. Apply tape (8, Chart 2, 30-00-00) to the deicer boot surface at trimouts and vertical splices. At the outboard ends and at the trailing edges, apply tape 1/8 to 1/4 inch away from the edge of the deicer boot. The tape line must be as straight as possible. Clean the masked areas with water so as not to remove the conductive coating.

NOTE

Apply primer (5, Chart 2, 30-00-00) to any bare composite surfaces. Allow the primer to dry for at least 30 minutes before applying the sealant. Epoxy primed surfaces should be scuff sanded with 400-grit sandpaper or a scotchbrite pad in the masked area (includes the masked surface of deicer boot also), and cleaned with a clean white cloth dampened with solvent (6, Chart 2, 30-00-00).

c. Sealant (10, Chart 2, 30-00-00) is a two-part material. The mixing ratio is 100 parts A to 50.4 parts B by weight. Silver conductive sealant (13, Chart 2, 30-00-00) is a single-component material. Shake or stir thoroughly before applying.

d. Use a flexible, flat-edged tool to apply the sealant with smooth, long continuous strokes. The sealant must overlap onto the deicer boot and airfoil surface to provide a smooth, continuous surface. To prevent runs, DO NOT apply excess sealant. The sealant around the vortilons must be installed with a .25-inch radius.

e. A second person must follow the sealer and remove the tape immediately after the sealant has been applied.

f. Allow the sealant to cure for approximately 6 to 12 hours. If conditions are cold or of high humidity, the sealant may take more time to dry. Allow the sealant adequate drying time before cleaning up.

g. After the sealant has cured, clean the deicer boot and surrounding area with a clean white rag dampened with solvent (6, Chart 2, 30-00-00).

CAUTION

Do not fly the airplane until the deicer boot adhesive has had at least 24 hours to cure from the time of installation. The deicer boots shall not be pressure tested until 48 hours after the installation is complete. Pressure testing shall not exceed 18 psi. and shall not be applied for longer than 4 minutes on the deicer boots.

SHORTENED 1300L DRYING TIME FOR STANDARD PNEUMATIC DEICERS (Effectivity: All) (Figure 14)

NOTE

The following instructions are provided to supply relief from the 48 hours of drying time recommended for 1300L adhesive on pneumatic deicers. This applies to the deicers with 1 1/4-inch or narrower tubes operating at 20 psi or less pressure.

The chart gives the reduced set times after installation for the applicable pneumatic deicers. Refer to Figure 14. The airplane may be operated after this set time as long as the deicer system is not used until after the 48-hour period.

If the deicers are flexed before a 48-hour set time, the deicer boots must be checked for signs of lifting as soon as possible and be reinstalled or replaced if visible signs of lifting are present.

CAUTION

Attempts to speed up drying time by applying heat externally through the deicers are not recommended since it is important for the bondline to be at the indicated temperature. The deicers will insulate the bondline from the heat being applied through it, and excess heat could cause deterioration of the deicers.

NOTE

While a reduction in set time after installation is allowed, it is recommended that the actual set time before inflation of the deicers be extended as much as possible.

The following instructions are provided for using the shortened drying-time chart:

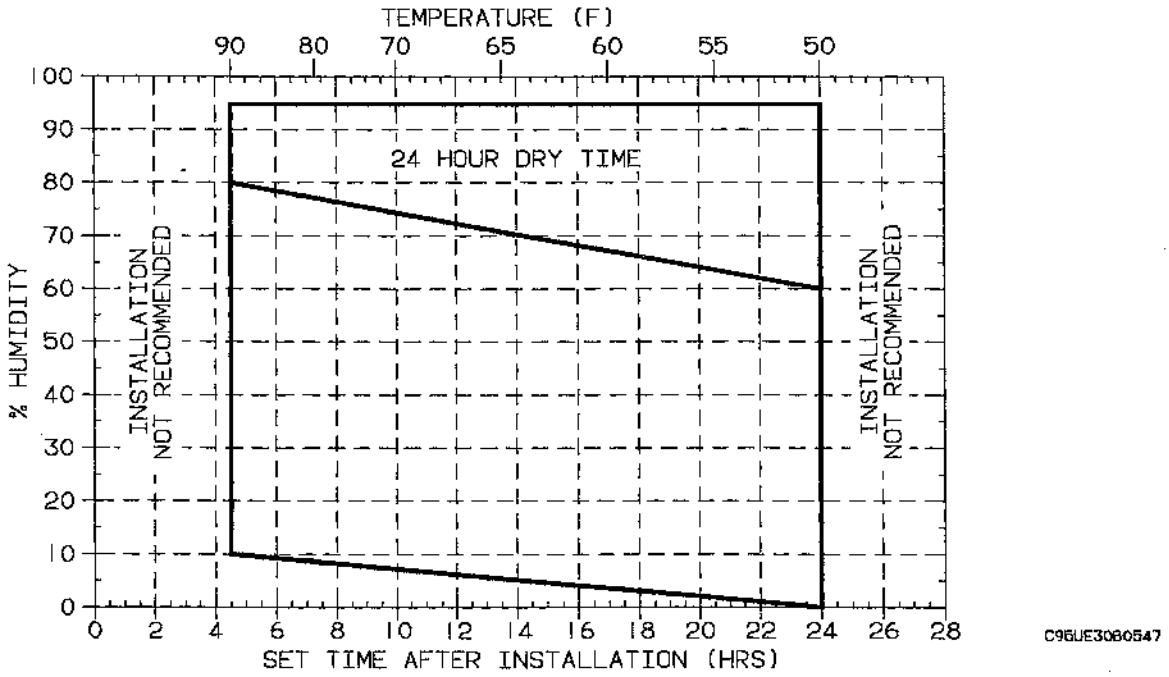
a. First determine the room temperature and humidity.

b. Locate the point on the chart where the temperature (scale at the top of the chart) and humidity (scale on the side of the chart) intercept.

c. If the intercept point falls within the "Installation Not Recommended" area, do not install the deicers until the temperature and humidity conditions fall within the acceptable range or the installation is moved to an area where the temperature and humidity are within the acceptable range.

d. If the intercept point falls within the "24-hour drying time" area, allow 24 hours of drying time.

e. If the intercept point falls within the middle area, read the recommended drying time by moving straight down to the drying time scale (at the bottom of the chart).



1300L Drying Time for Standard Pneumatic Deicers
(Effectivity: All)
Figure 14

APPLICATION OF ICEX COMPOUND (Effectivity: All)

ICEX compound (7, Chart 2, 30-00-00) is a silicone-based material specifically compounded to lower the strength of ice adhesion on the surfaces of the deicer boots. It will not harm the deicer boots and offers added ozone protection. When properly applied and renewed at periodic intervals, ICEX provides a smooth, polished film that evens out microscopic irregularities on the rubber surfaces; therefore, ice formations have less chance to cling and are removed faster and cleaner when the deicer boots are operated. Apply the ICEX compound according to the following instructions:

NOTE

Because ICEX contains silicone, it should not be applied prior to painting in the adjacent area.

a. Thoroughly clean the deicer boot with a clean white rag dampened with solvent (6, Chart 2, 30-00-00). Do not allow excess solvent to run down the side of the deicer boot.

b. Wash the deicer boot with mild soap and water. Rinse thoroughly with water and allow to air dry.

c. Mask the area adjacent to the deicer boots with tape (9, Chart 2, 30-00-00) to prevent staining.

d. Thin the ICEX compound 10% to 25% with solvent (4, Chart 2, 30-00-00) to obtain a thin coat.

e. Use a clean cloth saturated with the thinned ICEX compound for application. Hand buff with a clean soft cloth until a smooth glossy surface is obtained.

NOTE

The frequency of application will depend upon the number of flight hours and type of weather encountered. Reapply as required to maintain a thin polished film on the deicer boot surfaces.

DEICER BOOT SURFACE COATING INSPECTION (Effectivity: All)

The aft and forward wing surface deicer boots have a silver conductive-surface coating to assist in the electrical dissipation of precipitation static to the airframe via the conductive edge sealer. Proper application and maintenance of the silver conductive coating and sealer is important in maintaining proper radio and avionics operation and preventing excessive static charge buildup. If the static charge is allowed to build to a level high enough to penetrate the deicer boot, damage to the deicer boot in the form of pinhole leaks along the stitch lines between the deicer boot inflation paths will result. Inspect the silver conductive-surface coating of the deicer boot as follows:

a. Visually inspect the deicer boot surfaces along the leading edge for erosion of the silver conductive-surface coating down to the grey estane material on the outer layer of the deicer boot. Erosion of the silver conductive-surface coating must not exceed 2 inches in width on the leading edge.

b. If the surface of the deicer boot requires recoating, obtain the surface repair kit (12, Chart 2, 30-00-00) and apply the sealer and silver conductive-surface coating according to instructions provided with the kit.

SURFACE DEICER BOOT REPAIRS (Effectivity: All)

Minor scuffed areas on the boots normally require only restoring the conductive surface in the immediate area. However, if the entire surface ply has been removed, exposing the brown natural rubber underneath, or if the boot is cut, torn or ruptured, it is necessary to patch the damage. Obtain the patch kit (15, Chart 2, 30-00-00) for instructions and materials needed to repair estane deicers with severe damage and/or air leaks.

To maintain the maximum functional efficiency of the surface deicer boots. the following limits are recommended by B. F. Goodrich concerning the allowable number and concentration of patches.

- Patches 1.25 inch x 2.50 inches - 3 patches per 12-inch square.
- Patches 2.50 inches x 5 inches - 2 patches per 12-inch square.
- Patches 5 inches x 10 inches - 1 patches per 12-inch square.

Replace any deicer boot that has a cut, tear or rupture that exceeds .75 inch in length.

RESURFACING DEICER BOOTS (Effectivity: All)

An edge sealer kit (17, Chart 2, 30-00-00) is available for resurfacing of Estane boots. Refer to the instructions provided with the kit for resurfacing the boots.

SCUFF DAMAGE REPAIR
(Effectivity: All)

This damage is the type most commonly encountered and generally will require no repair; however, close visual inspection when the boots are cycled may expose deep scuffs and cuts. B. F. Goodrich Kit (15, Chart 2, 30-00-00) (for Estane boots) contains cold patches suitable for repairing the damaged area.

Cold patch repairs can be made as follows:

a. Clean the area around the damage with a clean, grease-free cloth dampened with solvent (6, Chart 2, 30-00-00).

b. Using steel wool, buff the area around the damage so that it is moderately, but completely roughened.

NOTE

A locally manufactured buffing shield, fabricated from any thin sheet material, will assure a neater repair.

c. Wipe the buffed area with a clean grease-free cloth dampened with solvent (6, Chart 2, 30-00-00) to remove loose particles.

d. Select a repair patch which will extend at least 1/2 inch beyond the damaged area in all directions.

e. Apply one even coat of No. 4 cement, provided in the repair kit, to the patch and the damaged area. Allow the cement to dry until tacky.

f. Apply the patch to the boot with an edge, or the center, adhering first. Work the patch down carefully to avoid air pockets. Thoroughly roll the patch down with a stitcher roller.

NOTE

The patches are manufactured to stretch in one direction only. Be sure to cut and apply the selected patch so that stretch is in the widthwise (chordwise) direction of the inflatable tubes. The length of the patch shall run along the length of the tubes.

g. Allow the patch to set for 10 to 15 minutes, then clean the patch and surrounding area with a clean cloth dampened with solvent (6, Chart 2, 30-00-00) for Estane boots.

h. Satisfactory adhesion of the patch will be reached in about four hours; however, the boot may be inflated to check the repair after a minimum of 30 minutes.

i. Apply a smooth light coat of sealing compound (16, Chart 2, 30-00-00) to seal and feather the edges of the patch to the boot.

j. After the cement and sealing compound has dried and cured, apply a thin coating of silicone coating (7, Chart 2, 30-00-00) to the patch and surrounding area.

TUBE AREA DAMAGE
(Effectivity: All)

Repair cuts, tears or ruptures to the tube area as follows:

a. Clean the area around the damage with a clean, grease-free cloth dampened with solvent (6, Chart 2, 30-00-00).

b. Buff the area around the damage with the buffing stick provided in the repair kit.

c. Wipe the buffed area with a clean, grease-free cloth dampened with solvent (6, Chart 2, 30-00-00) to remove loose particles.

d. Select a patch of ample size to cover the damaged area and extend at least 5/8 inch beyond the ends and edges of the cut or tear.

NOTE

If none of the patches are the size required, cut a patch of the desired size from one of the larger patches. If this is done, bevel the edges by cutting with shears at an angle.

e. Complete the installation of the patch as outlined in SCUFF DAMAGE REPAIR, steps d through j.

LOOSE SURFACE PLY IN NONINFLATABLE AREA (Effectivity: All)

Using the repair kit, repair the damaged area as follows:

a. Peel and trim the loose surface ply to a point where adhesion of the surface ply is good.

b. Scrub (roughen) the area in which the surface ply is removed with steel wool. The scrubbing motion must be parallel to the cut edge of the ply to prevent the ply from loosening.

c. Scrub with steel wool and solvent (6, Chart 2, 30-00-00) directly over and parallel to the edges of the ply to ensure proper taper.

d. Cut a piece of surface ply material large enough to cover the damaged area and extend at least one inch beyond the damaged area in all directions.

e. Mask off an area 1-1/2 inches larger in length and width than the size of the damaged area.

f. Apply one, even coat of No. 4 cement, provided in the repair kit, to the damaged area and the surface ply material. Allow the cement to dry until tacky.

g. Roll the surface ply to the deicer boot with a two-inch rubber roller, applying enough tension to prevent wrinkling. If air blisters appear, remove with a hypodermic needle and reroll the area.

h. Remove the masking tape and complete the installation as outlined in SCUFF DAMAGE REPAIR steps g through j.

LOOSE SURFACE PLY IN TUBE AREA
(Effectivity: All)

Loose surface ply in the tube area is usually an indication of incipient flex failure. This type of failure is easily detected as a blister under the surface ply when the boot is pressurized. If detected while a small blister (1/4 to 3/8 inch in diameter) and immediately patched, service life of the boot will be appreciably extended. Repair of this type damage is the same as SCUFF DAMAGE REPAIR.

FABRIC-BACK-PLY DAMAGE DURING REMOVAL
(Effectivity: All)

If the adhesive from the leading edge skin has adhered to back surface of the boot, remove the adhesive with steel wool and solvent (6, Chart 2, 30-00-00) for Estane boots. In spots where the coating has pulled off the fabric leaving bare fabric exposed, apply at least two additional coats of adhesive (3, Chart 2, 30-00-00). Allow each coat to dry thoroughly.

ENGINE AIR INTAKES - DESCRIPTION AND OPERATION - (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The engine air intake anti-icing consists of a heated air-inlet lip and an induction-air-duct inertial ice separator.

AIR INTAKE ANTI-ICE LIP (Effectivity: All) (Figure 1)

The engine air inlet lip is continuously heated by engine P_3 bleed air when the firewall bleed air shutoff valve is open. This air is then ducted to provide pneumatic air, cabin pressurization, heating and ventilation. Control of the bleed air passing through the anti-ice lip is by the BLEED AIR VALVES switch (A175) on the copilot's inboard subpanel.

INERTIAL ICE SEPARATION SYSTEM (Effectivity: All) (Figure 1)

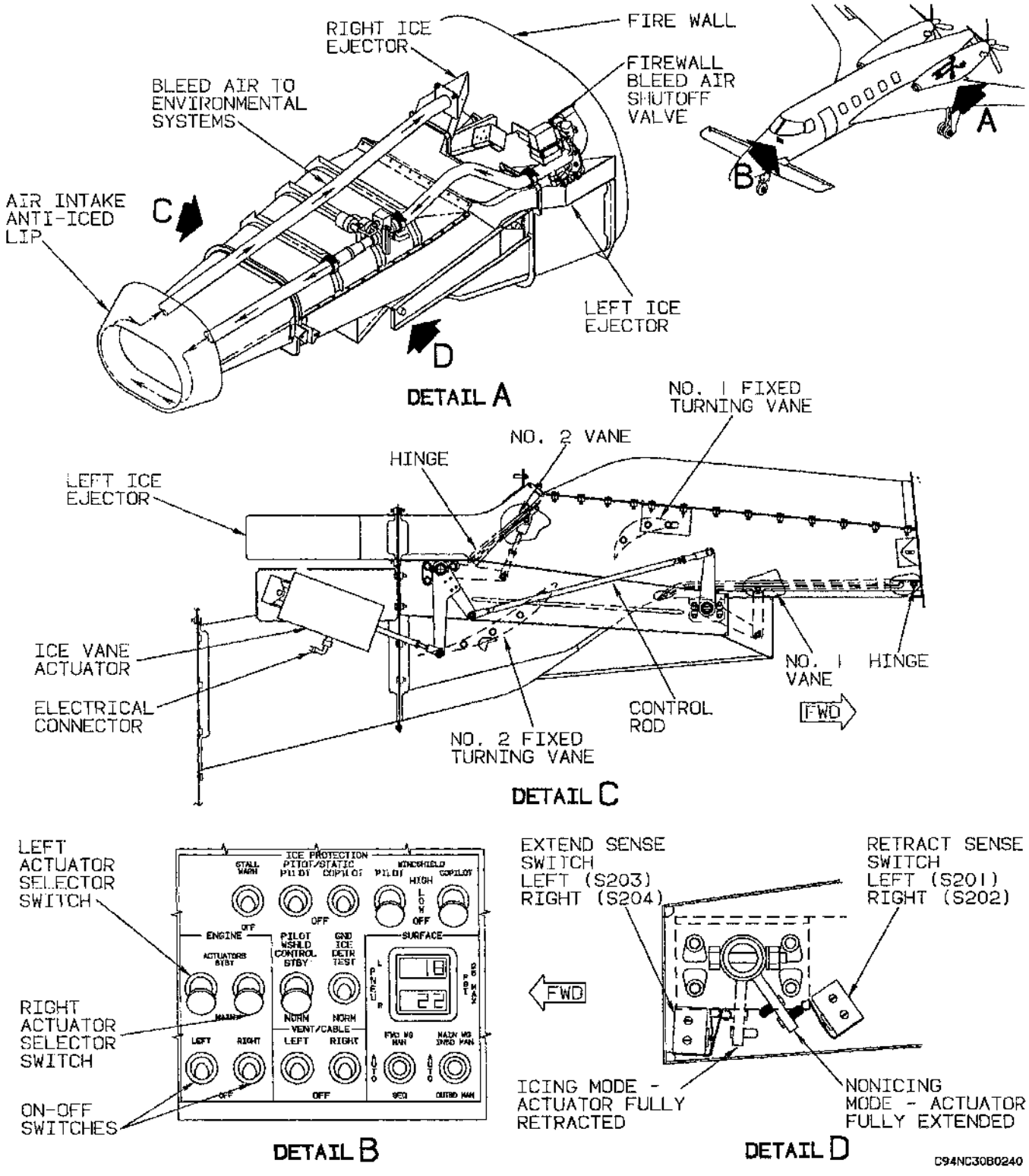
The induction air duct inertial ice separator consists of two movable vanes actuated by linkage, levers and an ice vane actuator having a standby and a main electric motor. The operating linkage and ice vane actuator are on the right side of both of the right and left inlet air ducts. The retract and extend sense switches are on the left side of the inlet air ducts under access panels 582AL and 682AL. In nonicing conditions, the No. 1 vane is down against the bottom of the duct and the No. 2 vane is up, closing off the ice exit duct. When the system is in the ICING mode, the No. 1 vane is up against the No. 1 fixed turning vane and the No. 2 vane is down against the leading edge of the No. 2 fixed turning vane. This forces all of the inlet air to go over the top of the No. 1 fixed turning vane. The No. 1 turning vane deflects the ram air downward to induce a sudden turn in the direction of the airstream, causing the moisture and ice particles to continue on undeflected and be discharged overboard through the left and right ice ejectors. Refer to Figure 2 for the flow of air through the inlet duct during nonicing conditions and icing conditions.

INERTIAL ICE SEPARATION SYSTEM CONTROLS (Effectivity: All) (Figures 1 and 3)

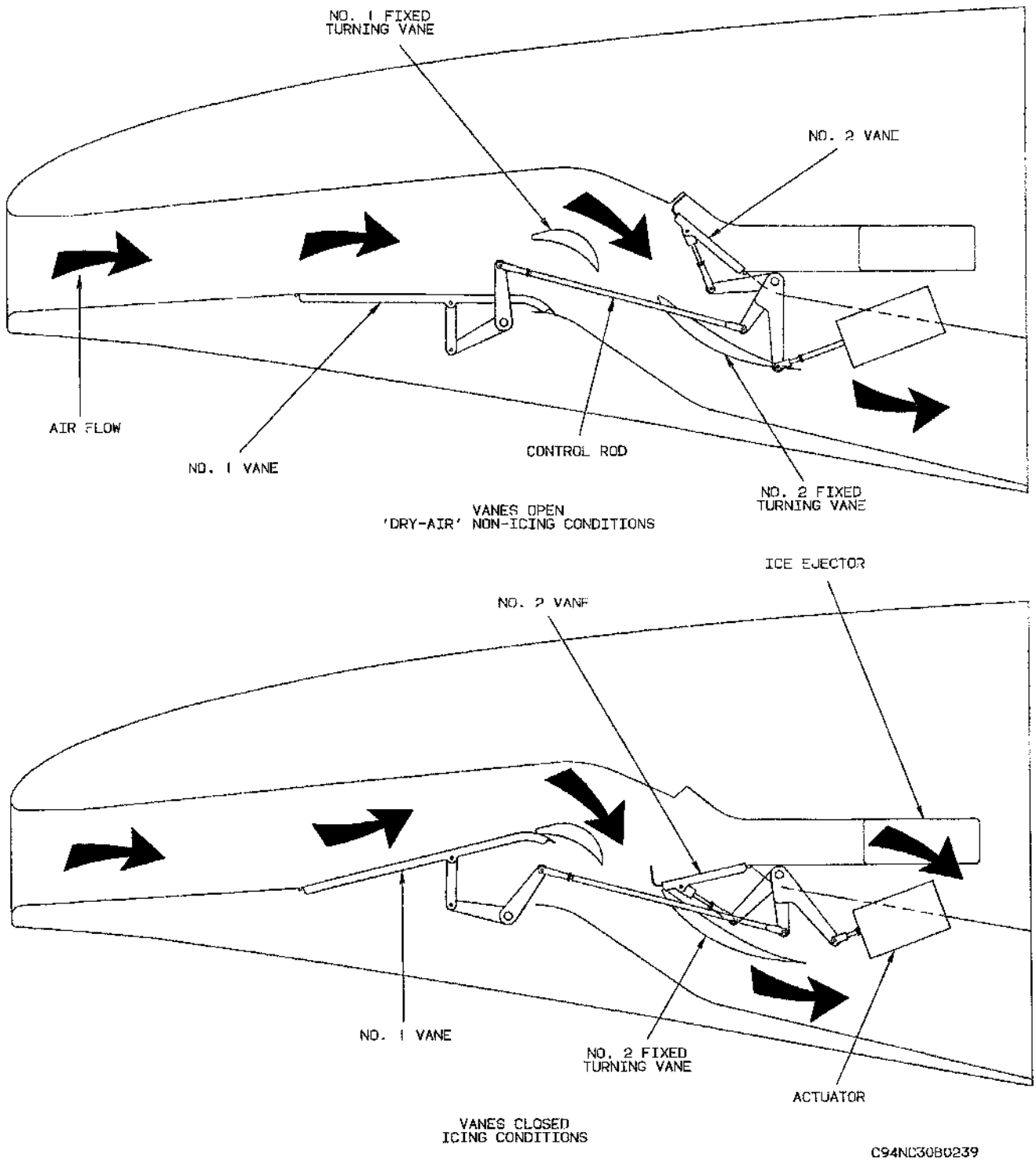
The engine air intake controls are identical and independent for the left and right engines. The controls consist of a two-position, lever-lock selector switch to select the MAIN or STBY motor on the inertial separator ice vane actuator and a two-position toggle switch for OFF or ON. Typically, the ice vane actuator will be operated by the MAIN motor. Selection of system ON will actuate the ice vane actuator motor.

ANNUNCIATION FOR THE ENGINE AIR INTAKE SYSTEMS (Effectivity: All) (Figure 3)

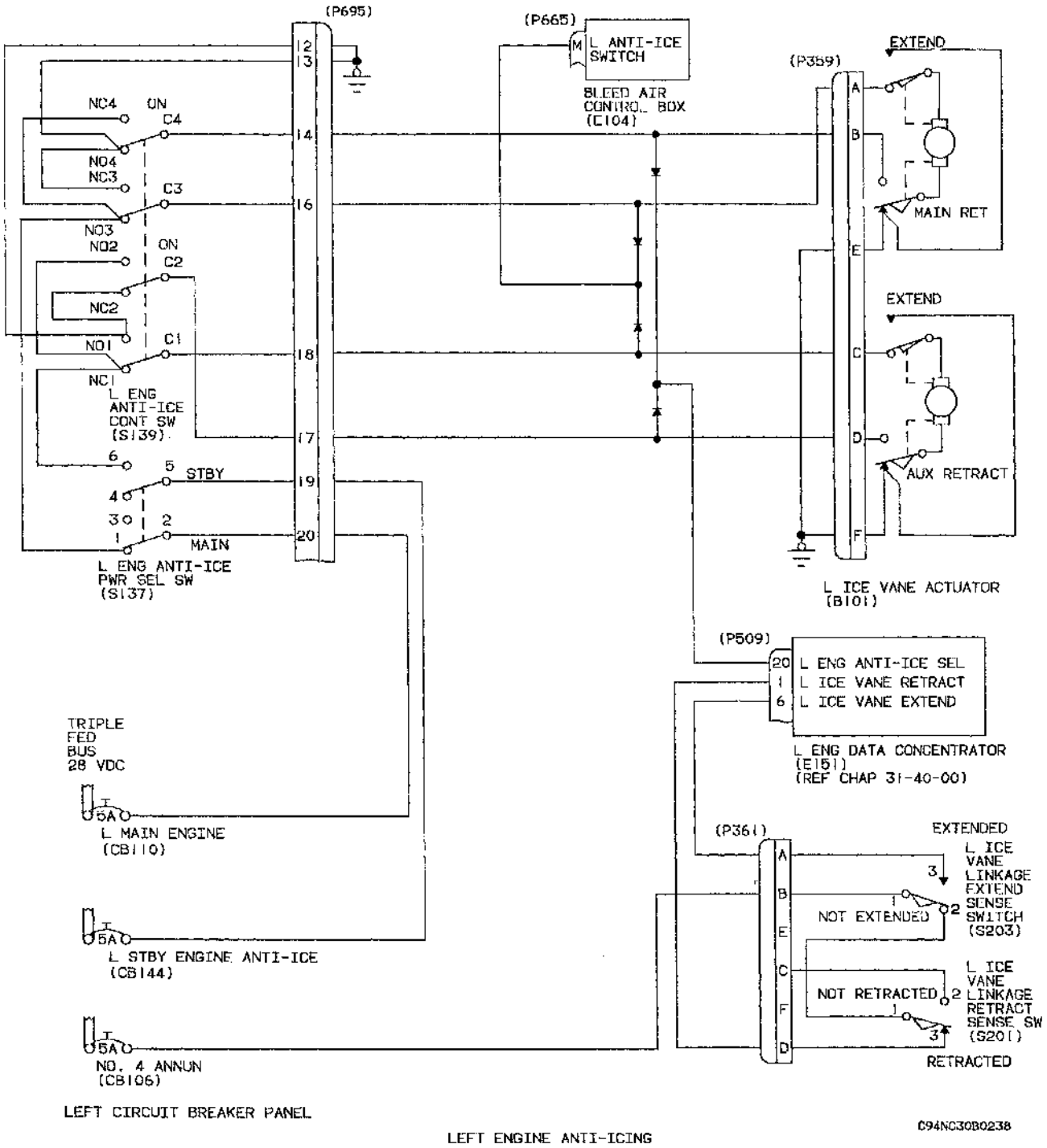
Annunciation for the left and right engines are identical and independent. The annunciation messages appear on the Engine Instrument, Crew Alerting System (EICAS) display. For the left-engine system, yellow annunciation is provided for the L ICE VANE FAIL, L BLEED OFF and L ENG ANTICE ON. If annunciation of L ICE VANE FAIL occurs under typical operation, the pilot must select the STBY actuator. The annunciation of the L BLEED OFF occurs because the firewall bleed air shutoff valve is not open. This condition does not allow for bleed air to be routed to the engine air inlet lip. The pilot must select the appropriate switch position to open the firewall bleed air shutoff valve. The firewall bleed air shutoff valves are controlled by the BLEED AIR VALVES switch (A175) on the copilot's inboard subpanel and is discussed in Chapter 21-40-00. Annunciation of the L ENGINE ANTICE ON occurs when the inertial separator ice vane actuator has attained its proper position and the system is operating properly. Loss of all aircraft electrical power results in the firewall bleed air valves remaining in the last selected position.



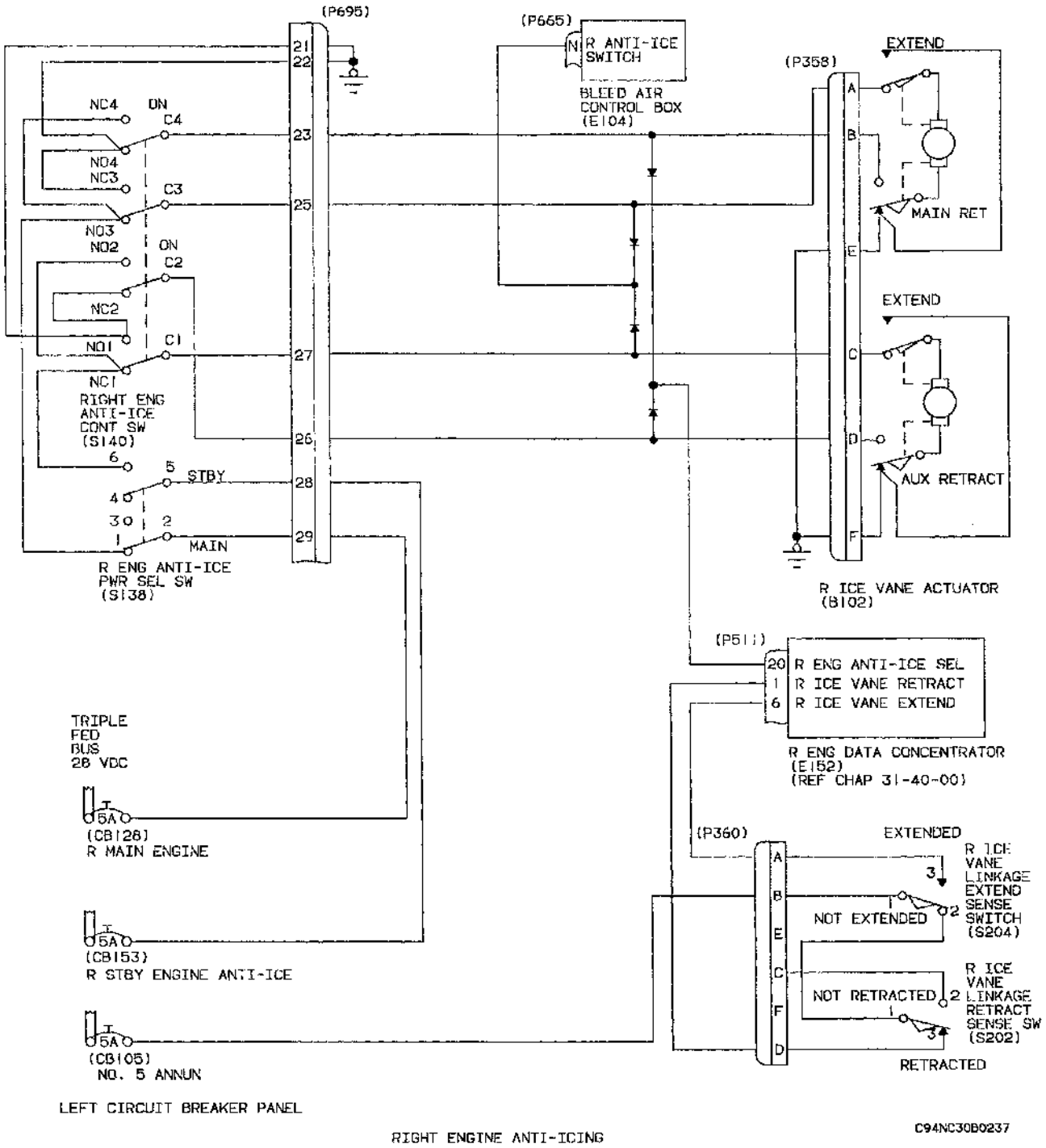
Air Inlet Anti-Icing (Effectivity: All)
Figure 1



Operation of Air Intake Anti-Icing
(Effectivity: All)
Figure 2



Air Inlet Anti-Icing Electrical Schematic
(Effectivity: All)
(Sheet 1 of 2)
Figure 3



Air Inlet Anti-Icing Electrical Schematic
(Effectivity: All)
(Sheet 2 of 2)
Figure 3

ENGINE AIR INTAKES - MAINTENANCE PRACTICES (Effectivity: All)

ICE VANE ACTUATOR REMOVAL (Effectivity: All) (Figure 4)

NOTE

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove all electrical power from the airplane.

b. Remove the access panel (582CR or 682CR) from the right side of the nacelle for access to the ice vane actuator (8).

c. Disconnect the electrical connector (15).

d. Remove the cotter pin (1), nut (2), washer (3) and bolt (4) connecting the ice vane actuator fork (6) to the lever (5).

e. Remove the nut (11), washer (10) and bolt (9) from the ice vane actuator (8) pivot and remove the actuator from the nacelle.

f. Disconnect the bonding jumper (13) from the ice vane actuator (8) case.

ICE VANE ACTUATOR INSTALLATION (Effectivity: All) (Figure 4)

NOTE

The ice vane actuator must meet the electrical bonding requirements specified in Chapter 20-00-00.

If the control linkage was properly adjusted before removing the actuator, adjust the replacement actuator to the same retracted length as the actuator that was removed.

a. Adjust the length of the retracted replacement ice vane actuator (8) to a length of 8.72 inches between the centerline of the actuator fork (6) to the centerline of the actuator pivot hole where bolt (9) is installed. The adjustment is made by loosening the locknut (7) and turning the fork (6) in or out. After adjustment, tighten the locknut (7).

b. Connect the bonding jumper (13) to the ice vane actuator (8) case with a screw (14).

c. Attach the completely retracted ice vane actuator (8) to the support (12) with the bolt (9), washer (10) and nut (11). Connect the actuator fork (6) to the lever (5) with a bolt (4), washer (3), nut (2) and cotter pin (1).

d. Connect the electrical connector (15).

e. Check the rigging as instructed by INERTIAL ICE SEPARATION SYSTEM RIGGING in this chapter.

f. Operate the ice vane actuator (8) to check that the system and the sense switches are operating properly.

g. install access panel (582CR or 682CR).

INERTIAL ICE SEPARATION SYSTEM RIGGING (Effectivity: All) (Figure 5)

NOTE

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove all electrical power from the airplane.

b. Remove access panels (583AT and 581AT or 683AT and 681AT) from the top of the nacelle.

c. Remove the duct top (10).

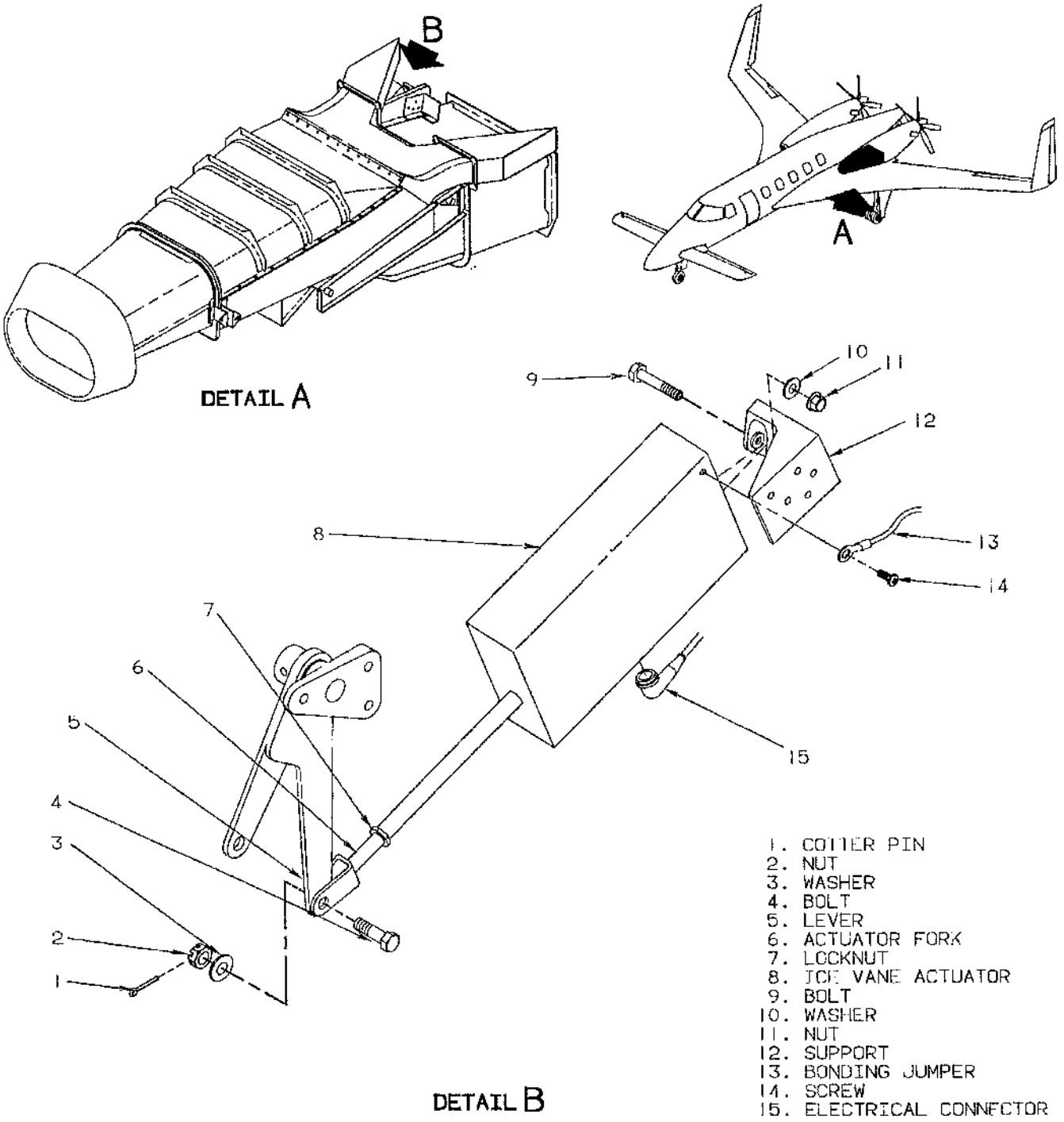
d. Remove the cotter pin, nut, washer and bolt (9) connecting the control rod (5) to the lever (11).

e. Remove the ice vane actuator as instructed in the ICE VANE ACTUATOR REMOVAL procedure in this section and adjust the length of the retracted replacement ice vane actuator (2) to a length of 8.72 inches between the centerline of the fork (16) to the centerline of the actuator pivot hole where bolt (1) is installed. The adjustment is made by loosening the locknut (17) and turning the fork (16) in or out. After adjustment, tighten the locknut (17).

f. Install the retracted ice vane actuator (2) as instructed in the ICE VANE ACTUATOR INSTALLATION procedure in this section.

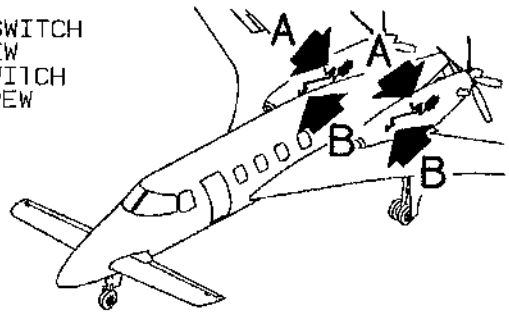
g. Adjust the length of the control rod (5) by loosening the locknut (7) and turning the clevis (8) so the No. 1 vane (12) just touches the No. 1 fixed turning vane (6). Then shorten the control rod (5) by turning the clevis (8) one turn. Tighten the locknut (7).

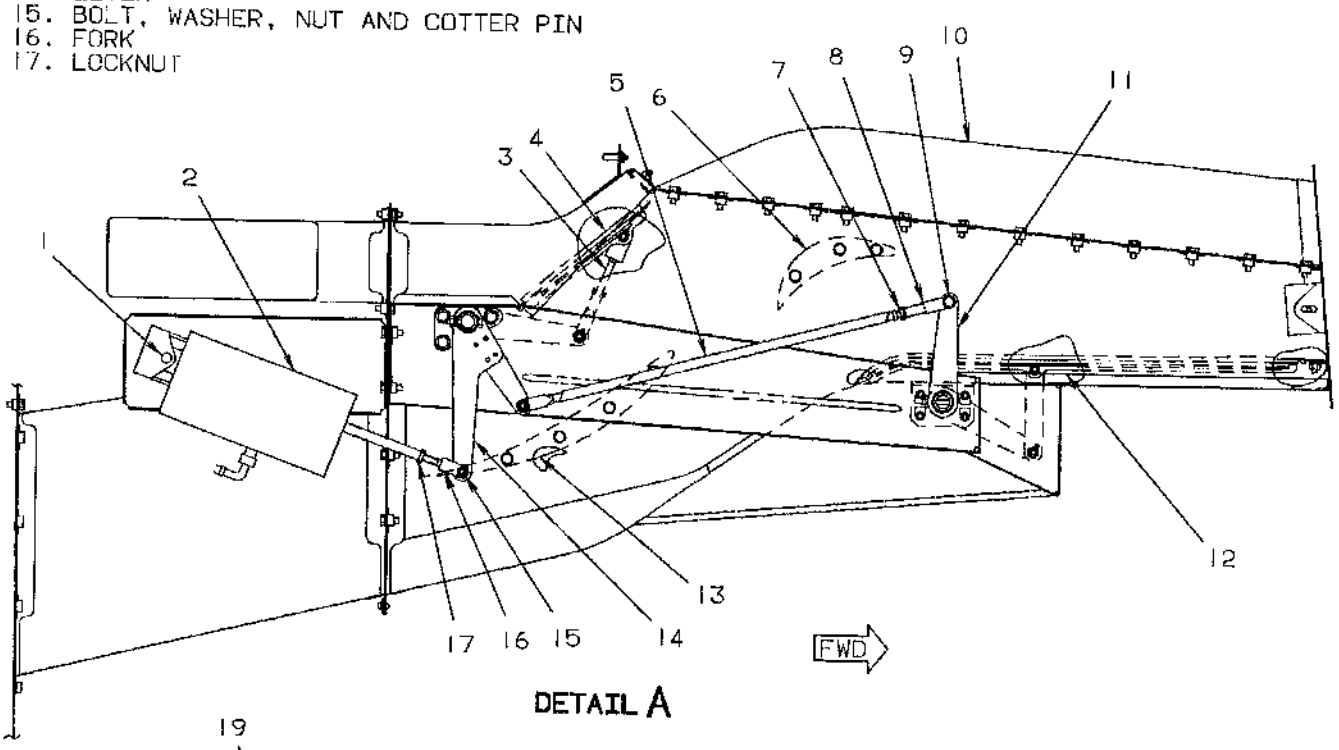
h. Remove the cotter pin, nut, washer and bolt (15) attaching the fork (16) to the lever (14) and operate the ice vane actuator (2) to the fully extended position.



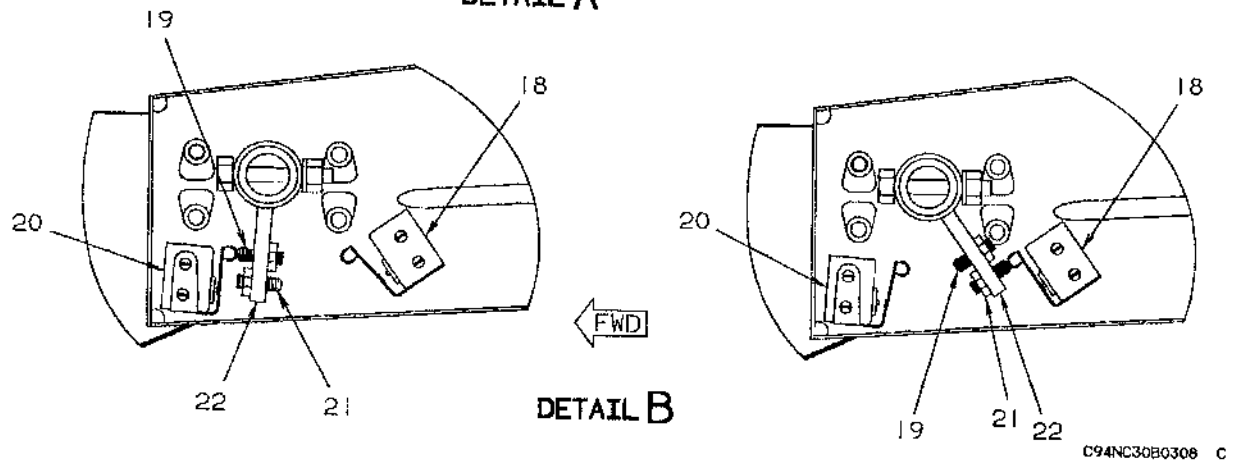
Ice Vane Actuator Installation
(Effectivity: All)
Figure 4

1. BOLT, WASHER, NUT
2. ICE VANE ACTUATOR
3. PUSH-PULL LINKS
4. NO. 2 VANE
5. CONTROL ROD
6. NO. 1 FIXED TURNING VANE
7. LOCKNUT
8. CLEVIS
9. BOLT, WASHER, NUT AND COTTER PIN
10. DUCT TOP
11. LEVER
12. NO. 1 VANE
13. NO. 2 FIXED TURNING VANE
14. LEVER
15. BOLT, WASHER, NUT AND COTTER PIN
16. FORK
17. LOCKNUT
18. RETRACT SENSE SWITCH
19. EXTEND SET SCREW
20. EXTEND SENSE SWITCH
21. RETRACT SET SCREW
22. SWITCH LEVER





DETAIL A



DETAIL B

C94NC30B0308 C

Inertial Ice Separation System Rigging
(Effectivity: All)
Figure 5

i. Manually operate the control linkage and check that the No. 2 vane (4) comes within .0625 to .125 inch of the No. 2 fixed turning vane (13) when open or closes to within .030 inch of touching the top of the duct when closed. Lengthening the push-pull links (3) will move the No. 2 vane (4) towards the top of the duct and away from the No. 2 fixed turning vane (13). Shortening the push-pull links (3) will move the No. 2 vane (4) in the opposite direction.

j. Check the alignment of the bolt hole in the fork (16) with the bolt hole in the lever (14). If bolt (15) cannot be installed, the ice vane actuator (2) extended length will have to be adjusted to permit installation of bolt (15). Make this adjustment if necessary.

k. Check adjusment of the extend (20) and retract (18) sense switches. Operate the linkage manually to make this adjustment.

1. Adjust the extend setscrew (19) to a position that actuates the extend sense switch (20) when the No. 1 vane (12) is within .125 to .200 inch of the No. 1 fixed turning vane (6).

2. Adjust the retract setscrew (21) to a position that actuates the retract sense switch (18) when the No. 1 vane (12) is within .125 to .200 inch of the bottom of the duct.

l. Connect the ice vane actuator fork (16) to the lever (14) with the bolt, washer, nut and cotter pin (15).

m. After completion of the rigging procedure, install duct top (10) and the access panels (583AT and 581AT or 683AT or 681AT) on top of the nacelle.

WINDOWS, WINDSHIELDS AND DOORS - DESCRIPTION AND OPERATION

WINDSHIELD WIPERS

Heavy duty windshield wipers for both the pilot and copilot provide visibility during rainy flight conditions. The windshield wipers are controlled by a rotary switch placarded WINDSHIELD WIPER, OFF-INTMT-SLOW-FAST located on the overhead switch panel. Circuit protection is provided by a 15 ampere circuit breaker on the circuit breaker panel on the LH side of the flight compartment. Rotary power for operating the wiper system is provided by 2 motor-converters located forward of the instrument panel. The motor contains an automatic reset thermal protector which will temporarily de-energize the motor in the event of an overload. Also included is a cam to automatically "park" both wipers when the rotary switch is turned to the OFF position.

CAUTION

Operating the wipers on a windshield covered with dust and sand will cause damage to the glass surface. During preflight or operation on the ground, lift the wiper blades off the glass surface, remove dust, sand or dirt from the blade section and apply water to the windshield.

WINDOWS, WINDSHIELDS AND DOORS - MAINTENANCE PRACTICES

WINDSHIELD WIPER MOTOR-CONVERTER REMOVAL

a. Remove all electrical power from aircraft.

b. Rotate the adjustment bolt counterclockwise to relax the tension on the wiper arm. Refer to Figure 1.

c. Loosen hub bolt and remove locknut and washer on wiper arm.

d. Remove the wiper arm assembly.

e. Refer to Chapter 31-10-00 and remove the Primary Flight Display and the Navigational Display unit.

f. Remove screws and let rack assembly drop down low enough to gain access to motor-converter unit.

g. Remove electrical connector.

h. Remove screws and washers securing the motor-converter in place and remove from mounting bracket.

WINDSHIELD WIPER MOTOR-CONVERTER INSTALLATION

a. Replace O-rings on the motor-converter shaft. Refer to Figure 1.

b. Insert shaft into mounting bracket and secure with screws and washers.

c. Install electrical connector.

d. Install equipment rack with screws.

e. Refer to Chapter 31-10-00 and install the Primary Flight Display and the Navigational Display unit.

f. Install wiper arm assembly in the park position.

g. Tighten hub bolt and install locknut and washer.

h. Rotate adjustment bolt clockwise to obtain 8 to 9 inch lbs. tension on the wiper arm.

i. Restore all electrical power to aircraft.

SPEED CONTROL ASSEMBLY REMOVAL

a. Remove all electrical power from the aircraft.

b. Refer to Chapter 53-30-00 and remove the RH Avionics Compartment Door.

c. Disconnect all electrical connectors from speed control located on forward pressure bulkhead.

d. Remove bolts and washers and remove from aircraft. Refer to Figure 1.

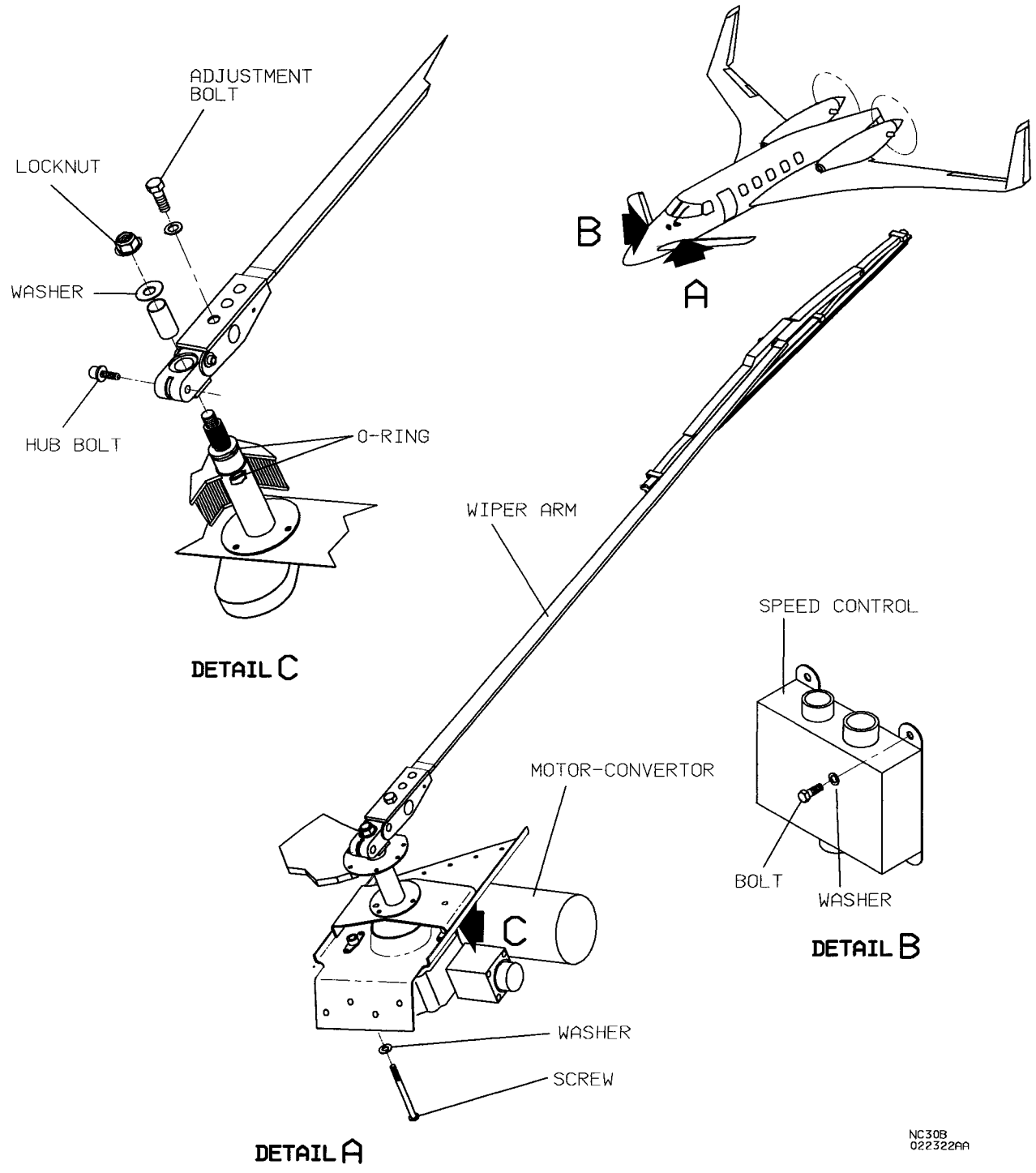
SPEED CONTROL ASSEMBLY INSTALLATION

a. Attach speed control to forward pressure bulkhead with bolts and washers. Refer to Figure 1.

b. Connect all electrical connectors to speed control.

c. Refer to Chapter 53-30-00 and install the RH Avionics Compartment Door.

d. Restore all electrical power to the aircraft.



Windshield Wiper Installation
Figure 1

WATER LINES - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapter 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

OVERBOARD DRAIN HEATER ELEMENT (Effectivity: All)

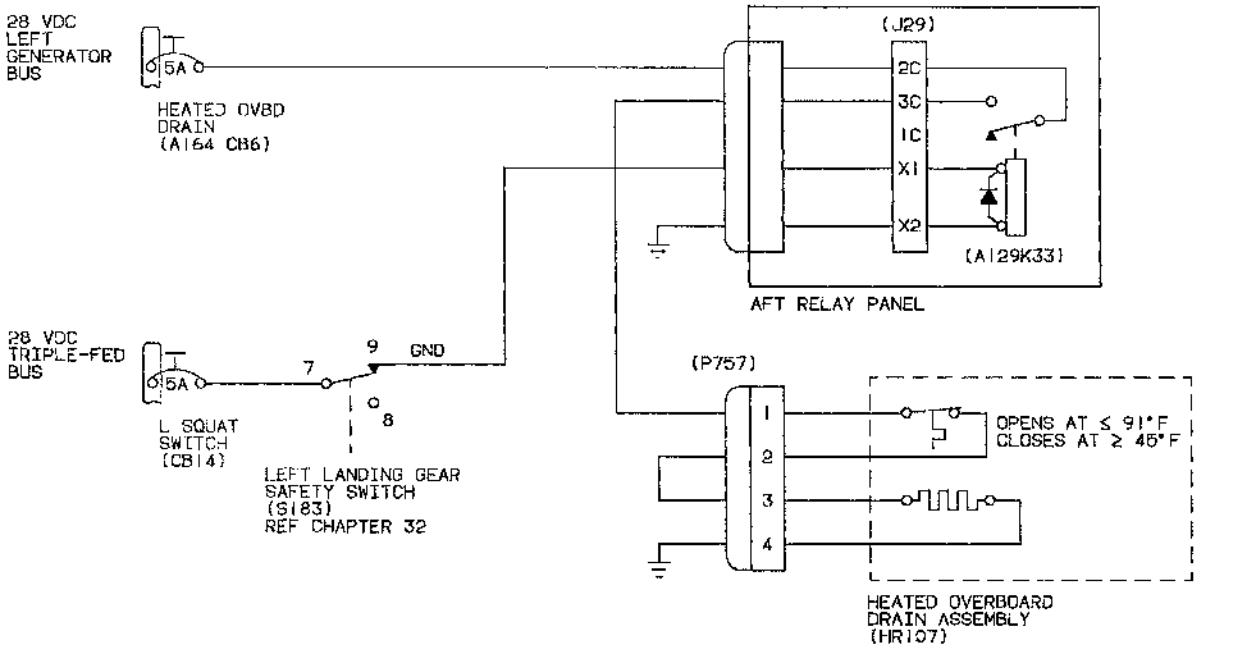
The overboard plumbing and drain is provided to drain fluid from the refreshment cabinet, relief tubes, and the toilet cabinet. A heater element with a pressure-sensitive adhesive backing is installed around the end of the overboard drain tube to keep the drain line from freezing. An in-line thermostat in the heater element closes at or above 45° F to keep the drain line thawed and opens at, or below, 91°F to keep excess heat from being transmitted to the fuselage. The heater element receives 28 vdc from the HEATED OVBD DRAIN circuit breaker. The circuit breaker is rated at 5 amps. Power to the heater element is cut off by the A129K33 relay when the airplane is on the ground. The A129K33 relay is energized by the left landing gear safety switch (S183). Note the schematic diagram in Figure 1.

WATER LINES - TROUBLESHOOTING (Effectivity: All)

Refer to Chart 1 to troubleshoot the overboard drain heater element. Note the schematic diagram in Figure 1.

CHART 1
TROUBLESHOOTING - WATER LINES (Effectivity: All)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Heater element does not operate.	a. Defective heater element or thermostat.	a. Disconnect the P757 connector and verify continuity. Replace the heater element if continuity exists between the circuit breaker and the P757 connector.
	b. Defective A129K33 relay.	b. Verify continuity at the A129K33 relay on the aft relay panel. Replace the relay if continuity exists between the circuit breaker and the J29 relay plug.
	c. Defective wiring.	c. Repair or replace the wire harness.



Overboard Drain Heater Element Schematic Diagram
(Effectivity: All)
Figure 1

WATER LINES - MAINTENANCE PRACTICES (Effectivity: All)

OVERBOARD DRAIN HEATER ELEMENT REMOVAL
(Effectivity: All)
(Figure 2)

a. Refer to Chapter 25 and remove the carpet from the floorboard (171BT).

b. Refer to Chapter 53-20-00 remove the left aft floorboard (171BT).

c. Refer to Chapter 38-30-00 and remove the overboard drain tube (7) from the airplane.

d. Identify, tag, remove the electric wires (2) from the electric connector (1).

e. Loosen the clamp (6) and remove the clamp (6) and the braided shield (3) from the electric wires (2).

f. Feed the electric wires (2) through the tube (8) and remove the heater element (4) from the drain tube (7).

OVERBOARD DRAIN HEATER ELEMENT INSTALLATION
(Effectivity: All)
(Figure 2)

a. Route the electric wires (2) through the tube (8) and position the heater element (4) around the drain tube (7).

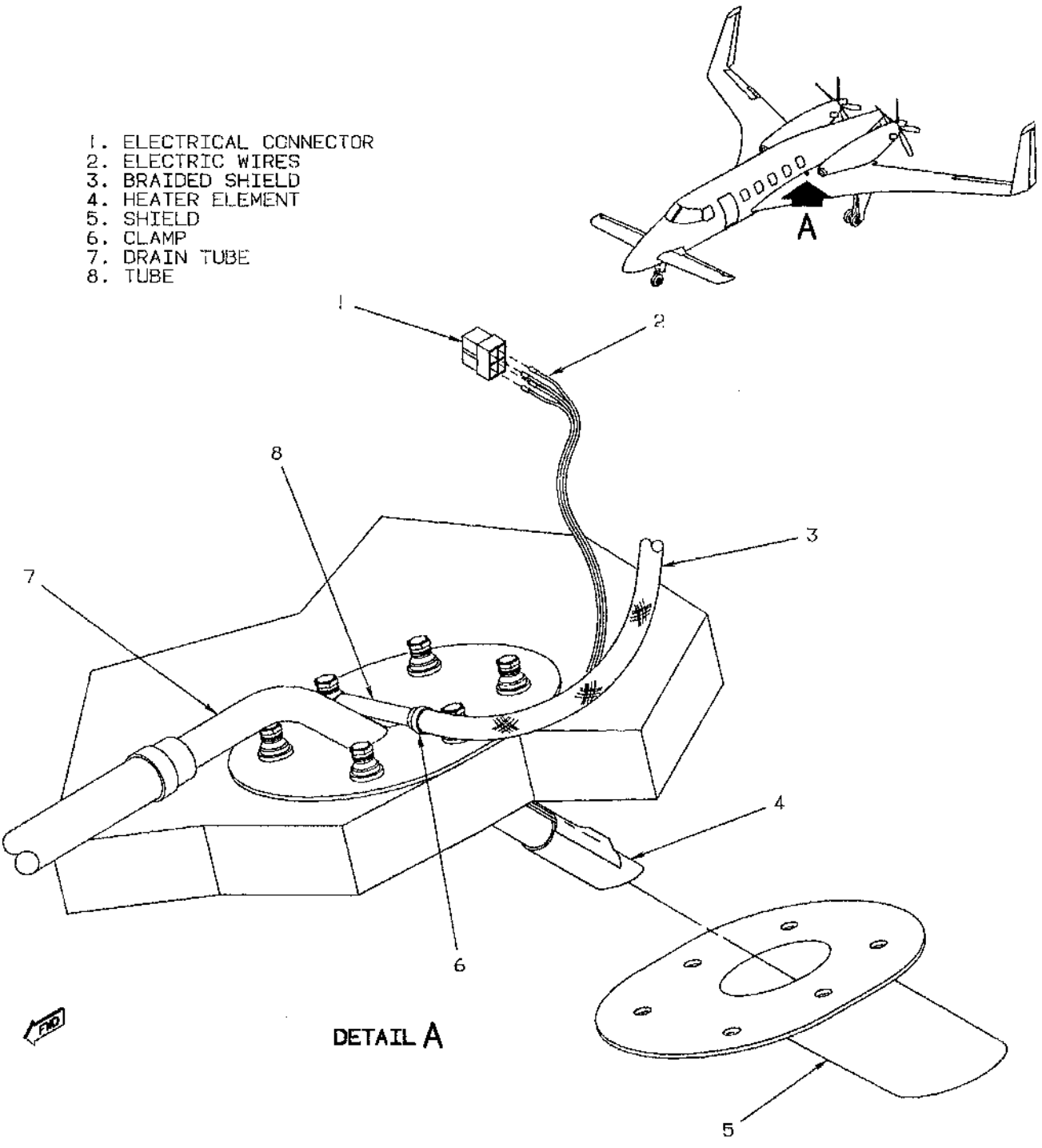
b. Slide the braided shield (3) and clamp (6) over the electric wires (2) onto the tube (8) and tighten the clamp (6).

c. Install the electric wires (2) into the electric connector (1).

d. Refer to Chapter 38-30-00 and install the overboard drain tube (7) into the airplane.

e. Refer to Chapter 53-20-00 and install the left aft floorboard (171BT).

f. Refer to Chapter 25-20-00 and install the carpet on the floorboard.



Overboard Drain Heater Element Installation
(Effectivity: All)
Figure 2

DETECTION - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors must be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

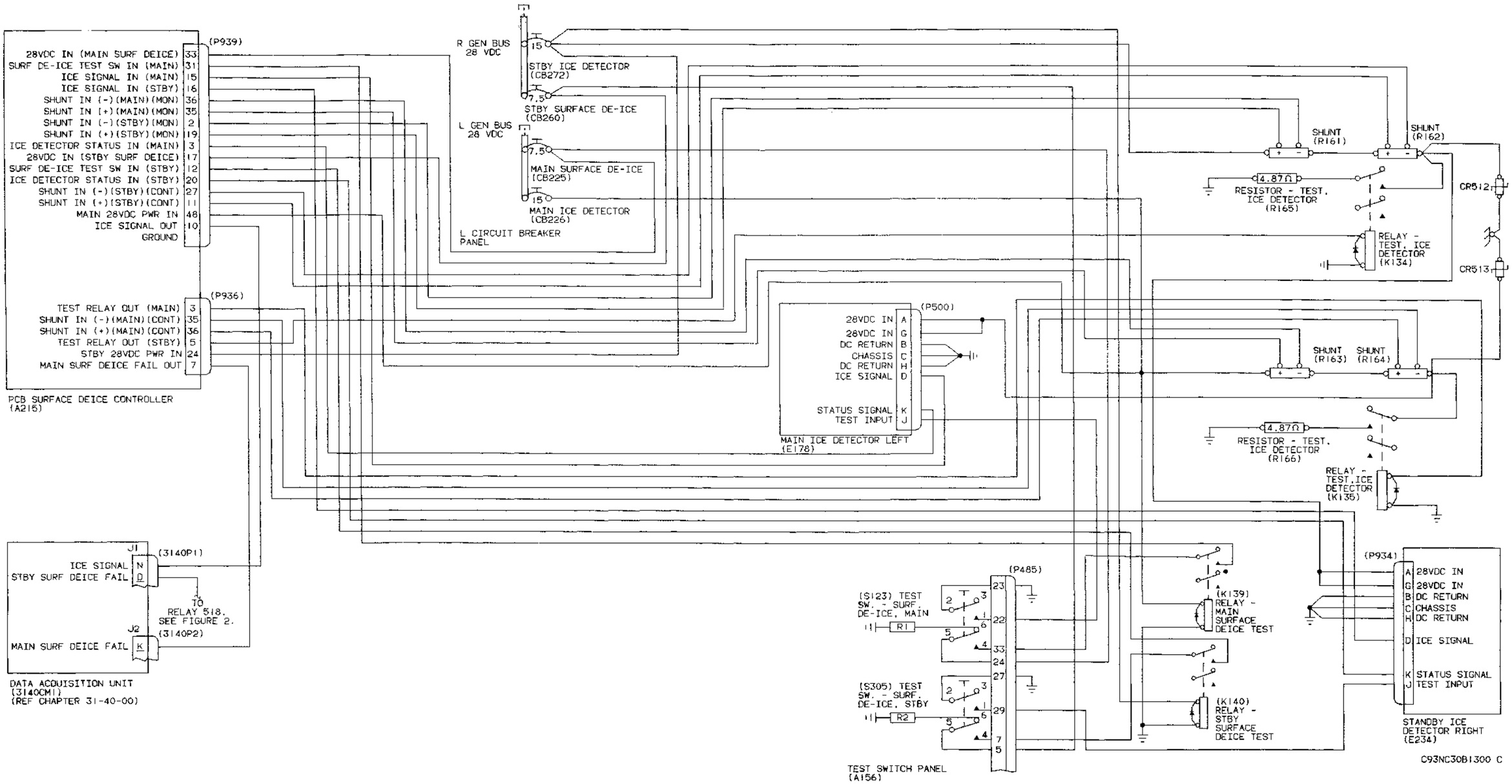
There are two ice detectors used to detect ice on the airplane during flight, the main ice detector (E178) and the standby ice detector (E234). These ice detectors automatically control the operation of the surface deice system. A ground ice detector (E263) is mounted on top of the nose of the airplane. The ground ice detector detects atmospheric icing conditions which may occur before takeoff and alerts the pilots to this condition. The main ice detector is located on the left nose skin panel (131AL) and the standby ice detector is located on the right nose skin panel (132AR). The ice detector test relays (K134 and K135), the shunts (R161, R162, R163 and R164) and the resistors (R165 and R166) are located under the right nose skin panel (132AR) shown in Chapter 39-20-00.

FLIGHT ICING DETECTOR SYSTEM (Effectivity: All) (Figure 1)

The flight icing detector system consists of two ice detectors. The main ice detector receives 28 vdc from the main ice detector circuit breaker (CB226) and the standby ice detector receives 28 vdc from the standby ice detector circuit breaker (CB272). The ice detectors provide the data acquisition unit with three signals: ICING, MAIN or STBY DEICE FAIL and a signal output for each cycle of the detector probe deice heaters. The ICING signal is actuated for 60 ± 10 seconds each time an ice level of .020 ± .005 inch in thickness is reached on the small detector shaft protruding from the ice detector. If the icing condition continues until the ice-thickness switching level is reached before 60 ± 10 seconds have elapsed, the ICING signal will be continuous. The sensing probe and strut deicing heaters will actuate ON with the ICING output signal to the EICAS display. The heaters turn off within 5 to 13 seconds after actuation (time for the ice on the probe to melt). After eight icing cycles of the detectors have been counted, the main monitor/logic controller actuates the forward wing deicer distributor valves to start the deicing of the wing leading edges. Refer to NORMAL DEICING OPERATION in Chapter 30-10-00 for a complete description of the detection, control and operation of the surface deice system.

The MAIN or STBY DEICE FAIL signal is actuated when a failure is detected by the built-in, continuous self-test of the detectors and micro-controllers and/or when the pilot initiates a test of the system by operation of the press-to-test switch which detects a failure of the ice detector or micro-controller internal circuity. The signal is displayed as a yellow EICAS message. The built-in, continuous self-test occurs every 250 milli-seconds after initial power on and following power interruptions that cause a micro-controller reset. Each ice detector does a more detailed check of additional internal circuitry every 60 seconds. During the more detailed, periodic test, the heater is pulsed for 10 milli-seconds to insure that it is operational. The pilot-initiated, press-to-test function performs the same test as described for the built-in self-test. The built-in test for the ice detectors is active when the airplane power is on. The ice detection mode is operational at all times, except during ground operations.

The ICING message is yellow on the EICAS display if the ice detector is sensing an icing condition and all ice protection systems are not active. The ICING message on the EICAS display is white if the ice detector is sensing icing conditions and all ice protection systems are on. The MAIN DEICE FAIL or STBY DEICE FAIL message on the EICAS display is yellow if either ice detector has failed a self-test and/or a pilot-initiated test. Any yellow displays actuate the master caution flasher. Refer to Chapter 31-50-00 for the conditions that cause a certain message to appear on the EICAS display.



Flight Icing Detector Electrical Schematic
(Effectivity: All)
Figure 1

GROUND ICING DETECTOR SYSTEM (Effectivity: NC-10, NC-44 And After, And Airplanes With BEECHCRAFT Kit No. 122-5024) (Figure 2)

The ground-icing detector is used only to alert the pilot of an atmospheric icing condition that occurs before takeoff. It will not actuate the surface deice system. The ground-ice detector will operate after the first generator is turned on until the left landing gear safety switch is opened at takeoff. The ground ice detector receives 28 vdc from the ground-ice detector circuit breaker (CB288) located on the auxiliary circuit breaker panel. The ice detector used with this system is mounted on top of the nose and is not associated with the flight-icing detector system. The operation of the ground detection probe is identical to that of the flight deice probes.

The ground icing detector can be checked by the crew by placing the GND ICE DETR switch on the pilot's inboard subpanel to TEST and holding for approximately 6 seconds. Icing messages will be illuminated on the EICAS display to determine if the icing equipment is on or off. Refer to Section IV Normal Procedures, ICING FLIGHT, BEFORE TAKEOFF (RUNUP) of the Starship 1 FAA Approved Airplane Flight Manual when the GND ICE DETR switch is pressed. Refer to Chapter 30-10-00 for the controller logic of the test.

DETECTION - MAINTENANCE PRACTICES (Effectivity: All)

ICE DETECTOR REMOVAL (Effectivity: All) (Figure 3)

NOTE

Removal of the ground ice detector and flight ice detectors are typical.

a. Remove all electrical power from the airplane.

b. Gain access to the ground ice detector by opening the forward nose landing gear doors and pinning the doors in the open position. Gain access to the flight ice detectors by removing the left (131AL) or right (132AR) nose skin panel as instructed in Chapter 53-30-00.

c. Disconnect the electrical connector.

d. Remove the five screws, washers, and nuts securing the ice detector to the skin panel.

e. Remove the ice detector from the airplane.

ICE DETECTOR INSTALLATION (Effectivity: All) (Figure 3)

NOTE

Installation of the ground ice detector and flight ice detectors are typical.

a. To install the ground ice detector, open the forward nose landing gear doors and pin the doors in the open position. The flight ice detectors are attached to the nose skin panels (131AL or 132AR) as instructed in Chapter 53-30-00.

b. Remove any old sealant from the ice detector and mounting plate.

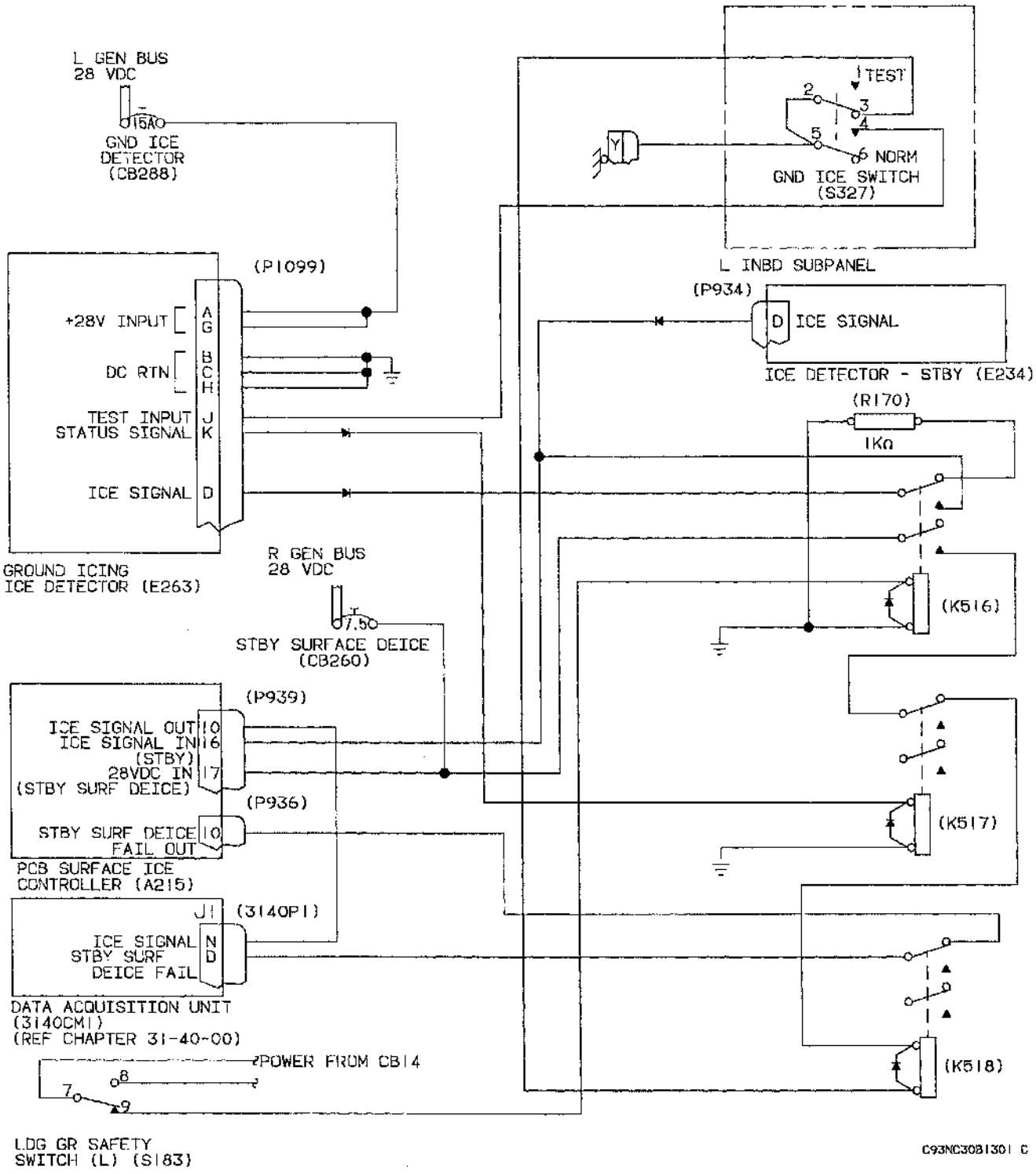
c. Apply a bead of sealant (14, Chart 2, 30-00-00) to the mounting plate.

d. Align the ice detector to the mounting plate and install five screws wet with sealant (14, Chart 2, 30-00-00), washers, and nuts.

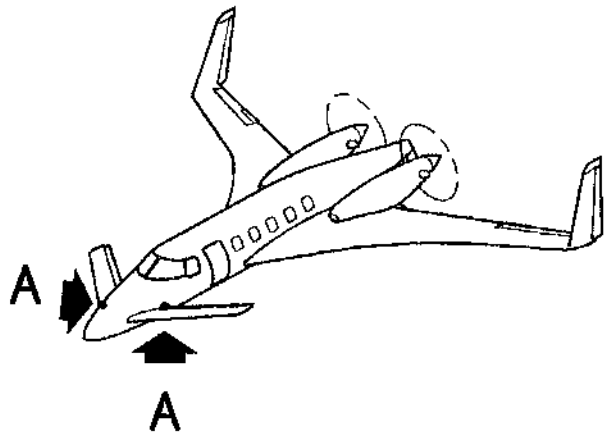
e. Connect the electrical connector to the ice detector.

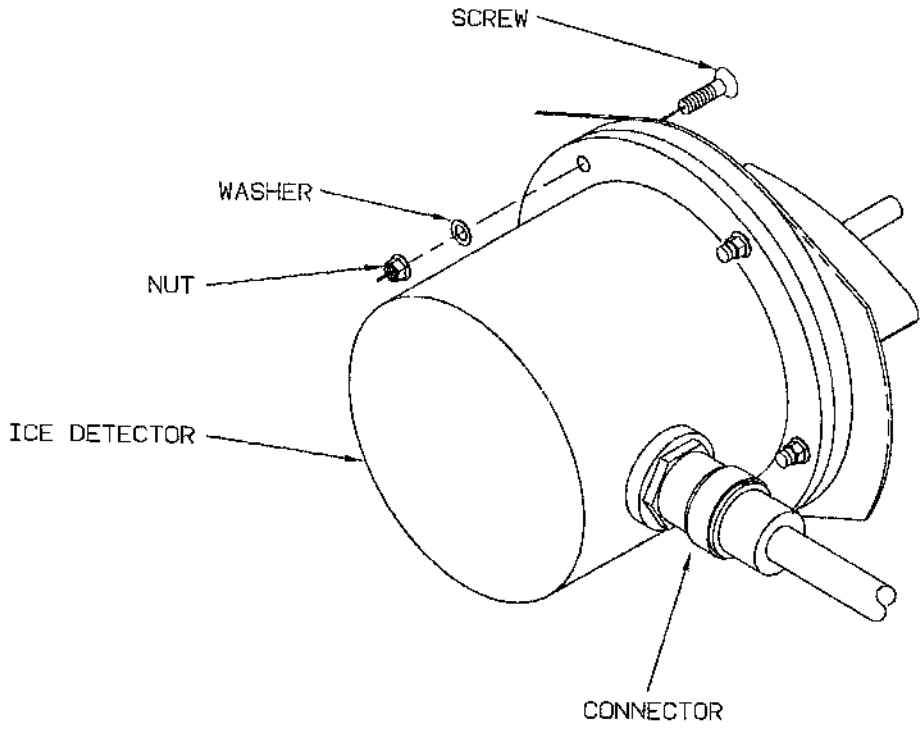
f. Restore electrical power to the airplane.

g. Remove the landing gear door safety pin and close the doors.



Ground Icing Detector Electrical Schematic
(Effectivity: NC-10, NC-44 And After, And Airplanes With BEECHCRAFT Kit No. 122-5024)
Figure 2





VIEW A

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Ice Detector Installation (Effectivity: All)
Figure 3

31 - INDICATING/RECORDING SYSTEMS

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Beechcraft
STARSHIP 1
MAINTENANCE MANUAL

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INSTRUMENT AND CONTROL PANELS - DESCRIPTION AND OPERATION
(Effectivity: All)
(Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

There are six large and eight small cathode ray tube (CRT) display units mounted on the instrument panel in the cockpit. The six large display units are part of the Instrument Display System (IDS). They consist of two Primary Flight Displays (PFD), two Navigation Displays (ND), a Multifunction Display (MFD) and an Engine Instrument, Crew Alerting System (EICAS) display. A cooling inlet is located on the back of the display units. The small CRTs consist of two Airspeed Indicators (ASI), two Altitude Indicators (ALI), two Control Display Units (CDU) and two Sensor Display Units (SDU). Two Radio Tuning Unit displays (RTU), three standby instruments and the pilot's and copilot's clocks are also mounted on the instrument panel in the cockpit.

The primary flight displays receive data bus inputs from the four input/output concentrators in the IAPS and from both attitude heading computers. They provide data bus outputs to the IAPS.

The navigation displays receive data bus inputs from the four input/output concentrators in the IAPS and from both attitude heading computers as well as from the weather radar. They provide data bus outputs to the IAPS.

The multifunction display receives data bus inputs from the four input/output concentrators in the IAPS, from both channels of the data acquisition unit, from the flight management computer (FMS, inside the IAPS), and from the weather radar system (WRX). They provide data bus outputs to the IAPS.

The EICAS display receives data bus inputs from the four input/output concentrators in the IAPS, and from both channels of the data acquisition unit. The EICAS provides a data bus output to the IAPS.

The airspeed and altitude indicators receive data bus inputs from the air data computers. The airspeed indicator transmits button/knob settings to the altitude indicator, where they are transmitted along with the altitude control settings to the air data computer.

Each control display unit is the primary controller for the primary flight displays, the navigation displays, the weather radar assembly, and the flight management system. The CDU can also tune the nav/com pulse radios.

Inputs to each CDU include ALT/MDA information from the appropriate course heading selector and a data bus input from the left FMC or right CDU coupler. The CDU output bus is applied to the left FMS computer or the right CDU coupler.

Each sensor display unit provides an RMI function and receives drive from the SDD sensor display driver. The SDD is a 2-channel processor. It receives inputs from both the left and right side radios, from both attitude/heading computers, and from a left and a right input/output concentrator in the IAPS. Each channel of the SDD provides drive for one SDU.

The left and right RTU's are the primary radio controls. Each RTU receives an input bus from both of the onside input/output concentrators of the IAPS, both left and right data engine concentrators, and one cross-side input/output concentrator.

Each RTU provides a tune bus output to both the left and right side com/nav/pulse radios.

Refer to Chapter 31-20-00 for detailed information on the clocks and to Chapters 24-32-00 and 34-00-00 for detailed information on the three standby instruments.

There are nine control panels mounted on the instrument panel in the cockpit. They consist of two reversionary control panels, two autopilot mode panels, two altitude awareness panels, an engine reversionary and audio volume control panel, the pressurization control panel and the course heading MFD joystick

control panel. Refer to the Collins Starship 1 Avionics System Manual (523-0774076-00111A or latest revision) for detailed information on operation and troubleshooting of the instrument panel display units and control panels.

INSTRUMENT AND CONTROL PANELS - MAINTENANCE PRACTICES (Effectivity: All)

Refer to Chapter 34-00-00 to remove the standby instruments and to Chapter 31-20-00 to remove the clocks. Refer to the Collins Starship 1 Avionics System Manual (523-0774076-00111A or latest revision) for additional information.

PRIMARY FLIGHT DISPLAY, MULTIFUNCTION DISPLAY AND ENGINE INSTRUMENT, CREW ALERTING SYSTEM DISPLAY UNIT REMOVAL (Effectivity: All) (Figure 2)

a. Remove all electrical power from the airplane.

b. Use an allen wrench to loosen the two integral turnlock fasteners (1 1/2 turns) on the lower corners of the display unit.

c. Reach under the instrument panel and locate the release lever at the bottom of the lower display rack.

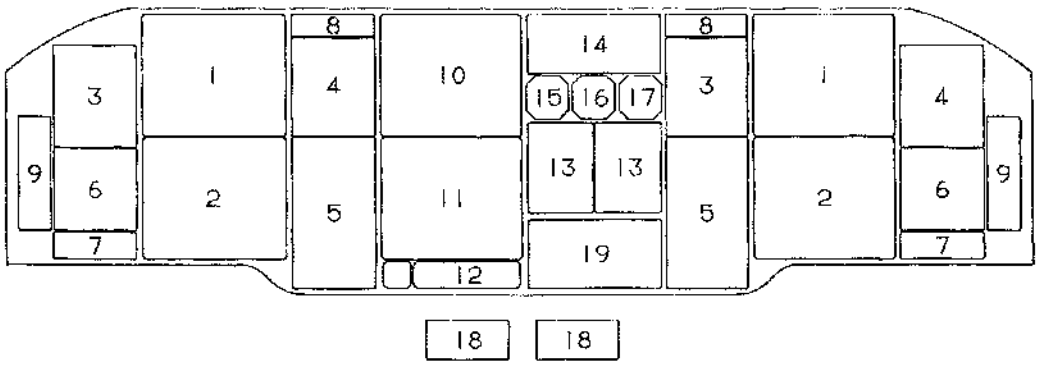
d. Rotate the release lever clockwise to release the PFD or EICAS display unit from its electrical connector. Rotate the release lever counterclockwise to release the MFD from its electrical connector.

e. Slide the display unit out of the rack.

PRIMARY FLIGHT DISPLAY, MULTIFUNCTION DISPLAY AND ENGINE INSTRUMENT, CREW ALERTING SYSTEM DISPLAY UNIT INSTALLATION (Effectivity: All) (Figure 2)

a. Ensure that all electrical power is removed from the airplane.

b. Reach under the instrument panel and locate the release lever at the bottom of the lower display rack.



1. PRIMARY FLIGHT DISPLAYS (PFD)
2. NAVIGATION DISPLAYS (ND)
3. AIRSPEED INDICATORS (ASI)
4. ALTITUDE INDICATORS (ALI)
5. CONTROL DISPLAY UNITS (CDU)
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11. MULTIFUNCTION DISPLAY (MFD)
12. COURSE HEADING MFD JOYSTICK CONTROL PANEL
13. RADIO TUNING UNITS (RTU)
14. ENGINE REVERSIONARY AND AUDIO VOLUME CONTROL PANEL
15. STANDBY AIRSPEED INDICATOR
16. STANDBY GYRO HORIZON INDICATOR
17. STANDBY PNEUMATIC ALTIMETER
18. AUTOPILOT MODE PANELS
19. PRESSURIZATION CONTROL PANEL

2000-392-05

Instrument Panel (Effectivity: All)
Figure 1

c. The release lever must be in a certain position to allow installation of the display unit. Rotate the release lever until it points outboard (parallel to the wing).

d. Insert the display unit in the mounting rack.

e. Firmly push the display unit into the mounting rack until the cooling duct and electrical connector are fully engaged.

f. Secure the display unit in the rack by using an allen wrench to tighten the two integral turnlock fasteners.

g. Restore electrical power to the airplane.

NAVIGATION DISPLAY UNIT REMOVAL (Effectivity: All) (Figure 2)

a. Remove all electrical power from the airplane.

b. Refer to PRIMARY FLIGHT DISPLAY, MULTIFUNCTION DISPLAY AND ENGINE INSTRUMENT, CREW ALERTING SYSTEM DISPLAY UNIT REMOVAL in this chapter and remove the PFD above the ND to be removed.

c. Use an allen wrench to loosen the two integral turnlock fasteners (1 1/2 turns) located on the lower corners of the ND.

d. Rotate the release lever counterclockwise to release the ND from the cooling duct and electrical connector.

e. Pull the control wheel back as far as it will go.

f. Loosen the two front screws on the middle shelf two full turns.

g. Pivot the middle shelf upward.

h. Carefully raise the ND up enough to clear the control wheel while pulling the ND from the lower rack.

NAVIGATION DISPLAY INSTALLATION (Effectivity: All) (Figure 2)

a. Ensure that all electrical power is removed from the airplane.

b. The release lever must be in a certain position to allow installation of the ND. Rotate the release lever until it points outboard (parallel to the wing).

c. Lower the back end of the ND over the control wheel and insert it in the lower rack.

NOTE

The middle shelf must be pivoted to the raised position and the control wheel must be pulled back as far as it will go.

d. Firmly push the ND into the mounting rack until the electrical connector and cooling duct are fully engaged.

e. Secure the ND in the rack by using an allen wrench to tighten the two integral turnlock fasteners.

f. Pivot the middle shelf downward and tighten the two screws to secure the middle shelf in place.

g. Refer to PRIMARY FLIGHT DISPLAY, MULTIFUNCTION DISPLAY AND ENGINE INSTRUMENT, CREW ALERTING SYSTEM DISPLAY UNIT INSTALLATION procedures in this chapter and install the applicable PFD.

h. Restore electrical power to the airplane.

AIRSPEED INDICATOR, ALTITUDE INDICATOR, SENSOR DISPLAY UNIT AND CONTROL DISPLAY UNIT REMOVAL (Effectivity: All)

a. Remove all electrical power from the airplane.

b. Use an allen wrench to remove the screws attaching the display unit to the instrument panel.

c. Pull the display unit far enough out to access the electrical connector on the back.

d. Tag, disconnect and stow the electrical connector.

e. Remove the display unit from the airplane.

AIRSPEED INDICATOR, ALTITUDE INDICATOR, SENSOR DISPLAY UNIT AND CONTROL DISPLAY UNIT INSTALLATION (Effectivity: All)

a. Ensure that all electrical power is removed from the airplane.

b. Reconnect the electrical connector to the display unit according to tags.

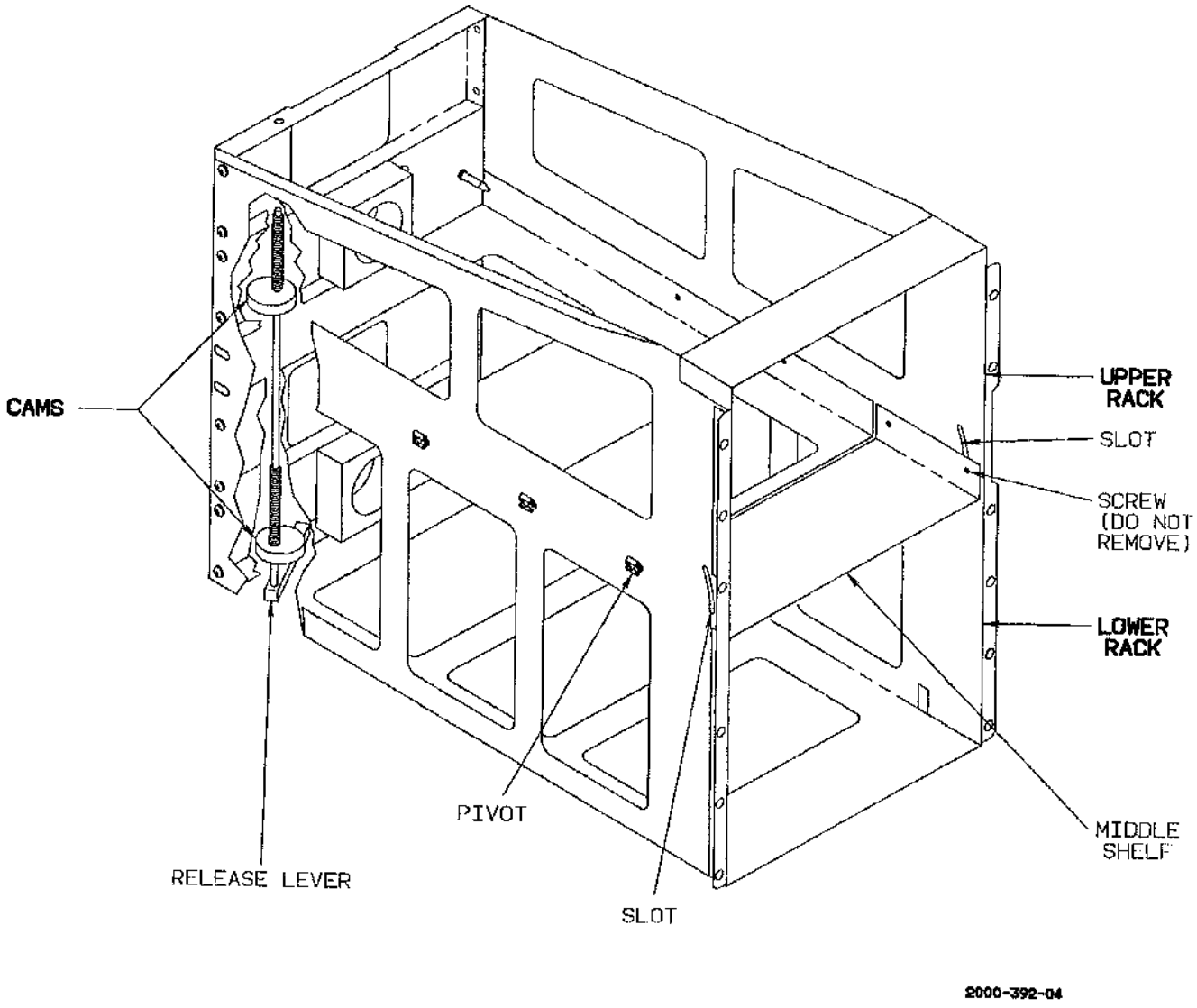
c. Insert the display unit in the instrument panel cutout.

d. Use an allen wrench to tighten the screws securing the display unit to the instrument panel.

e. Restore electrical power to the airplane.

RADIO TUNING UNIT REMOVAL (Effectivity: All)

a. Remove all electrical power from the airplane.



Primary Flight Display, Navigation Display, Multifunction Display and Engine Instrument, Crew Alerting System Display Unit Installation (Effectivity: All)
Figure 2

b. Using an allen wrench, turn the four integral turnlock fasteners counterclockwise until they stop.

c. Pull the RTU far enough out to access the electrical connector on the back.

d. Tag, disconnect and stow the electrical connector.

e. Remove the RTU from the airplane.

RADIO TUNING UNIT INSTALLATION (Effectivity: All)

a. Ensure that all electrical power is removed from the airplane.

b. Reconnect the electrical connector to the RTU according to tags.

c. Insert the RTU in the instrument panel cutout and use an allen wrench to tighten the four integral turnlock fasteners.

d. Restore electrical power to the airplane.

PRESSURIZATION CONTROL, AUTOPILOT MODE, REVERSIONARY CONTROL AND ENGINE REVERSIONARY PANEL REMOVAL (Effectivity: All)

a. Remove all electrical power from the airplane.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

b. Use an allen wrench to remove the screws attaching the control panel to the instrument panel.

c. Pull the control panel far enough out of the instrument panel to access the electrical connector.

d. Tag, disconnect and stow the electrical connector.

e. Remove the control panel from the airplane.

PRESSURIZATION CONTROL, AUTOPILOT MODE, REVERSIONARY CONTROL AND ENGINE REVERSIONARY PANEL INSTALLATION (Effectivity: All)

a. Ensure that all electrical power is removed from the airplane.

b. Reconnect the electrical connector and any wiring to the the back of the control panel according to tags.

c. Insert the control panel in the instrument panel cutout.

d. Use an allen wrench to tighten the screws securing the control panel to the instrument panel.

e. Restore electrical power to the airplane.

ALTITUDE AWARENESS AND COURSE HEADING MFD JOYSTICK CONTROL PANEL REMOVAL (Effectivity: All)

a. Remove all electrical power from the airplane.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

b. Use an allen wrench to loosen the turnlock fasteners (1 1/2 turns).

c. Pull the control panel far enough out of the instrument panel to access the electrical connector.

d. Tag, disconnect and stow electrical connector.

e. Remove the control panel from the airplane.

ALTITUDE AWARENESS AND COURSE HEADING MFD JOYSTICK CONTROL PANEL INSTALLATION (Effectivity: All)

a. Ensure that all electrical power is removed from the airplane.

b. Reconnect the electrical connector to the back of the control panel according to tags.

c. Insert the control panel in the instrument panel cutout.

d. Secure the control panel to the instrument panel by using an allen wrench to tighten the integral turnlock fasteners.

e. Restore electrical power to the airplane.

CATHODE RAY TUBE BRIGHTNESS CHECK (Effectivity: All)

This check is to be performed in accordance with the inspection interval specified in Chapter 4, or when the brightness of a CRT is questionable. The display unit containing the CRT should be removed from the airplane with procedures outlined in this chapter, then sent to a qualified Collins Avionics Service Center to have the brightness check performed.

INDEPENDENT INSTRUMENTS - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The Starship 1 independent instruments consist of two clocks (pilot's and copilot's) and the flight hour meter.

FLIGHT HOUR METER (Effectivity: All) (Figures 1 and 2)

The flight hour meter, placarded FLIGHT HOURS on the copilot's outboard subpanel, is an electromechanical device that counts total flight hours. The flight hour meter five digit counter is illuminated by the copilot's outboard subpanel electroluminescent panel.

Operating voltage and power return for the flight hour meter are controlled through two logic relays mounted on the logic relay PCB A134. When N1 (compressor shaft) speed is greater than 62 percent, relay A134K4 is energized, and triple-fed bus voltage from the No. 1 annunciator circuit breaker is applied to the flight hour meter. When the airplane begins flight, the left squat switch opens, and a ground is applied to the flight hour meter through logic relay A134K3. The flight hour meter will then begin adding flight time to the counter. When 9,999.9 hours is reached, the counter rolls back to 0,000.0.

CLOCKS (Effectivity: All) (Figure 1)

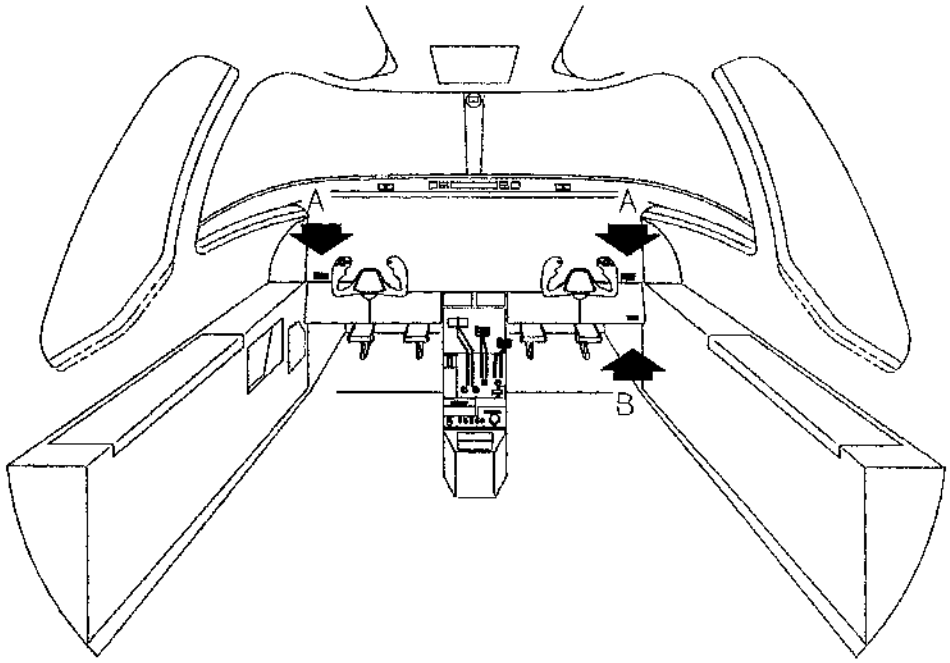
There are two interchangeable clocks located above the pilot's and copilot's outboard subpanels on the instrument panel in the cockpit. The pilot's clock functions as the master clock. Each clock has two liquid crystal displays (LCD) that indicate clock function and display time. Lighting voltage (5 vdc) to the clocks is controlled by the SUBPANEL DISPLAYS potentiometer on the overhead panel in the cockpit. Refer to Chapter 33-10-00 for detailed information on cockpit lighting. The clocks are continually powered by the "hot" battery bus for memory purposes and initialize with triple-fed bus voltage when the pilot's and copilot's avionics master switches are placed in the ON position.

The left LCD has three timer functions, and one set function controlled by four pushbutton switches located on the clock face. The functions are: trip time (TRP), stopwatch (SW), downcount (DC) and set (SET). There is also a display OFF mode where the left LCD is blank. The left LCD format is 00:00:00 for hours-minutes-seconds.

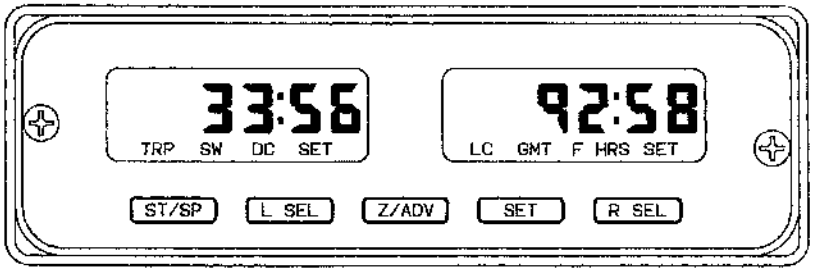
The TRP function counts trip time automatically for each flight and cannot be interrupted or changed with any of the push button switches. During engine start-up, the engine oil pressure switches send a 28 vdc signal to each clock, and the left LCD of each clock resets to the TRP function and displays the previous trip time from memory. As the airplane takes off, a 28 vdc signal is sent to the clocks from the landing gear squat switches. This signal resets trip time to 00:00:00 under the TRP function, and the display begins counting trip time. Active trip time is continually stored in volatile memory. When the airplane lands, the landing gear squat switches open, and the 28 vdc signal is removed from the clocks to stop the trip timer from counting. A 45-second delay is built into the TRP function, allowing touch-and-go landings without resetting the trip timer. After landing, TRP is stored in non-volatile memory for reference at the beginning of the next trip cycle.

The SW function is used to show elapsed time from any start point. The stopwatch timer is controlled with the ST/SP and Z/ADV push button switches. Once initiated, other clock functions may be selected without affecting stopwatch timer operation. Stopwatch time can be revised anytime; however, the value is returned to zero when the clock is initialized.

The DC function is used to count time down from a given starting point. This function is controlled using the SET, Z/ADV and ST/SP push button switches. A given amount of time is preset into the display by the operator. When ST/SP is pushed, the time begins

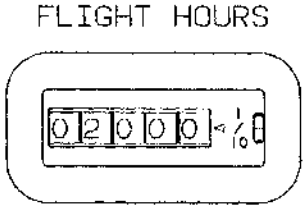


COCKPIT



CLOCK

DETAIL A



FLIGHT HOUR METER

DETAIL B

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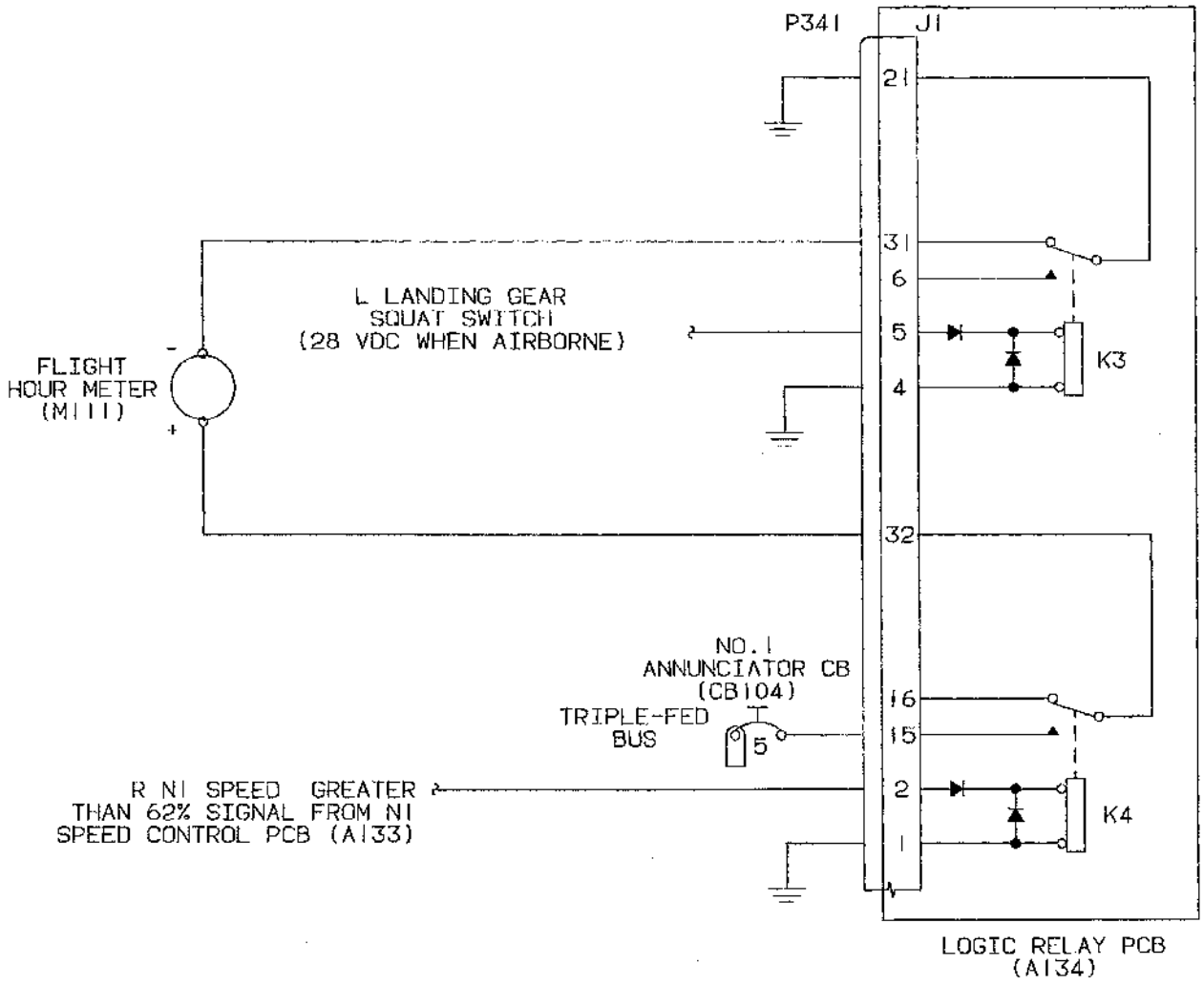
Independent Instruments (Effectivity: All)
Figure 1



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Flight Hour Meter Schematic (Effectivity: All)
Figure 2

counting down to zero in minutes, hours, and seconds, then counts upward from zero with a flashing display to denote that zero has been passed. Once initiated, other clock functions may be selected without affecting the DC timer operation. Downcount time can be revised anytime; however, the value is returned to zero when the clock is initialized.

RIGHT LCD (Effectivity: All)
(Figures 3 and 4)

The functions for the right LCD are: Greenwich Mean Time (GMT), local time (LC), total flight hours (F HRS) and set (SET). The right LCD has two display formats. The 00:00:00 format is used to display hours-minutes-seconds in the LC and GMT functions, or day-month-year in the DATE function. The 00000.0 format is used to display total flight hours up to 99999.9. The display OFF mode will cause the display to blank.

The LC function keeps independent time in a 24 hour format. Local time is automatically displayed when the triple-fed bus is powered.

NOTE

The minute and second values for local time are synchronized to Greenwich Mean Time.

The GMT function keeps independent time in a 24 hour format and automatically transmits GMT to the Data Acquisition Unit (DAU). The DAU codes and transmits GMT to the ARINC 429 data bus, where it is referenced by the avionics equipment.

NOTE

The copilot's clock GMT display is synchronized with the pilot's clock GMT display. The copilot's clock configuration does not allow it to assume clock data transmissions if the pilot's clock should fail.

The F HRS function automatically accumulates total flight hours. As the airplane takes off, the landing gear squat switches close and a 28 vdc signal to the clocks starts the counting of flight hours. When the airplane lands, the landing gear squat switches open and remove the 28 vdc signal from the clocks. After a 45-second delay, the TRP value is rounded off to the nearest tenth of an hour and added to the previous flight hour total to obtain a running total.

The DATE function maintains the date internally and automatically transmits it to the Data Acquisition Unit (DAU). The DAU codes and transmits the date to the ARINC 429 data bus, where it is referenced by the avionics equipment.

The SET function is enabled when the SET push button switch is pressed, and works independently on displayed LC or GMT. The SET push button switch is used in conjunction with the rotary set switch on the back of the clock to set the date and adjust the flight hour total.

CLOCK CONTROLS (Effectivity: All)
(Figure 3)

The five push button switches on the face of the clock, and the rotary set switch on the back, select the display functions and control the setting of both LCDs. A function acronym is shown at the bottom of each display to indicate the function mode. The push button switches and rotary set switch perform the following:

START/STOP (ST/SP) (Effectivity: All)

The ST/SP push button switch works on the left LCD only. ST/SP starts and stops the SW or DC timer functions but has no effect on the TRP function.

LEFT LCD SELECT (L SEL) (Effectivity: All)

The L SEL push button switch cycles the left LCD through each timer function and the display OFF mode. The functions will cycle in the following order: TRP, SW, DC, blank display, and back to TRP. L SEL has no effect when the rotary set switch on the back of the clock is in the SET DATE or F HRS position.

RIGHT LCD SELECT (R SEL) (Effectivity: All)

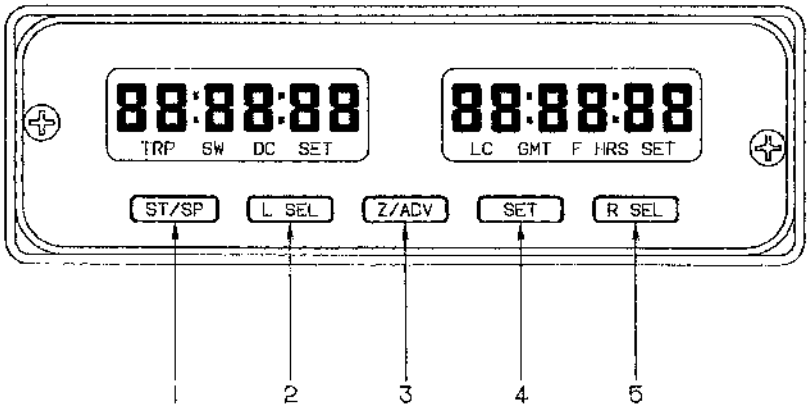
The R SEL push button switch cycles the right LCD through each timer function and the display OFF mode. The functions will cycle in the following order: LC, GMT, F HRS, blank display, and back to LC. R SEL has no effect when the rotary set switch on the back of the clock is in the SET DATE or F HRS position.

ZERO/ADVANCE (Z/ADV) (Effectivity: All)

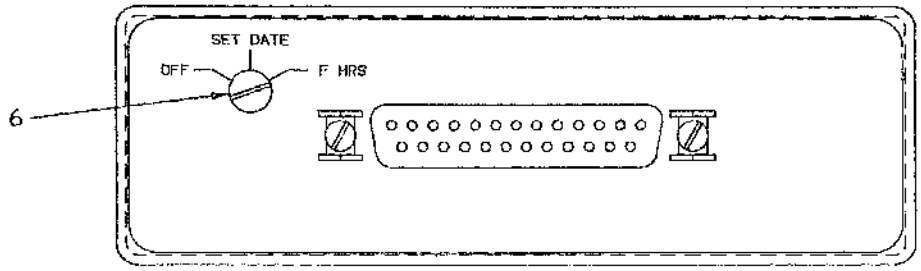
The Z/ADV push button switch will zero the displayed value for the SW and DC timer functions only. Z/ADV will advance the flashing digits to be revised one count each time it is pressed for the LC, GMT, DC, DATE, and F HRS functions. Holding Z/ADV in longer than one second will advance the digits 6 counts per second.

SET (SET) (Effectivity: All)

The SET push button switch enables and disables the SET function for setting LC, GMT, and DC time, but



FRONT VIEW



REAR VIEW

ITEM	NOMENCLATURE	DEFINITION
1	ST/SP	START/STOP
2	L SEL	LEFT LCD SELECT
3	Z/ADV	ZERO/ADVANCE
4	SET	SET
5	R SEL	RIGHT LCD SELECT
6	SET DATE/F HRS	SET DATE/FLIGHT HOURS

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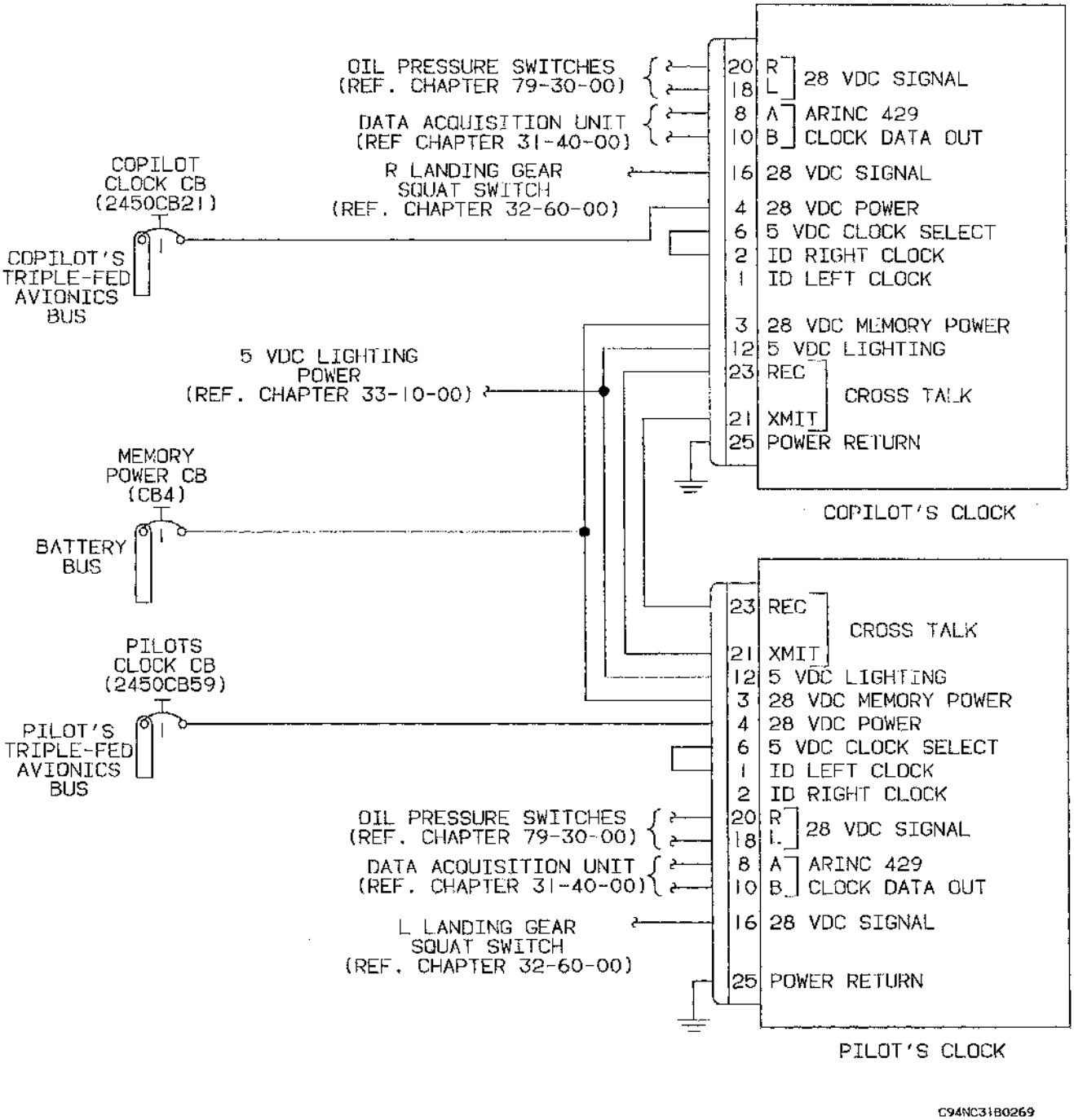
Clock Controls (Effectivity: All)
Figure 3



Pilot & Copilot Clock Signals (Effectivity: All)
Figure 4

has no effect on the SW or TRP functions. When the SET push button switch is pressed, the digits ready for revision will flash ON and OFF. The SET push button switch is also used in conjunction with the rotary set switch on the back of the clock to revise the date or flight hours. If no change has been made in the SET function, the timer or clock functions will resume as if there had been no interruption.

DATE (DATE) AND FLIGHT HOURS (F HRS) (Effectivity: All)

The rotary set switch on the back of the clock is used to revise the DATE and F HRS. The rotary set switch has no effect until the landing gear squat switches are closed, and both engine oil pressure switches are open, indicating the airplane is not in a flight mode. In the OFF position, all functions controlled by the push button switches on the front of the clock are operational. With the rotary set switch moved out of the OFF position, the internal battery is disabled, and other specific internal interrupts and resets take over. Only the date or flight hours can be displayed or revised on the right LCD in this mode. No other functions are operational until the rotary set switch on the back is returned to the OFF position.

CLOCK INTERNAL BATTERY (Effectivity: All)

An internal battery automatically powers the clock up to 500 hours, should voltage from the hot battery bus drop below 10 vdc. When the clock is forced to rely on operating voltage from its internal battery, both LCDs blank, the SW and DC timers stop and zero, and the TRP and F HRS values stored in volatile memory are stored in non-volatile memory. The DATE, LC and GMT continue to update and transmit, although they are not displayed. When voltage from the clock's internal battery drops to 0 vdc, all functions stop and zero. When voltage from the hot battery bus increases above 18 vdc, the internal battery begins to charge, and the TRP and F HRS times are entered in volatile memory. All other clock functions return to normal operation.

CAUTION

If the clock is to be without memory power from the hot battery bus for an extended time, the internal battery may be preserved by placing the set switch on the back in either the SET DATE or F HRS position.

INDEPENDENT INSTRUMENTS - TROUBLESHOOTING (Effectivity: All)

The BEECHCRAFT Starship 1 Wiring Diagram Manual should be used to trace out specific wire routes and locate pin and plug numbers. Many times a loose or faulty connection will cause a fault. Ensure that there are no loose, damaged, or corroded pins on any connectors in the system. Continuity checks should be made with power removed from the specific terminal or connector. Use a calibrated voltmeter for all voltage checks and ensure that the battery switch is ON during voltage checks unless otherwise indicated. Refer to the applicable chart in this chapter to isolate any faults. The charts indicate the problem, probable cause, and remarks to isolate or correct the problem. Refer to Chapter 32-60-00 to test or adjust the landing gear squat switches. Refer to Chapter 79-30-00 for information on the engine oil pressure switches.

FLIGHT HOUR METER TROUBLESHOOTING (Effectivity: All)

Troubleshooting the flight hour meter consists of checking the wiring circuits, including the logic relay PCB A134 inputs and outputs. The landing gear squat switch and engine oil pressure switch signals used to energize logic relays A134K3 and A134K4 are generated from in-flight conditions. Therefore, troubleshooting with the engines shut down and the airplane on the ground requires bypassing the logic relay PCB A134 and providing 28 vdc signals straight to the flight hour meter to simulate in-flight conditions. Refer to LOGIC RELAY A134K3 AND A134K4 FUNCTIONAL TEST in this chapter to determine if the relays are functioning properly.

CLOCK TROUBLESHOOTING (Effectivity: All)

Troubleshooting the clock consists of checking the required power inputs and associated wiring. Use the setting procedures in this chapter to validate the internal operation.

CHART 1
INDEPENDENT INSTRUMENTS - TROUBLESHOOTING (Effectivity: All)
FLIGHT HOUR METER

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. The flight hour meter does not operate in flight.	a. No. 1 annunciator circuit breaker is open.	a. Close circuit breaker.
	b. No. 1 annunciator circuit breaker is not functioning properly.	b. Check for continuity across the circuit breaker contacts. Replace the circuit breaker if required.
	c. Open circuit between logic relay PCB A134 and ground.	c. Check for continuity to ground.
	d. Relay A134K4 is not receiving 28 vdc to energize.	d. Check airplane wiring for 28 vdc from N_1 speed control PCB A133 when right N_1 speed is greater than 62%.
	e. Relay A134K4 contacts do not close when 28 vdc is applied to the coil.	e. Replace logic relay PCB A134.
	f. Relay A134K3 is not de-energizing as required.	f. Check left landing gear squat switch for proper actuation (open).
	g. Relay A134K3 contacts do not close when voltage is removed from the coil.	g. Replace logic relay PCB A134.
	h. Open circuit between flight hour meter and logic relay PCB A134.	h. Check airplane wiring for continuity.
	i. Flight hour meter is inoperative.	i. Replace flight hour meter.
2. Flight hour meter does not stop running.	a. Relay A134K3 is not receiving 28 vdc to energize.	a. Check left landing gear squat switch for proper actuation (closed).
	b. Relay A134K3 is not operating properly.	b. Replace logic relay PCB A134.
	c. Relay A134K4 is not de-energizing as required.	c. Check that voltage signal from N_1 speed control PCB A133 is removed from relay A134K4 when engine N_1 speed drops below 62%.
	d. Relay A134K4 is not operating properly.	d. Replace logic relay PCB A134.

CHART 2
INDEPENDENT INSTRUMENTS - TROUBLESHOOTING (Effectivity: All)
CLOCKS

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Clock does not display.	a. PILOT CLOCK or COPILOT CLOCK circuit breaker is open.	a. Close circuit breaker.
	b. PILOT CLOCK or COPILOT CLOCK circuit breaker is not functioning properly.	b. Check for continuity across the contacts of the circuit breaker and replace if necessary.
	c. Clock is not receiving 28 vdc operating voltage.	c. Check airplane wiring.
	d. Avionics MEMORY POWER circuit breaker is open.	d. Close circuit breaker.
	e. Avionics MEMORY POWER circuit breaker is not functioning properly.	e. Check for continuity across the contacts of the circuit breaker and replace if necessary.
	f. "Hot" battery bus voltage to the avionics MEMORY POWER circuit breaker has dropped below 10 vdc and the clock has internally shut off displays.	f. Press the BATT volts switch on the center subpanel and read battery voltage on the voltmeter. If the battery voltage is below 20 vdc, refer to Chapter 24-31-00 and remove the battery for charging.
	g. Clock not functioning properly.	g. Replace clock.
2. Clock displays are incomplete.	a. Clock not functioning properly.	a. Replace clock.
3. Clock does not keep satisfactory time.	a. Clock not functioning properly.	a. Replace clock.
4. Clock does not automatically display TRP and LC functions when engines are started.	a. Right and left engine oil pressure switch 28 vdc signals are not present.	a. Check airplane wiring for continuity and the engine oil pressure switches for proper actuation points.
	b. Clock not functioning properly.	b. Replace clock.
5. Clock does not automatically zero TRP and start counting trip time on take-off.	a. Landing gear squat switch 28 vdc signals are not present.	a. Check airplane wiring for continuity and the landing gear squat switches for proper actuation.
	b. Clock not functioning properly.	b. Replace clock.
6. The TRP function does not stop running when the airplane lands.	a. The landing gear squat switch 28 vdc signals are not present.	a. Check airplane wiring for continuity and the landing gear squat switches for proper actuation.
7. The F HRS function does not update after landing.	a. Clock not functioning properly.	a. Replace clock.
8. Clock controls do not respond.	a. Clock not functioning properly.	a. Replace clock.

INDEPENDENT INSTRUMENTS - MAINTENANCE PRACTICES (Effectivity: All)

FLIGHT HOUR METER REMOVAL (Effectivity: All) (Figure 5)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 33-10-00 and remove the electroluminescent panel from the copilot's outboard subpanel.

c. Remove the copilot's outboard subpanel.

d. Pull the subpanel out far enough to gain access to the two wires on back of the flight hour meter.

e. Remove two screws, washers and nuts attaching the flight hour meter to the copilot's outboard subpanel.

f. Tag and remove wiring from the flight hour meter.

g. Remove the flight hour meter from the airplane.

FLIGHT HOUR METER INSTALLATION (Effectivity: All) (Figure 5)

a. Reconnect wiring to the flight hour meter according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

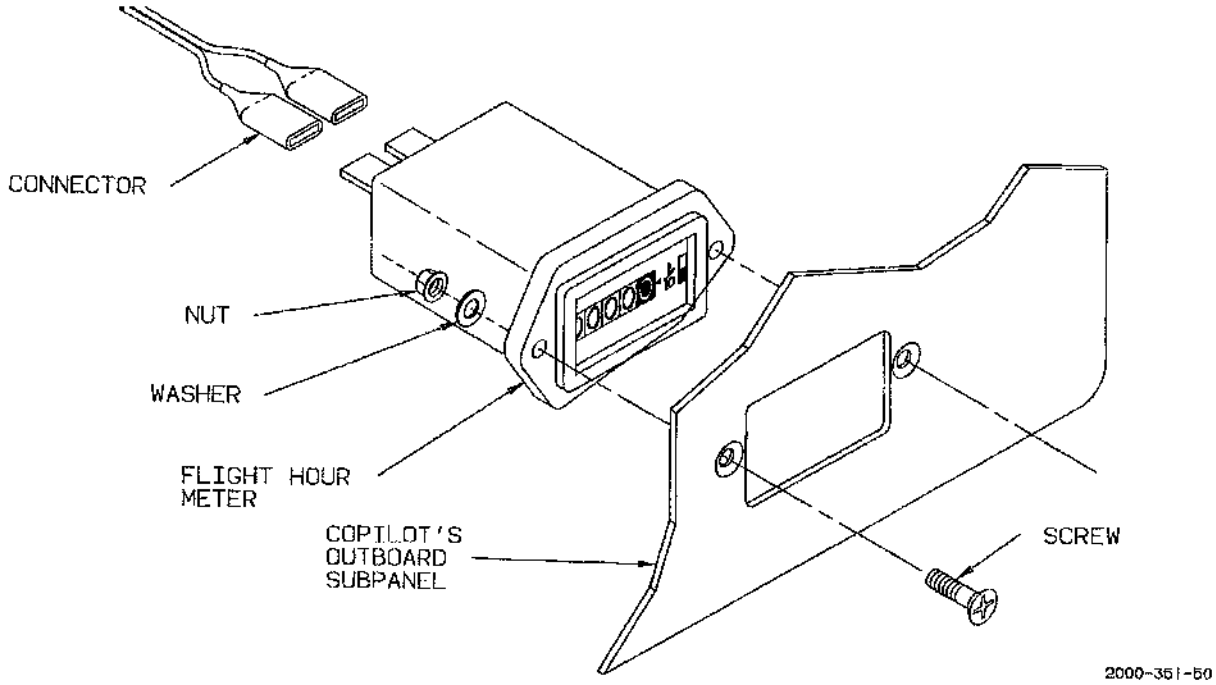
b. Align the flight hour meter with the mounting holes on the copilot's outboard subpanel.

c. Install two screws from the front of the panel and secure the flight hour meter to the panel by installing washers and nuts on the screws.

d. Install the copilot's outboard subpanel.

e. Refer to Chapter 33-10-00 and install the copilot's outboard subpanel electroluminescent panel.

f. Restore electrical power to the airplane.



Flight Hour Meter Installation (Effectivity: All)
Figure 5

CLOCK REMOVAL (Effectivity: All) (Figure 6)

a. Remove all electrical power from the airplane.

b. Turn the two screws on the clock face 1/2 turn counterclockwise to unlock the mounting feet.

c. Pull the clock far enough out of the instrument panel to access the electrical connector on the back of the clock.

d. Remove the two electrical connector mounting screws, and disconnect the electrical connector from the clock.

e. Remove the clock from the airplane.

CLOCK INSTALLATION (Effectivity: All) (Figure 6)

a. Reconnect the electrical connector and secure it to the back of the clock by installing two screws.

b. Align the clock with the instrument panel cut-out.

c. Turn the two screws on the clock face 1/2 turn clockwise to engage the mounting feet.

CLOCK SETTING PROCEDURES (Effectivity: All)

The following procedures are used to revise the timer functions, clock functions, flight hours and internal calendar date. The stopwatch timer (SW), downcount timer (DC), local time (LC) and Greenwich Mean Time (GMT) are changed with the push button switches on the face of the clock. The date (DATE) and flight Hours (F HRS) are revised only when the rotary set switch on the back is in the SET DATE or F HRS position.

NOTE

The hot battery bus and pilot's and copilot's triple-fed avionics buses must be powered to perform clock setting procedures.

STOPWATCH TIMER (SW) (Effectivity: All)

a. Press L SEL until the SW function is selected.

b. Press ST/SP to start the timer.

c. Press ST/SP to stop the timer.

d. Press Z/ADV to reset the timer to zero.

DOWNCOUNT TIMER (DC) (Effectivity: All)

a. Press L SEL until the DC function is selected.

b. Press SET. The hours flash, and SET is displayed in the lower RH corner. Minutes or seconds may be selected by pressing SET again. The digits subject to revision are indicated by a flashing display.

c. Press Z/ADV to zero and advance the selected digits to the desired setting.

NOTE

The selected digits zero when Z/ADV is first pressed. The digits will advance when Z/ADV is pressed a second time.

d. Press SET. The display stops flashing and SET is no longer displayed in the lower RH corner.

e. Press ST/SP to start the DC timer. The display counts down to zero, then begins to flash while counting upward to denote that zero has been passed.

f. Press ST/SP to stop the DC timer.

g. Press Z/ADV to reset DC timer to zero.

GREENWICH MEAN TIME (GMT) (Effectivity: All)

a. Press L SEL to select the TRP or SW function on the left display.

NOTE

If the left display is in the DC mode, SET will cycle through the DC digits before selecting digits on the right display.

b. Press R SEL until the GMT function is selected.

c. Press SET. The hours flash, and SET is displayed in the lower RH corner.

d. Press Z/ADV and advance the hours to the desired value.

e. Press SET again. The hours stop flashing and the minutes begin to flash.

f. Press Z/ADV and advance the minutes to the desired value.

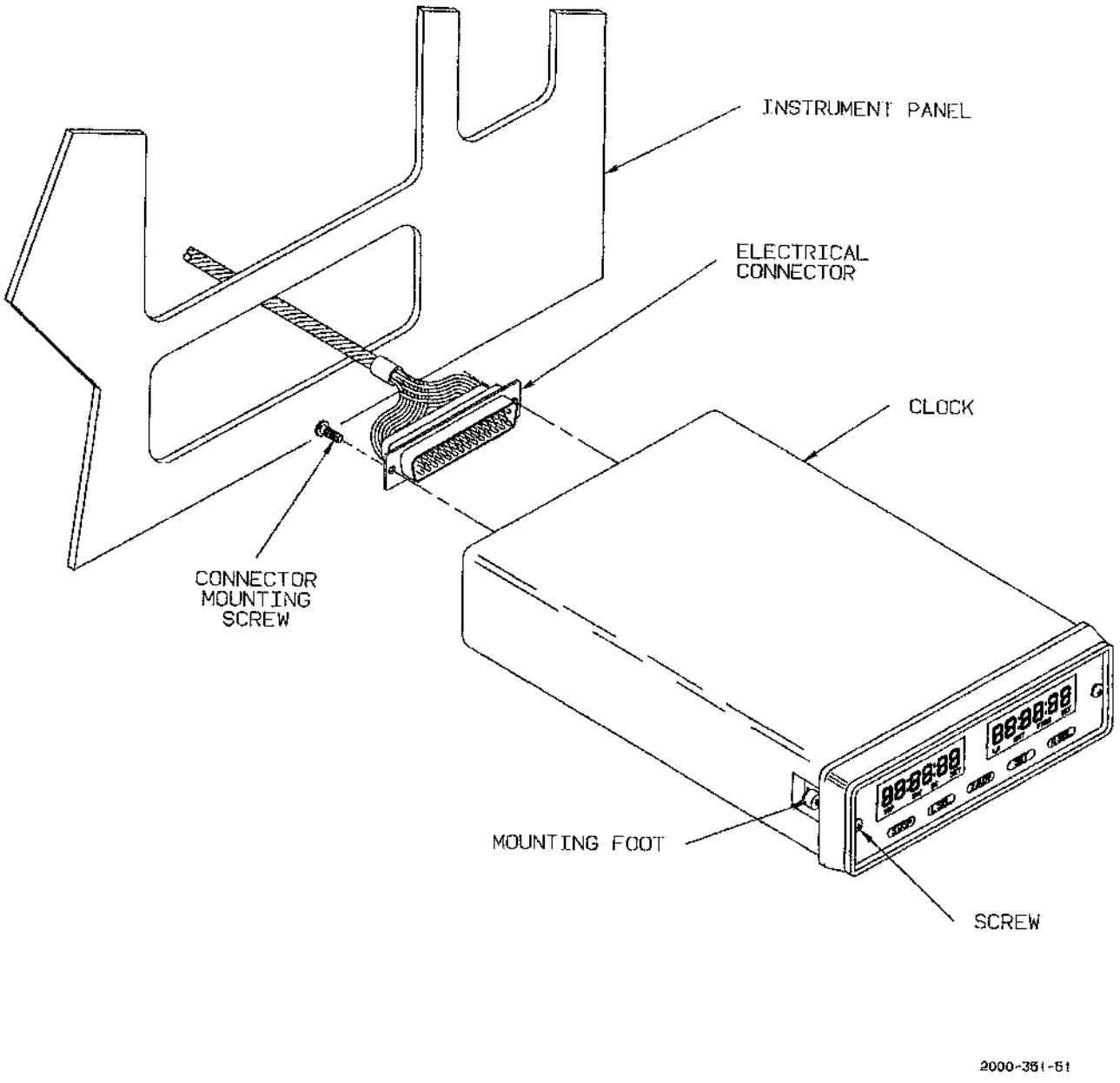
g. Press SET. SET is no longer displayed in the lower RH corner, the minutes stop flashing, and seconds begin counting from zero.

LOCAL TIME (LC) (Effectivity: All)

a. Press L SEL to select the TRP or SW function on the left display.

NOTE

If the left display is in the DC mode, SET will cycle through the DC digits before selecting digits on the right display.



Clock Installation (Effectivity: All)
Figure 6

b. Press R SEL until the LC function is selected.

c. Press SET. The hours flash, and SET is displayed in the lower RH corner.

d. Press Z/ADV and advance the hours to the desired value.

e. Press SET. The hours stop flashing and SET is no longer displayed in the lower RH corner. Minutes and seconds for local time are not set, but are automatically synchronized with the clock's current Greenwich Mean Time setting.

TOTAL FLIGHT HOURS (F HRS) (Effectivity: All)

a. Turn the two screws on the clock face 1/2 turn counterclockwise and pull the clock out far enough to access the rotary set switch on the back.

b. Use a small screwdriver to move the rotary set switch to the F HRS position. The left display will blank, and the right display will display total flight hours from memory in hours and tenths of hours. The ten-thousands and thousands digits will flash.

c. Press Z/ADV to advance the digits to the desired value.

d. Press SET to select the next pair of digits for revision.

e. Press Z/ADV to advance the digits to the desired value.

f. Press SET to select the last pair of digits for revision.

g. Press Z/ADV to advance the digits to the desired value.

h. Use a small screwdriver to move the rotary switch back to the OFF position.

NOTE

The clock will not operate if the rotary set switch on the back is not returned to the OFF position.

i. Align the clock with the instrument panel cut-out and turn the two screws on the clock face 1/2 turn clockwise to engage the mounting feet.

DATE (DATE) (Effectivity: All)

a. Turn the two screws on the clock face 1/2 turn counterclockwise and pull the clock out far enough to access the rotary set switch on the back.

b. Use a small screwdriver to move the rotary set switch to the SET DATE position. The left display will blank, and the right display will display the month, day and year, with the month digits flashing.

c. Press Z/ADV to advance the months.

d. Press SET. The day digits will flash.

e. Press Z/ADV to advance the days.

f. Press SET. The year digits will flash.

g. Press Z/ADV to advance the years.

h. Use a small screwdriver to move the rotary switch back to the OFF position.

NOTE

The clock will not operate if the rotary set switch on the back is not returned to the OFF position.

i. Align the clock with the instrument panel cut-out and turn the two screws on the clock face 1/2 turn clockwise to engage the mounting feet.

LOGIC RELAY A134K3 AND A134K4 FUNCTIONAL TEST (Effectivity: All) (Figure 7)

The test procedure for the logic relays consists of applying 28 vdc from a DC power source to close the logic relays, and then checking for continuity and infinity. Defective diodes on the PCB will prevent the logic relays from closing. Relay contacts may corrode and pit which will cause high resistance readings. Replace any PCB that has abnormal readings.

NOTE

Any inputs or outputs to relays A134K3 and A134K4 that are not flight hour meter related are not tested with this procedure.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-10-00 and remove the pilot's seat and console upholstery panel.

c. Ground pins 1 and 4 on the PCB to the airplane ground plane.

d. Connect a single-pole single-throw test switch (S2) to pin 2 of the PCB.

e. Connect a single-pole single-throw test switch (S1) to pin 5 of the PCB.

f. With S1 and S2 open, connect a 28 vdc power source to the contact of each switch not connected to the PCB.

g. Connect an ohmmeter between pins 21 and 31 of the PCB. The ohmmeter should indicate continuity.

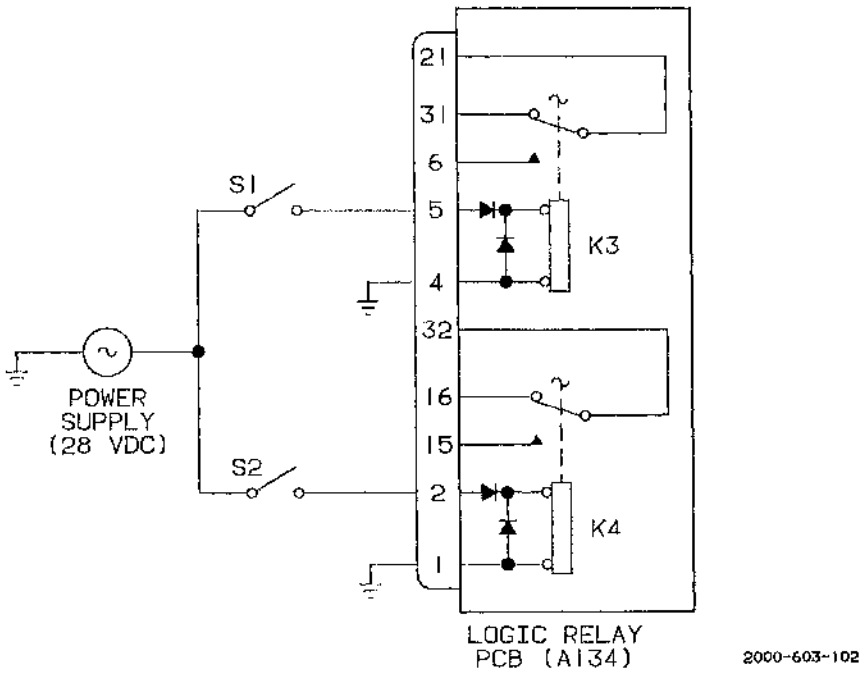
h. Close S1. The ohmmeter should indicate infinity.

i. Connect the ohmmeter between pins 32 and 15 of the PCB. The ohmmeter should indicate infinity.

j. Close S2. The ohmmeter should indicate continuity.

k. Disconnect the power supply and test switches from the PCB.

l. If the readings obtained are other than specified, refer to Chapter 39-00-00 and replace the logic relay PCB A134.



Logic Relay A134K3 and A134K4 Test Circuit Schematic (Effectivity: All)
Figure 7

CENTRAL COMPUTERS - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The central computer equipment consists of an Engine Data Concentrator (EDC) for each engine and a Data Acquisition Unit (DAU). This equipment is commonly refered to as the Aircraft Data Acquisition System (ADAS).

The ADAS encodes engine parameters and airplane discretes for use by the Instrument Display System (IDS) and the Integrated Avionics Processor System (IAPS). It also decodes aural warning information from the IAPS and caution/warning lamp driver information from the IDS and applies corresponding discrete outputs to the various airplane sensors.

EDC 1 gathers discrete data from the No. 1 engine and various airplane sensors. This data is converted to digital format and transmitted on two channels (A and B) to the DAU and to the Communication System (CS) for use in the reversionary mode.

EDC 2 performs the same function as EDC 1 except for the No. 2 engine.

The DAU is a dual-redundant data accumulator. It provides bidirectional interface between the avionics systems and the airplane engines, sensors, and warning/caution systems. The DAU contains two identical channels (A and B) which are isolated from each other to provide independent operation. Each channel processes engine data, airplane sensor/switch data, and clock data into digital bus outputs. These outputs are then applied to the IDS and IAPS systems. Each channel also processes IDS and IAPS data bus inputs into aural warning and exceedance caution/warn lamp driver discrete outputs.

Each clock provides synchronized data to one channel of the DAU. The clocks provide time and date reference to the processors.

The Attitude Heading Reference System (AHRS) computers monitor the landing gear strut logic to determine when the airplane is on the ground. When either attitude heading computer is initializing, an INIT discrete output is applied to the DAU.

ENGINE DATA CONCENTRATOR (EDC) (Effectivity: All)

The EDCs are located underneath the center section of the wing at locations 551AB and 651AB. Refer to chapter 6-50-00.

Each EDC receives +28 vdc power from two separate sources. This prevents the loss of either power supply from totally disabling the EDCs. EDC 1 receives power from L EDC CHA(CB219) and L EDC CHB(CB12) while EDC 2 receives power from R EDC CHA(CB220) and R EDC CHB(CB13). The EDCs are identified to the EDC software by strapping two pins on the rear connectors to ground. EDC 1 has P1-46 and P2-46 strapped to ground while EDC 2 has P1-38 and P2-38 strapped to ground.

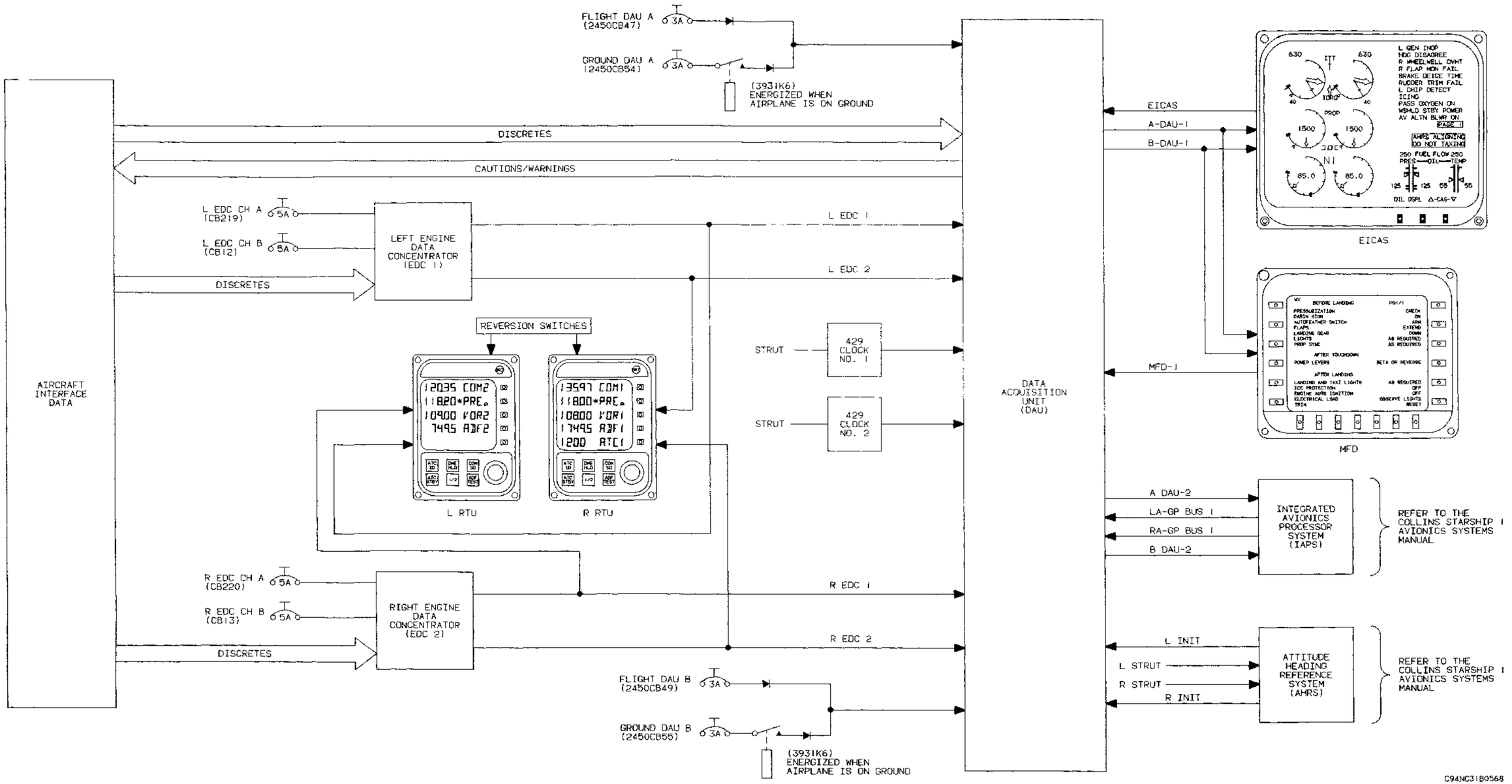
The EDCs convert analog engine data and airplane sensor data to digital format and collects discrete switch logic from various airplane switches. The digitized data is concentrated and then transmitted to the DAU and Radio Tuning Units (RTU) in the communications system.

DATA ACQUISITION UNIT (DAU) (Effectivity: All)

The DAU is located in the cockpit underneath the compartment coverplate 143CTC. Refer to chapter 6-50-00.

The DAU is a dual-redundant data accumulator that provides engine and airplane sensor data to the IAPS/IDS systems and provides exceedance discretes to airplane warn/caution annunciators and aural devices.

The DAU is divided in half. The left half is the A-side while the right half is the B-side. Each half of the DAU



Central Computer System Block Diagram
(Effectivity: All)
Figure 1

is an independent unit functioning with a separate power supply, digital bus input, discrete input, microprocessor, digital bus output, and discrete output circuits. Channel isolation is maintained so that a failure in either half of the DAU will not affect the operation of the other half.

Power for DAU A is provided by two circuits, one for ground operation and one for operation in flight.

When the airplane is on the ground with neither generator on line, power to DAU A is supplied from the battery through the center bus and the EICAS center bus feeder/circuit breaker.

The EICAS CENTER BUS FEEDER, GROUND DAU A, EICAS AUX BATT, and L SQUAT SWITCH circuit breakers must be engaged to complete the application of power to DAU A.

The input power is applied through an internal 4-amp fuse to the power supply circuit. This circuit generates the required internal power for A-side operation.

Power for DAU B is provided by two circuits, one for ground operation and one for operation in flight.

When the airplane is on the ground with neither generator on line, power to DAU B is supplied from the battery through the center bus and the EICAS center bus feeder/circuit breaker.

The EICAS CENTER BUS FEEDER, GROUND DAU B, EICAS AUX BATT, and L SQUAT SWITCH circuit breakers must be engaged to complete the application of power to DAU B.

The input power is applied through an internal 4-amp fuse to the power supply circuit. This circuit generates the required internal power for B-side operation.

Both the A-side and the B-side of the DAU operate in an identical manner.

CENTRAL COMPUTERS - TROUBLESHOOTING (Effectivity: All)

For assistance in troubleshooting this system, refer to the latest revision following:

- Collins Starship 1 Avionics System Manual.
- Chapter 77-40-00 of this manual.
- Starship 1 Wiring Diagram Manual.
- Starship 1 Avionics Wiring Diagrams.

CENTRAL COMPUTERS - MAINTENANCE PRACTICES (Effectivity: All)

ENGINE DATA CONCENTRATOR (EDC) REMOVAL (Effectivity: All) (Figure 2)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 57-00-00 and remove the appropriate wing access panel (551AB or 651AB).

c. Tag, disconnect and remove the two electrical connectors from the EDC (1).

d. Remove the screws (2), and washers (3) from the EDC (1).

e. Remove the EDC (1) from the airplane.

ENGINE DATA CONCENTRATOR (EDC) INSTALLATION (Effectivity: All) (Figure 2)

NOTE

If an EDC is REPLACED, perform the EDC TORQUE CALIBRATION procedure in Chapter 77-00-00.

a. Remove all electrical power from the airplane.

b. Position the EDC (1) in place with the connectors pointing inboard.

c. Install washers (3), and screws (2) to secure the EDC (1).

d. Untag and connect the two electrical connectors.

e. Refer to Chapter 77-40-00 in this manual and perform the ENGINE DATA SIMULATION TEST and test per the level 2 troubleshooting procedure as described in the Collins Starship 1 Avionics System Manual.

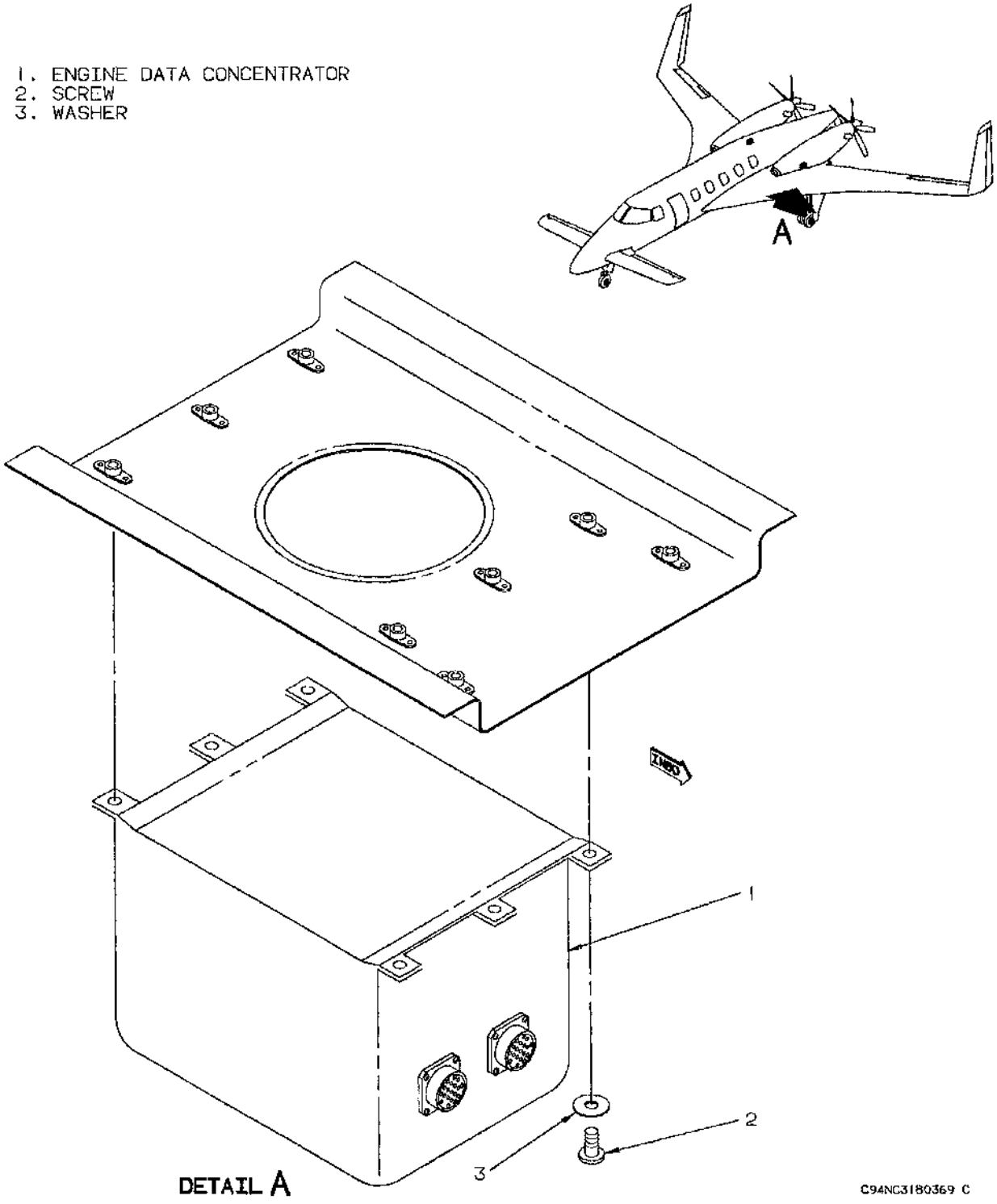
f. Refer to Chapter 57-00-00 and install the appropriate wing access panel (551AB or 651AB).

DATA ACQUISITION UNIT (DAU) REMOVAL (Effectivity: All) (Figure 3)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-20-00 and remove the cockpit floorboard panel (143CTC).

c. Tag and disconnect the four electrical connectors from the DAU (1).



Engine Data Concentrator (EDC) Installation (Effectivity: All)
Figure 2

d. Remove the screws (2), lockwashers (3) and washers (4) and remove the DAU (1) from the airplane.

DATA ACQUISITION UNIT (DAU) INSTALLATION (Effectivity: All) (Figure 3)

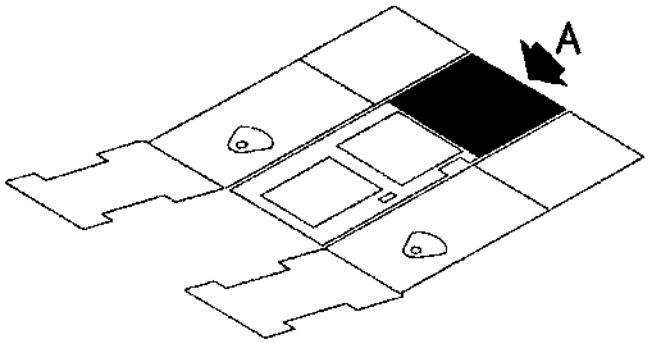
a. Set the DAU (1) in place and install the washers (4), lockwashers (3) and screws (2).

b. Untag and connect the four electrical connectors to the DAU (1).

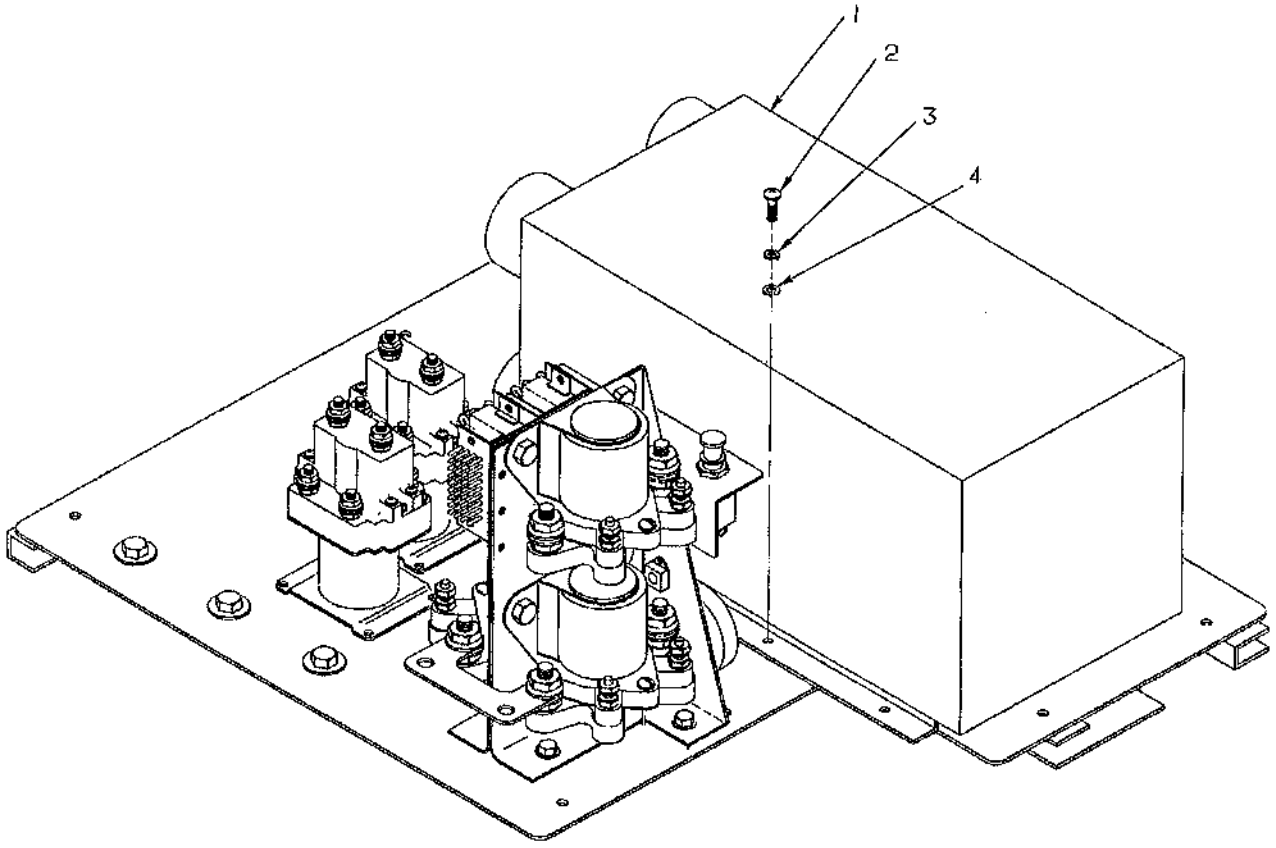
c. Refer to Chapter 53-20-00 and install the cockpit floorboard panel (143CTC).

d. Restore electrical power to the airplane.

e. Test the DAU by performing the level 2 troubleshooting procedure as described in the Collins Starship 1 Avionics System Manual.



1. DATA ACQUISTION UNIT
2. SCREW
3. LOCK WASHER
4. WASHER



DETAIL A

C94NC31B0306 C

Data Acquisition Unit (DAU) Installation (Effectivity: All)
Figure 3

INTEGRATED AVIONICS PROCESSOR SYSTEM - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The Integrated Avionics Processing System (IAPS) performs part of the integration function required to interface the various avionics systems on the aircraft.

The IAPS assembly includes the IAPS card cage, four input/output data concentrators, and four IAPS power supplies. Five other modules are located in the IAPS card cage but are part of other systems. They are: two Flight Control Computers (FCC's); one Flight Management Computer (FMC); one Navigation Computer (NC); and one Control Display Coupler (CDC). The flight management computer, the navigation computer and the Control Display Coupler are part of the Flight Management System (FMS). The two flight control computers are part of the Flight Control System (FCS).

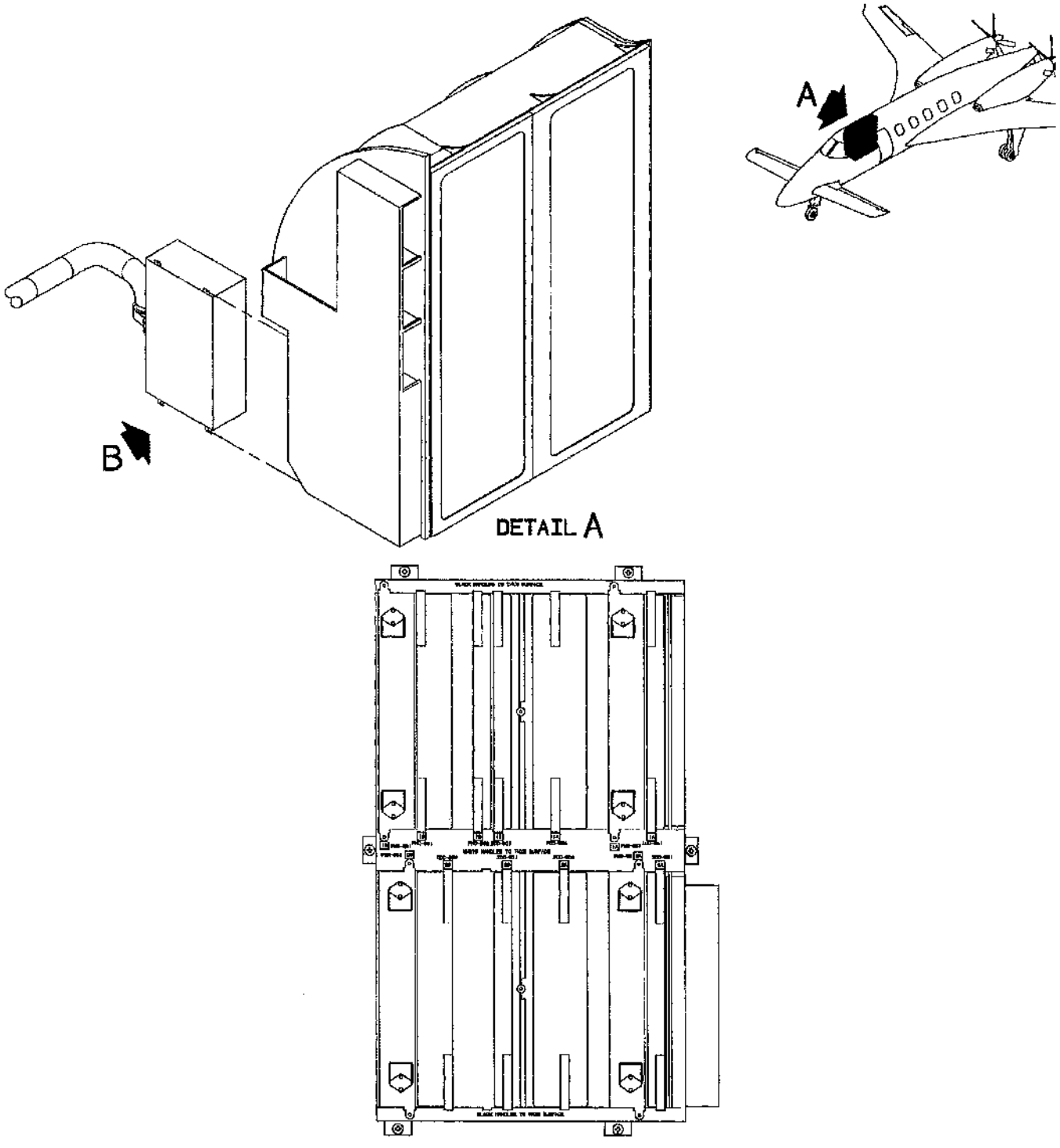
Refer to the Collins Starship 1 Avionics System Manual (523-0774076-00111A or latest revision) for a description of the IAPS components and for IAPS theory of operation.

IAPS MODULE LOCATIONS (Effectivity: All) (Figure 2)

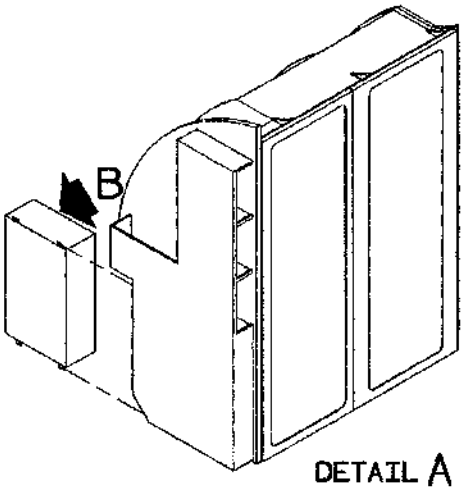
The top half of the IAPS card cage contains modules for the pilot's systems and the bottom half contains modules for the copilot's systems. Each half is further divided into channels A and channels B. The top right quadrant of the IAPS card cage contains input/output concentrator 1A, IAPS power supply 1A, and flight control computer 1A. The left quadrant contains input/output concentrator 1B, the NAV computer, the flight management computer, and IAPS power supply 1B. The bottom right quadrant of the IAPS card cage contains input/output concentrator 2A, IAPS power supply 2A, and flight control computer 2A. The left half contains input/output concentrator 2B, the CDC, and IAPS power supply 2B. A diagram is located on the IAPS card cage cover plate which shows the locations of modules in the IAPS card cage.

INTEGRATED AVIONICS PROCESSOR SYSTEM - TROUBLESHOOTING (Effectivity: All)

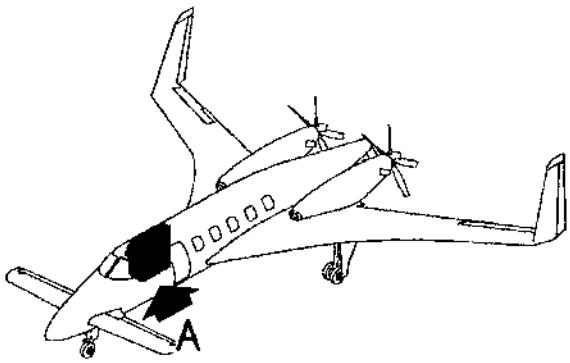
Refer to the Collins Starship 1 Avionics System Manual (523-0774076-00111A or latest revision) for detailed procedures on troubleshooting the IAPS system.

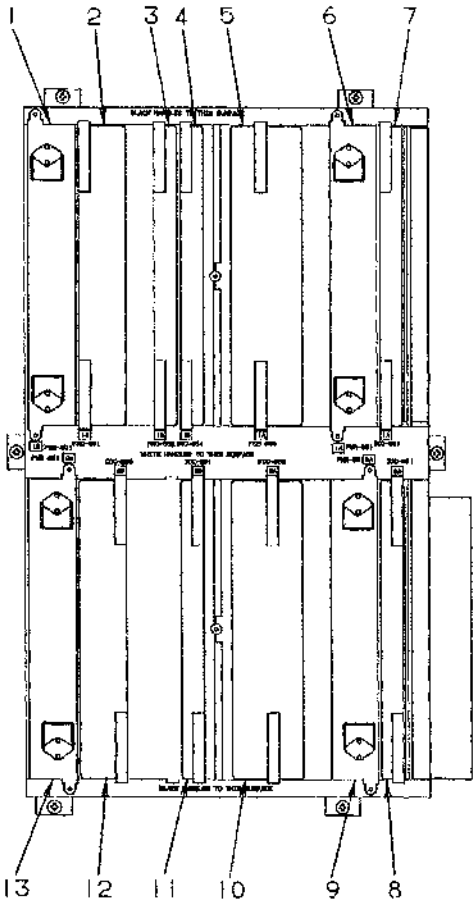


IAPS Location
(Effectivity: All)
Figure 1



DETAIL A





COVER REMOVED FOR CLARITY

DETAIL B

1. IAPS POWER SUPPLY 1B (PILOT)
2. FLIGHT MANAGEMENT COMPUTER
3. NAVIGATION COMPUTER
4. INPUT/OUTPUT CONCENTRATOR 1B (PILOT)
5. FLIGHT CONTROL COMPUTER 1A (PILOT)
6. IAPS POWER SUPPLY 1A (PILOT)
7. INPUT/OUTPUT CONCENTRATOR 1A (PILOT)
8. INPUT/OUTPUT CONCENTRATOR 2A (COPILOT)
9. IAPS POWER SUPPLY 2A (COPILOT)
10. FLIGHT CONTROL COMPUTER 2A (COPILOT)
11. INPUT/OUTPUT CONCENTRATOR 2B (COPILOT)
12. CONTROL DISPLAY COUPLER
13. IAPS POWER SUPPLY 2B (COPILOT)

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IAPS Module Locations (Effectivity: All)
Figure 2

INTEGRATED AVIONICS PROCESSOR SYSTEM - MAINTENANCE PRACTICES (Effectivity: All)

IAPS CARD CAGE REMOVAL (Effectivity: All) (Figure 3)

The IAPS card cage (3) is mounted on the aft side of the divider panel in the right forward baggage compartment. It is accessed by removing the forward panel of the baggage compartment.

a. Remove electrical power from the airplane.

b. Remove the screw (1) and washer (2) securing the plenum assembly (5) to the IAPS card cage (3). Move the plenum assembly clear of the card cage.

c. Support the card cage (3) and disengage the six captive fasteners securing the card cage (3) to the divider panel.

d. Identify, tag and disconnect the electrical connectors.

e. Remove the IAPS card cage.

IAPS CARD CAGE INSTALLATION (Effectivity: All) (Figure 3)

a. Ensure all electrical power is removed from the airplane.

b. Connect the electrical connectors to the card cage (3).

c. Place the IAPS card cage (3) in position on the aft side of the divider panel in the right forward baggage compartment. Engage the six captive fasteners securing the cage to the panel.

d. Position the plenum assembly (5) in place on the card cage (3). Make sure the seal (4) is in the proper position and will not obstruct the cooling air flow. Secure the plenum assembly with the screw (1) and washer (2).

e. Install the furnishings removed for access.

f. Restore electrical power to the airplane.

IAPS MODULE REMOVAL (Effectivity: All)

NOTE

The following procedure is typical for removal of all modules in the IAPS card cage.

a. Disengage the captive screws and remove the card cage cover.

NOTE

Two locking handles are located on the front of each module.

b. Pull the bottom of the top handle outward to disengage the handle.

c. Pull the top of the bottom handle outward to disengage the handle.

NOTE

The handles should now be pointing straight outward.

d. Gently pull the module outward by pulling the handles. Pull the module outward until the module can be grasped. Remove the module.

IAPS MODULE INSTALLATION (Effectivity: All)

CAUTION

IAPS modules must be installed in their proper slots with proper orientation of the handles maintained. Modules installed in the upper half of the IAPS card cage must be installed with their black handles toward the top of the card cage. Modules installed in the lower half of the IAPS card cage must be installed with their black handles toward the bottom of the card cage. Improper module installation may result in damage to equipment.

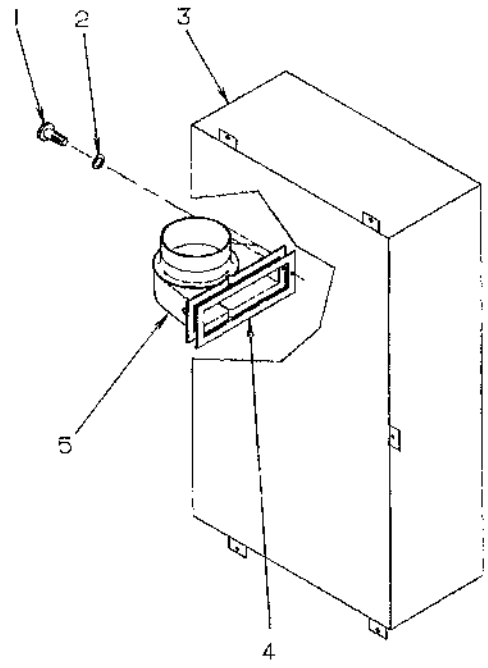
NOTE

The module handles must be sticking straight outward to ensure proper alignment of their attaching mechanism.

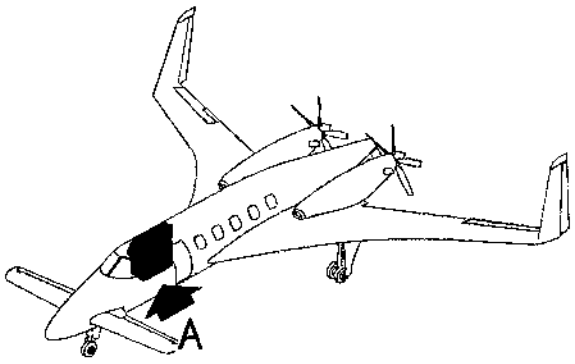
a. Position the module in it's proper position with the black handle oriented properly. Gently push the module until the module meets firm resistance.

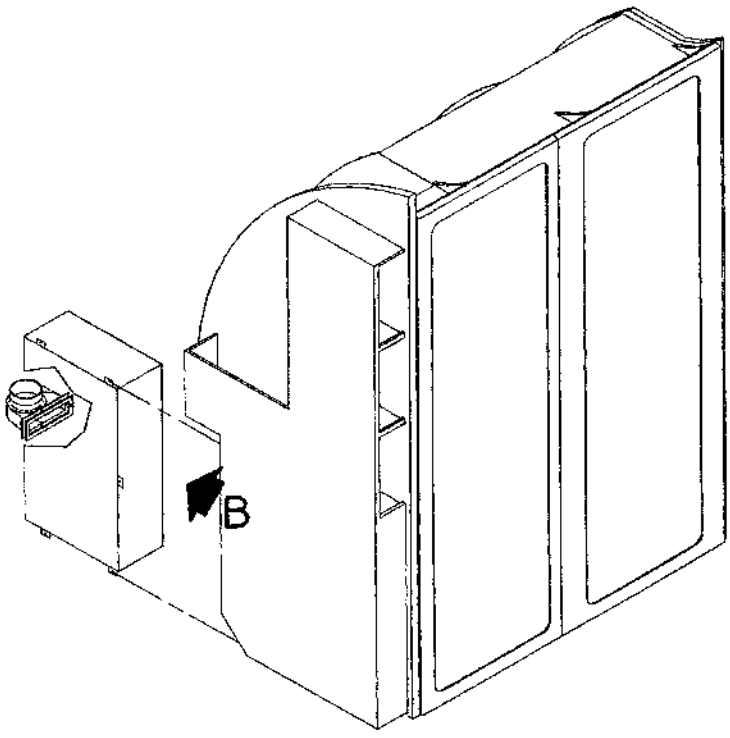
b. Engage the handles to their locked positions.

c. Install the card cage cover and engage the captive fasteners.



DETAIL B





DETAIL A

1. SCREW
2. WASHER
3. IAPS CARD CAGE
4. SEAL
5. PLENUM ASSY

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IAPS Card Cage Installation (Effectivity: All)
Figure 3

CENTRAL WARNING SYSTEM - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

WARNING ANNUNCIATOR SYSTEM (Effectivity: All) (Figures 1, 2 and 3)

The warning annunciator system indicates a hazard which requires immediate crew alerting and corrective action. The warning annunciator panel is mounted in the cockpit directly above the instrument panel in the center of the glareshield and contains fourteen red warning annunciators with black legends. There are two master warning lights, (pilot's and copilot's) mounted one each on either side of the warning annunciator panel. They have black legends that read MASTER WARNING when illuminated, and white letters that read PRESS TO RESET on the legend face. Each master warning light has a built-in switch that resets the master warning signal from the annunciator control PCB when the legend face is pressed. Triple-fed bus voltage through the number 1 annunciator circuit breaker powers annunciator control PCB A135, annunciator fault detection PCB A137, and both the annunciator test and master warning reset circuits.

Annunciator fault detection PCB A137 receives signals from unrelated systems in the airplane. These signals are generated by pressure switches, limit switches and other PCB's that have detected a fault parameter out of limits. Annunciator fault detection PCB A137 sends a signal to warning annunciator panel A103 to illuminate the appropriate annunciator. It also sends a red trigger signal to activate the annunciator control PCB timer circuit, which causes both master warning lights to illuminate and flash on-and-off. The master warning lights will flash until the legend face on either light is pressed, at which time they reset the annunciator control PCB and extinguish. The warning annunciator remains illuminated until the fault signal is removed from the annunciator fault detection PCB input. Any additional faults detected will cause the master warning lights to illuminate and flash again. They must be reset each time a fault is indicated. The test switch placarded ANNUN on the pilot's outboard subpanel may be used to test the warning annunciators and master warning lights.

NOTE

Signals from the door sense switches are input directly to annunciator control PCB A135. Annunciator control PCB A135 then sends a signal to annunciator fault detection PCB A137 to illuminate the DOOR UNLOCKED warning annunciator. Refer to Chapter 52-70-00 for detailed information on the cabin door sense switches.

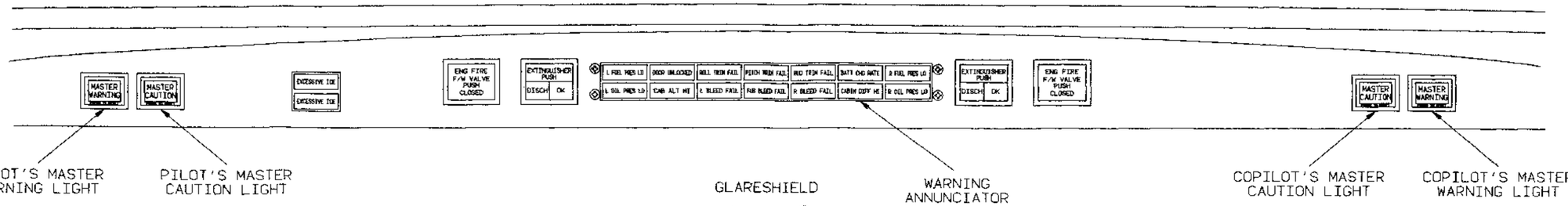
AMBIENT LIGHT SENSOR CIRCUIT (Effectivity: All)

The ambient light sensor, adjacent to the PILOT DISPLAYS potentiometer on the overhead panel in the cockpit, signals the annunciator control PCB to dim the warning annunciator lights according to cockpit light intensity. The master warning and master caution light power returns are also wired through the ambient light sensor circuit. Refer to Chapter 33-10-00 for detailed information on the ambient light sensor circuit.

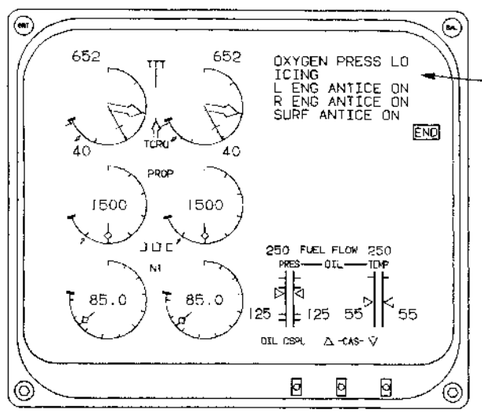
AURAL WARNING SYSTEM (Effectivity: All) (Figures 1 and 4)

The aural warning system consists of a tone generator and two audio amplifiers located in the RH nose avionics compartment, and two speakers and headphones located in the cockpit. Refer to Chapter 27-31-00 for detailed information on stall warning.

The aural warning tone generator is powered by the triple-fed bus through the aural warning circuit

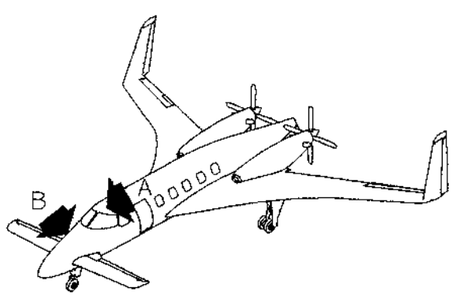


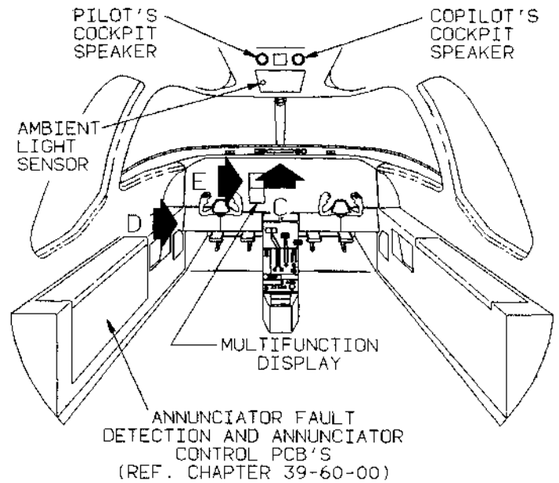
GLARESHIELD
DETAIL C



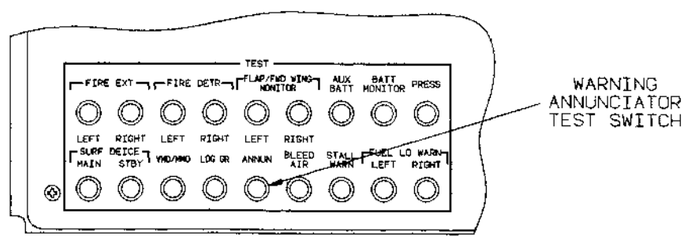
EICAS MESSAGES

EICAS DISPLAY
DETAIL E

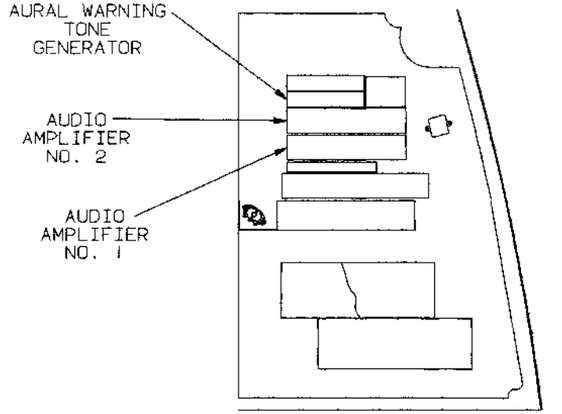




COCKPIT
DETAIL A



PILOT'S OUTBOARD SUBPANEL
DETAIL D



RH NOSE AVIONICS COMPARTMENT
DETAIL B

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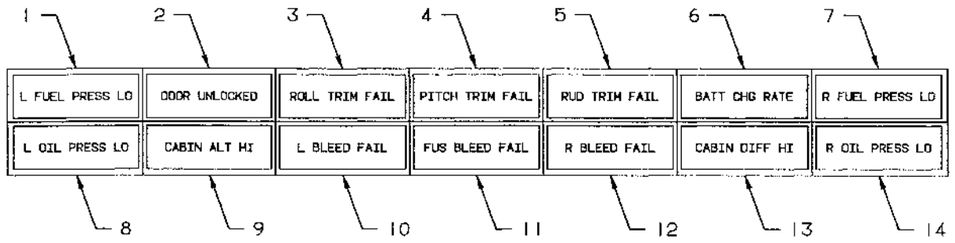
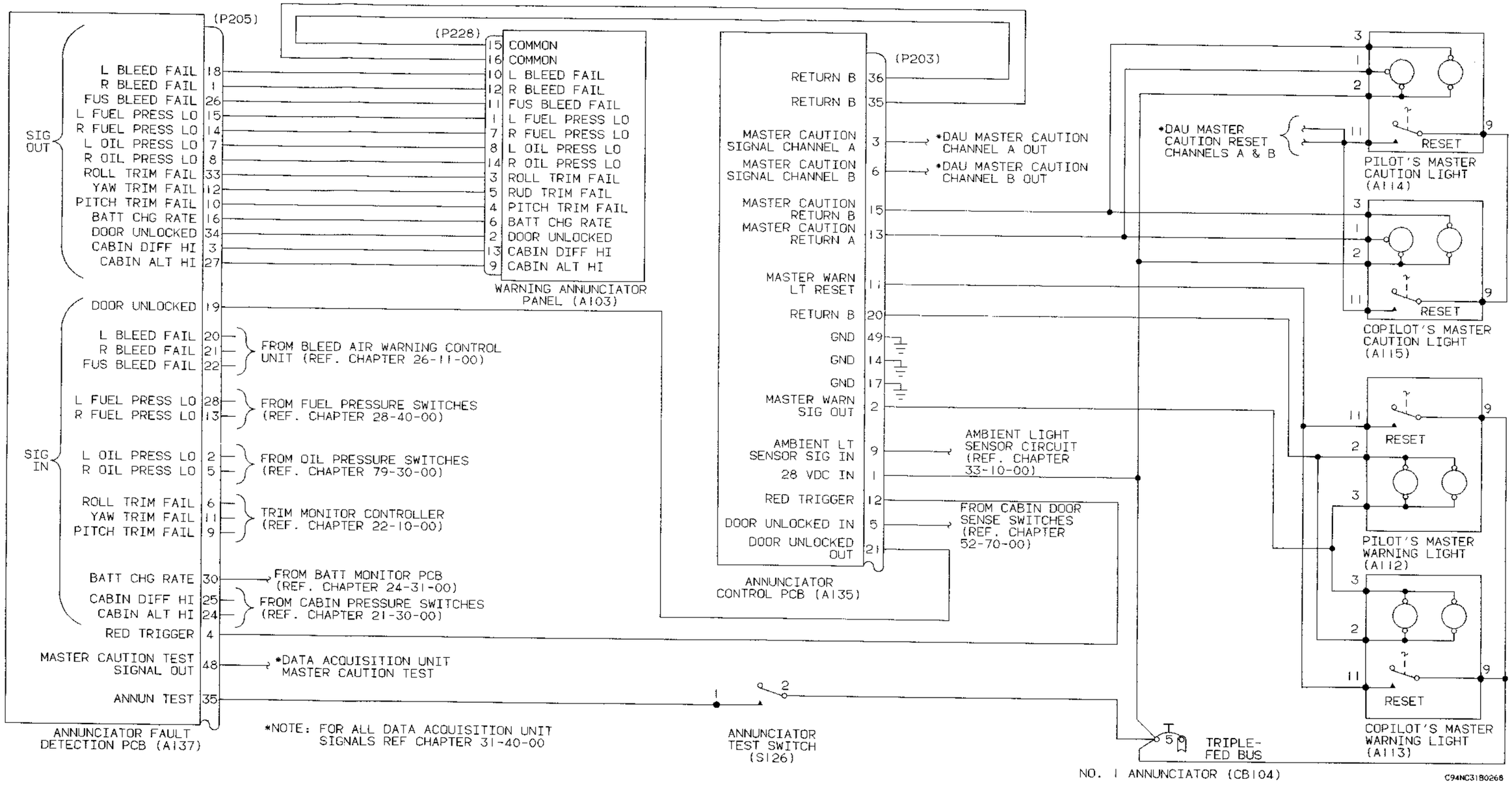
Central Warning System (Effectivity: All)
Figure 1



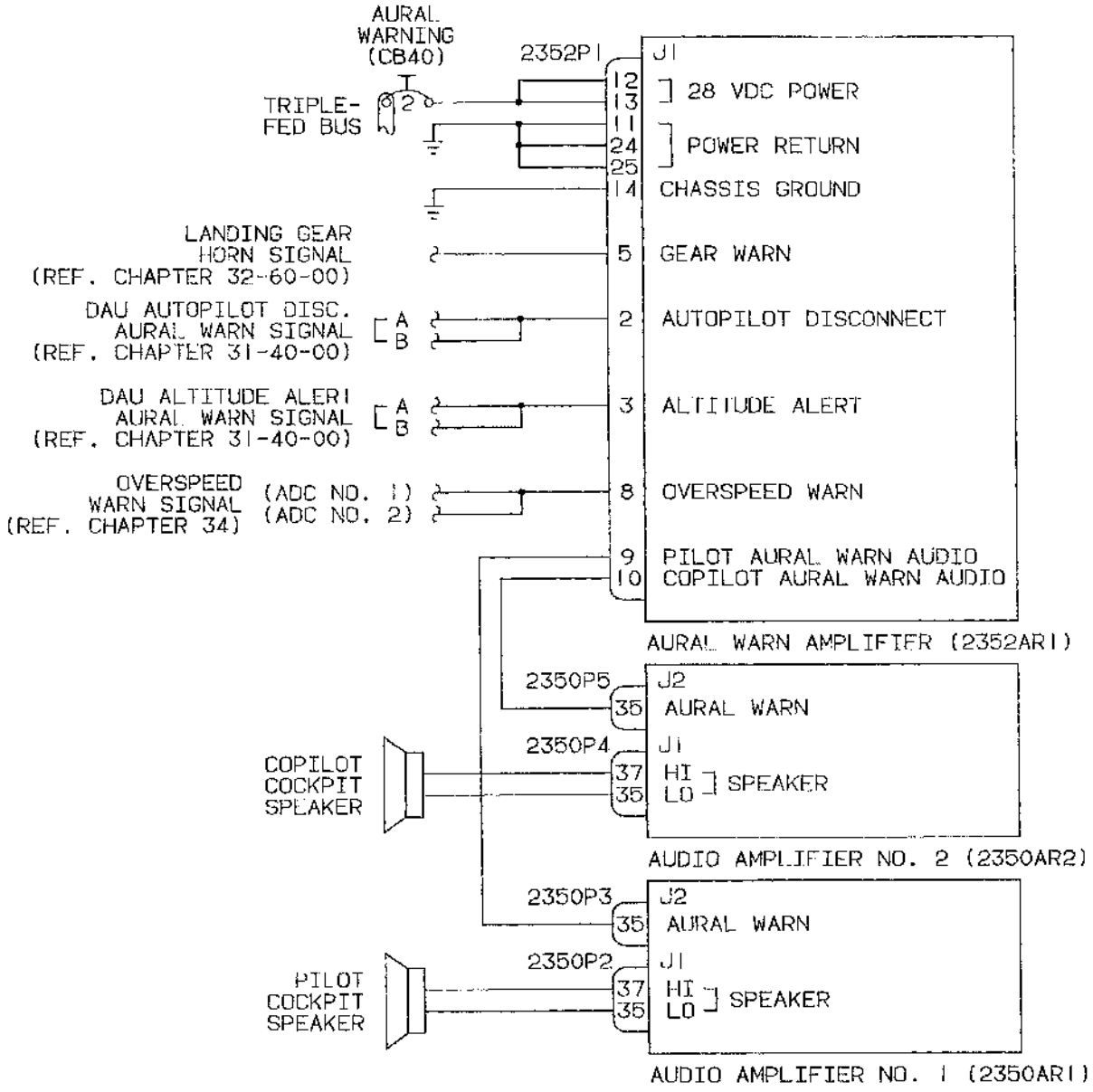
ITEM	NOMENCLATURE	COLOR	REASON FOR ILLUMINATION
1	L FUEL PRESS LO	RED	LEFT FUEL PRESSURE LOW
2	DOOR UNLOCKED	RED	CABIN DOOR NOT CLOSED OR LOCKED
3	ROLL TRIM FAIL	RED	ROLL TRIM HAS FAILED
4	PITCH TRIM FAIL	RED	PITCH TRIM HAS FAILED
5	RUD TRIM FAIL	RED	RUDDER TRIM HAS FAILED
6	BATT CHG RATE	RED	BATTERY CHARGING IS ABNORMAL
7	R FUEL PRESS LO	RED	RIGHT FUEL PRESSURE LOW
8	L OIL PRESS LO	RED	LEFT ENGINE OIL PRESSURE LOW
9	CABIN ALT HI	RED	CABIN ALTITUDE EXCEEDS 10,000 FT.
10	L BLEED FAIL	RED	LEFT BLEED AIR DUCT LEAK
11	FUS BLEED FAIL	RED	FUSELAGE BLEED AIR DUCT LEAK
12	R BLEED FAIL	RED	RIGHT BLEED AIR DUCT LEAK
13	CABIN DIFF HI	RED	CABIN DIFFERENTIAL PRESSURE EXCEEDS LIMITS
14	R OIL PRESS LO	RED	RIGHT ENGINE OIL PRESSURE LOW

2000-351-30

Warning Annunciators (Effectivity: All)
Figure 2



Warning Annunciator and Master Warning/Caution Schematic (Effectivity: All)
Figure 3



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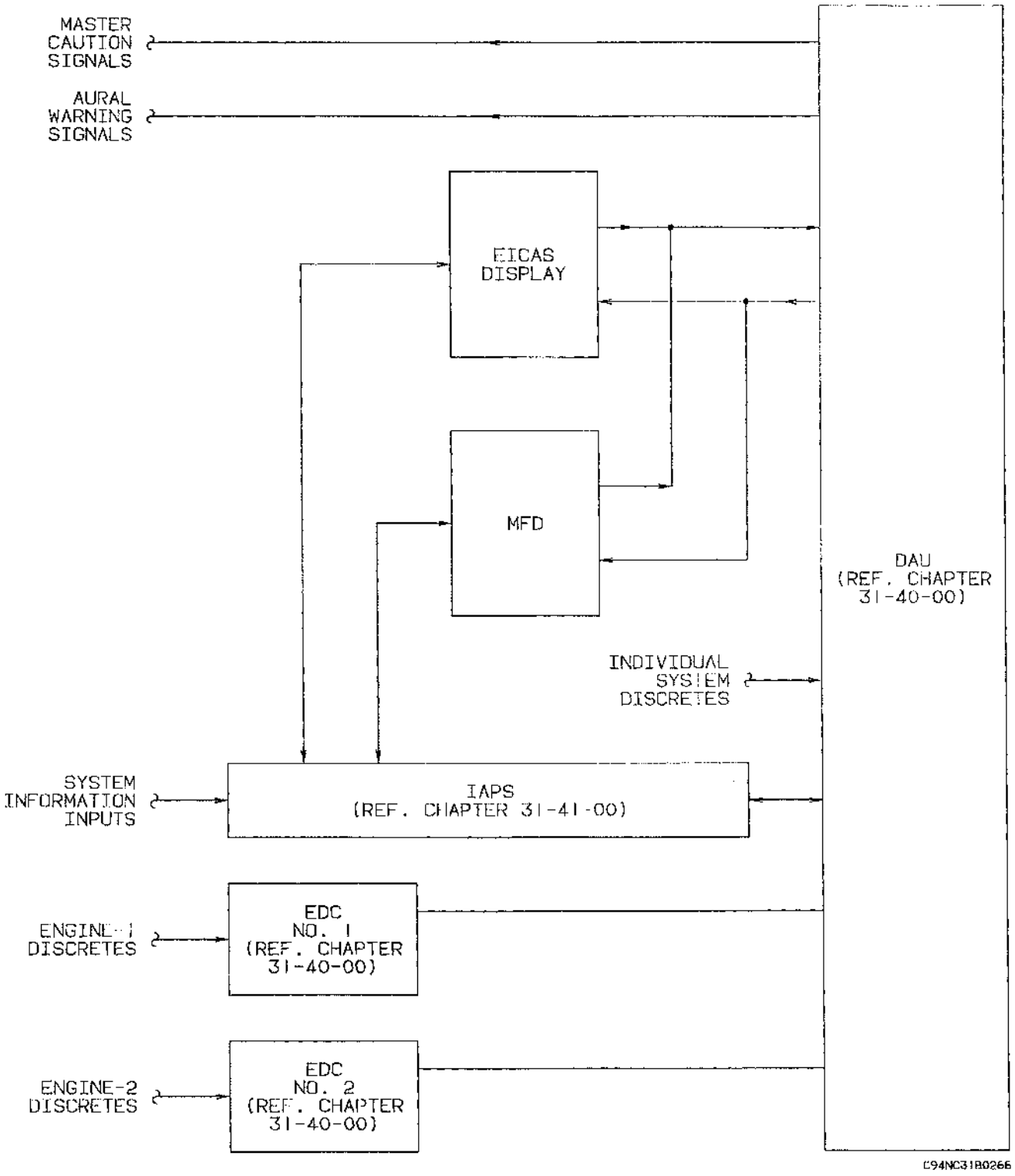
Aural Warning System Schematic (Effectivity: All)
Figure 4



EICAS Block Diagram (Effectivity: All)
Figure 5

breaker. It receives a ground signal from different systems in the airplane that require an aural warning. When an input on the aural warning tone generator receives a ground signal, a distinct tone signal is output on the pilot's and copilot's channels to the No. 1 and No. 2 audio amplifiers. Audio amplifier No. 1 sounds the warning tone through the pilot's cockpit speaker and headphones while audio amplifier No. 2 sounds the warning tone through the copilot's speaker and headphones.

There are four distinct warning tones used for the aural warning system. Two or more tones activated at once will cause a summation of tones. The different tones are described as follows:

a. OVERSPEED WARNING - Repetitive swept frequency tone, increasing from about 500 hz to 1670 hz 5 times per second.

b. LANDING GEAR - Intermittent tone at about 500 hz, cycling on-and-off every .375 second.

c. AUTOPILOT DISCONNECT - Warble tone at about 460 hz modulating about 5 times per second.

d. ALTITUDE ALERT - Steady musical "C" chord tone.

ENGINE INSTRUMENT, CREW ALERTING SYSTEM (Effectivity: All) (Figures 1 and 5)

The Engine Instrument, Crew Alerting System (EICAS) display is a three-division list of messages regarding airplane and avionic systems status. Within each division, messages are shown in chronological order with the most current at the top. When a new message occurs, it is inserted at the top of the division. Once a message is displayed, it will remain for a minimum of four seconds. The three divisions are categorized as follows:

a. YELLOW CAUTION - Indicates a fault which requires immediate crew awareness and possible future crew action.

b. WHITE STATUS - Indicates a condition which could be either normal or abnormal but does not require immediate crew alerting or action. These messages are displayed below the caution messages.

c. GREEN ADVISORY - Confirms that an operation initiated or selected by the flight crew did occur as required. It does not require immediate crew alerting or any further action. These messages are displayed below the status messages.

The Data Acquisition Unit (DAU) provides a dual channel central reference for the EICAS message discrete inputs. Localized discrete data from individual airplane systems are input directly to the DAU. Localized discrete data from individual airplane systems and engine data discretes are first input to the No. 1 and No. 2 Engine Data Concentrators (EDC), where they are encoded and output to an ARINC output bus. This bus transmits the coded data to both channels of the DAU. Discrete data is also input to the DAU from the four Integrated Avionics Processing System (IAPS) input/output concentrators. The EICAS and Multifunction Displays (MFD) receive inputs from both channels on the DAU, and also have outputs directly back to the DAU. During normal operation, the inputs from the DAU will cause the applicable EICAS message or messages to display on the RH side of the EICAS display. Refer to Chart 2 for a listing of EICAS messages.

NOTE

The EICAS messages may be reverted from the EICAS display to the MFD.

There are two master caution lights (pilot's and copilot's) mounted in the glareshield adjacent to the master warning lights. They have black legends that read MASTER CAUTION when illuminated, and white letters that read PRESS TO RESET on the legend face. When the DAU receives a discrete requiring an EICAS yellow caution message, it sends a signal to illuminate the master caution lights and cause them to flash on-and-off. The master caution lights will flash until the legend face on either light is pressed, at which time they will reset and extinguish. The test switch placarded ANNUN on the pilot's outboard subpanel may be used to test the master caution lights.

NOTE

When ANNUN is pressed the EICAS yellow CAUTION TEST caution message is displayed on the EICAS display.

CHART 1
ENGINE INSTRUMENT, CREW ALERTING SYSTEM MESSAGES (Effectivity: All)

NOMENCLATURE	COLOR	CAUSE FOR ILLUMINATION
STBY ATT GYRO OFF	Yellow	Standby gyro power is not selected on.
STBY ATT BATT LO	Yellow	Battery pack for the standby altitude gyro is not charged to an adequate level.
STBY ATT BAT OK	White	Battery pack for the backup attitude display is charged to an adequate level.
STBY ATT BAT TST	White	Wait for battery test to be completed.
ROLL DISAGREE	Yellow	Number 1 and 2 displayed roll attitudes not in agreement.
PITCH DISAGREE	Yellow	Number 1 and 2 displayed pitch attitudes not in agreement.
HDG DISAGREE	Yellow	Number 1 and 2 displayed headings not in agreement.
LOC DISAGREE	Yellow	Number 1 and 2 displayed localizer deviations not in agreement.
GS DISAGREE	Yellow	Number 1 and 2 displayed glideslope deviations not in agreement.
PITCH TRIM OFF	Yellow	Pitch trim is off.
PITCH TRIM SYNC	Yellow	Pitch trim is out of synchronization.
ROLL TRIM OFF	Yellow	Roll trim is off.
ROLL TRIM SYNC	Yellow	Roll trim is out of synchronization.
RUDDER TRIM OFF	Yellow	Rudder trim is off.
AVIONIC AIR FAIL	Yellow	Insufficient air flow within avionics air distribution duct.
AV ALTN BLWR ON	Green	Alternate blower is operating for avionics cooling.
DUCT OVERTEMP	Yellow	Environmental duct air is too hot.
L BLEED OFF	Yellow	Left bleed air valve is not open.
R BLEED OFF	Yellow	Right bleed air valve is not open.
EMER BLEED ON	White	Emergency bleed air valve is open.
L GEN TIE OPEN	Yellow	Left generator bus tie is open.
R GEN TIE OPEN	Yellow	Right generator bus tie is open.
L GEN INOP	Yellow	Left generator is off-line. Engine is not secured.
R GEN INOP	Yellow	Right generator is off-line. Engine is not secured.
BATTERY TIE OPEN	Yellow	Battery is isolated from generator buses.
EXT POWER CONN	White	External power unit is connected.
MAN TIES CLOSED	Green	Generator bus ties are manually closed.
L CHIP DETECT	White	Left engine oil contamination detected. Engine is not secured.
R CHIP DETECT	White	Right engine oil contamination detected. Engine is not secured.
L F/W VALVE FAIL	Yellow	Left fuel firewall valve has not reached its selected position within 2 seconds.
R F/W VALVE FAIL	Yellow	Right fuel firewall valve has not reached its selected position within 2 seconds.
L FUEL LEVEL LO	Yellow	Left fuel system contains approximately 135 pounds of fuel (level must be constant for 1 second).

CHART 1
ENGINE INSTRUMENT, CREW ALERTING SYSTEM MESSAGES (Effectivity: All) *(Continued)*

NOMENCLATURE	COLOR	CAUSE FOR ILLUMINATION
R FUEL LEVEL LO	Yellow	Right fuel system contains approximately 135 pounds of fuel (level must be constant for 1 second).
L ENG SECURED	White	Left engine intentionally shutdown with propeller lever in feather, condition lever in cut-off and firewall valve closed. This message replaces other messages normally displayed with a non-running left engine.
R ENG SECURED	White	Right engine intentionally shutdown with propeller lever in feather, condition lever in cut-off and firewall valve closed. This message replaces other messages normally displayed with a non-running right engine.
FUEL TRANSFER >	White	Fuel is being transferred from the left to right fuel tank.
FUEL TRANSFER <	White	Fuel is being transferred from the right to left fuel tank.
L IGNITION ON	Green	Left engine ignition is on.
R IGNITION ON	Green	Right engine ignition is on.
RUD TRIM SYNC	Yellow	Rudder trim is out of synchronization.
L FLAP MON FAIL	Yellow	Left flap/forward monitor has failed the ground functional test.
R FLAP MON FAIL	Yellow	Right flap/forward wing monitor has failed the ground functional test.
ROLL COMP OFF	White	Roll comparator function is inoperative.
PITCH COMP OFF	White	Pitch comparator function is inoperative.
HDG COMP OFF	White	Heading comparator function is inoperative.
ALL COMP OFF	White	All comparators are inoperative.
ICING	Yellow	Ice detector is sensing icing conditions and all ice protection systems are not selected on.
L WING BOOT FAIL	Yellow	A left wing boot failed to inflate.
R WING BOOT FAIL	Yellow	A right wing boot failed to inflate.
L FWD BOOT FAIL	Yellow	Left forward wing boot failed to inflate.
R FWD BOOT FAIL	Yellow	Right forward wing boot failed to inflate.
PNEU PRESS LOW	Yellow	Pneumatic system pressure is too low to operate the deice boots.
PITOT OVERHEAT	Yellow	Pilot and/or copilot pitot/static heat has been on for more than 2 minutes on the ground.
MAIN DEICE FAIL	Yellow	Left ice detector is inoperataive or primary auto deice controller has failed.
STBY DEICE FAIL	Yellow	Right ice detector is inoperative or standby auto deice controller has failed.
ICING	White	Ice detector is sensing icing conditions and all ice systems are selected on.
L ENG ANTICE ON	Green	Left engine ice vane is selected and operating.
R ENG ANTICE ON	Green	Right engine ice vane is selected and operating.
L ICE VANE FAIL	Yellow	Left engine ice vane has not reached its proper position within 30 seconds.

CHART 1
ENGINE INSTRUMENT, CREW ALERTING SYSTEM MESSAGES (Effectivity: All) *(Continued)*

NOMENCLATURE	COLOR	CAUSE FOR ILLUMINATION
R ICE VANE FAIL	Yellow	Right engine ice vane has not reached its proper position within 30 seconds.
WSHLD STBY POWER	Green	Pilot's windshield standby heat system is operating.
HYD FLUID LO	Yellow	Hydraulic fluid in power pack is low.
ANTI-SKID INOP	Yellow	Anti-skid system is inoperative.
POWER BRAKE INOP	Yellow	Power brake system is inoperative.
L GEAR UP	White	Left gear is fully retracted and alternate gear extension is selected.
R GEAR UP	White	Right gear is fully retracted and alternate gear extension is selected.
L GEAR IN TRANS	White	Left gear neither fully extended or retracted and alternate gear extension is selected.
R GEAR IN TRANS	White	Right gear neither fully extended or retracted and alternate gear extension is selected.
NS GEAR UP	White	Nose gear is fully retracted and alternate gear extension is selected.
NS GEAR IN TRANS	White	Nose gear neither fully extended or retracted and alternate gear extension is selected.
L STALL WRN FAIL	Yellow	Left stall warning computer has failed.
R STALL WRN FAIL	Yellow	Right stall warning computer has failed.
PUSHER MOTOR ON	Yellow	Column pusher motor is energized and the pusher clutch is not energized.
PUSHER CLUTCH ON	Yellow	Column pusher clutch is energized and the pusher motor is not energized.
PUSHER INOP	Yellow	Column pusher is inoperative.
OXYGEN NOT ARMED	Yellow	Oxygen system is not armed.
OXYGEN PRES LO	Yellow	Oxygen pressure is below 500 psi and the system is armed.
PASS OXYGEN ON	White	Passenger oxygen system is charged.
CAUTION TEST	Yellow	ANNUN test button has been depressed.
AFX OFF	Yellow	Autofeather system is not armed.
AFX DISABLED	Yellow	Autofeather is selected, power levers have been advanced to a high N_1 position, but the ARM signal has been lost while on the ground.
AFX DISABLED	White	Autofeather is selected, power levers have been advanced to a high N_1 position, but the ARM signal has been lost while airborne.
AFX ENABLED	Green	Both left and right autofeather systems are armed.
AHRS ALIGNING DO NOT TAXI	White	Left or right AHRS have not completed alignment procedures while on the ground.
AHRS ALIGNING	White	Left or right AHRS have not completed alignment procedures while airborne.

CENTRAL WARNING SYSTEM - TROUBLESHOOTING (Effectivity: All) (Figure 3)

The BEECHCRAFT Starship 1 Wiring Diagram Manual should be used to trace out specific wire routes and locate pin and plug numbers. Many times a loose or faulty connection will cause a fault. Ensure that there are no loose, damaged or corroded pins on any connectors in the system. Continuity checks should be made with power removed from the circuit being checked. Use a calibrated voltmeter and ensure that the battery switch is ON during all voltage checks. Press the TRIP FED volts button on the center subpanel to determine battery voltage on the triple-fed bus. The steps in each chart should be followed sequentially unless otherwise specified. An example would be to proceed to the next step when there are no directions following a NO condition.

Troubleshooting the warning annunciator system begins by performing the WARNING ANNUNCIATOR SYSTEM OPERATIONAL TEST as outlined in this chapter. This test will indicate a failure of the system in general, or isolate burned-out lamps. If an individual warning annunciator fails to illuminate, refer to Chapter 33-10-00 and replace the lamps. If the warning annunciator still fails to illuminate, check for triple-fed bus voltage at the warning annunciator panel inputs, and if necessary the annunciator fault detection PCB outputs. If an individual output on the annunciator fault detection PCB is without voltage during test, suspect a faulty PCB. If voltage is at the annunciator fault detection PCB output, but not at the annunciator warning panel, locate and correct the wiring fault. A failure of only one master warning or master caution light to illuminate normally indicates a fault in the wiring to the light, or a fault in the light itself. Refer to Chapter 33-10-00 to replace the master warning and master caution light lamps. If the master warning or master caution lights fail to flash during test, suspect an annunciator control PCB internal fault. If any of the warning annunciators, master warning lights or master caution lights appear to be illuminated at half brightness when compared to the others, suspect that only one of the two lamps has burned out.

Refer to the applicable chart in this chapter to isolate faults not identified with the general troubleshooting guidelines. If the fault is isolated to one of the system PCB's, use the functional test procedures in this chapter to test the applicable PCB.

CHART 2
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All)

WARNING ANNUNCIATORS DO NOT ILLUMINATE DURING TEST (1)

STEP 1	Is a reading slightly less than battery voltage displayed on the voltmeter when the TRIP FED voltmeter select switch on the center subpanel is pressed? (See Note 2.)	**YES**	Proceed to Step 2.
	NO		
	Refer to Chapter 24-50-00 for detailed information on electrical load distribution. The triple-fed bus must be powered for the warning annunciators to illuminate under test.		
STEP 2	Is the No. 1 annunciator circuit breaker open?	**YES**	Close circuit breaker.
	NO		
STEP 3	Is there continuity across the No. 1 annunciator circuit breaker terminals?	**YES**	Proceed to Step 4.
	NO		
	Faulty circuit breaker.		
STEP 4	Is there triple-fed bus voltage at pin 35 of the annunciator fault detection PCB?	**YES**	Faulty annunciator fault detection PCB.

CHART 2
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All) *(Continued)*

WARNING ANNUNCIATORS DO NOT ILLUMINATE DURING TEST (1)

NO

STEP 5	Check the ANNUN test switch for continuity across the contacts when it is pressed. Is there continuity?	**YES**	Proceed to Step 6.

NO

Faulty ANNUN test switch.

STEP 6	Is there continuity between the No. 1 annunciator circuit breaker and the ANNUN test switch?	**YES**	Proceed to Step 7.

NO

Restore continuity to circuit wiring.

STEP 7	Is there continuity between the two common pins on the warning annunciator panel and their dimming returns on the annunciator control panel?	**YES**	Proceed to Step 8.

NO

Restore continuity to dimming returns.

STEP 8	Is there continuity between the ANNUN test switch and the annunciator fault detection PCB?	**YES**	Check for a loose or damaged connector between the warning annunciator panel and the annunciator fault detection PCB. If no fault is evident, suspect a faulty annunciator fault detection PCB.

NO

Restore continuity to circuit wiring.

1. The battery switch should be in the ON position for all voltage checks. Continuity checks should be made with power removed from the circuit being checked.

2. Press the BATT voltmeter select switch on the center subpanel and read battery voltage on the voltmeter. Compare this reading with the reading obtained for triple-fed bus voltage. A drop of .7 vdc is normal.

CHART 3
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All)

MASTER WARNING LIGHTS DO NOT ILLUMINATE DURING TEST (1)

STEP 1	Is there triple-fed bus voltage at the master warning lights?	**YES**	Proceed to Step 2.

NO

STEP 2	Are the master warning lights properly grounded to the returns on the annunciator control PCB?	**YES**	Faulty master warning light or burned out lamps. Refer to Chapter 33-10-00 for lamp replacement procedures.

CHART 3
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All) *(Continued)*

MASTER WARNING LIGHTS DO NOT ILLUMINATE DURING TEST (1)

NO

Restore continuity to master warning light ground.

STEP 3	Is there triple-fed bus voltage at pin 2 of the annunciator control PCB?	**YES**	Restore continuity to wiring between the annunciator control PCB and the master warning lights.

NO

STEP 4	Is there triple-fed bus voltage (red trigger signal) at pin 12 of the annunciator control PCB?	**YES**	Faulty annunciator control PCB.

NO

STEP 5	Is there triple-fed bus voltage (red trigger signal) at pin 4 of the annunciator fault detection PCB?	**YES**	Restore continuity between pin 4 on the annunciator fault detection PCB and pin 12 on the annunciator control PCB.

NO

Faulty annunciator fault detection PCB.

1. Ensure that the warning annunciators illuminate during test before using this chart. The battery switch should be in the ON position for all voltage checks. Continuity checks should be made with power removed from the circuit being checked.

CHART 4
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All)

MASTER WARNING LIGHTS WILL NOT RESET (1)

STEP 1	Is there triple-fed bus voltage at pin 11 of the annunciator control PCB.	**YES**	Faulty annunciator control PCB.

NO

STEP 2	Is there triple-fed bus voltage at the reset switch output on the master warning light? (See Note 2.)	**YES**	Restore continuity to wiring between the master warning light reset switch output and pin 11 on the annunciator control PCB.

NO

STEP 3	Is there triple-fed bus voltage from the No. 1 annunciator circuit breaker at the master warning light reset switch input?	**YES**	Faulty master warning light reset switch.

NO

Restore continuity to wiring between the No. 1 annunciator circuit breaker and master warning light reset switch input.

CHART 4
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All) *(Continued)*

MASTER WARNING LIGHTS WILL NOT RESET (1)

1. The battery switch should be in the ON position for all voltage checks. Continuity checks should be made with power removed from the circuit being checked.

2. The master warning light reset switch must be actuated to receive this reading.

CHART 5
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All)

MASTER CAUTION LIGHTS DO NOT ILLUMINATE DURING TEST (1)

STEP 1	Is a reading slightly less than battery voltage displayed on the voltmeter when the TRIP FED voltmeter select switch on the center subpanel is pressed? (See Note 2.)	**YES**	Proceed to Step 2.
	NO		
	Refer to Chapter 24-50-00 for detailed information on electrical load distribution. The triple-fed bus must be powered for the master caution lights to illuminate during test.		
STEP 2	Is the No. 1 annunciator circuit breaker open? :tc3.YES		Close circuit breaker.
	NO		
STEP 3	Is there triple-fed bus voltage at the master caution lights?	**YES**	Proceed to Step 5.
	NO		
STEP 4	Is there continuity between the master caution lights and the No. 1 annunciator circuit breaker?	**YES**	Check the No. 1 annunciator circuit breaker for continuity across the contacts. Replace a faulty circuit breaker. If the circuit breaker is good, suspect a faulty master caution light.
	NO		
	Restore continuity to wiring between circuit breaker and master caution lights.		
STEP 5	Are the master caution lights properly grounded through the master caution returns on the annunciator control PCB?	**YES**	
		YES	Suspect either a faulty master caution light or burned-out lamps. Refer to Chapter 33-10-00 for lamp replacement procedures.

CHART 5
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All) *(Continued)*

MASTER CAUTION LIGHTS DO NOT ILLUMINATE DURING TEST (1)

NO

Restore continuity to the master caution light returns. Replace the annunciator control PCB if it is not providing a ground to the master caution lights through the dimming returns.

1. Ensure that the warning annunciators and master warning lights illuminate during test before using this chart. The battery switch should be in the ON position for all voltage checks. Continuity checks should be made with power removed from the circuit being checked.

2. Press the BATT voltmeter select switch on the center subpanel and read battery voltage on the voltmeter. Compare this reading with the reading obtained for triple-fed bus voltage. A drop of .7 vdc is normal.

CHART 6
TROUBLESHOOTING - CENTRAL WARNING SYSTEM (Effectivity: All)

MASTER CAUTION LIGHTS DO NOT RESET (1)

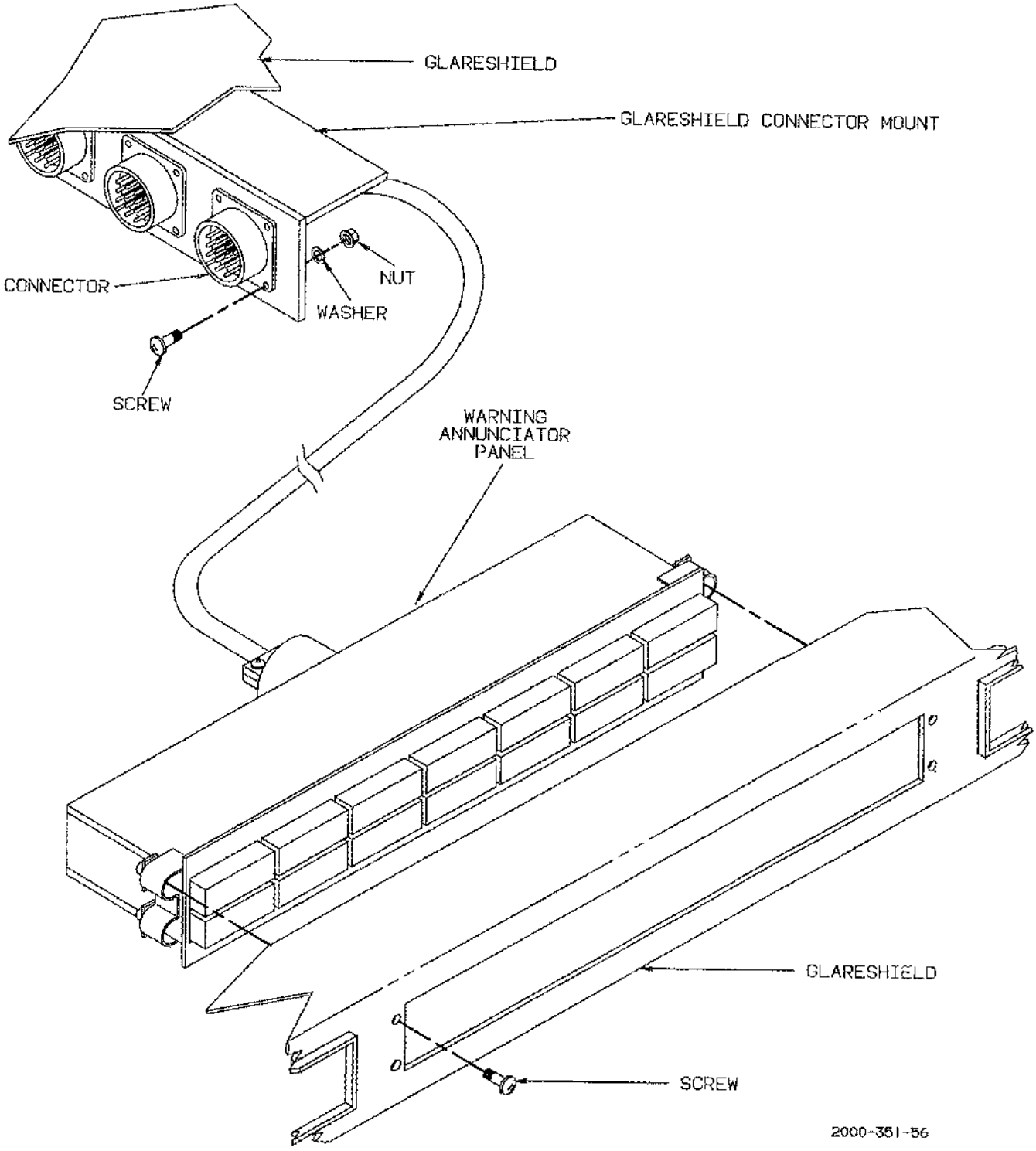
STEP 1	Is there triple-fed bus voltage at pin H of the DAU for both channels A and B?	**YES**	Suspect fault in the DAU. Refer to the Collins Starship 1 Avionics System Manual (523-0774076-00111A or later revision) for information on the DAU.
	NO		
STEP 2	Is there triple-fed bus voltage at the reset switch output on the master caution light? (See Note 2.)	**YES**	Restore continuity to wiring between the master caution light reset switch output and pin H on both channels of the DAU.
	NO		
STEP 3	Is triple-fed bus voltage from the No. 1 annunciator circuit breaker at the master caution light reset switch input?	**YES**	Faulty master caution light reset switch.
	NO		
	Restore continuity to wiring between the No. 1 annunciator circuit breaker and master caution light reset switch.		

1. The battery switch should be in the ON position for all voltage checks. Continuity checks should be made with power removed from the circuit being checked.

2. The master caution light reset switch must be actuated to receive this reading.

CENTRAL WARNING SYSTEM - MAINTENANCE PRACTICES (Effectivity: All)

Refer to Chapter 39-60-00 to remove the annunciator control and annunciator fault detection PCB's. Refer to Chapter 33-10-00 for warning annunciator, master warning and master caution light lamp replacement. Refer to Chapter 33-10-00 for ambient light sensor removal procedures. Refer to Chapter 31-10-00 to remove the EICAS display. Refer to Chapter 31-40-00 to remove the DAU and EDC.



Warning Annunciator Panel Installation (Effectivity: All)
Figure 6

WARNING ANNUNCIATOR PANEL REMOVAL (Effectivity: All) (Figure 6)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-10-00 and remove the glareshield.

c. Remove the four mounting screws, washers and nuts from the warning annunciator panel connector and remove it from the glareshield connector mount.

d. Remove the four mounting screws attaching the warning annunciator panel to the glareshield.

e. Remove the warning annunciator panel from the glareshield.

WARNING ANNUNCIATOR PANEL INSTALLATION (Effectivity: All) (Figure 6)

a. Align the warning annunciator panel with the glareshield panel cutout.

b. Secure the warning annunciator panel to the glareshield by installing the four mounting screws.

c. Align the warning annunciator panel connector with the glareshield connector mount and install the four screws, washers and nuts.

d. Refer to Chapter 25-10-00 and install the glareshield.

e. Restore electrical power to the airplane.

f. Perform the WARNING ANNUNCIATOR SYSTEM OPERATIONAL TEST as outlined in this chapter.

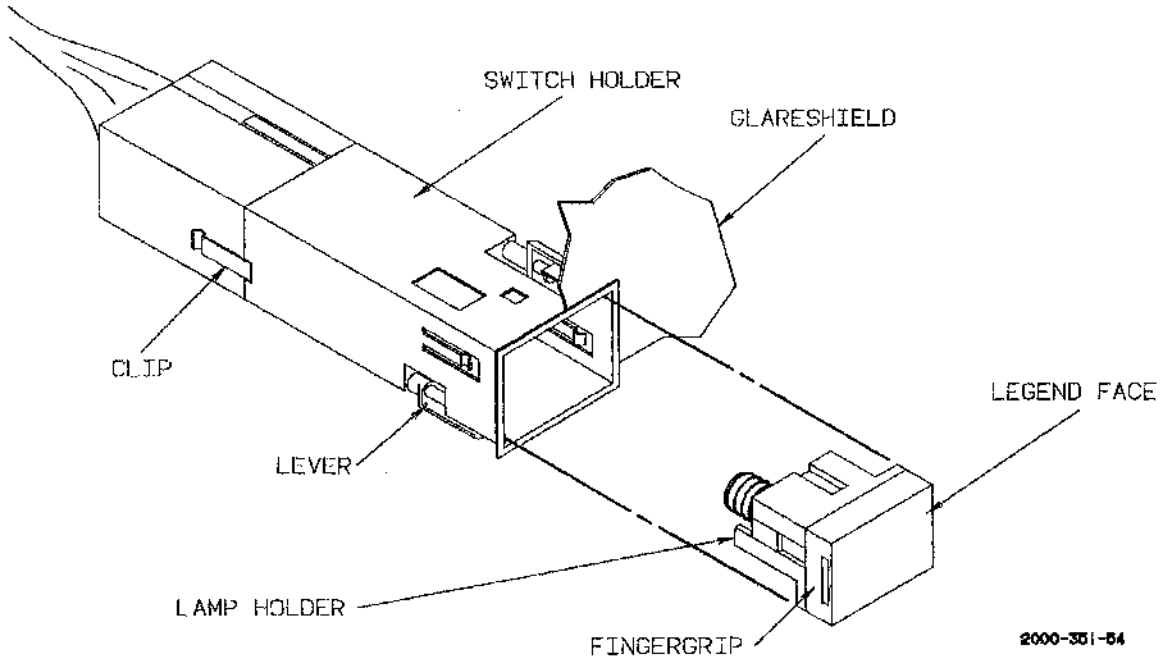
MASTER WARNING AND MASTER CAUTION LIGHT SWITCH REMOVAL (Effectivity: All) (Figure 7)

NOTE

Both the pilot's and copilot's master warning and master caution light switches are removed with the same procedure.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-10-00 and remove the glareshield.



Master Warning and Master Caution Light Switch Installation (Effectivity: All)
Figure 7

c. Tag and remove wiring from the back of the applicable switch.

d. Grasp the grips on the side of the legend face and pull outward to remove the lamp holder.

e. Ensure the levers on the side of the switch are rotated down to allow removal of the switch.

f. Slide the switch out until it snaps out of the clips on the switch holder and remove it from the holder.

MASTER WARNING AND MASTER CAUTION LIGHT SWITCH INSTALLATION (Effectivity: All) (Figure 7)

NOTE

Both the pilot's and copilot's master warning and master caution light switches are installed with the same procedure.

a. Slide the switch into the switch holder until it clicks into place. Ensure the clips are holding the switch in the holder.

b. Reconnect wiring to the back of the switch according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

c. Push the lamp holder back into the switch body until it clicks and locks into place.

NOTE

Ensure that the legend face is not inverted.

d. Refer to Chapter 25-10-00 and install the glareshield.

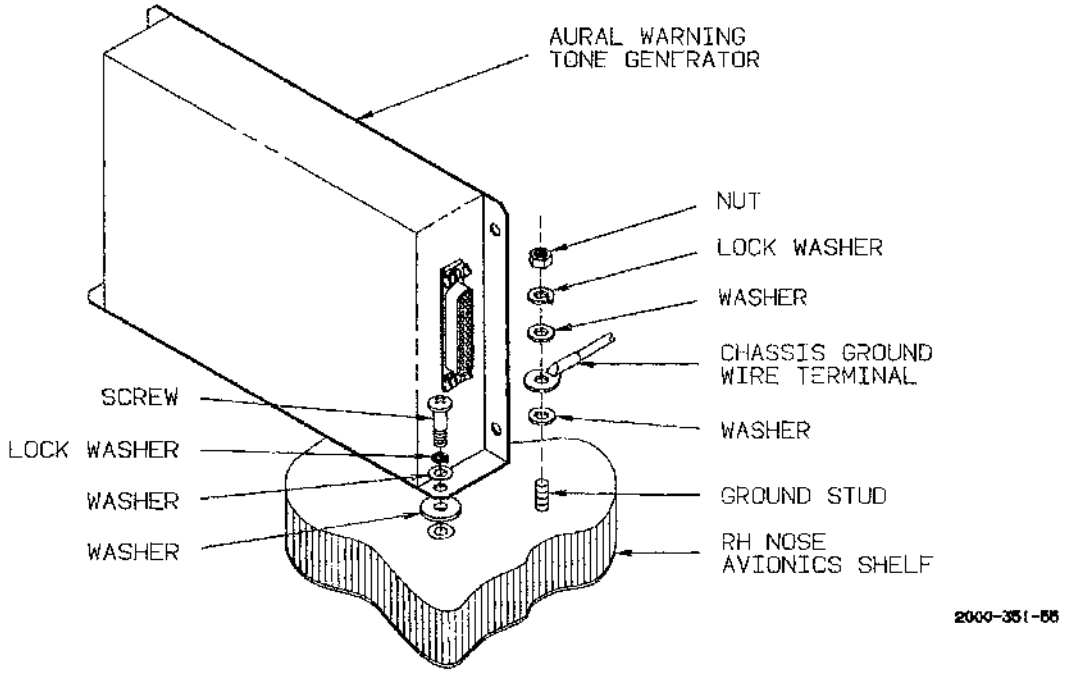
e. Restore electrical power to the airplane.

f. Perform the WARNING ANNUNCIATOR SYSTEM OPERATIONAL TEST as outlined in this chapter.

AURAL WARNING TONE GENERATOR REMOVAL (Effectivity: All) (Figure 8)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the RH nose avionics compartment door.



Aural Warning Tone Generator Installation (Effectivity: All)
Figure 8

c. Disconnect the electrical connector from the tone generator.

d. Remove nut, lock washer and washer attaching the chassis ground wire terminal to ground stud on the avionics shelf.

e. Remove the mounting screws, lock washers and washers attaching the tone generator to the avionics shelf.

f. Remove the tone generator.

AURAL WARNING TONE GENERATOR INSTALLATION (Effectivity: All) (Figure 8)

CAUTION

The aural warning tone generator must meet electrical bonding requirements as specified in Chapter 20-00-03.

a. Align washers over the inserts in the avionics equipment shelf.

b. Align the tone generator mounting holes with the inserts and washers.

c. Secure the tone generator to the avionics shelf with two washers, lock washers and screws.

d. Install the chassis ground wire terminal on the ground stud with one washer below the terminal, and a washer, lock washer and nut above the terminal.

e. Reconnect the electrical connector to the tone generator.

f. Refer to Chapter 53-30-00 and install the RH nose avionics compartment door.

g. Restore electrical power to the airplane.

ANNUNCIATOR CONTROL PCB FUNCTIONAL TEST (Effectivity: All) (Figure 9)

This functional test procedure checks the circuits on the annunciator control PCB associated with the warning annunciator system. Required outputs are connected to lamps for quick identification of operational circuits. Faults in the PCB are identified when a lamp fails to illuminate as specified in the test procedures. Replace a PCB that fails the functional test. A functional test of the ambient light sensor circuit is also included. Refer to Chapter 33-10-00 for detailed information on the ambient light sensor circuit.

MASTER WARNING CIRCUIT (Effectivity: All)

a. Press S1 to apply 28 vdc at pin 12. DS1 should flash on and off.

b. Change the setting of R1. DS1 intensity will decrease as the value of R1 is increased from 10 K to 60 M ohms.

c. Press S2 to reset the PCB. DS1 should extinguish.

MASTER CAUTION CIRCUIT (Effectivity: All)

a. Press S4 to apply a ground at pin 3. DS3 should illuminate.

b. Release S4. DS3 should extinguish.

c. Press S5 to apply a ground at pin 6. DS3 should illuminate.

d. Release S5. DS3 should extinguish.

DOOR UNLOCKED CIRCUIT (Effectivity: All)

a. Press S3 to apply 28 vdc at pin 5. DS2 should illuminate.

b. Release S3. DS2 should extinguish.

ANNUNCIATOR FAULT DETECTION PCB FUNCTIONAL TEST (Effectivity: All) (Figure 10)

Use the following procedures and test circuit to check the annunciator fault detection PCB for faults. Individually test each warning annunciator input and output with S1 and DS1. Single failures will not affect operation of other circuits on the PCB, but are cause enough to have it replaced.

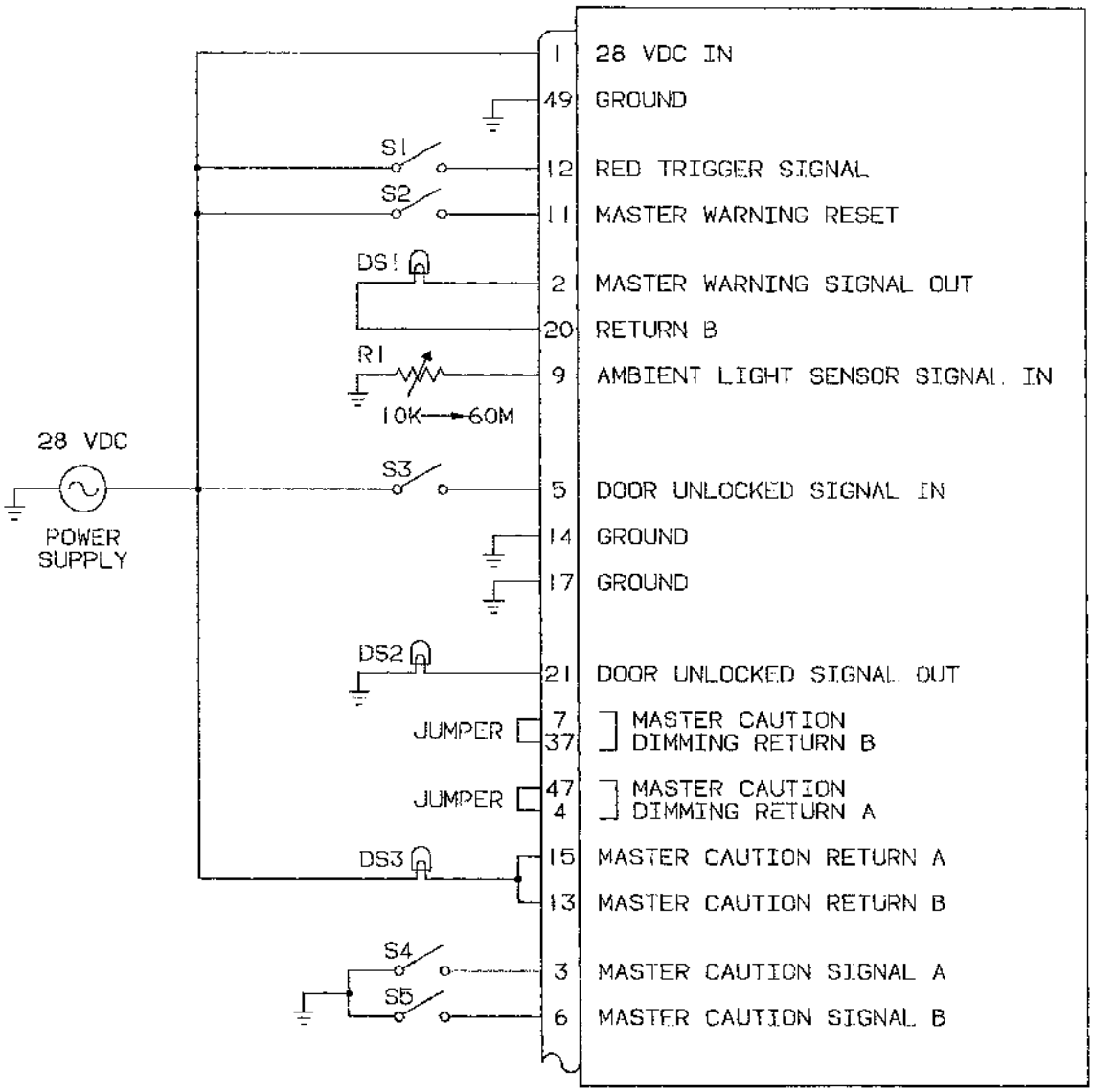
a. Connect S1 to one of the 14 warning annunciator inputs and connect DS1 to the associated output.

b. Press S1 to apply 28 vdc at the selected input on the PCB. A 28 vdc signal from the respective output should illuminate DS1. DS2 should also be illuminated each time 28 vdc is applied to a warning annunciator input.

c. Disconnect S1 and DS1 from the PCB.

d. Repeat steps a, b and c for the remaining warning annunciator inputs and outputs, including any spares.

e. Connect S2, DS1, DS2 and DS3 to the PCB.



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Annunciator Control PCB Functional Test Circuit (Effectivity: All)
Figure 9

f. Press S2. All of the warning annunciator outputs should have 28 vdc present. Check each output individually with DS1 to verify the test circuit. Both DS2 and DS3 should remain illuminated until S2 is released.

WARNING ANNUNCIATOR SYSTEM OPERATIONAL TEST (Effectivity: All)

The operational test for the warning annunciator system consists of pressing the ANNUN test switch on the center subpanel in the cockpit and observing the warning annunciators. If any part of the operational test indicates a fault, refer to the applicable troubleshooting chart in this chapter to further isolate the fault. An operational check of the ambient light sensor circuit is also performed in this test.

a. Place the battery switch in the ON position to power the triple-fed bus.

b. Press and hold the ANNUN test switch on the pilot's outboard subpanel. Observe the following:

1. All fourteen warning annunciators in the warning annunciator panel should illuminate.

2. The pilot and copilot master warning and master caution lights should illuminate and flash on and off.

3. The EICAS yellow CAUTION TEST caution message should display on the EICAS display.

c. Release the ANNUN test switch. The warning annunciators should extinguish.

d. Press the legend face on one master warning light. Both master warning lights should stop flashing and extinguish.

NOTE

The EICAS must be ON for any EICAS messages to display.

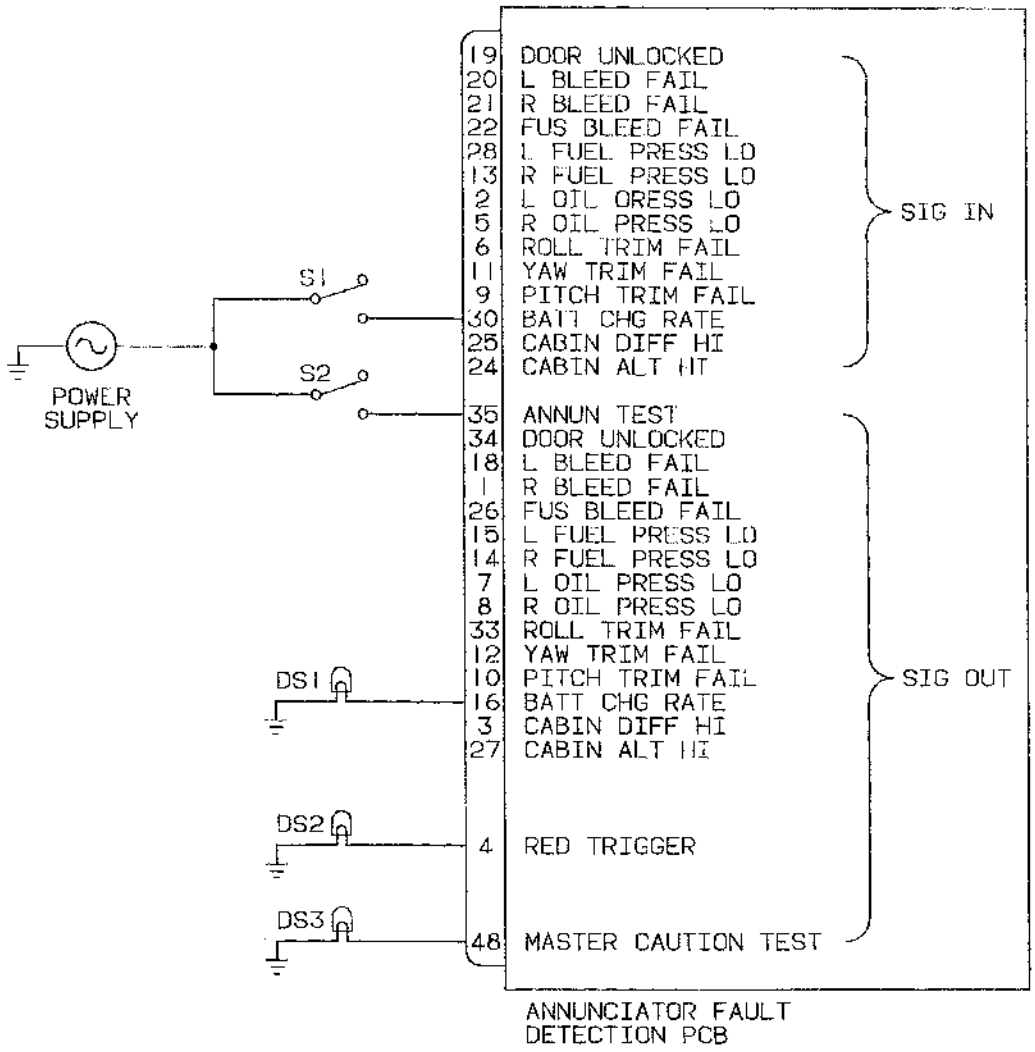
e. Press the legend face on one master caution light. Both master caution lights should stop flashing and extinguish.

NOTE

The EICAS yellow CAUTION TEST message should no longer be displayed on the EICAS display.

f. Repeat steps b thru e to test the reset switch on the opposite master warning and master caution lights.

g. With the warning annunciators illuminated, check operation of the ambient light sensor circuit by changing the light intensity directly in front of the ambient light sensor. The ambient light sensor is located adjacent to the PILOT DISPLAYS potentiometer on the overhead panel in the cockpit. The warning annunciators, master warning lights and master caution lights should increase or decrease in brightness as light intensity at the ambient light sensor is changed. Refer to Chapter 33-10-00 to replace the ambient light sensor.



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Annunciator Fault Detection PCB Functional Test Circuit (Effectivity: All)
Figure 10

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WARNING

When an airplane has experienced abnormal landing gear procedures of any type, as a safety precaution, place the airplane on jacks prior to performing any inspection or maintenance. Ensure that all three landing gears are down and locked prior to removing the airplane from jacks.

CAUTION

Jacking of an airplane for the purpose of landing gear operation, inspection, servicing or maintenance, should be accomplished within an enclosed building or hangar. In the intrest of safety, should it become necessary to jack the airplane in the open, wind velocity in any direction and terrain variations, must be compensated for prior to jacking the airplane.

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer Chapters 32-10-00 and 32-20-00 for the proper lockpin installation procedures.

The main landing gear are conventional oleo (air-hydraulic) units that act as shock absorbers. Each main landing gear is mounted in the wheel well to trunnion fittings by pivot pins. When the actuator piston is extended, the landing gear is retracted inboard into the wheel well. A landing gear door is connected to the outboard side of each landing gear by door rods. The doors are pulled closed and pushed open as the landing gear is retracted or extended.

The nose landing gear is a conventional oleo (air-hydraulic) unit that acts as a shock absorber. The nose landing gear is mounted in the wheel well to trunnion fittings by trunnion pins. The landing gear is extended and retracted by a hydraulic actuator connected to the nose structure at one end and to the nose gear at the other end. The nose wheel well is covered by four clam-shell doors when the landing gear is fully retracted. The aft set of doors are connected to the landing gear and are opened and closed as the gear is retracted and extended. The forward set of doors are opened and closed by the hydraulic door actuator. The forward doors are open only while the landing gear is retracting or extending and remain closed when the landing gear is fully extended or retracted.

The extension and retraction system consists of two independent systems. The primary system is powered by an electric motor to provide hydraulic pressure to retract and extend the landing gear and nose gear forward doors. The alternate extension system utilizes a hand pump to extend the gear in the event that the primary system is disabled in any way.

The airplane is equipped with four hydraulically operated brake assemblies. Each main landing gear incorporates two multidisc, carbon-lined brake assemblies bolted together, one on each side of the strut.

A nose landing gear steering system is installed to facilitate turning while the airplane is on the ground. Steering action is transmitted from the rudder pedals to the nose gear through a series of pushrods and bellcranks.

Visual indication of landing gear position is provided by two red in-transit lights located in the landing gear control handle and by a green GEAR DOWN indicator light assembly (labeled NOSE-L-R) located adjacent to the landing gear control handle on the pilot's inboard subpanel.

SPECIAL TOOLS AND RECOMMENDED MATERIALS (Effectivity: All)

The special tools and recommended materials listed in Charts 1 and 2 as meeting federal, military or supplier specifications are provided for reference only and

are not specifically prescribed by Raytheon Aircraft Company. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Raytheon Aircraft Company, the supplier, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the supplier of the product to be used.

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All)

TOOL NAME	PART NO.	SUPPLIER	USE
1. Main Gear Lockpin	122-590017-1	Raytheon Aircraft Company Wichita, Kansas	Lock Main Gear in Down and Locked Position.
2. Nose Gear Lockpin	122-590017-1	Raytheon Aircraft Company Wichita, Kansas	Lock Nose Gear in Down and Locked Position.
3. Nose Gear Door Lockpin	122-590017-3	Raytheon Aircraft Company Wichita, Kansas	Lock Nose Gear Doors in Open Position.
4. Hydraulic Hand Pump	Model 06-4005-0501 (0-3000 psi with reservoir and shutoff valve)	Tronair, Inc. S. 1740 Eber Rd. Holland, OH 43528-9635	Position Gear for Rigging.
5. Bleeder Hose Adapter	TK889-2 D928	Raytheon Aircraft Company Wichita, Kansas	Bleed Brakes.
6. Bleeder Hose Assembly	None	Fabricate as described in this chapter	Bleed Brakes.
7. Drill Motor (800 to 1200 rpm)	None	Commercially Available	Spin Wheel Speed Transducer for Functional Check.
8. Wheel Drive Clip Fixture	114-810055-939	Raytheon Aircraft Company Wichita, Kansas	Connect Wheel Speed Transducer to Drill Motor.
9. Pressure Gages (2) (0-1000 psi)	None	Commercially Available	Measure Brake Pressure at Main Landing Gear.
10. Nose Gear Test Box	122-820001/935-2	Raytheon Aircraft Company Wichita, Kansas	Adjustment of Nose Gear Switches.
11. Ohmmeter	None	Commercially Available	Functional Test of Switches.
12. Main Gear Test Box	122-810001/935-1	Raytheon Aircraft Company Wichita, Kansas	Adjustment of Main Gear Switches.
13. Manometer	None	Commercially Available	Functional Test of Airspeed Switch.
14. Pressure Gage	None	Commercially Available	Test of Landing Gear System.

CHART 2
RECOMMENDED MATERIALS (Effectivity: All).

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
1. Hydraulic Fluid	MIL-H-5606	2126 Hydraulic Oil	Exxon Company U.S.A. P.O.Box 2108 Houston, TX 77001
		PED 3337	Standard Oil of California 225 Bush St. San Francisco, CA 94120
2. Grease	MIL-G-81322	Mobilgrease 28	Mobil Oil Corp. Shoreham Bldg. Washington, DC 20005
		Royco 22	Royal Lubricants Co. River Rd & Merry Ln. East Hanover, NJ 07936
3. Solvent (Mineral Spirits)	PD680, Type III		Obtain Locally
4. Retaining Compound		Retaining Compound 680	Loctite Corp. 705 N. Mountain Rd. Newington, CT 06111
5. Threadlock Adhesive		Loctite 222	Loctite Corp. 705 N. Mountain Rd. Newington, CT 06111
6. Sealant	RTV-162 RTV-738 RTV-3145	Silicone Sealant	General Electric Silicone Products Business Dept. Hudson River Rd. Waterford, NY 12188
7. Adhesive	No. 609	Adhesive	Loctite Corp. 705 N. Mountain Rd. Newington, CT 06111
8. Wheel Bearing Grease		AmsOil Heavy Duty (GHD)	AMS/OIL 2206 Winter Street Superior, Wisconsin

MAIN GEAR AND DOORS - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to this chapter and Chapter 32-20-00 for the proper lockpin installation procedures.

The main landing gear are conventional oleo (air-hydraulic) units that act as shock absorbers. A charging valve is installed in the gear for use when servicing the shock strut with hydraulic fluid and air.

Each main landing gear is mounted in the wheel well-to-trunnion fittings by pivot pins. The trunnion fittings contain spherical bearings into which the pivot pins protrude when the landing gear is installed. A folding side brace is connected to the gear assembly and extends inboard where it is attached to the wing structure.

The landing gear is extended and retracted by a hydraulic actuator that is connected to the upper end of the side brace and to the landing gear assembly through two drive links and a jury-brace retract link. When the actuator piston is extended, the landing gear is retracted inboard into the wheel well. When fully retracted, an uplock roller on the landing gear engages an uplock hook that holds the landing gear in the retracted position. When the actuator piston is retracted, the landing gear is extended to the down and locked position. Before the gear can be extended, hydraulic pressure retracts the uplock actuator piston to release the uplock hook from the uplock roller. The landing gear is secured in the down and locked position by the overcenter action of the lock link assemblies.

A landing gear door is connected to the outboard side of each landing gear by door rods. The doors are pulled closed and pushed open as the landing gear is retracted or extended. When closed, the doors cover the wheel well except for part of the wheel and tire.

MAIN GEAR AND DOORS - MAINTENANCE PRACTICES (Effectivity: All)

WARNING

When performing maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance. Cycle the landing gear and ensure that all three landing gears are down and locked prior to removing the aircraft from jacks.

MAIN LANDING GEAR SHOCK ABSORBER SERVICING (Effectivity: All)

To check the fluid level and fill the main landing gear shock absorbers, refer to Chapter 12-10-00.

LUBRICATION (Effectivity: All)

Lubricate the main gear wheel bearings and the torque knee grease fittings as detailed in the Lubrication Schedule in Chapter 12-20-00.

MAIN LANDING GEAR LOCKPIN INSTALLATION (Effectivity: All) (Figure 2)

Anytime the airplane is on the ground (whether on jacks or on the wheels), the main landing gear must be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. Install the lockpin (1, Chart 1, 32-00-00) as detailed below:

WARNING

Use only a 122-590017-1 lockpin (1, Chart 1, 32-00-00) to lock the landing gear in the down and locked position, DO NOT use any pin other than the one listed. DO NOT push on, or in any way move the lock link assembly when inserting the lockpin.

a. Ensure that the landing gear is in the down and locked position.

b. Install a 122-590017-1 lockpin (1, Chart 1, 32-00-00) into the lock link assembly of each main gear. Refer to Figure 2 for the lockpin hole location.

MAIN LANDING GEAR REMOVAL (Effectivity: All) (Figure 3)

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. After the airplane is on jacks remove the lockpin from the lock link assembly.

c. Remove all electrical power from the airplane.

d. Remove the nuts (30), washers (28) and bolts (29) to disconnect the door rods (31) from the landing gear door.

e. Remove the downlock spring assembly (7) as follows:

1. Remove the cotter pin, washer (11) and pin (10) to disconnect the lower end of the downlock spring assembly (7) from main gear brace.

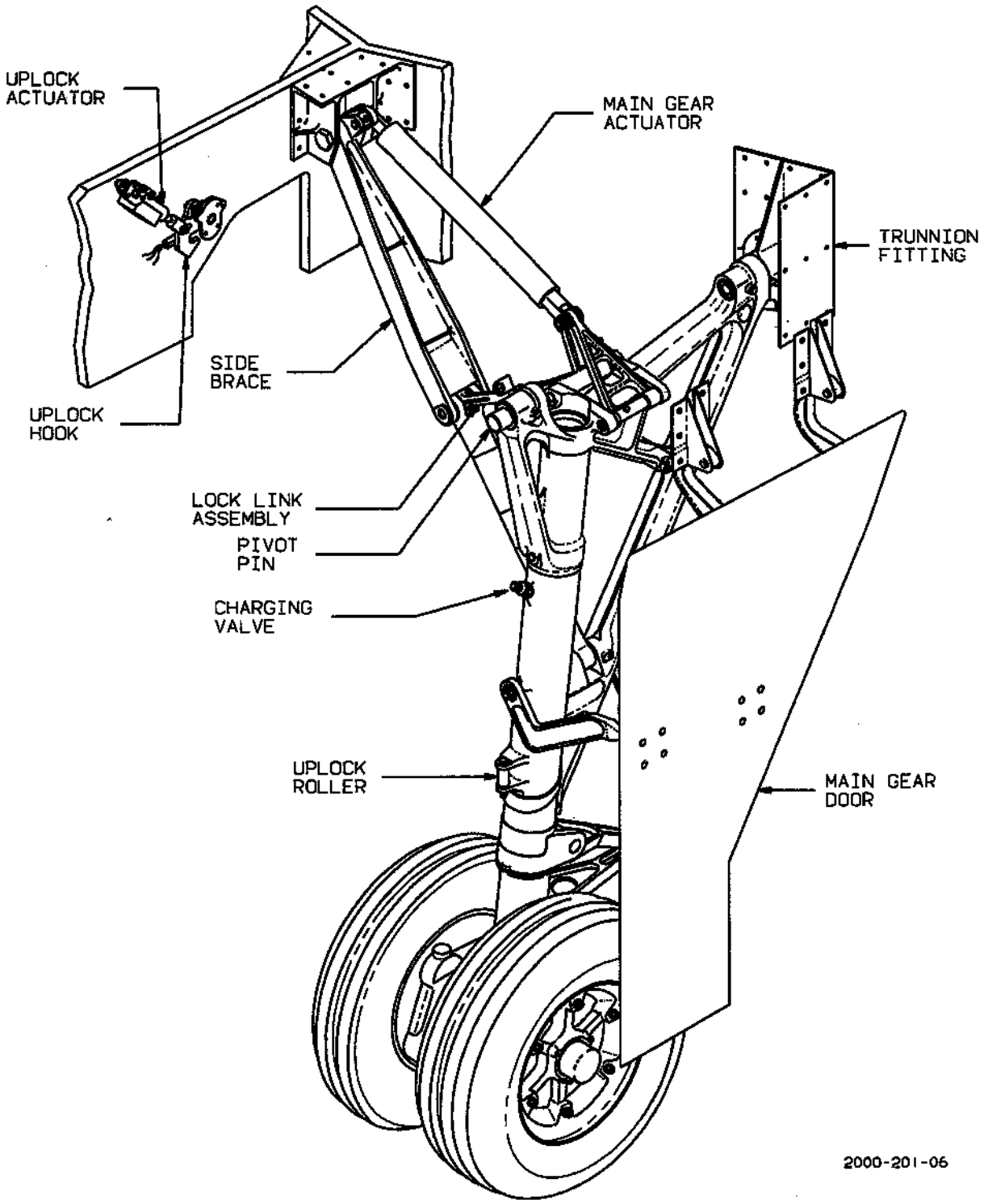
2. Remove the cotter pin, nut (5), washers (2 and 3) and bolt (1) to disconnect the upper end of the spring assembly from the spring crank (6). Discard washer (2).

f. Remove the cotter pin, nut (9), washer (8) and bolt (4) to remove the spring crank (6) from pin (21).

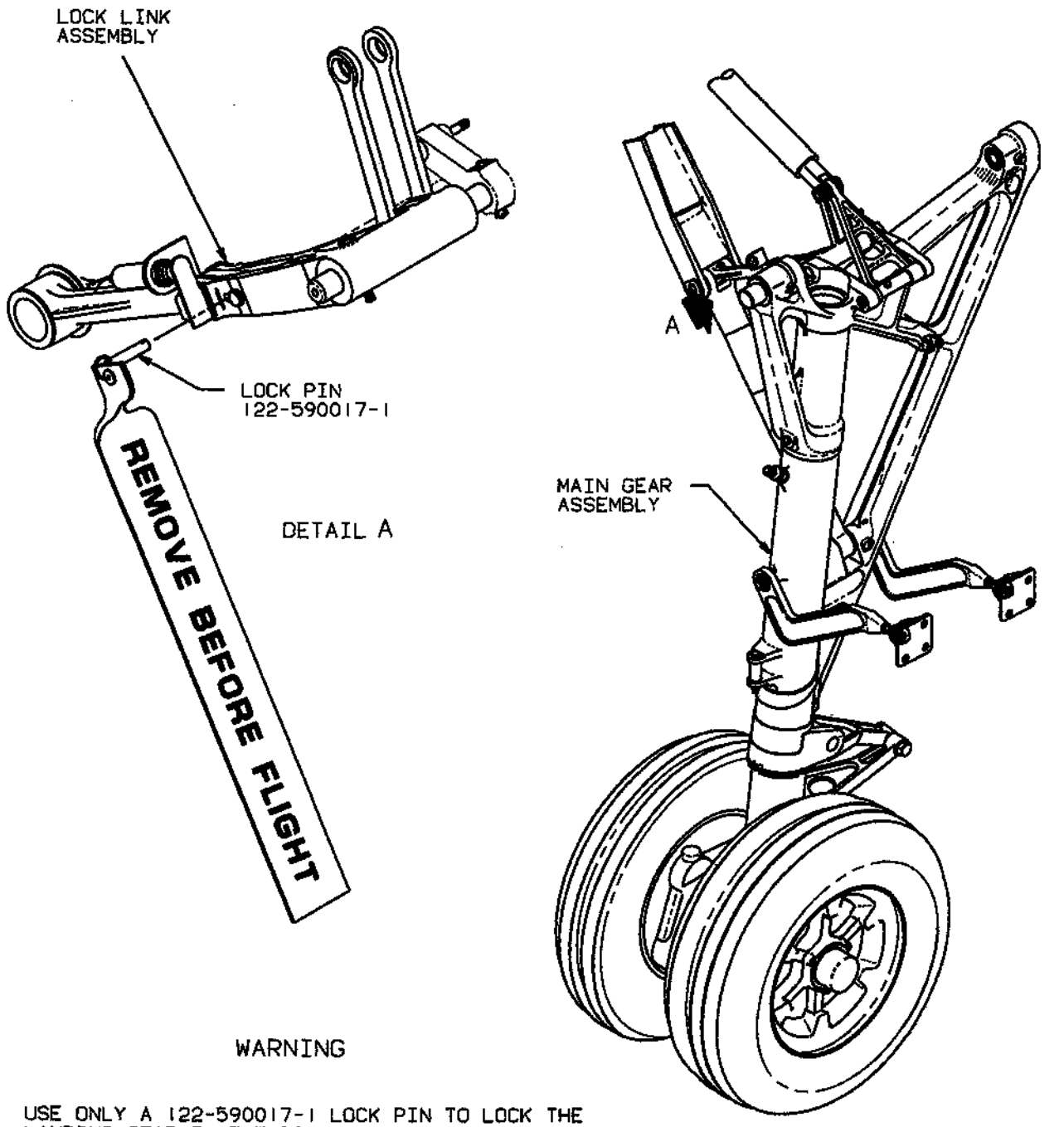
g. Remove the cotter pin, washer (14) and pin (12) from the pin (15) and remove sleeve (13). Remove the pin (15) to disconnect the actuator from the jury brace retract link (17) and drive links (16).

h. Remove the cotter pin, nut (20), washers (19) and bolt (18) from the pin (21) and lock link assembly (22).

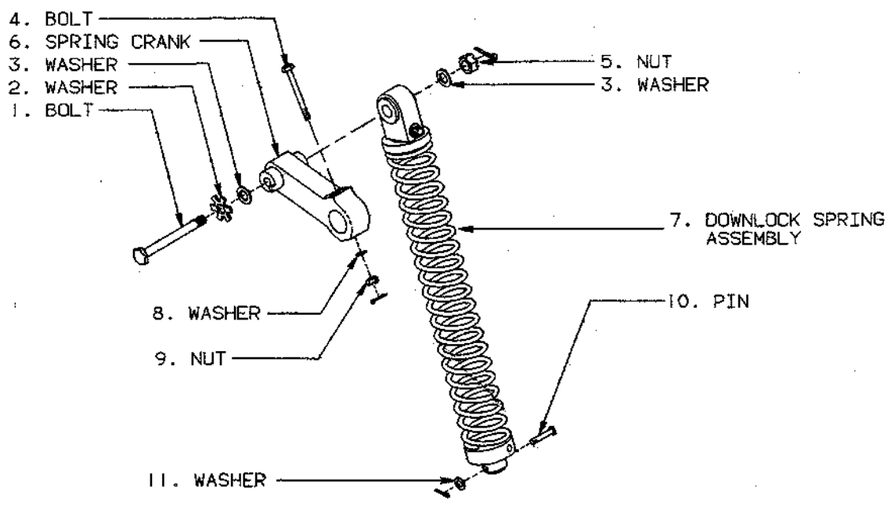
i. Remove pin (21) to free the lock link assembly (22) from the main gear brace.



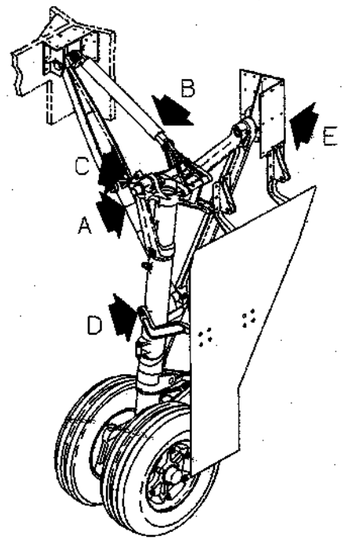
Main Landing Gear and Door (Effectivity: All)
Figure 1

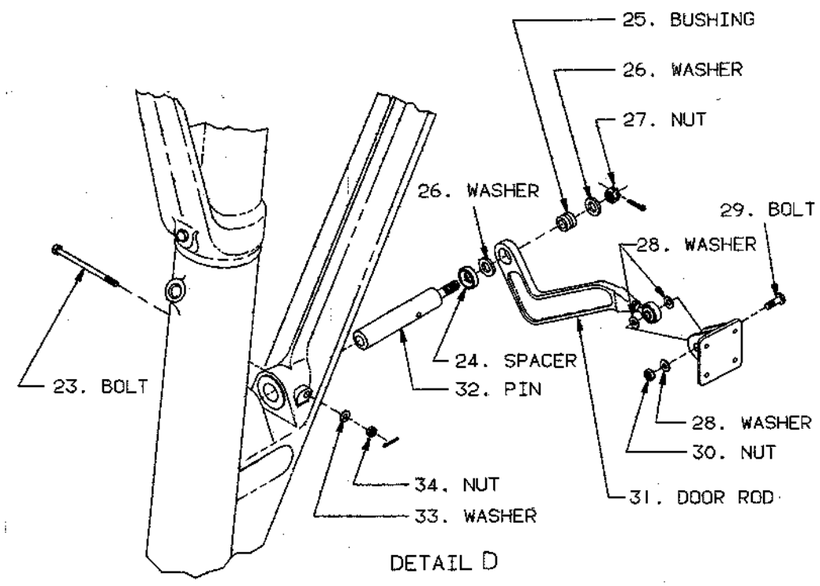


Main Landing Gear Lockpin Installation (Effectivity: All)
Figure 2

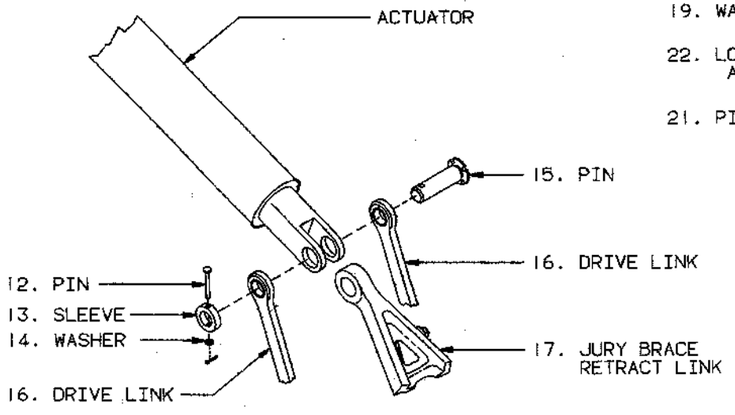


DETAIL A

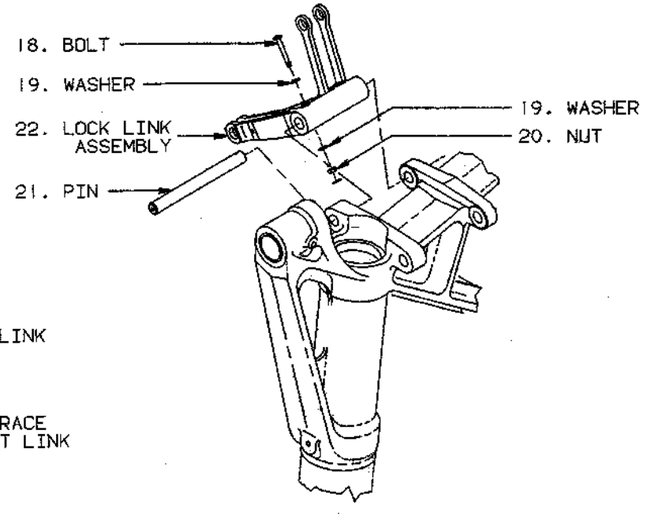




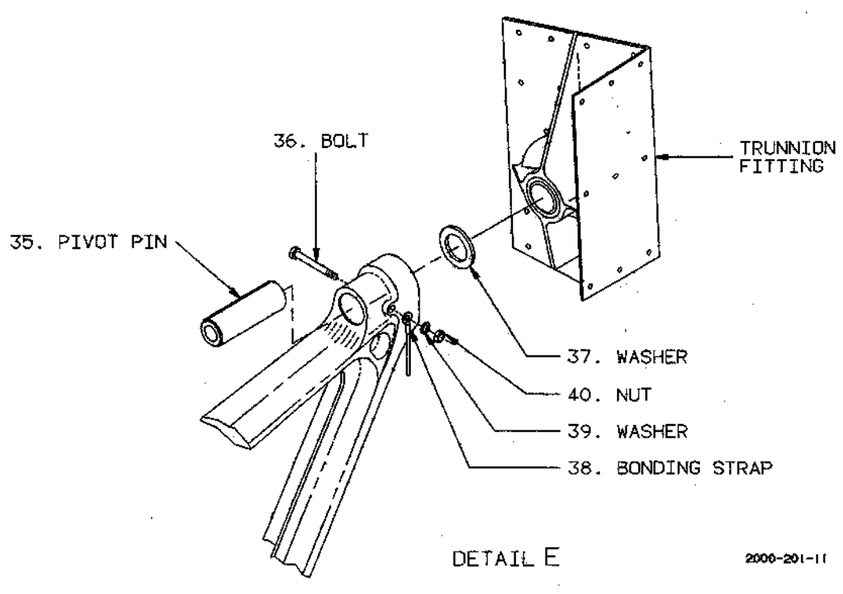
DETAIL D



DETAIL B



DETAIL C



DETAIL E

2000-201-11

Main Landing Gear Installation (Effectivity: All)
Figure 3

j. Remove the cotter pin, nut (27), washers (26), bushing (25), door rod (31) and spacer (24) from the pin (32).

k. Remove the cotter pin, nut (34), washer (33) and bolt (23) from the main gear brace and pin (32).

l. Remove the pin (32) to disconnect the side brace from the main gear.

m. When the pivot pins (35) are removed from the trunnion bearings, the main gear will be released from the airplane. Before removing the pivot pins, place a block under the main wheels to support the main gear.

n. Remove the cotter pins and remove the bolts (36), washers (39) and nuts (40) that secure the pivot pins (35) to the main gear.

o. Remove the pivot pins (35) until they are free of the bearings in the trunnion fittings and lower the main gear away from the airplane.

NOTE

When the pivot pins are removed from the trunnion bearings, the washers (37) installed between the main gear and the trunnion bearings will be released. Note the positions of the washers to facilitate alignment of the main gear to the side brace when the main gear is installed.

MAIN LANDING GEAR INSTALLATION (Effectivity: All) (Figure 3)

a. Install the pivot pins (35) into the main gear brace.

b. Position the main gear in the wheel well and install the pivot pins (35) into the trunnion bearings with the washers (37) in their original positions between the main gear and the trunnion bearings. Use washers (37) as required to provide a total clearance of .005 to .03 inch between the main gear and the trunnion fittings. Align the holes in the pivot pins with those in the main gear and install the bolts (36), washers (39) and nuts (40) to secure the pivot pins to the main gear, install the bonding strap (38) under the washer (39) of the aft pivot pin.

c. Check the alignment of the side brace with the attachment lug on the main gear; there shall be no side loads on the side brace when the landing gear is in the extended or retracted position. Correct any misalignment by shifting the washers (37) from side to side of the main gear and the trunnion bearings. A minimum of one washer must be installed between the main gear and each trunnion bearing.

d. After the side brace and main gear are properly aligned, connect the side brace to the main gear with the pin (32). Align the hole in the pin (32) with the hole in the main gear and install the bolt (23), washer (33), nut (34) and cotter pin.

e. Connect the lock link assembly (22) to the main gear with the pin (21).

f. Align the hole in the pin (21) with the hole in the lock link assembly (22) and install the bolt (18), washers (19) and nut (20). Use AN960KD10 and/or AN960KD10L washers (19) under the bolt head and nut as required to allow engagement of the cotter pin with the nut when the bolt and nut are tightened finger-tight. A minimum of one washer (19) should be placed under both the bolt head and nut (20). Install the cotter pin.

g. Install the spring crank (6) on the pin (21) and secure with the bolt (4), washer (8), nut (9) and cotter pin.

h. Align the actuator rod end, jury brace retract link (17) and drive links (16) and install the pin (15). Install the sleeve (13) on the pin (15) and install pin (12), washer (14) and cotter pin.

i. Install the downlock spring assembly (7) as follows:

1. Install the bolt (1), new washer (2), washers (3) and nut (5) to connect the upper end of the spring assembly (7) to the spring crank (6). Use AN960KD10 and/or AN960KD10L washers (3) under the bolt head and nut as required to allow engagement of the cotter pin with the nut when the bolt and nut are tightened finger-tight. A minimum of one washer should be placed under both the bolt head and the nut. Install the cotter pin and bend one tang of the washer (2) over the flat of the bolt head and bend one over the spring crank (6).

2. Install the pin (10), washer (11) and cotter pin to connect the lower end of the spring assembly to the main gear.

j. Install the spacer (24), door rod (31), bushing (25), washers (26), nut (27) and cotter pin on the pin (32).

NOTE

Install the bushing (25) with the countersink toward the main gear brace.

k. Connect the door rods (31) to the main gear door with bolts (29), washers (28) and nut (30).

l. Check the landing gear and door rigging as outlined in Chapter 32-30-00, and adjust the downlock switch and the uplock switch as outlined in Chapter 32-60-00.

m. Ensure the landing gear is in the down and locked position and install the lockpin in the main gear as detailed in this chapter.

n. Remove the airplane from the jacks.

o. Restore electrical power to the airplane.

MAIN LANDING GEAR SIDE BRACE REMOVAL (Effectivity: All) (Figure 4)

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. After the airplane is on jacks remove the lockpin from the lock link assembly.

c. Remove all electrical power from the airplane.

d. Remove the nuts (30), washers (28) and bolts (29) to disconnect the door rods (31) from the landing gear door.

e. Remove the downlock spring assembly (7) as follows:

1. Remove the cotter pin, washer (11) and pin (10) to disconnect the lower end of the spring assembly (7) from main gear brace.

2. Remove the cotter pin, nut (5), washers (2 and 3) and bolt (1) to disconnect the upper end of the spring assembly from the spring crank (6). Discard the washer (2).

f. Remove the cotter pin, nut (9), washer (8) and bolt (4) to remove the spring crank (6) from pin (21).

g. Remove the cotter pin, washer (14) and pin (12) from the pin (15) and remove sleeve (13). Remove the pin (15) to disconnect the actuator from the jury brace retract link (17) and drive links (16).

h. Remove the cotter pin, washer (42) and pin (38) that attach the pin (39) to the side brace.

i. Remove the pin (39) to disconnect the actuator from the side brace. Tie the actuator up out of the way.

j. Remove the cotter pin, nut (20), washers (19) and bolt (18) from the pin (21) and lock link assembly (22).

k. Remove pin (21) to free the lock link assembly (22) from the main gear brace.

l. Remove the cotter pin, nut (27), washers (26), bushing (25), door rod (31) and spacer (24) from the pin (32).

m. Remove the cotter pin, nut (34), washer (33) and bolt (23) from the main gear brace and pin (32).

n. Remove the pin (32) to disconnect the side brace from the main gear.

o. Remove the cotter pin, nut (35), and washer (36). Remove bolt (43) from the retract fitting to free the side brace and bushing (41) from the fitting.

NOTE

When the bolt (43) is removed from the retract fitting, the washers (37) installed between the side brace and fitting will be released. Note the positions of the washers to facilitate alignment of the side brace to the main gear when the side brace is installed.

MAIN LANDING GEAR SIDE BRACE INSTALLATION (Effectivity: All) (Figure 4)

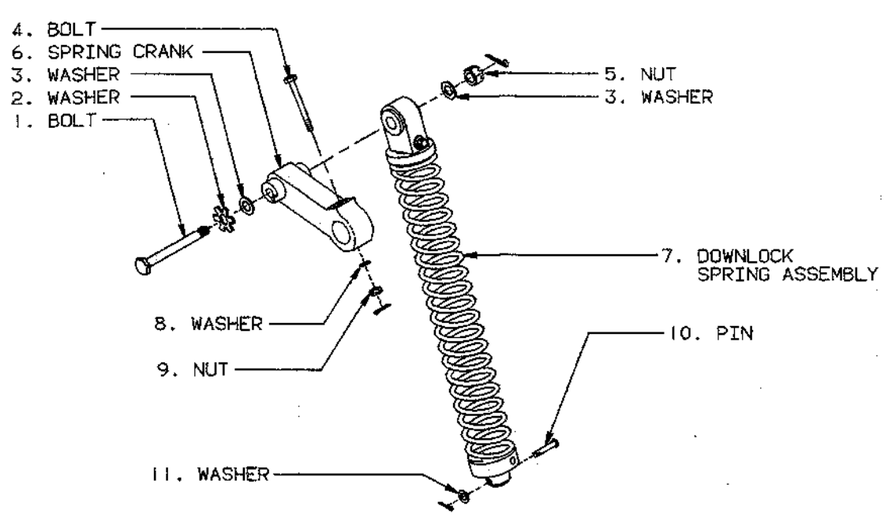
a. Install the bushing (41) into the upper end of the side brace and position the upper end of the side brace in the retract fitting with washers (37) in their original positions on either side of the side brace and install bolt (43). Use washers (37) as required to remove any noticeable end play. Install the washer (36), nut (35) and cotter pin.

b. Check the alignment of the side brace with the attachment lug on the main gear; there shall be no side loads on the side brace when the landing gear is in the extended or retracted position. Correct any misalignment by shifting the washers (37) from side to side.

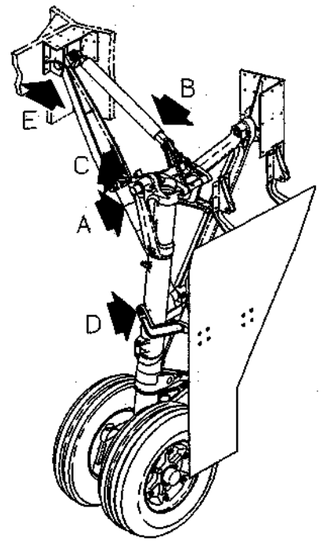
c. After the side brace and main gear are properly aligned, connect the side brace to the main gear with the pin (32). Align the hole in the pin (32) with the hole in the main gear and install the bolt (23), washer (33), nut (34) and cotter pin.

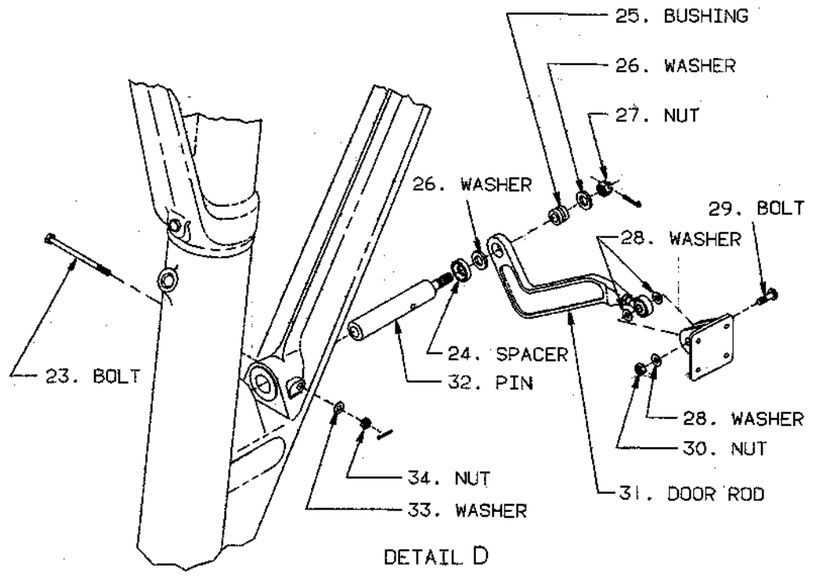
d. Connect the lock link assembly (22) to the main gear with pin (21).

e. Align the hole in the pin (21) with the hole in the lock link assembly (22) and install the bolt (18), washers (19) and nut (20). Use AN960KD10 and/or AN960KD10L washers (19) under the bolt head and nut as required to allow engagement of the cotter pin

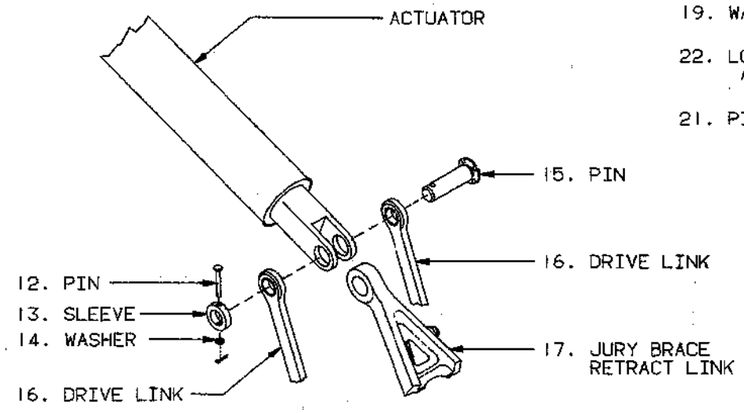


DETAIL A

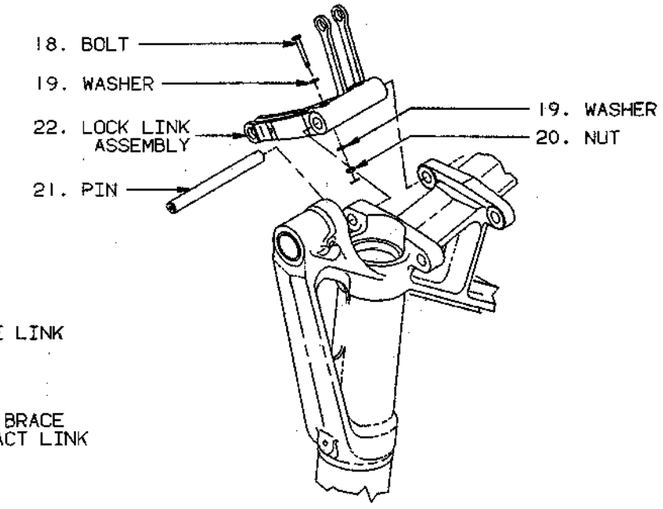




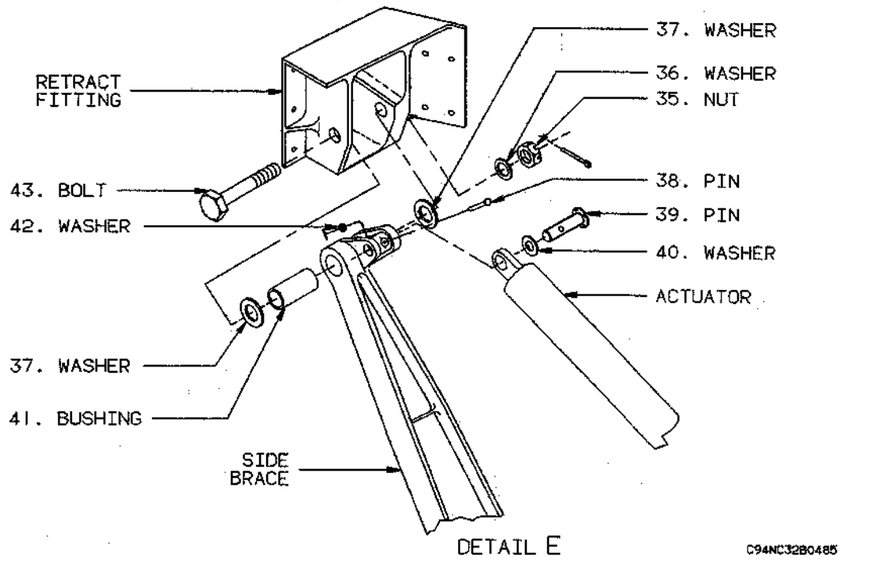
DETAIL D



DETAIL B



DETAIL C



DETAIL E

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Main Landing Gear Side Brace Installation
(Effectivity: All)
Figure 4

with the nut when the bolt and nut are tightened finger-tight. A minimum of one washer (19) should be placed under both the bolt head and nut. Install the cotter pin.

f. Install the spring crank (6) on the pin (21) and secure with the bolt (4), washer (8), nut (9) and cotter pin.

g. Install the pin (39) and washers (40) to connect the actuator to the side brace. Use washers (40) as required under the head of pin (39) to remove play in the joint when pin (38) is installed.

h. Align the hole in pin (39) with the hole in the side brace and install pin (38), washer (42) and cotter pin.

i. Align the actuator rod end, jury brace retract link (17) and drive links (16) and install the pin (15). Install the sleeve (13) on the pin (15) and install pin (12), washer (14) and cotter pin.

j. Install the downlock spring assembly (7) as follows:

1. Install the bolt (1), new washer (2), washers (3) and nut (5) to connect the upper end of the spring assembly to the spring crank (6). Use AN960KD10 and/or AN960KD10L washers (3) under the bolt head and nut as required to allow engagement of the cotter pin with the nut when the bolt and nut are tightened finger-tight. A minimum of one washer should be placed under both the bolt head and the nut. Install the cotter pin and bend one tang of washer (2) over the flat of the bolt head and bend one over the spring crank (6).

2. Install the pin (10), washer (11) and cotter pin to connect the lower end of the spring assembly (7) to the main gear.

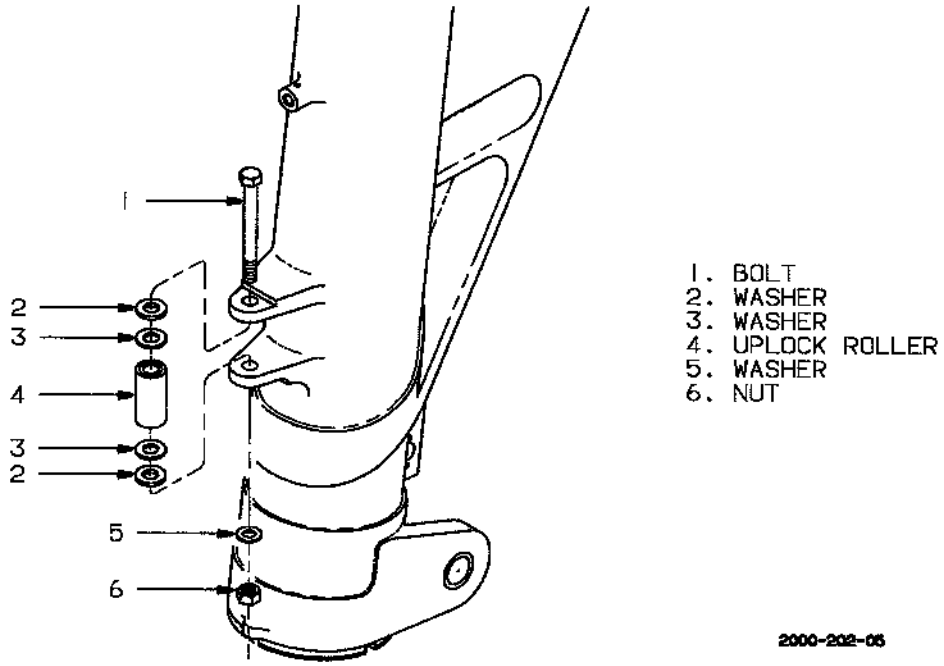
k. Install the spacer (24), door rod (31), bushing (25), washers (26), nut (27) and cotter pin on the pin (32).

NOTE

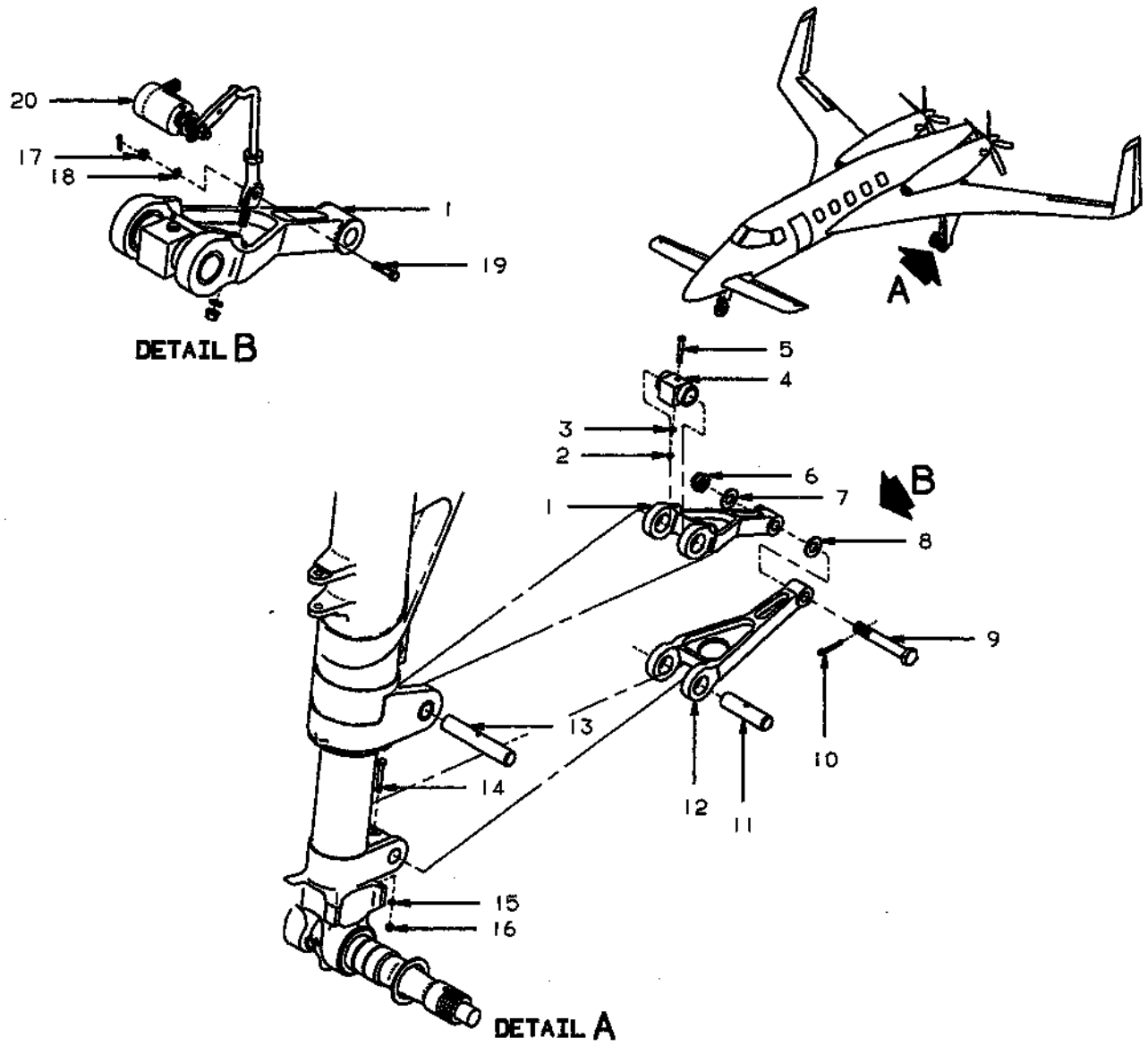
Install bushing (25) with the countersink toward the main gear brace.

l. Connect the door rods (31) to the main gear door with bolts (29), washers (28) and nut (30).

m. Check the landing gear and door rigging as outlined in Chapter 32-30-00, and adjust the downlock switch and the uplock switch as outlined in Chapter 32-60-00.



Uplock Roller Installation (Effectivity: All)
Figure 5



1. UPPER TORQUE KNEE
2. NUT
3. WASHER
4. SPACER
5. BOLT
6. NUT
7. WASHER
8. WASHER
9. PIN
10. COTTER PIN
11. LOWER TORQUE KNEE PIN
12. LOWER TORQUE KNEE
13. UPPER TORQUE KNEE PIN
14. BOLT
15. WASHER
16. NUT
17. NUT
18. WASHER
19. BOLT
20. SAFETY SWITCH

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Main Gear Torque Knee Installation (Effectivity: All)
Figure 6

n. Ensure the landing gear is in the down and locked position and install the lockpin in the lock link assembly as detailed in this chapter.

o. Remove the airplane from the jacks.

p. Restore electrical power to the airplane.

UPLOCK ROLLER REMOVAL (Effectivity: All) (Figure 5)

a. Remove the nut (6) and washer (5) from bolt (1).

b. Remove the uplock roller (4) and washers (2 and 3) from the main gear brace.

UPLOCK ROLLER INSTALLATION (Effectivity: All) (Figure 5)

a. Pack the inside diameter of uplock roller (4) and bearings and coat the outside diameter of bolt (1) with grease (2, Chart 2, 32-00-00).

b. Position the uplock roller (4) between the lugs on the main gear brace. Install washers (2 and 3) on each side of the uplock roller (4) and install the attaching bolt (1).

c. Install the washer (5) and nut (6) on the bolt (1). Tighten the nut only enough to remove any noticeable end play.

MAIN GEAR TORQUE KNEE REMOVAL (Effectivity: All) (Figure 6)

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Remove all electrical power from the airplane.

c. Remove the cotter pin, nut (17), washer (18) and bolt (19) to disconnect the safety switch (20) from the upper torque knee (1).

d. Remove the cotter pin (10), nut (6), washers (7 and 8) and bolt (9) to separate the torque knees.

e. Remove nut (2), washer (3) and bolt (5) that secure the upper torque knee (1) and pin (13) to the main gear brace.

f. Remove pin (13) to free the spacer (4) and upper torque knee (1).

g. Remove nut (16), washer (15) and bolt (14).

h. Remove pin (11) to free the lower torque knee (12) from the main gear shock absorber piston.

MAIN GEAR TORQUE KNEE INSTALLATION (Effectivity: All) (Figure 6)

a. Place the lower torque knee (12) on the main gear shock absorber piston and install pin (11) to hold the knee in place.

b. Align the hole in the pin (11) with the hole in the main gear shock absorber piston and install bolt (14), washer (15) and nut (16).

c. Place the upper torque knee (1) and spacer (4) on the main gear brace and install pin (13) to hold the spacer and knee in place.

d. Align the hole in the pin (13) with the hole in the spacer (4) and install bolt (5), washer (3) and nut (2).

e. Connect the upper and lower torque knees with bolt (9), washers (7 and 8) and nut (6). Use a new washer (7). Tighten nut (6) as required to remove any noticeable end play but not enough to create a tight joint. Install cotter pin (10) and bend a minimum of one tang of washer (7) over the flat of the nut (6).

f. Connect the safety switch (20) to the upper torque knee (1) with bolt (19), washer (18), nut (17) and cotter pin.

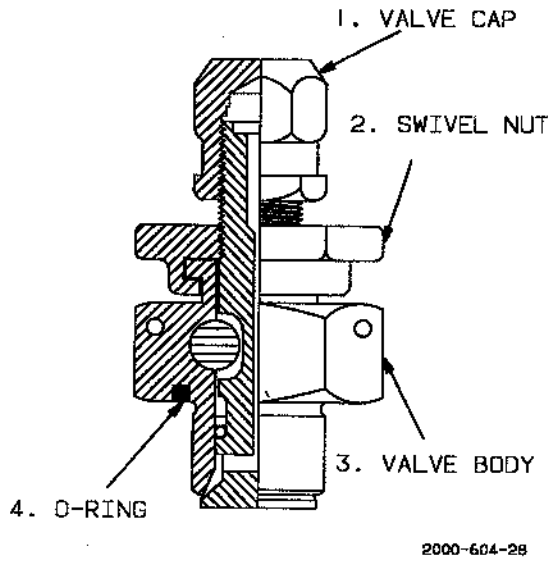
CHARGING VALVE REMOVAL (Effectivity: All) (Figure 7)

WARNING

Never remove the charging valve from the landing gear assembly until all of the pressure is released from the strut.

a. Loosen the valve cap (1) about one turn to release the trapped air, then remove the cap from the charging valve.

b. Release the air pressure in the strut by slowly turning the swivel nut (2) counterclockwise. Do not allow the valve body (3) to turn while loosening the swivel nut. Two and one-fourth turns will open the valve completely. Allow the strut to fully compress.



Charging Valve Installation (Effectivity: All)
Figure 7

WARNING

Stand to one side of the valve while releasing the air pressure.

c. Remove the valve body (3) from the main gear assembly.

CHARGING VALVE INSTALLATION (Effectivity: All) (Figure 7)

a. Ensure that the charging valve is clean.

b. Lubricate a new O-ring (4) with hydraulic fluid (1, Chart 2, 32-00-00). Install the O-ring on the charging valve, use care to avoid damaging the O-ring during installation.

c. Screw the valve body (3) into the main gear assembly until the hexagonal portion of the valve body contacts the boss. Tighten the valve body to a torque of 100 to 110 inch-pounds and safety wire.

d. Tighten the swivel nut (2) to a torque of 50 to 70 inch-pounds

e. Install the valve cap (1) finger-tight.

f. Refer to Chapter 12-10-00 and service the main gear shock absorber assembly.

NOSE GEAR AND DOORS - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to this chapter and Chapter 32-10-00 for the proper lockpin installation procedures.

The nose landing gear is a conventional oleo (air-hydraulic) unit that acts as a shock absorber. A charging valve is installed in the landing gear for use when servicing the shock strut with hydraulic fluid and air.

The nose landing gear is mounted in the wheel well to trunnion fittings by trunnion pins. The trunnion fittings contain spherical bearings into which the trunnion pins protrude when the landing gear is installed. A folding drag brace is connected to the gear assembly and extends forward into the wheel well where it is attached to the nose structure.

The landing gear is extended and retracted by a hydraulic actuator connected to the nose structure at one end and to nose gear at the other end. When the actuator piston is retracted, the landing gear is pulled forward into the wheel well. When fully retracted, an uplock roller on the landing gear engages an uplock hook that holds the landing gear in the retracted position. When the actuator piston is extended, the landing gear is extended to the down and locked position. Before the gear can be extended, hydraulic pressure retracts the uplock actuator piston to release the uplock hook from the uplock roller. The landing gear is secured in the down and locked position by the over-center action of the drag brace and lock links.

The wheel well is covered by four clam-shell doors when the landing gear is fully retracted. The aft set of doors are connected to the landing gear and are opened and closed as the gear is retracted and extended. The forward set of doors are opened and closed by the hydraulic door actuator. The forward doors are open only while the landing gear is retracting or extending and remain closed when the landing gear is fully extended or retracted. When the airplane is on the ground, the forward set of doors can be opened by pumping the alternate extension hand-pump handle in the cockpit.

The nose gear is steerable through the rudder pedals.

A tow bar fitting is mounted on the forward side of the landing gear to allow for the connection of a tow bar.

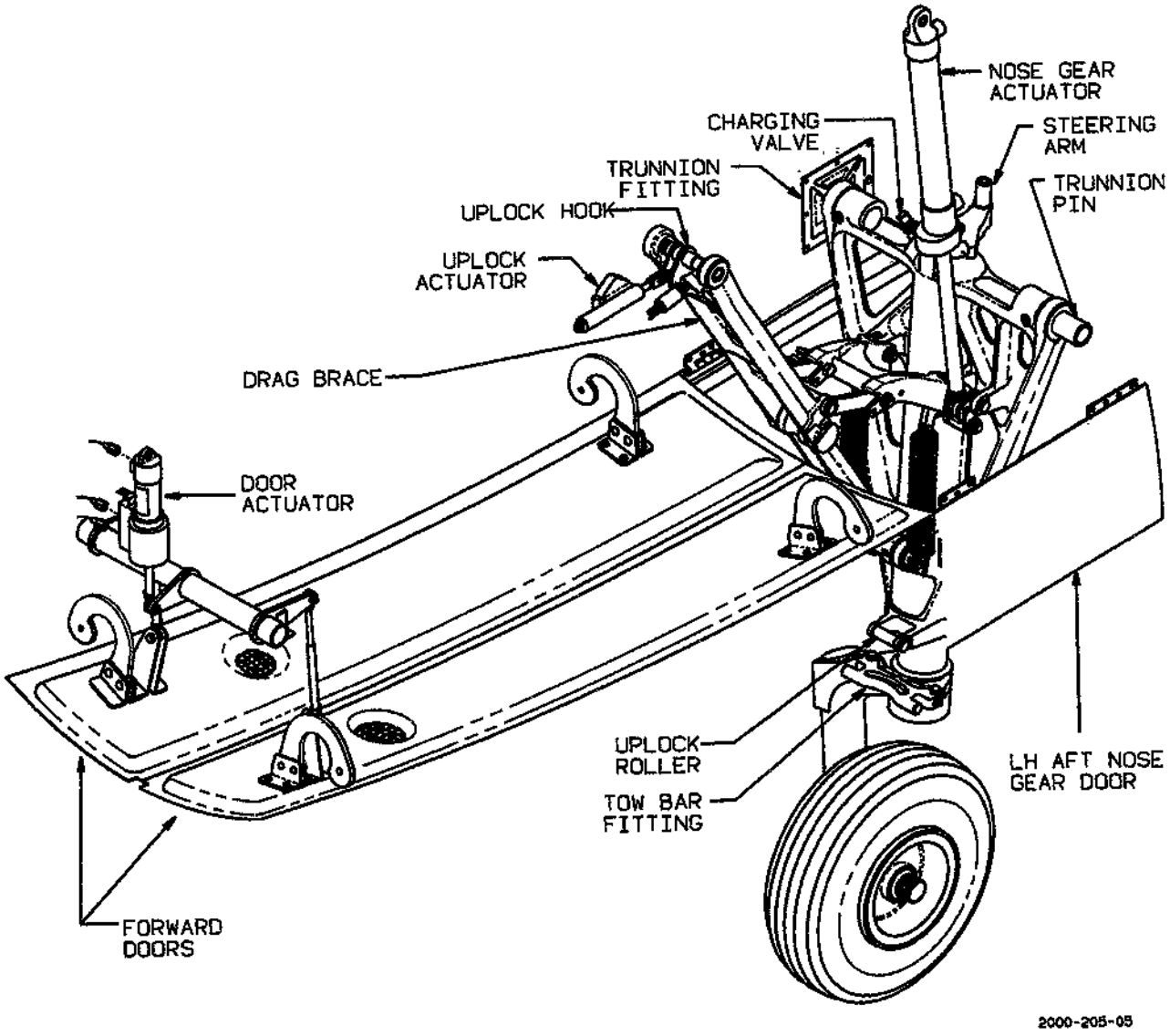
NOSE GEAR AND DOORS - MAINTENANCE PRACTICES (Effectivity: All)

WARNING

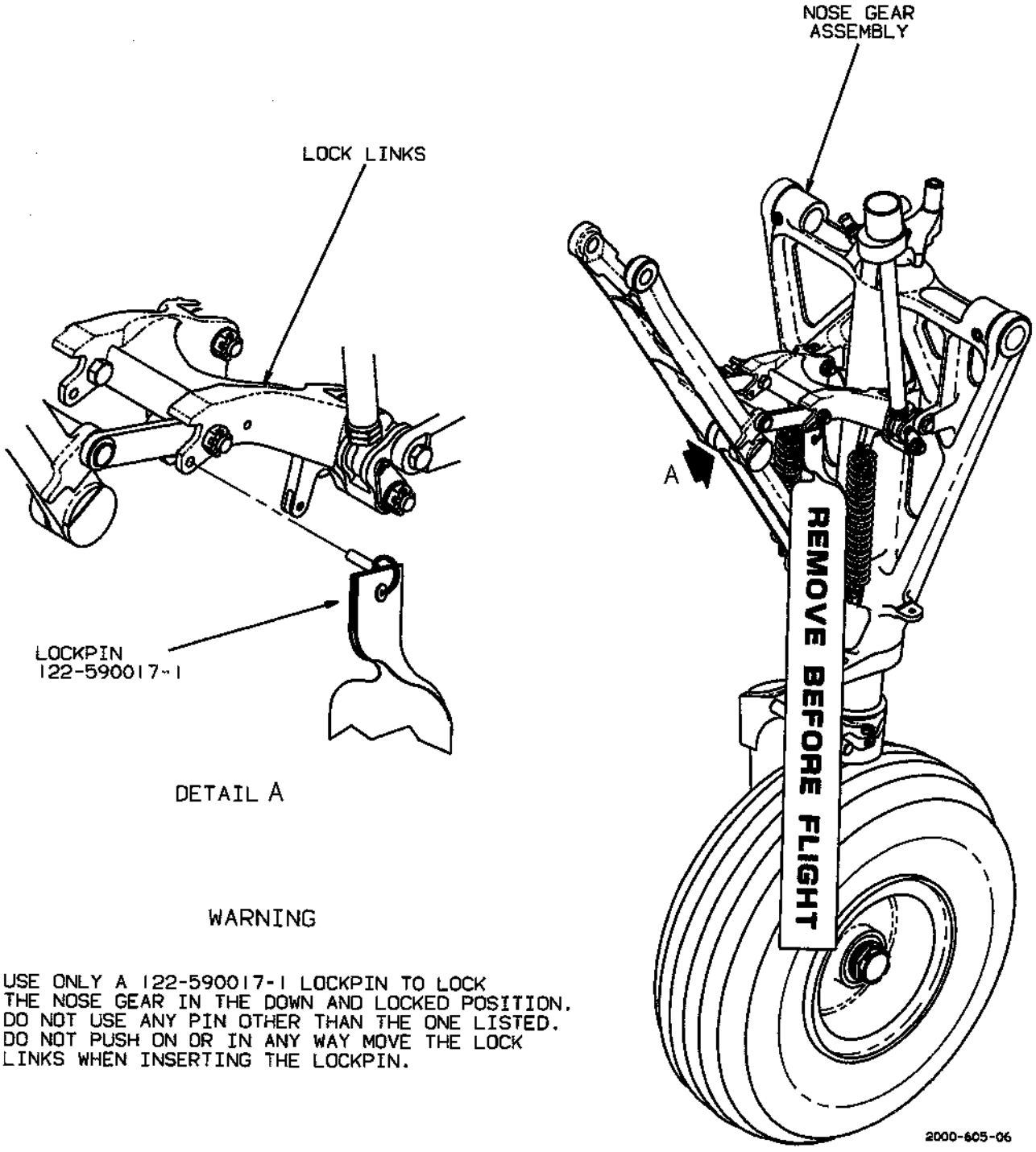
When performing maintenance on a hydraulically operated landing gear system, be aware that any movement of a hydraulic actuator cylinder can displace hydraulic fluid and cause unanticipated movement of other actuator cylinders in the landing gear retraction system. Servicing of the landing gear hydraulic accumulator can also result in unanticipated movement of an actuator. Either action can result in an unsafe, unlocked landing gear system. Therefore, place the airplane on jacks prior to performing any inspection or maintenance. Cycle the landing gear and ensure that all three landing gears are down and locked prior to removing the aircraft from jacks.

NOSE GEAR SHOCK ABSORBER SERVICING (Effectivity: All)

To check the fluid level and fill the nose gear strut, refer to Chapter 12-10-00.



Nose Landing Gear and Doors (Effectivity: All)
Figure 1



Nose Landing Gear Lockpin Installation (Effectivity: All)
Figure 2

LUBRICATION (Effectivity: All)

Lubricate the nose gear wheel bearings as detailed in the Lubrication Schedule in Chapter 12-20-00.

NOSE LANDING GEAR LOCKPIN INSTALLATION (Effectivity: All) (Figure 2)

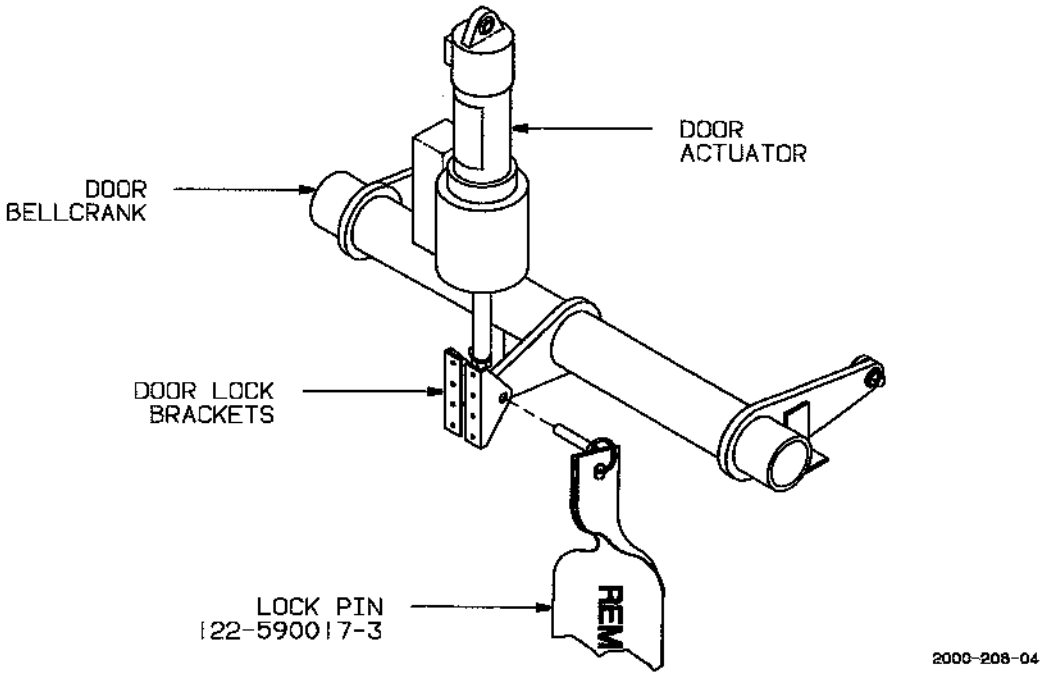
Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. Install the lockpin (2, Chart 1, 32-00-00) as detailed below.

WARNING

Use only a 122-590017-1 lockpin (2, Chart 1, 32-00-00) to lock the nose gear in the down and locked position, DO NOT use any pin other than the one listed. DO NOT push on, or in any way move the lock links when inserting the lockpin.

a. Ensure that the landing gear is in the down and locked position.

b. Install a 122-590017-1 lockpin (2, Chart 1, 32-00-00) into the lock links of the nose gear. Refer to Figure 2 for the lockpin hole location.



Nose Landing Gear Door Lockpin Installation (Effectivity: All)
Figure 3

NOSE LANDING GEAR DOOR LOCKPIN INSTALLATION (Effectivity: All) (Figure 3)

When any work is being performed in the nose gear wheel well, the forward nose landing gear doors must be pinned in the open position.

a. Ensure that the landing gear is in the down and locked position.

b. Pump the alternate-extension hand pump in the cockpit until the forward nose landing gear doors are fully open.

c. Install a 122-590017-3 lockpin (3, Chart 1, 32-00-00) into the nose gear door bellcrank and door-lock bracket. Refer to Figure 3 for lockpin hole location.

UPLOCK ROLLER REMOVAL (Effectivity: All) (Figure 4)

a. Remove the cotter pin, nut (4) and washer (2) from the bolt (1).

b. Remove the bolt (1) and the remaining washers (2), then remove the uplock roller (3) from the nose gear brace.

UPLOCK ROLLER INSTALLATION (Effectivity: All) (Figure 4)

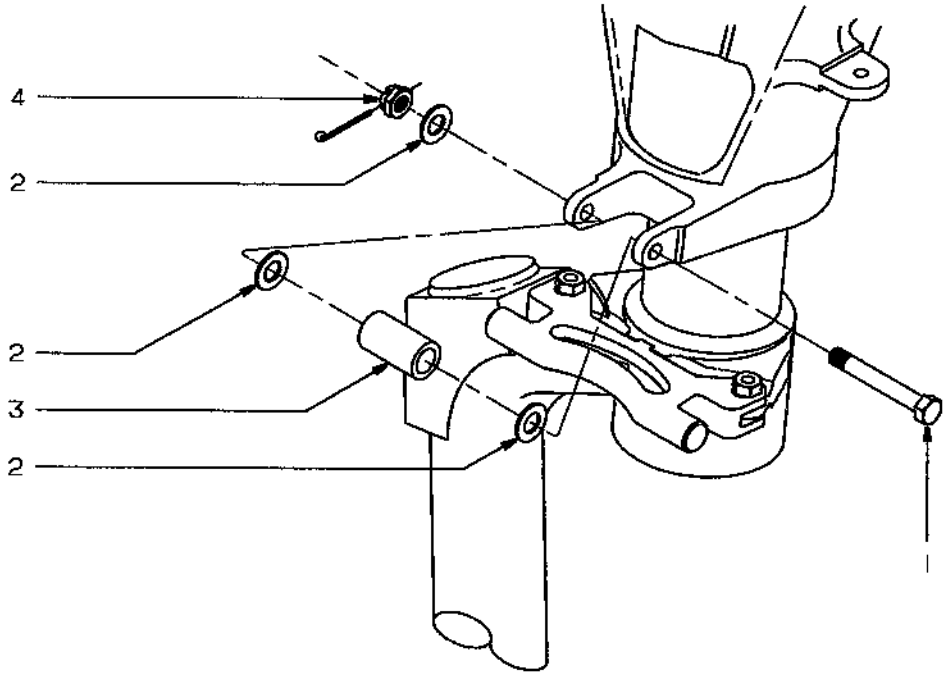
a. Pack the inside diameter of the uplock roller (3) and bearings and coat the outside diameter of bolt (1) with grease (2, Chart 2, 32-00-00).

b. Position the uplock roller (3) between the lugs on the nose gear brace. Install washers (2) on each side of the roller (3) and install the attaching bolt (1).

c. Install the washer (2) and nut (4) on the bolt (1). Tighten the nut only enough to remove any noticeable end play and install the cotter pin.

TOW BAR FITTING REMOVAL (Effectivity: All) (Figure 5)

a. Remove the cotter pins, nuts (4) and washers (3) from the fuses (2).



Uplock Roller Installation (Effectivity: All)
Figure 4

b. Remove the fuses (2) attaching the tow bar fitting (1) to the nose gear piston assembly.

TOW BAR FITTING INSTALLATION
(Effectivity: All)
(Figure 5)

a. Position the tow bar fitting (1) on the nose gear brace.

b. Install the fuses (2) attaching the tow fitting to the nose gear piston assembly.

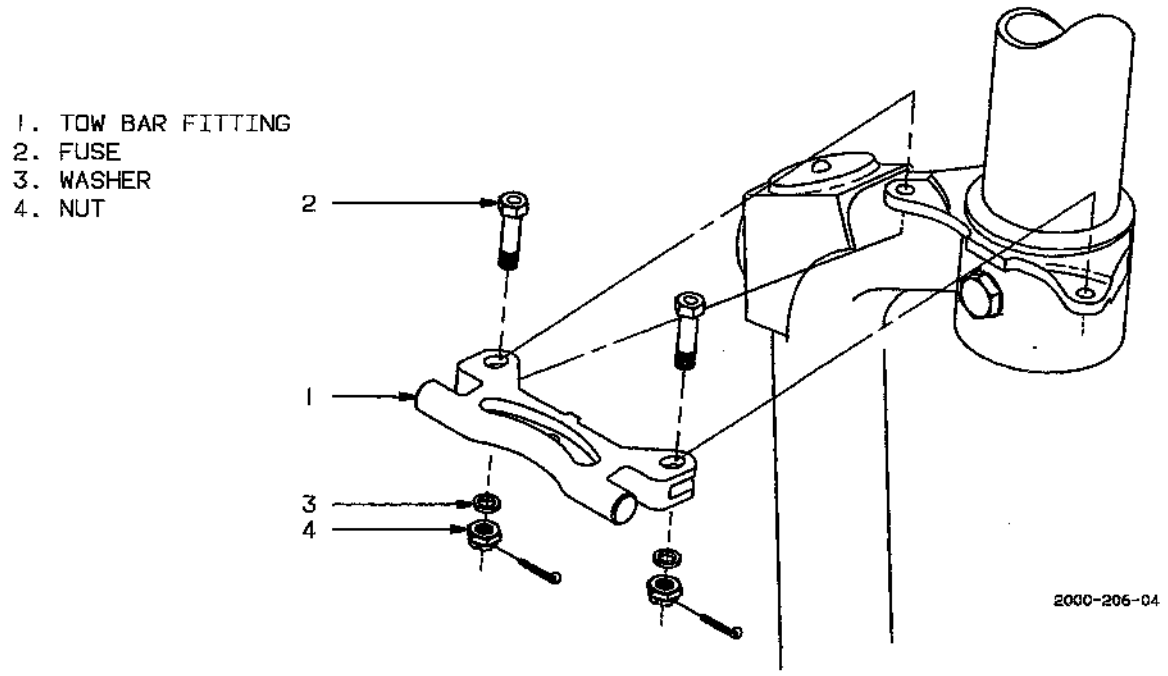
c. Install the washers (3) and nuts (4) on the fuses. Tighten the nuts only enough to remove any end play from the fuses and install the cotter pins.

CHARGING VALVE REMOVAL
(Effectivity: All)
(Figure 6)

WARNING

Never remove the charging valve from the landing gear assembly until all of the pressure is released from the strut.

a. Loosen the valve cap (1) about one turn to release the trapped air, then remove the cap from the charging valve.



Tow Bar Fitting Installation (Effectivity: All)
Figure 5

b. Release the air pressure in the strut by slowly turning the swivel nut (2) counterclockwise. Do not allow the valve body (3) to turn while loosening the swivel nut. Two and one-fourth turns will open the valve completely. Allow the strut to fully compress.

WARNING

Stand to one side of the valve while releasing the air pressure.

c. Cut the safety wire and remove the valve body (3) from the nose gear assembly.

CHARGING VALVE INSTALLATION (Effectivity: All) (Figure 6)

a. Ensure that the charging valve is clean.

b. Lubricate a new O-ring (4) with hydraulic fluid (1, Chart 2, 32-00-00). Install the O-ring on the charging valve, use care to avoid damaging the O-ring during installation.

c. Screw the valve body (3) into the nose gear assembly until the hexagonal portion of the valve body contacts the boss. Tighten the valve body to a torque of 100 to 110 inch-pounds and safety wire.

d. Tighten the swivel nut (2) to a torque of 50 to 70 inch-pounds

e. Install the valve cap (1) finger tight.

f. Refer to Chapter 12-10-00 and service the nose gear shock absorber assembly.

NOSE LANDING GEAR REMOVAL (Effectivity: All) (Figure 7)

a. Remove all electrical power from the airplane.

b. Using the alternate-extension hand pump in the cockpit, pump the forward nose gear doors open.

c. Install a 122-590017-3 lockpin (3, Chart 1, 32-00-00) into the nose gear door bellcrank and door lock bracket.

d. Place the airplane on jacks as outlined in Chapter 7-10-00.

e. After the airplane is on jacks, remove the nose gear lockpin (2, Chart 1, 32-00-00) from the lock link.

f. Refer to Chapter 52-80-00 and disconnect the nose gear doors and secure in the fully open position.

g. Remove the cotter pins, washers (3), bushings (2) and pins (1) and remove the downlock springs (4) from the nose gear.

h. Remove the cotter pin, nut (13), washers (12) and lock plate (14) from the bolt (17).

i. To disconnect the actuator from the nose gear insert a .138-32 threaded bolt into the end of the actuator pivot pin (15) and withdraw the pin from the actuator bracket (10).

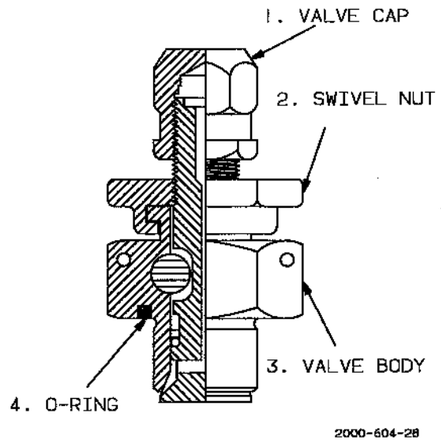
j. Remove the bolt (17) and spacer (16) from the aft lock link (18) and actuator bracket (10).

k. Remove the cotter pins and remove the nuts (5 and 21), washers (6, 7, 8 and 20) and bolts (11 and 19) that connect the aft lock link (18) to the nose gear.

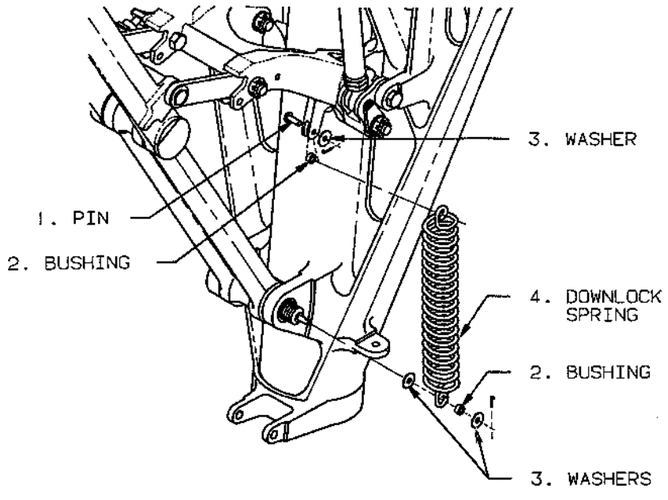
l. Remove the cotter pin and remove the nut (24) and washer (23) from the drag brace pin (22).

m. Remove the drag brace pin (22) to disconnect the drag brace from the nose gear.

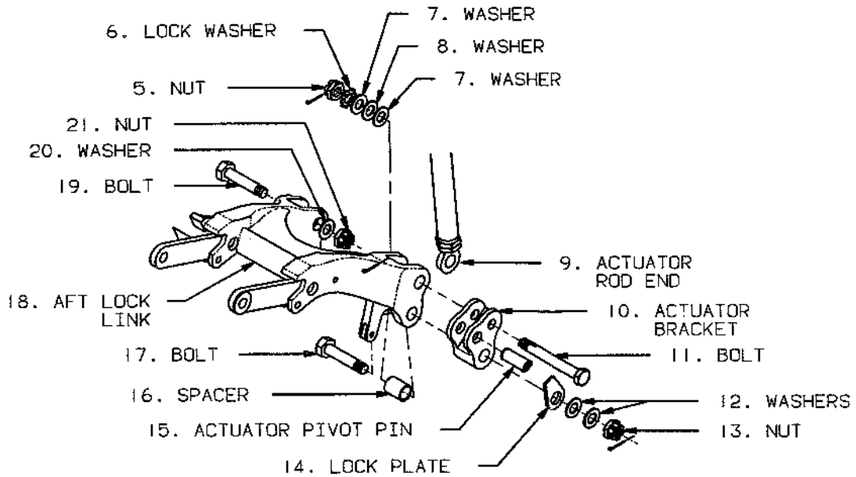
n. When the trunnion pins (31) are removed from the trunnion bearings, the nose gear will be released from



Charging Valve Installation (Effectivity: All)
Figure 6



DETAIL A



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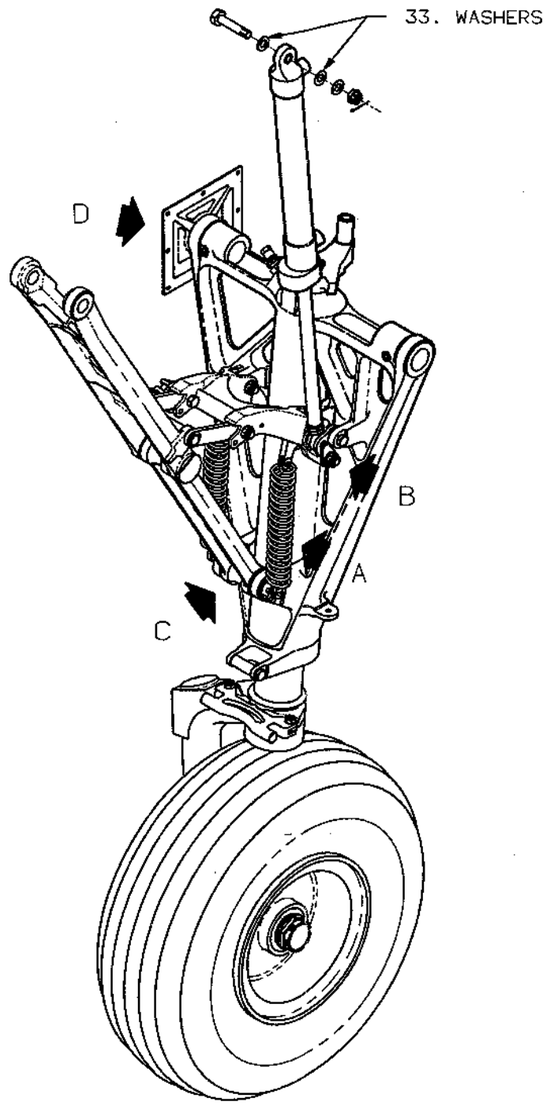

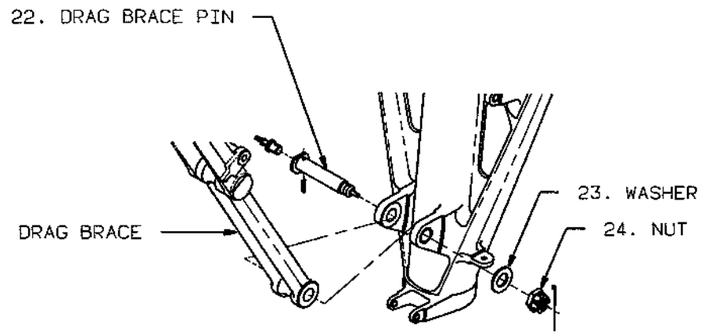

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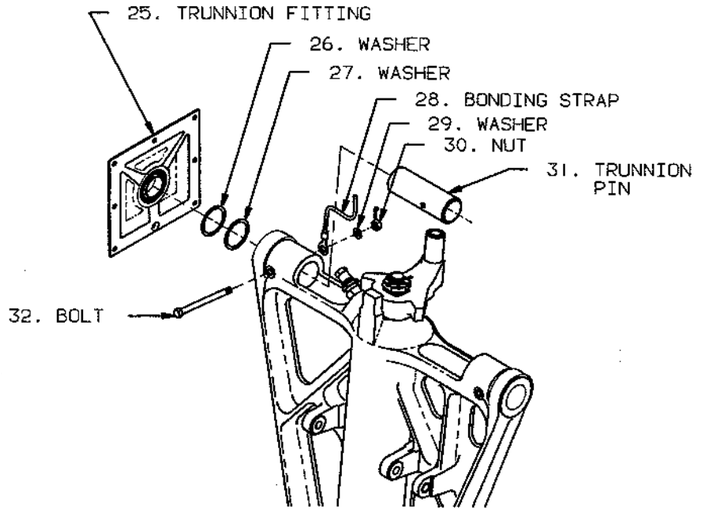


DETAIL D

2000-205-06

Nose Landing Gear Installation (Effectivity: All)
Figure 7

the airplane. Before removing the trunnion pins, place a block under the nose wheel to support the nose gear.

o. Remove the cotter pins and remove the bolts (32), washers (29) and nuts (30) that secure the trunnion pins (31) to the nose gear.

p. Move the trunnion pins (31) inboard (toward the nose gear brace) until they are free of the bearings in the trunnion fittings. Lower the nose gear away from the airplane.

NOTE

When the trunnion pins are removed from the trunnion bearings the washers (26 and 27) installed between the nose gear and the trunnion bearings will be released. Note the positions of the washers to facilitate alignment of the nose gear to the drag brace when the nose gear is installed.

NOSE LANDING GEAR INSTALLATION
(Effectivity: All)
(Figure 7)

a. Install the trunnion pins (31) into the nose gear brace.

b. Position the nose gear in the wheel well and insert the trunnion pins (31) into the trunnion bearings with the washers (26 and 27) in their original positions between the nose gear and trunnion bearings. Align the holes in the trunnion pins with those in the nose gear brace and install the bolts (32), washers (29) and nuts (30) securing the trunnion pins to the nose gear brace; install the bonding strap (28) under the washer (29) on the right side. Tighten the nuts only enough to remove any perceptible end play and install the cotter pins.

c. Check the alignment of the drag brace with the attachment lugs on the nose gear brace; there shall be no side loads on the drag brace when the landing gear is in the extended or retracted position. Correct any misalignment by adding or shifting the washers (26 and 27) from side to side of the nose gear brace and the trunnion bearing. A minimum of one washer must be installed between the nose gear brace and each trunnion bearing.

d. After the drag brace and nose gear are properly aligned, connect the drag brace to the nose gear with the drag brace pin (22), washer (23), nut (24) and the cotter pin.

e. Connect the aft lock link (18) and actuator bracket (10) to the nose gear with the bolts (11 and 19), washers (6, 7, 8 and 20) and nuts (5 and 21). Install the washers (6, 7, 8) on the bolt (11) in the same positions as when they were removed. Tighten the nuts (5 and 21) only enough to remove any perceptible end play and install the cotter pins.

f. Install the bolt (17) and spacer (16) to the aft lock link (18) and actuator bracket (10).

g. To eliminate any side load on the actuator, check the alignment of the actuator rod end (9) with the actuator bracket (10). The actuator rod end must slip into the actuator bracket without causing any side loads on the actuator piston rod. If any misalignment is found, shift the washers (33) on either side of the actuator upper attach point to bring the actuator into alignment.

h. Position the actuator rod end (9) in the actuator bracket (10) and install the pivot pin (15). Install the lock plate (14), washers (12), and nut (13) on the bolt (17) and install the cotter pin.

NOTE

Position the pivot pin so that it is properly retained by the lock plate.

i. Install the downlock springs (4) to the nose gear with the pin (1), bushings (2), washers (3) and cotter pins.

NOTE

Use the washers (3) as required on the pin (1) and on both sides of the lower bushing (2) to provide for a snug fit of the cotter pins.

j. Refer to Chapter 52-80-00 and connect the landing gear doors.

k. Check the landing gear and door rigging as outlined in Chapter 32-30-00, and adjust the downlock switch and the uplock switch as outlined in Chapter 32-60-00.

l. Ensure the landing gear is in the down and locked position and install the lockpin (2, Chart 1, 32-00-00) in the nose gear as detailed in this chapter.

m. Remove the airplane from the jacks.

n. Remove the lockpin from the forward nose gear doors and push the forward nose gear doors closed.

o. Restore electrical power to the airplane.

NOSE LANDING GEAR DRAG BRACE REMOVAL (Effectivity: All) (Figure 8)

a. Remove all electrical power from the airplane.

b. Using the alternate-extension hand pump in the cockpit, pump the forward nose gear doors open.

c. Install a 122-590017-3 lockpin in the nose gear door bellcrank and door lock bracket.

d. Place the airplane on jacks as outlined in Chapter 7-10-00.

e. After the airplane is on jacks, remove the lockpin (2, Chart 1, 32-00-00) from the lock link.

f. Refer to Chapter 52-80-00 and disconnect the nose gear doors and secure in the full open position.

g. Remove the cotter pins, washers (3), bushings (2) and pins (1) and remove the downlock springs (4) from the nose gear.

h. Remove the cotter pin, nut (13), washers (12) and lock plate (14) from the bolt (17).

i. To disconnect the actuator from the nose gear insert a .138-32 threaded bolt into the end of the actuator pivot pin (15) and withdraw the pin from the actuator bracket (10) and actuator rod end (9).

j. Remove the bolt (17) and spacer (16) from the aft lock link (18) and actuator bracket (10).

k. Remove the cotter pins and remove the nuts (5 and 21), washers (6, 7, 8 and 20) and bolts (11 and 19) that connect the aft lock link (18) to the nose gear.

l. Remove the cotter pin and remove the nut (24) and washer (23) from the drag brace pin (22).

m. Remove the drag brace pin (22) to disconnect the drag brace from the nose gear.

n. Remove the cotter pin, nut (30), washers (27), bushings (28) and bolt (26) connecting the uplock actuator (25) to the uplock hook (29).

o. Remove the cotter pins and remove the pins (32) securing the upper drag brace pin (33) to the drag brace fittings (31).

p. Disconnect the spring (36) from the uplock hook (29) and the drag brace.

q. Withdraw the upper drag brace pin (33) from the drag brace fittings and the drag brace. Retain the washers (35), spring (36), bushing (34) and uplock hook (29) as the upper drag brace pin (33) is removed.

r. Remove the drag brace from the airplane.

NOSE LANDING GEAR DRAG BRACE INSTALLATION (Effectivity: All) (Figure 8)

a. Install the upper drag brace pin (33) through the drag brace and the drag brace fittings (31). Install the washers (35), spring (36), bushing (34) and uplock hook (29) as the upper drag brace pin (33) is installed.

NOTE

Position the spring (36), bushing (34), uplock hook (29) and washers (35) in the same positions as when removed. Use washers (35) as required on each side of the uplock hook (29) to center the uplock hook and maintain a maximum of .03 inch end play.

b. Align the holes in the upper drag brace pin (33) with the holes in the drag brace fittings (31) and install the pins (32) and cotter pins securing the upper drag brace pin (33) to the drag brace fittings (31).

c. Connect the uplock actuator (25) to the uplock hook (29) with the bolt (26), bushings (28), washers (27), nut (30) and cotter pin.

d. Check the alignment of the drag brace with the attachment lugs on the nose gear brace; there shall be no side loads on the drag brace when the landing gear is in the extended or retracted position. Correct any misalignment by adding or shifting the washers (26 and 27, Figure 7) from side to side of the nose gear brace and the trunnion bearing. A minimum of one washer must be installed between the nose gear brace and each trunnion bearing.

e. After the drag brace and nose gear are properly aligned, connect the drag brace to the nose gear with the drag brace pin (22), washer (23), nut (24) and cotter pin.

f. Connect the aft lock link (18) and actuator bracket (10) to the nose gear with the bolts (11 and 19), washers (6, 7, 8 and 20) and nuts (5 and 21). Install the washers (6, 7, and 8) on the bolt (19) in the same positions as when they were removed. Tighten the nuts (5 and 21) only enough to remove any perceptible end play and install the cotter pins.

g. Install the bolt (17) and spacer (16) to the aft lock link (18) and actuator bracket (10).

h. To eliminate any side load on the actuator, check the alignment of the actuator rod end (9) with the actuator bracket (10). The actuator rod end must slip into

the actuator bracket without causing any side loads on the actuator piston rod. If any misalignment is found, shift the washers (37) on either side of the actuator upper attach point to bring the actuator into alignment.

i. Position the actuator rod end (9) in the actuator bracket (10) and install the pivot pin (15). Install the lock plate (14), washers (12), and nut (13) on the bolt (17) and install the cotter pin.

NOTE

Position the pivot pin so that it is properly retained by the lock plate.

j. Install the downlock springs (4) to the nose gear with the pins (1), bushings (2), washers (3) and cotter pins.

NOTE

Use the washers (3) as required on the pins (1) and on both sides of the lower bushings (2) to provide for a snug fit of the cotter pins.

k. Refer to Chapter 52-80-00 and connect the landing gear doors.

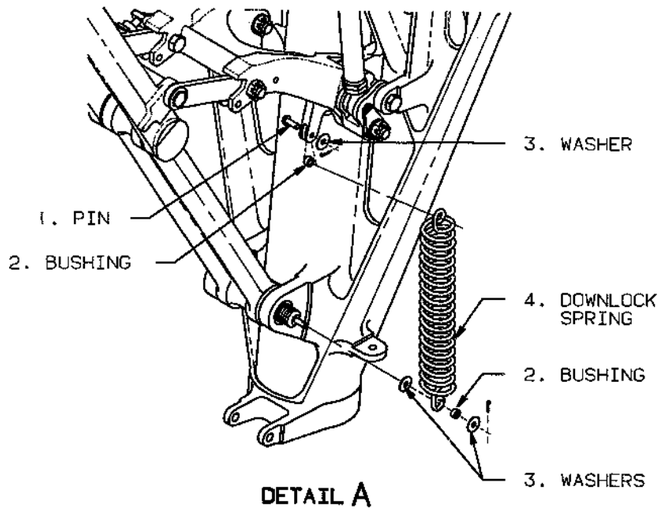
l. Check the landing gear and door rigging as outlined in Chapter 32-30-00, and adjust the downlock switch and uplock switch as outlined in Chapter 32-60-00.

m. Ensure the landing gear is in the down and locked position and install the lockpin (2, Chart 1, 32-00-00) in the nose gear as detailed in this chapter.

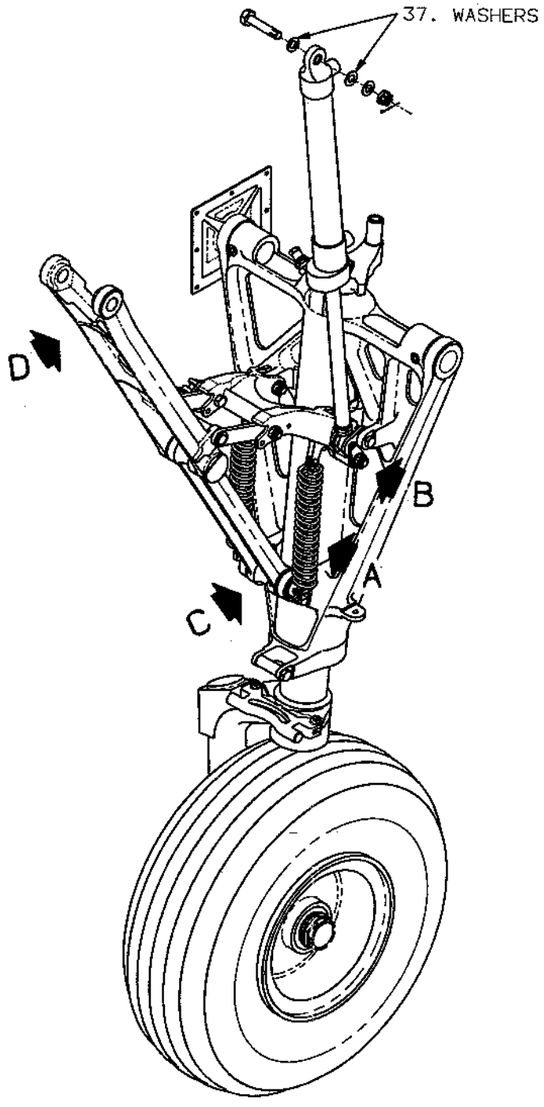
n. Remove the airplane from the jacks.

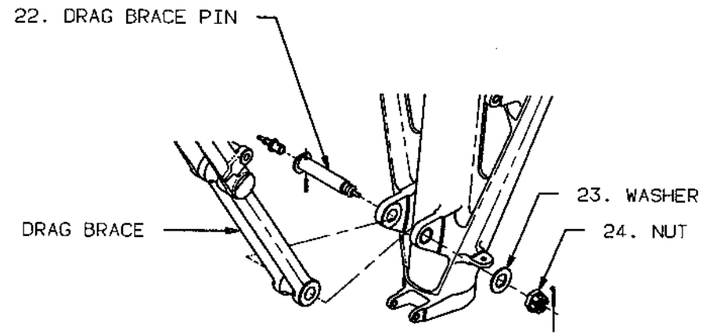
o. Push the forward nose gear doors closed.

p. Restore the electrical power to the airplane.



DETAIL A





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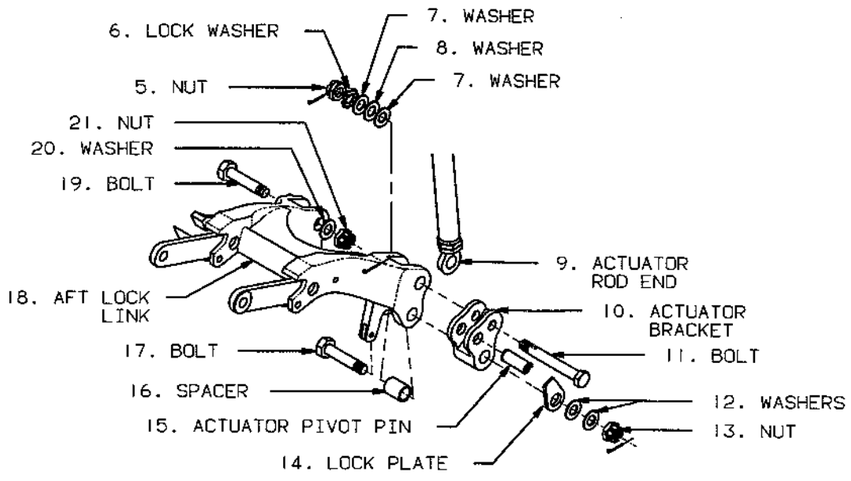
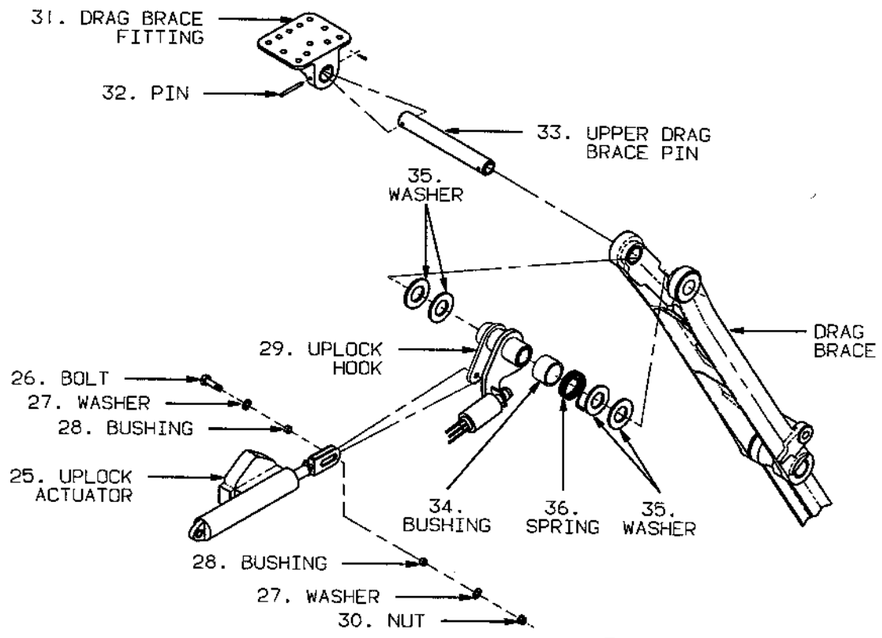


DETAIL B



DETAIL D

C94NC32B0191

Nose Landing Gear Drag Brace Installation
(Effectivity: All)
Figure 8

EXTENSION AND RETRACTION - DESCRIPTION AND OPERATION (EFFECTIVITY: ALL)

WARNING

Anytime the airplane is on the ground (whether on jacks or on wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock-pin installation procedures.

This system consists of two (2) independent gear extension systems. The primary system is powered by an electric motor that provides hydraulic pressure to retract and extend the landing gears and nose gear forward doors. The alternate extension system utilizes a hand pump to extend the gear in the event that the primary system is disabled in any way.

The landing gear hydraulic retract system consists of a power pack assembly, one hydraulic retract actuator and uplock actuator at each gear, a nose gear door sequence valve, and a locking actuator for the nose gear doors.

The power pack assembly includes an electric motor and pump, dual reservoirs (one for the primary system and one for the alternate extension system), two solenoid selector valves (one for the landing gears and one for the nose gear doors), a system pressure relief valve, a gear-down pressure relief valve, filters and a low fluid level sensor.

PRIMARY SYSTEM OPERATION (EFFECTIVITY: ALL)

For primary system operation, the motor pump provides fluid under pressure using fluid from the primary reservoir. This fluid is directed to and from the nose gear door actuator by the solenoid selector valve to open the nose gear forward doors prior to gear extension or retraction and, subsequently, close the doors after the appropriate gear cycle. After the nose gear forward doors are open, fluid is directed to and from the gear retract actuators and uplock actuators by means of the landing-gear-solenoid selector valve as required for the desired gear cycle. The application of power to the pump relay and the sequencing of the solenoid selector valves to the landing gear and nose gear door are controlled by an electronic logic circuit. This circuit functions on the basis of inputs from the landing gear control handle, the squat switches, and the switches sensing the position of the landing gear and nose gear door in the fully retracted and extended modes.

To secure the landing gear in the retracted position, a pivoting spring-loaded hook is provided at each landing gear. A separate hydraulic actuator is also provided at each gear to retract the uplock hook for gear extension. Proper sequencing, which ensures that the uplock hook is fully retracted prior to gear extension, is provided by the uplock actuator by means of a mechanical check valve that senses the fully retracted position before directing fluid to the landing gear actuator for extension.

ALTERNATE EXTENSION SYSTEM OPERATION (EFFECTIVITY: ALL)

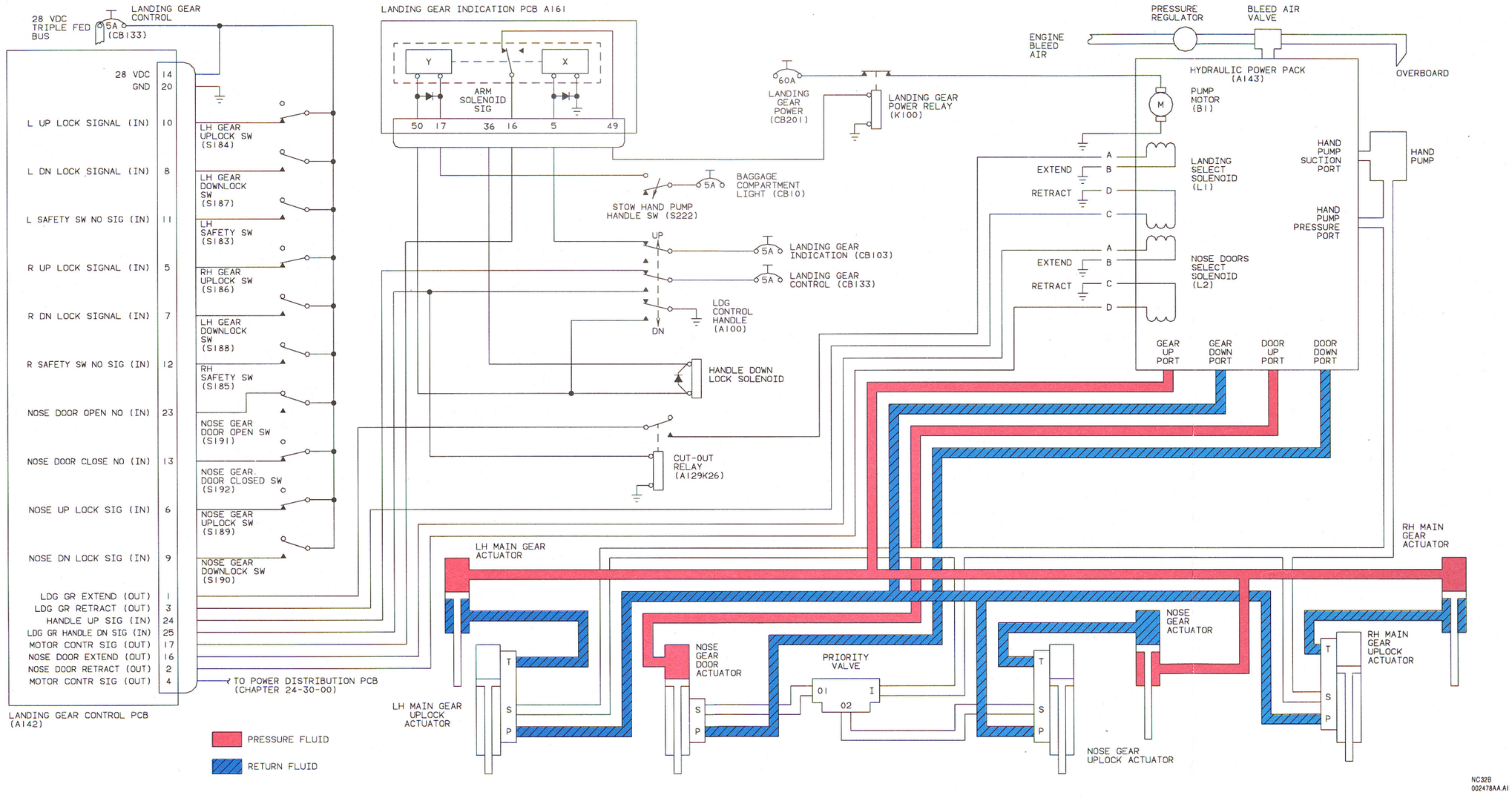
The alternate extension system is used to manually lower the gear to the down-and-locked position whenever the primary system is inoperative. The system consists of a secondary reservoir in the power pack, a hand pump, the hand pump stow switch (S222), a priority valve, and the associated hydraulic plumbing. The uplock actuators, landing gear retract actuators, and the indication system are common to both systems. The primary and alternate extend

systems are separated by a shuttle valve at the uplock actuators. The priority valve ensures that the nose gear doors open before the gear extends. The landing gear is extended by pumping the hand pump and monitoring the position of the gear with the indicator lights and EICAS display messages.

LANDING GEAR CONTROL PCB
(EFFECTIVITY: ALL)
(FIGURE 1)

The control PCB contains the electronic-logic circuitry which provides the sequencing of the landing gear and nose gear doors solenoid selector valves and power to the pump motor relay.

The PCB receives inputs from switches sensing landing gear and nose gear door position in the full-retracted and extended modes, the landing gear control handle and squat-switch positions. The PCB is located in the pilot's aft console. For removal and installation procedures refer to Chapter 39-60-00.



Landing Gear Control System Schematic
(Retract Mode)
(Effectivity: All) (Sheet 1 of 3)
Figure 1

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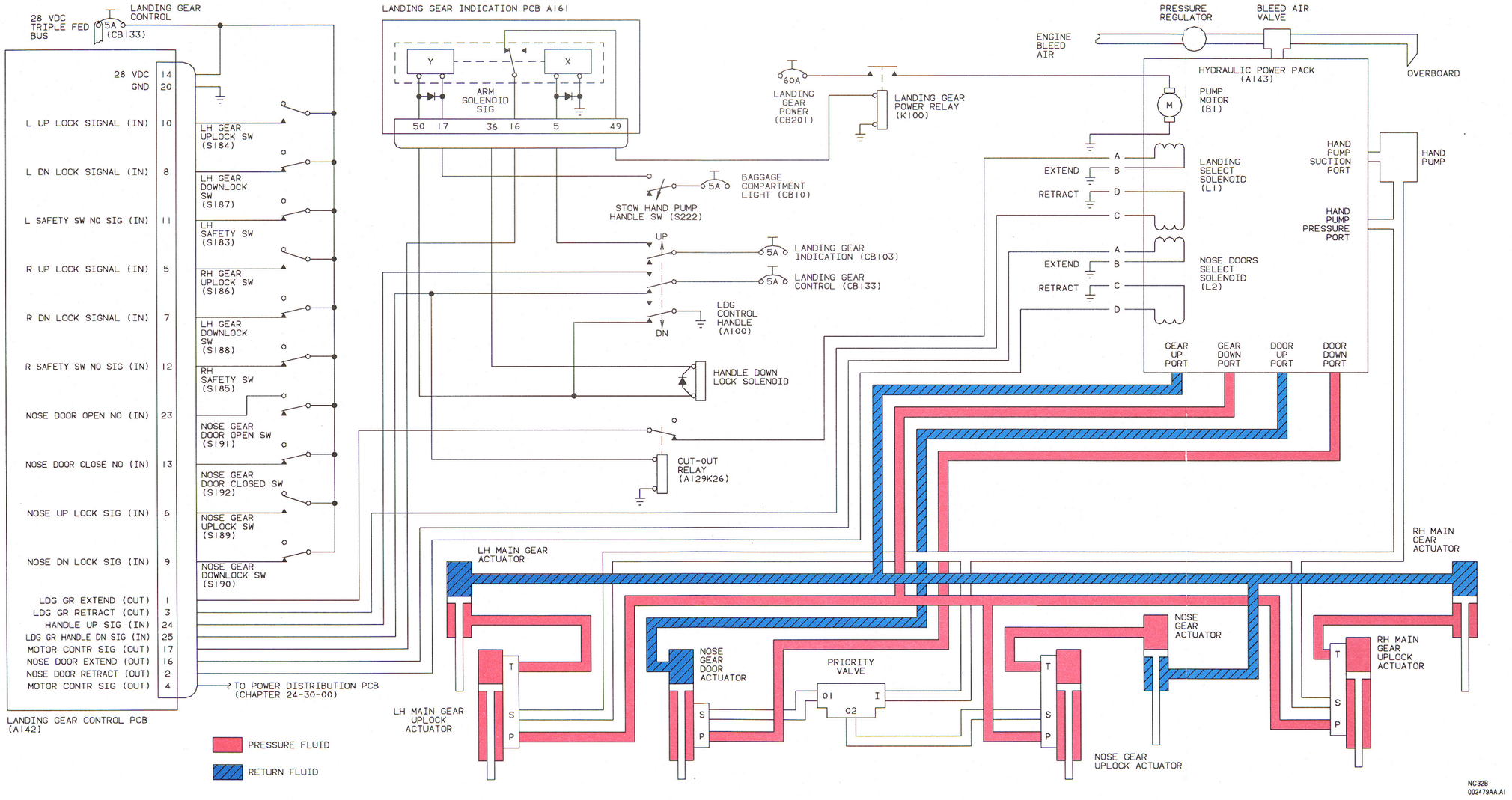

Landing Gear Control System Schematic (Extend Mode) (Effectivity: All) (Sheet 2 of 3) Figure 1

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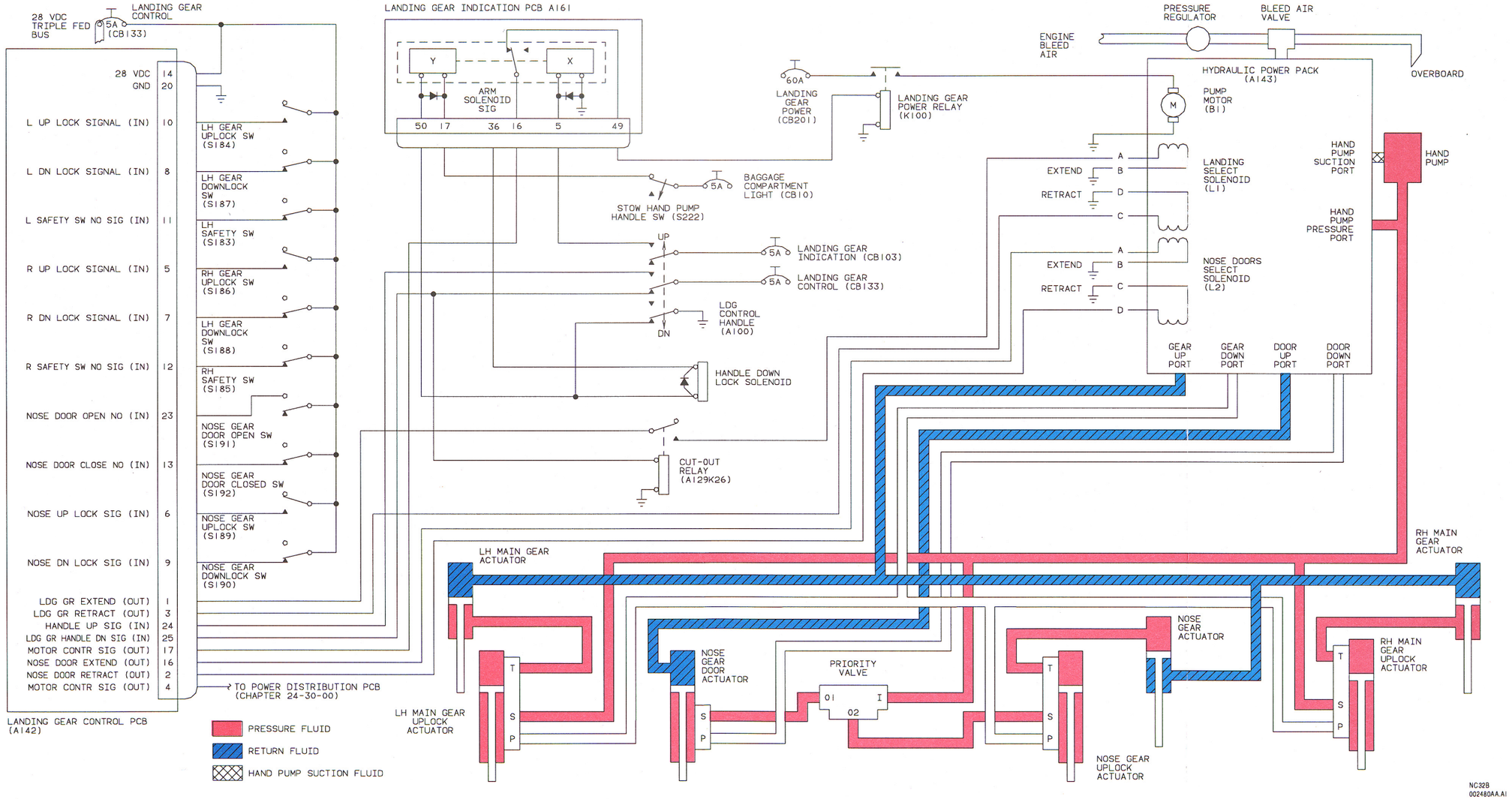

Landing Gear Control System Schematic
(Alternate Extension Mode)
(Effectivity: All) (Sheet 3 of 3)
Figure 1

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EXTENSION AND RETRACTION - TROUBLESHOOTING (EFFECTIVITY: ALL)

WARNING

To avoid damage to the airplane in case of accidental landing gear retraction, place the airplane on jacks as outlined in Chapter 7-10-00 when troubleshooting the landing gear system.

NEVER SERVICE THE HYDRAULIC SYSTEM OR DO MAINTENANCE OR RIGGING OF THE LANDING GEAR WITHOUT FIRST PLACING THE AIRPLANE ON JACKS. STAY CLEAR OF THE WHEEL WELLS, LANDING GEARS, AND GEAR DOORS WHILE THE LANDING GEAR IS IN OPERATION.

WHEN JACKING THE AIRPLANE IN AN UNSHELTERED AREA WHERE WINDS IN EXCESS OF 35 KNOTS MAY BE ENCOUNTERED, NEVER JACK MORE THAN ONE GEAR CLEAR OF THE GROUND AT A TIME.

ANYTIME THE LANDING GEAR IS ONLY PARTIALLY RETRACTED DURING MAINTENANCE, ALWAYS CYCLE THE GEAR WITH THE POWER PACK THROUGH AT LEAST ONE COMPLETE CYCLE BEFORE REMOVING THE AIRPLANE FROM THE JACKS.

Many faults can be traced to a loose or damaged electrical connection. Ensure that all connectors are clean and free of corrosion. Repair or replace any damaged or loose connector pins. Refer to the applicable wiring diagram manual for specific pin and plug locations.

Normal landing gear transition takes between 6 and 8 seconds. The timing circuits on the Landing Gear Control PCB (A142) are preset for 17 seconds. If the landing gear does not complete transitioning in the 17-second time frame, the outputs from the PCB are disabled. The 17-second time frame begins when the landing gear handle is placed in the up or down position.

CAUTION

Printed circuit boards are electrostatic discharge sensitive (ESD) components and can be easily damaged if not handled properly. Refer to Chapter 20-00-02 for information regarding the handling of ESD sensitive equipment.

CHART 1
TROUBLESHOOTING
LANDING GEAR DOES NOT EXTEND
(EFFECTIVITY: ALL)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Pump Motor does not operate.	a. Damaged Pump Motor (B1).	a. Replace Power Pack.
	b. Hand Pump Handle not stowed.	b. Stow Pump Handle.
	c. Pump Handle Switch failed.	c. Replace Pump Handle Switch (S222).
2. No voltage to the Pump Motor	a. Landing Gear Power Relay (K100) failed.	a. Replace Landing Gear Power Relay (K100).
	b. Landing Gear Indication PCB (A161) failed.	b. Replace Landing Gear Indication PCB (A161).
	c. Landing Gear Control Handle (A100) does not switch power to the Landing Gear Indication PCB (A161).	c. Replace the Landing Gear Control Handle (A100).
	d. No input signal from Pin 17 of Landing Gear Control PCB (A142).	d. Check position of the uplock, downlock, and door switches, then replace Landing Gear Control PCB (A142).
	e. Timer circuit in Landing Gear Control PCB (A142) fails.	e. Replace Landing Gear Control PCB (A142).
2. Nose Door Select Solenoid Fails to operate.	a. Damaged Select Solenoid Valve.	a. Replace Power Pack (A143).
	b. Safety switches fail to open and signal PCB (A142).	b. Adjust or replace safety switches.
	c. Uplock switches fail to make and signal PCB (A142).	c. Adjust or replace uplock switches.
	d. No output from pin 16 of the Landing Gear Control PCB (A142).	d. Replace the Landing Gear Control PCB (A142).
3. No hydraulic pressure at the Door Down Port.	a. Select valve fails in the extend or neutral position.	a. Replace the Power Pack (A143).
4. Nose Gear Door actuator fails to open Nose Doors.	a. Lack of hydraulic pressure at actuator.	a. Replace actuator hydraulic lines.
	b. Failed or leaking actuator.	b. Replace nose door actuator.

CHART 2
TROUBLESHOOTING
NOSE DOORS OPEN, BUT GEAR DOES NOT EXTEND (EFFECTIVITY: ALL)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Landing Gear Select Solenoid does not receive power.	a. Nose Gear Door Open Switch (S191) does not make.	a. Adjust or replace nose door open switch (S191).
	b. Landing Gear Control Handle (A100) does not power Cutout Relay (A129K26).	b. Replace Landing Gear Control Handle (A100).
	c. Cutout relay (A129K26) does not make.	c. Replace Cutout relay (A129K26).
2. No Hydraulic flow or pressure at the Gear Down Port.	a. Select Valve fails in the retract or neutral position.	a. Replace the Power Pack (A143).
3. Gear Uplock Actuators fail to release gear.	a. Lack of hydraulic pressure at the uplock actuators.	a. Inspect hydraulic lines, tighten fittings, and replace lines.
	b. Shuttle Valve does not open Primary Port.	b. Replace Uplock Actuator.

CHART 3
TROUBLESHOOTING
LANDING GEAR DOES NOT RETRACT (EFFECTIVITY: ALL)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Pump Motor does not operate.	a. Damaged pump motor (B1).	a. Replace Power Pack (A143).
	b. Hand Pump Handle not stowed.	b. Stow Hand Pump Handle.
	c. Hand Pump Handle switch (S222) failed.	c. Replace Pump Handle switch (S222).
2. No voltage to the pump motor (B1).	a. Landing gear power relay (K100) failed.	a. Replace Landing Gear Power Relay (K100).
	b. No input signal from the Landing Gear Control Handle (A100) to pin 5 of the LANDING GEAR INDICATION PCB (A161)	b. Verify voltage from the LANDING GEAR INDICATION circuit breaker (CB133) and replace Landing Gear Control Handle (A100).
	c. Landing Gear Indication PCB Relay (A161K2) failed.	c. Verify input signal at pins 5 and 16, then replace Landing Gear Indication PCB (A161).
	d. No motor control signal from pin 17 of the Landing Gear Control PCB (A142)	d. Check position of the safety, uplock, downlock, and door switches, then replace the Landing Gear Control PCB (A142).
	e. Timer circuit in landing gear Control PCB (A142) failed.	e. Replace the Landing Gear Control PCB (A142).
3. Nose Door Select Solenoid fails to operate.	a. Damaged Door Select Solenoid Valve.	a. Replace Power Pack (A143).

CHART 3
TROUBLESHOOTING
LANDING GEAR DOES NOT RETRACT (EFFECTIVITY: ALL) (CONTINUED)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
	b. No voltage at the Door Select Solenoid (L2).	b. Check the position of the safety, uplock, downlock, and door switches.
	c. No voltage at the select solenoid valve (L2).	c. Verify output at Landing Gear Control PCB (A142) and replace PCB.
3. No hydraulic pressure at the Door Up Port	a. Select Valve fails in the retract or neutral position.	a. Replace the Power Pack (A143).

CHART 4
TROUBLESHOOTING
LANDING GEAR DOES NOT EXTEND WITH THE HAND PUMP (EFFECTIVITY: ALL)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Landing Gear does not extend with the hand pump.	a. Relief valve fails in the open position.	a. Replace hand pump.
	b. Priority Valve fails in nonsequenced position.	b. Replace priority valve.

CHART 5
TROUBLESHOOTING
SINGLE GEAR DOES NOT EXTEND WITH THE HAND PUMP (EFFECTIVITY: ALL)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Uplock Actuator does not release gear.	a. No hydraulic pressure at the secondary port.	a. Check hydraulic lines and fittings for leakage.
	b. Shuttle valve does not open secondary port.	b. Replace uplock actuator.

CHART 6
TROUBLESHOOTING
LANDING GEAR EXTENDS OR RETRACTS SLOWLY (EFFECTIVITY: ALL)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Low hydraulic pressure or flow.	a. System Pump does not receive proper voltage.	a. Correct input voltage to pump motor.
	b. Filter Screens for Gear Up or Gear Down, Nose Door Up or Down Ports are restricted.	b. Replace Power Pack (A143).
	c. Low fluid level in Power Pack reservoir.	c. Check Power Pack and hydraulic lines for leakage.

CHART 7
TROUBLESHOOTING
NOSE GEAR DOORS FAIL TO CLOSE (EFFECTIVITY: ALL)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Landing Gear Select Solenoid does not receive power.	a. Nose Gear Door Open Switch (S191) does not make.	a. Adjust or replace Nose Door Open Switch (S191).
	b. Nose Gear Uplock Switch (S189) does not make.	b. Adjust or replace Nose Gear Uplock Switch (S189).

EXTENSION AND RETRACTION - MAINTENANCE PRACTICES (EFFECTIVITY: ALL)

SERVICING (EFFECTIVITY: ALL)

For servicing of the landing gear system refer to Chapter 12-10-00.

HYDRAULIC SYSTEM CLEANLINESS (EFFECTIVITY: ALL)

When performing maintenance on the hydraulic system, cleanliness is essential for maintaining a functional and trouble-free system. Any foreign materials that are allowed to enter the system are damaging to the O-rings, seals, valves and moving parts within the system. Such damage can result in fluid leakage and/or failure of the system components. Observe the following guidelines when working on the hydraulic system:

a. Hydraulic storage containers should be sealed completely to prevent contamination of the hydraulic fluid (1, Chart 2, 32-00-00) during storage.

b. Use only clean hydraulic fluid when filling the system.

c. Use only clean hydraulic fluid when flushing the system or testing any of the system components.

d. When any lines or hoses are disconnected and/or any components are removed from the system, plug or cap all openings to prevent the entry of foreign material into the system.

e. When the system is open, prevent contamination of the hydraulic fluid with other oils or water.

f. Hands and tools should be free of contaminants (grease, oil, dirt, etc.) while working on the hydraulic system.

g. If the system becomes contaminated, drain and flush the system with clean hydraulic fluid.

h. To reduce the possibility of leaks in the hydraulic system, care should be taken when installing hydraulic fittings or components. Any time a fitting or component is loosened or removed, discard the O-ring and install the fitting or component with a new O-ring. To prevent damage to the O-ring, coat the O-ring and the threads of the fitting or component with hydraulic fluid before installing the O-ring; do not contaminate the O-ring after it is lubricated.

TUBING REPAIR AND REPLACEMENT (EFFECTIVITY: ALL)

For tubing repair and replacement refer to Chapter 20-00-06.

FILLING AND BLEEDING THE HYDRAULIC SYSTEM (EFFECTIVITY: ALL) (FIGURE 2)

The landing gear system should not need to be filled and bled unless the power pack is removed and any of the system lines are opened at the actuators, resulting in a considerable loss of fluid and entry of air into the system.

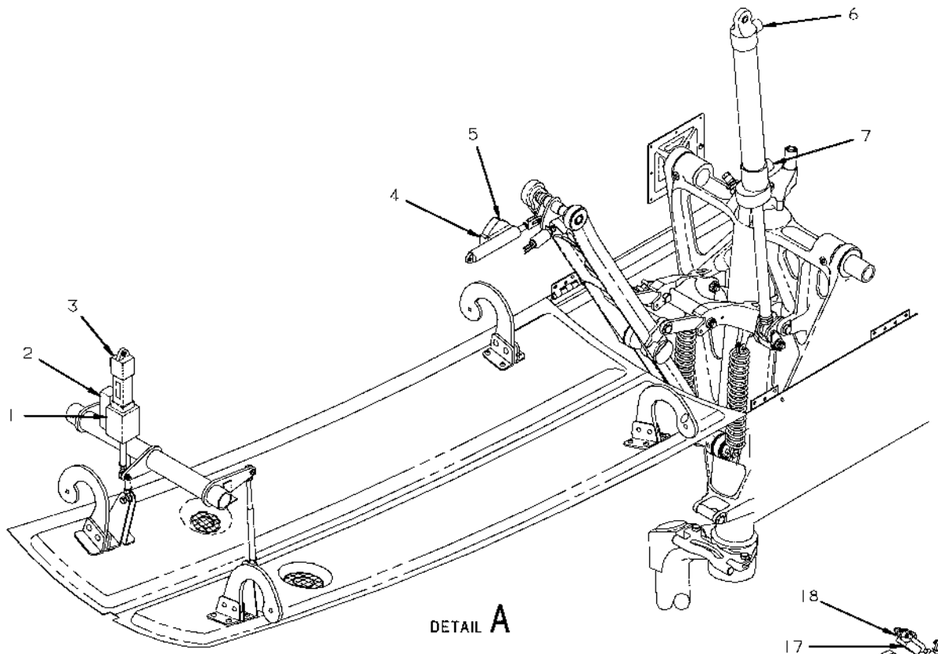
a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Refer to Chapter 24-40-00 and connect 28 vdc external power to the airplane.

c. Refer to Chapter 54-00-00 and remove the upper access panel (581AT) from the LH nacelle.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling (19) and restrictor (20) from the bleed air valve (11) when bleeding the hydraulic system.



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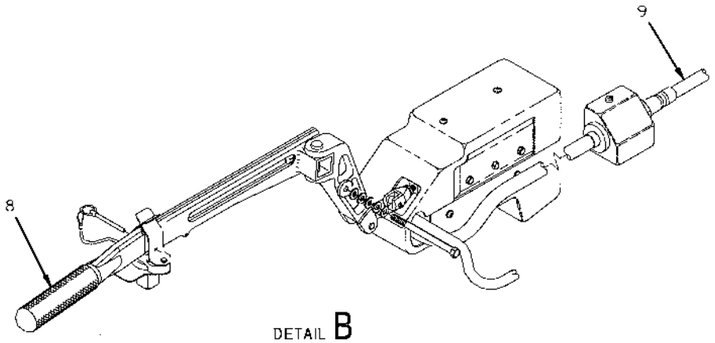

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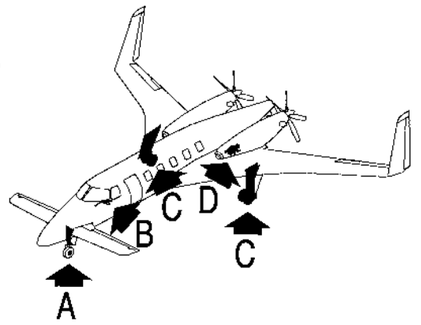

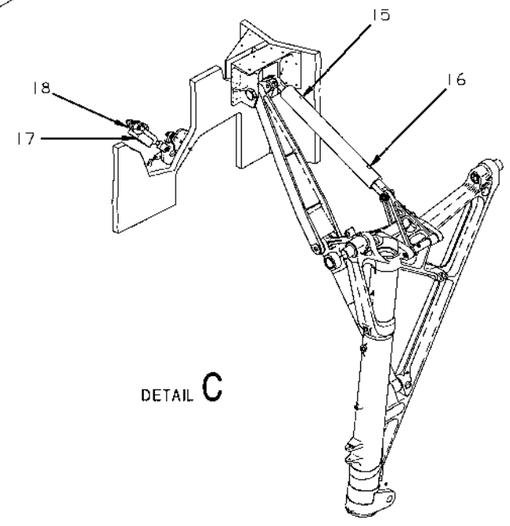

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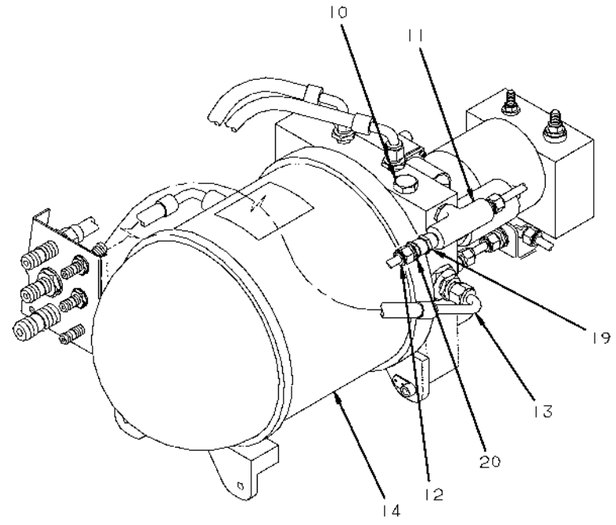

DETAIL D

1. NOSE GEAR DOOR ACTUATOR DOWN PORT "P"
2. NOSE GEAR DOOR ACTUATOR
3. NOSE GEAR DOOR ACTUATOR UP PORT
4. NOSE UPLOCK ACTUATOR GEAR DOWN PORT "P"
5. NOSE UPLOCK ACTUATOR GEAR UP PORT
6. NOSE GEAR ACTUATOR EXTEND PORT
7. NOSE GEAR ACTUATOR RETRACT PORT
8. HAND PUMP HANDLE
9. HAND PUMP SUCTION LINE
10. RESERVOIR FILL CAP
11. BLEED AIR VALVE
12. BLEED AIR INLET LINE
13. HAND PUMP PRESSURE LINE
14. HYDRAULIC POWER PACK
15. MAIN GEAR ACTUATOR RETRACT PORT
16. MAIN GEAR ACTUATOR EXTEND PORT
17. MAIN UPLOCK ACTUATOR GEAR UP PORT
18. MAIN UPLOCK ACTUATOR GEAR DOWN PORT "P"
19. COUPLING *
20. RESTRICTOR *

* ADDED WITH KIT 122-5036-1

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Filling and Bleeding the Hydraulic System
(Effectivity: All)
Figure 2

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d. Locate the bleed air valve (11) on the power pack (14), remove the air inlet line (12) from the bleed air valve (11) and connect a regulated supply of 18 to 20 psi of dry air to the valve. The air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack (14).

e. Open the LANDING GEAR POWER (CB201) circuit breaker. Unstow the alternate extension hand pump handle (8) and place in the full up position.

f. Loosen the hand pump suction line (9) at the hand pump, loosen the hand pump pressure line (13) at the power pack.

g. With the air pressure shut off, turn the electrical power on and place the landing gear control switch in the down position.

h. Remove the reservoir fill cap (10) and add approximately three quarts of hydraulic fluid (1, Chart 2, 32-00-00) to the reservoir as fast as the system will accept it. Do not exceed the maximum level marked on the fill cap dipstick. Tighten the suction line (9) to the hand pump when fluid appears at the end of the line.

i. Operate the hand pump through full strokes as fast as possible until fluid appears at the end of the hand pump pressure line (13) at the power pack (14); stop pumping and connect the hand pump pressure line (13) to the power pack (14).

j. Ensure that the fill cap (10) is on the power pack (14).

k. Disconnect the alternate extension line (the actuator alternate extend port is identified with an "S") from the nose gear door actuator (2). Pump the hand pump until fluid appears at the end of the line. Connect the hose to the actuator. Crack the B-nut at the actuator (2), pump the hand pump and bleed the line until it is free of air.

l. Crack the B-nut at the nose gear actuator extend port (6). Pump the alternate extension hand pump and bleed the line until free of air.

m. Crack the B-nut at the RH main gear actuator extend port (16). Pump the alternate extension hand pump and bleed the line until free of air.

n. Crack the B-nut at the LH main gear actuator extend port (16). Pump the hand pump and bleed the line until it is free of air.

o. Stow the hand pump handle (8).

p. Add hydraulic fluid (1, Chart 2, 32-00-00) to the fill reservoir and replace the cap (10).

q. Open the LANDING GEAR CONTROL (CB133) circuit breaker.

r. Pressurize the power pack reservoir to 18 to 20 psi.

s. Crack the B-nuts individually at the following locations and bleed each line until it is free of air:

1. Nose gear actuator retract (gear-up) port (7).
2. Nose gear uplock actuator (gear-down) port (4) (the actuator gear-down port is identified with a "P").
3. Nose gear door actuator up port (3).
4. Nose gear door actuator down port (1) (the door actuator down port is identified with a "P").
5. RH main gear actuator retract (gear-up) port (15).
6. RH main uplock actuator gear-down port (18) (the actuator port is identified with a "P").

7. LH main gear actuator retract (gear-up) port (15).

8. LH main uplock actuator gear-down port (18) (the actuator port is identified with a "P").

t. Shut off the air supply.

u. Close the LANDING GEAR CONTROL circuit breaker (CB133) and restore electrical power to the airplane.

v. With the hand pump handle (8) in the stowed position and the system pressurized with air, operate the landing gear a minimum of ten complete cycles with the hydraulic power pack. Between every cycle of the landing gear, shut off the air supply, check the fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required. While cycling the landing gear, check for any signs of hydraulic fluid leakage.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to the power pack reservoir during ground operation of the power pack.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period every five cycles.

w. Use caution during filling to avoid overfilling the system, as excess fluid will be dumped overboard. A container should be positioned under the overboard drain tube to catch any excess fluid.

x. Shut off the air supply. Disconnect the air supply line from the bleed air valve (11). Connect the bleed air line (12) from the regulator to the bleed air valve (11).

y. Refer to Chapter 54-00-00 and install the upper access panel (581AT) on the LH nacelle.

z. Refer to Chapter 24-40-00 and disconnect the external power supply from the airplane.

CAUTION

Before removing the airplane from the jacks, make sure that the hand pump handle is in the stowed position and pinned, and that the landing gear is in the down-and-locked position with the lockpins installed as detailed in Chapters 32-10-00 and 32-20-00.

aa. Remove the airplane from the jacks.

POWER PACK ASSEMBLY REMOVAL (EFFECTIVITY: ALL) (FIGURE 3)

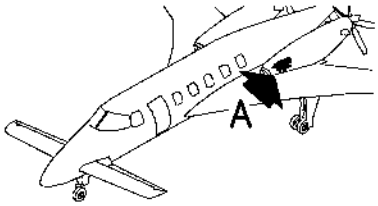
a. Place the airplane on jacks as outlined in Chapter 7-10-00.

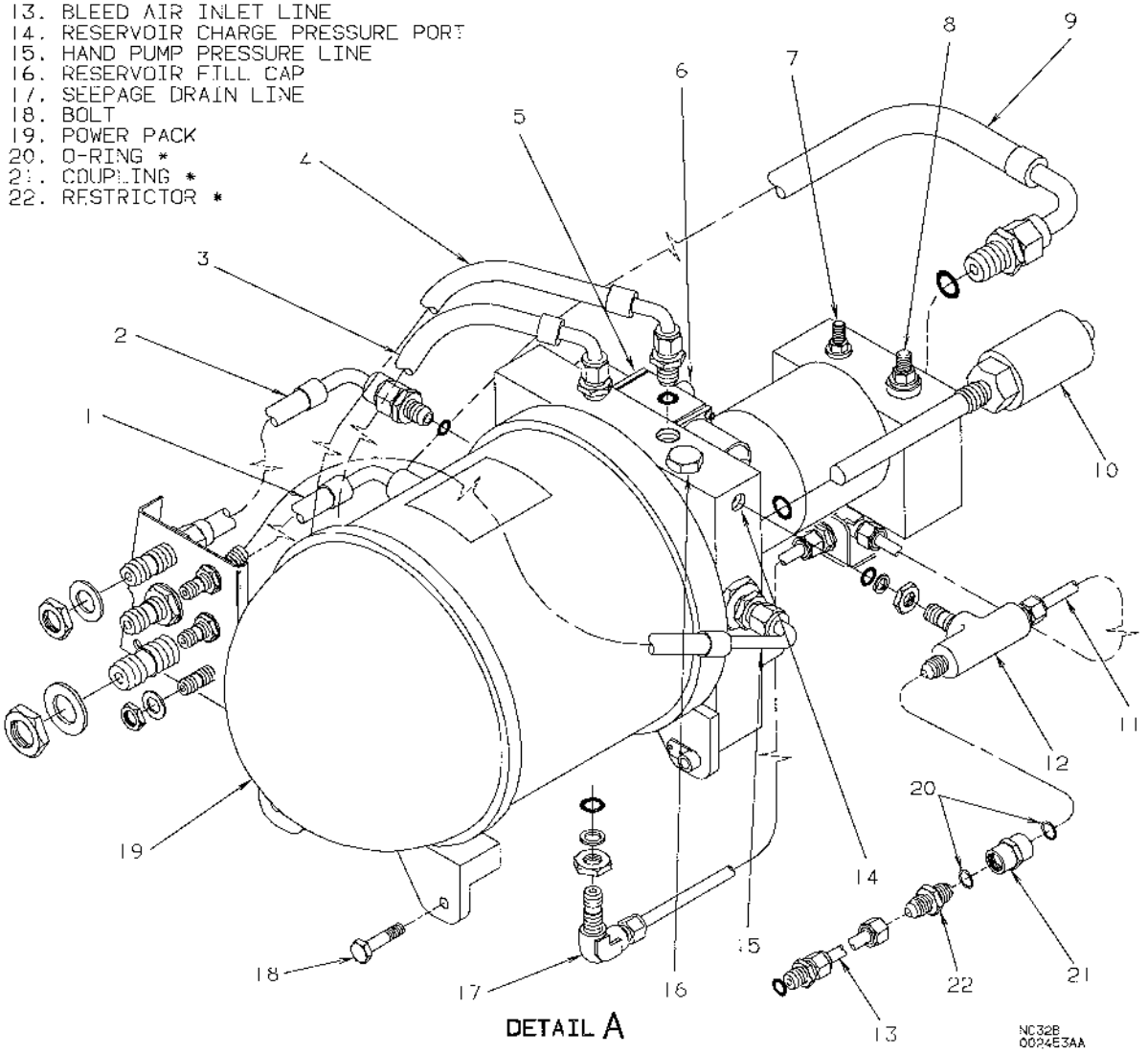
b. Remove all electrical power from the airplane.

c. Refer to chapter 54-00-00 and remove the LH nacelle upper access panels (581AT).

1. GEAR DOWN HYDRAULIC LINE
2. GEAR UP HYDRAULIC LINE
3. DOOR DOWN HYDRAULIC LINE
4. DOOR UP HYDRAULIC LINE
5. LANDING GEAR SOLENOID CONNECTOR
6. NOSE DOOR SOLENOID CONNECTOR
7. TERMINAL
8. TERMINAL
9. HAND PUMP SUCTION LINE
10. LOW LIQUID SENSOR CONNECTOR
11. OVERBOARD LINE
12. BLEED AIR VALVE
13. BLEED AIR INLET LINE
14. RESERVOIR CHARGE PRESSURE PORT
15. HAND PUMP PRESSURE LINE
16. RESERVOIR FILL CAP
17. SEEPAGE DRAIN LINE
18. BOLT
19. POWER PACK
20. O-RING *
21. COUPLING *
22. RESTRICTOR *

* ADDED WITH KIT 122-5036-1





Power Pack Assembly Installation (Effectivity: All)
Figure 3

d. Identify, tag and disconnect the electrical connectors from the motor terminals (7 and 8), the landing gear solenoid connector (5), the nose door solenoid connector (6), and the low liquid sensor connector (10),

CAUTION

As the hydraulic lines are disconnected from the power pack, plug or cap all openings to prevent entry of foreign material into the lines or power pack.

e. Disconnect the following hydraulic lines from the power pack:

1. The gear-down hydraulic line (1).
2. The gear-up hydraulic line (2).
3. The door-up hydraulic line (4).
4. The door-down hydraulic line (3).
5. The hand pump suction line (9).
6. The overboard drain line (11).
7. The bleed air inlet line (13).
8. The hand pump pressure line (15).
9. The seepage drain line (17).

f. Remove the four bolts (18) securing the power pack (19) to the mounting brackets on the nacelle shelf.

g. Remove the power pack (19) from the nacelle.

POWER PACK ASSEMBLY INSTALLATION (EFFECTIVITY: ALL) (FIGURE 3)

a. Position the power pack (19) on the mounting brackets on the LH nacelle and install four bolts (18).

b. Connect the following hydraulic lines to the power pack:

1. The seepage drain line (17).
2. The hand pump pressure line (15).
3. The bleed air inlet line (13).
4. The overboard drain line (11).
5. The hand pump suction line (9).
6. The door-up hydraulic line (4).
7. The door-down hydraulic line (3).
8. The gear-up hydraulic line (2).
9. The gear-down hydraulic line (1).

c. Connect the electrical connectors to the low liquid sensor connector (10), the nose door solenoid connector (6), the landing gear solenoid connector (5), and the motor terminals (7 and 8).

d. Refer to Chapter 24-40-00 and connect a 28 vdc external power supply to the airplane.

e. Remove the reservoir fill cap (16) and add hydraulic fluid (1, Chart 2, 32-00-00) to fill the reservoir to the appropriate mark on the dipstick.

f. Replace the reservoir fill cap (16).

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling (21) and restrictor (22) from the bleed air valve (12). Connect the dry air supply to the restrictor.

g. Locate the bleed air valve (12) on the power pack (19), remove the inlet line (13), and connect a regulated supply of 18 to 20 psi dry air to the valve (12), the air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack (19).

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to the air valve during ground operation of the power pack.

CAUTION

When cycling the landing gear with the power pack, allow a one minute cooling period between each cycle and a five-minute cooling period every five cycles.

h. With the alternate extension hand pump handle in the stowed position and the system pressurized with air, operate the landing gear a minimum of ten complete cycles with the hydraulic power pack until the fluid level stabilizes at the appropriate level on the dipstick. Every third or fourth cycle of the landing gear; shut off the air supply, check the fluid level, and add fluid as required. While cycling the landing gear, check for any signs of hydraulic fluid leakage.

i. Shut off the air supply and disconnect the air supply line from the bleed air valve (12) and reconnect the inlet line (13) from the pressure regulator to the bleed air valve.

j. Refer to Chapter 54-00-00 and install the LH nacelle upper access panels.

k. Disconnect the external power supply from the airplane.

CAUTION

Before removing the airplane from the jacks, make sure that the hand pump handle is in the stowed position and pinned and that the landing gear is in the down and locked position.

l. Remove the airplane from the jacks.

m. Restore the electrical power to the airplane.

LOW LIQUID LEVEL SENSOR REMOVAL (EFFECTIVITY: ALL) (FIGURE 4)

a. Remove all electrical power from the airplane.

b. Place the airplane on jacks as outlined in Chapter 7-10-00.

c. Refer to Chapter 54-00-00 and remove the LH nacelle upper access panels (581AT).

d. Disconnect the electrical connector from the sensor (2).

e. Remove the reservoir fill cap (1) to monitor the fluid level in the reservoir.

f. Disconnect the hand pump pressure line (4) from the power pack (5) and drain the hydraulic fluid from the reservoir down to a level lower than the sensor (2).

g. Connect the hand pump pressure line (4) to the power pack (5).

h. Unscrew the sensor (2) from the power pack (5).

i. Discard the O-ring (3).

LOW LIQUID LEVEL SENSOR INSTALLATION (EFFECTIVITY: ALL) (FIGURE 4)

a. Install a new MS28778-8 O-ring (3) on the sensor (2).

b. Screw the sensor (2) into the power pack (5). The sensor (2) should be installed hand-tight only.

c. Connect the electrical connector to the sensor (2).

d. Add hydraulic fluid (1, Chart 2, 32-00-00) to the reservoir to the appropriate fill mark on the dipstick and install the reservoir fill plug (1).

e. Refer to Chapter 54-00-00 and install the LH upper access panels (581AT).

f. Remove the airplane from the jacks.

g. Restore the electrical power to the airplane.

POWER PACK BLEED AIR VALVE REMOVAL (EFFECTIVITY: AIRPLANES WITHOUT KIT 122-5036-1 INSTALLED) (FIGURE 5)

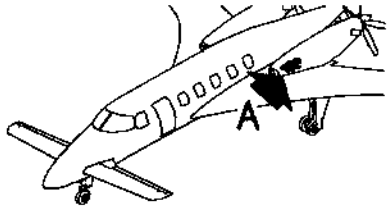
a. Refer to Chapter 54-00-00 and remove the LH nacelle upper access panel (581AT).

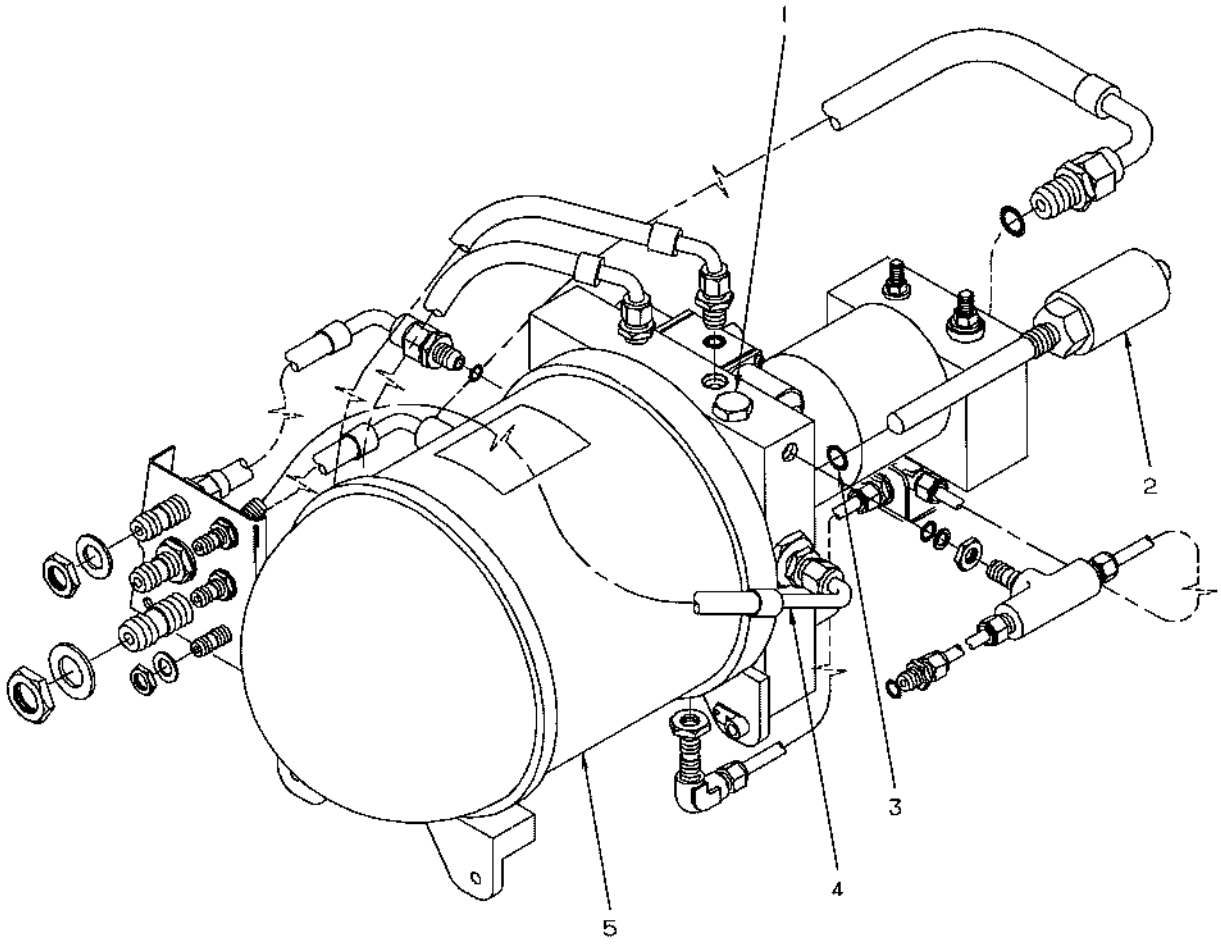
b. Disconnect the bleed air inlet line (7) from the inlet port ("I") of the bleed air valve (6).

c. Disconnect the overboard drain line (5) from the air pressure relief port ("R") of the bleed air valve (6).

d. Loosen the nut (4) and remove the bleed air valve (6) from the power pack (8). Discard the O-ring (2) and the backup ring (3).

1. RESERVOIR FILL PLUG
2. LOW LIQUID LEVEL SENSOR
3. O-RING
4. HAND PUMP PRESSURE LINE
5. HYDRAULIC POWER PACK





DETAIL A

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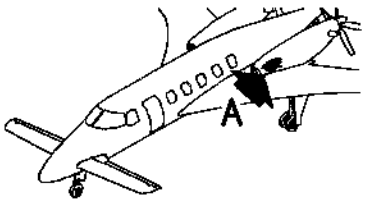
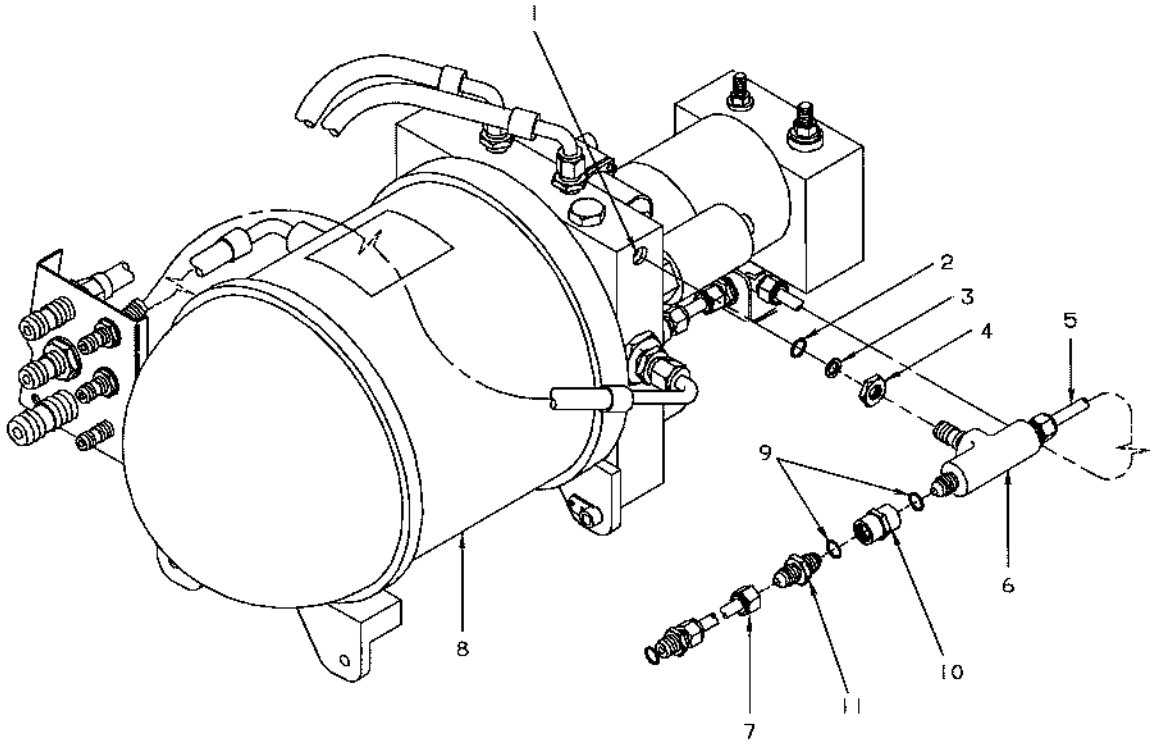
Low Liquid Level Sensor Installation (Effectivity: All)
Figure 4



1. RESERVOIR CHARGE PRESSURE PORT
2. O-RING
3. BACKUP RING
4. NUT
5. OVERBOARD DRAIN LINE
6. BLEED AIR VALVE
7. BLEED AIR INLET LINE
8. POWER PACK
9. O-RING *
10. COUPLING *
11. RESTRICTOR *

* ADDED WITH KIT 122-5036-1



DETAIL A

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Power Pack Bleed Air Valve Installation (Effectivity: All)
Figure 5

POWER PACK BLEED AIR VALVE INSTALLATION (EFFECTIVITY: AIRPLANES WITHOUT KIT 122-5036-1 INSTALLED)
(FIGURE 5)

a. Install the nut (4), new backup ring (3) and new MS28778-4 O-ring (2) on the valve (6).

b. Install the valve (6) in the power pack (8).

c. Connect the overboard drain line (5) and bleed air inlet line (7) to the bleed air valve (6).

d. Refer to Chapter 54-00-00 and install the LH upper access panel (581AT).

POWER PACK BLEED AIR VALVE REMOVAL (EFFECTIVITY: AIRPLANES WITH KIT 122-5036-1 INSTALLED)
(FIGURE 5)

a. Refer to Chapter 54-00-00 and remove the LH nacelle upper access panel (581AT).

b. Disconnect the bleed air inlet line (7) from the restrictor (11).

c. Disconnect the restrictor (11) from the coupling (10).

d. Disconnect the coupling (10) from the inlet port ("I") of the bleed air valve (6).

e. Discard the O-rings (9).

f. Disconnect the overboard drain line (5) from the air pressure relief port ("R") of the bleed air valve (6).

g. Loosen the nut (4) and remove the bleed air valve (6) from the power pack (8). Discard the O-ring (2) and the backup ring (3).

POWER PACK BLEED AIR VALVE INSTALLATION (EFFECTIVITY: AIRPLANES WITH KIT 122-5036-1 INSTALLED)
(FIGURE 5)

a. Install the nut (4), new backup ring (3) and new MS28778-4 O-ring (2) on the valve (6).

b. Install the valve (6) in the power pack (8).

c. Install the coupling (10) and a new MS28778-4 O-ring (9) on the inlet port ("I") of the bleed air valve (6).

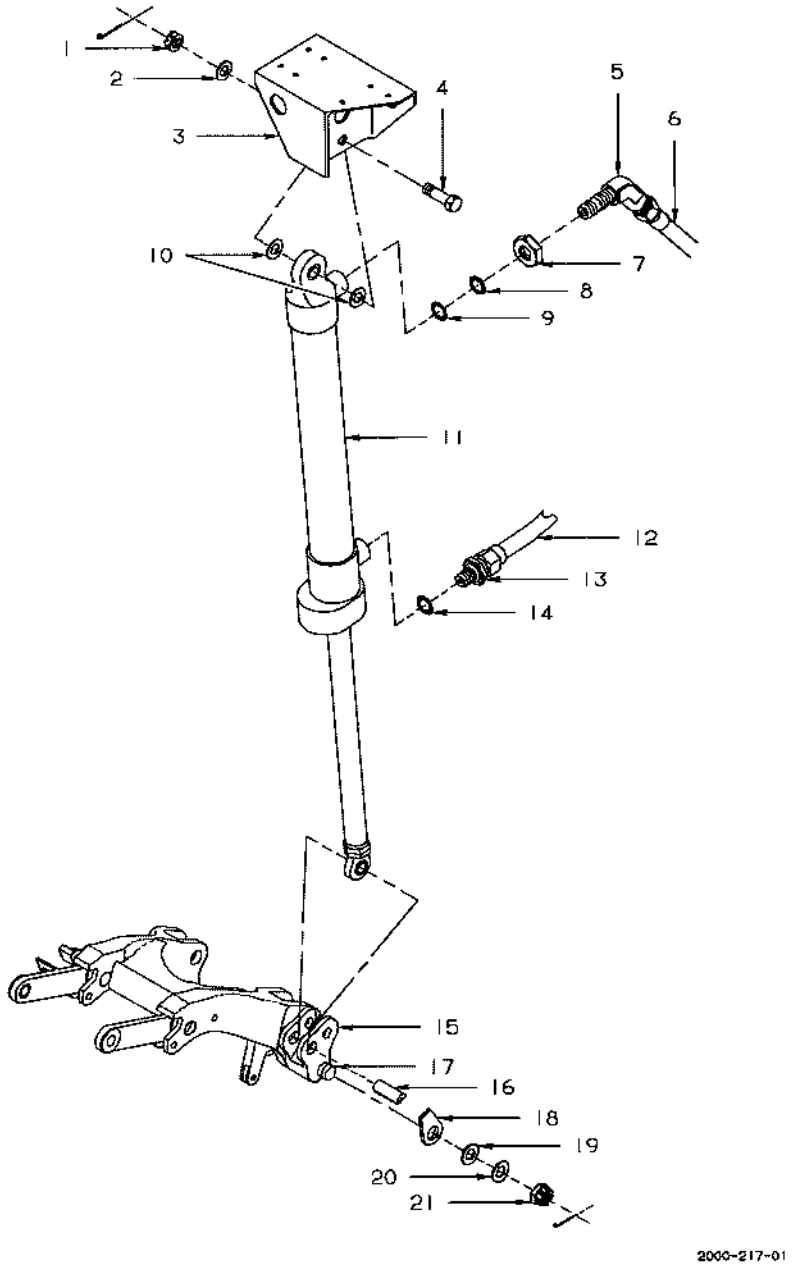
d. Install the restrictor (11) and a new MS28778-4 O-ring (9) on the coupling (10).

e. Connect the bleed air inlet line (7) to the restrictor (11).

f. Connect the overboard drain line (5) to the air pressure relief port ("R") of the bleed air valve (6).

g. Refer to Chapter 54-00-00 and install the LH upper access panel (581AT).

1. NUT
2. WASHER
3. ACTUATOR FITTING
4. BOLT
5. ELBOW
6. GEAR EXTEND HOSE
7. NUT
8. RING
9. O-RING
10. WASHER
11. ACTUATOR
12. GEAR RETRACT HOSE
13. RESTRICTOR
14. O-RING
15. ACTUATOR BRACKET
16. ACTUATOR PIVOT PIN
17. BOLT
18. LOCK PLATE
19. WASHER
20. WASHER
21. NUT



Nose Gear Actuator Installation (Effectivity: All)
Figure 6

NOSE GEAR ACTUATOR REMOVAL (EFFECTIVITY: ALL) (FIGURE 6)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL) to gain access to the upper actuator fitting (3).

c. Using the alternate extension hand pump in the cockpit, pump the forward nose gear doors open.

d. Install the lockpin (3, Chart 1, 32-00-00) in accordance with the NOSE LANDING GEAR DOOR LOCK PIN INSTALLATION procedure in Chapter 32-20-00.

e. Place the airplane on jacks as outlined in Chapter 7-10-00.

CAUTION

When disconnecting hydraulic lines, install plugs or caps on all openings to prevent the entry of foreign material.

f. Disconnect the two hydraulic hoses (6 and 12) from the actuator (11).

g. If required, remove the elbow (5), ring (8), restrictor (13) and O-rings (9 and 14). Discard the O-rings (9 and 14).

h. Remove the cotter pin, nut (1) and washer (2) from the bolt (4).

i. Remove the bolt (4) to disconnect the actuator (11) from the actuator fitting (3). Retain the washers (10) in their original positions when the actuator (11) is removed.

j. Support the actuator (11) and remove the cotter pin, nut (21), washers (19 and 20), lock plate (18) and actuator pivot pin (16) from the actuator bracket (15).

k. Remove the actuator (11) from the actuator bracket (15).

NOSE GEAR ACTUATOR INSTALLATION (EFFECTIVITY: ALL) (FIGURE 6)

a. If removed from the actuator, install a new ring (8) and O-rings (9 and 14) with the elbow (5) and restrictor (13).

b. Position the actuator (11) into the actuator fitting (3) with the washers (10) in the same positions as when the actuator was removed and install the bolt (4).

c. Install the washer (2), nut (1) and cotter pin on the bolt (4).

d. To eliminate any side load on the actuator (11), check the alignment of the actuator (11) with the actuator bracket (15). With the nose gear in both the retract and extend positions, the actuator rod end must slip into the actuator bracket (15) without causing side loads on the actuator piston rod. If any misalignment is found, remove the bolt (4) and shift the washers (10) on either side of the actuator fitting (3) to bring the actuator (11) into alignment. A minimum of one washer is required between the actuator (11) and the actuator fitting (3).

e. Position the actuator (11) in the actuator bracket (15) and install the actuator pivot pin (16), lock plate (18), washers (19 and 20), nut (21) and cotter pin.

f. Remove the caps or plugs and connect the two hydraulic hoses (6 and 12) to the actuator (11).

g. Bleed the actuator as follows:

1. Check the system fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required, refer to Chapter 12-10-00.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

2. Locate the bleed air valve on the power pack, remove the air inlet line from the valve and connect a regulated supply of 18 to 20 psi dry air to the air valve; the air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack.

3. Pressurize the system. Crack the B-nuts individually at the actuator (11) and bleed each hose (6 and 12) until it is free of air.

4. Shut off the air supply and check the system fluid level in the power pack and add hydraulic fluid (1, Chart 2, 32-00-00) as required, refer to Chapter 12-10-00.

h. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

i. Remove the nose and main landing gear lockpins. Remove the nose landing gear door lockpin.

j. Check the nose gear rigging as outlined under NOSE GEAR RIGGING in this chapter, and adjust the uplock and downlock switches as outlined in Chapter 32-60-00.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

k. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down-and-locked position.

l. After the landing gear is in the down-and-locked position, ensure that the lockpins are installed in the landing gear (refer to Chapters 32-10-00 and 32-20-00).

m. Shut off the air supply and disconnect the air supply line from the bleed air valve.

n. Connect the air inlet line to the bleed air valve on the power pack.

o. Refer to Chapter 53-50-00 and install the forward wing upper fairing (211BL).

p. Before removing the airplane from the jacks, make sure the hand pump handle is in the stowed position and pinned.

q. Remove the airplane from the jacks.

r. Restore electrical power to the airplane.

NOSE GEAR DOOR ACTUATOR REMOVAL (EFFECTIVITY: ALL) (FIGURE 7)

a. Remove all electrical power from the airplane.

b. Using the alternate extension hand pump in the cockpit, pump the forward nose gear doors open. Install the lockpin (3, Chart 1, 32-00-00) in accordance with the NOSE LANDING GEAR DOOR LOCK PIN INSTALLATION procedure in Chapter 32-20-00.

c. Place the airplane on jacks as outlined in Chapter 7-10-00.

CAUTION

When disconnecting the hydraulic lines, install plugs or caps on all openings to prevent the entry of foreign material.

d. Disconnect the three hydraulic hoses (1, 4 and 9) from the actuator (14).

e. Remove the nut (17), washer (16) and bolt (15) to disconnect the actuator (14) from the bellcrank (18).

f. Remove the nut (12), washer (11) and bolt (10) to disconnect the actuator (14) from the actuator fitting (13).

g. If required, remove the unions (2 and 7), restrictor (5) and O-rings (3, 6 and 8). Discard the O-rings (3, 6 and 8).

NOSE GEAR DOOR ACTUATOR INSTALLATION (EFFECTIVITY: ALL) (FIGURE 7)

a. If the unions (2 and 7) or restrictor (5) were removed from the actuator (14), install them with new O-rings (3, 6 and 8).

b. Position the actuator (14) into the actuator fitting (13) and install the bolt (10), washer (11) and nut (12).

c. Connect the actuator (14) to the bellcrank (18) with the bolt (15), washer (16), and nut (17).

d. Remove the caps or plugs and connect the three hydraulic hoses (1, 4 and 9) to the actuator (14).

e. Bleed the actuator as follows:

1. Refer to Chapter 54-00-00 and remove the LH nacelle upper access panel (581AT).

2. Check the system fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required, refer to Chapter 12-20-00.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

3. Locate the bleed air valve on the power pack, remove the air inlet line from the valve and connect a regulated supply of 18 to 20 psi dry air to the air valve. The air supply should incorporate a shutoff valve to control the

flow of pressurized air to the power pack.

4. Pressurize the system. Crack the B-nuts individually at the actuator and bleed each line until it is free of air.

5. Shut off the air supply.

6. Check the system fluid level. Refer to Chapter 12-10-00 and add hydraulic fluid (1, Chart 2, 32-00-00) as required.

f. Refer to Chapter 24-40-00 and apply 28-vdc external power to the airplane.

g. Remove the nose landing gear door lockpin. Remove the nose and main landing gear lockpins.

h. Check the nose gear door rigging as outlined under NOSE GEAR DOOR RIGGING in this chapter, and adjust the door open switch as outlined in Chapter 32-60-00.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

i. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down-and-locked position.

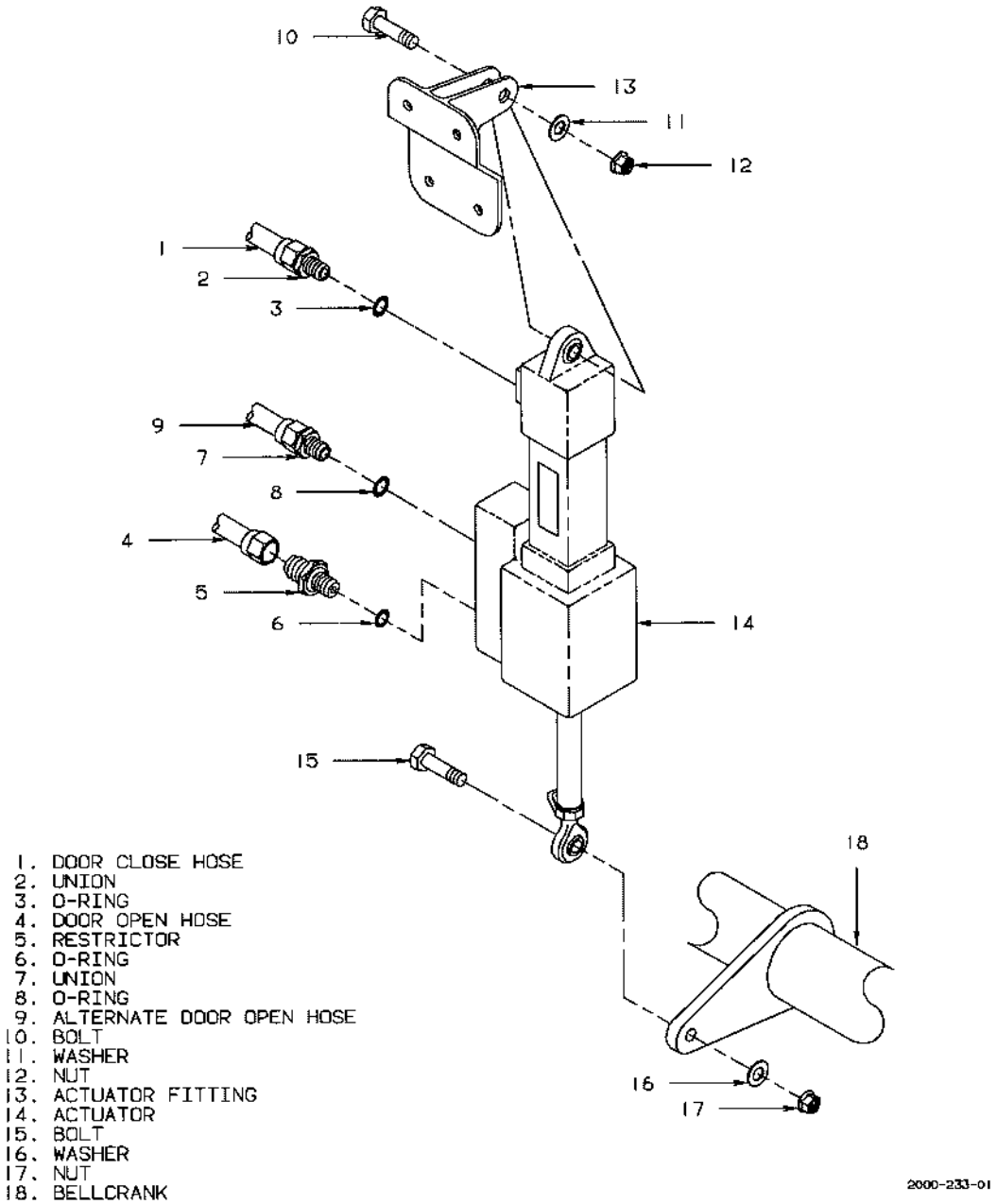
j. After the landing gear is in the down-and-locked position, ensure that the lockpins are installed in the landing gear, refer to Chapters 32-10-00 and 32-20-00.

k. Shut off the air supply and disconnect the air supply from the bleed air valve. Connect the air inlet line to the bleed air valve on the power pack.

l. Before removing the airplane from the jacks, make sure the alternate extension hand pump handle is in the stowed position and pinned.

m. Remove the airplane from the jacks.

n. Restore electrical power to the airplane.



1. DOOR CLOSE HOSE
2. UNION
3. O-RING
4. DOOR OPEN HOSE
5. RESTRICTOR
6. O-RING
7. UNION
8. O-RING
9. ALTERNATE DOOR OPEN HOSE
10. BOLT
11. WASHER
12. NUT
13. ACTUATOR FITTING
14. ACTUATOR
15. BOLT
16. WASHER
17. NUT
18. BELLCRANK

Nose Gear Door Actuator Installation (Effectivity: All)
Figure 7

NOSE GEAR UPLOCK ACTUATOR REMOVAL (EFFECTIVITY: ALL) (FIGURE 8)

a. Remove all electrical power from the airplane.

b. Using the alternate extension hand pump in the cockpit, pump the forward nose gear doors open. Install the lockpin (3, Chart 1, 32-00-00) in accordance with the NOSE LANDING GEAR DOOR LOCK PIN INSTALLATION procedure in Chapter 32-20-00.

c. Place the airplane on jacks as outlined in Chapter 7-10-00.

CAUTION

When disconnecting the hydraulic lines, install plugs and caps on all openings to prevent entry of foreign material into the hoses or actuator.

d. Disconnect the three hydraulic hoses (1, 2 and 11) from the uplock actuator (22).

e. Remove the cotter pin, nut (16), washers (13), bushings (14) and bolt (12) to disconnect the uplock actuator (22) from the uplock hook (15).

f. Remove the cotter pin, nut (17), washer (18), bushing (21) and bolt (20) to disconnect the uplock actuator (22) from the actuator fitting (19).

g. If required, remove the elbow (10), unions (3 and 4), nut (9), ring (8) and O-rings (5, 6 and 7). Discard the O-rings (5, 6 and 7) and ring (8).

NOSE GEAR UPLOCK ACTUATOR INSTALLATION (EFFECTIVITY: ALL) (FIGURE 8)

a. If the elbow (10) or unions (3 and 4) were removed from the actuator (22), install them with new O-rings (5, 6 and 7) and ring (8).

b. Position the actuator (22) into the actuator fitting (19) and install bolt (20), bushing (21), washer (18), nut (17) and cotter pin.

c. Connect the uplock actuator (22) to the uplock hook (15) with the bolt (12), washers (13), bushings (14), nut (16) and cotter pin.

d. Connect the three hydraulic hoses (1, 2 and 11) to the uplock actuator (22).

e. Bleed the actuator as follows:

1. Refer to Chapter 54-00-00 and remove the LH nacelle access panels.

2. Check the system fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required, refer to Chapter 12-10-00.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

3. Locate the bleed air valve on the power pack, remove the air inlet line from the valve and connect a regulated supply of 18 to 20 psi of dry air to the air valve. The air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack.

4. Pressurize the system. Crack the B-nuts individually at the actuator (22) and bleed each line (1, 2, and 11) until it is free of air.

5. Shut off the air supply.

6. Check the system fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required, refer to Chapter 12-10-00.

f. Remove the nose landing gear door lockpin.

g. Manually push the nose gear to the retracted position. Adjust the rod end of the uplock actuator (if necessary) to allow the uplock hook (15) to bottom out on the uplock roller with .03-.09 inches of uplock rod end slot remaining with the uplock actuator in the extended position. After any adjustment is complete, tighten the jam nut and safety wire to the key washer.

h. Check the nose gear rigging as outlined under NOSE GEAR RIGGING in this chapter, and adjust the nose gear uplock switch as outlined in Chapter 32-60-00.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

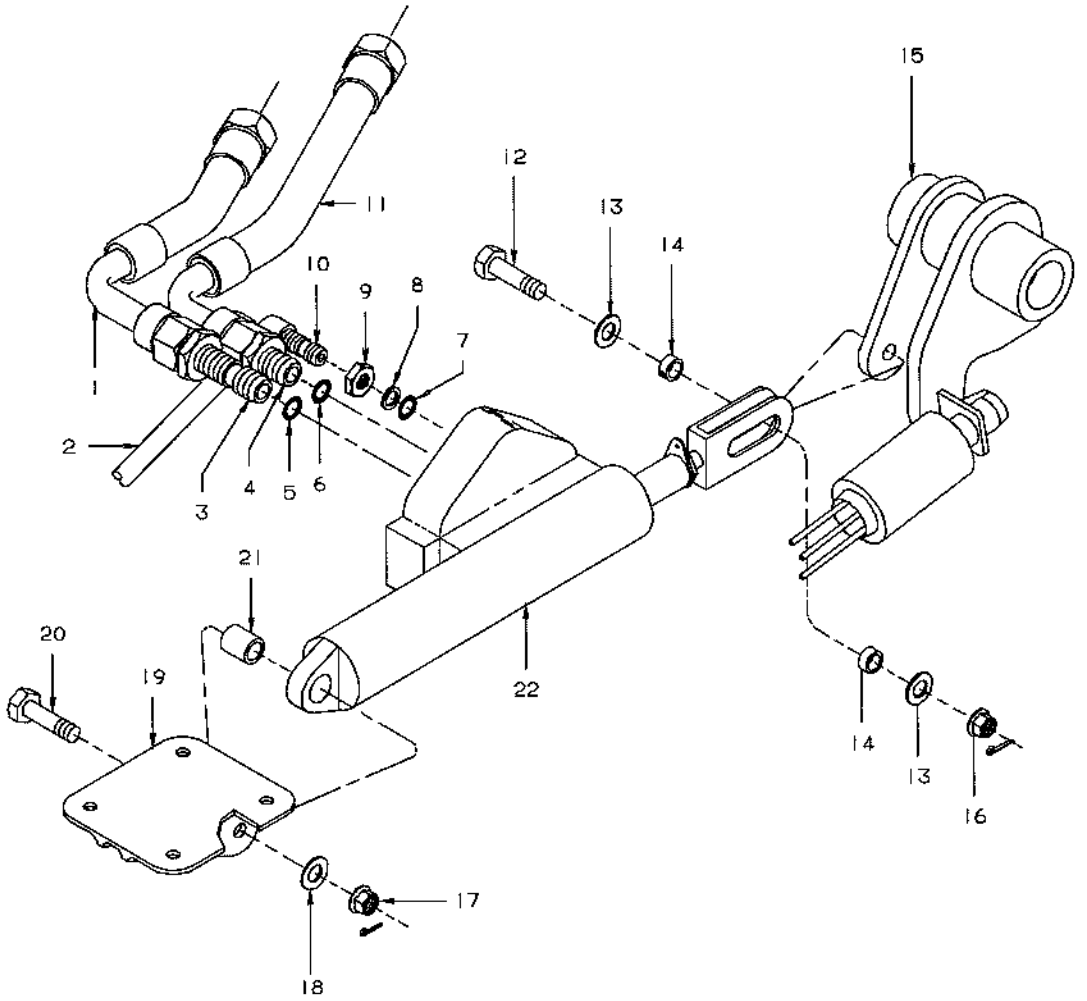
i. Before removing the airplane from the jacks, make sure the alternate extension handle is in the stowed position and pinned. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down-and-locked position.

j. After the landing gear is in the down and locked position, install the lockpins in the landing gear (refer to Chapters 32-10-00 and 32-20-00).

k. Shut off the air supply and disconnect the air supply line from the air valve. Reconnect the inlet line from the pressure regulator to the bleed air valve on the power pack.

l. Remove the airplane from the jacks.

m. Restore electrical power to the airplane.



1. HYDRAULIC HOSE
2. HYDRAULIC HOSE
3. UNION
4. UNION
5. O-RING
6. O-RING
7. O-RING
8. RING
9. NUT
10. ELBOW
11. HYDRAULIC HOSE
12. BOLT
13. WASHER
14. BUSHING
15. UPLOCK HOOK
16. NUT
17. NUT
18. WASHER
19. ACTUATOR FITTING
20. BOLT
21. BUSHING
22. UPLOCK ACTUATOR

Nose Gear Uplock Actuator Installation (Effectivity: All)
Figure 8

NOSE GEAR UPLOCK HOOK ASSEMBLY REMOVAL (EFFECTIVITY: ALL) (FIGURE 9)

a. Remove all electrical power from the airplane.

b. Using the alternate extension hand pump in the cockpit, pump the forward nose gear doors open. Install the lockpin (3, Chart 1, 32-00-00) in accordance with the NOSE LANDING GEAR DOOR LOCK PIN INSTALLATION procedure in Chapter 32-20-00.

c. Place the airplane on jacks as outlined in Chapter 7-10-00.

d. Remove the nose gear uplock actuator as detailed in NOSE GEAR UPLOCK ACTUATOR REMOVAL.

e. Remove the cotter pin (2) and pivot pin retaining pin (1) from the upper drag brace pivot pin (3).

f. Remove the upper drag brace pivot pin (3) from the upper drag brace (4).

g. Remove the four washers (5, 6, 12, and 13), spring (7), and two bushings (8 and 9) from the uplock hook (11).

h. Disconnect the wiring from the uplock switch (10).

i. Remove the nose gear uplock hook (11) from the airplane.

NOSE GEAR UPLOCK HOOK ASSEMBLY INSTALLATION (EFFECTIVITY: ALL) (FIGURE 9)

a. Install the two bushings (8 and 9), spring (7) and four washers (5, 6, 12, and 13) on the uplock hook (11).

b. Position the uplock hook assembly (11) on the upper drag brace (4) and install the upper drag brace pivot pin (3), the pivot pin retaining pin (1) and the cotter pin (2).

c. Connect the wiring to the uplock switch (10).

d. Install the nose gear uplock actuator as detailed in NOSE GEAR UPLOCK ACTUATOR INSTALLATION.

e. Remove the nose gear doors lockpin.

CAUTION

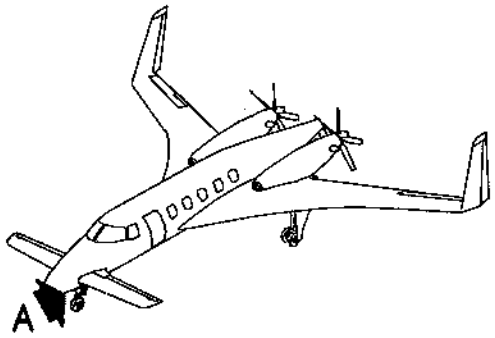
To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly.

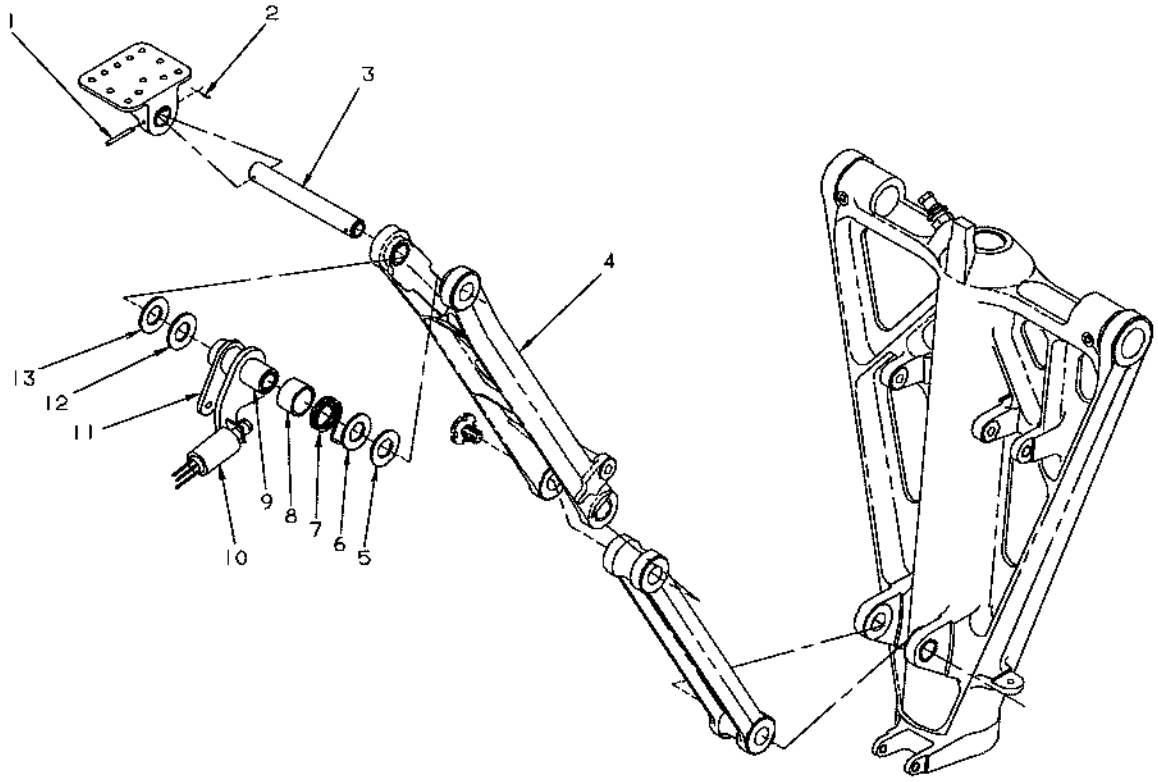
CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

f. Before removing the airplane from the jacks, make sure the alternate extension handle is in the stowed position and pinned. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down-and-locked position.

1. RETAINER PIN
2. COTTER PIN
3. PIVOT PIN
4. UPPER DRAG BRACE
5. WASHER
6. WASHER
7. SPRING
8. BUSHING
9. BUSHING
10. UPLOCK SWITCH
11. UPLOCK HOOK
12. WASHER
13. WASHER





DETAIL A

C94NC32B0407

Nose Gear Uplock Hook Assembly Installation (Effectivity: All)
Figure 9

g. Remove the airplane from the jacks.

h. Restore electrical power to the airplane.

PRIORITY VALVE REMOVAL (EFFECTIVITY: ALL) (FIGURE 10)

a. Remove all electrical power from the airplane.

b. Using the alternate extension hand pump in the cockpit, pump the forward nose gear doors open. Install the lockpin (3, Chart 1, 32-00-00) in accordance with the NOSE LANDING GEAR DOOR LOCK PIN INSTALLATION procedure in Chapter 32-20-00.

CAUTION

When disconnecting the hydraulic lines, install caps and plugs on all openings to prevent the entry of foreign objects.

c. Loosen the fitting and disconnect the hand pump tube (4), the nose gear uplock actuator line (7), and the nose gear door line (9) from the priority valve (1).

d. Remove the two bolts (5) and washers (6) from the clamps (8).

e. Remove the priority valve (1) from the airplane.

PRIORITY VALVE INSTALLATION (EFFECTIVITY: ALL) (FIGURE 10)

NOTE

When removing the union (3) from the priority valve (1), replace the ring (2).

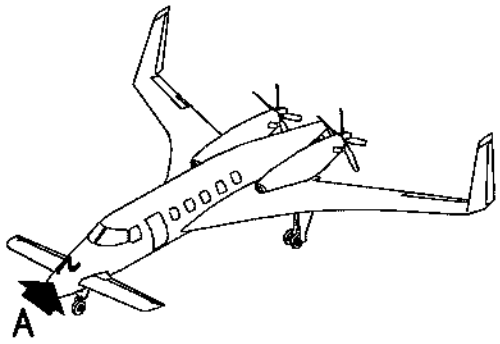
a. Position the priority valve (1) in nose gear wheel well and install the two clamps (8), washers (6) and bolts (5).

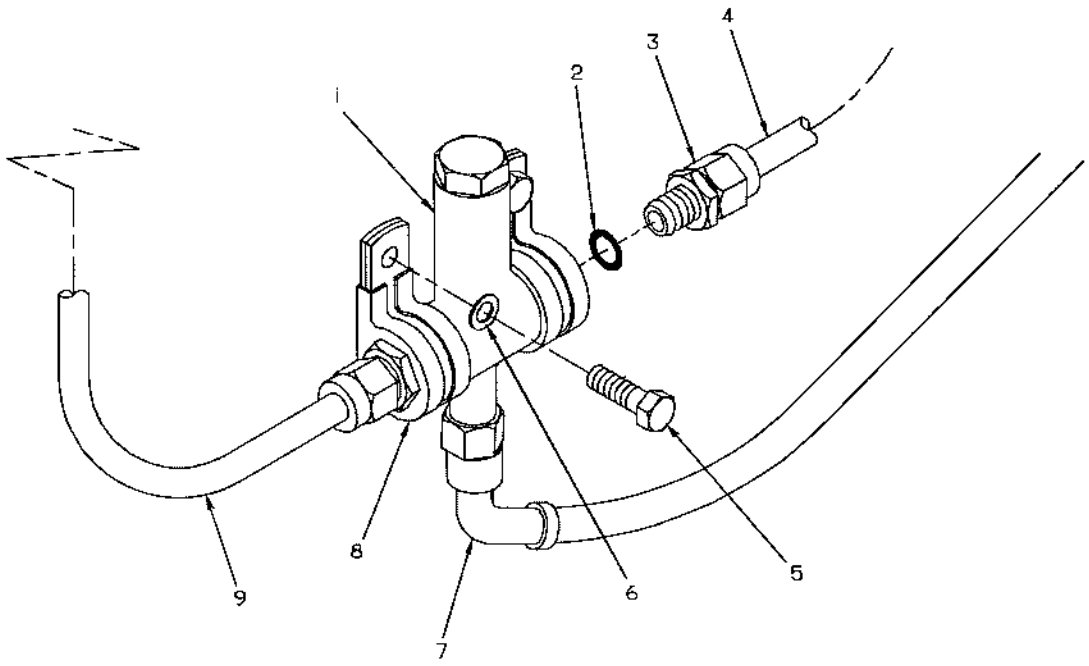
b. Remove the caps and plugs and connect the hand pump line (4), the uplock actuator line (7), and the gear door line (9) to the priority valve (1) and tighten the fittings.

c. Remove the nose gear door lockpin.

d. Restore electrical power to the airplane.

1. PRIORITY VALVE
2. O-RING
3. UNION
4. HAND PUMP LINE
5. BOLT
6. WASHER
7. NOSE GEAR UPLOCK ACTUATOR LINE
8. CLAMP
9. NOSE GEAR DOOR LINE





DETAIL A

C94NC32B0406

Priority Valve Installation (Effectivity: All)
Figure 10

MAIN GEAR ACTUATOR REMOVAL (EFFECTIVITY: ALL)
(FIGURE 11)

CAUTION

When disconnecting the hydraulic lines, install plugs or caps on all openings to prevent the entry of foreign material into the hoses or actuator.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 7-10-00 and place the airplane on jacks.

c. Remove lockpin (1, Chart 1, 32-00-00) from the landing gear.

d. Disconnect the two hydraulic hoses (8 and 12) from the actuator (9).

e. Remove the cotter pin and washer (2) from the pin (5).

f. Remove the pin (5), washer (3), and pin (4) from the actuator (9).

g. Support the actuator (9) and remove the pin, and washer (16) from the pin (18).

h. Remove the pin (18), and sleeve (17) from the pin (13).

i. Remove the pin (13), and the actuator (9) from the jury brace retract link (15) and drive links (14).

j. Remove the actuator (9) from the wheel well.

MAIN GEAR ACTUATOR INSTALLATION (EFFECTIVITY: ALL)
(FIGURE 11)

a. If the union (11) and restrictor (7) were removed from the actuator (9), install them in the actuator (9) with new O-rings (6 and 10).

b. Align the actuator (9) with the upper side brace (1) and install the pin (4), and washer (3). Use washers (3) as required under the head of the pin (4) to remove play in the joint when the pin (5) is installed.

c. Align the hole in pin (4) with the hole in the upper side brace (1) and install the pin (5) through the upper side brace (1) and the pin (4). Install the washer (2) and cotter pin.

d. Align the actuator (9), jury brace retract link (15), and drive links (14) and install the pin (13). Install the sleeve (17) on the pin (13) and install the pin (18), washer (16) and cotter pin.

e. Connect the two hydraulic hoses (8 and 12) to the actuator (9) and tighten the fittings.

f. Bleed the actuator as follows:

1. Check the system fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

2. Locate the bleed air valve on the power pack, remove the air inlet line from the valve and connect a regulated supply of 18 to 20 psi dry air to the air valve. The air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack.

3. Pressurize the system. Crack the B-nuts individually at the actuator and bleed each line (8 and 12) until it is free of air.

4. Shut off the air supply.

5. Check the system fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required.

g. Remove the landing gear lockpins.

h. Check the landing gear rigging as outlined under MAIN GEAR RIGGING in this chapter, and adjust the uplock and downlock switches as outlined in Chapter 32-60-00.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly.

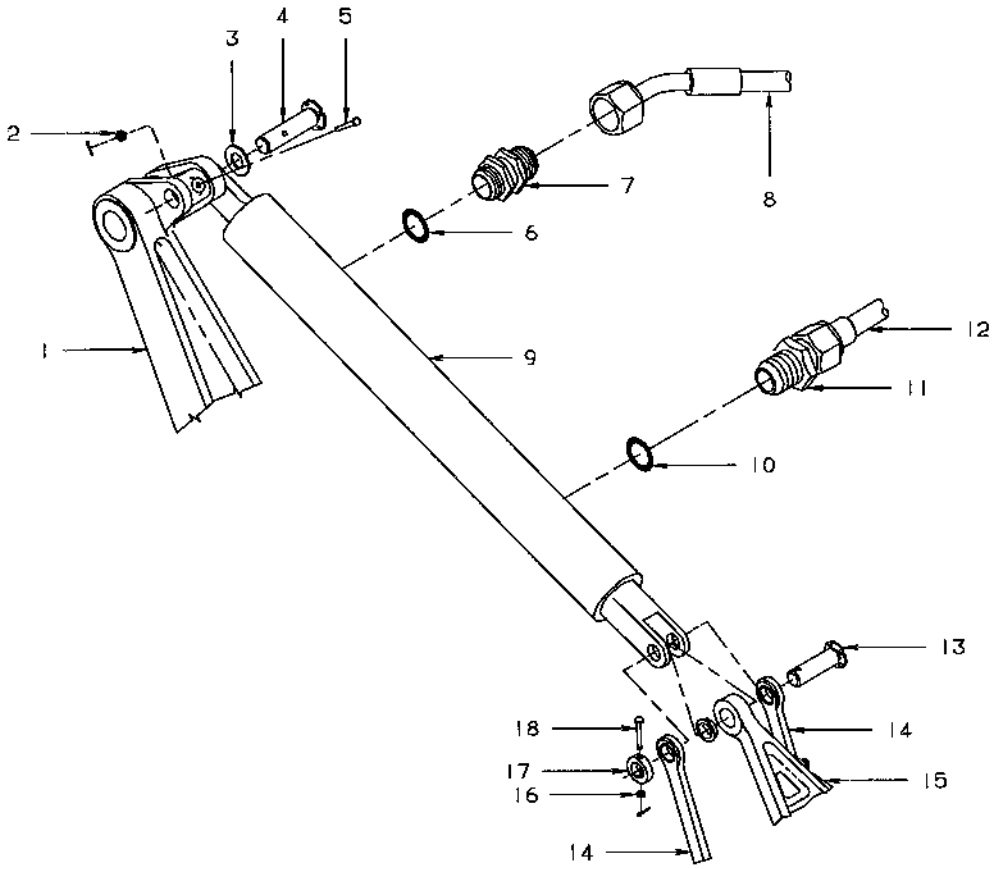
CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

i. Before removing the airplane from the jacks, make sure the alternate extension handle is in the stowed position and pinned. Retract and extend the landing gear, using the hydraulic power pack to ensure that the landing gear is in the down and locked position. After the landing gear is in the down and locked position, ensure that the lockpins are installed in the landing gear.

j. Shut off air supply to the air valve and disconnect the air supply line from the air valve. Reconnect the air inlet line from the pressure regulator to the bleed air valve.

k. Remove the airplane from the jacks.



1. SIDE BRACE
2. WASHER
3. WASHER
4. PIN
5. PIN
6. O-RING
7. RESTRICTOR
8. GEAR RETRACT HOSE
9. MAIN GEAR ACTUATOR
10. O-RING
11. UNION
12. GEAR EXTEND HOSE
13. PIN
14. DRIVE LINK
15. JURY BRACE RETRACT LINK
16. WASHER
17. SLEEVE
18. PIN

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Main Gear Actuator Installation (Effectivity: All)
Figure 11

MAIN GEAR UPLOCK ACTUATOR REMOVAL (EFFECTIVITY: ALL) (FIGURE 12)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 7-10-00 and place the airplane on jacks.

c. Remove lockpin (1, Chart 1, 32-00-00) from the landing gear.

CAUTION

When disconnecting the hydraulic lines, install plugs or caps on all openings to prevent the entry of foreign material.

d. Loosen the fittings, disconnect the three hoses (1, 2, and 3) from the uplock actuator (15), and install plugs in the fittings.

e. If required, remove the union (4), nut (18), and O-rings (5 and 16). Discard the O-rings (5 and 16) and ring (17).

f. Remove the cotter pin, nut (13), washer (12), bushing (11) and bolt (14) to disconnect the uplock actuator (15) from the uplock hook (10).

g. Remove the cotter pin, nut (6), washer (7), and bushing (8) to remove the uplock actuator (15) from the actuator support (9).

MAIN GEAR UPLOCK ACTUATOR INSTALLATION (EFFECTIVITY: ALL) (FIGURE 12)

a. If the elbow or unions (4) were removed from the actuator (15), install a new ring (17) and O-rings (5, and 16) with them.

b. Position the uplock actuator (15) on the actuator support (9) and install the bushing (8), washer (7), nut (6) and cotter pin.

c. Connect the uplock actuator (15) to the uplock hook (10) and install the bolt (14), bushing (11), washer (12), nut (13) and cotter pin.

d. Connect the hand pump pressure hose (1) to the "S" port elbow on the actuator (15) and tighten the fitting.

e. Connect the primary hose (2) to the "P" port union on the actuator (15) and tighten the fitting.

f. Connect the transfer hose (3) to the "T" port union on the actuator (15) and tighten the fitting.

g. Bleed the actuator as follows:

1. Refer to Chapter 54-00-00 and remove the LH nacelle upper access panels.
2. Check the system fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

3. Locate the bleed air valve on the power pack, remove the air inlet line from the valve and connect a regulated supply of 18 to 20 psi dry air to the valve. The air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack.

4. Pressurize the system. Crack the B-nuts individually at the actuator and bleed each line until it is free of air.

5. Shut off the air supply.

6. Check the system fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required.

h. Check the main gear rigging as outlined under MAIN GEAR RIGGING in this chapter and adjust the main gear uplock switch as outlined in Chapter 32-60-00.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

i. Before removing the airplane from the jacks, make sure the alternate extension handle is in the stowed position and pinned. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down-and-locked position.

j. After the landing gear is in the down and locked position, ensure that the lockpins are installed in the landing gear. (Refer to Chapters 32-10-00 and 32-20-00).

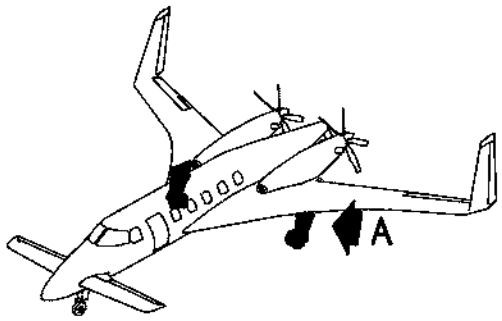
k. Shut off the air supply and disconnect the air supply line from the air valve. Reconnect the line from the pressure regulator to the air valve.

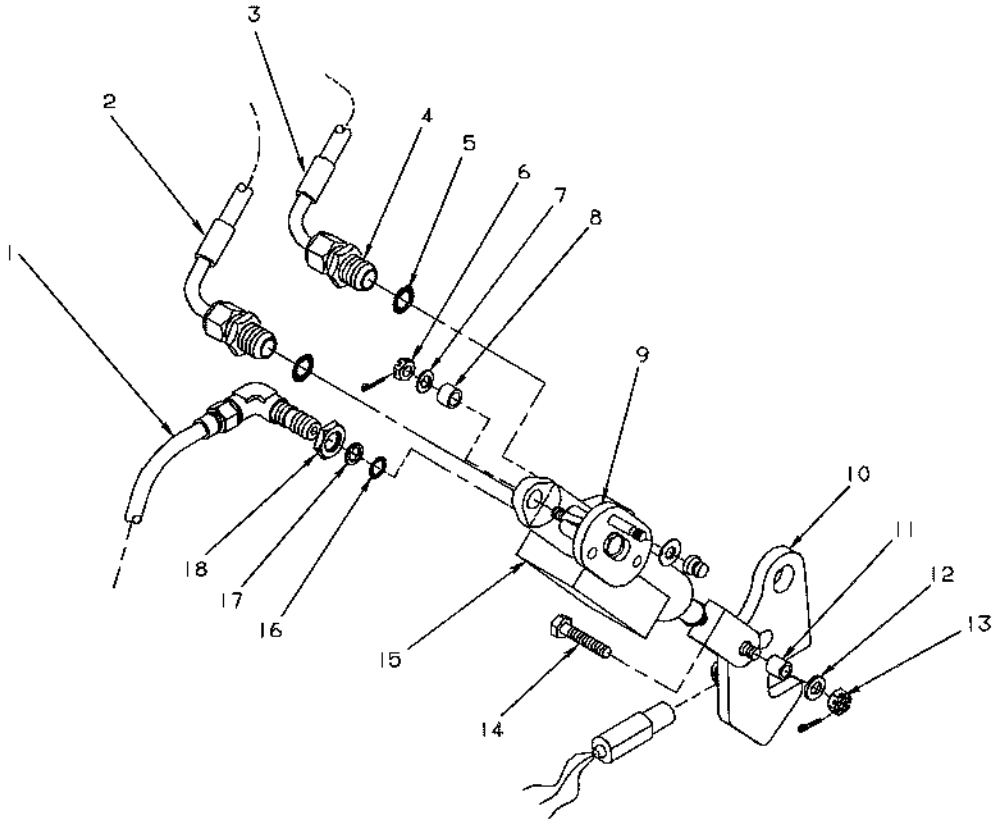
l. Refer to Chapter 32-60-00 and perform the MAIN GEAR UPLOCK SWITCH ADJUSTMENT procedure.

m. Remove the airplane from the jacks.

n. Restore electrical power to the airplane.

1. HAND PUMP PRESSURE HOSE
2. UPLOCK ACTUATOR PRIMARY HOSE
3. UPLOCK ACTUATOR TRANSFER HOSE
4. UNION
5. O-RING
6. NUT
7. WASHER
8. BUSHING
9. ACTUATOR SUPPORT
10. UPLOCK HOOK
11. BUSHING
12. WASHER
13. NUT
14. BOLT
15. MAIN GEAR UPLOCK ACTUATOR
16. O-RING
17. RING
18. NUT





DETAIL A

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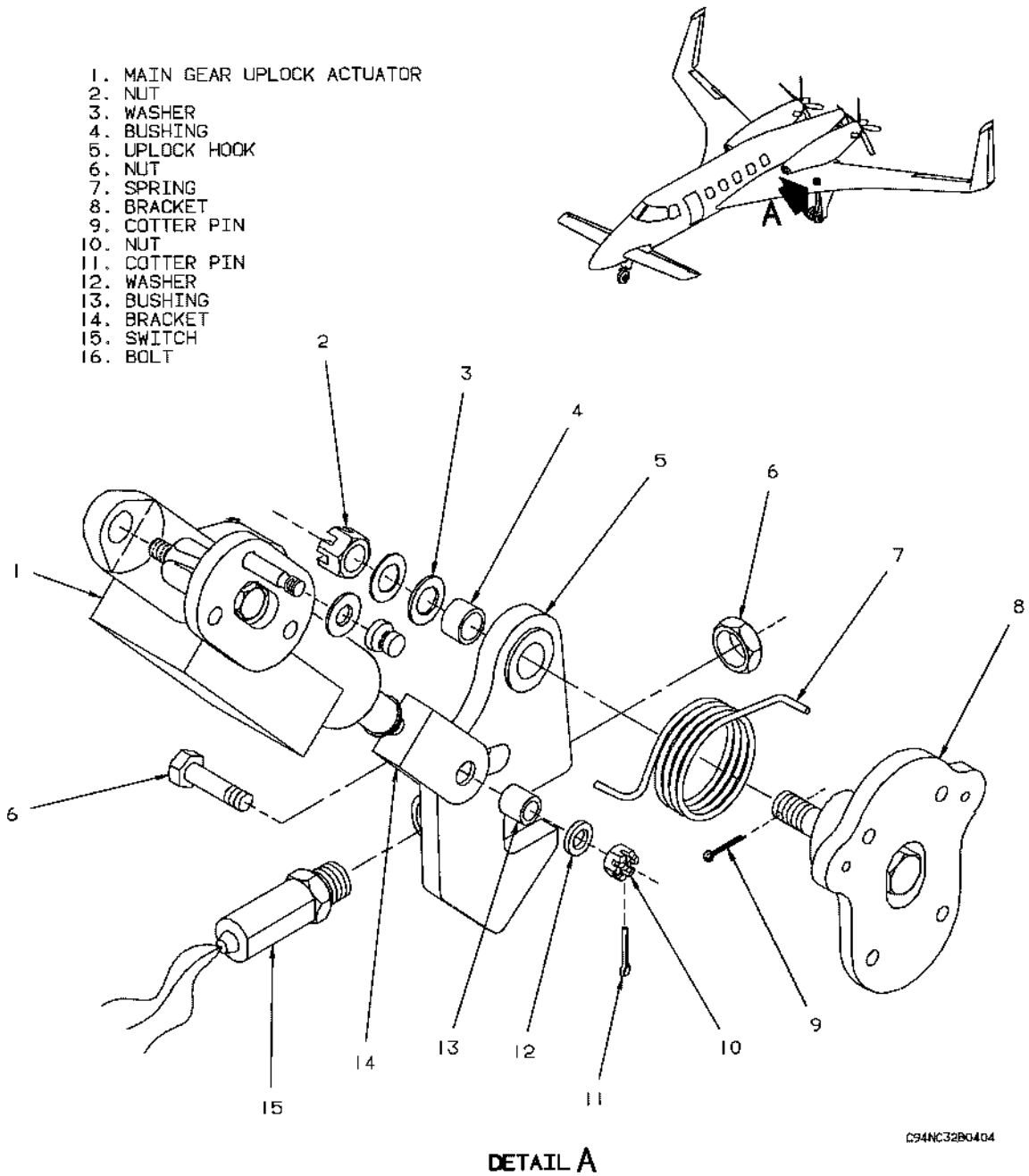
Main Gear Uplock Actuator Installation (Effectivity: All)
Figure 12



Main Gear Uplock Hook Assembly (Effectivity: All)
Figure 13

MAIN GEAR UPLOCK HOOK ASSEMBLY REMOVAL (EFFECTIVITY: ALL) (FIGURE 13)

a. Remove all electrical power from the airplane.

b. Place the airplane on jacks as outlined in Chapter 7-10-00.

c. Remove the cotter pin (11), nut (10), washer (12), bushing (13) and bolt (16) and disconnect the uplock actuator (1) from the uplock hook (5).

d. Remove the cotter pin (9), nut (2), two washers (3), bushing (4), uplock hook (5) and spring (7) from the bracket (8).

e. Remove the nut (6) and remove the uplock sensor switch (15) from the uplock hook (5).

f. Remove the uplock hook (5) from the airplane.

MAIN GEAR UPLOCK HOOK ASSEMBLY INSTALLATION (EFFECTIVITY: ALL) (FIGURE 13)

a. Position the switch (15) in uplock hook (5) and install the nut (6).

b. Install the spring (7), uplock hook (5), bushing (4), two washers (3), nut (2) and cotter pin (9) on the bracket (8).

c. Connect the uplock actuator (1) to the uplock hook (5) and install the bolt (16), bushing (13), washer (12), nut (10), and cotter pin (11).

d. Refer to Chapter 32-60-00 and perform the MAIN LANDING GEAR UPLOCK SWITCH ADJUSTMENT procedure.

e. Remove the airplane from the jacks.

f. Restore electrical power to the airplane.

ALTERNATE EXTENSION HAND PUMP HANDLE REMOVAL (EFFECTIVITY: ALL) (FIGURE 14)

a. Refer to Chapter 25-20-00 and remove the pilot's seat and the cockpit floorboards (143CTC and 141CT).

b. Remove the cotter pin (17), washer (16), pin (14), and spacer (10) from the link assembly (12).

c. Remove the cotter pin (5), nut (4), washer (6), spacer (7), eight washers (8, 9, 10, and 15) and bolt (13) from the handle (1).

d. Remove the pin (2) from the clip (3) and remove the handle (1) from the airplane.

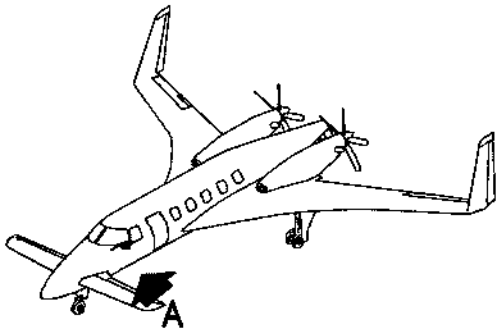
ALTERNATE EXTENSION HAND PUMP HANDLE INSTALLATION (EFFECTIVITY: ALL) (FIGURE 14)

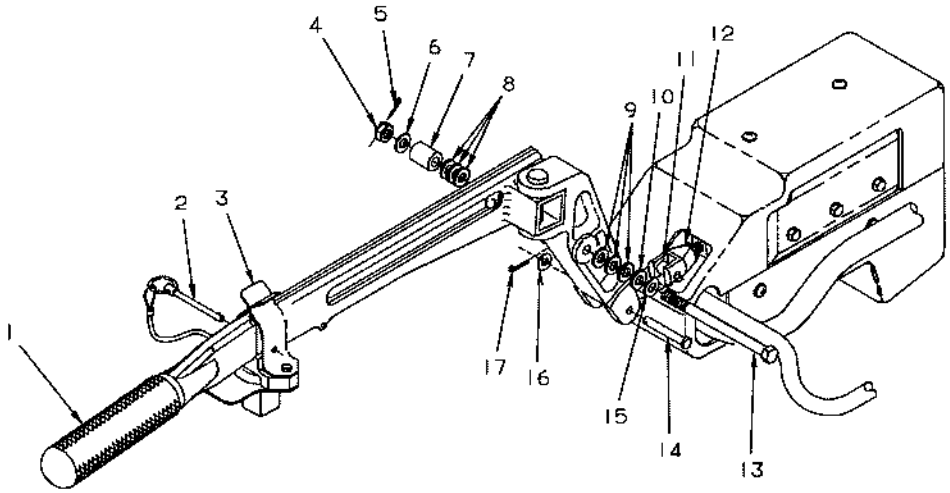
a. Install the five washers (9, 10, and 15) on the bolt (13), align the handle (1) with the fuselage and install the bolt at the pivot point on the handle (1).

b. Install the three washers (8), spacer (7), washer (6), and nut (4) on the bolt (13) and secure the nut (4) with the cotter pin (5).

c. Align the handle (1) with the link assembly (12) and install the pin (14), washer (16), and cotter pin (17).

1. HANDLE
2. PIN
3. CLIP
4. NUT
5. COTTER PIN
6. WASHER
7. SPACER
8. WASHER
9. WASHER
10. WASHER
11. SPACER
12. LINK ASSEMBLY
13. BOLT
14. PIN
15. WASHER
16. WASHER
17. COTTER PIN





DETAIL A

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Alternate Extension Hand Pump Handle Installation (Effectivity: All)
Figure 14

d. Secure the handle (1) in the clip (3) and install the pin (2).

e. Refer to Chapter 25-20-00 and install the cockpit floorboards (143CTC and 141CT) and the pilot's seat.

ALTERNATE EXTENSION HAND PUMP REMOVAL (EFFECTIVITY: ALL) (FIGURE 15)

a. Remove all electrical power from the airplane.

b. Ensure that the landing gear lockpins are installed as detailed in Chapters 32-10-00 and 32-20-00.

c. Refer to Chapter 25-20-00 and remove the four cockpit floorboards (141BT, 141CT, 143BTC, and 143CTC).

d. Refer to ALTERNATE EXTENSION HAND PUMP HANDLE REMOVAL and remove the hand pump handle.

e. Remove pressure from the landing gear hydraulic system.

f. Remove the seven screws (2) and the upper shield (1).

g. Remove the cotter pin (25), washer (24), pin (22), and link assembly (23) from the hand pump (4).

h. Loosen the fitting (21), disconnect the hand pump pressure hose (20), and remove the O-ring. Discard the O-ring and install a plug in the hose (20).

i. Remove the two screws (14), and two washers (13) from the lower shield (18).

j. Remove the three bolts (7), six washers (6 and 26), three nuts (27), and the side shield (3) from the hand pump (4).

k. Loosen the fitting (8), disconnect the elbow (9) and hand pump suction hose (19) from the hand pump (4), and cap the fitting (8).

l. Remove the hand pump (4) from the airplane.

m. Remove the nut (10), three washers (11, 12, and 15) and the fitting (16) from the lower shield (18). Install a cap on the drain hose (17).

n. Remove the lower shield (18) from the airplane.

ALTERNATE EXTENSION HAND PUMP INSTALLATION (EFFECTIVITY: ALL) (FIGURE 15)

a. Remove the cap from the drain hose (17).

b. Install the washer (15) and fitting (16) in the lower shield (18), and secure with the two washers (12 and 11) and nut (10).

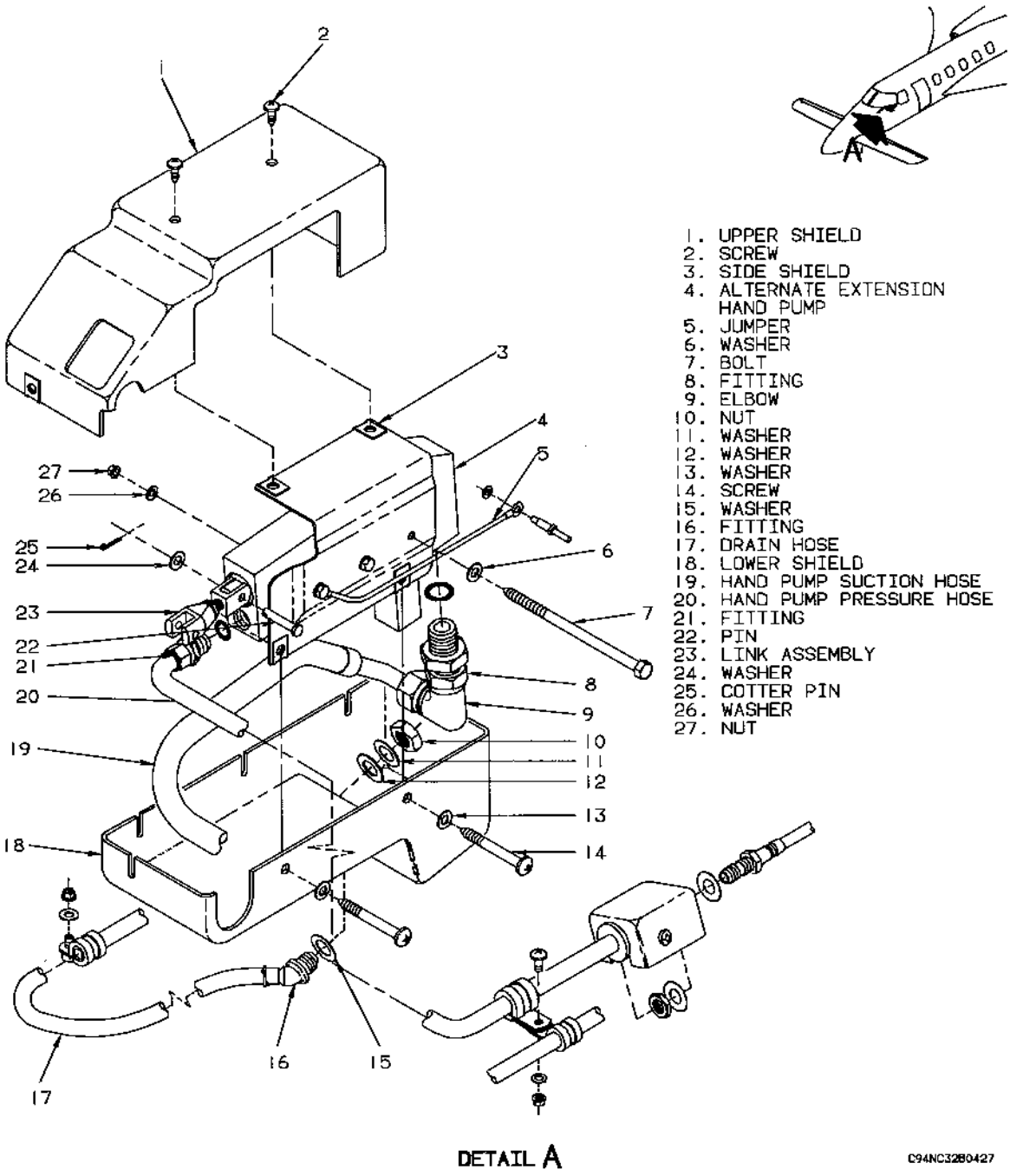
c. Remove the cap from the hand pump suction hose (19) and connect the fitting (8) to the hand pump union.

d. Remove the cap from the hand pump pressure hose (20) and connect the fitting (21) to the hand pump union.

e. Align the side shield (3) with the hand pump (4) and install three bolts (7), six washers (6 and 26), and three nuts (27) securing the hand pump (4) to the fuselage. Install the jumper (5) on the forward bolt.

f. Align the lower shield (18) with the side shield (3) and install two screws (14) and washers (13).

g. Align the link assembly (23) with the hand pump (4) and install the pin (22), washer (24), and cotter pin (25).



Alternate Extension Hand Pump Installation (Effectivity: All)
Figure 15

h. Position the upper shield (1) on the hand pump (4) and install seven screws (2)

i. Refer to ALTERNATE EXTENSION HAND PUMP HANDLE INSTALLATION and install the hand pump handle.

j. Refer to Chapter 25-20-00 and install the four cockpit floorboards (141BT, 141CT, 141BTC, and 143CTC).

k. Refill and bleed the hydraulic system as detailed in FILLING AND BLEEDING HYDRAULIC SYSTEM.

l. Restore electrical power to the airplane.

LANDING GEAR RIGGING (EFFECTIVITY: ALL)

For safety reasons the 5-ampere LANDING GEAR CONTROL circuit breaker on the circuit breaker panel in the pilot's console should be pulled while rigging the landing gear. The landing gear should not be cycled electrically until the landing gear is properly rigged.

A Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00) or it's equivalent, may be utilized to retract or extend the landing gear during the rigging procedure. When connected to the primary extend and retract ports of one uplock actuator, the pump will allow that landing gear to be extended and retracted individually.

Before removing the airplane from the jacks, make sure that the alternate extension hand pump handle is in the stowed position and pinned. Connect a 28 vdc external power supply, then retract and extend the landing gear with the hydraulic power pack to ensure that the landing gear is in the down-and-locked position before removing the airplane from the jacks. Install the lockpins in the landing gear as instructed in Chapters 32-10-00 and 32-20-00.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

The power pack reservoir must be pressurized during ground operation. To pressurize the system while on the ground, remove the air inlet line from the bleed air valve located on the power pack and connect a regulated supply of 18 to 20 psi dry air to the valve. The air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

MAIN GEAR RIGGING (EFFECTIVITY: ALL)

CAUTION

Do not fully cycle the landing gear electrically until all the gears and doors are properly rigged. Use of a Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00) or it's equivalent, is recommended for initial rigging. Unless the hydraulic system is properly bled, the main landing gear must be supported when it is extended. An improperly bled system does not provide for proper damping of the gear extension rate and could damage the retraction linkage. If the system requires bleeding refer to FILLING AND BLEEDING THE HYDRAULIC SYSTEM in this chapter.

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Remove the lockpin (1, Chart 1, 32-00-00) from the landing gear.

c. Disconnect the door rods from the door and secure the door in the full open position.

d. With the main gear in the extended position, check to see that the lock link assembly falls overcenter .04 to .06 inch. If not, examine the assembly and correct the deficient parts or alignment. Ensure the jury brace assembly moves freely without any binding.

e. With the hydraulic actuator in the fully retracted position and the landing gear in the down and locked position, adjust actuator clevis so that the attach bolt will install freely in the drive links, then turn clevis in one full turn. Tighten the jam nut and safety wire the nut to the key washer. Ensure the jury brace assembly moves freely without any binding.

f. To eliminate any side load on the actuator, the alignment of the actuator to the upper side brace and to the jury brace retract link assembly must be checked. The actuator may be shimmed at it's attachment to the upper side brace by adding washers to the pin (note Figure 11).

g. Manually push the main gears to the retracted position. With the uplock actuator in the extended position, adjust the rod ends of the uplock actuators (if necessary) to allow the uplock hooks to bottom out on the uplock rollers with .03 to .09 inch of uplock hook slot remaining. After rod end adjustment is complete, tighten the jam nut and safety wire the nut to key washer.

h. Adjust the uplock and downlock switches as outlined in Chapter 32-60-00, then cycle the landing gear to check for proper operation of the in-transit and gear-down lights.

i. Connect the landing gear doors and check the rigging of the doors as outlined under MAIN GEAR DOOR RIGGING.

j. Remove the airplane from the jacks.

NOSE GEAR RIGGING (EFFECTIVITY: ALL)

CAUTION

Do not cycle the landing gear electrically until all the gears and doors are properly rigged. Use of a Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00) or it's equivalent, is recommended for initial rigging.

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Using the alternate extension hand pump in the cockpit, pump the forward nose landing gear doors open.

c. Remove the lockpin (2, Chart 1, 32-00-00) from the nose gear.

d. Disconnect the door rods from both the aft and forward doors and secure the doors in the full open position.

e. With the nose gear extended, disconnect the actuator piston rod end from the nose gear. Refer to NOSE GEAR ACTUATOR REMOVAL in this chapter to disconnect the actuator.

f. Check to see that the actuator in the gear fully-down position still has .03 to .18 inch of extension travel remaining. If not, adjust actuator rod end to have .03 to .18 inch of extension travel remaining. After the adjustment is complete, tighten the jam nut and safety to key washer.

g. With the gear still in the extended position, check to see that the aft lock assembly falls overcenter by .04 to .065 inch. If not, examine the assembly and correct the deficient parts or alignment. Ensure the jury brace moves freely without any binding.

h. To eliminate any side load on the actuator, check the alignment of the actuator rod end with the actuator bracket attached to the nose gear. With the nose gear in both the retracted and extended positions the actuator rod end must slip into the actuator bracket without causing any side loads on the actuator piston rod. If any misalignment is found, shift the washers on either side of the actuator upper attach point to bring the actuator into alignment. A minimum of one washer is required between actuator and actuator support bracket. Refer to the nose gear actuator removal and installation procedures in this chapter if shifting of the washers is necessary.

i. Refer to NOSE GEAR ACTUATOR INSTALLATION in this chapter and connect the actuator rod end to the actuator bracket on the nose gear.

j. Manually push the nose gear to the retracted position. Adjust the rod end of the uplock actuator (if necessary) to allow the uplock hook to bottom out on the uplock roller with .03-.09 inch of uplock rod end slot remaining with the uplock actuator in the extended position. After any adjustment is complete, tighten the jam nut and safety wire the jam nut to the key washer.

k. Adjust the downlock and uplock switches as outlined in Chapter 32-60-00, and cycle the landing gear to check for proper operation of the in-transit and gear-down lights.

l. Reconnect the nose gear doors. Check the rigging of the doors as outlined under NOSE GEAR DOOR RIGGING in this chapter.

m. Cycle landing gear to the down and locked position.

n. Install the lockpin (2, Chart 1, 32-00-00) in the nose gear as detailed in Chapter 32-20-00.

o. Remove the airplane from the jacks.

LANDING GEAR DOOR RIGGING (EFFECTIVITY: ALL)

The landing gear doors consist of two sets of nose gear doors and one door for each main gear. Before the landing gear doors can be rigged, the landing gear must first be properly rigged as outlined in this chapter.

A Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00), or it's equivalent, may be utilized to retract or extend the landing gear during the door rigging procedure. When connected to the primary extend and retract ports of one uplock actuator, the pump will allow that landing gear to be extended and retracted individually. Connection of the above pump to the nose gear door actuator will allow the forward set of doors to be opened and closed during the rigging procedure.

MAIN GEAR DOOR RIGGING (EFFECTIVITY: ALL)

Prior to rigging the main gear doors, the main landing gear must be properly rigged as outlined under MAIN GEAR RIGGING in this chapter.

A Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00), or it's equivalent, may be utilized to retract or extend the landing gear during the door rigging procedure. When connected to the primary extend and retract ports of one uplock actuator, the pump will allow that landing gear to be extended and retracted individually.

a. Place the airplane on jacks as detailed in Chapter 7-10-00.

b. Connect the above pump to the main gear uplock actuator.

c. Using the pump, fully retract the main gear.

d. Adjust the door pushrods so that the doors close snugly without any gaps.

e. After adjustment of both main gear doors, cycle the landing gear several times with the power pack, checking each time that the doors close properly.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir in place of the engine bleed air.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period every five cycles.

f. Before removing the airplane from the jacks, make sure the emergency extension handle is in the stowed position and pinned. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down-and-locked position before removing the airplane from the jacks.

g. Refer to Chapter 32-10-00 and install the lockpins in the gear.

h. Remove the airplane from the jacks.

FORWARD NOSE GEAR DOORS RIGGING (EFFECTIVITY: ALL)

Prior to rigging the nose gear forward doors, the landing gear must be properly rigged as outlined under NOSE GEAR RIGGING in this chapter.

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Using the alternate extension hand pump in the cockpit, pump the forward nose landing gear doors open.

c. Disconnect the forward door pushrods from the doors.

d. Connect a pump (4, Chart 1, 32-00-00) to the forward door actuator.

e. Push the nose gear forward doors closed by hand and check for fit.

f. Connect the pushrods to the forward doors and adjust the rods to ensure a snug fit when doors are closed and locked, but do not exceed 10 pounds of preload per door.

g. Open the doors to the fully open position and retract the nose gear using the pump (4, Chart 1, 32-00-00), while checking to ensure adequate clearance (more than .5 inch) between the nose gear and the forward doors.

h. With the nose gear retracted and the aft doors closed, close the forward doors and check that the forward doors overlap the aft doors by a minimum of .19 inch.

i. Refer to Chapter 32-60-00 and check the adjustment of the door-open switch and door closed switch.

j. Cycle the landing gear several times with the power pack, checking each time that the doors close properly when the landing gear is retracted.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir in place of the engine bleed air.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after every five cycles.

k. Before removing the airplane from the jacks, make sure that the emergency extension handle is in the stowed position. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down-and-locked position before removing the airplane from the jacks.

l. Install the lockpins in the landing gear as detailed in Chapters 32-10-00 and 32-20-00.

m. Remove the airplane from the jacks.

AFT NOSE GEAR DOORS RIGGING (EFFECTIVITY: ALL)

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Using the alternate extension hand pump in the cockpit, pump the forward nose landing gear doors open.

c. Rig the nose gear aft doors to provide the proper fit with the nose gear in the fully retracted position by adjusting the rod ends on the aft door links so that the aft doors close snugly without excessive preload when the nose gear is fully retracted. Do not exceed 10 pounds of preload along the aft edge of doors. No gaps should exist along any

of the door seals.

d. With the forward doors open, cycle the nose gear with a pump (4, Chart 1, 32-00-00), and check for smooth operation of the aft doors and a minimum clearance of .13 inch between the aft and forward doors when both sets of doors are fully open.

e. When adjustment of the door is complete, tighten rod end jam nuts on the door links and safety wire.

f. Cycle the landing gear several times with the power pack, checking each time that the doors close properly when the landing gear is retracted.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir in place of the engine bleed air.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period every five cycles.

g. Before removing the airplane from the jacks, make sure that the emergency extension handle is in the stowed position. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down-and-locked position before removing the airplane from the jacks.

h. Install the lockpins in the landing gear as detailed in Chapters 32-10-00 and 32-20-00.

i. Remove the airplane from the jacks.

LANDING GEAR RETRACTION AND EXTENSION CHECK (POWER PACK) (EFFECTIVITY: ALL)

a. Place the airplane on jacks as instructed in Chapter 7-10-00.

b. Refer to Chapter 24-40-00 and connect a 28-vdc external power supply to the airplane.

c. Before operating the landing gear with the hydraulic power pack, check the fluid level in the power pack reservoir and add hydraulic fluid (1, Chart 2, 32-00-00) as required. Refer to Chapter 12-10-00. If the hydraulic fluid level is low, the landing gear should not be operated using the power pack until the fluid level is corrected.

d. Refer to Chapters 32-10-00 and 32-20-00 and remove the lockpins from the landing gear.

e. Locate the air valve on the power pack, remove the air inlet line from the valve and connect a regulated supply of 18 to 20 psi of dry air to the air valve. The air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly.

f. Check that the in-transit lights and gear down lights illuminate when the LDG GR TEST switch on the pilot's outboard subpanel is pressed.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

g. Ensure that the landing control circuit breaker is pushed in, then move the gear handle to the UP position. The red-in transit lights should illuminate in the control switch handle to signify that the gear is in transit or unlocked. When the gear is fully retracted, the red lights should extinguish.

h. Place the landing gear control handle in the DN position. The red lights should illuminate while the gear is in transit. When the gear is fully down and locked, the red lights should extinguish and the green NOSE-L-R down-and-locked lights should illuminate.

i. Before removing the airplane from the jacks, make sure the alternate extension handle is in the stowed position and pinned. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down and locked position.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

j. After the landing gear is in the down and locked position, install lockpins in the landing gear as outlined in Chapters 32-10-00 and 32-20-00.

k. Shut off the air supply and disconnect the air supply line from the air valve. Reconnect the line from the pressure regulator to the air valve.

l. Remove the airplane from the jacks.

m. Disconnect the external power.

MANUAL EXTENSION CHECK (EFFECTIVITY: ALL)

a. Place the airplane on jacks as instructed in Chapter 7-10-00.

b. Refer to Chapter 24-40-00 and connect a 28-vdc external power supply to the airplane.

c. Before operating the landing gear with the hydraulic power pack, check the fluid level in the power pack reservoir and add hydraulic fluid (1, Chart 2, 32-00-00) as required. Refer to Chapter 12-10-00. If the hydraulic fluid level is low, the landing gear should not be operated using the power pack until the fluid level is corrected.

d. Remove the lockpins from the landing gear.

e. Locate the bleed air valve on the power pack, remove the air inlet line from the valve and connect a regulated supply of 18 to 20 psi of dry air to the air valve. The air supply should incorporate a shutoff valve to control the flow of pressurized air to the power pack.

NOTE

If Kit 122-5036-1 is installed, do not disconnect the coupling and restrictor from the bleed air valve. Connect the dry air supply to the restrictor.

CAUTION

To prevent damage to the pump, never operate the power pack without supplying 18 to 20 psi of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly.

f. Press the LDG GR TEST switch on the pilot's outboard subpanel. Make sure that the in-transit lights and gear-down lights illuminate. Wait approximately 30 seconds for the message, HYD FLUID LO, to appear on the EICAS display.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

g. Ensure that the LANDING GEAR CONTROL (CB133) circuit breaker is pushed in, then move the gear handle to the UP position. The red in-transit lights should illuminate in the control switch handle to signify that the gear is in transit or unlocked. When the gear is fully retracted, the red lights should extinguish.

h. Pull the LANDING GEAR CONTROL (CB133) circuit breaker and place the control switch handle in the DN position.

i. Remove the pin and unstow the alternate extension hand pump handle from the securing clip. When the handle is removed from the clip, the following white messages should be displayed on the EICAS display: L GEAR UP, R GEAR UP and NS GEAR UP.

j. Pump the handle up and down to extend the landing gear. Avoid moving the handle down to the stow block while pumping as this will cause the relief valve in the hand pump to open, resulting in a loss of pressure.

k. As the landing gear are released, the following white EICAS messages will replace the GEAR UP messages: L GEAR IN TRANS, R GEAR IN TRANS, and NS GEAR IN TRANS.

l. Continue pumping the alternate extension hand pump until the GEAR IN TRANS messages are no longer displayed and the green NOSE-L-R down and locked lights illuminate.

m. To insure proper operation of the hydraulic system for the subsequent retraction, stow the hand pump handle in the securing clip and install the pin.

n. Push the forward nose doors by hand into the closed-and-locked position. They will be left open after an alternate extension of the landing gear.

o. Engage the LANDING GEAR CONTROL (CB133) circuit breaker.

p. Retract and extend the landing gear with the hydraulic power pack to ensure the landing gear is in the down and locked position.

CAUTION

When cycling the landing gear with the power pack, allow a one-minute cooling period between each cycle and a five-minute cooling period after five cycles.

q. After the landing gear is in the down-and-locked position, install the lockpins in the landing gear as outlined in Chapters 32-10-00 and 32-20-00.

r. Shut off the air supply and disconnect the air supply line from the air valve. Reconnect the line from the pressure regulator to the air valve.

s. Remove the airplane from the jacks.

t. Disconnect the external power from the airplane.

LANDING GEAR CUTOUT RELAY CONTACTS INSPECTION (EFFECTIVITY: ALL)

At the interval specified in Chapter 4-00-00, inspect the landing gear cutout relay contacts as follows:

a. The landing gear cutout relay (K26) is located in the logic relay panel (A129) mounted on the aft fuselage equipment shelf. To gain access to the relay panel, remove the access panel on the LH side of the aft fuselage as outlined in Chapter 53-30-00.

b. Locate and disconnect the plug (P289A) from the relay panel.

c. Connect an ohmmeter (11, Chart 1, 32-00-00) between pin 55 and pin 56 of the relay panel connector.

d. There should be NO continuity between pins 55 and 56.

e. Replace any relay that has continuity between pins 55 and 56. Refer to Chapter 24-50-00 for the relay removal and installation procedures.

HYDRAULIC FLUID LEVEL SENSOR INSPECTION (EFFECTIVITY: ALL)

Inspect the hydraulic fluid low level sensor at the interval specified in Chapter 5-20-00 as follows: Press the LDG GR test switch on the pilot's outboard subpanel and check that the in-transit lights in the landing gear control switch illuminate, the landing gear warning horn sounds, and the HYD FLUID LO message is displayed on the EICAS display within approximately 30 seconds.

WHEELS AND BRAKES - DESCRIPTION AND OPERATION

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing.

When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

MAIN GEAR WHEEL ASSEMBLIES

Two wheels are installed on each main landing gear. (See TIRES in this section for size and type.) Each wheel consists of an inner and outer wheel half. The halves are held together with six bolts and nuts, with one washer under each bolt head and nut to prevent galling and stress concentration. An O-ring mounted in a groove on the inner wheel half seals the wheel against air leakage when the wheel halves are joined together. Three keys are riveted to each inboard wheel half and project into the rotating disc drive slots of the brake assembly. The wheels rotate on tapered roller bearings seated in bearing cups that are shrink-fitted into each wheel half. Grease seals are installed over each wheel bearing. The wheel assemblies are retained on each axle with two washers, a nut, and a lock ring. Three fuse plugs are installed in each inner wheel half to relieve tire inflation pressure in case of heat buildup.

NOSE GEAR WHEEL ASSEMBLY

One wheel is installed on the nose landing gear. (See TIRES in this section for size and type.) The wheel consists of an inner and outer wheel half held together with eight bolts and nuts, with one washer under each bolt head and nut to prevent galling and stress concentration. An O-ring mounted in a groove on the inner wheel half seals the wheel against air leakage when the wheel halves are joined together. The wheel rotates on tapered roller bearings seated in bearing cups that are shrink-fitted into each wheel half. Grease seals are installed over each wheel bearing. The wheel is retained on the axle with two washers, a nut, and a cotter pin.

TIRES

The nose landing gear wheel is equipped with a 19.5 X 6.75-8, 10-ply-rated, tubeless tire. Each main landing gear wheel is equipped with an H19.5 X 6.75-10, 8-ply-rated, tubeless, rim-inflated tire.

BRAKE ASSEMBLY

The airplane is equipped with four hydraulically operated brake assemblies. Each main landing gear incorporates two multi-disc, carbon-lined brake assemblies bolted together, one on each side of the strut. Each brake assembly contains two inlet ports equipped with swivel fittings. The forward swivel fitting is installed to allow installation of a pressure gage for testing the brakes and is plugged during normal operation. The aft swivel fitting is connected to the airplane brake line. The brake assemblies are interchangeable by changing the locations of the plugged swivel fitting and the inlet swivel fitting. A bleeder screw is located adjacent to each inlet port.

Two rotating discs on each brake assembly are keyed to rotate with the wheel. A stationary disc, a pressure plate, and a backplate are attached to the brake housing. Braking action occurs when hydraulic pressure is applied to the five small pistons in the brake housing, forcing the disc stack together and creating friction between the rotating discs and the stationary components of the brake.

HYDRAULIC BRAKE SYSTEM

The dual hydraulic brakes are operated by depressing the toe portion of either the pilot's or copilot's rudder pedals. Refer to Figure 1. The depression of either set of pedals compresses the fluid below the piston rod in the master cylinder attached to each pedal. The hydraulic pressure resulting from the movement of the pistons in the master

cylinders is transmitted through flexible hoses and fixed aluminum tubing to the disc brake assembly on each main landing gear wheel. Braking action occurs when hydraulic pressure at the brake pistons forces the disc stack of each brake assembly together, creating friction between the rotating discs and the stationary components of the brake assembly. Hydraulic fluid is supplied to the master cylinders from a reservoir that is accessible for servicing through the LH forward wing upper fairing.

The system incorporates a number of electrical and hydraulic components. A pallet assembly mounted on the keel in the lower LH nose contains a pump and motor assembly, relief valve, filter, check valve, power brake valve, and high and low pressure switches. Two restrictors and a two-position, three-way-solenoid bypass valve are plumbed between each of the copilot's master cylinders and the power brake valve. When the system is operating in the manual mode, the fluid pressure from the master cylinder is routed by the solenoid bypass valve through one of the restrictors. When the system is operating in the power brake mode or the antiskid and power brake mode, the fluid pressure from the master cylinder is allowed to flow through both of the restrictors. An ANTI SKID OFF switch is mounted on the LH inboard subpanel to serve as the control switch for antiskid operation. A pump motor relay and a two-position, three-way-solenoid valve are mounted above the pallet assembly. An accumulator is mounted just forward of the pallet in the LH nose. An antiskid control box is mounted in the aft wing center section. Wheel speed transducers, one per wheel, are mounted in each main gear axle. Two restrictors and a bypass solenoid valve are plumbed in the system between each copilot's master cylinder and the power brake valve.

The system control circuit is protected by a 5-ampere circuit breaker, placarded ANTI SKID, on the LH circuit breaker panel. The control switch, placarded ANTI SKID-OFF, energizes the antiskid control box when placed in the ANTI SKID position. The antiskid control box contains test logic which detects open or short wheel speed transducer or power brake valve circuits, loss of power from the control switch, and excessive full brake release. If a fault is detected, the antiskid control box sends a signal to the Engine Instrument, Crew Alerting System (EICAS), which displays a yellow ANTI-SKID INOP message. Power for the pump motor is supplied through a 20-ampere limiter, located in the RH nacelle electrical distribution panel, when the pump motor relay is activated. The pump motor relay is grounded through the high pressure switch and the A182 PCB relay. The relief valve is set to act as a safety valve and relieve pump outlet pressure at 2,100 ± 100 psi. The actuation points of the high pressure switch are set to maintain the system pressure between 1,200 ± 50 and 1,500 ± 75 psi. Fluid under pressure from the pump is directed through the solenoid valve to the pressure inlet port of the power brake valve. An accumulator (precharged to 600 ± 50 psi) is plumbed in the line from the pump to the power brake valve to aid in maintaining the system pressure. Anytime the system pressure drops below 650 ± 32 psi, the low pressure switch will provide a ground signal to the brake deice 2-second timer PCB. This illuminates a yellow POWER BRAKE INOP message on the EICAS display, if the ground signal remains for at least two seconds. When the LH forward wing upper fairing is removed, a maintenance switch, mounted on the brake fluid reservoir, provides power from the No. 3 ANNUNCIATOR circuit breaker on the RH circuit breaker panel to open the solenoid valve. This allows maintenance to be performed on the system components.

MANUAL MODE

When the brake system is in the manual mode (master switch in the OFF position or ANTI-SKID circuit breaker pulled) there will be no fluid pressure from the pump at the pressure inlet port of the power brake valve. This allows the pressure generated at the master cylinders to force the shuttle valves in the power brake valve to open. The manual mode is used for braking when the airplane is being towed and no electrical power is applied to the system. Refer to Figure 2.

POWER BRAKE MODE

The power brake mode (control switch set to OFF, and electrical power applied to pump) operates similar to the antiskid and power brake mode except that the antiskid control switch must be set to OFF. No electrical signals will be sent to the skid control servo valve. The system will operate as a power-assisted brake system. This mode is used if the antiskid portion of the system is malfunctioning or if the pilot desires not to use the antiskid and power brake mode. The system will also revert to this mode if the airplane taxiing speed falls below 10 knots. Refer to

Figure 3.

ANTISKID AND POWER BRAKE MODE

With the brake system in the antiskid and power brake mode (control switch set to ANTI-SKID, and electrical power applied to the pump), the pump pressure present at the inlet port of the power brake valve will position the internal shuttle valves to isolate the master cylinder pressure from the brake assemblies and to open a passage for metered pump pressure to the brake assemblies. Master cylinder pressure in the power brake valve acts on the control valve piston to close the return passage and to open the passage for pump pressure into the control valve. The resulting metered pressure is applied to the control valve spool to balance the pressure delivered to the brake assemblies with the master cylinder pressure. As the master cylinder pressure is increased or decreased, the control valve spool moves to allow fluid to escape through the return passage, decreasing the metered pressure, or moves to close the return port, allowing the metered pressure to increase. Refer to Figure 4.

The wheel speed transducers monitor the wheel RPM and relay the information to the antiskid control box to be processed. The antiskid control box sends electrical signals, based on the information provided by the wheel transducers, to the servo valve mounted on the power brake valve. The servo valve responds to the input from the antiskid control box by varying its control pressure. The control pressure is created in the cavity of the servo valve between the system pressure nozzle and the system return nozzle. Signals from the antiskid control box regulate the opening and closing of the two nozzles. When sensed rotational speed of the wheels drops below 10 knots, the control signals to the servo valve will drop out. When no electrical signal from the antiskid control box is present, the pressure nozzle is open and the return nozzle is closed; therefore, the control pressure of the servo valve is the same as the pump pressure and has no effect on the operation of the control valve.

At taxi speeds above 10 knots, when a skid condition is imminent, the appropriate signal is sent to the servo valve and the servo valve control pressure is reduced as the pressure nozzle is closed. When the control pressure is reduced, pump pressure at the opposite end of the control valve spools moves the spools to open the return passages, allows fluid in the control valves to bleed off, and results in a reduced pressure to all four brake assemblies. As the wheel RPM begins to increase, the servo valve control pressure increases, resulting in an increased brake pressure. If the wheel transducers indicate that the wheels are still in a skid condition, the brake pressure will be decreased again. This cycle will continue until the tendency for the wheels to skid has ended. Differential braking between the LH and RH master cylinders will be maintained while the antiskid function is operating.

PARKING BRAKE VALVE

The parking brake valve is plumbed in the lines between the power brake valve and the wheel brakes. The lever-type valve is mounted on the LH keel under the floorboard below the pilot's rudder pedals. A push-pull cable from the valve control lever is routed to the pedestal, terminating with a knob. After the brake pedals have been depressed to build up pressure in the brake lines, the valve can be closed by pulling out the parking brake knob on the pedestal. This closes the valve to retain the pressure that was previously pumped into the brake lines. The parking brake is released when the rudder pedals are pressed briefly to equalize the pressure on both sides of the valve and the parking brake knob is pushed in.

CHART 1
TROUBLESHOOTING - WHEELS AND BRAKES

Step 1	Solid pedal and brakes?	**YES**	Proceed to Step 2.
	NO		
Step 2	Brakes have no spongy feeling?	**YES**	Proceed to Step 3.
	NO		

CHART 1
TROUBLESHOOTING - WHEELS AND BRAKES (CONTINUED)

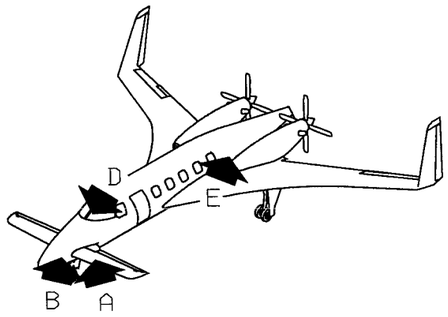
Step 3	Brakes able to hold pressure?	**YES**	Proceed to Step 4.
	NO		
	Check system for leaks or scored master cylinder.		
Step 4	Does the parking brake hold.	**YES**	Proceed to Step 5.
	NO		
	Bleed the brakes or replace the parking brake valve.		
Step 5	Brakes are free from grabbing.	**YES**	Proceed to Step 6.
	NO		
	Check for stones or foreign matter locking brake disc. Wash brakes.		
Step 6	"POWER BRAKE INOP" displayed?	**YES**	Proceed to Step 7.
	NO		
	Proceed to Step 14.		
Step 7	Are lines and seals free of leaks?	**YES**	Proceed to Step 8.
	NO		
	Tighten or replace lines or seals		
Step 8	Accumulator has 600 ± 50 psi?	**YES**	Proceed to Step 9.
	NO		
	Charge accumulator.		
Step 9	Is power brake valve functioning properly	**YES**	Proceed to Step 10.
	NO		
	Repair or replace power brake valve.		
Step 10	Is solenoid valve functioning properly?	**YES**	Proceed to Step 11.
	NO		
	Repair or replace solenoid valve.		
Step 11	Are pump and motor functioning properly?	**YES**	Proceed to Step 12.
	NO		
	Repair or replace pump or motor.		
Step 12	Is the pump motor relay functioning properly?	**YES**	Proceed to Step 13.
	NO		
	Replace pump motor relay.		

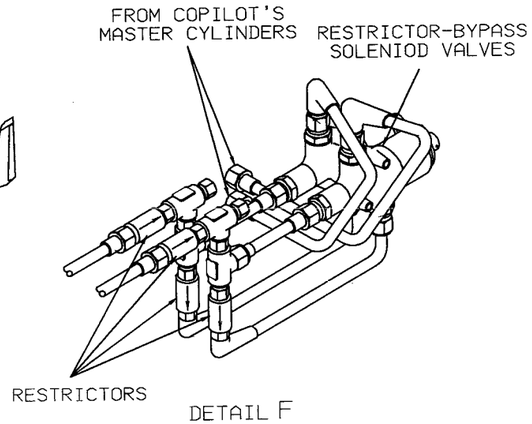
CHART 1
TROUBLESHOOTING - WHEELS AND BRAKES (CONTINUED)

Step 13	Are the pressure switches functioning properly?	**YES**	Proceed to Step 14.
	NO		
	Replace faulty switch.		
Step 14	"ANTI-SKID INOP" displayed.	**YES**	Proceed to Step 15.
	NO		
	Proceed to step 22.		
Step 15	Are the wheel speed transducers operational?	**YES**	Proceed to Step 16.
	NO		
	Replace faulty wheel speed transducers.		
Step 16	Is input power correct to control box per BEECH Starship 1 Wiring Diagram Manual?	**YES**	Proceed to Step 17.
	NO		
	Repair faulty wiring.		
Step 17	Electrical circuits OK in servo valve or transducer?	**YES**	Proceed to Step 18.
	NO		
	Repair faulty electrical circuit.		
Step 18	Tires showing flat spotting?	**YES**	Check amount of wear per Beech Starship Component Maintenance Manual. Replace control box.
	NO		
	Proceed to Step 20.		
Step 19	Control valve shows signs of internal contamination.	**YES**	Clean control valve per Beech Starship 1 Component Maintenance Manual. Replace system filter element.
	NO		
	Proceed to Step 20		
Step 20	Does parking brake open fully?	**YES**	Proceed to Step 21.
	NO		
	Adjust parking brake cable.		
Step 21	Brakes show loss or reduction of braking at low speeds?	**YES**	Recheck wheel speed transducers.

CHART 1
TROUBLESHOOTING - WHEELS AND BRAKES (CONTINUED)

	NO		
	Proceed to Step 22.		
Step 22	Is "ANTI-SKID INOP" still displayed.	**YES**	Replace control box.
	NO		
	Perform Brake System Functional Check		





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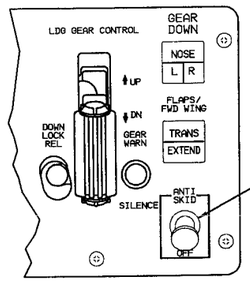
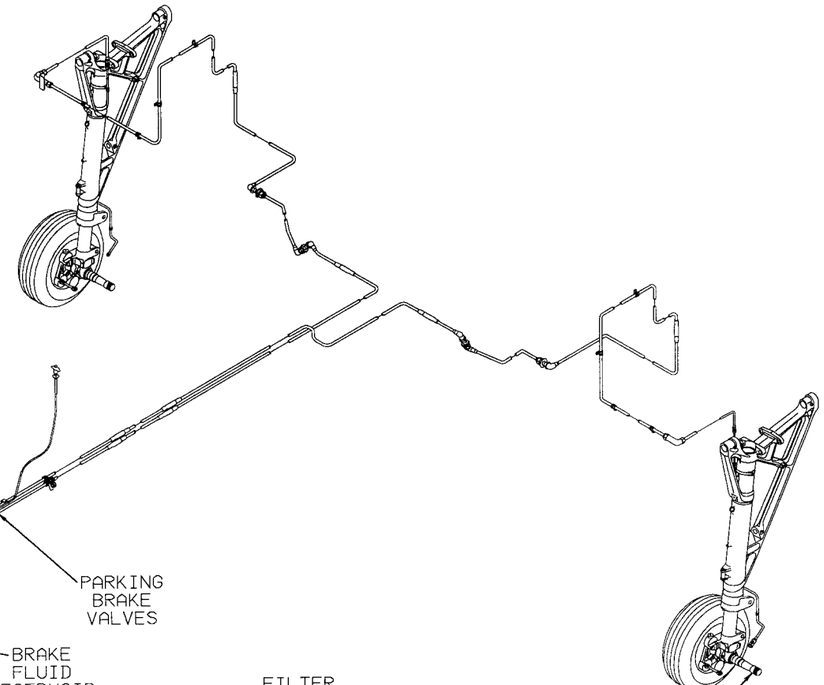
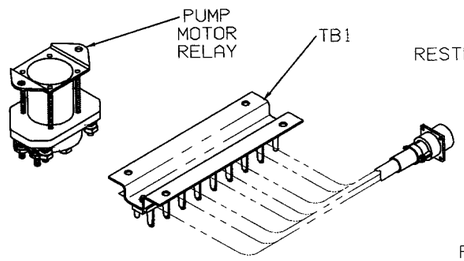


ANTISKID-OFF SWITCH

PILOT'S INBOARD SUBPANEL

DETAIL D





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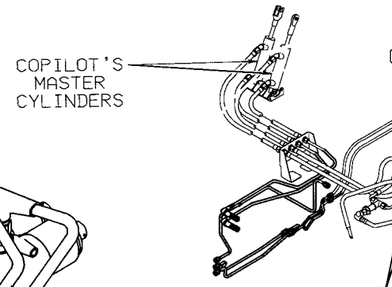

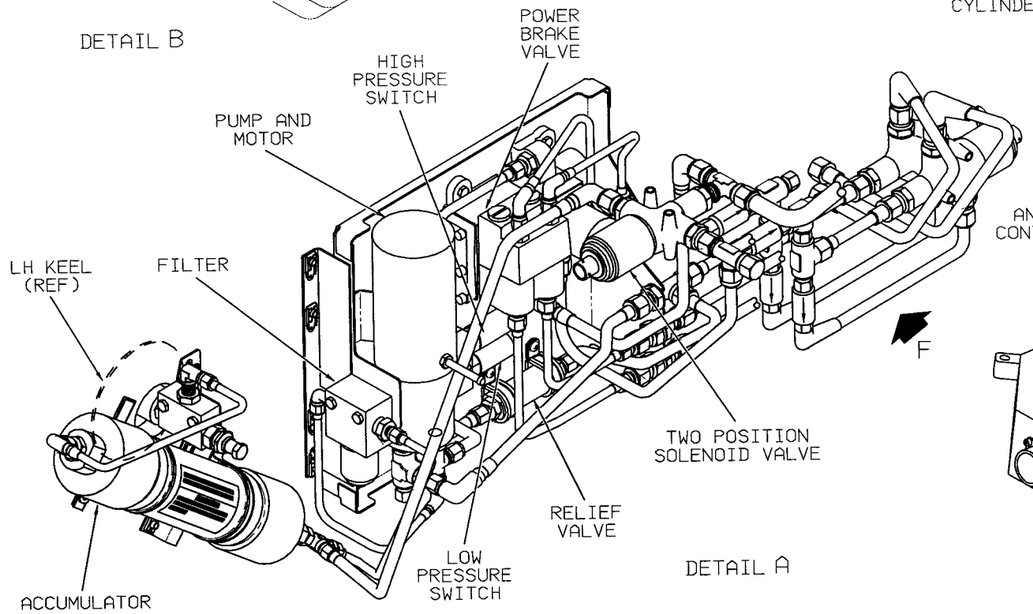

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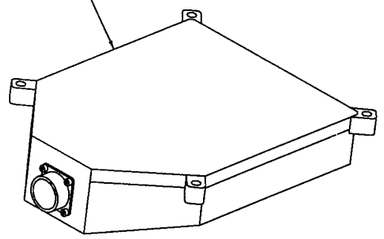

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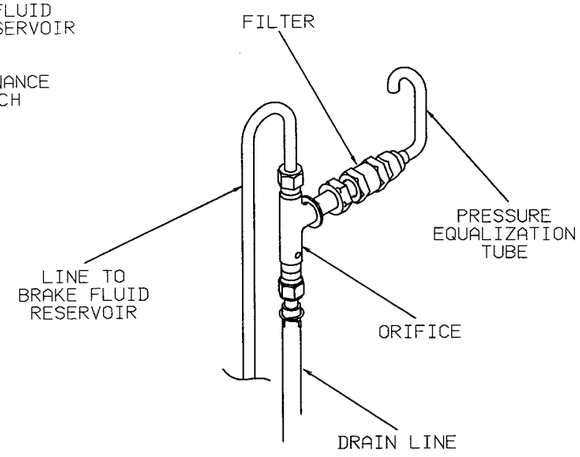

DETAIL C

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Brake System Locations
Figure 1

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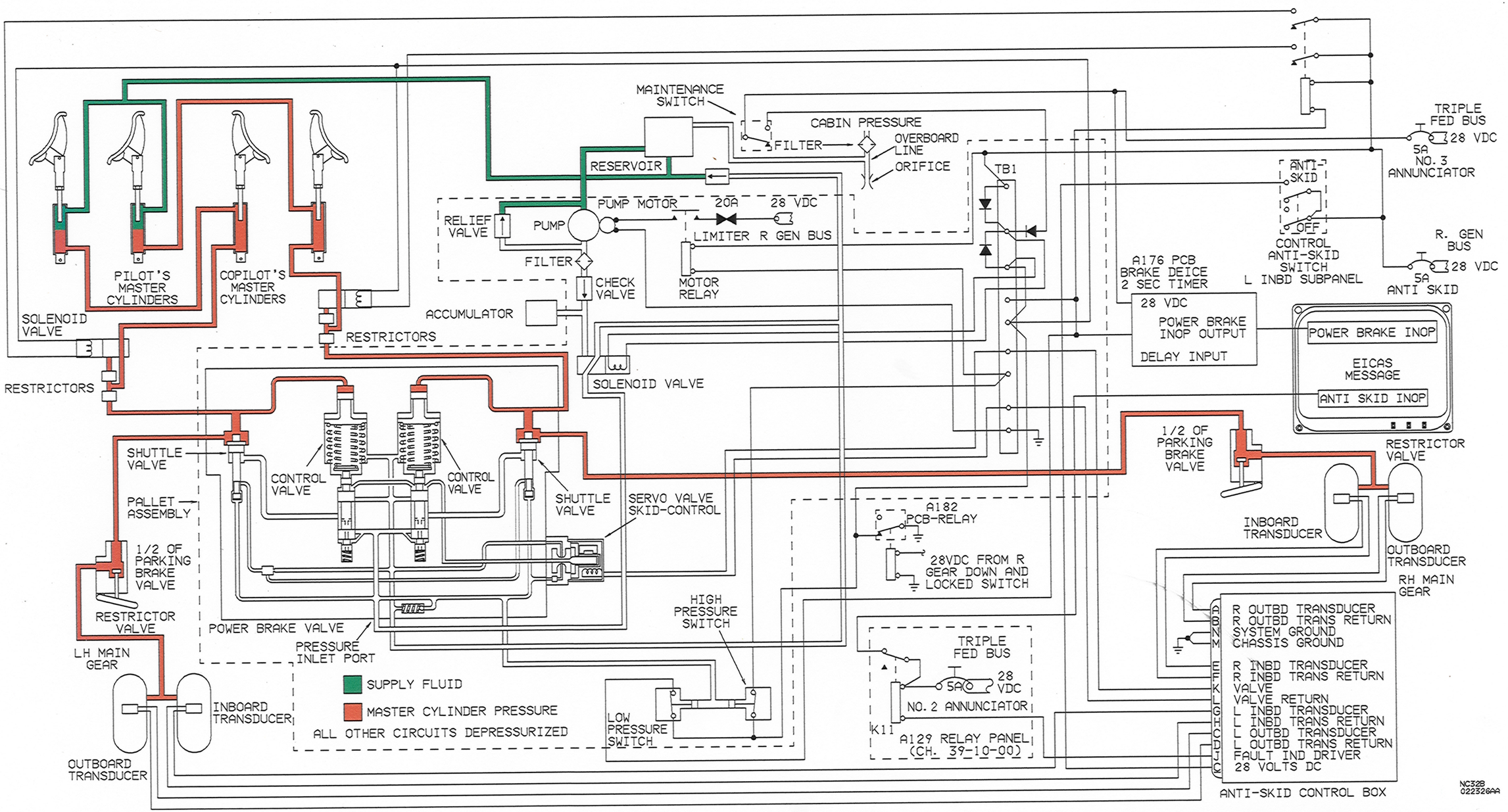

Brake System Schematic (Manual Mode)
Figure 2

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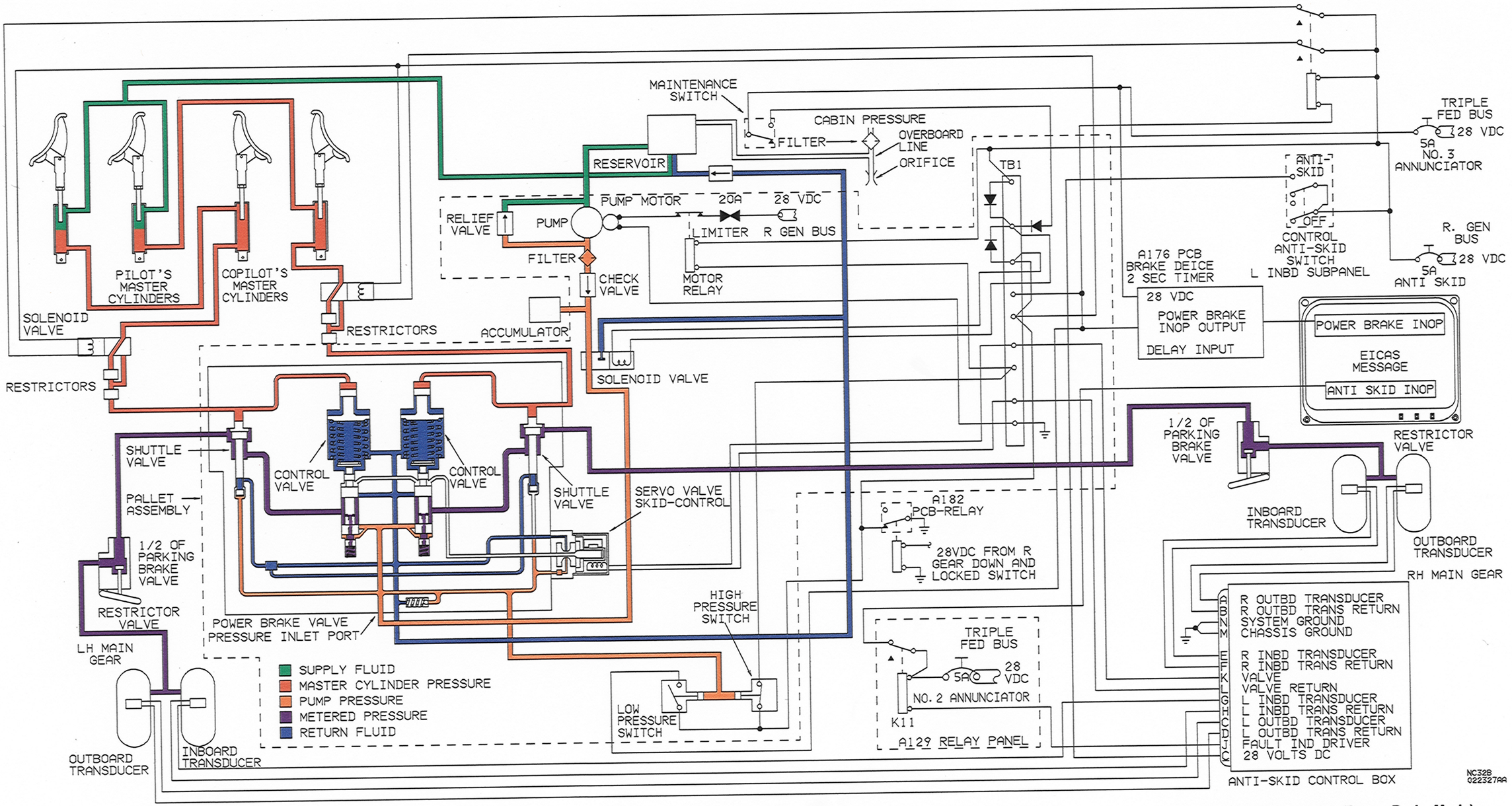

Brake System Schematic (Power Brake Mode)
Figure 3

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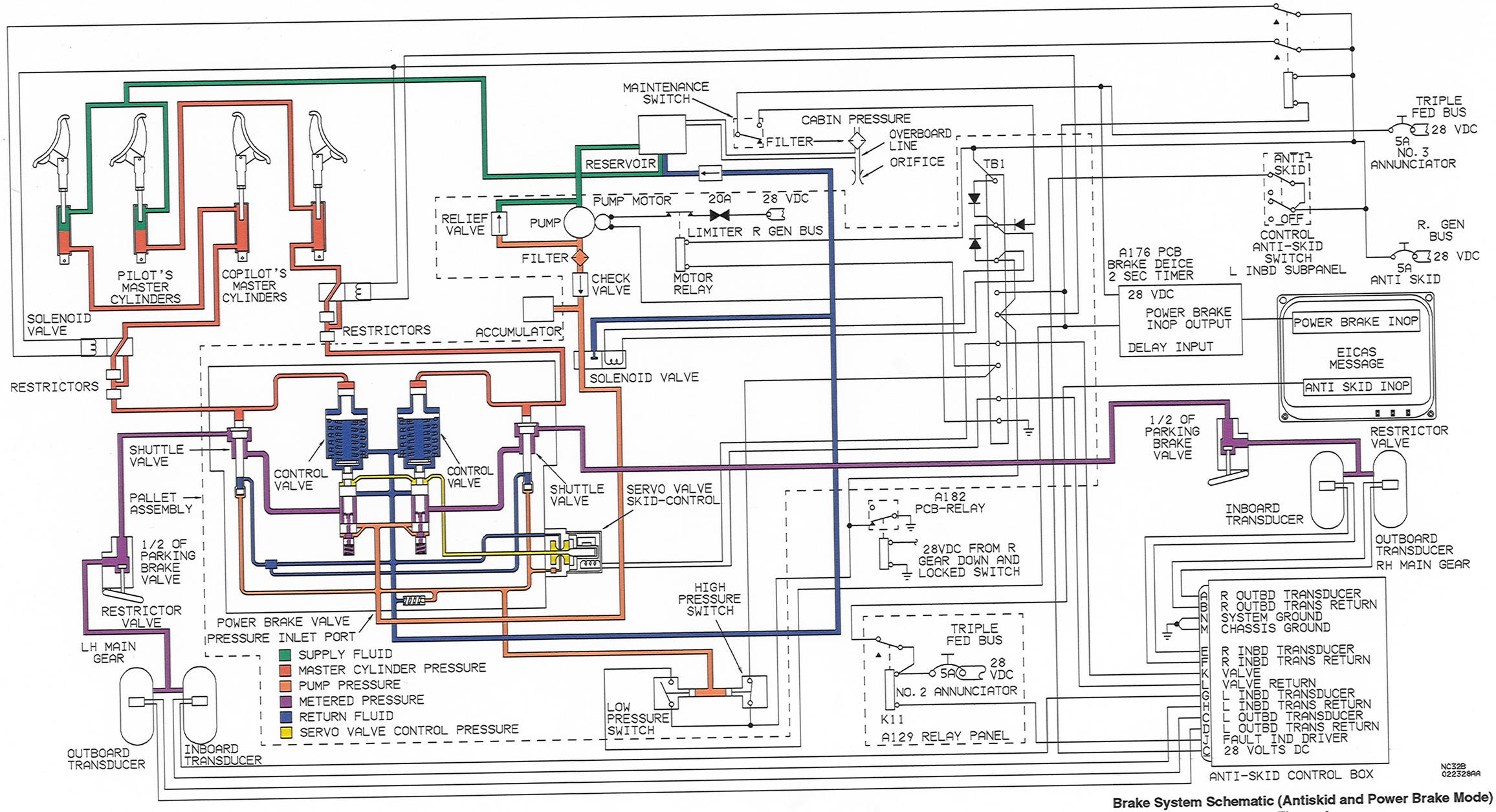

Brake System Schematic (Antiskid and Power Brake Mode)
Figure 4

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WHEELS AND BRAKES - MAINTENANCE PRACTICES

WHEELS AND BRAKES SERVICING

For detailed information relating to the proper inspection and repair procedures for the wheels and brakes refer to the Beech Starship 1 Component Maintenance Manual.

TIRE SERVICING

Refer to Chapter 12-10-00 for the tire inflation pressures and all servicing of the tires.

ANTI-SKID COMPONENTS SERVICING

Refer to the Beech Starship 1 Component Maintenance Manual for detailed information relating to the proper inspection and repair procedures for the antiskid control box, the wheel speed transducers, and the power brake valve.

BRAKE WEAR LIMITS

To check the total disc wear, measure the distance from the end of the wear indicator tube to the brake housing as shown in Figure 5. If the distance is 0.207 inch or less with normal brake pressure applied from the master cylinder, the brake assembly should be removed from the landing gear and the individual discs checked for wear as outlined in the Beech Starship 1 Component Maintenance Manual.

PRESSURE EQUALIZATION LINE CLEANING

The brake fluid reservoir is plumbed so that cabin air is directed into the top of the reservoir through the pressure equalization line. A filter is located in the pressure equalization line just aft of the forward pressure bulkhead. An orifice is located in the bottom of the tee just forward of the forward pressure bulkhead. Cabin air contaminants (fuzz, lint, nicotine, tar, etc.) can find their way into the pressure equalization line and eventually block the filter and the orifice so that excessive air is diverted into the brake fluid reservoir, causing excessive pressure to be applied to the master cylinders. The filter is to be removed and replaced and the orifice cleaned after the first 100 hours. At the intervals specified in Chapter 5-20-00, the filter and orifice are to be cleaned. Clean the filter and orifice as follows:

a. Working inside the cockpit, disconnect the pressure equalization tube from the filter, remove the filter from the tee at the forward pressure bulkhead, and reconnect the pressure equalization tube to the tee. Refer to Figure 1.

b. Clean the filter with solvent (3, Chart 2, 32-00-00) and blow dry with compressed air.

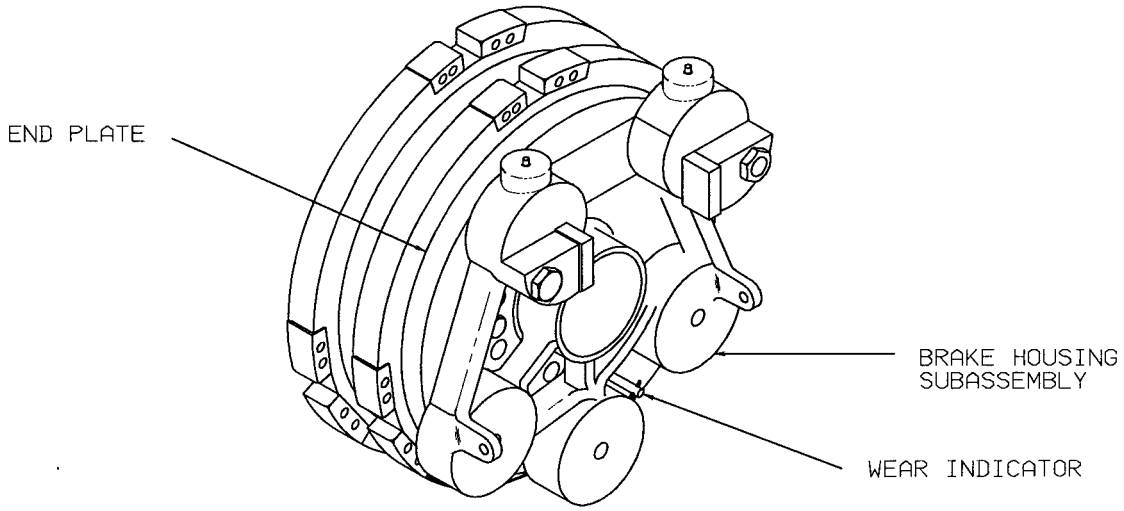
c. Disconnect the line from the brake fluid reservoir to the pressure equalization line tee. Cap the top of the tee in the pressure equalization line.

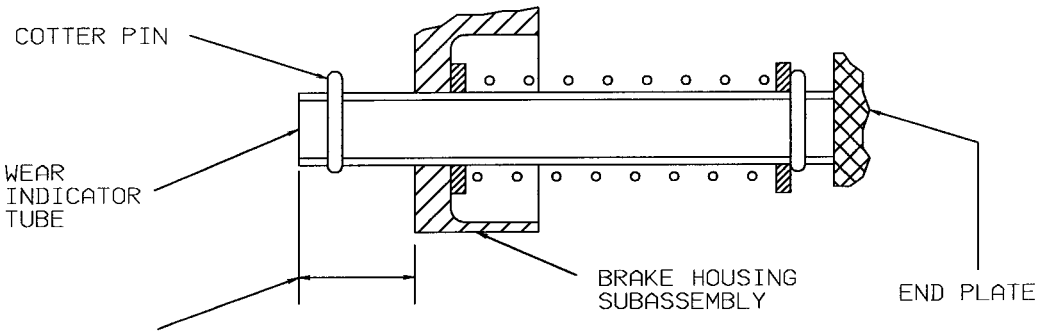
d. Working inside the cockpit, locate the pressure equalization line tee that passes through the forward pressure bulkhead. Select a length of rubber tubing that will fit tightly over the end of the pressure equalization tube in the tee; slide one end of the tubing over the open end of the tube in the fitting.

e. Inject a small amount of solvent (3, Chart 2, 32-00-00) into the rubber tubing and blow out the line with low air pressure. Inspect the solvent flowing from the bottom drain for cleanliness. Repeat as necessary until clean solvent flows from the bottom drain.

f. After clean solvent flows from the bottom drain, blow out the line thoroughly with low air pressure.

g. Working inside the cockpit, remove the rubber tubing from the pressure equalization line tube. Remove the pressure equalization line tube, then install the filter and the pressure equalization tube.





WHEN THE BRAKE IS APPLIED AND THE END OF THE WEAR INDICATOR TUBE IS .207 INCH OR LESS FROM THE BRAKE HOUSING SUBASSEMBLY, THE BRAKE SHOULD BE REMOVED FOR OVERHAUL.

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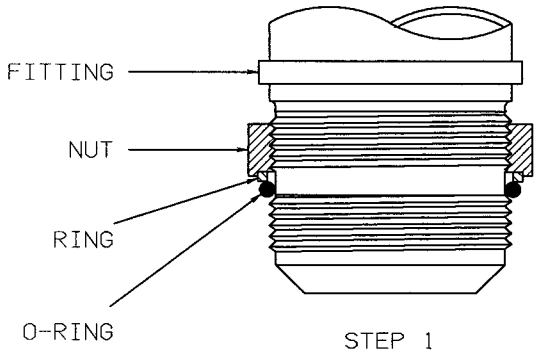
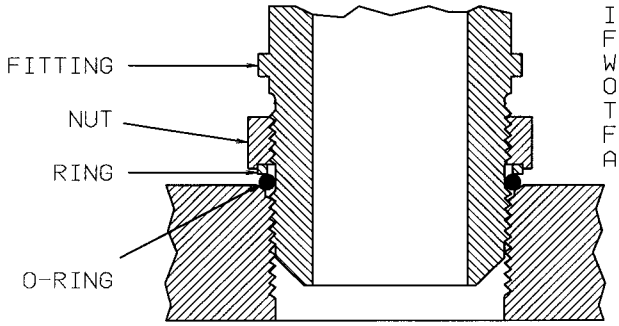
Break Wear Limits
Figure 5



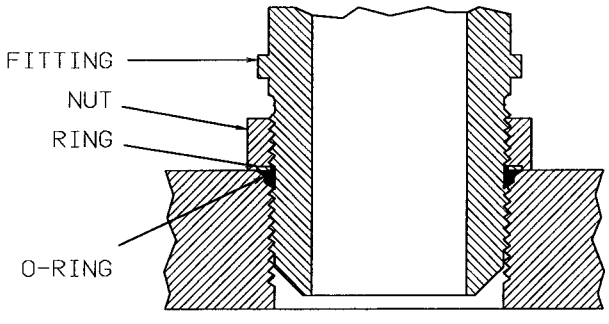
STEP 1

COAT THE RING, O-RING, AND THE MALE THREADS OF THE FITTING WITH HYDRAULIC FLUID (1, CHART 2, 32-00-00) AND ASSEMBLE AS SHOWN. WORK THE RING INTO THE COUNTERBORE OF THE NUT AND TURN THE NUT DOWN UNTIL THE O-RING IS PUSHED FIRMLY AGAINST THE LOWER THREADED PORTION OF THE FITTING.



STEP 2

INSTALL THE FITTING, TURNING THE NUT WITH THE FITTING, UNTIL THE O-RING CONTACTS THE SURFACE. WITH THE FITTING IN THIS POSITION, PUT A WRENCH ON THE NUT TO PREVENT IT FROM TURNING, AND TURN THE FITTING IN 1-1/2 TURNS. POSITION THE FITTING BY TURNING IT IN NOT MORE THAN ONE ADDITIONAL TURN.



STEP 3

WHILE HOLDING THE FITTING TO PREVENT MOVEMENT, TURN THE NUT DOWN AGAINST THE SURFACE. SLIGHT EXTRUSION OF THE RING IS PERMISSIBLE.

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Hydraulic Fitting Installation
Figure 6

h. Uncap the tee and reconnect the line from the brake fluid reservoir to the top of the tee.

BRAKE SYSTEM HYDRAULIC FITTINGS INSTALLATION

To reduce the possibility of leaks in the hydraulic system, care should be taken when installing the hydraulic fittings. Anytime a fitting is loosened or removed, discard the O-ring and install the fitting with a new O-ring. To prevent damage to the O-ring, coat the O-ring and the threads of the fitting with hydraulic fluid (1, Chart 2, 32-00-00) before installing the O-ring on the fitting. Some fittings in the brake system employ both an O-ring and another ring. When this type of fitting is loosened or removed, both the O-ring and the other ring must be replaced when the fitting is installed. Figure 6 illustrates the correct method of installing these fittings.

BRAKE SYSTEM BLEEDING

Brake system bleeding will be required whenever the system is opened at any point between the master cylinder and the wheel brake assembly, whenever the brakes become spongy in service, or whenever the parking brakes will no longer hold. In the latter instance, the system should be further checked for leakage.

If it is suspected that air is trapped in the brake housing, remove the brake assembly from the airplane and lay it on its side. Add brake fluid as needed through the bleed port and tap the brake assembly lightly with a rubber hammer to dislodge any air bubbles. When air bubbles no longer appear at the port, reinstall the brake assembly and bleed the brake system.

CAUTION

If brakes feel soft or "spongy" or exhibit asymmetric or "jerky" operation, it is likely that air is trapped in the system. The air should be purged using the following procedures. Failure to correct this condition could result in reduced directional control and reduced braking response, which could be hazardous.

BRAKE SYSTEM GRAVITY BLEEDING

This method of bleeding is done from the master cylinders down to the brake assemblies. The brake fluid reservoir must be kept full during the bleeding operation. Since the pilot's and copilot's master cylinders are plumbed in series, the system may be bled by operating the pilot's rudder pedals only. Bleed the brakes in the following manner:

NOTE

The parking brake valve must be released while bleeding the brake system.

a. Connect an external power supply to the airplane and adjust the voltage to 28 ±0.25 vdc.

b. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.
2. Turn the master switch ON.
3. Ensure that the parking brake is off.
4. Refer to Chapter 53-50-00 to remove the LH forward wing upper fairing (211BL).
5. Engage the No. 3 ANNUNCIATOR circuit breaker.

6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Check the brake reservoir fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required to obtain a full reading.

d. Open the No. 3 ANNUNCIATOR circuit breaker.

e. Cut the safety wire and remove the screws from the forward bleeder ports of both brake assemblies on one main landing gear. Install a bleeder hose adapter (5, Chart 1, 32-00-0) into each brake bleeder port. Fabricate a bleeder hose assembly (6, Chart 1, 32-00-00) and connect the bleeder hose assembly to the bleeder hose adapters. Refer to Figure 8.

f. Connect 25 feet of tygon tubing with an inline filter to the bleeder hose assembly. Remove the brake fluid reservoir cap and place the end of the tygon tubing into the reservoir.

g. Pump the pilot's corresponding brake pedal a minimum of 25 to 30 times, then continue to pump the brake pedal until the fluid flowing from the brake assemblies is free of air bubbles.

h. Engage the ANTI SKID and No. 3 ANNUNCIATOR circuit breakers and allow the system to normalize.

i. Pump pilot's brake pedal slowly and smoothly 15 to 20 times to allow the system pump to slowly purge all the remaining air from that side of the brake system. When the brake fluid returning to the reservoir is free of air, open the ANTI SKID and No. 3 ANNUNCIATOR circuit breakers.

j. Disconnect the tygon tubing from the bleeder hose assembly and remove the bleeder hose assembly from the bleeder port adapters. Remove the bleeder port adapters from the forward bleeder ports and install the bleeder port screws.

k. Loosen the aft bleeder port screws of the brake assemblies just bled.

l. Depress the pilot's corresponding brake pedal slowly and smoothly to eliminate air trapped in the system.

m. Hold the brake pedal in the depressed position and close the aft bleeder port screws.

n. Release the brake pedal.

o. Repeat steps "k" through "n" until the fluid flowing from the brake assemblies is free from air bubbles.

p. Tighten the aft bleeder port screws and safety wire.

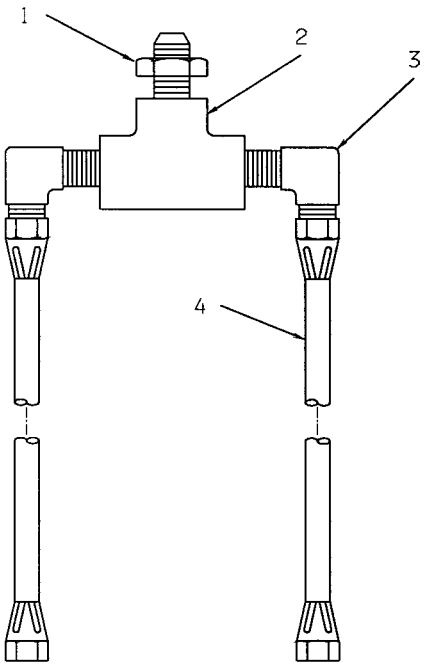
q. To bleed the other side of the system, repeat steps "b" through "p" on the other brake assemblies.

r. Refer to step "b" and remove brake fluid pressure from accumulator.

s. Check the brake reservoir fluid level and add hydraulic fluid (1, Chart 2, 32-00-00) as required to obtain a full reading.

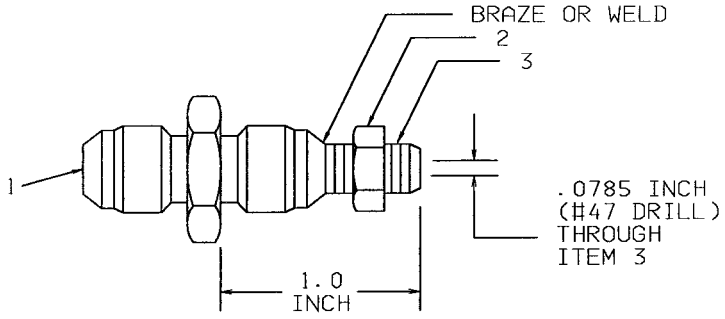
t. Refer to Chapter 53-50-00 and install the LH forward wing upper fairing (211BL).

u. Pull circuit breakers, check the brakes for proper operation. When the brake pedals are depressed, there should be no spongy feeling and the pedal pressure should be equal for both brakes.



BLEEDER HOSE ASSEMBLY
PARTS LIST

ITEM	QTY	DESCRIPTION
1	1	AN816-4 UNION
2	1	AN917-1D TEE
3	2	AN822-4D ELBOW
4	2	250 INCH FLEX HOSE WITH END FITTINGS TO MATCH ITEM 3 AND BLEEDER HOSE ADAPTER (LENGTH OPTIONAL)



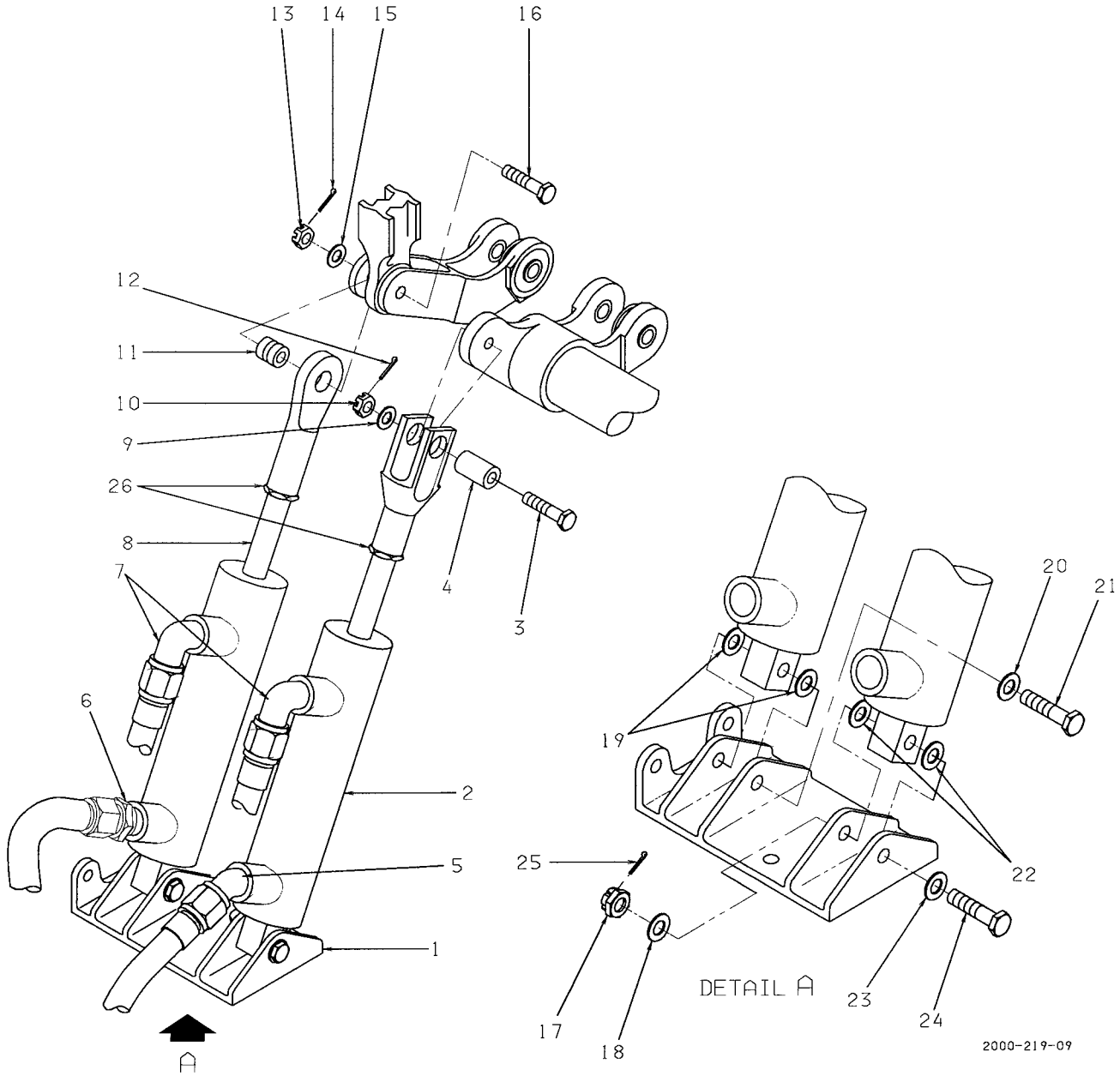
TK889-2 D928
BLEEDER HOSE ADAPTER
PARTS LIST

ITEM	QTY	DESCRIPTION
1	1	AN815-4 UNION
2	1	10-32 HEX NUT
3	1	10-32 X .75 SOCKET HEAD CAP SCREW

2000-218-09

Bleeder Hose Assembly and Adapter
Figure 7

1. ATTACH PLATE
2. OUTBOARD MASTER CYLINDER
3. BOLT
4. BUSHING
5. OUTBOARD LOWER FITTING
6. INBOARD LOWER FITTING
7. UPPER FITTINGS
8. PISTON SHAFT
9. WASHER
10. NUT
11. BUSHING
12. COTTER PIN
13. NUT
14. COTTER PIN
15. WASHER
16. BOLT
17. NUT
18. WASHER
19. WASHERS
20. WASHER
21. BOLT
22. WASHERS
23. WASHER
24. BOLT
25. COTTER PIN
26. LOCKNUT



Master Cylinder Installation
Figure 8

BRAKE MASTER CYLINDER REMOVAL

a. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.

2. Turn the master switch ON.

3. Ensure that the parking brake is off.

4. Remove the LH forward wing upper fairing (211BL) (refer to Chapter 53-50-00).

5. Engage the No. 3 ANNUNCIATOR circuit breaker.

6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

b. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

c. Refer to Chapter 53-20-00 to remove the floorboards (141AT) from around the master cylinder.

d. Disconnect hydraulic lines at the fittings (5 and 7, or 6 and 7) from the master cylinder. Refer to Figure 8. Discard old O-rings.

CAUTION

As the hydraulic lines are disconnected, plug or cap the openings to prevent entry of foreign material into the lines or master cylinder.

e. If removing inboard master cylinder, cut and remove safety wire on bolt (21) on lower end of master cylinder.

f. Remove the master cylinder cotter pins (12 and 25, or 14), nuts (17 and 10, or 13), attaching bolts (3 and 24 or 16 and 21), bushings (4 or 11) and washers (18, 22 and 9 or 20, 19 and 15) at the upper and lower end of the master cylinder.

g. Remove the master cylinder from the airplane.

BRAKE MASTER CYLINDER INSTALLATION

a. Place the brake master cylinder in position and secure the lower end with the attaching bolts (21 or 24), washers (18 and 22 or 19 and 20), nut (17) and cotter pin (25). Refer to Figure 8.

b. Adjust the brake master cylinder linkage as instructed in BRAKE MASTER CYLINDER LINKAGE ADJUSTMENT.

c. Place the upper end of the brake master cylinder in position on the rudder pedal and install the attaching bolts (3 or 16), washers (9 or 15), nuts (10 or 13) and cotter pins (12 or 14).

d. If installing inboard master cylinder, safety wire bolt (21) on lower end of master cylinder.

e. Install new O-rings and connect the hydraulic lines at the fittings (5 and 7 or 6 and 7) to the master cylinder.

f. Bleed the brake system as described in BRAKE SYSTEM BLEEDING in this chapter.

g. Refer to Chapter 53-20-00 and replace the floorboards (141AT).

h. Refer to Chapter 53-50-00 and reinstall LH forward wing upper fairing (211BL).

i. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE MASTER CYLINDER LINKAGE ADJUSTMENT

a. Remove the cotter pin (12 or 14), nut (10 or 13), bolt (3 or 16) and washer (9 or 15) attaching the upper end of the master cylinder clevis to the rudder pedal. Slide the clevis free of the lug on the rudder pedal. Refer to Figure 8.

b. Loosen the locknut (26) on the piston shaft (8).

c. The length of the cylinder piston can be adjusted by turning the clevis on or off the piston shaft. Adjust the clevis so that the distance between the centerline of the clevis hole and the centerline of the master cylinder bottom attach hole is 9 inches. The length of the shaft should be adjusted so that the piston does not "bottom" when the brake is applied.

NOTE

The above adjustment must be maintained for correct operation of the master cylinder. Do not re-adjust the overall dimension between centers of the clevis hole and the master cylinder mounting pivot hole to accommodate physical stature of operating personnel.

d. Tighten the locknut (26) on the piston shaft (8).

e. Move the master cylinder clevis into position to mate with the lug on the rudder pedal and install the bolt (3 or 16), washer (9 or 15), nut (10 or 13) and cotter pin (12 or 14).

f. Check the upper and lower attachment points of the master cylinder for possible interference with the attachment lugs on the rudder pedal. Adjust alignment of clevis with attachment lug if necessary.

PARKING BRAKE VALVE REMOVAL

a. Remove brake fluid pressure from accumulator as follows:

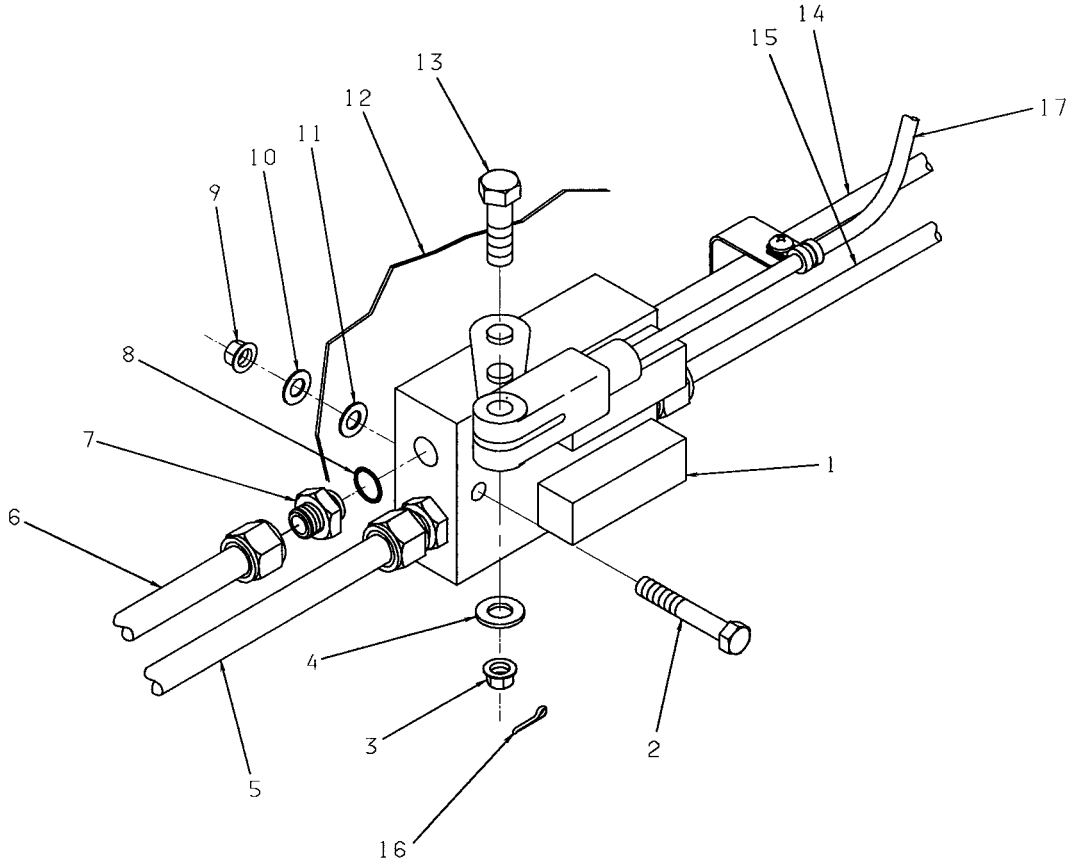
1. Open the ANTI SKID circuit breaker.
2. Turn the master switch ON.
3. Ensure that the parking brake is off.
4. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).
5. Engage the No. 3 ANNUNCIATOR circuit breaker.
6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

b. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

c. Refer to Chapter 53-20-00 and remove the floorboard (141AT) below the pilot's rudder pedals.

d. Remove the cotter pin (16), nut (3), washer (4) and bolt (13) connecting the control cable to the valve assembly. Refer to Figure 9.

1. PARKING BRAKE VALVES
2. BOLT
3. NUT
4. WASHER
5. RH BRAKE LINE
6. LH BRAKE LINE
7. REDUCER
8. O-RING
9. NUT
10. WASHER
11. WASHER
12. KEEL
13. BOLT
14. LH BRAKE LINE
15. RH BRAKE LINE
16. COTTER PIN
17. CONTROL CABLE



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Parking Break Valve Installation
Figure 9

e. Disconnect the hydraulic brake lines (5, 6, 14 and 15) from the valve. Discard old O-rings.

CAUTION

As the hydraulic lines are disconnected, plug or cap the openings to prevent entry of foreign material into the lines or valve.

f. Remove the nuts (9), washer (10 and 11) and bolts (2) securing the valve (1) to the keel (12).

g. Remove the valve (1) from the airplane.

PARKING BRAKE VALVE INSTALLATION

a. Position the valve (1) in the airplane and install the bolts (2), washers (10 and 11) and nuts (9) securing the valve (1) to the keel (12). Refer to Figure 9.

b. Attach the control cable to the valve (1) with bolt (13), washer (4), nut (3) and cotter pin (16). Tighten nut only enough to remove end play.

c. Install new O-rings (8) and connect the hydraulic brake lines (5, 6, 14 and 15) to the valve (1).

d. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

e. Adjust the parking brake control cable as instructed under PARKING BRAKE CONTROL CABLE ADJUSTMENT in this chapter.

f. Refer to Chapter 53-20-00 and replace the floorboard (141AT).

g. Refer to Chapter 53-50-00 and install the LH forward wing upper fairing (211BL).

h. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

PARKING BRAKE CONTROL CABLE ADJUSTMENT

NOTE

Overhaul of the parking brake valve is not recommended. If leakage is encountered, replace the valve.

To ensure proper operation of the parking brake valve, it is important that the full travel of the valve be maintained. Adjust the control cable as follows:

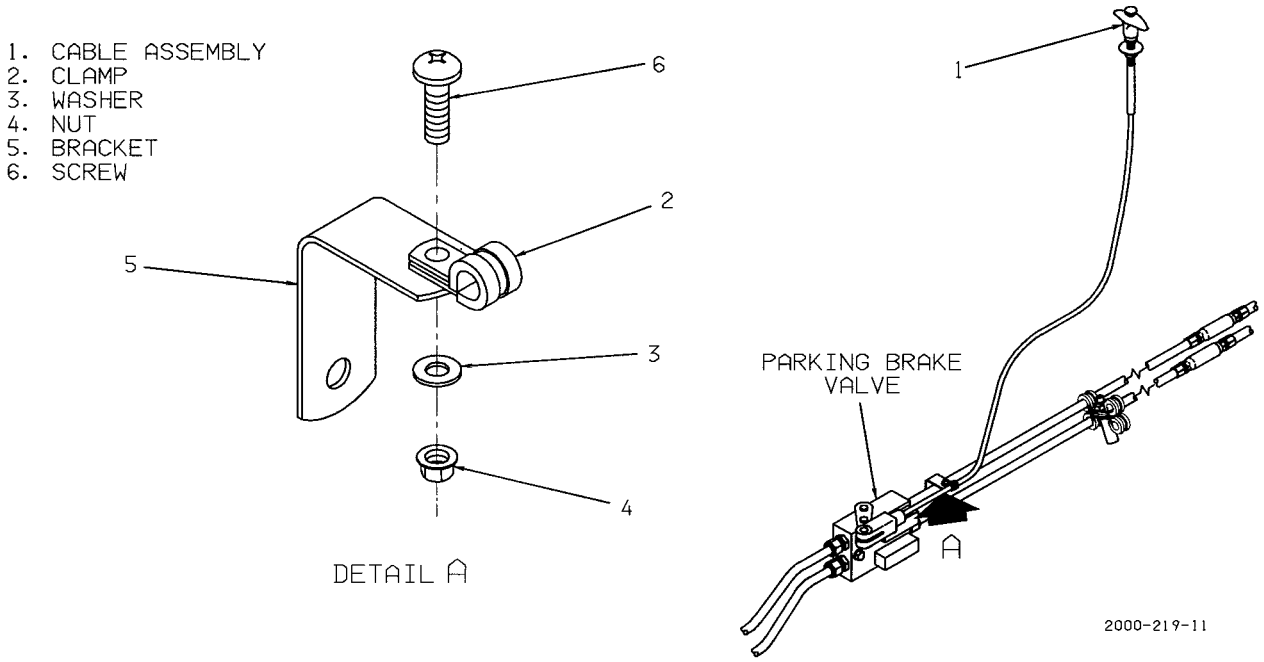
a. Refer to Chapter 53-20-00 and remove the floorboard (141AT) below the pilot's rudder pedals.

b. Loosen the nut (4). Position the outer housing of the cable (1) in the clamp (2) until pushing in on the control knob fully opens the valve. Tighten the nut (4). Refer to Figure 10.

c. Pull the control knob out and check that the valve closes fully.

d. If necessary, loosen the nut (4) and reposition the outer housing of the cable (1) in the clamp (2) until the valve opens and closes fully.

e. Refer to Chapter 53-20-00 and replace the floorboard (141AT).



Parking Brake Control Cable Adjustment
Figure 10

NOSE WHEEL REMOVAL

a. Place the airplane on jacks as described in Chapter 7-10-00.

b. Completely deflate the tire before removing the axle nut (2). Refer to Figure 11.

c. Remove the cotter pin (1), axle nut (2), key washer (3) and washer (4).

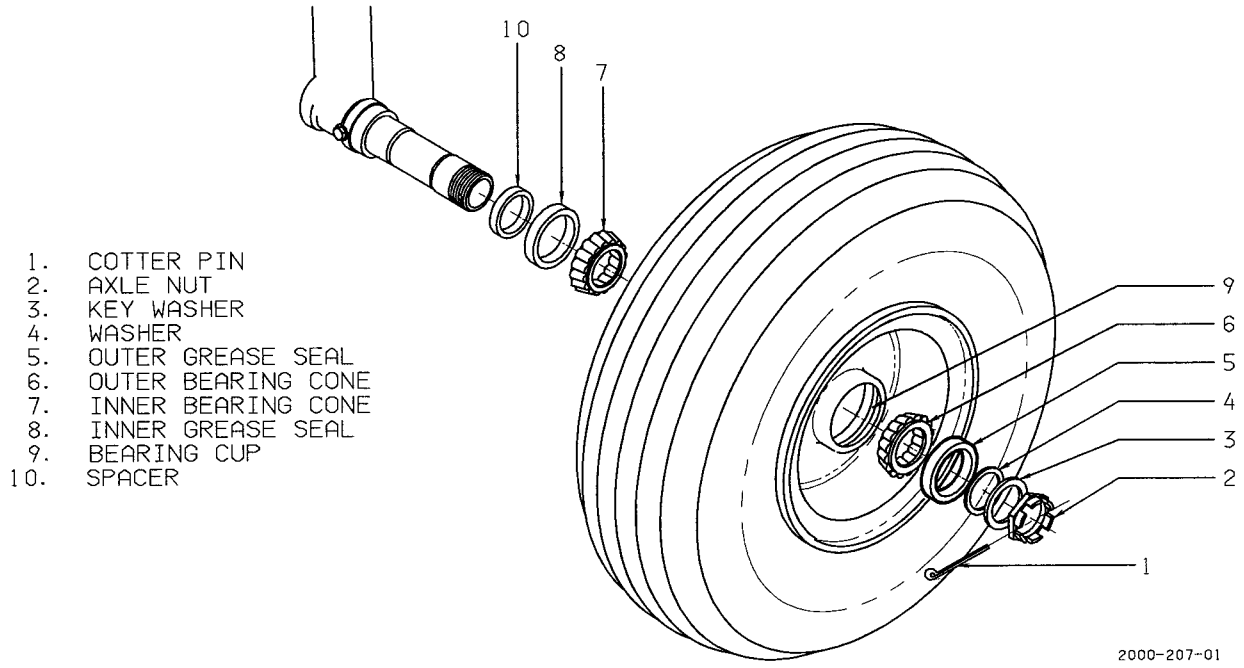
CAUTION

Use care when removing all internal and external nose wheel components to avoid damaging the axle surface and/or nicking the axle threads.

d. Remove the outer grease seal (5) and bearing cone (6), then remove the wheel and tire from the axle.

e. Remove the inner grease seal (8), inner bearing cone (7) and spacer (10) from the axle.

f. Refer to the Raytheon Aircraft Beech Starship 1 Component Maintenance Manual for further maintenance procedures on the wheel and tire.



Nose Wheel Installation
Figure 11

NOSE WHEEL INSTALLATION

a. Pack grease (8, Chart 2, 32-00-00) into the bearing cones (6 and 7) and smear grease on the ends of the rollers. Do not over lubricate. Spread a thin coat of bearing grease on the surface of the bearing cups (9). Refer to Figure 11.

b. Check the axle and nut for burrs or rough threads.

c. Install the spacer (10) on the axle against the axle shoulder. Carefully place the inner grease seal (8) on the axle against the spacer with the lips of the seal facing the bearing. Place the inner bearing cone (7) on the axle against the seal. Place the wheel and tire on the axle and seat the bearing cone in the bearing cup. Install the outer bearing cone (6) on the axle and seat the bearing cone in the outer bearing cup.

d. Install the outer grease seal (5) on the axle with the lips of the seal facing the bearing.

e. Apply a light coat of grease (8, Chart 2, 32-00-00) to the threads of the axle and the nut. Install the washer (4), key washer (3) and the axle nut (2) on the axle. Use the same grease as that used on the wheel bearings.

f. Tighten the axle nut to 150 to 200 inch-pounds of torque while rotating the wheel to ensure proper seating of the bearings.

g. Back off the axle nut to zero torque (DO NOT ALLOW THE WHEEL TO ROTATE WHILE BACKING OFF THE AXLE NUT), then torque the nut to 30 inch-pounds while rotating the wheel. Check to be certain there is no side motion of the wheel.

h. Install the cotter pin (1). If the holes do not align, tighten the nut to the next available keying position.

MAIN WHEEL AND BRAKE REMOVAL

a. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.

2. Turn the master switch ON.

3. Ensure that the parking brake is off.

4. Refer to Chapter 52-50-00 and remove the LH forward wing upper fairing (211BL).

5. Engage the No. 3 ANNUNCIATOR circuit breaker.

6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

b. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

c. Place the airplane on jacks as described in Chapter 7-10-00.

d. Remove the screws (1) securing the hubcap (2) to the wheel and remove the hubcap. Refer to Figure 12.

e. Cut the safety wire and back off the lock ring screws (25) far enough to allow the lock ring (3) to be removed from the nut (4). Remove the nut (4), key washer (7) and washer (8) securing the wheel and tire to the axle.

f. Remove the wheel and tire from the axle, being careful not to drop or damage the grease seals (5 and 11) and bearing cones (6 and 10).

CAUTION

Use care when removing all internal and external wheel components to avoid damaging the axle surface and/or nicking the axle threads.

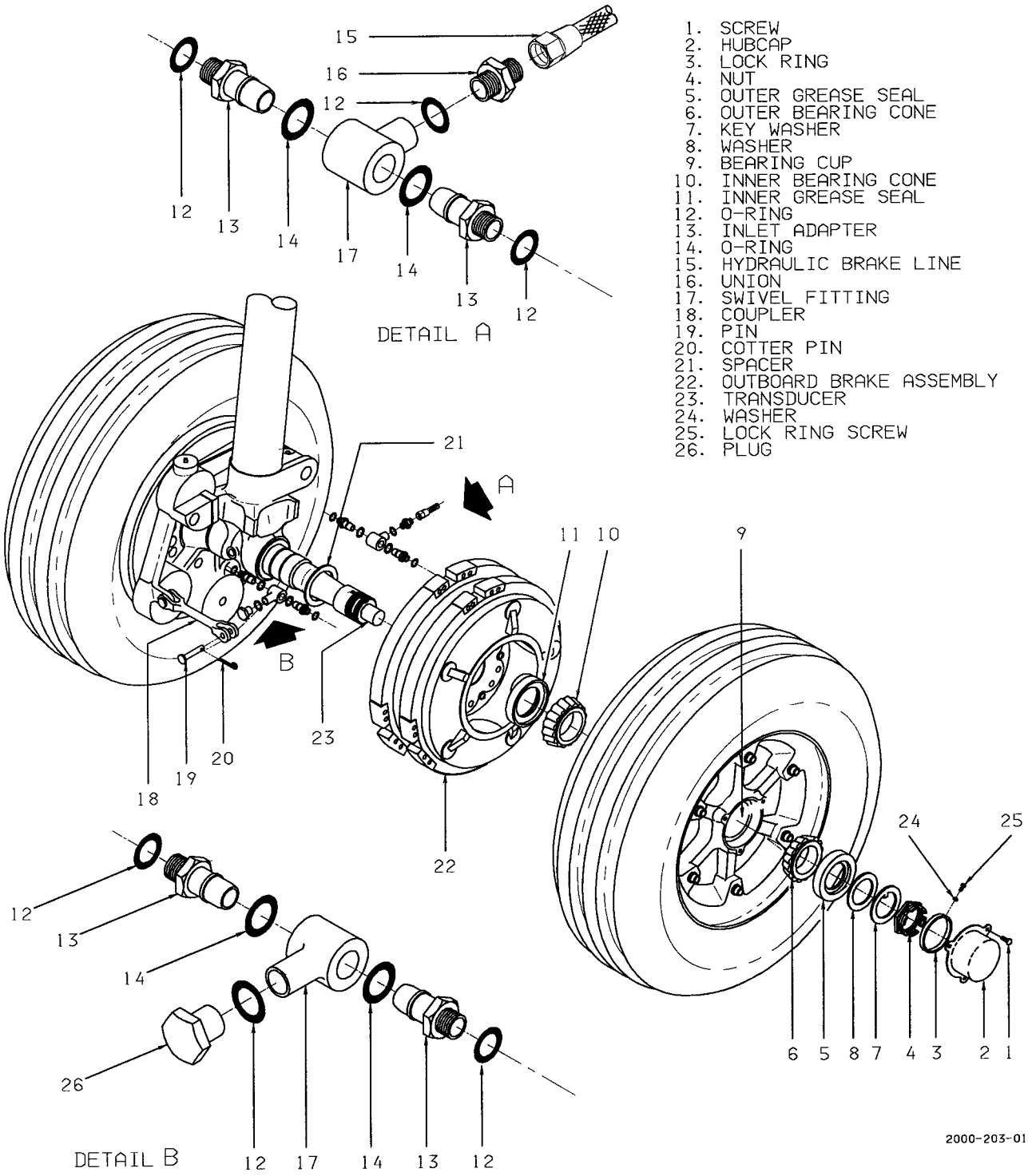
g. Disconnect the hydraulic brake line (15) from the union (16) and cap the line and fitting.

CAUTION

As the hydraulic lines are disconnected, plug or cap the openings to prevent entry of foreign material into the lines.

h. Remove the cotter pins (20) and pins (19) from the couplers (18) which secure the two brake assemblies to one another.

i. Remove brake assembly from the axle, pulling inlet adapters (13) out of swivel fittings (17) for both swivel fittings. Discard old O-rings (14).



Main Wheel and Brake Installation
Figure 12

j. Remove spacer (21) from the axle.

k. Refer to the Raytheon Aircraft Beech Starship 1 Component Maintenance Manual for further maintenance procedures on the wheel and tire.

MAIN WHEEL AND BRAKE INSTALLATION

a. Position the spacer (21) on the axle. Refer to Figure 12.

b. Apply a thin coat of grease (8, Chart 2, 32-00-00) to the inside diameter of the brake housing and on the axle at assembly. Position the brake assembly on the axle.

c. Install new O-rings (14), lubricated with hydraulic fluid (1, Chart 2, 32-00-00), on the inlet adapters (13).

d. Align inlet adapters (13) with swivel fittings (17) and push brake assembly (22) fully against spacer (21), being careful not to displace or damage the O-rings (14).

e. Connect brake assemblies together with couplers (18), pins (19) and cotter pins (20).

f. Ascertain that both brake assemblies are correctly seated on the strut and axle, and that the swivel fittings (17) swivel freely.

g. Connect the hydraulic brake line (15) to the union (16).

h. Pack grease (8, Chart 2, 32-00-00) into the bearing cones (6 and 10) and smear grease on the ends of the rollers. Do not over lubricate. Spread a thin coat of bearing grease on the surface of the bearing cups (9) and the lips on the grease seals (5 and 11).

i. Check the axle and nut for burrs or rough threads.

j. Apply grease (8, Chart 2, 32-00-00) to the axle threads and all bearing surfaces of the washer (7), key washer (8) and nut (4).

k. Install the inner grease seal (11) and the inner bearing cone (10) on the axle. The lips of the grease seal must be toward the wheel.

l. Place the wheel and tire on the axle and install the outer bearing cone (6) and grease seal (5).

CAUTION

Ascertain that the grease seals are properly seated in the wheel against the bearing cups and that the seals are not damaged.

m. Install the washer (8) and key washer (7) on the axle. Place the axle nut (4) on the axle and tighten to 150 to 200 inch-pounds of torque while rotating the wheel to ensure proper seating of the bearings.

n. Back off the nut to zero torque (DO NOT ALLOW THE WHEEL TO ROTATE WHILE BACKING OFF THE AXLE NUT), then torque the nut to 40 inch-pounds while rotating the wheel.

o. Install the lock ring (3) over the nut (4). Install the screws and washers securing the lock ring (3) and safety wire the screws. If the holes do not align, tighten the nut to the next available keying position. If necessary, use thin washers as required so as not to exceed 100 inch-pounds of torque.

p. Apply a small amount of sealant (6, Chart 2, 32-00-00) to the mating flange of the hubcap.

q. Secure the hubcap (2) to the wheel with the three screws (1).

CAUTION

Ensure that the transducer drive clip is properly engaged with the transducer during hubcap installation.

r. Bleed the brake system as described in BRAKE SYSTEM BLEEDING in this chapter.

s. Refer to Chapter 53-50-00and install the LH forward wing upper fairing (211BL).

t. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE SYSTEM SOLENOID VALVE REMOVAL

a. Refer to Chapter 53-50-00 and remove the lower LH nose access panel (131AL).

b. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.
2. Turn the master switch ON.
3. Ensure that the parking brake is off.
4. Remove the LH forward wing upper fairing (211BL) (refer to Chapter 53-50-00).
5. Engage the No. 3 ANNUNCIATOR circuit breaker.
6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

d. Remove electrical power from the airplane and disconnect the battery.

e. Cut safety wire and disconnect the electrical connector from the solenoid valve (5). Refer to Figure 13.

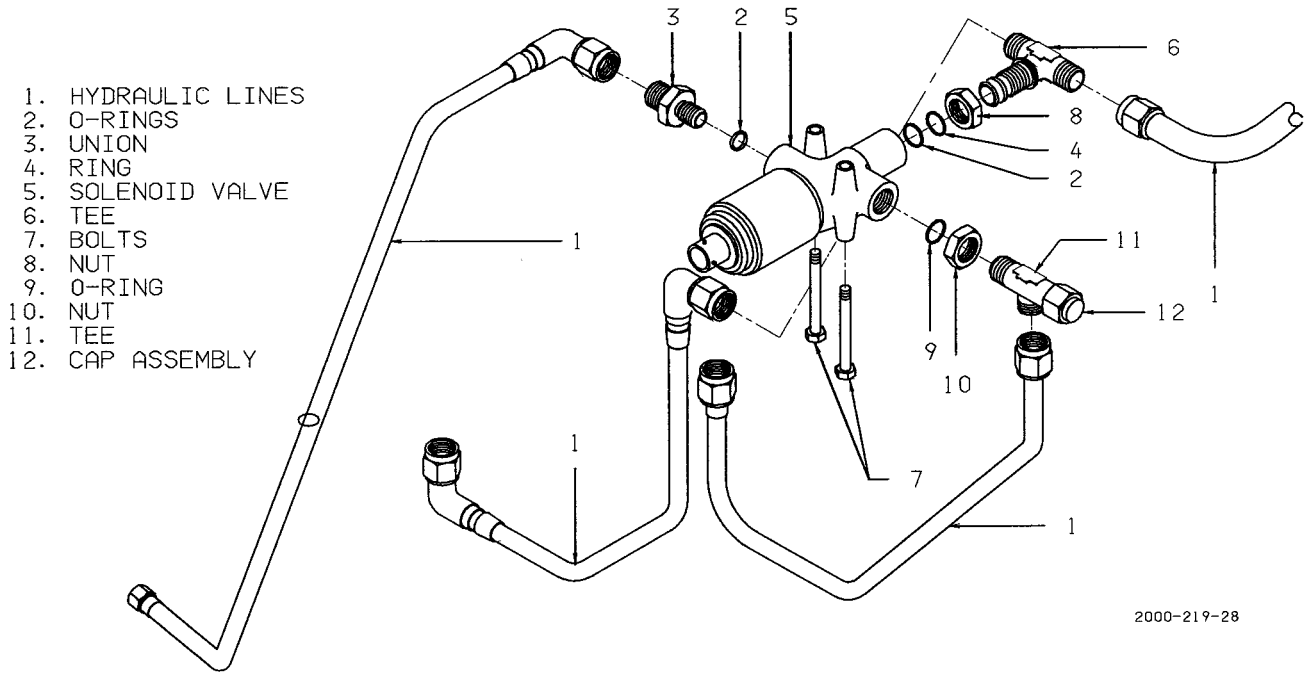
f. Disconnect the hydraulic lines (1) from the solenoid valve (5).

CAUTION

As the hydraulic lines are disconnected, plug or cap the openings to prevent entry of foreign material into the lines or valve.

g. Remove the two bolts (7) securing the solenoid valve (5) to the shelf.

h. Remove the solenoid valve (5) from the airplane.



Solenoid Valve Installation
Figure 13

i. Remove the union (3), O-rings (2), tees (6 and 11), nut (8), ring (4), O-ring (9), nut (10) and cap (12) from the solenoid valve (5). Discard the old O-rings (2 and 9) and ring (4).

BRAKE SYSTEM SOLENOID VALVE INSTALLATION

a. Install the union (3), O-rings (2), tees (6 and 11), nut (8), ring (4), O-ring (9), nut (10) and cap (12) on the solenoid valve (5). Use new O-rings (2 and 9) and ring (4). Refer to Figure 13.

b. Position the solenoid valve (5) in the airplane.

c. Install the two bolts (7) securing the solenoid valve (5) to the shelf.

d. Connect the hydraulic lines (1) to the solenoid valve (5).

e. Connect the electrical connector to the solenoid valve (5) and safety wire.

f. Connect the battery and restore electrical power to the airplane.

g. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

h. Refer to Chapter 53-50-00 and replace the access panel (131AL) removed from the lower LH nose.

i. Refer to Chapter 53-50-00 and reinstall the LH forward wing upper fairing (211BL).

j. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE SYSTEM POWER BRAKE VALVE REMOVAL

a. Refer to Chapter 53-50-00 and remove the lower LH nose access panel (131AL)

b. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.

2. Turn the master switch ON.

3. Ensure that the parking brake is off.

4. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).

5. Engage the No. 3 ANNUNCIATOR circuit breaker.

6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

d. Remove electrical power from the airplane and disconnect the battery.

e. Disconnect power brake valve wiring from the terminal board (TB1). Tag wiring.

f. Disconnect the hydraulic lines (1) from the power brake valve (5). Refer to Figure 14.

CAUTION

As the hydraulic lines are disconnected, plug or cap the openings to prevent entry of foreign material into the lines or valve.

g. Remove the three bolts (4) securing the power brake valve (5) to the pallet assembly.

h. Remove the power brake valve (5) from the airplane.

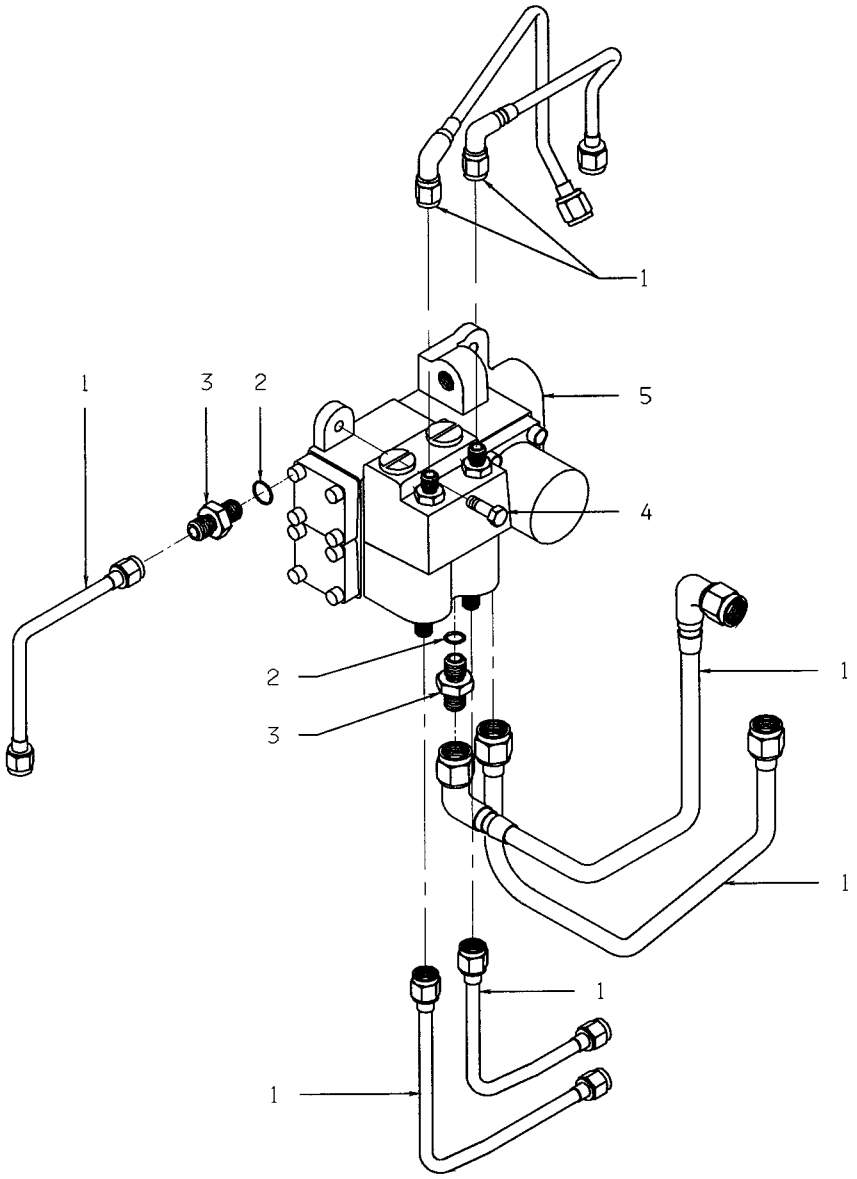
NOTE

Remove only the unions shown removed in Figure 14. The remainder of the unions are filtered fittings. Refer to the Raytheon Aircraft Beech Starship 1 Component Maintenance Manual for further information on the power brake valve.

i. Remove unions (3) from power brake valve (5). Discard old O-rings (2).

BRAKE SYSTEM POWER BRAKE VALVE INSTALLATION

a. Install unions (3) on power brake valve (5), using new O-rings (2). Refer to Figure 14.



1. HYDRAULIC LINES
2. O-RINGS
3. UNIONS
4. BOLTS
5. POWER BRAKE VALVE

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Power Brake Valve Installation
Figure 14

b. Position the power brake valve (5) in the pallet assembly and secure the power brake valve (5) to the pallet assembly with the three bolts (4).

c. Connect the hydraulic lines (1) to the power brake valve (5).

d. Connect power brake valve wiring to the terminal board (TB1) as tagged.

e. Connect the battery and restore electrical power to the airplane.

f. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

g. Refer to Chapter 53-50-00 and replace the access panel (131AL) removed from the lower LH nose.

h. Refer to Chapter 53-50-00 and reinstall the LH forward wing upper fairing (211BL).

i. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE SYSTEM PUMP AND MOTOR REMOVAL

a. Refer to Chapter 53-50-00 and remove the lower LH nose access panel (131AL).

b. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.
2. Turn the master switch ON.
3. Ensure that the parking brake is off.
4. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).
5. Engage the No. 3 ANNUNCIATOR circuit breaker.
6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

d. Remove electrical power from the airplane and disconnect the battery.

e. Disconnect the pump and motor input wire (red) from the pump motor relay. Tag the pump motor relay. Disconnect the ground wire (black) for the pump and motor from the terminal board (TB1) and tag TB1. Refer to 32-40-00, Figure 1.

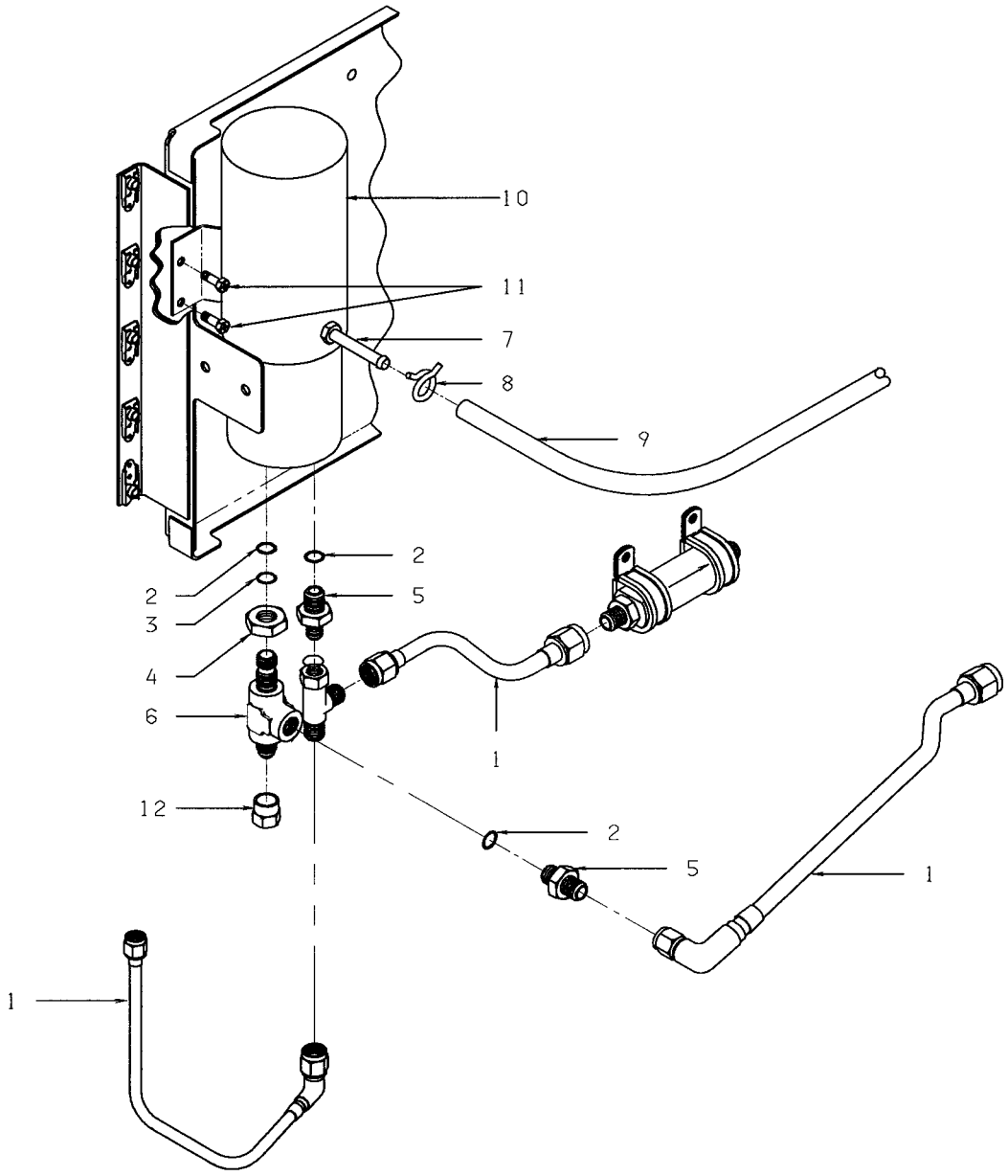
f. Disconnect the hydraulic lines (1) from the pump and motor (10). Refer to Figure 15.

CAUTION

As the hydraulic lines are disconnected, plug or cap the openings to prevent entry of foreign material into the lines or pump.

g. Remove clamp (8) and tube (9) from the adapter (7).

h. Remove the four screws (11) securing the pump and motor (10) to the pallet assembly.



1. HYDRAULIC LINES
2. O-RINGS
3. RING
4. NUT
5. UNIONS
6. TEES
7. ADAPTER
8. CLAMP
9. TUBE
10. PUMP AND MOTOR
11. SCREWS
12. CAP ASSEMBLY

2000-219-29

Pump and Motor Installation
Figure 15

i. Remove the pump and motor (10) from the airplane. Remove the tees (6), nut (4) and union (5) from the pump and motor (10). Discard old O-rings (2) and ring (3).

j. Remove adapter (7) from the pump and motor (10).

BRAKE SYSTEM PUMP AND MOTOR INSTALLATION

a. Install the adapter (7) on the pump and motor (10). Refer to Figure 15.

b. Install tees (6), nut (4) and union (5) on the pump and motor (10) using new O-rings (2) and ring (3).

c. Position the pump and motor (10) in the pallet assembly and secure with the four screws (11).

d. Install the ground wire (black) on the terminal board (TB1) as tagged. Install the power input wire (red) for the pump and motor to the pump motor relay as tagged.

e. Connect the hydraulic lines (1) to the pump and motor (10).

f. Install tube (9) and clamps (8) on the pump and motor (10).

g. Connect the battery and restore electrical power to the airplane.

h. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

i. Refer to Chapter 53-50-00 and install the access panel (131AL) removed from the lower LH nose.

j. Refer to Chapter 53-50-00 and install the LH forward wing upper fairing (211BL).

k. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE SYSTEM ACCUMULATOR REMOVAL

a. Refer to Chapter 53-50-00 and remove the lower LH nose access panel (131AL).

b. Remove brake fluid pressure from accumulator as follows:

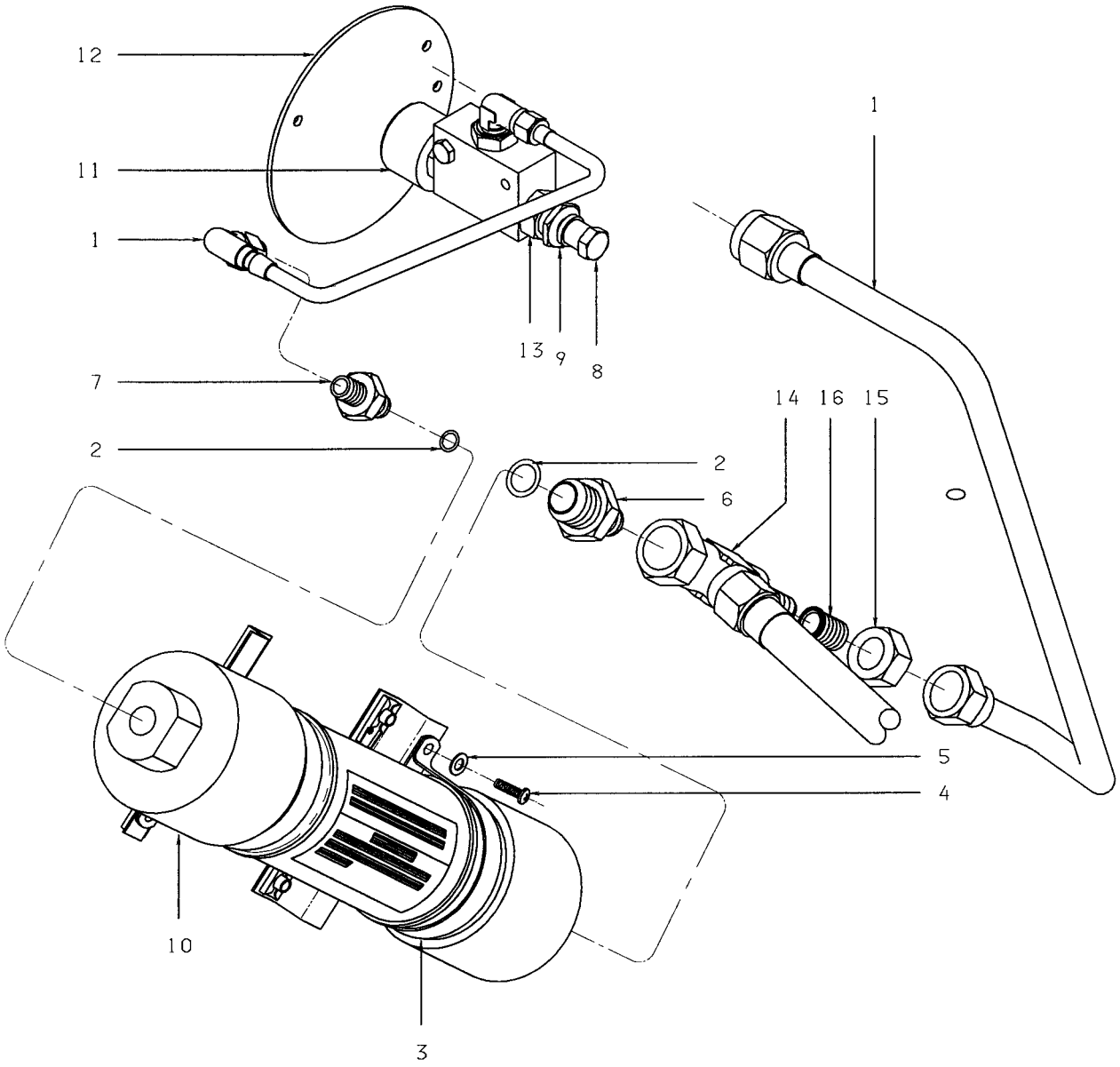
1. Open the ANTI SKID circuit breaker.
2. Turn the master switch ON.
3. Ensure that the parking brake is off.
4. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).
5. Engage the No. 3 ANNUNCIATOR circuit breaker.
6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Open the No. 3 Accumulator circuit breaker controlling the maintenance switch.

CAUTION

Wear eye protection when releasing gas pressure from accumulator.

d. Release gas pressure from the accumulator (10) by removing the valve cap (8) from the charge valve (13) and slowly opening the swivel nut (9) to release nitrogen gas pressure from the accumulator (10). Refer to Figure 16.



1. HYDRAULIC LINES
2. O-RINGS
3. CLAMP
4. SCREW
5. WASHER
6. REDUCER
7. REDUCER
8. VALVE CAP
9. SWIVEL NUT
10. ACCUMULATOR
11. ACCUMULATOR PRESSURE GAGE
12. COVER
13. CHARGE VALVE
14. TEE
15. NUT
16. REDUCER

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Accumulator Installation
Figure 16

e. Disconnect the hydraulic lines (1) from the accumulator (10).

CAUTION

As the hydraulic lines are disconnected, plug or cap the openings to prevent entry of foreign material into the lines or accumulator.

f. Remove the screws (4), washers (5) and clamps (3) securing the accumulator (10) to the mounting brackets.

g. Remove the accumulator (10) from the airplane.

h. Remove the reducer (7), tee (14), nut (15) and reducers (6 and 16) from the accumulator (10). Discard old O-rings (2).

BRAKE SYSTEM ACCUMULATOR INSTALLATION

a. Install the reducer (7), tee (14), nut (15) and reducers (6 and 16) to the accumulator (10) using new O-rings (2). Refer to Figure 16.

b. Position the accumulator (10) in the airplane and secure to brackets with screws (4), washers (5) and clamps (3).

c. Connect the hydraulic lines (1) to accumulator (10).

d. Charge accumulator (10) as described in Chapter 12-10-00.

e. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

f. Refer to Chapter 53-50-00 and install the lower LH nose access panel (131AL)

g. Refer to Chapter 53-50-00 and install the LH forward wing upper fairing (211BL).

h. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE SYSTEM FILTER REMOVAL

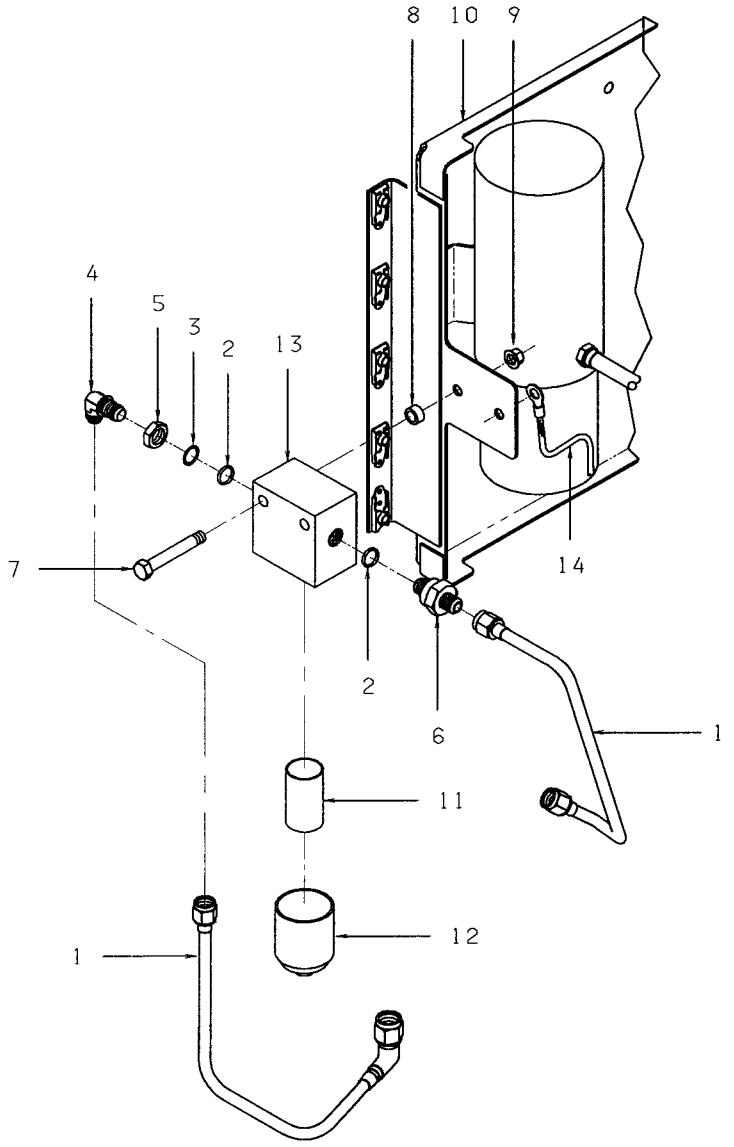
a. Refer to Chapter 53-50-00 and remove the lower LH nose access panel (131AL).

b. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.
2. Turn the master switch ON.
3. Ensure that the parking brake is off.
4. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).
5. Engage the No. 3 ANNUNCIATOR circuit breaker.
6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

d. Disconnect the hydraulic lines (1) from elbow (4) and check valve (6). Refer to Figure 17.



1. HYDRAULIC LINES
2. O-RINGS
3. RING
4. ELBOW
5. NUT
6. CHECK VALVE
7. BOLTS
8. SPACER
9. NUTS
10. PALLET
11. FILTER ELEMENT
12. FILTER BOWL
13. FILTER BASE
14. BONDING JUMPER

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Brake System Filter Installation
Figure 17

CAUTION

As the hydraulic lines are disconnected, plug or cap the openings to prevent entry of foreign material into the lines.

e. Remove the bolts (7), nuts (9) and spacers (8) securing the filter base (13) to the pallet (10).

f. Remove the filter base (13) from the airplane.

g. Remove the elbow (4), nut (5) and check valve (6) from the filter base (13). Discard old O-rings (2) and ring (3).

BRAKE SYSTEM FILTER INSTALLATION

a. Install elbow (4), nut (5) and check valve (6) on filter base (13), use new O-rings (2) and ring (3). Refer to Figure 17.

b. Position filter base (13) on pallet (10) and secure with bolts (7), nuts (9) and spacers (8). Install bonding jumper (14) between nut (9) and bracket.

c. Connect hydraulic lines (1) to elbow (4) and check valve (6).

d. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

e. Refer to Chapter 53-50-00 and replace the lower LH nose access panel (131AL).

f. Install the LH forward wing upper fairing (211BL) (refer to Chapter 53-50-00).

g. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE SYSTEM FILTER ELEMENT REPLACEMENT

Replace the system filter element after the first 100 hours. Inspect the filter element at the intervals specified in Chapter 5-20-00 and replace as necessary.

a. Refer to Chapter 53-50-00 and remove the lower LH nose access panel (131AL) to gain access to the filter element.

b. Open the ANTI SKID and No. 3 ANNUNCIATOR circuit breakers.

c. Remove safety wire securing filter bowl to base. Refer to Figure 17.

d. Unscrew filter bowl from base.

e. Remove filter element and replace with a new filter element.

f. Screw filter bowl onto base.

g. Safety wire filter bowl to base.

h. Refer to Chapter 53-50-00 to replace lower LH nose access panel (131AL).

i. Close the ANTI SKID and No. 3 ANNUNCIATOR circuit breakers.

j. Bleed the brake system as described in BRAKE SYSTEM BLEEDING in this chapter.

BRAKE SYSTEM PRESSURE SWITCH REMOVAL

This procedure is applicable for both the high pressure and the low pressure switch. Remove either switch as follows:

a. Refer to Chapter 53-50-00 and remove the lower LH nose access panel (131AL).

b. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.

2. Turn the master switch ON.

3. Ensure that the parking brake is off.

4. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).

5. Engage the No. 3 ANNUNCIATOR circuit breaker.

6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

d. Remove electrical power from the airplane and disconnect the battery.

e. Disconnect the pressure switch (1 or 2) electrical wiring from the terminal board (TB1). Tag the wiring and terminal board. Refer to Figure 18.

f. Cut and remove safety wire from pressure switch (1 or 2).

g. Unscrew the pressure switch (1 or 2) from the hydraulic fitting (4). Discard old O-ring (3).

PRESSURE SWITCH INSTALLATION

This procedure is applicable for both the high pressure and the low pressure switch. Install either switch as follows:

a. Screw the pressure switch (1 or 2) into the hydraulic fitting (4); use new O-rings (3). Refer to Figure 18.

b. Safety wire the pressure switch (1 or 2).

c. Connect pressure switch (1 or 2) electrical wiring to terminal board (TB1) as tagged.

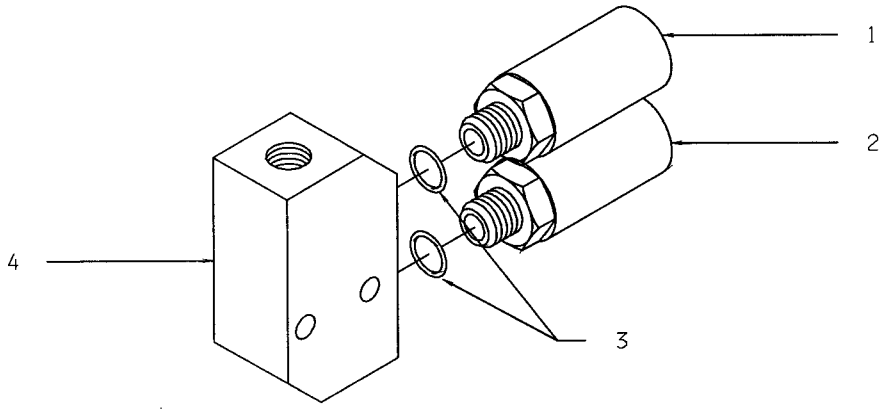
d. Connect the battery and restore electrical power to the airplane.

e. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

f. Refer to Chapter 53-50-00 and install the lower LH nose access panel (131AL).

g. Refer to Chapter 53-50-00 and install the LH forward wing upper fairing (211BL).

h. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.



1. HIGH PRESSURE SWITCH
2. LOW PRESSURE SWITCH
3. O-RINGS
4. HYDRAULIC FITTING

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Pressure Switch Installation
Figure 18

BRAKE SYSTEM RELIEF VALVE REMOVAL

a. Refer to Chapter 53-50-00 and remove the lower LH nose access panel (131AL).

b. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.

2. Turn the master switch ON.

3. Ensure that the parking brake is off.

4. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).

5. Engage the No. 3 ANNUNCIATOR circuit breaker.

6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

d. Disconnect hydraulic lines (1) from unions (6). Refer to Figure 19.

e. Remove screws (2), clamps (3) relief valve (5) from the airplane.

f. Remove unions (6) from the relief valve (5). Discard old O-rings (4).

BRAKE SYSTEM RELIEF VALVE INSTALLATION

a. Install unions (6) on relief valve (5), use new O-rings (4). Refer to Figure 19.

b. Position relief valve (5) on pallet, ensuring that the flow arrow points forward. Secure with clamps (3) and screws (2).

c. Connect hydraulic lines (1) to relief valve (5).

d. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

e. Refer to Chapter 53-50-00 and install the lower LH nose access panel (131AL).

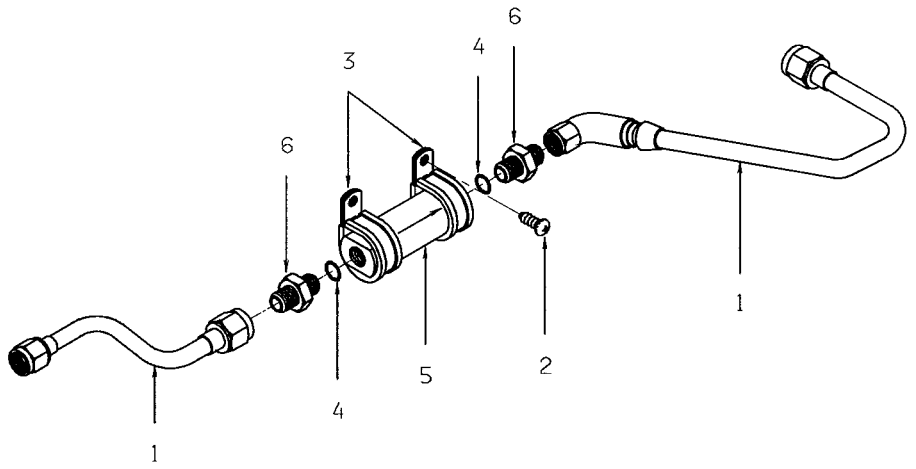
f. Refer to Chapter 53-50-00 and install the forward wing upper fairing (211BL).

g. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE RESERVOIR REMOVAL

a. Refer to Chapter 53-50-00 and install the lower LH nose access panel (131AL).

b. Remove brake fluid pressure from accumulator as follows:



Relief Valve Installation
Figure 19

1. Open the ANTI SKID circuit breaker.

2. Turn the master switch ON.

3. Ensure that the parking brake is off.

4. Refer Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).

5. Engage the No. 3 ANNUNCIATOR circuit breaker.

6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

c. Open the No. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

d. Remove electrical power from the airplane and disconnect the battery.

e. Disconnect hydraulic lines (1) from top, bottom and inboard side of brake reservoir (4). Refer to Figure 20.

f. Disconnect wiring from inboard side of maintenance switch (7). Tag wiring.

g. Remove three bolts (2) and three washers (3) securing brake reservoir (4) to forward pressure bulkhead and remove brake reservoir (4) from airplane.

BRAKE RESERVOIR INSTALLATION

a. Secure brake reservoir (4) to forward pressure bulkhead with three bolts (2) and three washers (3). Install chain (6) on filler plug under bolt (2) and washer (3) in upper outboard mounting hole. Refer to Figure 20Figure 21.

b. Connect wiring, as tagged, to inboard side of maintenance switch (7).

c. Connect hydraulic lines (1) to brake reservoir (4) in three places.

d. Loosen locking nuts securing the maintenance switch (7) to mounting bracket and adjust the maintenance switch so that 0.06 to 0.12 inch overtravel is maintained when LH forward wing fairing (211BL) is installed.

e. Connect the battery and restore electrical power to the airplane.

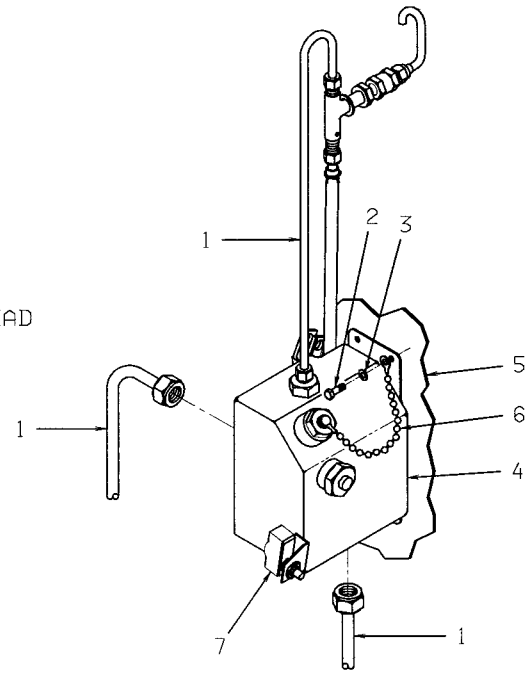
f. Ensure system is filled properly by completing BRAKE SYSTEM SERVICING as described in Chapter 12-10-00.

g. Bleed the brakes as described in BRAKE SYSTEM BLEEDING in this chapter.

h. Refer to Chapter 53-50-00 and install the LH forward wing upper fairing (211BL).

i. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

1. HYDRAULIC LINES
2. BOLTS
3. WASHERS
4. BRAKE RESERVOIR
5. FORWARD PRESSURE BULKHEAD
6. CHAIN
7. MAINTENANCE SWITCH



Brake Reservoir Installation
Figure 20

BRAKE SYSTEM ANTISKID CONTROL BOX REMOVAL

a. Open the ANTI SKID circuit breaker.

b. Refer to Chapter 53-50-00 and remove the aft wing box shear panel (528AB) located below the aft wing to gain access to the antiskid control box (2). The control box (2) is located in the aft manhole and is mounted on a bracket attached to the underside of the upper wing skin (5) just forward of the rear spar (6). Refer to Figure 21.

c. Disconnect the electrical connector from the control box (2).

d. Remove four screws (4) and washers (3) securing control box (2) to mounting bracket (1).

e. Remove control box (2) from airplane.

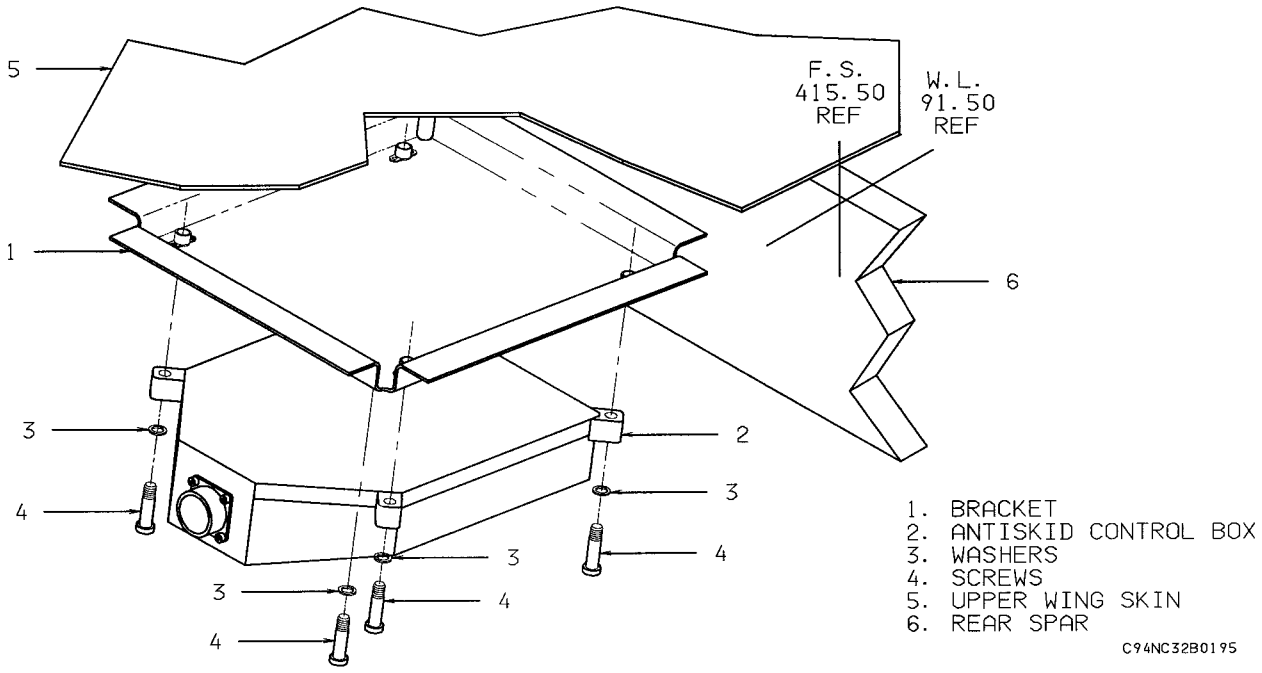
BRAKE SYSTEM ANTISKID CONTROL BOX INSTALLATION

a. Position control box (2) in airplane and secure to mounting bracket (1) with four screws (4) and washers (3). Refer to Figure 21.

b. Connect electrical connector to control box (2).

c. Refer to Chapter 53-50-00 and install aft wing box shear panel (528AB).

d. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.



Antiskid Control Box Installation
Figure 21

WHEEL SPEED TRANSDUCER REMOVAL

a. Open the ANTI SKID circuit breaker.

b. Place the airplane on jacks as described in Chapter 7-10-00.

c. Remove the three screws (1) securing the hubcap (2) to the wheel and remove the hubcap (2). Refer to Figure 22.

d. Cut the safety wire and back off the two lock ring screws (7) and washers (8) far enough to allow the lock ring (3) to be removed from the nut (4). Remove the nut (4) securing the wheel and tire to the axle (11).

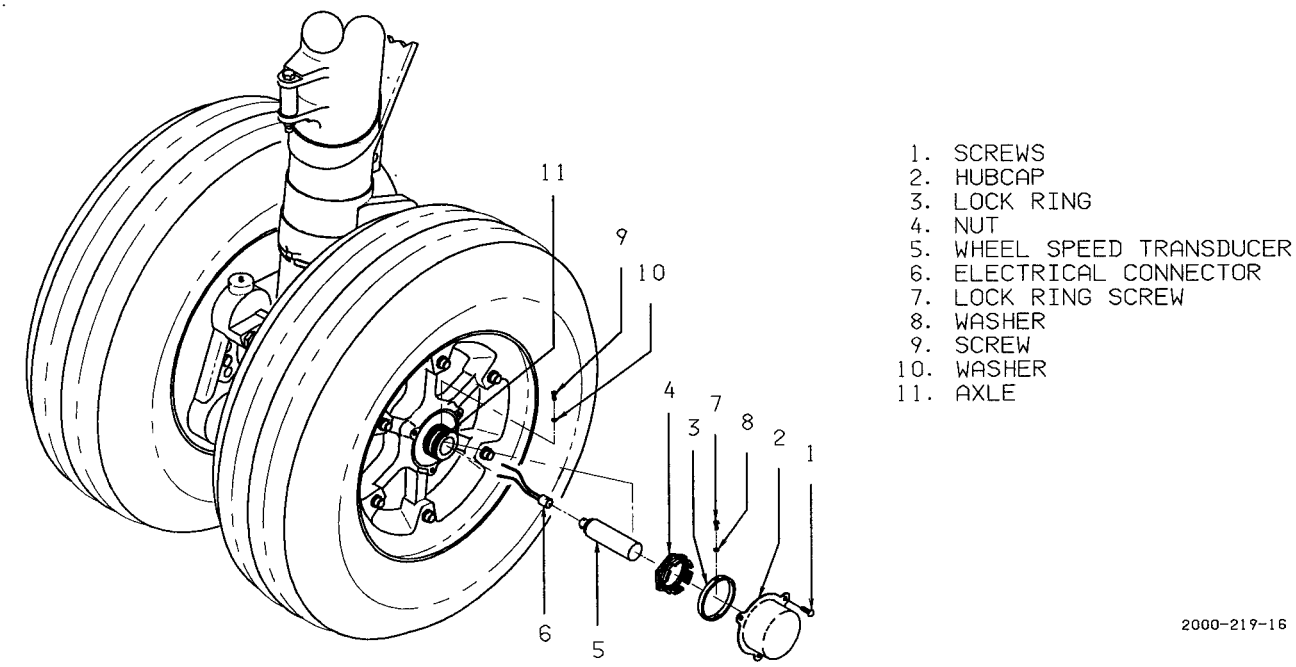
e. Remove the screws (9) and washers (10) from counterbored holes in axle (11). Discard the screws and washers.

f. Pull the wheel speed transducer (5) out of axle (11) far enough to gain access to electrical connector (6).

g. Disconnect electrical connector (6) from wheel speed transducer (5) and remove wheel speed transducer (5) from airplane.

WHEEL SPEED TRANSDUCER INSTALLATION

a. Connect electrical connector (6) to wheel speed transducer (5). Refer to Figure 22.



Wheel Speed Transducer Installation
Figure 22

b. Insert wheel speed transducer (5) into axle (11), being careful to slide wiring in before the wheel speed transducer (5).

c. Secure wheel speed transducer (5) to axle (11) with new screws (9) and new washers (10) inserted into counterbored holes in axle (11). Verify a minimum clearance of 0.010 inch between screw ends and transducer tang. Use extra washers as required to maintain clearance.

d. Place the axle nut (4) on the axle (11) and tighten to 150 to 200 inch-pounds of torque while rotating the wheel to ensure proper seating of the bearings.

e. Back off the axle nut (4) to zero torque (DO NOT ALLOW THE WHEEL TO ROTATE WHILE BACKING OFF THE AXLE NUT), then torque the nut (4) to 40 inch-pounds while rotating the wheel.

f. Install the lock ring (3) over the nut (4).

g. Install the two screws (7) and washers (8) securing the lock ring (3) and safety wire the screws (7). If the holes do not align, tighten the nut (4) to the next available keying position. If necessary, use thin washers as required so as not to exceed 100 inch-pounds of torque.

h. Apply a thin coat of sealant (6, Chart 2, 32-00-00), to the mating flange of the hubcap.

i. With the transducer tang 90° to the drive clip to prevent engagement, measure the distance between the hubcap and the hubcap gasket. Refer to Figure 23. Measurement should be at least 0.170 in. Should adjustments be needed a shim may be installed between the hubcap and the drive clip.

j. Check drive clip opening dimension, should be 0.080 to 0.170, if not replace.

k. Secure the hubcap (2) to the wheel with the three screws (1).

CAUTION

Ensure that the transducer drive clip on the inside of the hubcap is properly engaged with the transducer during hubcap installation.

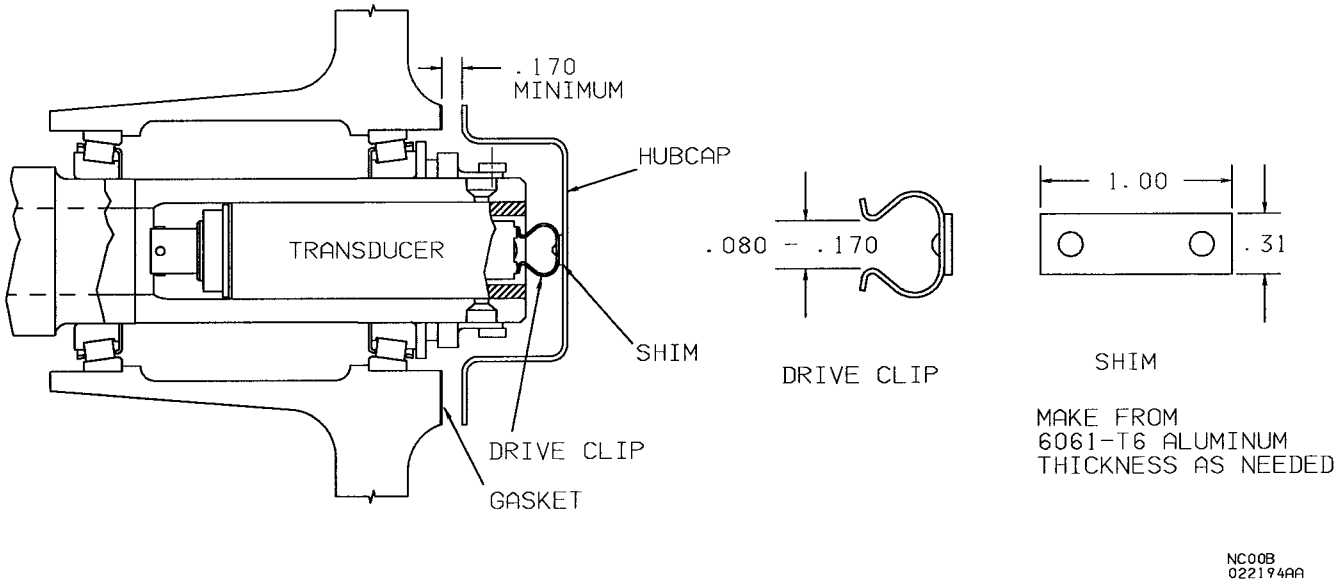
l. Perform functional check as described in BRAKE SYSTEM FUNCTIONAL CHECK in this chapter.

BRAKE SYSTEM FUNCTIONAL CHECK

a. The following tools are required to perform the brake system functional check (refer to Chart 1, 32-00-00).

1. Drill motor (7, Chart 1, 32-00-00).
2. Wheel drive clip fixture. Refer to Figure 24. (8, Chart 1, 32-00-00).
3. Pressure gages (2) (9, Chart 1, 32-00-00).

b. Install pressure gages at forward swivel fittings on left and right main landing gear brakes.



Wheel Speed Transducer Dimension Check
Figure 23

c. Fully depress pilot's LH brake pedal and verify pressure on gage installed on LH brake. Release LH brake pedal.

d. Fully depress pilot's RH brake pedal and verify pressure on gage installed on RH brake. Release RH brake.

e. Repeat steps "c" and "d" for the copilot's brake pedals.

f. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.

2. Turn the master switch ON.

3. Ensure that the parking brake is off.

4. Refer to Chapter 53-50-00 and remove the LH forward wing upper fairing (211BL).

5. Engage the No. 3 ANNUNCIATOR circuit breaker.

6. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

g. Verify "POWER BRAKE INOP" message on EICAS display is displayed.

h. Open the NO. 3 ANNUNCIATOR circuit breaker controlling the maintenance switch.

i. Engage the ANTI SKID circuit breaker and allow system to stabilize. Verify "POWER BRAKE INOP" message is not displayed on EICAS display.

j. Set the ANTI SKID-OFF switch to ANTI SKID. Verify "ANTI SKID INOP" message is not displayed on EICAS display.

k. Remove all four main gear hubcaps.

l. Install wheel drive clip fixture (Refer to Figure 23) (8, Chart 1, 32-00-00) in a drill motor (7, Chart 1, 32-00-00). Fully depress pilot's LH and RH brake pedals and hold.

NOTE

There should be a maximum pressure of 750 psi at left and right brakes.

m. With brake pedal pressure applied, rotate left outboard wheel speed transducer to approximately 800 to 1,200 rpm using drill motor (7, Chart 1, 32-00-00) and wheel drive clip (8, Chart 1, 32-00-00) Stop drill motor abruptly. Actuation of servo valve by the antiskid control box should cause hydraulic pressure at gages to drop to 0 to 50 psi immediately and return to original pressure within 15 to 20 seconds maximum. Verify that pressure switch maintains system pressure between 1,150 psi minimum to 1,575 psi maximum by observing the accumulator pressure gage in the nose wheel well.

n. Repeat above step for remaining three transducer positions.

o. Remove brake fluid pressure from accumulator as follows:

1. Open the ANTI SKID circuit breaker.

2. Turn the master switch ON.

3. Ensure that the parking brake is off.

4. Engage the No. 3 ANNUNCIATOR circuit breaker.

5. Depress the brake pedals 15 to 20 times to deplete the brake fluid pressure of the accumulator. While pumping the pedals, have an assistant observe the accumulator pressure gage in the nose wheel well. When the reading on the gage reaches 600 ± 50 psi and remains stationary, the fluid pressure has been depleted.

p. Remove pressure gages.

q. To allow any air that may be in the forward swivel fitting to circulate to the top of the brake assembly, rotate the swivel fitting so that the plug is pointing toward the ground.

r. To remove any air allowed to enter the system while removing the test gages, loosen the forward bleeder port screws from both brake assemblies on one main landing gear. Depress the pilot's corresponding brake pedal slowly and smoothly. With the brake pedal in the depressed position close the bleeder port screws. Repeat as necessary to remove any air in the brake assemblies. Safety wire the bleeder port screws.

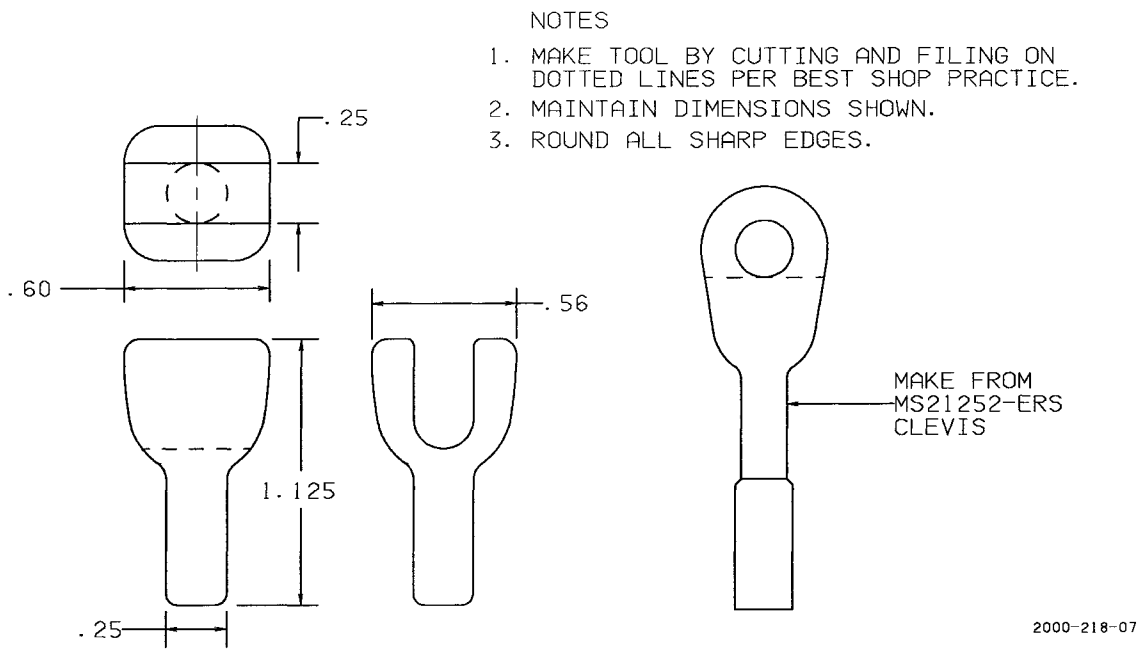
s. Repeat step "q" on the brake assemblies of the other side of the system.

t. Apply a small amount of sealant (6, Chart 2, 32-00-00) to the mating flange of the hubcap.

u. Install all four main gear hubcaps.

v. Refer to Chapter 53-50-00 and install the LH forward wing upper fairing (211BL).

w. Ensure system is filled properly by completing BRAKE SYSTEM SERVICING as described in Chapter 12-10-00.



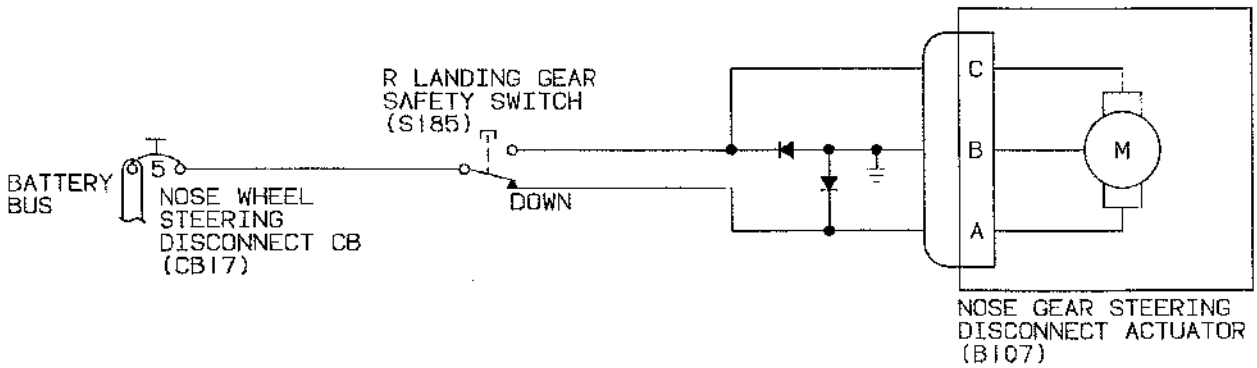
Wheel Drive Clip Fixture
Figure 24

NOSE LANDING GEAR STEERING - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 2)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The nose landing gear steering system is installed to facilitate turning while the airplane is on the ground. Steering action is transmitted from the rudder pedals to the nose gear through a series of pushrods and bell cranks. The mechanical steering linkage extends from the rudder pedals, through the forward bulkhead pressure seal assembly (F.S. 95.00), and into the upper area of the nose wheel well. A spring link connects the forward bulkhead bell crank assembly (F.S. 95.00) to an idler bell crank. The spring link prevents the transmission of excessive shock loads to the rudder pedals and allows for brake or tow bar induced steering angles greater than that realized by actuation of the rudder pedals. Normal nose gear steering travel is a full 24° (12° to each side). The idler bell crank converts the forward/aft travel of the spring link to a side-to-side motion and transfers that leverage to the steering bell crank via a push-pull rod. The steering bell



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Nose Landing Gear Steering Disconnect System Schematic (Effectivity: All)
Figure 1

crank mounts on the pivot fitting above the nose gear assembly and provides an attach point for the steering cam.

A hydraulic shimmy damper connected to the steering bell crank prevents oscillation of the nose gear wheel while the airplane is rolling. A straightening bracket attached to the forward side of the F.S. 85.00 bulkhead provides an alignment reference for the straightener guide on the steering bell crank, centering the nose gear assembly during extension and retraction.

NOSE LANDING GEAR STEERING DISCONNECT (Effectivity: All) (Figures 1 and 2)

A nose gear steering disconnect system is installed on the airplane to disable steering action from the rudder pedals during flight. The system consists of an electrical actuator assembly mounted on the LH rudder sector, the RH landing gear safety switch (S186) mounted on the RH main landing gear assembly, and all associated wiring and mechanical linkage.

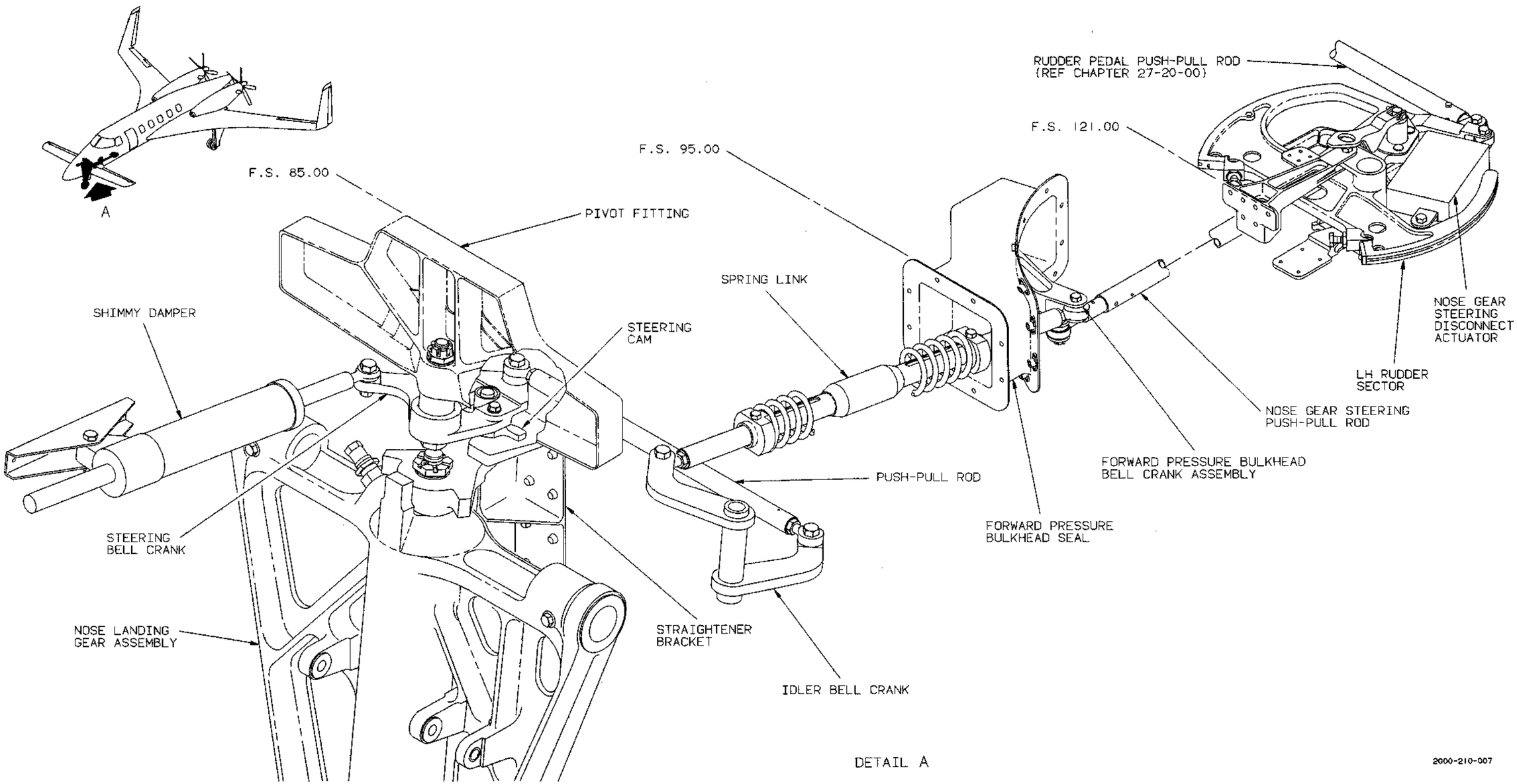
As the weight of the airplane compresses the RH main landing gear strut during landing, actuation of the RH safety switch applies voltage to pin A of the actuator motor to extend the actuator rod and engage the rudder pedals with the steering linkage. As the weight of the airplane is removed from the landing gear during take-off, extension of the RH main landing gear strut actuates the RH landing gear safety switch to the "in air" position. Voltage is then applied to pin C of the actuator motor to retract the actuator rod and disengage the steering linkage from the rudder pedals during flight. The actuator assembly incorporates limit switches to control actuator length of travel and has internal mechanical stops which prevent excess overtravel should a limit switch fail. Normal full travel of the actuator rod from the extended postion to full retraction is approximately 1.82 inches. The actuator motor is powered from the "hot" battery bus through the nose wheel steering disconnect circuit breaker.

NOSE LANDING GEAR STEERING - TROUBLESHOOTING (Effectivity: All)

If drag or limited movement is noticed during operation of the steering system, perform a visual inspection of the steering system components while a second person actuates the rudder pedals. Visually inspect the steering system components for any worn or loose parts and repair or replace them as necessary to return the system to normal operating status. Chart 1 contains troubleshooting information used to isolate specific faults in the steering system. Match the fault description with the symptom/trouble in Chart 1 and check the items listed under probable cause to isolate the fault. Once the fault is located, perform the corrective action listed for that item.

NOTE

If a fault involves both the steering and rudder control systems, refer to Chapter 27-20-00 for detailed information on the rudder control system.



DETAIL A

2000-210-007

Nose Landing Gear Steering (Effectivity: All)
Figure 2

CHART 1
NOSE GEAR STEERING SYSTEM TROUBLESHOOTING (Effectivity: All)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. The steering system feels loose or appears to have excessive slack when operated.	a. Loose or damaged push-pull rods.	a. Remove and replace damaged push-pull rods or tighten any loose connections.
	b. Worn or damaged bearings or bushings in steering linkage or bell crank assemblies.	b. Inspect all bearings and bushings that appear to be loose. Remove and replace as necessary.
2. The steering system drags when operated.	a. Steering linkages and push-pull rod connections overtorqued.	a. Loosen attachment points and retorque to prevent binding.
	b. Worn bearings in the forward pressure bulkhead bell crank assembly.	b. Inspect the bearings for wear. Refer to FORWARD PRESSURE BULKHEAD BELL CRANK ASSEMBLY REMOVAL to replace the bearings.
3. The steering system shimmies or vibrates during operation.	a. Worn or defective shimmy damper.	a. Refer to NOSE LANDING GEAR SHIMMY DAMPER REMOVAL and replace the defective shimmy damper.
	b. Shimmy damper low on hydraulic fluid.	b. Refer to NOSE GEAR SHIMMY DAMPER SERVICING and check the fluid level of the shimmy damper. Service the unit as needed.
	c. Loose, worn or misaligned steering system components.	c. Replace, rig or repair components that are loose, damaged or misaligned.
	d. Uneven tire wear or out-of-round condition.	d. Refer to Chapter 32-40-00 and perform a detailed inspection of the wheel and brake assembly.

NOSE LANDING GEAR STEERING - MAINTENANCE PRACTICES (Effectivity: All)

NOSE LANDING GEAR SHIMMY DAMPER SERVICING (Effectivity: All)

FLUID LEVEL CHECK (Effectivity: All) (Figure 3)

The shimmy damper fluid level may be checked without removing the unit from the airplane. Check the fluid level as follows and refer to FLUID LEVEL SERVICING to refill the reservoir if the fluid level is low.

a. Insert a wire, approximately 1/16 inch in diameter, through the hole in the washer at the forward end of the piston rod tube assembly , until the wire touches the bottom of the hole in the floating piston.

NOTE

A wire insertion depth of 8 inches indicates an empty reservoir.

b. Mark the wire, remove it and measure the depth of insertion. Inserting the wire in the hole of the floating piston, rather than letting it rest against the face of the piston, will give a more accurate check. To determine if the wire is properly inserted in the hole of the floating piston, insert the wire several times, noting each insertion depth. When the wire is in the hole, the depth will be approximately 1/2 inch greater than when against the face of the piston.

NOTE

With the shimmy damper reservoir full, measurable distance from the forward floating piston to the end of the tube assembly is 4.75 to 5.00 inches. When measurable distance from the forward floating piston to the end of the tube assembly reaches 6.5 to 7.0 inches, service the shimmy damper reservoir with hydraulic fluid (1, Chart 2, 32-00-00) as outlined by the following FLUID LEVEL SERVICING procedure.

FLUID LEVEL SERVICING (Effectivity: All) (Figure 3)

a. Refer to NOSE GEAR SHIMMY DAMPER REMOVAL and remove the shimmy damper from the airplane.

b. Secure the shimmy damper in a fixed position with the rod end of the piston pointing downward (preferably held by a vise).

CAUTION

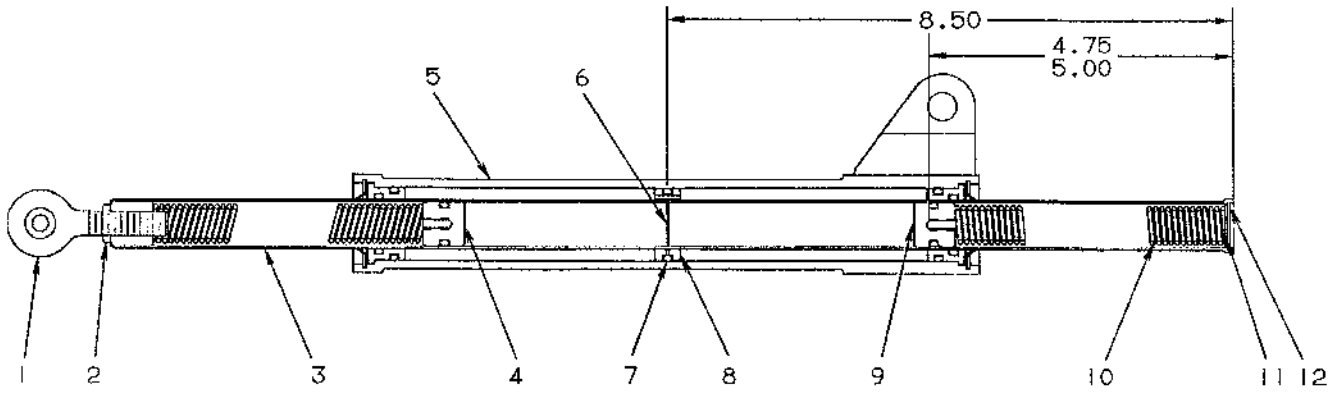
The spring in the tube assembly is compressed. Exercise care when removing the snap ring (12), washer (11) and spring (10).

c. Remove the snap ring (12), washer (11) and spring (10) from the forward end of the tube assembly (3).

d. Remove the forward floating piston (9) from the tube assembly (3) by threading a long 10-32 bolt into the piston and pulling out.

e. Remove the rod end fitting (1) from the aft end of the tube assembly (3) and thread a 10-32 bolt into the rod end of the piston until the aft floating piston (4) is engaged.

f. Pull the bolt until the aft floating piston (4) moves to the fully extended position.



1. ROD END FITTING
2. JAM NUT
3. PISTON ROD TUBE ASSY
4. AFT FLOATING PISTON
5. BARREL
6. PISTON RETAINING PIN
7. O-RING
8. PISTON
9. FORWARD FLOATING PISTON
10. SPRING
11. WASHER
12. SNAP RING

2000-209-02

Nose Landing Gear Shimmy Damper Servicing (Effectivity: All)
Figure 3

NOTE

A pair of vise grips may be installed on the bolt shank to hold the aft floating piston in position for proper servicing.

g. Fill the tube assembly (3) approximately 1/2 inch from the top with hydraulic fluid (1, Chart 2, 32-00-00).

h. Allow the aft floating piston (4) to move from the fully extended postion until the fluid level is even with the end of the piston rod opening.

i. While holding the aft floating piston (4) stationary, insert the forward floating piston (9) into the tube assembly (3). Extend the aft floating piston (4) to draw the forward floating piston (9) into the tube assembly (3). Use care to prevent any air from being trapped in the tube assembly (3).

j. Install the spring (10) on the forward piston (9).

k. Install the washer (11) and snap ring (12) to retain the spring (10) and forward piston (9) in the tube assembly (3).

l. Remove the 10-32 bolt from the aft floating piston (4) and reinstall the rod end fitting (1).

m. Refer to NOSE GEAR SHIMMY DAMPER INSTALLATION and reinstall the shimmy damper on the airplane.

NOSE GEAR SHIMMY DAMPER REMOVAL (Effectivity: All) (Figure 4)

a. Remove all electrical power from the airplane and pull the landing gear circuit breaker.

b. Use the alternate extension hand pump in the cockpit to open the nose landing gear doors, then stow the hand pump handle and pin the nose gear doors in the open position as outlined in Chapter 32-20-00.

c. Remove the nut (12), washer (11) and bolt (8) from the steering bell crank (10).

d. Remove the nut (4), washer (3), bolt (1), washers (103 and 104), bushing (5) and shimmy damper (6) from the nose wheel well.

NOSE GEAR SHIMMY DAMPER INSTALLATION (Effectivity: All) (Figure 4)

a. Insert the bushing (5), then align the washers (103 and 104) and shimmy damper (6) with the mounting bracket (2). Secure the assembly by installing the bolt (1), washer (3) and nut (4).

NOTE

The nut (4) should be tightened just enough to remove any noticeable end play from the bolt (1).

b. Loosen the jam nut (7) and adjust the rod end (9) until it is within one-half turn of matching the steering bell crank (10) attach point. Tighten the jam nut (7) to secure the shimmy damper rod end (9).

c. Install bolt (8), washer (11) and nut (12) to secure the assembly.

d. Remove the lock pin and push the nose gear doors closed.

e. Close the landing gear circuit breaker and restore electrical power to the airplane.

NOSE GEAR STEERING SPRING LINK REMOVAL (Effectivity: All) (Figure 4)

a. Remove all electrical power from the airplane and pull the landing gear circuit breaker.

b. Use the alternate extension hand pump in the cockpit to open the nose gear doors, then stow the hand pump handle and pin the nose gear doors in the open position as outlined in Chapter 32-20-00.

c. Refer to Chapter 53-30-00 and remove the LH forward nose skin panel (211AL).

d. Refer to Chapter 53-20-00 and remove the LH forward cockpit floorboard (141AT) beneath the pilot's rudder pedals.

e. Remove the forward rod end of the spring link (46) from the idler bell crank (56) by removing nut (57), washer (58) and bolt (59).

f. Remove bolts (92) and washers (89), then remove the cover (91) and gasket (90) from the pressure seal assembly (87) to access the aft end of the spring link (46).

g. Remove cotter pin (34), nut (35), washer (36) and bolt (39), then remove the spring link assembly (46) from the nose wheel well.

NOTE

Perform the next step only if a visual inspection indicates the need for further disassembly of the spring link (46).

h. Remove nuts (44 and 50), washers (43 and 49), bolts (41 and 47), collars (42 and 48) and springs (45) to disassemble the the spring link (46)

NOSE GEAR STEERING SPRING LINK INSTALLATION (Effectivity: All) (Figure 4)

NOTE

Begin installation procedures with step b if the spring link (46) is already assembled.

a. Install the springs (45) and collars (42 and 48) on the spring link (46), then secure with bolts (41 and 47), washers (43 and 49) and nuts (44 and 50).

b. Set the rudder pedals in the neutral position and install rig pin as described in Chapter 27-20-00. Position the spring link assembly (46) between the idler bell crank (56) and the upper bell crank (40) on the forward pressure bulkhead bell crank assembly. Loosen the jam nuts (37 and 60) and adjust the rod ends of the spring link until each is within one-half turn of being directly aligned with its respective bell crank attach point.

c. Tighten the jam nuts (37 and 60) and attach the spring link (46) with bolts (39 and 59), washers (36 and 58) and nuts (35 and 57).

NOTE

Tighten the nuts (35 and 57) just enough to remove any noticeable end play from the bolts (39 and 59).

d. Install a cotter pin (34) on the nut (35).

e. Align the gasket (90) and cover (91) with the forward bulkhead pressure seal assembly (87), then secure with bolts (92) and washers (89).

f. Refer to Chapter 53-30-00 and install the LH forward nose skin panel (211AL).

g. Refer to Chapter 53-20-00 and install the LH forward cockpit floorboard (141AT).

h. Remove the lock pin and push the nose gear doors closed.

i. Restore electrical power to the airplane and close the landing gear circuit breaker.

j. If the nose landing gear requires alignment, refer to NOSE GEAR STEERING SYSTEM ALIGNMENT for the adjustment procedure.

IDLER BELL CRANK PUSH-PULL ROD REMOVAL (Effectivity: All) (Figure 4)

a. Remove all electrical power from the airplane and pull the landing gear circuit breaker.

b. Use the alternate extension hand pump in the cockpit to open the landing gear doors, then stow the hand pump handle and pin the nose gear doors in the open position as outlined in Chapter 32-20-00.

c. Refer to Chapter 53-30-00 and remove the LH forward nose skin panel (211AL).

d. Remove nut (55), washer (54) and bolt (52) to disconnect the push-pull rod end (53) from the idler bell crank (56).

e. Remove nut (29), washer (28) and bolt (27) from the steering bell crank (10), then remove the push-pull rod (17) from the nose wheel well.

IDLER BELL CRANK PUSH-PULL ROD INSTALLATION (Effectivity: All) (Figure 4)

a. Position the push-pull rod (17) between the steering and idler bell cranks (10 and 56).

b. Set the rudder pedals in the neutral position and install rig pin as outlined in Chapter 27-20-00. Loosen the jam nuts (16 and 51) and adjust the rod ends (15 and 53) until they are within one-half turn of matching their respective bell crank attach points. Tighten the jam nuts (16 and 51) to secure the rod ends (15 and 53).

c. Install bolts (27 and 52), washers (28 and 54) and nuts (29 and 55).

NOTE

Tighten the nuts (29 and 55) just enough to remove any noticeable end play from the bolts (27 and 52).

d. Remove the lock pin and push the nose gear doors closed.

e. Refer to Chapter 53-30-00 and install the LH forward nose skin panel (211AL).

f. Restore electrical power to the airplane and close the landing gear circuit breaker.

IDLER BELL CRANK REMOVAL (Effectivity: All) (Figure 4)

a. Remove all electrical power from the airplane and pull the landing gear circuit breaker.

b. Use the alternate extension hand pump in the cockpit to open the nose gear doors, then stow the hand pump handle and pin the nose gear doors in the open position as outlined in Chapter 32-20-00.

c. Refer to Chapter 53-30-00 and remove the LH forward nose skin panel (211AL).

d. Remove bolt (59), washer (58) and nut (57) to disconnect the spring link (46) from the idler bell crank (56).

e. Remove nut (55), washer (54) and bolt (52) to disconnect the push-pull rod (17) from the idler bell crank (56).

f. Remove cotter pin (69), nut (68), washer (67) and bolt (63), then pull the idler bell crank (56) away from the brackets (61 and 62).

g. Remove the bearings (64 and 66) and bushing (65) from the idler bell crank (56).

IDLER BELL CRANK INSTALLATION
(Effectivity: All)
(Figure 4)

a. Insert the bushing (65) and bearings (64 and 66) into the idler bell crank (56).

b. Align the idler bell crank (56) with the brackets (61 and 62), and install the bolt (63), washer (67), nut (68) and cotter pin (69).

c. Reconnect the push-pull rod (17) to the idler bell crank (56) by installing bolt (52), washer (54) and nut (55).

d. Reconnect the spring link (46) to the idler bell crank (56) by installing bolt (59), washer (58) and nut (57).

e. Refer to Chapter 53-30-00 and install the LH forward nose skin panel (211AL).

f. Remove the lock pin and push the nose gear doors closed.

g. Restore electrical power to the airplane and close the landing gear circuit breaker.

h. If the nose landing gear requires alignment, refer to NOSE GEAR STEERING SYSTEM ALIGNMENT for the adjustment procedure.

STEERING BELL CRANK REMOVAL
(Effectivity: All)
(Figure 4)

a. Remove all electrical power from the airplane and pull the nose landing gear circuit breaker.

b. Use the alternate extension hand pump in the cockpit to open the nose landing gear doors, then stow the hand pump handle and pin the nose the nose gear doors in the open position as outlined in Chapter 32-20-00.

c. Disconnect the shimmy damper (6) from the steering bell crank (10) by removing bolt (8), washer (11) and nut (12).

d. Disconnect the idler bell crank push-pull rod (17) from the steering bell crank (10) by removing bolt (27), washer (28) and nut (29).

e. Remove the cotter pin (22), nut (23), washer (24), bolt (33) and washer (32).

f. Remove the steering bell crank (10) from the pivot fitting (25).

g. Remove the bearings (26 and 31), and bushing (30) from the steering bell crank (10).

h. Remove the cotter pins (21), nuts (20), washers (14 and 19), bolts (13) and cam (18) from the steering bell crank (10).

STEERING BELL CRANK INSTALLATION
(Effectivity: All)
(Figure 4)

a. Align the cam (18) with the steering bell crank (10) and secure with bolts (13), washers (14 and 19), nuts (20) and cotter pins (21).

b. Insert the bushing (30) and bearings (26 and 31) into the steering bell crank (10).

c. Align the steering bell crank with the pivot fitting (25) and secure with the bolt (33), washers (32 and 24), nut (23) and cotter pin (22).

d. Align the idler bell crank push-pull rod end (15) with the steering bell crank (10) and secure with bolt (27), washer (28) and nut (29).

e. Align the shimmy damper rod end (9) with the steering bell crank (10) and secure with bolt (8), washer (11) and nut (12).

f. Remove the lock pin and push the nose gear doors closed.

g. Restore electrical power to the airplane and close the landing gear circuit breaker.

h. If the nose landing gear requires alignment, refer to NOSE GEAR STEERING SYSTEM ALIGNMENT for the adjustment procedure.

FORWARD PRESSURE BULKHEAD BELL CRANK ASSEMBLY REMOVAL
(Effectivity: All)
(Figure 4)

a. Remove all electrical power from the airplane and pull the landing gear circuit breaker.

b. Use the alternate extension hand pump in the xockpit to open the nose landing gear doors, then stow the hand pump handle and pin the nose gear doors in the open position as outlined in Chapter 32-20-00.

c. Refer to Chapter 53-20-00 and remove the LH forward cockpit floorboard (141AT).

d. Remove bolts (92) and washers (89), then remove the gasket (90) and cover (91) from the forward bulkhead pressure seal assembly (87).

e. Remove cotter pin (34), nut (35), washer (36) and bolt (39) to disconnect the spring link (46) from the upper bell crank (40).

f. Remove cotter pin (97), nut (96), washer (95) and bolt (93) to disconnect the push-pull rod (98) from the lower bell crank (82).

g. Remove bolts (79), washers (78) and nuts (77), then remove the housing assembly (74) from the forward bulkhead pressure seal assembly (87).

h. Remove bolt (83), washer (81) and nut (80), to separate the bell cranks (40 and 82).

i. Remove nuts (76), washers (75), bolts (71) and keeper plate (72) from the housing assembly (74).

j. Remove the bearings (73, 99 and 100), from the housing assembly (74) as outlined by procedures in Chapter 20-00-05.

FORWARD PRESSURE BULKHEAD BELL CRANK ASSEMBLY INSTALLATION (Effectivity: All) (Figure 4)

a. Apply retaining compound (4, Chart 2, 32-00-00) to the housing assembly (74) bearing mating surfaces and to the outer mating surface on each bearing.

b. Follow the general guidelines provided in Chapter 20-00-05 to properly install the bearings (73, 99 and 100) in the housing assembly (74).

c. Align the keeper plate (72) with the housing assembly (74) and secure with bolts (71), washers (75) and nuts (76).

d. Insert the bell cranks (40 and 82) into the housing assembly (74), align the shafts, and install bolt (83), washer (81) and nut (80).

e. Position the bell crank and housing assembly in the forward bulkhead pressure seal assembly (87) and install bolts (79), washers (78) and nuts (77).

f. Align the push-pull rod (98) with the lower bell crank (82) and secure with bolt (93), washer (95), nut (96) and cotter pin (97).

NOTE

Tighten the nut just enough to remove any noticeable end play from the bolt.

g. Align the spring link (46) with the upper bell crank (40) and secure with bolt (39), washer (36), nut (35) and cotter pin (34).

h. Align the gasket (90) and cover (91) with the forward bulkhead pressure box (87) and secure with bolts (92) and washers (89).

i. Refer to Chapter 53-20-00 and install the LH forward cockpit floorboard(141).

j. Remove the lock pin and push the nose gear doors closed.

k. Restore electrical power to the airplane and close the landing gear circuit breaker.

l. If the nose landing gear requires alignment, refer to NOSE GEAR STEERING SYSTEM ALIGNMENT for the adjustment procedure.

NOSE GEAR STEERING DISCONNECT ACTUATOR REMOVAL (Effectivity: All) (Figure 4)

a. Remove all electrical power from the airplane and pull the nose wheel steering disconnect circuit breaker.

b. Refer to Chapter 53-20-00 and remove the LH forward cockpit floorboard (141AT).

c. Tag and disconnect the electrical connector from the actuator (112).

d. Remove cotter pin (111), nut (110), washer (109) and bolt (108) to disconnect the actuator (112) from the steering system linkage.

e. Remove cotter pin (116), nut (115), washer (114) and bolt (113).

f. Remove the actuator (112) from the LH rudder sector.

NOSE GEAR STEERING DISCONNECT ACTUATOR INSTALLATION (Effectivity: All) (Figure 4)

NOTE

Before mounting the actuator (112), ground pin B and apply 28 vdc to pin C until the actuator is fully extended. When fully extended, the actuator should measure 7.92 ±.03 inches from the center of one actuator rod end to the other.

a. Align the actuator (112) with the LH rudder sector and install bolt (113), washer (114), nut (115) and cotter pin (116).

b. Align the actuator rod end within one-half turn of the bell crank (102) attach point.

c. Apply threadlock adhesive (5, Chart 2, 32-00-00) to the threads on the actuator rod end, then tighten the jam nut.

d. Secure the actuator rod end to the bell crank (102) with bolt (108), washer (109), nut (110) and cotter pin (111).

e. Reconnect the electrical connector to the actuator (112) according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

f. Verify that total actuator length of travel is approximately 1.82 inches (calulated by measuring the difference in exposed actuator rod from the fully extended to the fully retracted position). If not, it may be necesary to repair or replace the actuator assembly.

g. Refer to Chapter 53-20-00 and install the LH forward cockpit floorboard (141AT).

h. Restore electrical power to the airplane and close the nose wheel steering disconnect circuit breaker.

NOSE GEAR STEERING SYSTEM ALIGNMENT (Effectivity: All) (Figure 4)

Perform alignment procedures anytime a steering component has been overhauled or replaced, or when a visual inspection indicates that the nose gear assembly is not properly aligned. The rudder control system should be properly adjusted before adjusting the steering system.

a. Place the airplane on jacks as instructed in Chapter 7-10-00 and allow the landing gear struts to extend.

b. Center the rudder pedals and install rig pin as outlined in Chapter 27-20-00.

c. Disconnect the idler bell crank push-pull rod (17) from the idler bell crank (56) by removing bolt (52), washer (54) and nut (55).

d. Loosen the jam nut (51) and adjust the rod end (53) so that the nose gear tire points straight ahead (parallel with the centerline of the airplane) when the push-pull rod (17) is reconnected.

e. Tighten the jam nut (51) and reconnect the push-pull rod (17) to the idler bell crank (56) by installing bolt (52), washer (54) and nut (55).

WARNING

Personnel should remain clear of the nose wheel well during actuation of the nose steering disconnect system.

f. Compress the RH main landing gear shock strut more than 2 inches from the fully extended position. The actuator motor should extend the actuator rod to engage the steering linkage.

NOTE

Many times, the nose gear disconnect actuator will drive the nose wheel a couple degrees to one side as it actuates. If so, readjust nose gear alignment as outlined by the remaining steps.

Ensure that the rudder pedals remain centered for the entire adjustment procedure.

g. Approximate the number of degrees that the nose gear assembly traveled from its straight ahead position after engagement of the actuator. Divide this value in half and readjust the push-pull rod to return the nose gear assembly halfway back to the original straight ahead position.

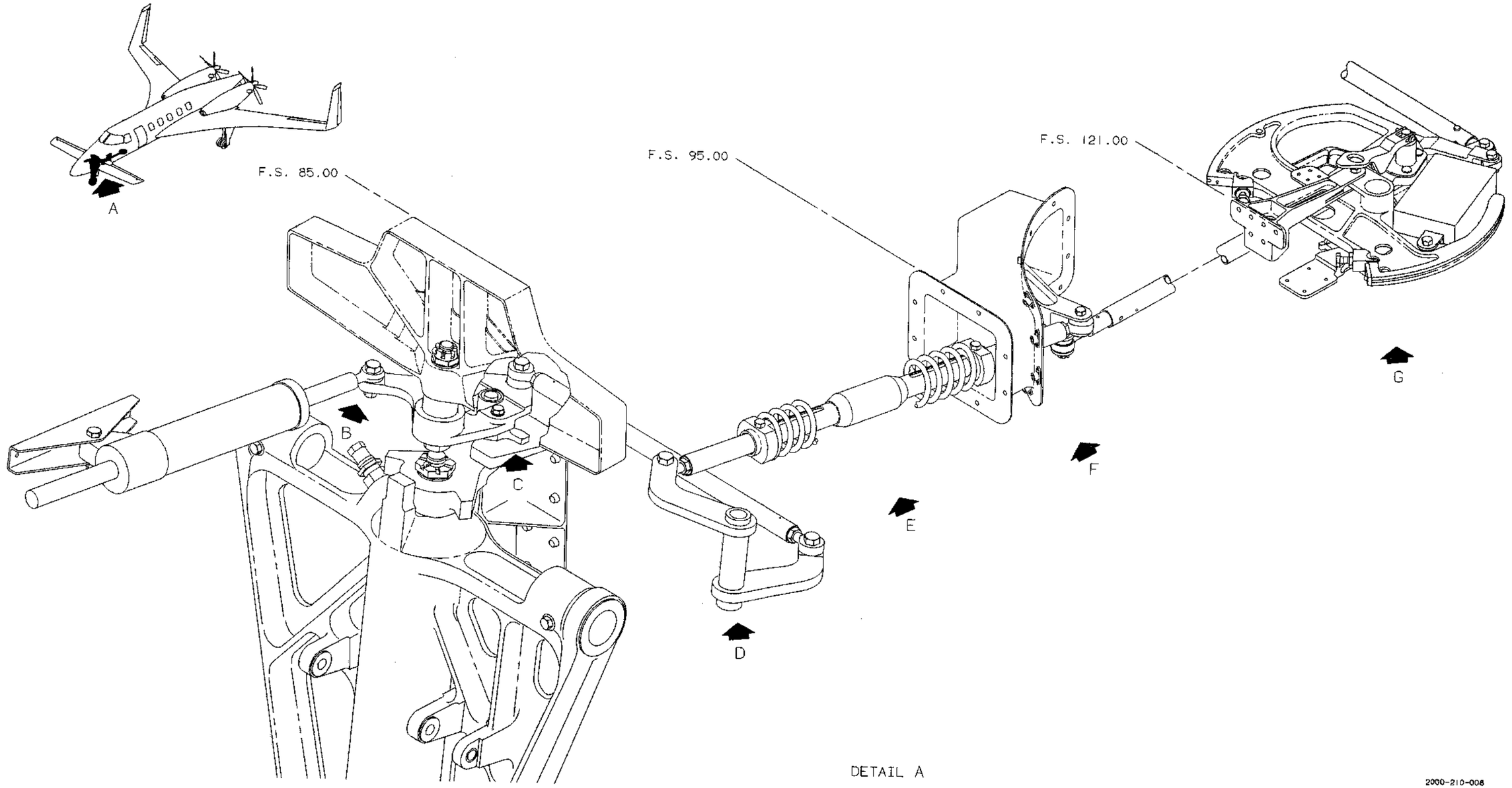
WARNING

Personnel should remain clear of the nose wheel well during actuation of the nose gear steering disconnect system.

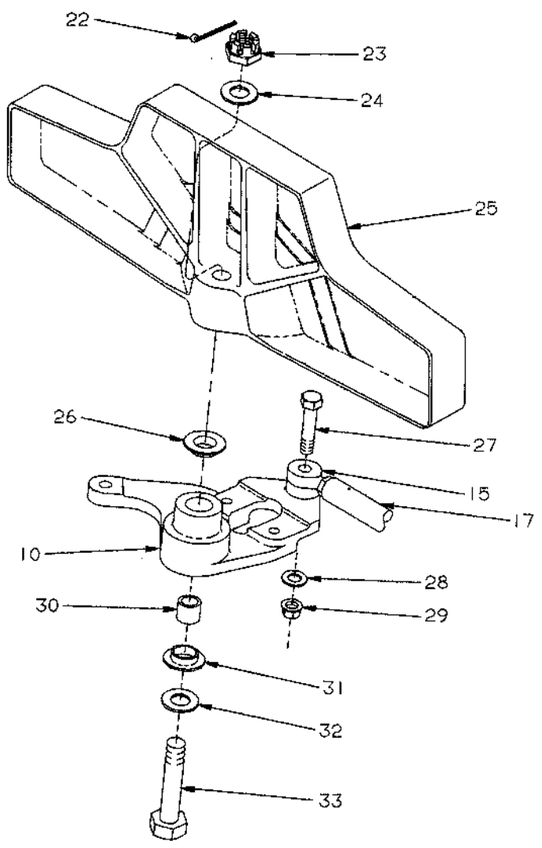
h. Allow the shock strut to fully extend. The disconnect actuator should retract and disengage steering action from the rudder pedals after the RH safety switch actuates.

i. Note the position of the nose gear assembly after disconnect. It shall be no more than one-half of the original distance noted in step g from center.

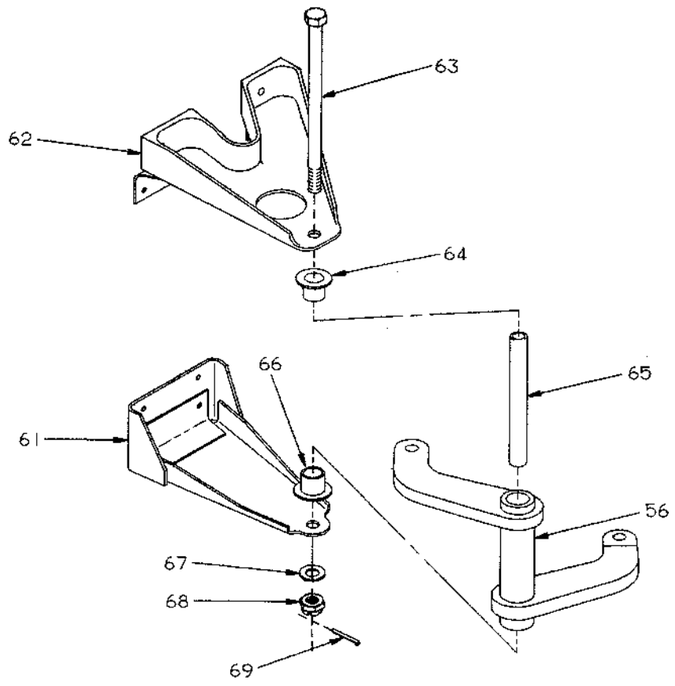
j. Remove the airplane from the jacks.



Nose Landing Gear Steering Installation
(Effectivity: All)
(Sheet 1 of 3)
Figure 4



DETAIL C



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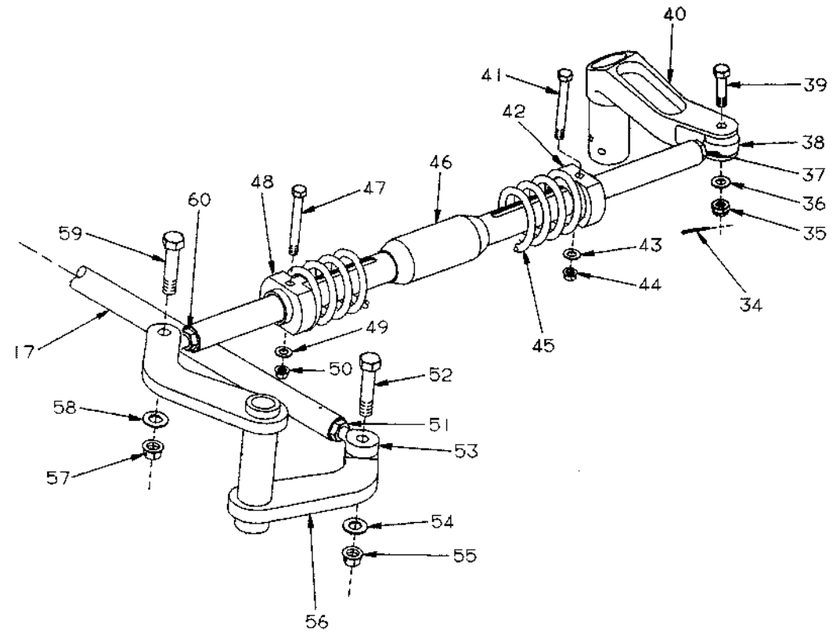

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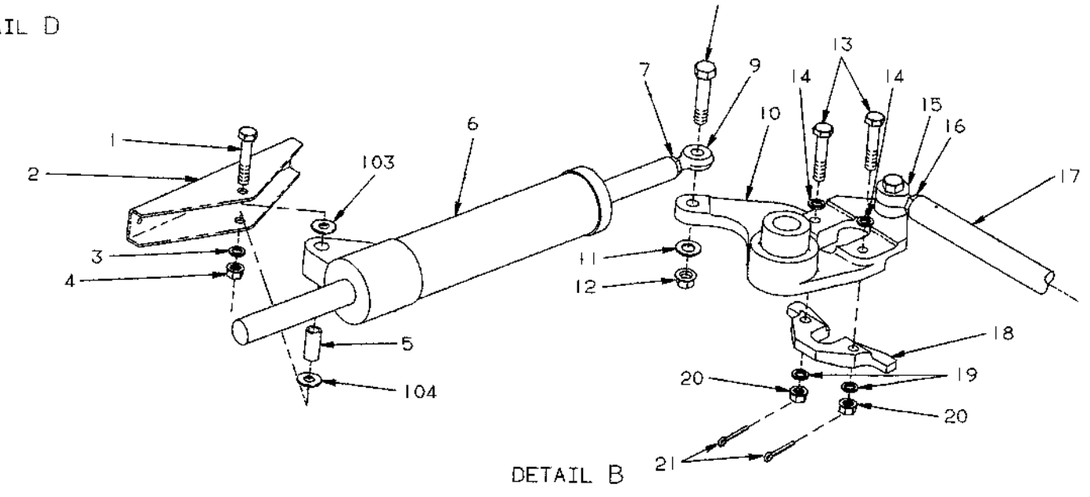
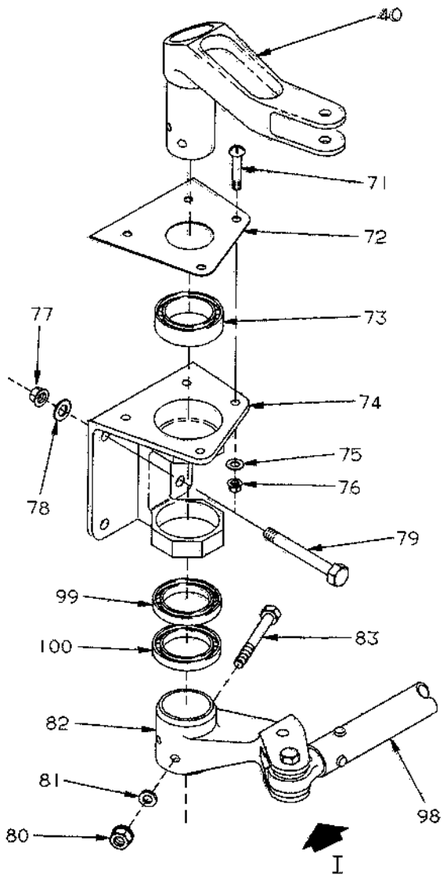


DETAIL B

2000-210-009

Nose Landing Gear Steering Installation
(Effectivity: All) (Sheet 2 of 3)
Figure 4



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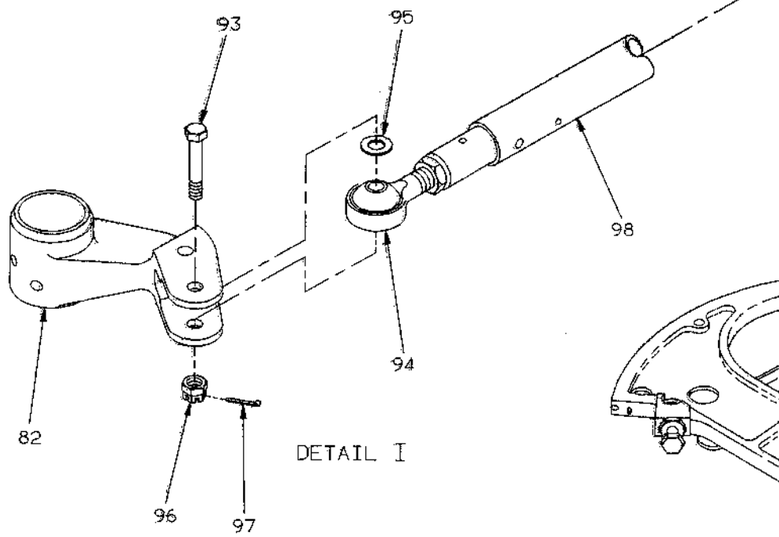
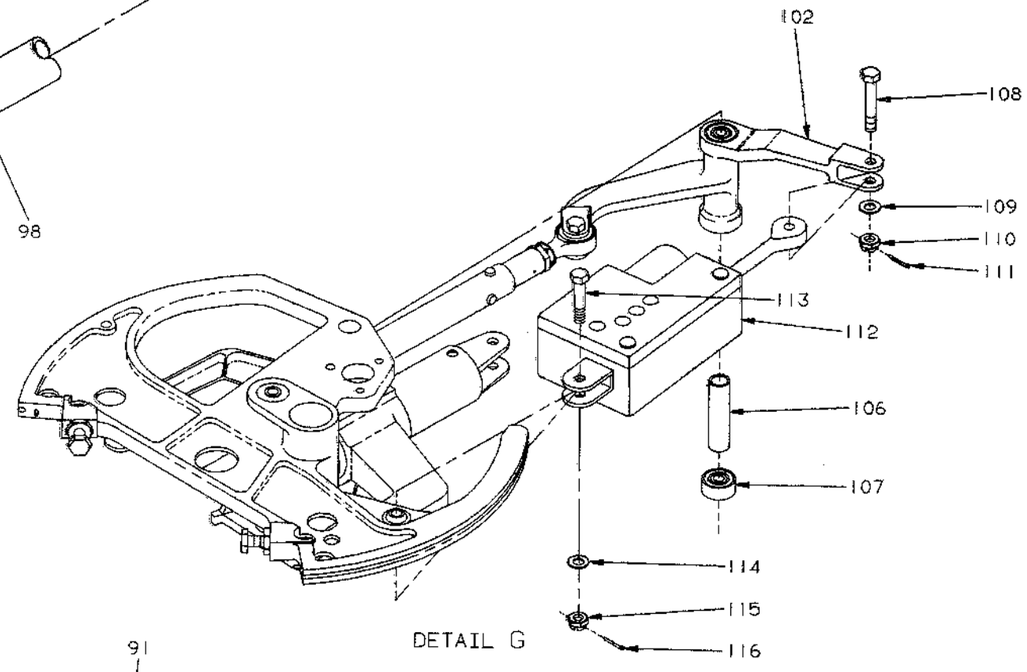
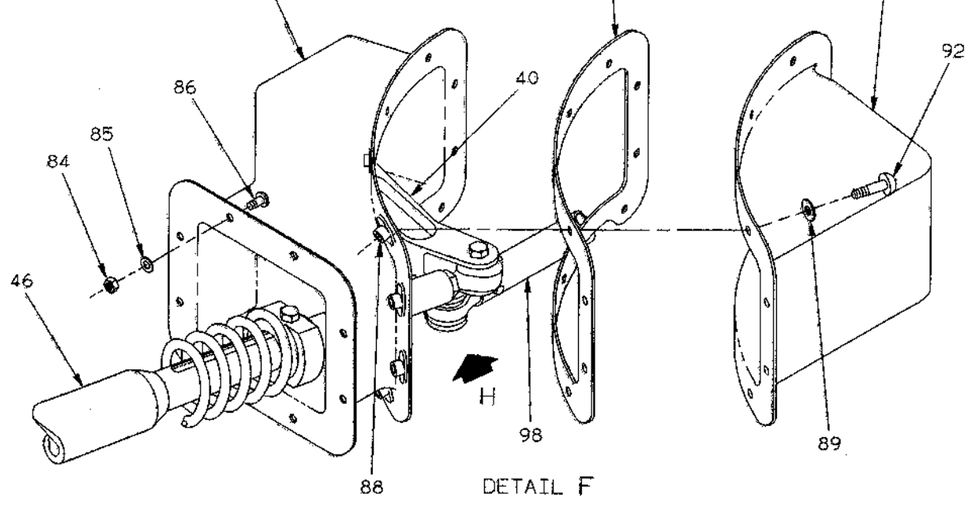


DETAIL I



DETAIL G



DETAIL F

2000-210-010

Nose Landing Gear Steering Installation (Effectivity: All)
(Sheet 3 of 3)
Figure 4

LANDING GEAR POSITION AND WARNING - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

LANDING GEAR POSITION (Effectivity: All)

Visual indication of landing gear position is provided by two red in-transit lights located in the landing gear control handle and by a green GEAR DOWN indicator light assembly, labeled NOSE - L - R, located adjacent to the landing gear control handle on the pilot's inboard subpanel. Illumination of the red in-transit lights indicates that the landing gear is in transit when the airplane is in the air, or that the landing gear control handle is in the UP position when the airplane is on the ground (refer also to LANDING GEAR WARNING for other conditions which will illuminate the in-transit lights). The in-transit lights are extinguished when all three landing gear assemblies are up and locked or down and locked. Illumination of the NOSE - L - R portions of the green GEAR DOWN light indicates that the respective landing gear is down and locked. The position lights can be checked by pressing the LDG GR TEST push-button switch located on the pilot's outboard subpanel.

Power for the in-transit and GEAR DOWN lights is supplied through a 5-ampere LANDING GEAR - IND circuit breaker located on the LH circuit breaker panel. Circuit ground for the in-transit and GEAR DOWN lights is through the ambient light sensor circuit on the annunciator controls pcb. Each landing gear has a downlock switch and an uplock switch. The main landing gear also has a squat switch located on each strut. A nose gear door open switch is provided to indicate when the forward nose gear door is open. A landing gear emergency select switch is provided to indicate when the landing gear alternate extension pump handle is not in a stowed position. Signals from all of these switches are sent to the landing gear indication pcb, where the system logic is located. Signals to illuminate the position lights are sent from the landing gear indication pcb and through the annunciator test and fire extinguisher pcb to the respective position lights.

When the airplane leaves the ground, the squat switches will open. When the landing gear control handle is moved to the UP position, the downlock switches will open as the landing gear retracts, the GEAR DOWN lights will extinguish, and the in-transit lights will illuminate. When all three uplock switches are actuated and the nose gear door open switch is open, the in-transit lights will extinguish, indicating gear-up.

When the airplane is preparing to land and the landing gear moves from the fully retracted position, the respective uplock switches will open and the in-transit lights will illuminate. When the lock link of each landing gear actuates its respective downlock switch, the in-transit lights will extinguish, indicating gear-down. In this position, the downlock switches will also close the circuit to illuminate their respective GEAR DOWN light (NOSE, L, or R).

If the green GEAR DOWN lights do not illuminate, the landing gear alternate extension hand pump in the cockpit (refer to Chapter 32-30-00) can be used to manually lower the landing gear. When the pump handle is raised out of its retaining clips, a landing gear emergency select switch is activated to send a signal to the landing gear indication pcb and to the Data Acquisition Unit (DAU). White EICAS messages L GEAR UP, R GEAR UP, and NS GEAR UP will be displayed when system logic is as shown on the schematic in Figure 2. The signal from the landing gear emergency select switch and a ground from the landing gear control handle when in the DN position will energize coil Y1-Y2 of relay K2 on the landing gear indication pcb. When coil Y1-Y2 is energized, the 28 vdc signal from the landing gear control pcb to the landing gear power relay will be interrupted, shutting off the landing gear pump motor. The landing gear can

then be lowered using the landing gear alternate extension pump handle.

After the landing gear alternate extension pump has been pumped enough to open the uplock switches on each landing gear, the following white EICAS messages will be displayed to aid the operator: L GEAR IN TRANS, R GEAR IN TRANS, and NS GEAR IN TRANS. After each landing gear is fully extended, the landing gear downlock switches will be closed, causing the white EICAS messages to no longer be displayed and the green GEAR DOWN lights to illuminate. The LDG GR TEST switch will also test these white EICAS messages.

CAUTION

Never rely on the squat switches to keep the gear down during taxiing, or landing rollout, or take-off roll. Always check the position of the landing gear control handle.

NOTE

To prevent accidental gear retraction on the ground, either of the squat switches on the main landing gear struts breaks the landing gear control circuit described in Chapter 32-30-00 whenever the strut is compressed.

The squat switches also actuate a solenoid-operated downlock hook, which is part of the landing gear control handle, to prevent the landing gear control handle from being moved to UP when the airplane is on the ground. The hook automatically unlocks when the airplane leaves the ground, but it can be manually overridden by pressing down on the red button placarded DOWN LOCK REL, located adjacent to the landing gear control handle. The right hand squat switch will complete a circuit to the in-transit lights through the landing gear indication pcb to illuminate the in-transit lights anytime the landing gear control handle is moved to the UP position while the airplane is on the ground with the shock strut compressed.

NOTE

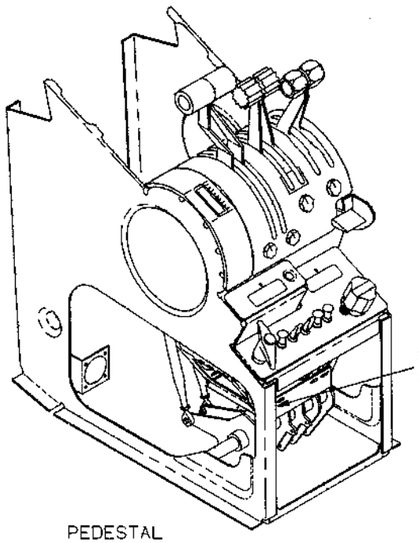
Power from the LANDING GEAR-IND circuit breaker is also supplied to the liquid level sensor in the landing gear power pack (refer to Chapter 32-30-00) to illuminate a yellow HYD FLUID LO message on the EICAS if the landing gear hydraulic fluid level becomes too low to extend the landing gear. The LDG GR TEST switch will also test this EICAS message.

LANDING GEAR WARNING (Effectivity: All)

A landing gear warning system is provided to warn the pilot that the landing gear is not down and locked during specific flight regimes. Power for the landing gear warning system is supplied through a 5-ampere LANDING GEAR-WARNING circuit breaker located on the LH circuit breaker panel. Two landing gear warning horn switches are installed in the pedestal and are actuated by cams on the power lever linkage whenever the power levers are retarded below 90% N1. An airspeed switch is installed in the pitot/static system in the RH forward nose compartment to sense when airspeed falls below 140 knots. A FLAPS/FWD WING switch, described in Chapter 27-50-00, indicates when the flaps and forward wing are in an extended position. These switches furnish signals to the system logic contained on the landing gear indication pcb. A landing gear test switch, located on the pilot's outboard subpanel and placarded LDG GR TEST, is furnished to test the horn signal.

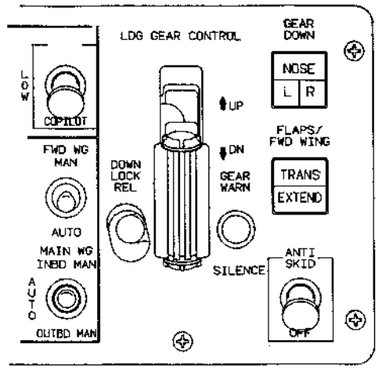
With the flaps and forward wings in the retracted position, airspeed below 140 knots, and either or both power levers retarded below 90% N1, the warning horn will sound intermittently and the landing gear control handle in-transit lights will illuminate. The warning horn can be silenced by depressing the GEAR WARN-SILENCE button located adjacent to the landing gear control handle, but the landing gear control handle in-transit lights cannot be extinguished. The landing gear warning system will be rearmed by sufficiently advancing the power lever(s).

With the flaps and forward wing extended, airspeed below 140 knots, and either or both power levers retarded below 90% N1, the warning horn and landing gear control handle in-transit lights will be activated and neither can be cancelled.



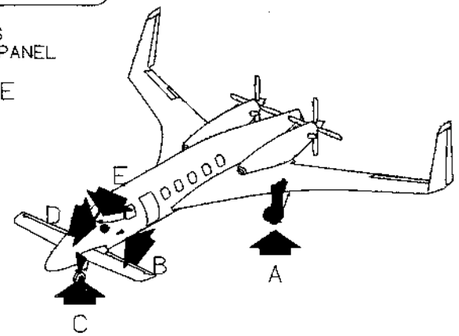
PEDESTAL

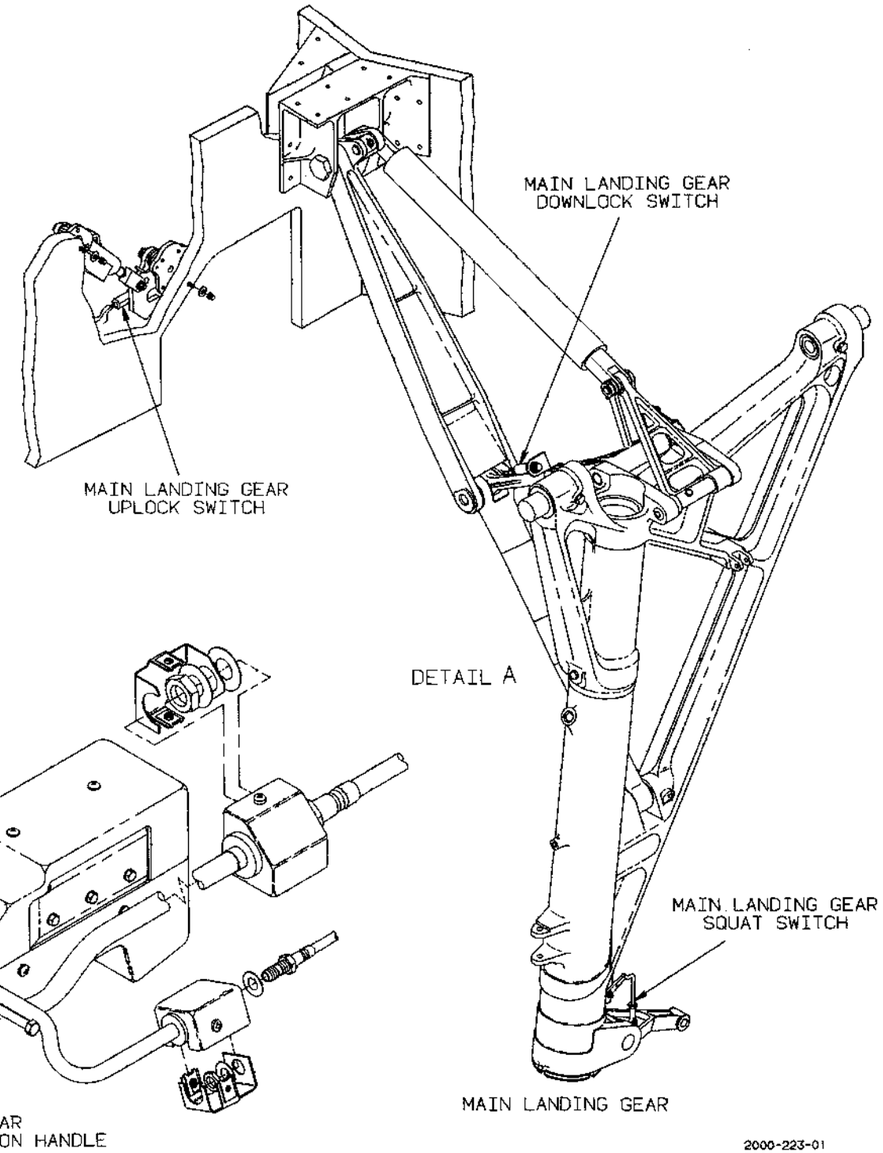
DETAIL D

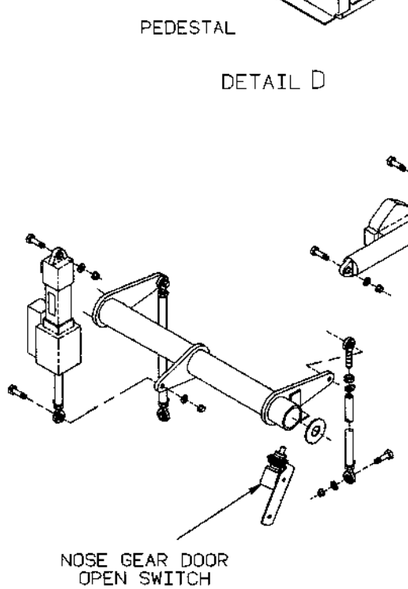


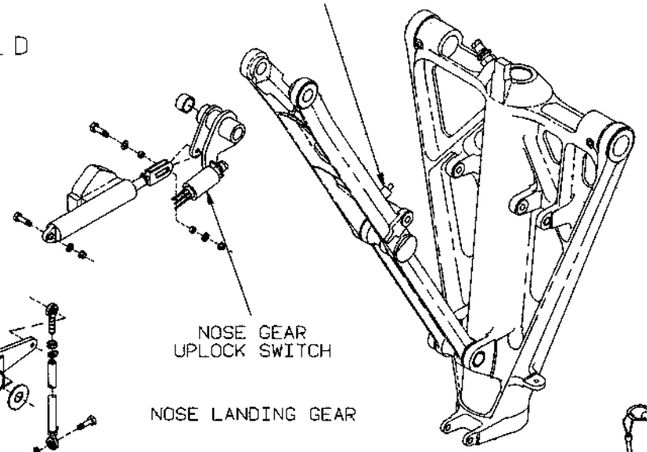
PILOT'S
INBOARD SUBPANEL

DETAIL E



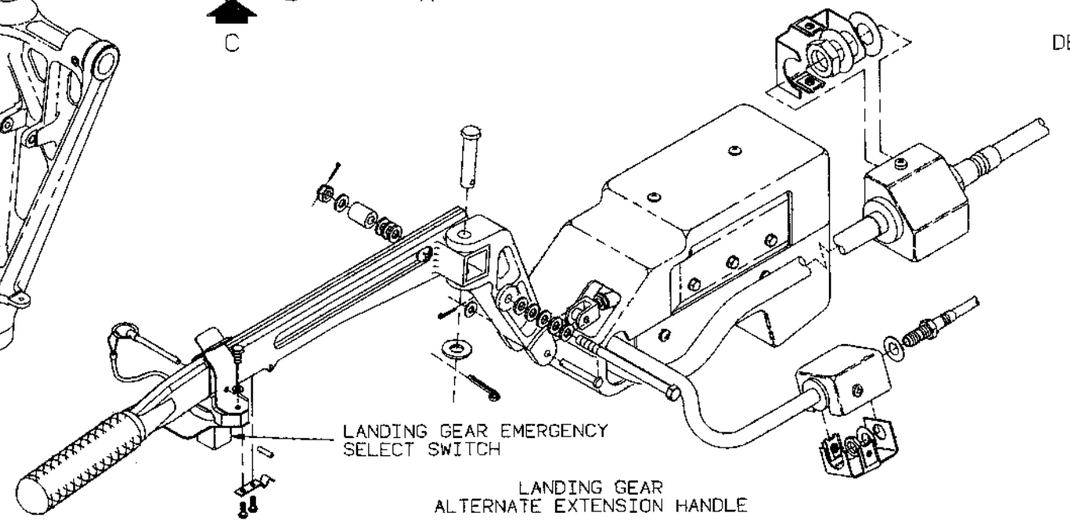






NOSE LANDING GEAR

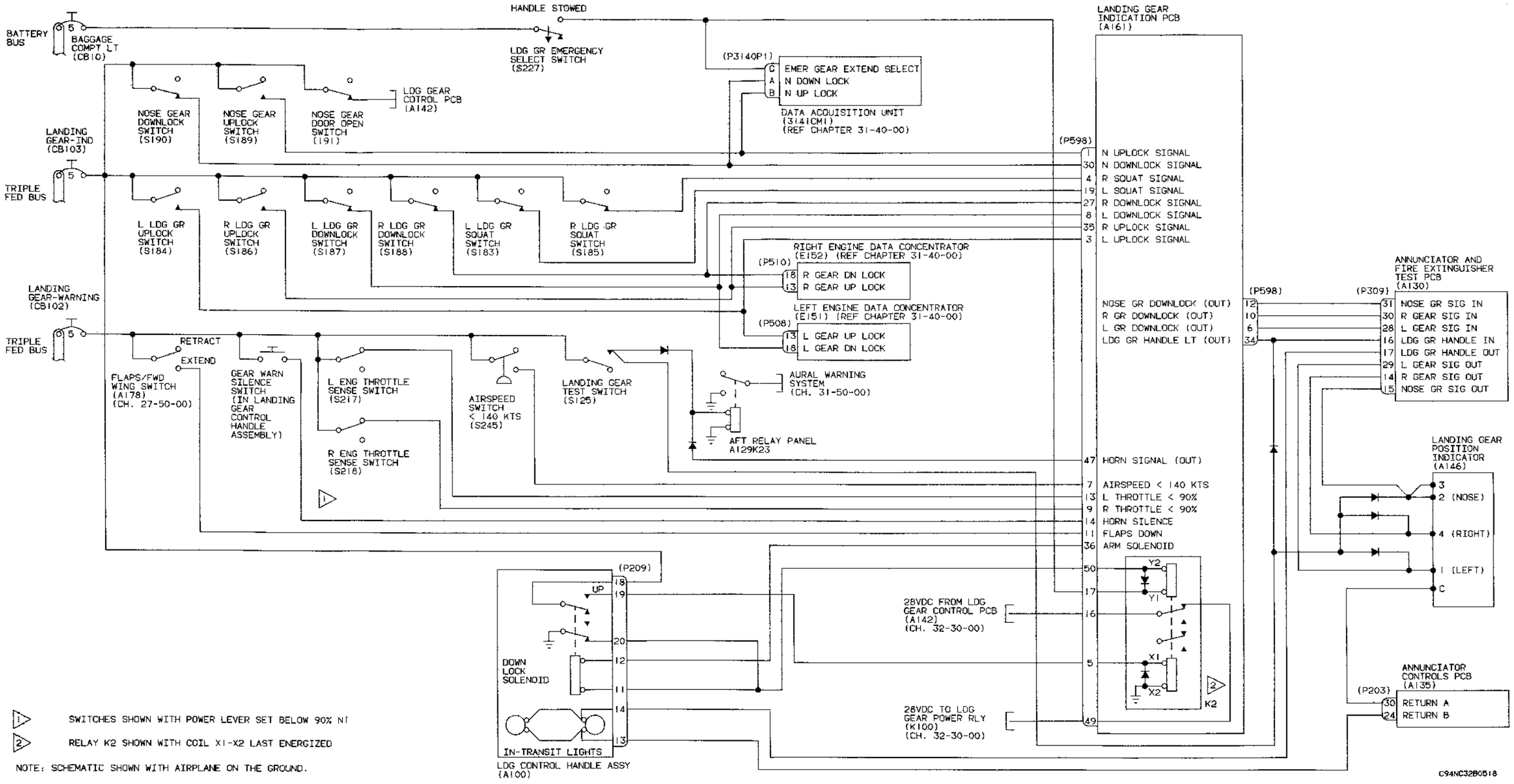
DETAIL C



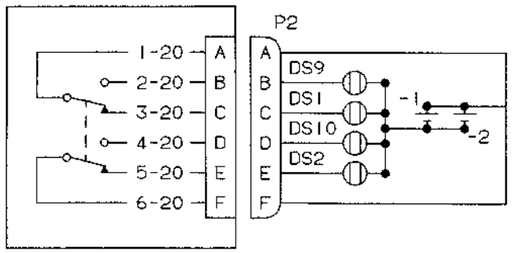
DETAIL B

2000-223-01

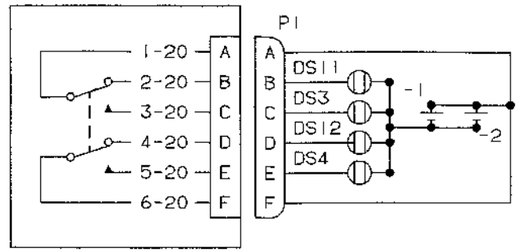
Landing Gear Position and Warning System
(Effectivity: All)
Figure 1



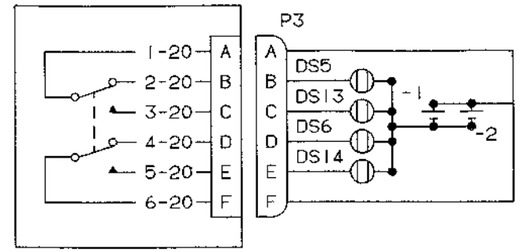
Landing Gear Position and Warning Schematic (Effectivity: All)
Figure 2



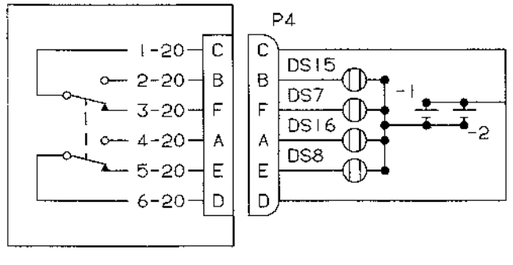
S190 NOSE GEAR DOWNLOCK SWITCH ASSY



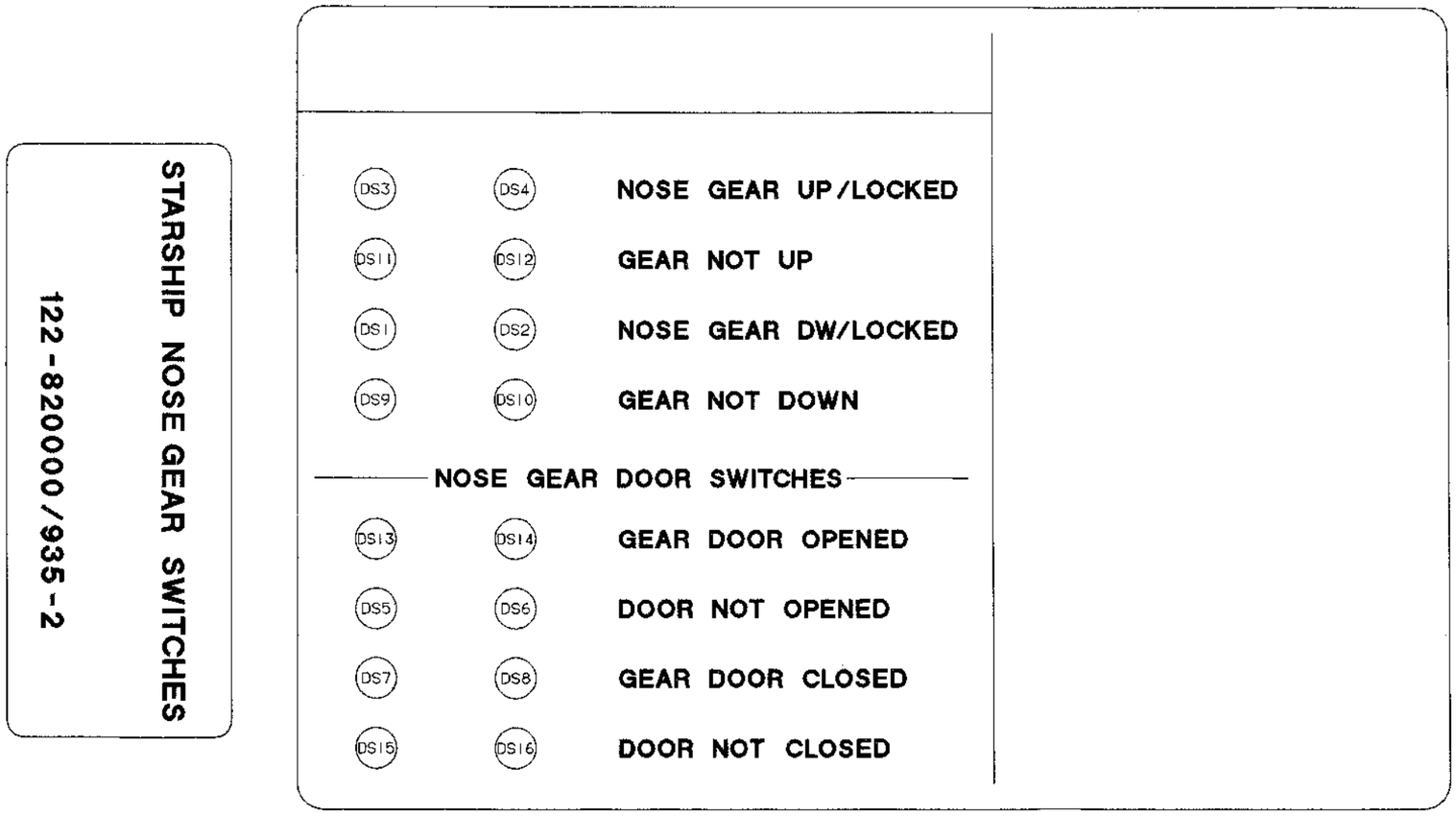
S189 NOSE GEAR UPLOCK SWITCH ASSY



S191 NOSE GEAR DOOR OPEN SWITCH



S192 NOSE GEAR DOOR CLOSED SWITCH



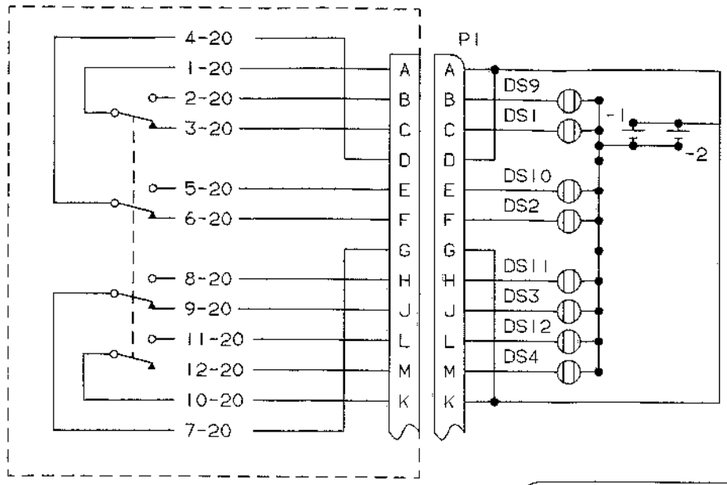
GENERAL NOTES

1. IDENTIFY TOOL WITH NAME AND PART NUMBER PER BEST SHOP PRACTICE.
2. USE #22 A.W.G. STRANDED, INSULATED WIRE FOR ALL HOOKUP UNLESS OTHERWISE NOTED.
3. MAKE LENGTH OF HARNESS 7 FEET IN LENGTH AND ENCLOSE IN EXPANDO SLEEVING.
4. SUBSTITUTION OF PARTS AND MATERIAL TO BE FUNCTIONALLY AND PHYSICALLY INTERCHANGABLE.
5. ALL SWITCHES ARE SHOWN WITH GEAR DOWN AND LOCKED.
6. WEIGHT ON WHEELS.
7. NOSE GEAR DOOR CLOSED.
8. UPLOCK SWITCHES ACTUATE WITH GEAR UP.
9. DOWNLOCK SWITCHES ACTUATE WITH GEAR DOWN.

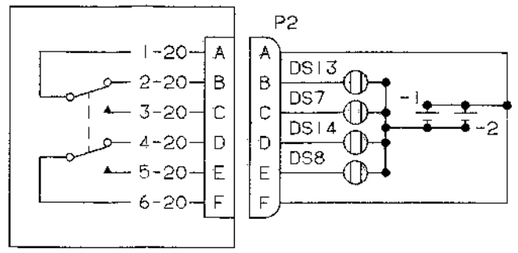
DESIGNATOR	QUANTITY	NOMENCLATURE
P4	1	CONNECTOR MS3476L-10SW
P3	1	CONNECTOR MS3476L-10S
P2	1	CONNECTOR MS3476L14-12SW
P1	1	CONNECTOR MS3476L14-12S
-2	8	BATTERIES SIZE "D" DRY CELL
DS1-8	8	LAMP HOLDER MS25256-4 (GREEN)
DS9-16	16	LAMP HOLDER MS25256-6 (RED)
-3	1	ZERO CARRY CASE ZC7110 9.19 X 13.25

C94NC32B0189

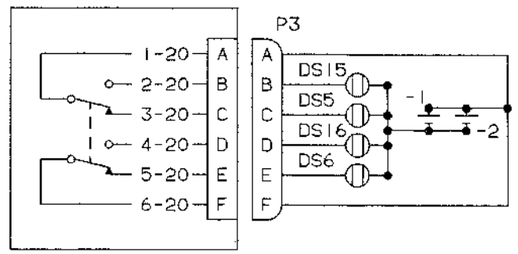
Nose Landing Gear Test Box (Effectivity: All)
Figure 3



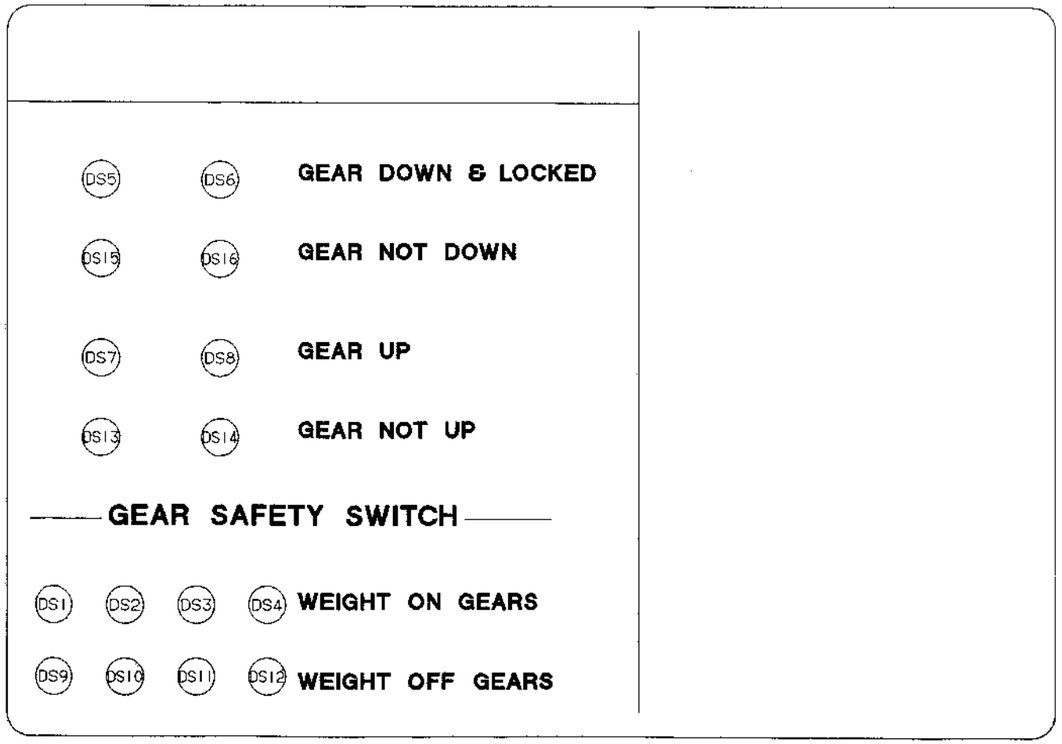
S183/S185 LANDING GEAR SAFETY SWITCH



S184/S186 MAIN GEAR UPLOCK SWITCH ASSEMBLY



S187/S188 MAIN GEAR DOWNLOCK SWITCH ASSEMBLY



GENERAL NOTES

1. IDENTIFY TOOL WITH NAME AND PART NUMBER PER BEST SHOP PRACTICE.
2. USE #22 A.W.G. STRANDED, INSULATED WIRE FOR ALL HOOKUP UNLESS OTHERWISE NOTED.
3. MAKE LENGTH OF HARNESS 7 FEET IN LENGTH AND ENCLOSE IN EXPANDO SLEEVING.
4. SUBSTITUTION OF PARTS AND MATERIAL TO BE FUNCTIONALLY AND PHYSICALLY INTERCHANGABLE.
5. ALL SWITCHES ARE SHOWN WITH GEAR DOWN AND LOCKED.
6. UPLOCK SWITCHES ARE ACTUATED WITH GEAR UP.
7. DOWNLOCK SWITCHES ACTUATED WITH GEAR DOWN.
8. SAFETY SWITCH SHOWN WITH WEIGHT ON GEARS.

DESIGNATOR	QUANTITY	NOMENCLATURE
P3	1	CONNECTOR MS3476L10-6SW
P2	1	CONNECTOR MS3476L10-6S
P1	1	CONNECTOR MS3476L14-18S
-1	3	BATTERY HOLDER KEYSTONE TYPE #2176
-2	6	BATTERIES SIZE "D" DRY CELL
DS1-DS8	8	LAMP HOLDER MS25256-4 (GREEN)
DS9-DS16	8	LAMP HOLDER MS25256-6 (RED)
-3	1	ZERO CARRY CASE ZC7110 9.19 X 13.25

C94NC32B0187

STARSHIP MAIN GEAR SWITCHES
122-810001 /935-1

Main Landing Gear Test Box (Effectivity: All)
Figure 4

CHART 1
TROUBLESHOOTING - LANDING GEAR WARNING HORN AND SWITCHES
(Effectivity: All)

Step 1	Is warning horn inoperative or malfunctioning?	**YES**	Proceed to Step 2.
Step 2	Is LANDING GEAR-WARNING circuit breaker engaged?	**YES**	Proceed to Step 3.
	NO		
	Engage LANDING GEAR-WARNING circuit breaker.		
Step 3	Is there continuity from circuit breaker to system switches to ground?	**YES**	Proceed to Step 4.
	NO		
	Repair open or grounded circuit.		
Step 4	Are system switches operational and in adjustment?	**YES**	Proceed to Step 5.
	NO		
	Replace or adjust system switches.		
Step 5	Are main or nose gear downlock switches operational and in adjustment?	**YES**	Operationally check system.
	NO		
	Replace or adjust main or nose gear downlock switches.		

CHART 2
TROUBLESHOOTING - LANDING GEAR POSITION INDICATORS AND SWITCHES
(Effectivity: All)

Step 1	Are all indicators inoperative?	**YES**	Engage LANDING GEAR-IND circuit breaker.
	NO		
	Proceed to Step 2.		
Step 2	Is electrical wiring OK between circuit breaker and indicators?	**YES**	Proceed to Step 3.
	NO		
	Repair open or shorts in the electrical wiring.		
Step 3	Is only one indicator inoperative?	**YES**	Proceed to Step 4.

CHART 2
TROUBLESHOOTING - LANDING GEAR POSITION INDICATORS AND SWITCHES
(Effectivity: All) *(Continued)*

	NO	
	Inspect landing gear switches for a defective switch or a switch out of adjustment.	
Step 4	Are landing gear switches operational and in adjustment?	**YES** Proceed to Step 5.
	NO	
	Replace or adjust system switches.	
Step 5	If one indicator is still inoperative, check for an open circuit or a defective indicator.	

LANDING GEAR POSITION AND WARNING - MAINTENANCE PRACTICES (Effectivity: All)

It is recommended that a test circuit be used to verify proper operation of the downlock, uplock, and squat switches installed on the nose gear and main gear. Schematics and materials required to build the test circuits are shown in Figures 3 and 4. Completed test boxes can also be purchased from BEECHCRAFT Commercial Product Support Parts and Equipment Marketing.

NOSE LANDING GEAR UPLOCK SWITCH ADJUSTMENT (Effectivity: All) (Figure 5)

NOTE

Always use a continuity checker to ascertain switch actuation. DO NOT depend upon the click of the switch as an indication of actuation.

Before adjusting the uplock switch, the landing gear must be rigged correctly as instructed in Chapter 32-30-00.

CAUTION

Use of a Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00) or it's equivalent, is recommended for retracting or extending the landing gear while adjusting the uplock switch. The procedure for using the pump can be found under LANDING GEAR RIGGING in Chapter 32-30-00.

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Disable the nose gear forward doors by disconnecting the pushrods from the actuating cams.

c. Ensure that the landing gear is fully retracted before adjusting the switch.

d. Disconnect the switch connector from airplane wiring and connect the switch connector to plug P1 on the nose gear test box (10, Chart 1, 32-00-00) shown in Figure 3.

e. Remove safety wire and loosen the locking nuts (3 and 6) that hold the switch in position. Adjust the switch so that the green lights (DS3 and DS4) on the test box illuminate. If using an ohmmeter (11, Chart 1, 32-00-00) it should indicate continuity when the wire leads are connected to pins E and F.

f. Adjust the switch so that the switch bushing (2) is 0.01 to 0.04 inch from the uplock roller (7) and secure the switch using the locking nuts (3 and 6) and safety wire.

NOTE

Excessive compression or bottoming out of the switch plunger (1) may cause damage to the switch.

g. Using the pump (4, Chart 1, 32-00-00), cycle the gear to unlock and back to lock while monitoring the lights on the test box.

h. As the gear is cycled to unlock, the red lights (DS11/DS12) on the test box, should illuminate. If using an ohmmeter it should indicate an open circuit.

i. Disconnect the switch connector from the test box and connect the switch connector to the airplane wiring.

j. Check for proper operation of the in-transit lights located in the landing gear control switch handle. Oporate the gear through at least one complete cycle with the power pack.

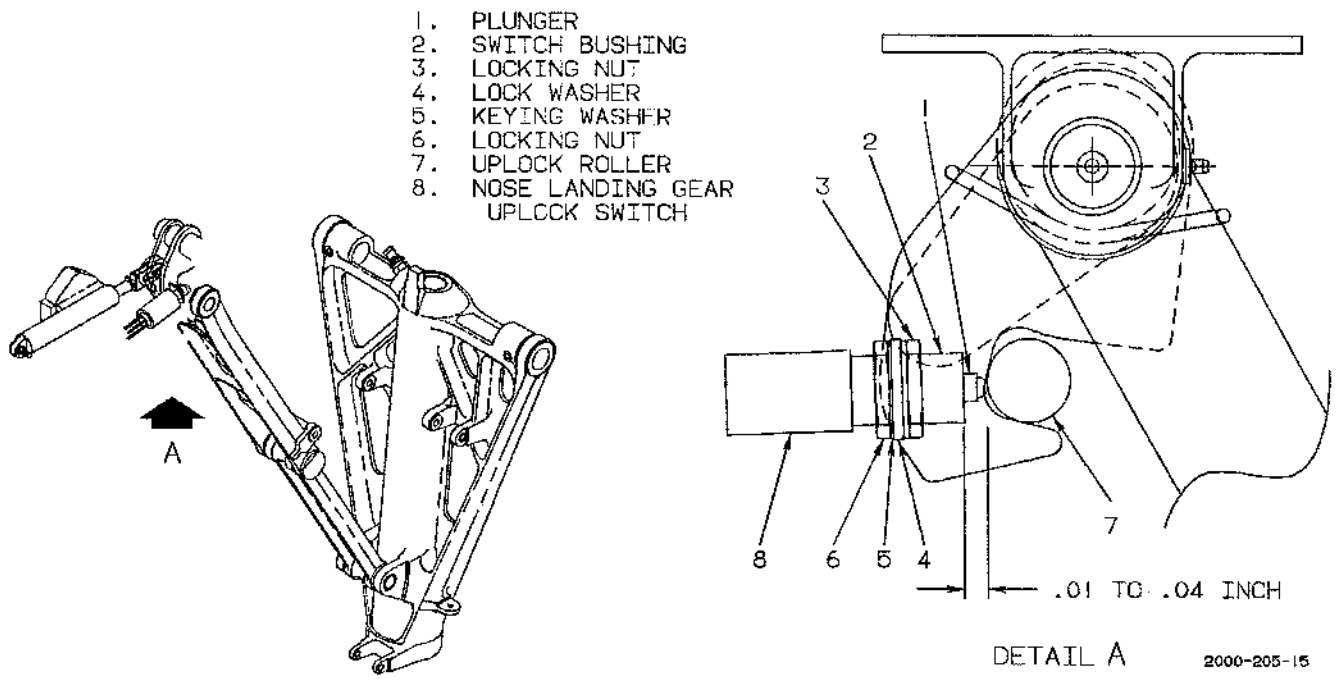
CAUTION

To prevent serious damage to the pump never operate the power pack without supplying 18 to 20 psig of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly. Connect the air source as instructed in Chapter 32-30-00.

CAUTION

When cycling the landing gear with the power pack, allow a one minute cooling period between cycles and a five minute cooling period after five cycles.

k. Disconnect the nose gear test box from the switch and reconnect the switch to the airplane electrical connector.



Nose Landing Gear Uplock Switch (Effectivity: All)
Figure 5

l. Connect the nose gear forward doors to the actuating cams and remove the airplane from the jacks.

MAIN LANDING GEAR UPLOCK SWITCH ADJUSTMENT (Effectivity: All) (Figure 6)

NOTE

Always use a continuity checker to ascertain switch actuation. DO NOT depend upon the click of the switch as an indication of actuation.

Before adjusting the uplock switch, the landing gear must be rigged correctly as instructed in Chapter 32-30-00.

CAUTION

Use of a Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-30-00), or it's equivalent, is recommended for retracting or extending the landing gear while adjusting the uplock switch. The procedure for using the pump can be found under LANDING GEAR RIGGING in Chapter 32-30-00.

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Disconnect the main gear door rods from the landing gear (refer to Chapter 32-10-00).

c. Ensure that the landing gear is fully retracted before adjusting the switch.

d. Disconnect the switch connector from airplane wiring and connect the switch connector to plug P2 on the main gear test box (12, Chart 1, 32-30-00), shown in Figure 4.

e. Remove safety wire and loosen the locking nuts (3 and 6) that hold the switch in position. Adjust the switch so that the green lights (DS7/DS8) on the test box illuminate. If using an ohmmeter (11, Chart 1, 32-00-00), it should indicate continuity when the wire leads are connected to pins E and F.

f. Adjust the switch so that the switch bushing (2) is 0.01 to 0.04 inch from the uplock roller (7) and secure the switch using the locking nuts (3 and 6) and safety wire.

NOTE

Excessive compression or bottoming out of the switch plunger may cause damage to the switch.

g. Using the hand pump (4, Chart 1, 32-00-00), cycle the gear to unlock and back to lock while monitoring the lights on the test box (12, Chart 1, 32-00-00).

h. As the gear is cycled to unlock, the red lights (DS13/DS14) on the test box should illuminate. If using an ohmmeter it should indicate an open circuit.

i. Disconnect the switch connector from the test box and connect the switch connector to the airpane wiring.

j. Check for proper operation of the in-transit lights located in the landing gear control switch handle. Operate the gear through at least one complete cycle with the power pack.

CAUTION

To prevent serious damage to the pump, never operate the power pack without supplying 18 to 20 psig of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly. Connect the air source as instructed in Chapter 32-30-00.

CAUTION

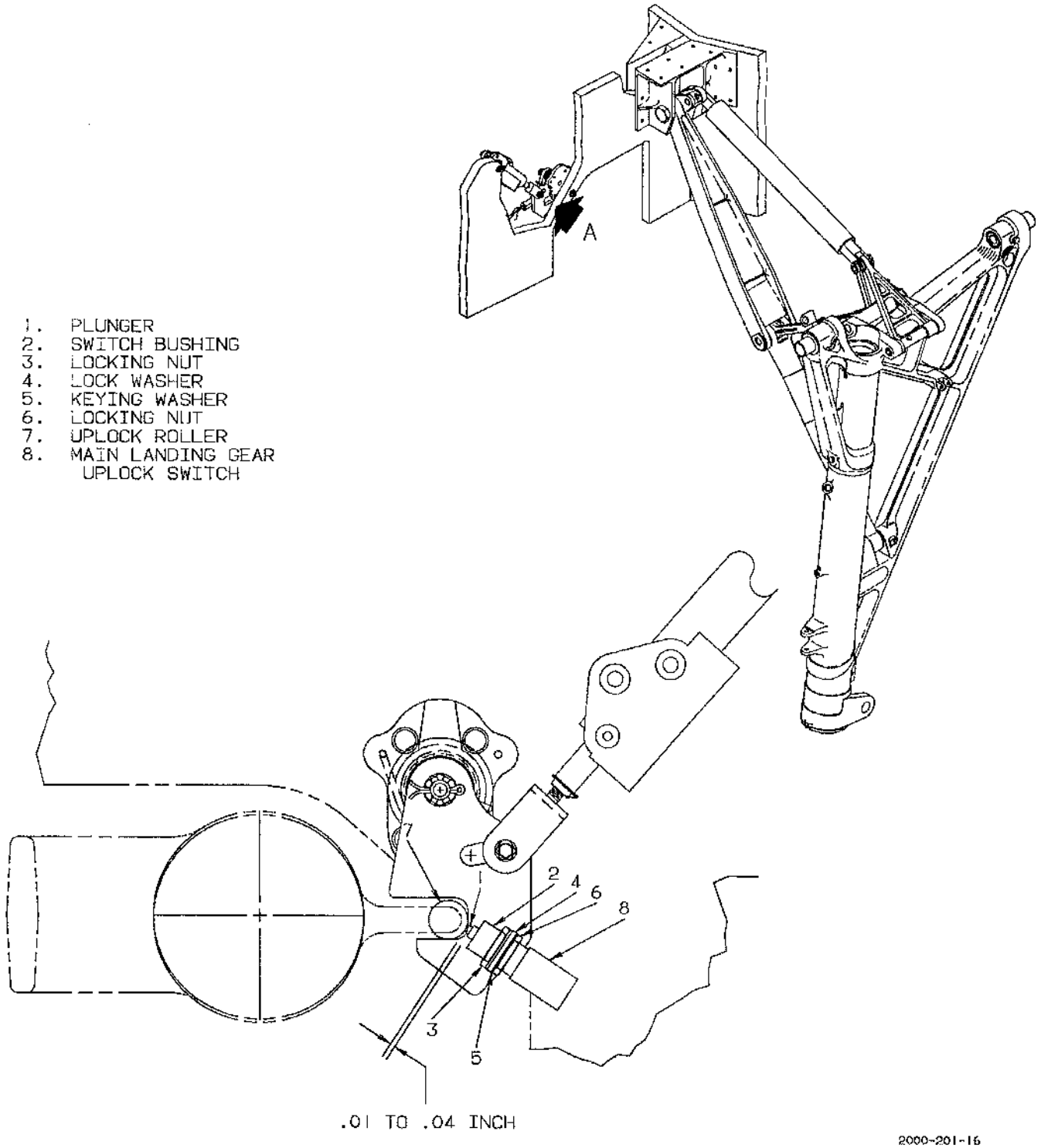
When cycling the landing gear with the power pack, allow a one minute cooling period between cycles and a five minute cooling period after five cycles.

k. Connect the main gear door rods to the landing gear (refer to Chapter 32-10-00) and remove the airplane from the jacks.

NOSE LANDING GEAR DOWNLOCK SWITCH ADJUSTMENT (Effectivity: All) (Figure 7)

NOTE

Always use a continuity checker to ascertain switch actuation. DO NOT depend upon the click of the switch as an indication of actuation.



Main Landing Gear Uplock Switch (Effectivity: All)
Figure 6

Before attempting any adjustment of the switch, the landing gear must be rigged correctly. Refer to Chapter 32-30-00 to rig the landing gear. Adjust the downlock switch on the nose gear drag brace as follows:

CAUTION

Use of a Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00) or it's equivalent, is recommended for retracting or extending the landing gear while adjusting the downlock switch. The procedure for using the pump can be found under LANDING GEAR RIGGING in Chapter 32-30-00.

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Disable the nose gear forward doors by disconnecting from the actuating cams.

c. Disconnect the nose gear downlock switch connector from airplane wiring and connect the switch connector to plug P2 on the nose gear test box (10, Chart 1, 32-00-00), shown in Figure 3.

d. The green lights (DS1/DS2) on the test box should illuminate as the actuator piston is extended to the down and locked position.

e. Remove safety wire and loosen the locking nuts (3 and 6) that hold the switch in position. Adjust the switch so that the green lights (DS1 and DS2) on the test box illuminate. If using an ohmmeter (11, Chart 1, 32-00-00), it should indicate continuity when the wire leads are connected to pins E and F.

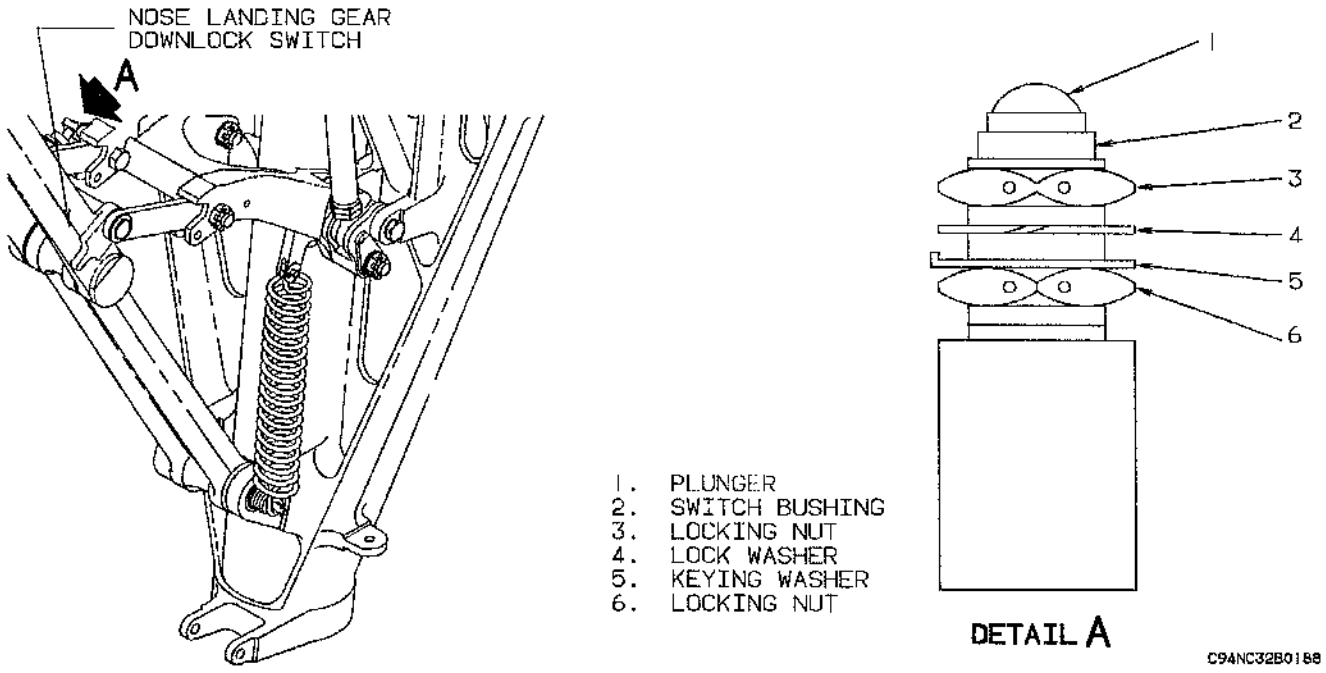
f. Turn the locking nuts (3 and 6) an additional one (1) turn for switch overtravel.

g. Secure the switch with the locking nuts (3 and 6) and safety wire.

NOTE

Excessive compression or bottoming out of the switch plunger may cause damage to the switch.

h. Using the hand pump (4, Chart 1, 32-00-00), cycle the gear to unlock and back to lock while monitoring the lights on the test box.



Nose Landing Gear Downlock Switch Adjustment (Effectivity: All)
Figure 7

i. As the gear is cycled to unlock, the red lights (DS9 and DS10) on the test box should illuminate. If using an ohmmeter it should indicate an open circuit.

j. Disconnect the switch connector from the test box and connect the switch connector to the airplane wiring.

k. Cycle the gear and check for proper operation of the GEAR DOWN light, located on the pilot's inboard subpanel, and the landing gear warning horn. Operate the gear through at least one complete cycle with the power pack.

CAUTION

To prevent serious damage to the pump, never operate the power pack without supplying 18 to 20 psig of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly. Connect the air source as instructed in Chapter 32-30-00.

CAUTION

When cycling the landing gear with the power pack, allow a one minute cooling period between cycles and a five minute cooling period after five cycles.

l. Connect the nose gear forward doors to the actuating cams and remove the airplane from the jacks.

MAIN LANDING GEAR DOWNLOCK SWITCH ADJUSTMENT (Effectivity: All) (Figure 8)

NOTE

Always use a continuity checker to ascertain switch actuation. DO NOT depend upon the click of the switch as an indication of actuation.

Before adjusting the downlock switch, the landing gear must be rigged correctly as instructed in Chapter 32-30-00.

CAUTION

Use of a Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00), or it's equivalent, is recommended for retracting or extending the landing gear while adjusting the downlock switch. The procedure for using the pump can be found under LANDING GEAR RIGGING in Chapter 32-30-00.

a. Place the airplane on jacks as outlined in Chapter 7-10-00.

b. Disable the main gear door by disconnecting the main gear door rods from the landing gear (refer to Chapter 32-10-00).

c. Disconnect the switch connector from airplane wiring and connect the switch connector to plug P3 on the main gear test box (12, Chart 1, 32-00-00), shown in Figure 4.

d. Ensure that the landing gear is down and locked.

e. Remove safety wire, loosen the locking nuts (3 and 6) on the switch, and back the switch out to the point that the plunger (1) is contacted, but not compressed.

f. Adjust the switch until the green lights (DS5 and DS6) on the test box illuminate. If using an ohmmmeter (11, Chart 1, 32-00-00), it should indicate continuity when the wire leads are connected to pins E and F.

g. Turn the locking nuts (3 and 6) one (1) full turn for switch overtravel.

h. Tighten the locking nuts (3 and 6) to hold the switch in position and safety wire.

NOTE

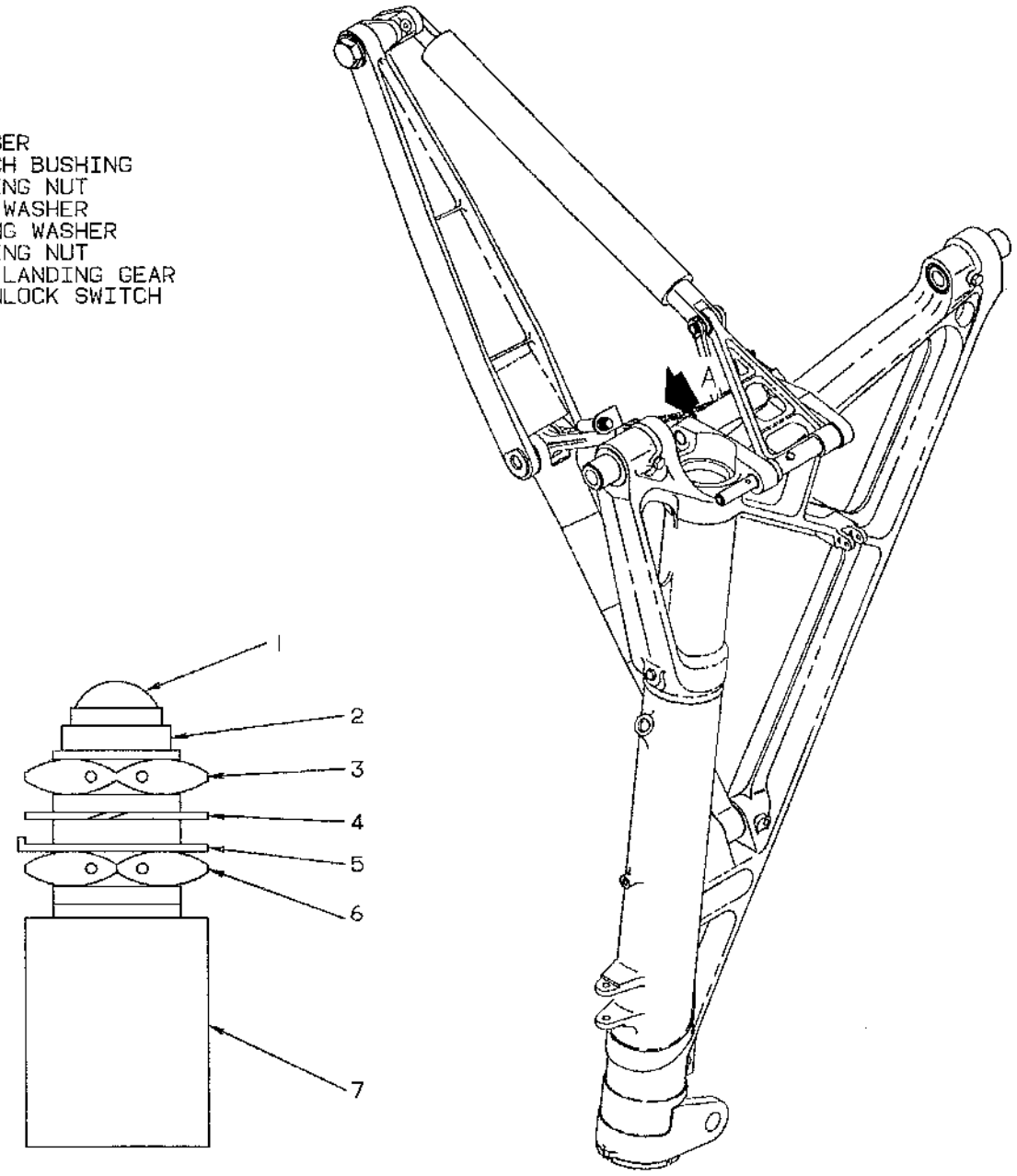
Excessive compression or bottoming out of the switch plunger may cause damage to the switch.

i. Using the hand pump cycle the gear to unlock and back to lock while monitoring the lights on the main gear test box.

j. As the gear is cycled to unlock, the red lights (DS15 and DS16) on the test box should illuminate. If using an ohmmeter it should indicate an open circuit.

k. Disconnect the switch connector from test box and connect the switch connector to the airplane wiring.

1. PLUNGER
2. SWITCH BUSHING
3. LOCKING NUT
4. LOCK WASHER
5. KEYING WASHER
6. LOCKING NUT
7. MAIN LANDING GEAR DOWNLOCK SWITCH



Main Landing Gear Downlock Switch (Effectivity: All)
Figure 8

l. Operate the gear through at least one complete cycle with the power pack and check for proper operation of the GEAR DOWN light and the landing gear warning horn.

CAUTION

To prevent serious damage to the pump, never operate the power pack without supplying 18 to 20 psig of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly. Connect the air source as instructed in Chapter 32-30-00.

CAUTION

When cycling the landing gear with the power pack, allow a one minute cooling period between cycles and a five minute cooling period after five cycles.

m. Connect the main gear door rods to the landing gear (refer to Chapter 32-10-00) and remove the airplane from the jacks.

MAIN LANDING GEAR SQUAT SWITCH ADJUSTMENT (Effectivity: All) (Figure 9)

NOTE

Always use a continuity checker to ascertain switch actuation. DO NOT depend upon the click of the switch as an indication of actuation.

This procedure is applicable to the RH and LH squat (safety) switches.

a. Place the airplane on jacks as instructed in Chapter 7-10-00.

b. Deflate the shock strut as instructed in Chapter 12-10-00.

c. Disconnect the switch connector from the airplane wiring and connect the switch connector to plug P1 on the main gear test box (12, Chart 1, 32-00-00), shown in Figure 4.

WARNING

Make all adjustments with the power off at the master switch and no external power connected.

d. Remove the squat switch actuator rod (1) from the attaching bracket on the upper torque knee, then remove the retaining nut (2) and the switch arm (3) from the switch shaft.

NOTE

Use a gear puller to remove the switch arm (3) from the switch shaft to prevent damage to the internal mechanism of the switch.

e. Jack the landing gear so the shock strut is compressed to .38 to .62 inch from the fully extended position.

f. If not using the test box connect the wire leads from an ohmmeter (11, Chart 1, 32-00-00), to pins G and H (both RH and LH installations) of the receptacle plug located in the upper rear of each main landing gear wheel well.

g. Rotate the switch shaft (right hand switch clockwise, left hand switch counterclockwise) until the ohmmeter indicates continuity or the red test lights (DS9, DS10, DS11 and DS12) on the test box illuminate.

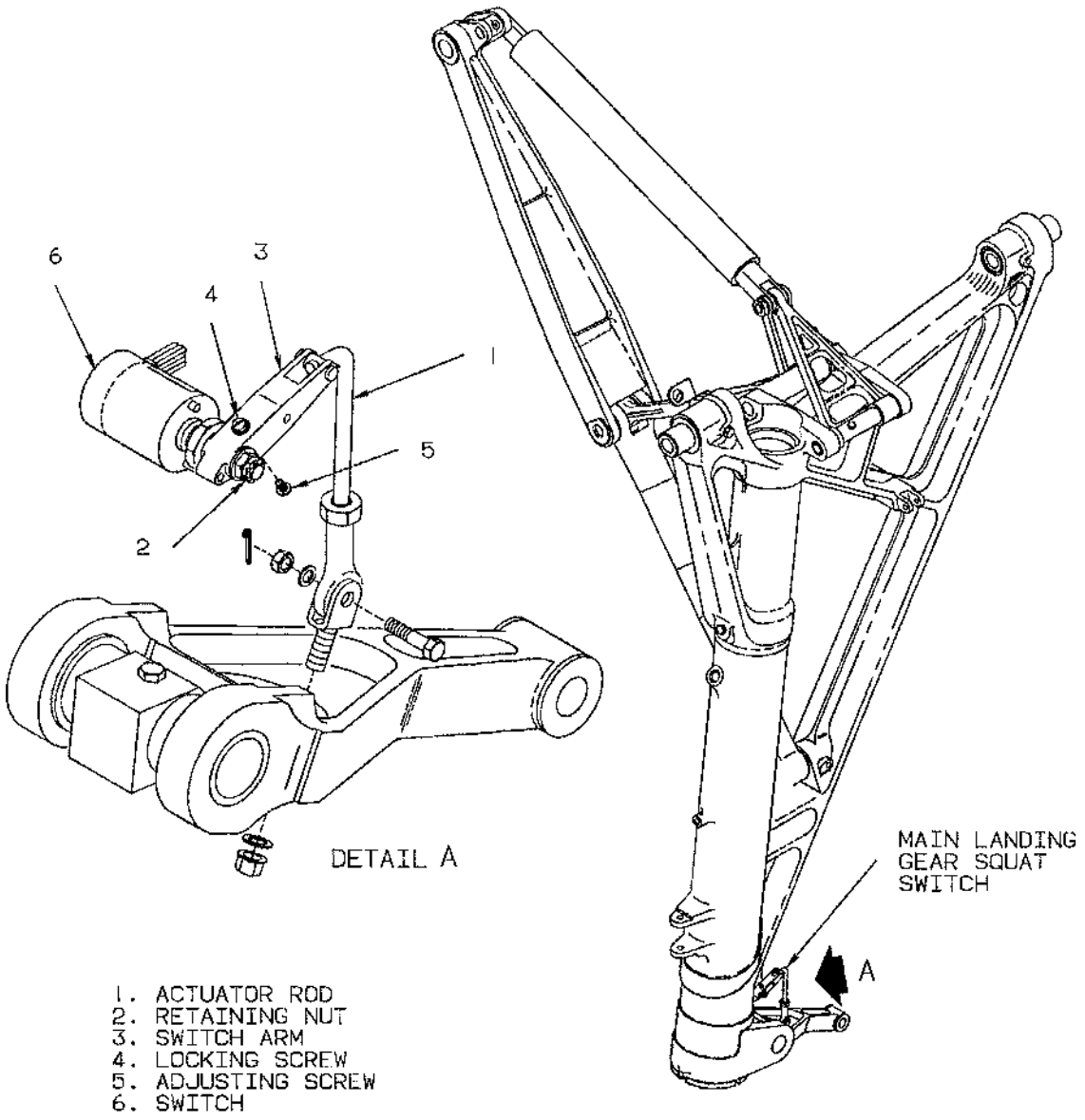
h. Remove the safety wire from the locking screw (4) on the switch arm (3) and back off the locking screw (4).

i. Install the switch arm (3) on the switch shaft in a position parallel to the upper torque knee and adjust the actuating rod (1) to align with the attaching bracket on the torque knee.

j. Compress the shock strut to 2 inches from the fully extended position, and adjust the switch shaft (right hand switch counterclockwise, left hand switch clockwise) at the adjusting screw (5) until the ohmmeter indicates an open circuit or the green test lights (DS1, DS2, DS3 and DS4) on the test box illuminate. When satisfactory adjustment is reached, tighten the locking screw (4) and retaining nut (2).

k. Before safety wiring the locking screw (4) to the switch arm (3), recheck the safety switch rigging as follows:

1. As the shock is compressed, the ohmmeter or green test lights (DS1, DS2, DS3 and DS4) on the test



Main Landing Gear Squat Switch (Effectivity: All)
Figure 9

box should indicate an open circuit when the shock strut reaches a position .75 to 1.50 inch from the fully extended position. The ohmmeter or green test lights (DS1, DS2, DS3 and DS4) on the test box should continue to indicate an open circuit as the shock strut is compressed to the fully compressed position.

2. As the shock strut is extended from the fully compressed position, the ohmmeter or the green test lights (DS1, DS2, DS3 and DS4) on the test box should continue to indicate an open circuit until the shock strut reaches a position .38 to .62 inch from the fully extended position. At this point, the ohmmeter or the red test lights (DS9, DS10, DS11 and DS12) on the test box should indicate continuity up to and including the fully extended position.

NOTE

The point at which the switch is actuated during compression and extension of the shock strut differs because of tolerances in the switch and its attendant linkage on the landing gear.

l. Safety wire the locking screw (4) to the switch arm (3).

m. Disconnect the switch connector from the test box and connect the switch connector to the airplane wiring.

n. Inflate the shock strut as instructed in Chapter 12-10-00.

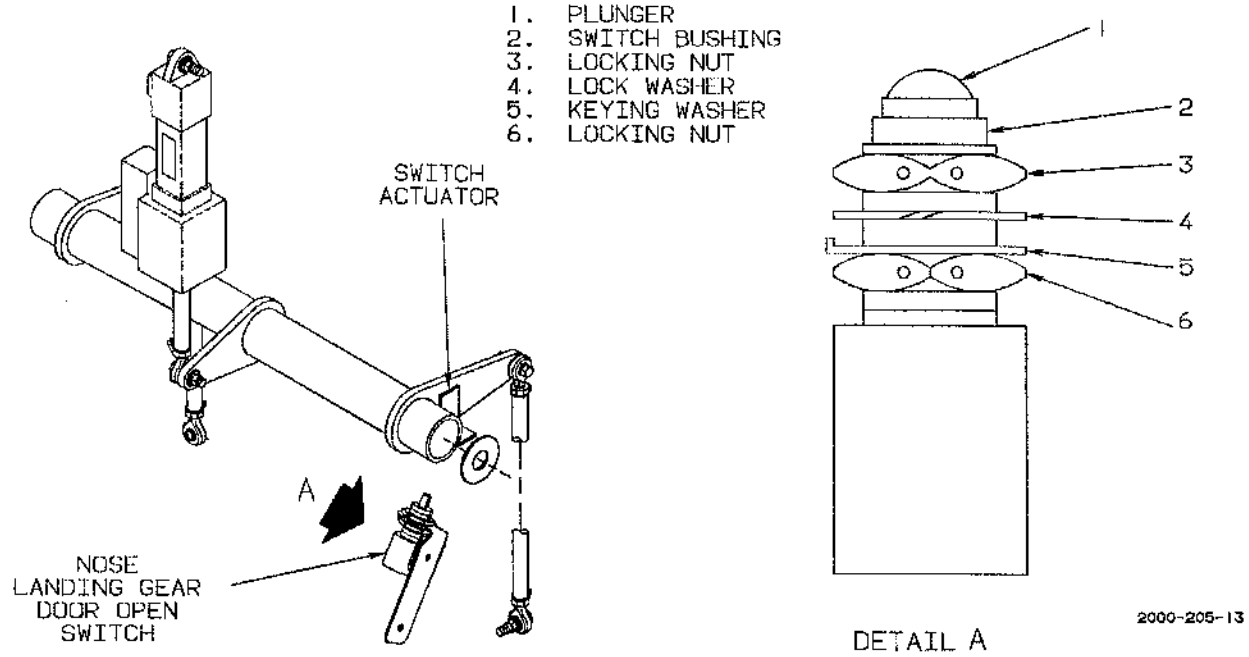
o. Remove the airplane from the jacks.

NOSE LANDING GEAR DOOR OPEN SWITCH ADJUSTMENT (Effectivity: All) (Figure 10)

The nose landing gear door open switch is located in the top of the nose gear wheelwell, and is closed by a cam arm on the forward door bell crank when the nose door is open.

NOTE

Always use a continuity checker to ascertain switch actuation. DO NOT depend upon the click of the switch as an indication of actuation.



Nose Landing Gear Door Open Switch (Effectivity: All)
Figure 10

a. Before adjusting the door open switch, the landing gear must be rigged correctly as instructed in Chapter 32-30-00.

CAUTION

Use of a Model 06-4005-0501 hydraulic hand pump (4, Chart 1, 32-00-00), or it's equivalent, is recommended for retracting or extending the nose gear forward door actuator while adjusting the door open switch. The procedure for using the pump can be found under LANDING GEAR RIGGING in Chapter 32-30-00.

b. Place the airplane on jacks as outlined in Chapter 7-10-00.

c. Remove all electrical power from the airplane.

d. Disconnect the nose gear forward doors by removing bolts and washers securing pushrods to bell crank assembly.

e. Using the hand pump cycle the nose gear forward door bell crank to the open position as instructed in Chapter 32-30-00.

f. Disconnect the switch connector from airplane wiring and connect the switch connector to plug P3 on the test box (10, Chart 1, 32-00-00), shown in Figure 3.

g. Remove safety wire and loosen the locking nuts (3 and 6) securing either side of the nose gear door open switch.

h. Adjust the switch until the red lights (DS13 and DS14) on the test box illuminate. If using an ohmmeter (11, Chart 1, 32-00-000), it should indicate continuity when the wire leads are connected to pins E and F.

i. Adjust the switch until it actuates and then adjust in one complete turn plus additional adjustment as required to obtain the next keying position.

NOTE

Excessive compression or bottoming out of the switch plunger (1) may cause damage to the switch.

j. Tighten the locking nuts (3 and 6) and safety wire.

k. Using the hand pump cycle the nose gear forward door bell crank to the open position and back to the closed position while monitoring the lights on the test box.

l. As the nose gear forward door is cycled to the open position, the red lights (DS13 and DS14) on the test box should illuminate. If using an ohmmeter it should indicate a closed circuit between pins E and F.

m. Disconnect the switch connector from the test box and connect the switch connector to the airplane wiring.

n. Reconnect pushrods to bellcrank assembly using attaching bolts and washers.

o. Restore electrical power to the airplane.

p. Check for proper operation of the in-transit lights located in the landing gear control handle through at least one complete cycle.

CAUTION

To prevent serious damage to the pump, never operate the power pack without supplying 18 to 20 psig of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly. Connect the air source as instructed in Chapter 32-30-00.

CAUTION

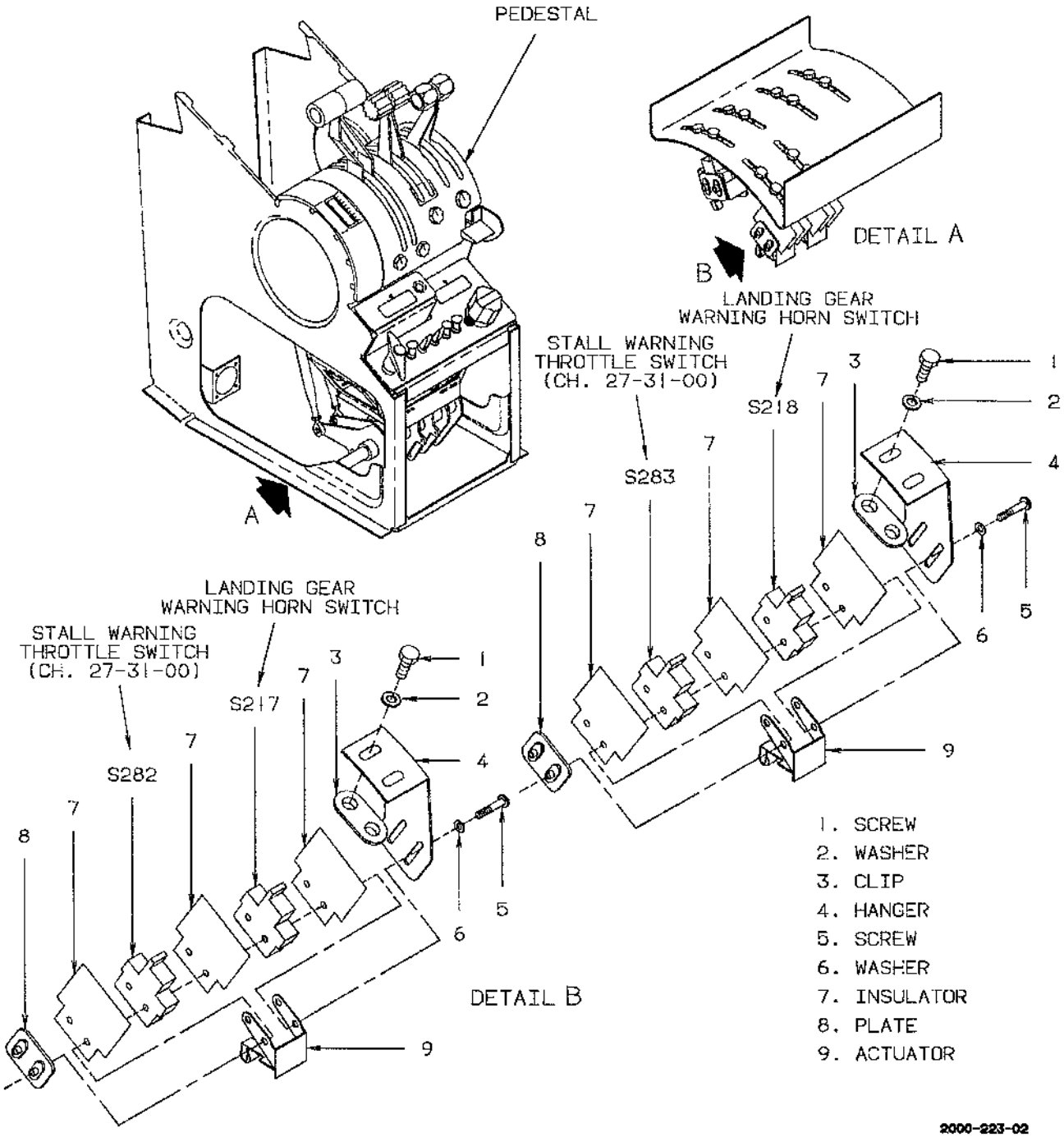
When cycling the landing gear with the power pack, allow a one minute cooling period between cycles and a five minute cooling period after five cycles.

q. Remove the airplane from the jacks.

LANDING GEAR WARNING HORN SWITCH ADJUSTMENT (Effectivity: All) (Figure 11)

A cam-operated switch is provided for each power lever linkage in the pedestal. The switches are wired so that retarding either or both levers beyond the safe flight setting with the gear retracted will sound the warning horn intermittently. The airplane must be in flight when determining the position of the power levers, which correspond to 88% to 90% N_1 (gas generator speed), since the ram air effect will alter the 88% to 90% N_1 position indicated during ground operation.

a. With the airplane in flight, advance the power levers until 88% to 90% N_1 is attained on each engine.



Landing Gear Warning Horn Switches (Effectivity: All)
Figure 11

b. Mark the position of the power levers on the pedestal with a piece of tape. Land the airplane.

c. Remove the panel from the LH side of the pedestal (refer to Chapter 25-10-00).

d. Place the power levers in alignment with the tape, positioned during flight test at 88% to 90% N_1, on the pedestal.

NOTE

The landing gear warning horn switches and the stall warning throttle switches are actuated by the same actuator for both power levers. The actuation point of all four switches is 88% to 90% N_1. Refer to Chapter 27-31-00 for further information on the stall warning throttle switches.

e. Adjust the landing gear warning horn switches by loosening the screws (1) securing them in their slotted mounting hangers (4) so that continuity is just shown between pins 30 and 7 and pins 30 and 20 of J239 (warning horn switch connector) in the airplane wiring. Verify continuity with an ohmmeter (11, Chart 1, 32-00-00).

f. Secure the landing gear warning horn switches in their slotted mounting hangers (4) by tightening the screws (1).

g. Move the power levers aft of the tape and listen for an audible click. Using an ohmmeter verify continuity between pins 30 and 7 and pins 30 and 20 of J239 in the airplane wiring.

h. Move the power levers forward of the tape. Using an ohmmeter verify an open circuit between pins 30 and 7 and pins 30 and 20 of J239 in the airplane wiring.

i. Remove the tape and install the panel on the LH side of the pedestal (refer to Chapter 25-10-00).

j. Flight test the airplane with the landing gear retracted and the power levers retarded until the warning horn sounds at an N_1 speed of 88% to 90%.

LANDING GEAR EMERGENCY SELECT SWITCH ADJUSTMENT (Effectivity: All) (Figure 12)

The landing gear emergency select switch is located under the landing gear alternate extension hand pump handle in the cockpit.

NOTE

Always use a continuity checker to ascertain switch actuation. DO NOT depend upon the click of the switch as an indication of actuation.

a. Place the airplane on jacks as instructed in Chapter 7-10-00.

b. Remove electrical power from the airplane.

c. Raise the landing gear alternate extension hand pump handle out of retention clips.

d. Remove floorboard (141BT) adjacent to pump handle (refer to Chapter 53-20-00).

e. Remove bolts (10) and washers (8) securing the lanyard (9), seat (3) and clips (2) to the floorboard.

f. Remove safety wire and loosen the locking nuts (6) that hold the switch on either side of the base (4).

g. Adjust the switch so that a minimum of .06 inch to a maximum of .12 inch overtravel is used when the hand pump handle is in the stowed position.

NOTE

Excessive compression or bottoming out of the switch plunger may cause damage to the switch.

h. Tighten the locking nuts (6) and safety wire.

i. Replace the seat (3), clips (2) and lanyard (9) and secure with the attaching bolts (10) and washers (8).

j. Stow the landing gear alternate extension hand pump handle.

k. Restore electrical power to the airplane.

CAUTION

To prevent serious damage to the pump, never operate the power pack without supplying 18 to 20 psig of regulated dry air to pressurize the power pack reservoir during ground operation of the power pack assembly. Connect the air source as instructed in Chapter 32-30-00.

CAUTION

When cycling the landing gear with the power pack, allow a one minute cooling period between cycles and a five minute cooling period after five cycles.

l. Move the landing gear control handle to UP to raise the landing gear.

NOTE

The landing gear will move from the raised position to the down position in approximately 7 seconds.

m. Move the landing gear control handle to the DN position to start the gear moving down, and within less than 7 seconds raise the landing gear alternate extension handle out of the stowed position. Verify that the the landing gear power pack motor stops running.

n. Stow the landing gear alternate extension handle. Verify that the landing gear does not change position.

o. Move the landing gear control handle to the UP position. Verify that the landing gear raises to the full up position.

p. Move the landing gear control handle to DN to lower the landing gear.

q. Remove the airplane from the jacks.

LANDING GEAR AIRSPEED SWITCH REMOVAL (Effectivity: All) (Figure 13)

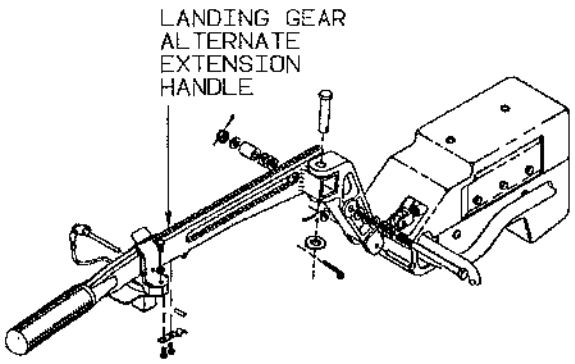
The airspeed switch is located in the pitot/static plumbing in the RH forward nose of the airplane.

a. Pull LANDING GEAR WARNING circuit breaker.

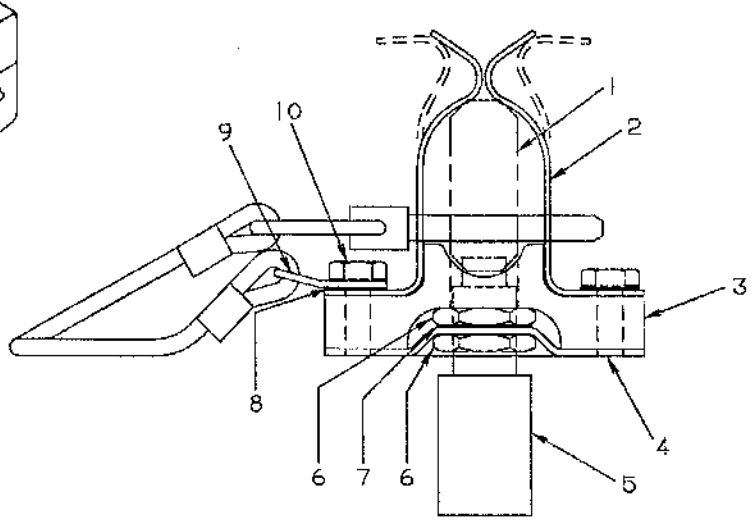
b. Remove the RH nose avionics compartment door (214AR) (refer to Chapter 53-30-00).

c. Disconnect electrical connector from airspeed switch (2).

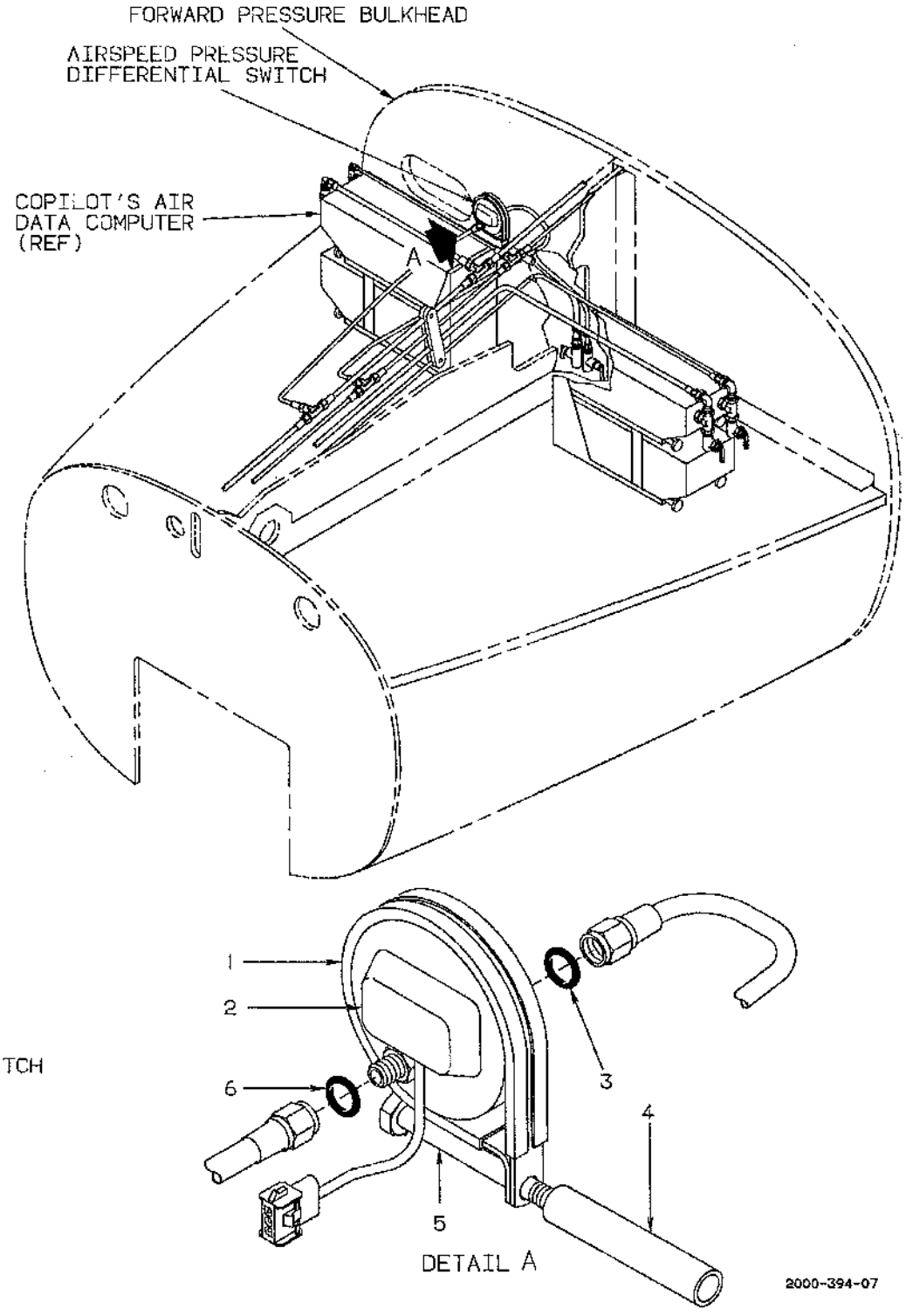
d. Disconnect plumbing from both sides of the airspeed switch (2). Cap the plumbing lines. Discard old O-rings (3 and 6).



1. LANDING GEAR ALTERNATE EXTENSION HANDLE
2. CLIP
3. SEAT
4. BASE
5. LANDING GEAR EMERGENCY SELECT SWITCH
6. LOCKING NUTS
7. WASHER
8. WASHER
9. LANYARD
10. BOLT



Landing Gear Emergency Select Switch (Effectivity: All)
Figure 12



1. CLAMP
2. AIRSPEED SWITCH
3. O-RING
4. SPACER
5. BOLT
6. O-RING

Landing Gear Airspeed Switch (Effectivity: All)
Figure 13

e. Remove clamp (1), spacer (4), and bolt (5) securing airspeed switch (2). Remove airspeed switch (2) from airplane.

LANDING GEAR AIRSPEED SWITCH INSTALLATION (Effectivity: All) (Figure 13)

a. Position airspeed switch (2) in airplane with high pressure port pointing forward.

b. Secure airspeed switch (2) with clamp (1), spacer (4), and bolt (5).

c. Install new O-rings (3 and 6) and connect plumbing to both sides of the airspeed switch (2).

d. Connect electrical connector to airspeed switch.

e. Perform a Pitot/Static Leakage Test as instructed in Chapter 34-10-00.

f. Engage LANDING GEAR WARNING circuit breaker.

g. Install the RH nose avionics compartment door (214AR) (refer to Chapter 53-30-00).

LANDING GEAR AIRSPEED SWITCH FUNCTIONAL TEST (Effectivity: All)

a. The following equipment is required to perform the airspeed switch functional test:

1. Ohmmeter (11, Chart 1, 32-00-00)

2. Manometer, calibrated in inches of water (13, Chart 1, 32-00-00)

3. Regulated air or dry nitrogen

b. Remove airspeed switch from airplane as instructed in LANDING GEAR AIRSPEED SWITCH REMOVAL in this chapter.

c. Check for continuity between pin 1 (red) and pin 3 (blue).

d. Connect both pressure ports to air supply. Increase pressure to 15 psig and hold for 1 minute. There shall be no loss of pressure.

e. Vent the low pressure port to atmosphere. Slowly increase pressure to the high pressure port. The switch must actuate with continuity between pin 1 (red) and pin 2 (white) by 15.2 inches of water.

f. Slowly decrease pressure and note that switch opens with continuity between pin 1 (red) and pin 3 (blue) between 13.7 and 14.7 inches of water.

g. Remove switch from test equipment and install in airplane as instructed in LANDING GEAR AIRSPEED SWITCH INSTALLATION in this chapter.

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CHAPTER 33 - LIGHTS

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

This chapter is concerned with the description, operation and maintenance practices for the airplane's lighting system. Information is provided for the following areas:

- Cockpit
- Cabin
- Baggage Compartment
- Exterior

See the appropriate subchapter for detailed information on the lighting equipment in that area.

RECOMMENDED MATERIALS AND SPECIAL TOOLS (Effectivity: All)

The recommended materials listed in Chart 1 and special tools in Chart 2 as meeting federal, military or supplier specifications are provided for reference only and are not specifically recommended by Beech Aircraft Corporation. The products included in this chart have been tested and approved for aviation usage by Beech Aircraft Corporation, by the supplier or by compliance with the applicable specification. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included herein. Only the basic number of each product specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the supplier of the product to be used.

CHART 1
RECOMMENDED MATERIALS (Effectivity: All)

MATERIALS	SPECIFICATION	PRODUCT	SUPPLIER
1. Cleaner	TT-I-735	Isopropyl Alcohol	Locally available
2. Sealant	MIL-S-8802, Type II	Pro Seal 890	Pro-Seal Corp., Div. of Essex Chemical Corp., Aerospace Products Compton, CA 90220
3. Sealant		Presstite # 576	Presstite-Keystone Engineering Co., 3900 Chateau Ave., St. Louis, MO
4. Cleaner	TT-N-95	Naphtha	Locally available
5. Adhesive		PS 30	Cadillac Plastics and Chemical Co., Detroit, MI

CHART 2
SPECIAL TOOLS (Effectivity: All)

TOOL NAME	PART NUMBER	SUPPLIER	USE
1. T-light Comparator	45-B7	Self-Powered Lighting Inc., 8 Westchester Plaza, Elmsford, New York 10523	For checking illuminated sign brightness.

COCKPIT LIGHTS - DESCRIPTION AND OPERATION (Effectivity: All) (Figures 1 thru 7)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

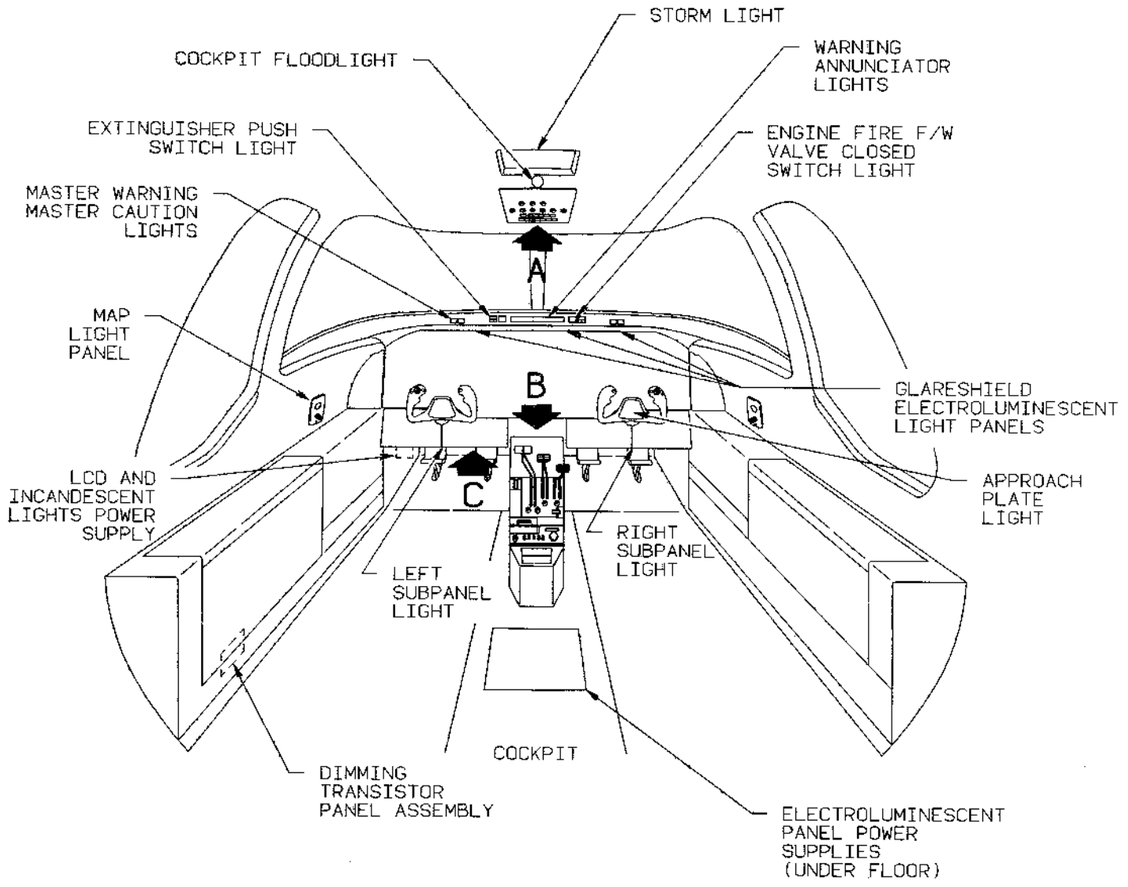
The cockpit lighting system provides lighting for both day and night operation of the airplane. Overall control of the cockpit lights is accomplished through eight potentiometers and the LIGHTS MASTER pushbutton switch located on the overhead panel in the cockpit. The only exceptions are the approach plate, map and storm lights, which are controlled individually. The LIGHTS MASTER switch shuts off all but two of the potentiometers through three master relays. The INSTR INDIRECT and COCKPIT AREA potentiometers are not controlled by the LIGHTS MASTER switch.

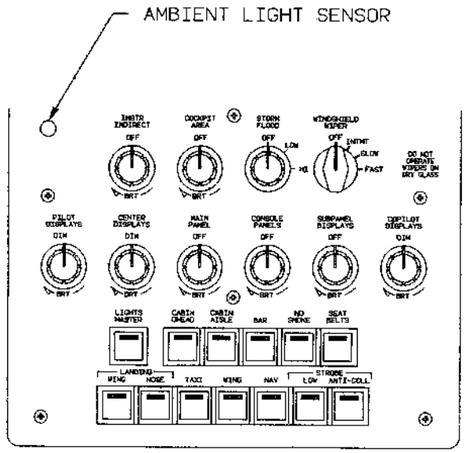
Indirect lighting of the instrument panel is accomplished with three electroluminescent light panels mounted under the glareshield. The 115 vac power supply (PS103) for the glareshield electroluminescent panel is mounted under the cockpit floor and is controlled by the INSTR INDIRECT potentiometer. This potentiometer rotates clockwise from the OFF position to increase brightness of the glareshield electroluminescent light panels. The approach plate lights are powered by the triple-fed bus through the flight instrument lights circuit breaker. When the button placarded PUSH on the approach plate is pressed, the cover will flip open for access to a switched potentiometer. This potentiometer controls the five lamps at the top of the approach plate cover.

The cockpit map lights are mounted on individual panels with a potentiometer. The two panels are mounted, one each, in the upholstery panel directly outboard of each pilot. They are powered by the left generator bus through the storm light circuit breaker, and increase in brightness as the potentiometer is rotated clockwise from OFF. The pilot and copilot storm lights are also powered through the storm light circuit breaker. When the rotary switch, placarded STORM FLOOD, OFF-LOW-HIGH, on the overhead panel is rotated to the LOW position, the copilot's storm light is illuminated. When the switch is rotated to the HIGH position, the pilot's storm light is also illuminated.

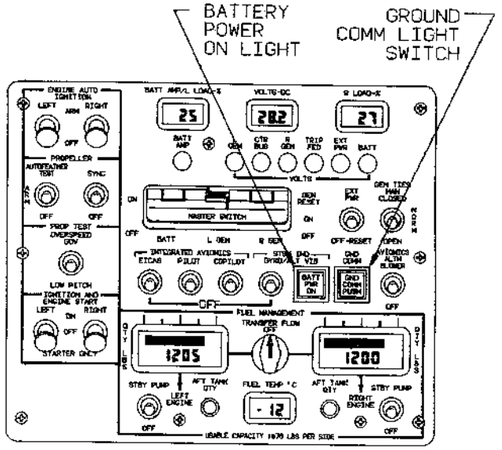
A subpanel light is mounted directly below each control column for illumination of the floor area near the rudder pedals. Both subpanel lights are controlled with the COCKPIT AREA potentiometer which is powered by the triple-fed bus through the overhead floodlight circuit breaker. This potentiometer is located on the overhead panel and rotates clockwise from the OFF position to increase brightness. The COCKPIT AREA potentiometer also controls the cockpit floodlight. The floodlight is mounted between two speakers in the overhead upholstery panel immediately aft of the overhead panel. The cockpit floodlight and both subpanel lights are illuminated full bright by the "hot" battery bus through the floodlight circuit breaker any time the cabin door is unlocked and the COCKPIT switch on the entry light switch panel is in the ON position. When the cabin door is locked, control of the cockpit flood and subpanel lights is switched back to the COCKPIT AREA potentiometer. Refer to Chapter 33-20-00 for complete information on the entry light switch panel.

Electroluminescent panels provide outline illumination of the subpanel controls, overhead panel controls and the three cockpit circuit breaker panels. Some of the individual instrument panel display units and control panels have electroluminescent panels for line select key and switch illumination. These electroluminescent panels are controlled by the MAIN PANEL potentiometer and powered by a 115 vac power supply (PS116) mounted under the cockpit floor. The circuit breaker and subpanel electroluminescent panels are controlled by the CONSOLE PANELS potentiometer and powered by two 115 vac power supplies (PS106 and PS107) mounted under the cockpit floor. When the LIGHTS MASTER switch is placed in the ON position, voltage from the triple-fed bus through the flight instrument light circuit breaker will close relay A110K2 located on the overhead panel. Triple-fed bus voltage

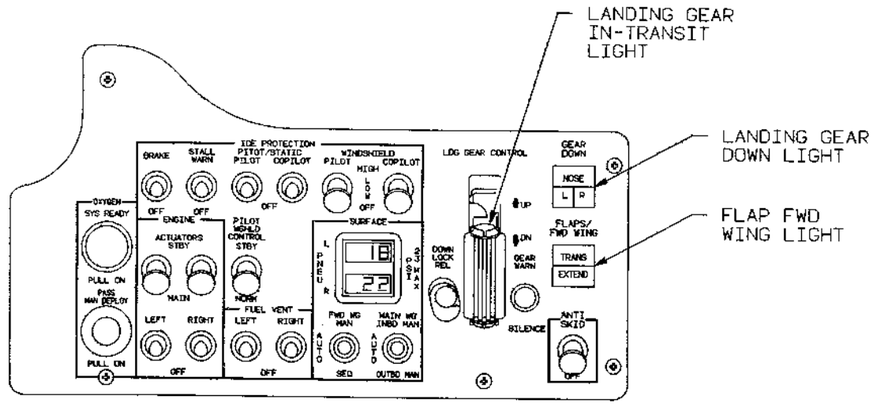




OVERHEAD PANEL
DETAIL A



CENTER SUBPANEL
DETAIL B



PILOT'S INBOARD SUBPANEL
DETAIL C

C93NC33B4443

Cockpit Lights (Effectivity: All)
Figure 1

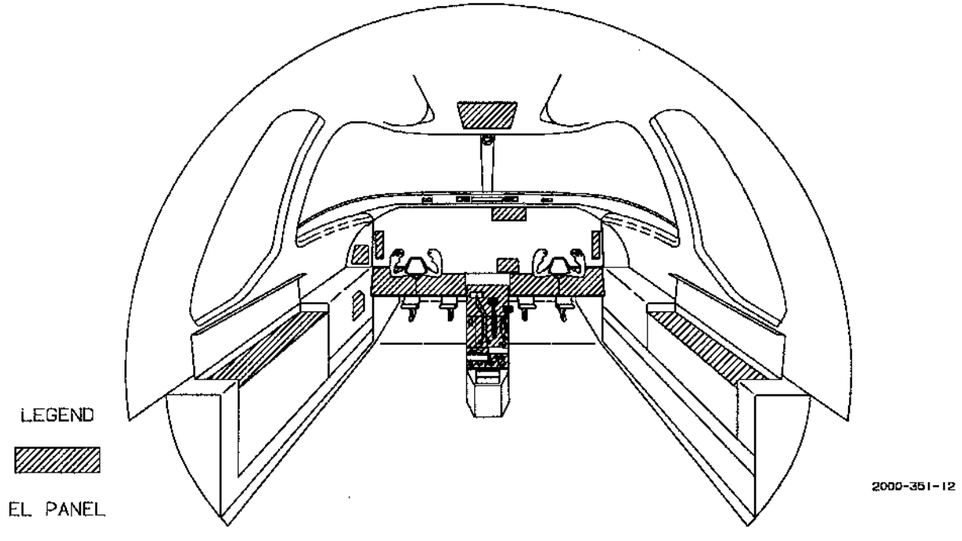
is then applied to the MAIN PANEL potentiometer and right generator bus voltage is applied to the CONSOLE PANELS potentiometer. Both potentiometers rotate clockwise from the OFF position to increase power supply (PS116) output to the electroluminescent panels.

The instrument panel CRT display units receive 28 vdc for light control. CRT dimming is accomplished through the lighting control PCB by three potentiometers placarded CENTER DISPLAYS, PILOT'S DISPLAYS and COPILOT'S DISPLAYS on the overhead panel. These potentiometers control the center, pilot's and copilot's dimming buses. When light control voltage on the dimming bus drops below a preset level, the display unit CRTs revert back to full bright. Refer to Chapter 31-10-00 for brightness check procedures if the brightness of a display unit CRT is questionable.

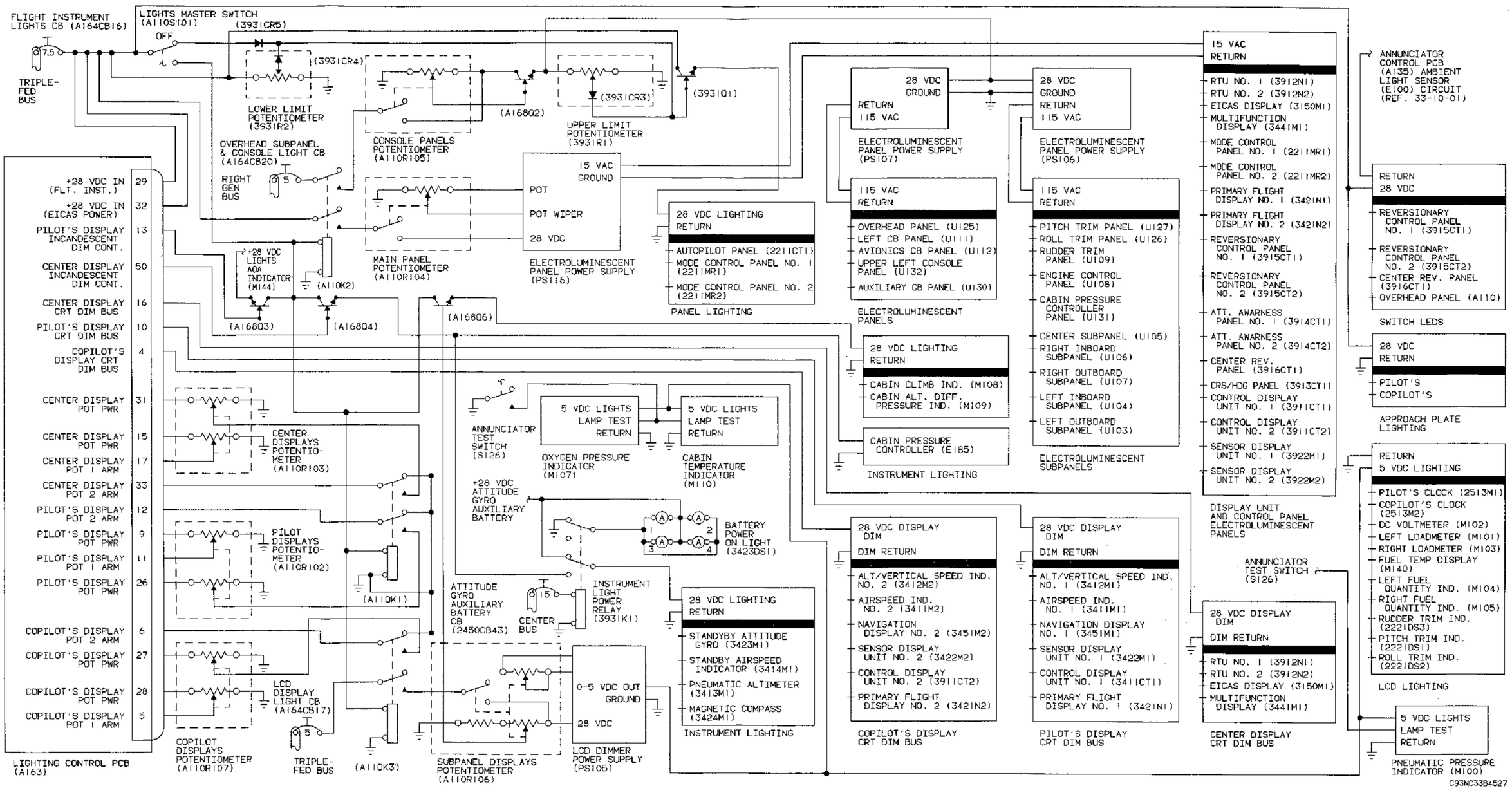
The autopilot panel, both mode select panels and the angle-of-attack indicator panel receive 28 vdc lighting voltage through a dimming transistor. The cabin-climb indicator, cabin-altitude differential-pressure indicator and cabin-pressure controller, along with the cockpit flood and subpanel lights, also receive 28 vdc through dimming transistors. Six dimming transistors are clustered with heat sinks on a panel assembly in the lower portion of the pilot's console.

The instrument panel LCD lights and incandescent displays are illuminated by a 5-vdc power supply mounted outboard of the instrument panel on the pilot's side. When the LIGHTS MASTER switch on the overhead panel is placed in the ON position, relay A110K3 located on the overhead panel is closed, connecting triple-fed bus voltage from the LCD display light circuit breaker to the SUBPANEL DISPLAYS potentiometer. The SUBPANEL DISPLAYS potentiometer increases brightness of the displays as it is rotated clockwise from OFF. The incandescent displays revert back to full bright when the SUBPANEL DISPLAYS potentiometer is turned OFF.

Each of the pushbutton switches on the overhead panel contains an LED that illuminates any time the switch is placed in the ON position. The three reversionary panels each have toggle switches with LEDs in the tip that illuminate any time the switch is placed in the ON position. The switch LEDs are powered by the triple-fed bus through the instrument light circuit



Electroluminescent Panel Locations (Effectivity: All)
Figure 2



Cockpit Lights Schematic (Effectivity: All)
Figure 3

breaker and grounded to the returns on the annunciator control PCB (A135) for automatic dimming control by the ambient light sensor circuit.

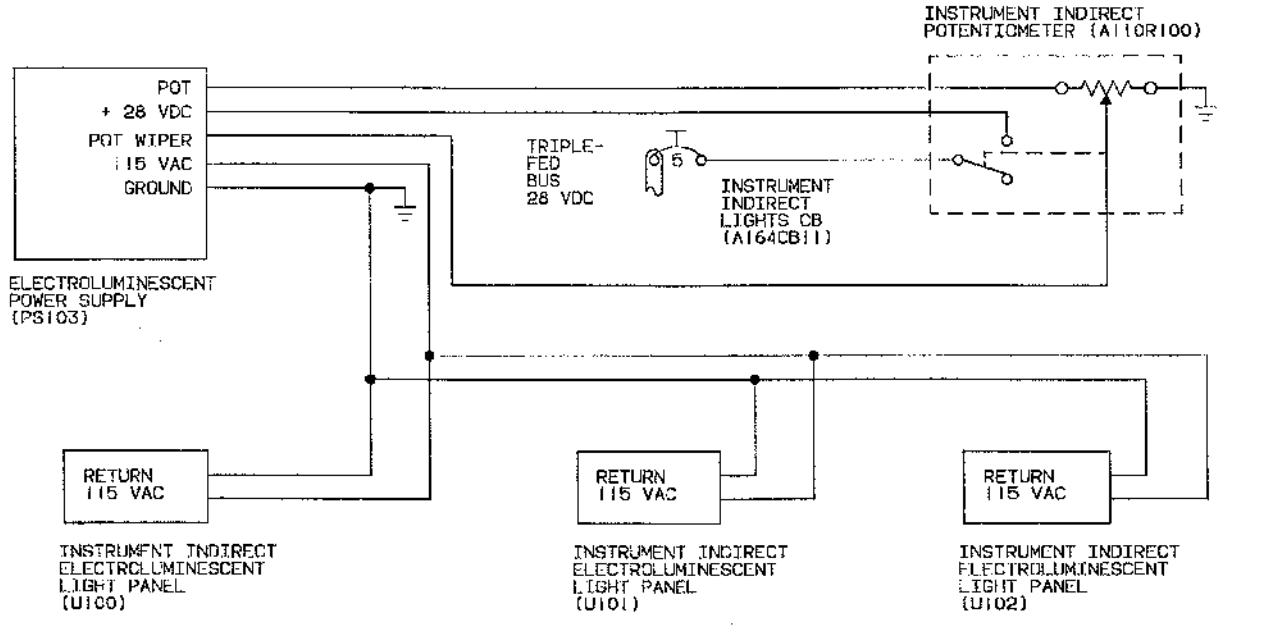
STANDBY INSTRUMENT AND MAGNETIC COMPASS LIGHTING (Effectivity: All) (Figures 1 and 3)

The three standby instruments and the magnetic compass receive lighting voltage from the triple-fed bus or the attitude gyro auxiliary battery. When the center bus is powered, the instrument light power relay is closed, illuminating the standby instrument and magnetic compass lights with triple-fed bus voltage. When power is removed from the center bus, the instrument light relay opens, illuminating the standby instrument and magnetic compass lights with voltage from the attitude gyro auxiliary battery. A ground is supplied through the relay to illuminate the BATT PWR ON light, which also receives voltage from the attitude gyro auxiliary battery. Refer to Chapter 24-32-00 for detailed information on auxiliary power.

WARNING ANNUNCIATOR AND INDICATOR LIGHTS (Effectivity: All) (Figure 1)

Fourteen red warning annunciators with black legends are mounted together in the warning annunciator panel in the glareshield. Each warning annunciator legend assembly contains two lamps replaceable from the front. Master warning and master caution lights, engine fire extinguisher controls and firewall valve shutoff controls are also mounted in the glareshield on either side of the warning annunciator panel. Each light is mounted with a switch assembly and has lamps replaceable from the front without removing the switch assembly. A test switch (placarded ANNUN) on the pilot's outboard subpanel allows for testing of the warning annunciator, master warning and master caution lights. Refer to Chapter 31-50-00 for detailed information on the warning annunciator system.

The battery-power-on light and ground communication switch light are both mounted on the center subpanel. The landing-gear-down and flap/forward wing lights are mounted on the left inboard subpanel. These



Instrument Indirect Lights Schematic (Effectivity: All)
Figure 4

lights also contain lamps replaceable from the front of the assembly. Two landing gear in-transit lamps are located in the landing gear handle knob.

AMBIENT-LIGHT SENSOR CIRCUIT (Effectivity: All) (Figure 7)

An ambient-light sensor is mounted in the overhead panel for sensing cockpit light intensity. Depending upon intensity, the ambient-light sensor will signal the annunciator- control PCB to adjust all lights grounded at returns "A" or "B" of the PCB. Triple-fed bus voltage is provided through the number-1- annunciator circuit breaker to power the ambient-light sensor circuit. All warning annunciators, master warning and master caution lights, and some of the lighted switches on the instrument panel are dimmed by the ambient-light sensor circuit.

COCKPIT LIGHTS - TROUBLESHOOTING (Effectivity: All)

Most troubleshooting of the cockpit lights will consist of lamp replacement. If a substitution of the lamp does not solve the problem, a check of all connections in the wiring to the lamp holder should be made. A loose, corroded, or damaged pin connection is one cause of lighting failure.

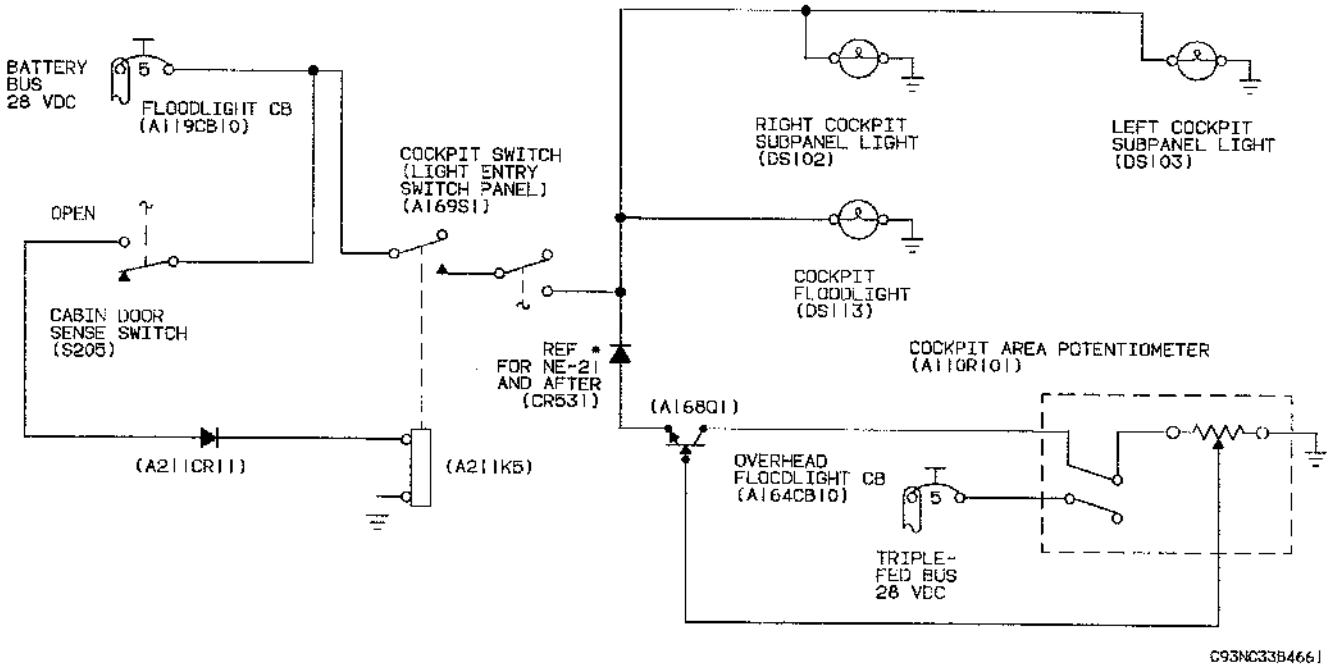
When troubleshooting instrument panel display units and control panels that have lighting as an integral part of their design, refer to the Collins Starship 1 Avionics System Manual (523-0774076-00111A or later revision).

When troubleshooting electroluminescent panels note the following:

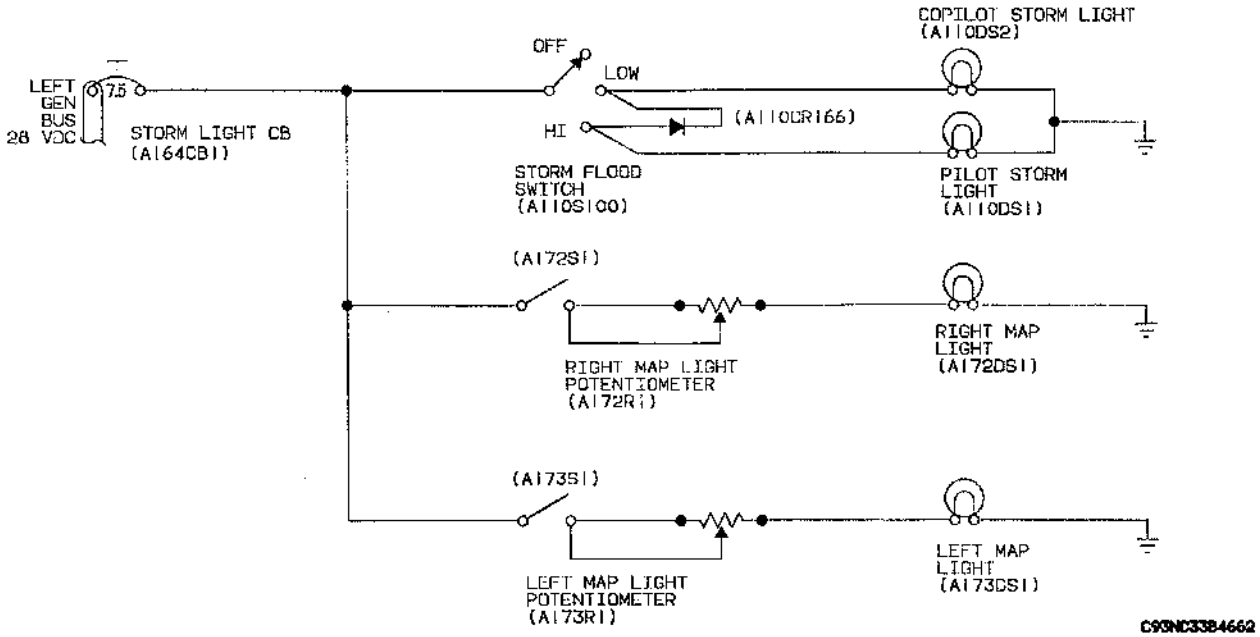
a. A short in one electroluminescent panel, or a faulty power supply, will prevent all of the electroluminescent panels in that circuit from illuminating.

b. Disconnect the output connector at the power supply to check power supply output. Use a potentiometer to vary output of the power supply.

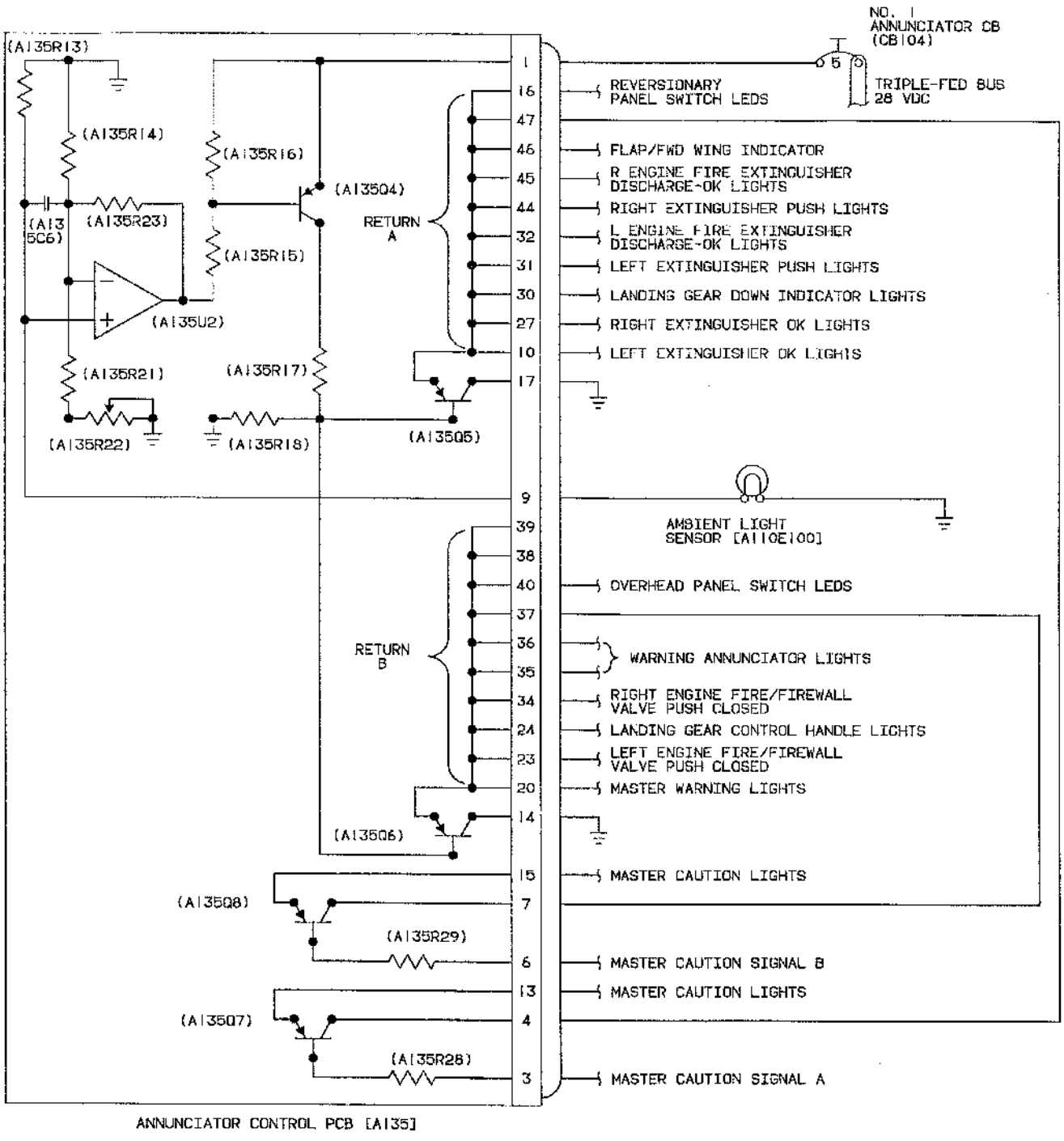
c. If the power supply output is good, check each individual electroluminescent panel by isolating it from the other panels in the circuit and applying the correct operating voltage.



Cockpit Subpanel and Flood Lights Schematic (Effectivity: All)
Figure 5



Cockpit Storm and Map Lights Schematic (Effectivity: All)
Figure 6



Ambient Light Sensor Circuit Schematic (Effectivity: All)
Figure 7

d. Electroluminescent panels cannot be repaired; therefore, replacement is the only option if the panel fails to illuminate.

COCKPIT LIGHTS - MAINTENANCE PRACTICES (Effectivity: All)

Ensure that the proper replacement lamp is used. The lamp number can usually be obtained from the base of the lamp. If not, reference the LAMP REPLACEMENT GUIDE in this chapter. Visually inspect the new lamp for a properly formed glass envelope. If it is distorted or misaligned with the lamp base, discard the lamp.

CHART 1
COCKPIT LIGHTS LAMP REPLACEMENT GUIDE
(Effectivity: All)

Location	Lamp #
Map Lights	1495
Subpanel Lights	1495
Master Warning Lights	90KF30
Master Caution Lights	90KF30
Annunciator Lights	327AS15
Flap/Fwd Wing Light	17-043
Landing-Gear-Down Light	17-043
Fire/FW Valve Light	OL-327
Engine Fire Ext. Lights	OL-327
Ground Comm. Light	OL-6839
Battery-Pwr.-On Light	OL-6839
Ldg. Gear In-transit Light	OL-327
Approach Plate Light	OL-6839
Storm Light	1939X
Cockpit Floodlight	1308

SUBPANEL ELECTROLUMINESCENT PANELS REMOVAL (Effectivity: All)

a. Remove all electrical power from the airplane.

WARNING

When removing knobs from the subpanel controls, ensure that all electrical power has been removed from the airplane to prevent accidental operation of any airplane system.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

NOTE

Removal of the instrument panel electroluminescent panels is the same for all panels with the exception of removing the different knobs.

b. Loosen the set screws and remove the knobs obstructing removal of the desired electroluminescent panel.

c. Remove all the electroluminescent panel attaching screws and remove the electroluminescent panel.

SUBPANEL ELECTROLUMINESCENT PANELS INSTALLATION (Effectivity: All)

CAUTION

When performing maintenance around electroluminescent panels take precautions not to damage the panels as they cannot be repaired.

a. Align the new electroluminescent panel with its subpanel.

b. Carefully match the power plug on the panel to the receptacle.

c. Install the mounting screws.

d. Install any knobs that were previously removed and tighten the set screws.

e. Restore electrical power to the airplane.

WARNING ANNUNCIATOR LAMP REPLACEMENT (Effectivity: All) (Figure 8)

a. Remove all electrical power from the airplane.

b. Push in on the legend plate (3). It should pop out when released.

NOTE

Pull on the legend plate (3) with fingers only. Damage to the legend plate (3) may occur if tools such as pliers are used for removal.

c. Pull the lamp holder (2) from the warning annunciator panel (1).

d. Extract both lamps (4) from the lamp holder (2) with fingers or a small screwdriver. Always replace the lamps (4) as a set.

e. Install the new lamps (4).

f. Ensure that the top of the lamp holder (2) and legend plate (3) are facing up.

g. Slide the lamp holder (2) into the warning annunciator panel (1) and push on the legend plate (3) until the lamp holder (2) snaps into place.

h. Restore electrical power to the airplane.

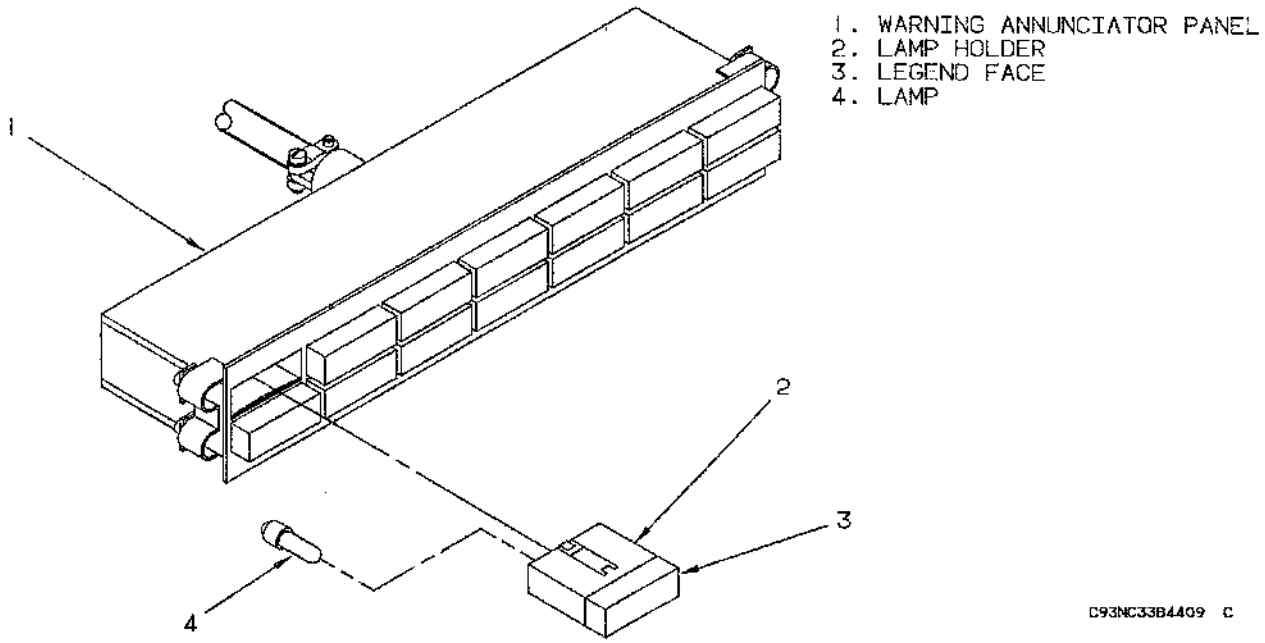
i. Check for illumination of the new lamps (4) by pressing the ANNUN test switch on the pilot's outboard subpanel.

MASTER CAUTION AND MASTER WARNING LIGHTS LAMP REPLACEMENT (Effectivity: All) (Figure 9)

a. Pull the No. 1 annunciator circuit breaker.

b. Grasp the finger grips (4) on the side of the legend face (3) and pull outward to remove the lamp holder (6).

c. Extract both lamps (5) from the lamp holder (6) with fingers or a small screwdriver. Always replace the lamps (5) as a set.



Annunciator Lamp Replacement (Effectivity: All)
Figure 8

d. Insert the new lamps (5) in the lamp holder (6).

e. Push the lamp holder (6) back into the switch body (1) until it clicks and locks into place.

NOTE

Ensure that the legend face (3) is not upside down.

f. Close the No. 1 annunciator circuit breaker.

g. Check for illunination of the new lamps (5) by pressing the ANNUN test switch on the pilots outboard subpanel.

EXTINGUISHER PUSH AND ENGINE FIRE F/W VALVE SWITCH LAMP REPLACEMENT (Effectivity: All) (Figure 10)

a. Remove all electrical power from the airplane.

CAUTION

Failure to disconnect the battery from the "hot" battery bus before lamp replacement may result in accidental discharge of the engine fire extinguishers or closing of the firewall valves.

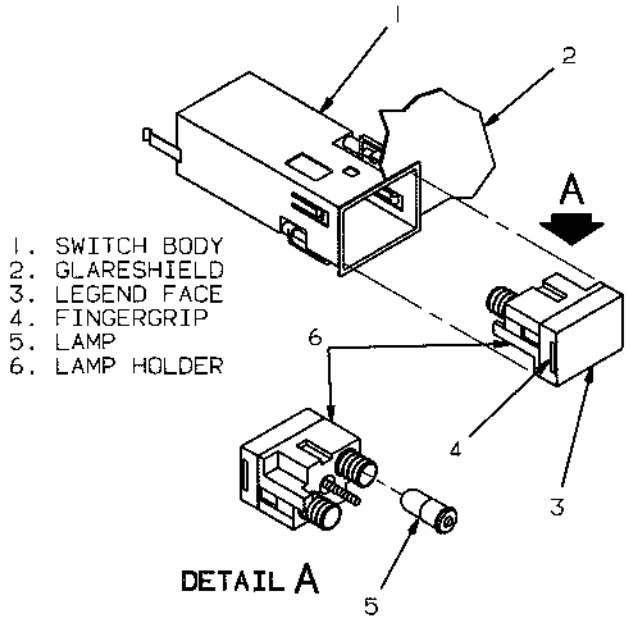
NOTE

The plastic guard of the extinguisher push switch is safety wired whereas the F/W valve switch guard is not.

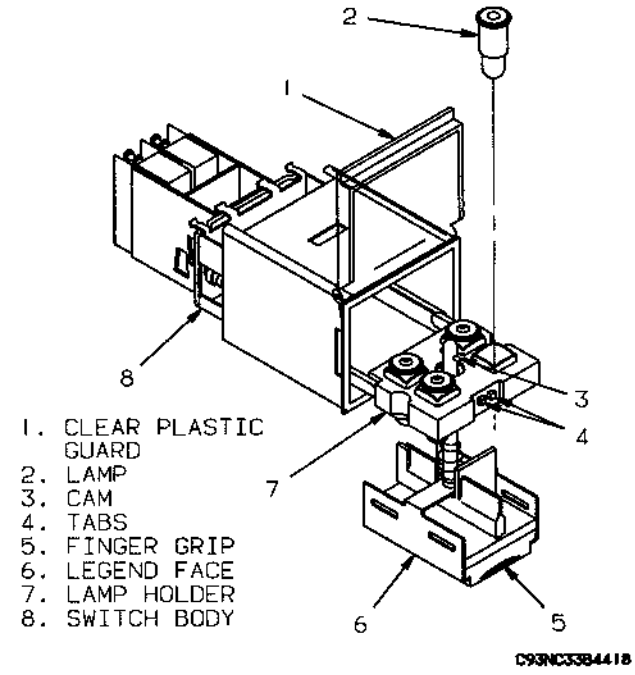
b. Remove the copper safety wire from the clear plastic guard (1) if applicable.

c. Flip the clear plastic guard (1) up, grasp the finger grips (5) on the sides of the legend face (6), and pull outward.

d. While fully extended, rotate the legend face (6) a 1/4 turn in either direction.



Master Caution and Master Warning Lights Lamp Replacement (Effectivity: All)
Figure 9



Typical Switch Lights Lamp Replacement (Effectivity: All)
Figure 10

e. Push in on the legend face (6) to release the lamp holder tabs (4) from their slots.

f. Pull the lamp holder (7) out and swivel it down to access the lamps (2).

g. Remove colored boots from the lamps (2) as required.

h. Extract the four lamps (2) from the lamp holder (7) with fingers or a small screwdriver. Always replace the lamps (2) as a set.

i. Install the new lamps (2).

j. Install any colored boots previously removed.

k. Fully extend the legend face (6) to engage the tab cam (3).

l. Retract the tabs (4) with the tab cam (3) by rotating the legend face (6) 1/4 turn in either direction.

m. Once the tabs (4) are retracted, swivel the lamp holder (7) up and slide it all the way into the switch body (8).

NOTE

Install the legend face (6) with the nomenclature facing upright. The lamp holder (7) will not seat properly if the legend face (6) is inverted.

n. Extend the tabs (4) by rotating the legend face (6) 1/4 turn. The lamp holder (7) will lock into place as the tabs (4) extend.

o. Push the legend face (6) into the switch body (8) until it clicks. An audible click indicates the legend face (6) has locked in place.

p. Safety wire the clear plastic guard (1) of the extinguisher push switch closed with MS20995CY15 copper safety wire.

NOTE

The F/W valve switch guard is not safety wired.

q. Restore electrical power to the airplane.

BATTERY-POWER-ON, LANDING-GEAR-DOWN, FLAP/FORWARD WING AND GROUND COMMUNICATIONS SWITCH LIGHTS LAMP REPLACEMENT (Effectivity: All) (Figure 11)

a. Remove all electrical power from the airplane.

b. Grasp the finger grips (5) on the sides of the legend face (4) and pull outward.

c. Swivel the lamp holder (6) down to access the lamps (3).

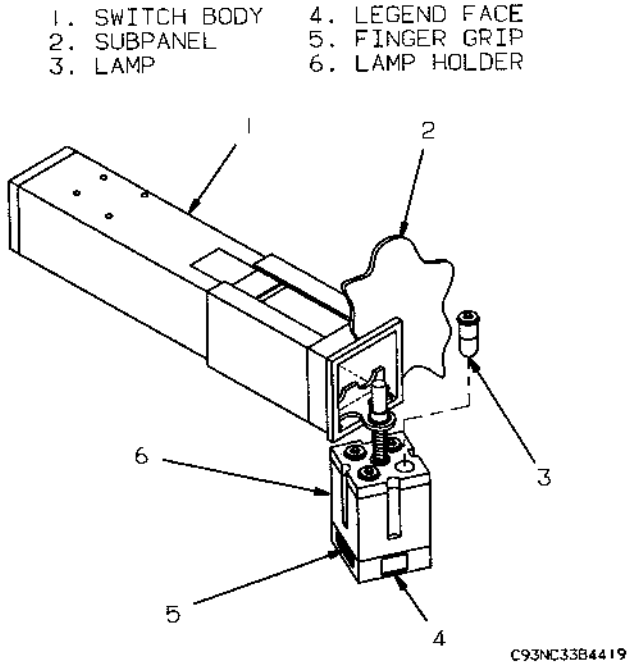
d. Extract the four lamps (3) from the lamp holder (6) with fingers or a small screwdriver. Always replace the lamps (3) as a set.

NOTE

Install the legend face (4) with the nomenclature facing upright. The lamp holder (6) will not seat properly if the legend face (4) is inverted.

e. Swivel the lamp holder (6) up and slide it into the switch body (1). An audible click is an indication that the lamp holder (6) and legend face (4) are locked in place.

f. Restore electrical power to the airplane.



Typical Switch Lights Lamp Replacement (Effectivity: All) Figure 11

LANDING GEAR IN-TRANSIT LIGHT LAMP REPLACEMENT (Effectivity: All) (Figure 12)

a. Pull the landing gear power circuit breaker and remove all electrical power from the airplane.

WARNING

When replacing the lamp (2), never move the landing gear handle (4) from the extended (down) position.

b. Unscrew the lamp retainer (1) and remove the defective lamp (2).

c. Install a new lamp (2) in the knob assembly (3).

d. Screw the lamp retainer (1) back on the knob assembly (3).

WARNING

Verify that the landing gear handle (4) is in the extended (down) position before restoring electrical power to the airplane.

e. Restore electrical power to the airplane and close the landing gear power circuit breaker.

APPROACH PLATE LAMP REPLACEMENT (Effectivity: All) (Figure 13)

a. Pull the flight instrument light circuit breaker.

b. Press the PUSH button on the approach plate to open the holder (1).

c. Allow the cover to swing up and expose the lamp holder (4).

d. Pull the faulty lamp (3) out of the lamp holder (4).

e. Install a new lamp (3).

f. Close the approach plate holder (1).

g. Close the flight instrument light circuit breaker.

SUBPANEL LIGHT LAMP REPLACEMENT (Effectivity: All) (Figure 14)

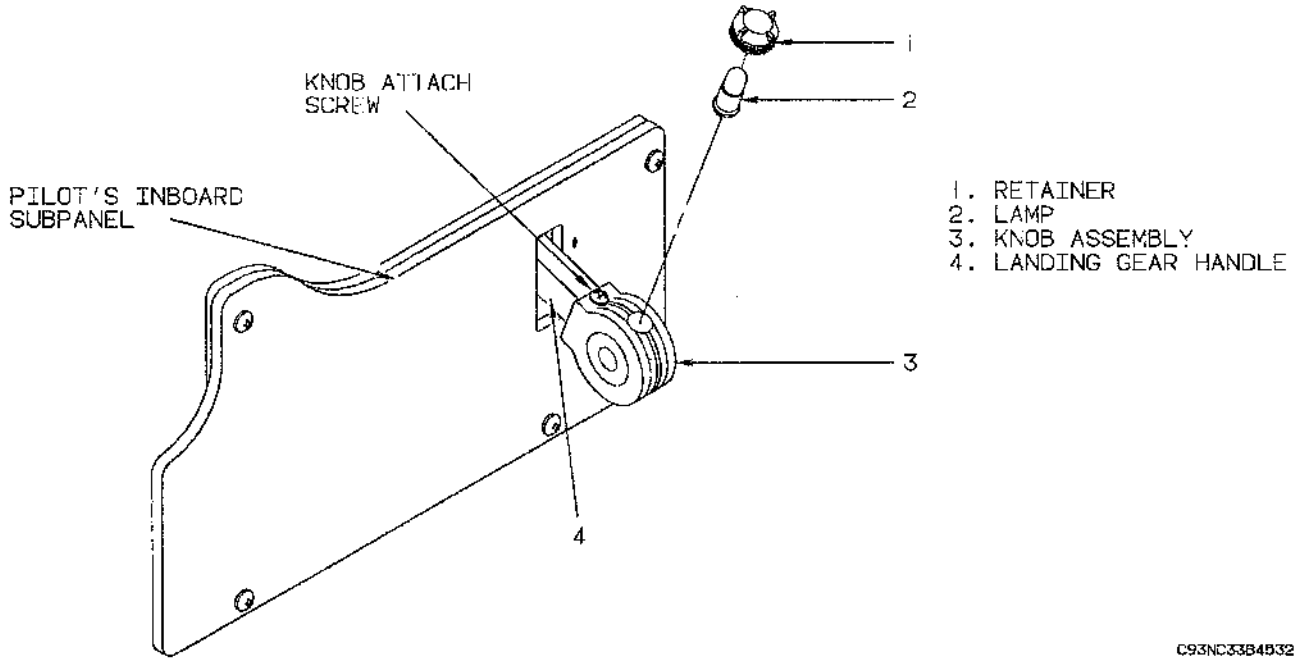
a. Pull the floodlight and overhead floodlight circuit breakers.

b. Reach through the lamp access opening under the control column (1), push the lamp (2) in and turn counterclockwise to disengage the lamp pins.

c. Pull the defective lamp (2) from the lamp holder (5).

d. Reach through the lamp access opening under the control column (1), and install a new lamp (2) by pushing in, turning clockwise and releasing to engage the lamp pins.

e. Close the floodlight and overhead floodlight circuit breakers.



Landing Gear In-Transit Light Lamp Replacement (Effectivity: All)
Figure 12

MAP LIGHT LAMP REPLACEMENT (Effectivity: All) (Figure 15)

a. Pull the storm light circuit breaker.

b. Remove the screw (4) attaching the map light panel (3) to the upholstery panel bracket (7).

c. Push up on the map light panel (3) and swing the bottom out until the map light panel (3) separates from the upholstery panel bracket (7).

d. Snap the lamp holder (6) out of the reflector assembly (2) to access the lamp (5).

e. Remove the lamp (5) and install a new one.

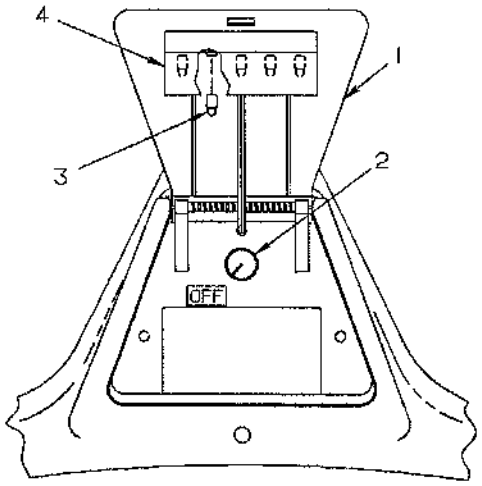
f. Snap the lamp holder (6) back into the reflector assembly (2).

g. Place the map light panel (3) over the upholstery panel bracket (7) and swing the bottom of the map light (3) panel over the upholstery panel to align the mounting screw hole with the bracket (7).

h. Install the mounting screw (4).

i. Close the storm light circuit breaker.

1. APPROACH PLATE HOLDER
2. POTENTIOMETER
3. LAMP
4. LAMP HOLDER



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Approach Plate Lamp Replacement (Effectivity: All) Figure 13

COCKPIT STORM LIGHT LAMP REPLACEMENT (Effectivity: All) (Figure 16)

a. Pull the storm light circuit breaker.

b. Turn the two fasteners (6) 1/4 turn counterclockwise and remove the lens (5) from the storm light assembly (3).

c. Remove the defective lamp (4) and install a new one.

d. Align the lens (5) with the storm light assembly (3) and turn fasteners (6) 1/4 turn clockwise to lock it in place.

e. Close the storm light circuit breaker.

COCKPIT FLOODLIGHT LAMP REPLACEMENT (Effectivity: All) (Figure 16)

a. Pull the floodlight and overhead floodlight circuit breakers.

b. Pull the nozzle and lens (7) off the light assembly (1) to access the lamp (8).

c. Remove the defective lamp (8) and install a new one.

d. Install the lens and nozzle (7).

e. Close the floodlight and overhead floodlight circuit breakers.

AMBIENT-LIGHT-SENSOR REPLACEMENT (Effectivity: All) (Figure 17)

CAUTION

The overhead panel must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Remove all electrical power from the airplane.

b. Loosen the set screws and remove the knobs (7) from the overhead panel controls.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

c. Remove the eight electroluminescent panel mounting screws (6).

d. The overhead switch panel assembly (5) is hinged and will swing down once the electroluminescent panel (8) is removed. Pull it down far enough to disconnect and tag the electrical leads on the ambient-light sensor (11).

e. Remove the two screws (10) attaching the ambient-light sensor (11) and remove it from the panel (5).

f. Align the new ambient-light sensor (11) with the panel mounting holes and install the attaching screws (10).

g. Connect wiring to the new ambient-light sensor (11) according to the BEECHCRAFT Starship 1 Wiring Diagram Manual or tags.

h. Fold the hinges up to position the overhead switch panel assembly (5), then align the electroluminescent panel (8) over it.

i. Ensure that the electroluminescent panel (8) power receptacle is connected, then install the eight mounting screws (6).

j. Install the knobs (7) and tighten the set screws.

k. Restore electrical power to the airplane.

LIGHTS-MASTER-SWITCH RELAY REPLACEMENT (Effectivity: All) (Figure 17)

CAUTION

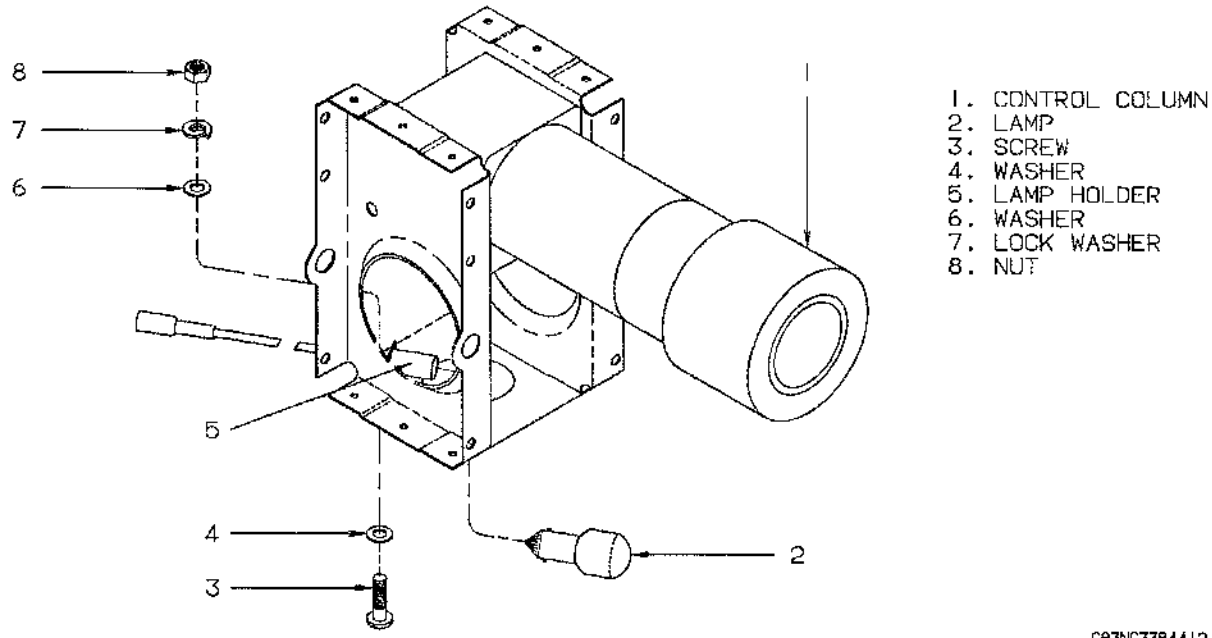
The overhead panel must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Remove all electrical power from the airplane.

b. Loosen the set screws and remove the knobs (7) from the overhead panel controls.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.



Subpanel Light Lamp Replacement (Effectivity: All)
Figure 14

c. Remove the eight electroluminescent panel mounting screws (6).

d. The overhead switch panel assembly (5) is hinged and will swing down once the electroluminescent panel (8) is removed. Pull it down far enough to access the lights-master-switch relays (4).

e. Disconnect and tag the wiring to the desired lights-master-switch relay (4).

f. Remove the two screws (12), washers (14) and nuts (13) attaching the lights-master-switch relay (4) to the overhead switch panel assembly (5).

g. Remove the relay (4) and align the new one with the mounting holes.

h. Install the mounting screws (12) from the front side of the panel (5).

i. Install the washers (14) and nuts (13) on the screws (12).

j. Reconnect wiring to the lights-master-switch relay (4) according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

k. Fold the hinges (3) back to position the overhead switch panel assembly (5), then align the electroluminescent panel (8) over it.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

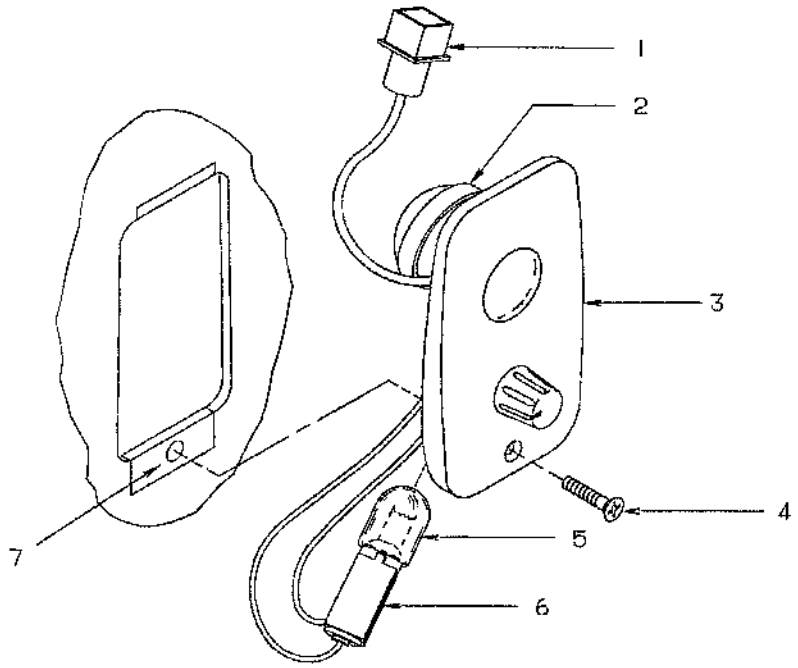
l. Ensure that the electroluminescent panel power receptacle is connected, then install the eight mounting screws (6).

m. Install the knobs (7) and tighten the set screws.

n. Restore electrical power to the airplane.

LIGHT SWITCH MODULE REMOVAL (Effectivity: All) (Figure 17)

a. Remove all electrical power from the airplane.



1. ELECTRICAL CONNECTOR
2. REFLECTOR ASSEMBLY
3. MAP LIGHT PANEL
4. SCREW
5. LAMP
6. LAMP HOLDER
7. UPHOLSTERY PANEL BRACKET

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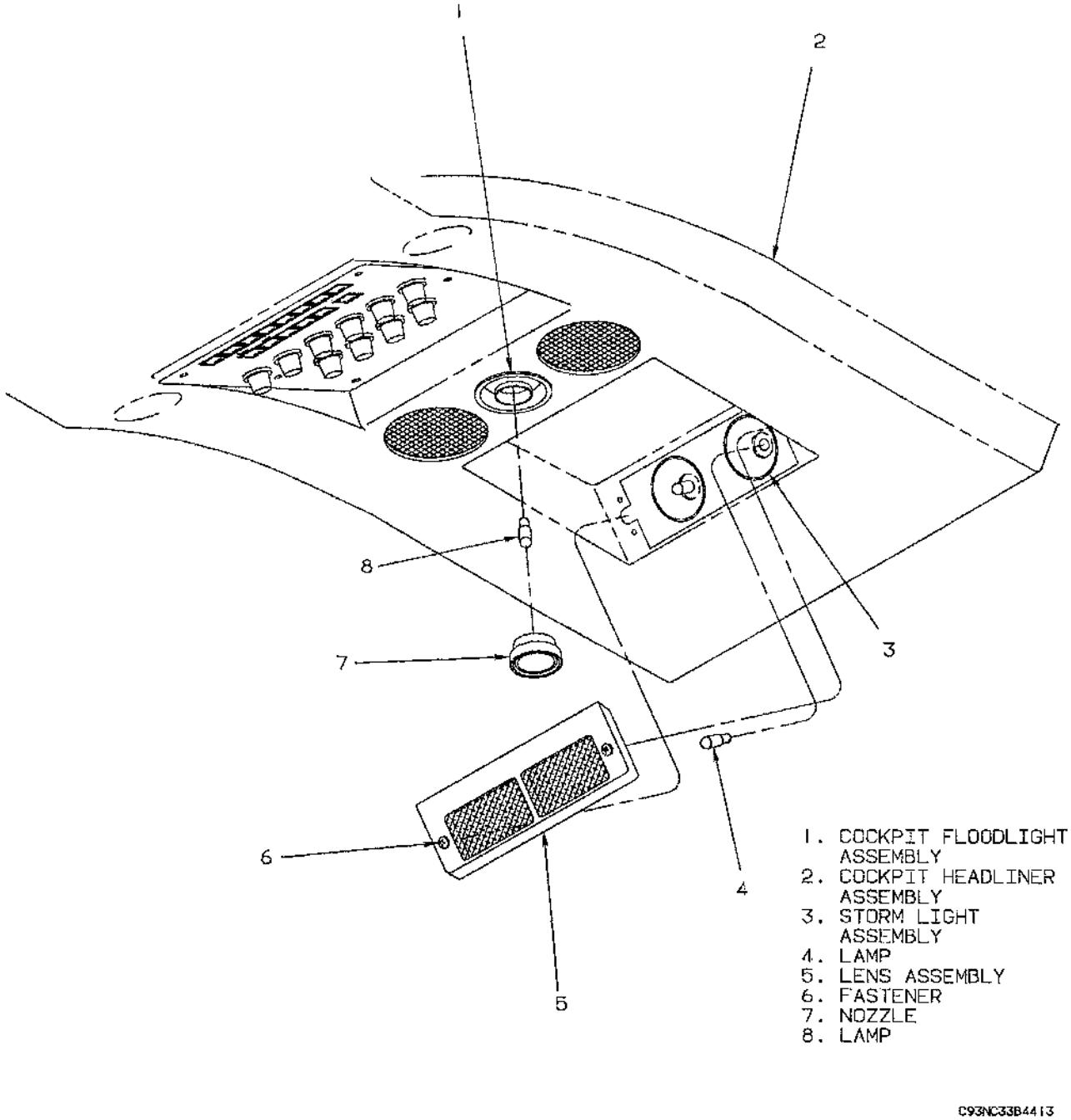
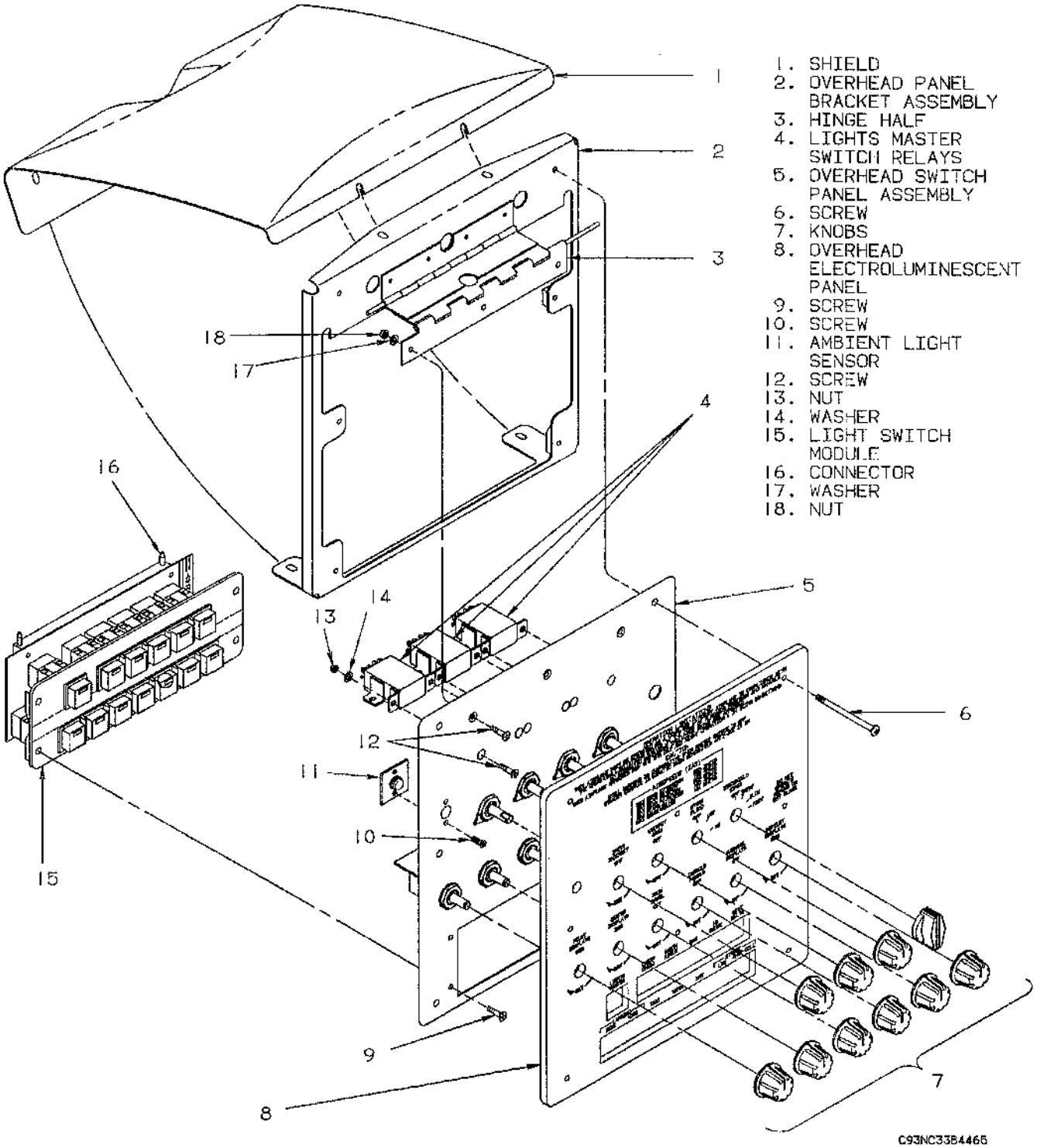
Map Light Replacement (Effectivity: All)
Figure 15



Cockpit Flood and Storm Lights Lamp Replacement (Effectivity: All)
Figure 16



Overhead Switch Panel Assembly Component Replacement (Effectivity: All)
Figure 17

b. Loosen the set screws and remove the knobs (7) from the overhead panel controls.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

c. Remove the eight electroluminescent panel mounting screws (6).

d. The overhead switch panel assembly (5) is hinged and will swing down once the electroluminescent panel (8) is removed. Pull it down far enough to disconnect the electrical connector (16) from the light switch module (15).

e. Remove the four mounting screws (9) from the light switch module (15).

f. Remove the light switch module (15).

LIGHT SWITCH MODULE INSTALLATION (Effectivity: All) (Figure 17)

CAUTION

The overhead panel must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Align the mounting holes in the light switch module (15) wthe the overhead switch panel assembly (5) and install the four mounting screws (9).

b. Reconnect the light switch module electrical connector (16).

c. Fold the hinges (3) back to position the overhead switch panel assembly (5), then align the electroluminescent panel (8) over it.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

d. Ensure that the electroluminescent panel power receptacle is connected, then install the eight mounting screws (6).

e. Install the knobs (7) and tighten the set screws.

f. Restore electrical power to the airplane.

POTENTIOMETER MODULE, POTENTIOMETER AND ROTARY SWITCH REMOVAL (Effectivity: All) (Figure 18)

a. Remove all electrical power from the airplane.

b. Loosen the set screws and remove the knobs (5) from the overhead panel controls.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

c. Remove the eight electroluminescent panel mounting screws (8) and remove the electroluminescent panel (6).

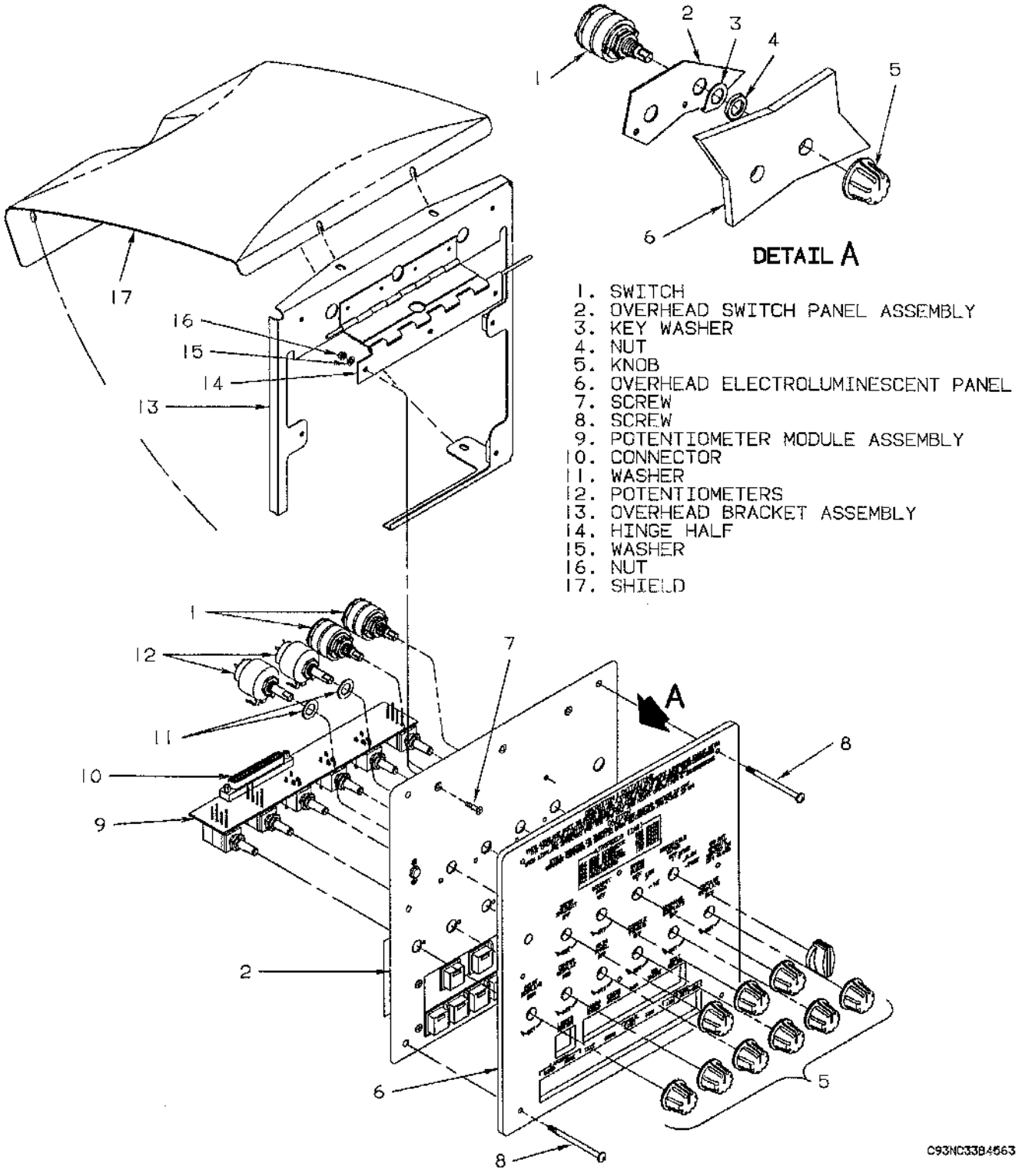
d. The overhead switch panel assembly (2) is hinged and will swing down once the electroluminescent panel (6) is removed. Pull it down far enough to access the potentiometers (12) and switches (1).

e. Disconnect the electrical connector (10) to the potentiometer module assembly (9) or disconnect and tag the leads to the applicable switch (1) or potentiometer (12).

f. Remove the nut (4) and key washer (3) from the applicable potentiometer (12) or switch (1), and remove it from the backside of the panel (2).

NOTE

Mounting hardware for all six potentiometers on the potentiometer module assembly (9) must be removed before the module assembly (9) can be removed from the overhead switch panel assembly (2).



Overhead Switch Panel Assembly Component Replacement (Effectivity: All)
Figure 18

POTENTIOMETER MODULE, POTENTIOMETER AND ROTARY SWITCH INSTALLATION (Effectivity: All) (Figure 18)

CAUTION

The overhead panel must meet electrical bonding requirements specified in Chapter 20-00-00.

NOTE

The INSTR INDIRECT and COCKPIT AREA potentiometers (12) must be installed with a washer (11) between the potentiometer (12) and the overhead switch panel assembly (2).

a. Align the potentiometer (12) or switch (1) shafts with the applicable overhead switch panel (2) mounting holes.

NOTE

Mounting hardware for all six potentiometers on the potentiometer module assembly (9) must be installed after the potentiometer shafts are aligned with the holes in the overhead switch panel assembly (2).

b. Install a key washer (3) and nut (4) on each shaft.

c. Connect the electrical connector (10) to the potentiometer module assembly (9) or the wiring to the new switch (1) or potentiometer (12) according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

d. Fold the hinges back to position the overhead switch panel assembly (2), then align the electroluminescent panel (6) over it.

CAUTION

When performing maintenance around electroluminescent panels, take precautions not to damage the panels as they cannot be repaired.

e. Ensure that the electroluminescent panel (6) power receptacle is connected, then install the eight mounting screws (8).

f. Install the knobs (5) and tighten the set screws.

g. Restore electrical power to the airplane.

GLARESHIELD ELECTROLUMINESCENT LIGHT PANEL REMOVAL (Effectivity: All) (Figure 19)

a. Pull the instrument indirect lights circuit breaker.

b. Remove the mounting screws (3) attaching the electroluminescent light panel (2) to the glareshield (1).

c. Lower the glareshield electroluminescent light panel (2) to access the electrical connector (4).

d. Disconnect the electrical connector (4).

GLARESHIELD ELECTROLUMINESCENT LIGHT PANEL INSTALLATION (Effectivity: All) (Figure 19)

a. Reconnect the electrical connector (4).

b. Feed the wiring and connector into the glareshield (1) while aligning the mounting holes on the glareshield electroluminescent light panel (2) with mounting bracket holes.

c. Install the mounting screws (3).

d. Close the instrument indirect lights circuit breaker.

LCD AND INCANDESCENT LIGHTS DIMMER POWER SUPPLY REMOVAL (Effectivity: All) (Figure 20)

a. Pull the LCD display light circuit breaker located on the aft circuit breaker panel.

NOTE

The power supply (2) is mounted outboard and forward of the pilot between the fuselage and the cockpit instrument panel structure. The mounting screws (1) are accessible from the inboard side of the panel, and the power supply (2) can be accessed through an opening in the panel directly forward of the mounting screws (1).

b. Reach behind the panel and disconnect the electrical connector.

c. Remove the two mounting screws (1) and washers (3) while supporting the power supply (2).

d. Remove the power supply (2) from the airplane.

LCD AND INCANDESCENT LIGHTS DIMMER POWER SUPPLY INSTALLATION
(Effectivity: All)
(Figure 20)

CAUTION

The LCD and incandescent lights power supply (2) must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Align the power supply nutplates with the panel mounting holes and install the two screws (1) and washers (3).

b. Reconnect the electrical connector.

c. Close the LCD display light circuit breaker.

ELECTROLUMINESCENT PANELS POWER SUPPLY REMOVAL
(Effectivity: All)
(Figure 21)

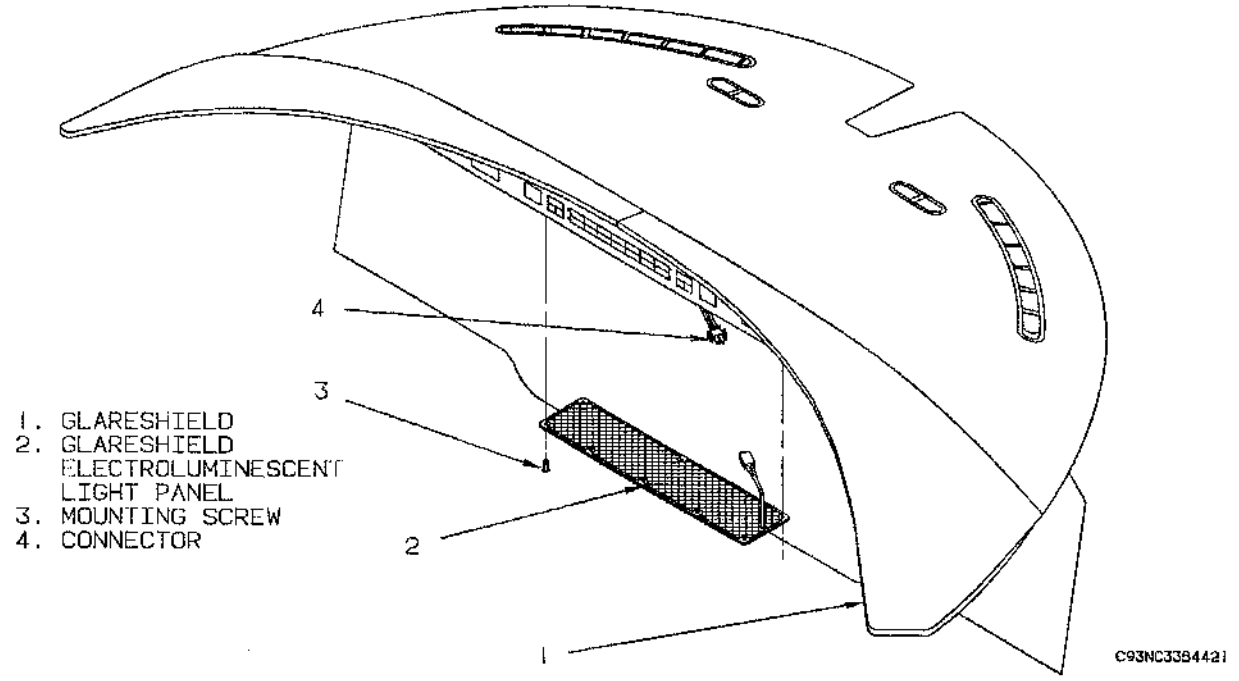
a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-20-00 and remove the cockpit center floorboard panel (143BTC).

c. Tag and disconnect the electrical connectors from the power supply (3, 4, 6 or 7) to be removed.

d. Remove the mounting screws (1) and washers (2).

e. Remove the power supply (3, 4, 6 or 7).



Glareshield Electroluminescent Light Panel Installation (Effectivity: All)
Figure 19

ELECTROLUNINESCENT PANELS POWER SUPPLY INSTALLATION
(Effectivity: All)
(Figure 21)

CAUTION

The electroluminescent panel power supplies must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Align the power supply (3, 4, 6 or 7) with the panel assembly nutplates.

b. Install the four mounting screws (1) and washers (2).

c. Reconnect the electrical connectors according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

d. Refer to Chapter 53-20-00 and install the cockpit center floorboard panel (143BTC).

e. Restore electrical power to the airplane.

DIMMING TRANSISTOR REMOVAL
(Effectivity: All)
(Figure 22)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-10-00 and remove the pilot's seat and console upholstery panel.

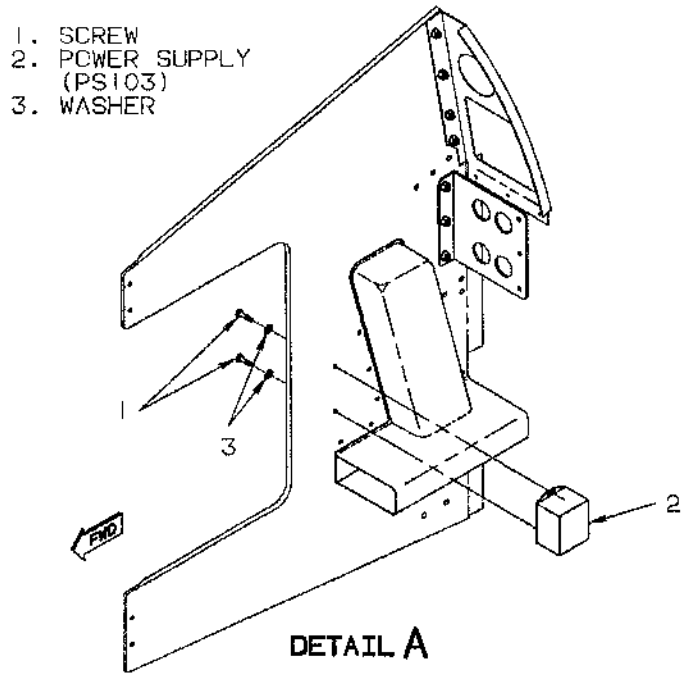
c. Remove the upper wire tray cover from the pilot's console.

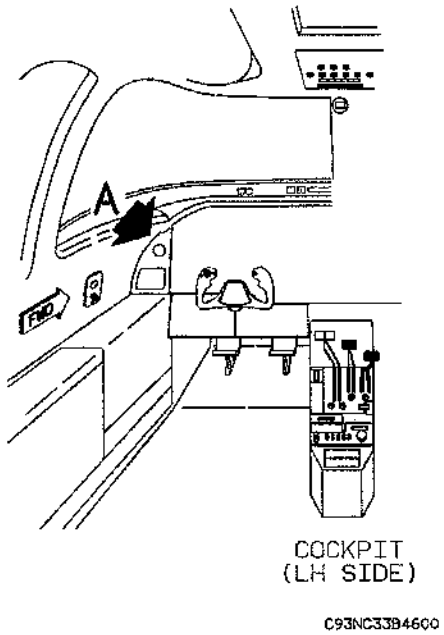
d. Remove the two screws (1) and washers (2) attaching the dimming transistor panel assembly (14) to the pilot's console.

e. Pull the dimming transistor panel assembly (14) out far enough to disconnect the electrical connector.

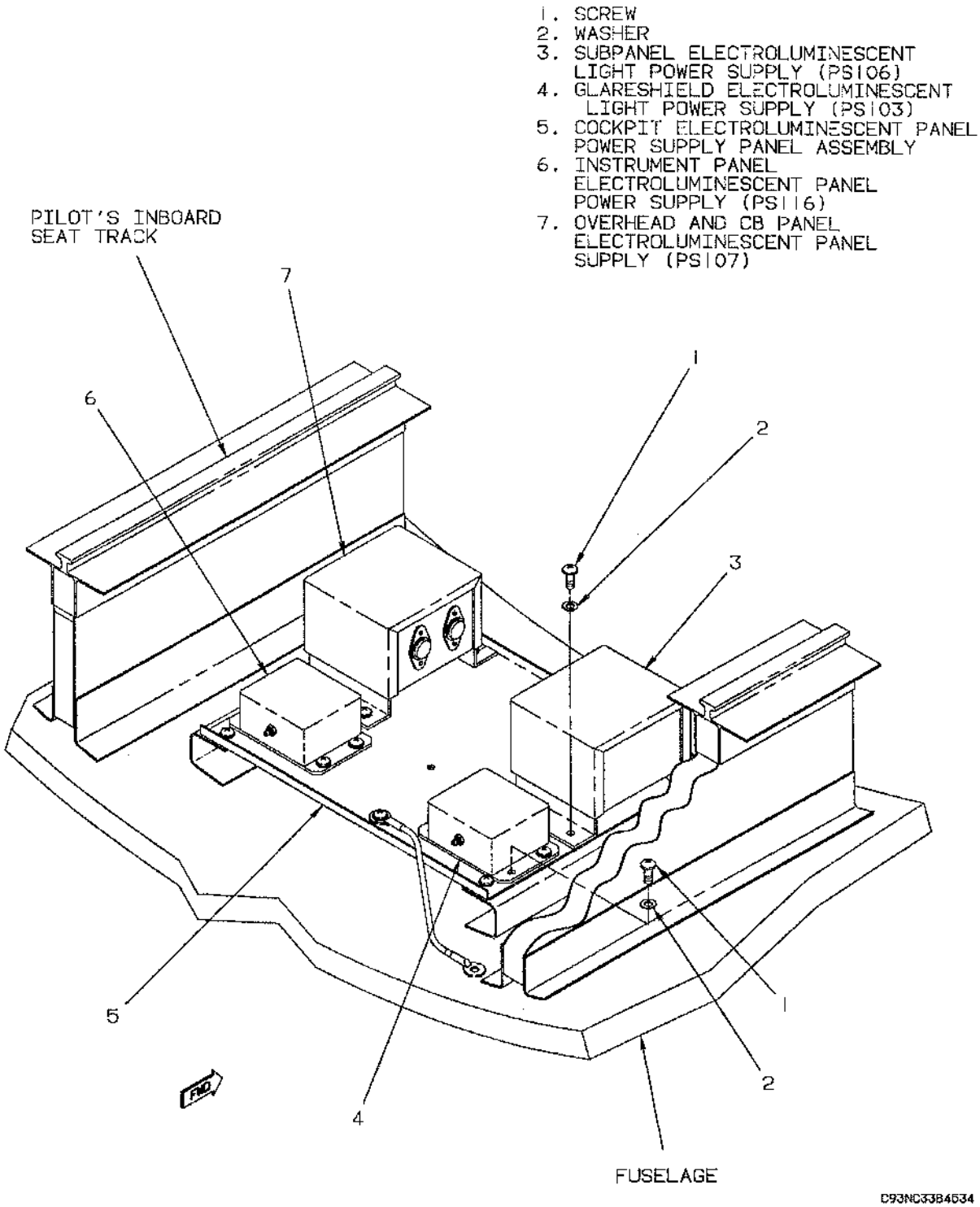
f. Tag and unsolder the wires to the dimming transistor (10) being removed.

g. Remove two attaching screws (8) and washers (9) and pull the dimming transistor (10) away from the insulator (11) and heat sink (12).

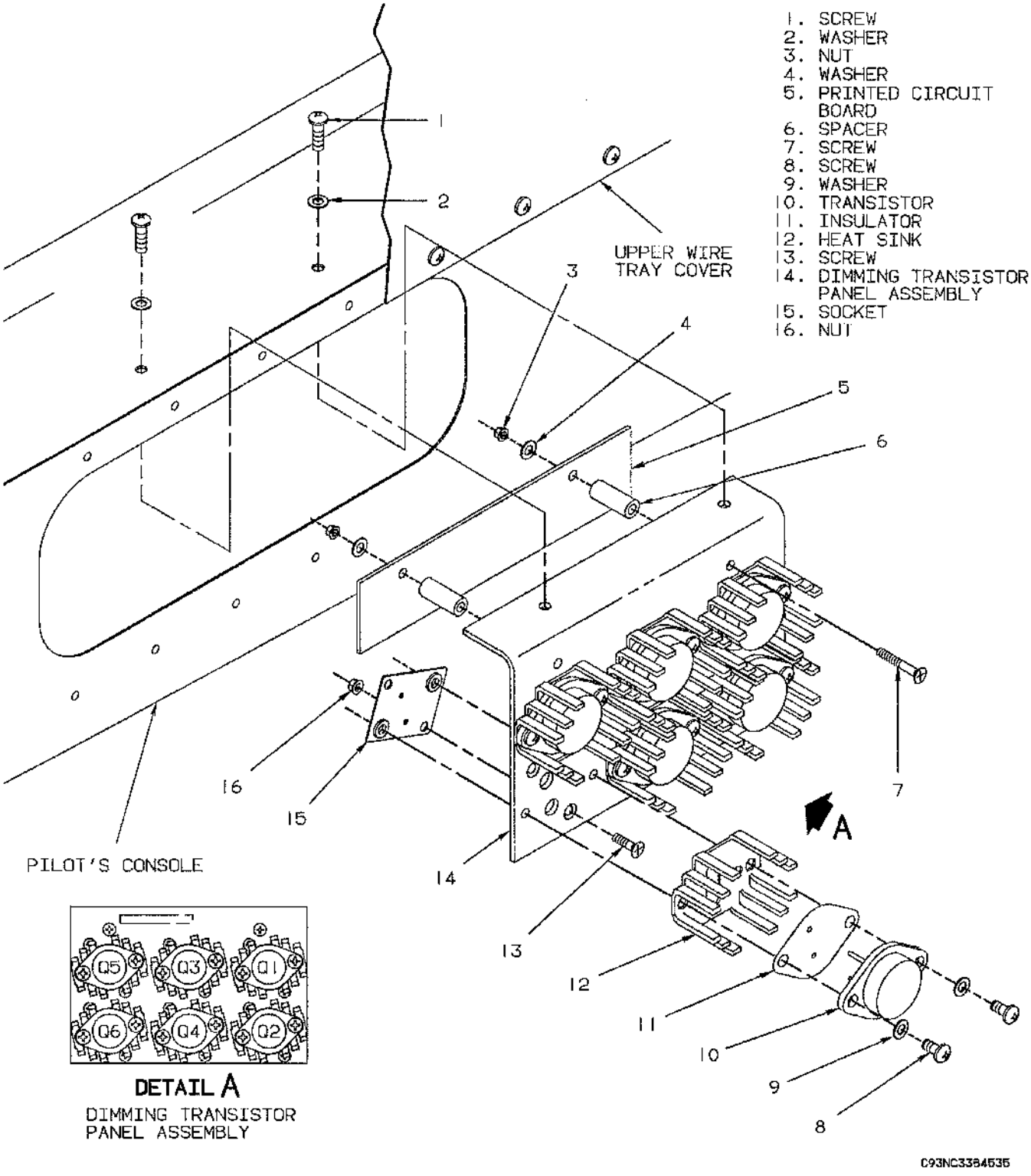




LCD and Incandescent Lights Dimmer Power Supply Installation (Effectivity: All)
Figure 20



Electroluminescent Panels Power Supply Installation (Effectivity: All)
Figure 21



Dimming Transistor Installation (Effectivity: All)
Figure 22

DIMMING TRANSISTOR INSTALLATION
(Effectivity: All)
(Figure 22)

CAUTION

The dimming transistor panel assembly must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Position the new dimming transistor (10) over the insulator (11) and heat sink (12), aligning all three with the panel assembly (14) and socket (15).

b. Install the two mounting screws (8) and washers (9).

c. Solder the leads to the dimming transistor (10) according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

d. Align the dimming transistor panel assembly (14) with the mounting holes in the pilot's console, and attach with the two mounting screws (1) and washers (2).

e. Reconnect the electrical connector.

f. Install the upper wire tray cover.

g. Refer to Chapter 25-10-00 and install the pilot's console upholstery panel and seat.

h. Restore electrical power to the airplane.

CABIN LIGHTS - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

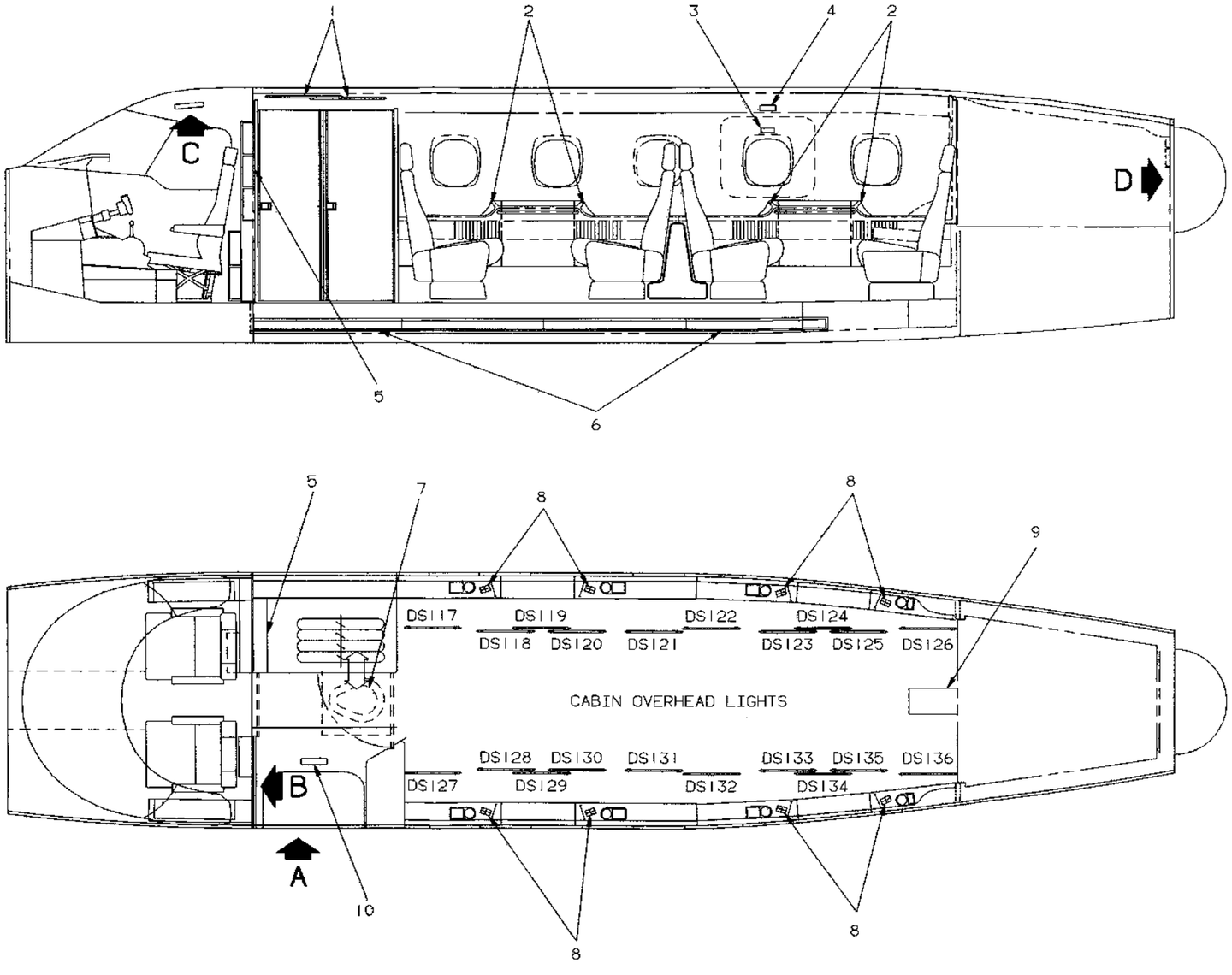
The cabin lights consist of entry lights, aisle lights, overhead lights, no-smoking sign lights, fasten-seatbelt sign lights and exit signs. These lights provide warnings and illuminate the cabin area.

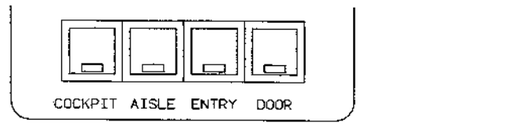
ENTRY LIGHTS (Effectivity: All) (Figures 1, 2, and 3)

The entry lights allow illumination of the interior before the airplane is entered. The entry-light switch panel is mounted on the LH forward partition for easy access from the exterior of the airplane. When the cabin door is open, the cabin door sense switch (S205) closes relay A211K5 with voltage from the "hot" battery bus. Relay A211K5 connects battery voltage to the switches placarded COCKPIT (S1), AISLE (S2), ENTRY (S3) and DOOR (S4) on the entry light switch panel. The COCKPIT switch (S1) illuminates the cockpit with the cockpit flood and subpanel lights. The AISLE switch (S2) illuminates the cabin floor area with the aisle lights. The ENTRY switch (S3) illuminates the cabin entry area by powering two fluorescent entry lights (a single, longer fluorescent light in NC-21 and after) with a 115 vac power supply. A light inside the entry light switch panel is turned on by the DOOR switch (S4) to illuminate the area in front of the cabin door and the panel itself. The DOOR switch (S4) also provides 28 vdc to the step lights power supply (PS111) which inverts the DC input to a 115 vac output for illumination of the cabin door steps with the three step light electroluminescent panels. When the cabin door is locked, relay A211K5 opens, disconnecting the "hot" battery bus from all four switches. Only the ENTRY switch is powered from two sources, powering the entry lights from the triple-fed bus when the cabin door is locked and the bus is energized.

CABIN AISLE LIGHTS (Effectivity: All) (Figures 1, 2 and 3)

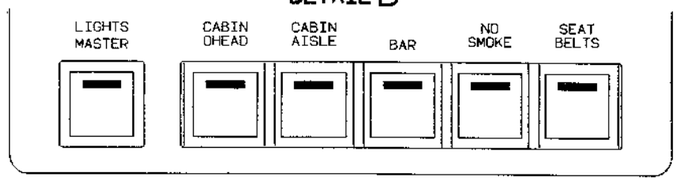
When the cabin door is locked, the aisle lights are controlled from the cockpit by the pushbutton switch (A110S103) placarded CABIN AISLE on the overhead panel. The triple-fed bus powers the two aisle light power supplies (PS 108 and PS109) through its cabin light circuit breaker (A119CB6 or A164CB9) and the CABIN AISLE switch (A110S103). Each aisle light power supply (PS108 and PS109) powers the lights on one side of the aisle, with four electroluminescent light panels per side.





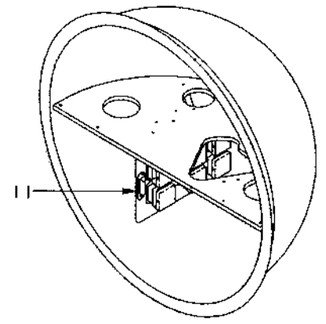
ENTRY LIGHT SWITCH PANEL

DETAIL B



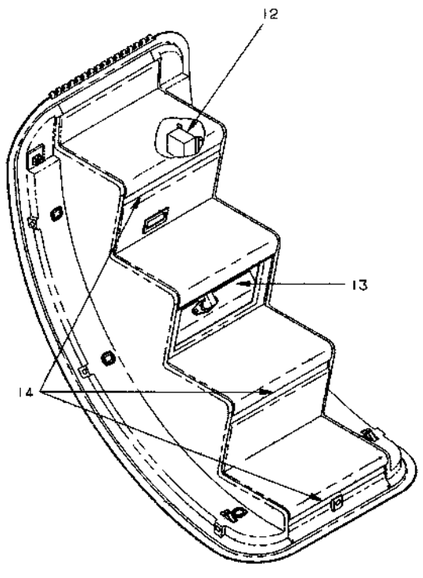
OVERHEAD PANEL

DETAIL C



AFT PRESSURE DOME

DETAIL D



CABIN DOOR

DETAIL A

1. CABIN ENTRY LIGHTS
2. NO SMOKING/FSB SIGNS
3. EXIT/PULL SIGN
4. EXIT SIGN
5. ENTRY AND OVERHEAD LIGHT POWER SUPPLIES
6. AISLE LIGHTS (RH)
7. TOILET
8. MOTORIZED WINDOW SHADE AND READING LIGHT SWITCHES
9. AFT STEP ELECTRICAL EQUIPMENT PANEL
10. EXIT SIGN
11. OVERHEAD LIGHT POWER SUPPLIES
12. STEP LIGHT POWER SUPPLY (PS111)
13. DOOR INSTRUCTION LIGHT PANEL
14. CABIN DOOR STEP LIGHTS

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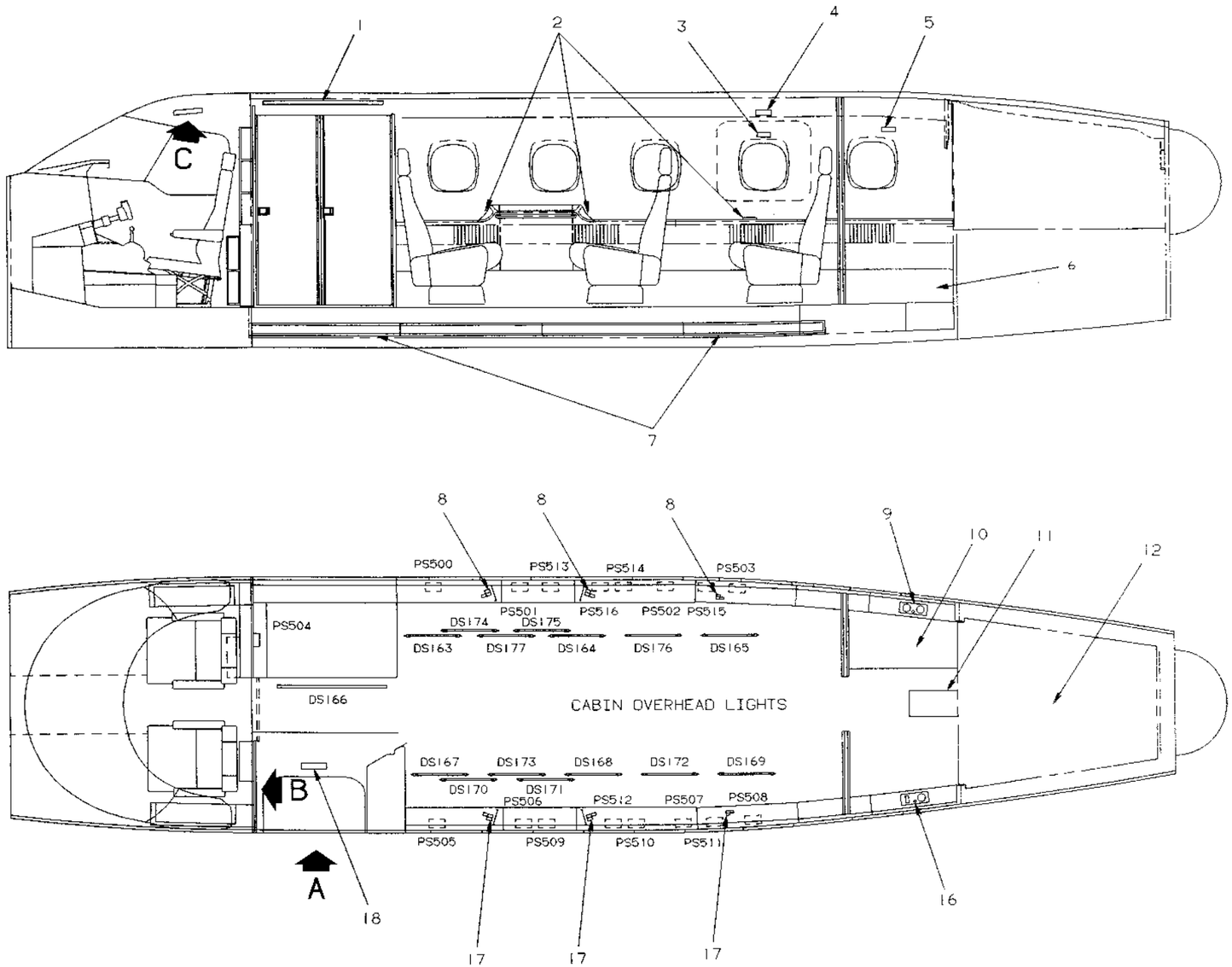
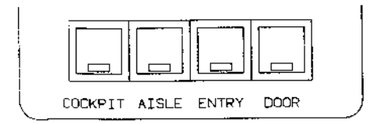
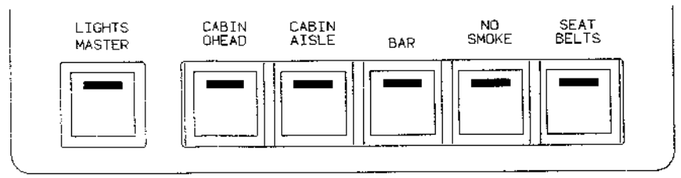
Cabin Lights (Effectivity: NC-4 through NC-20)
Figure 1





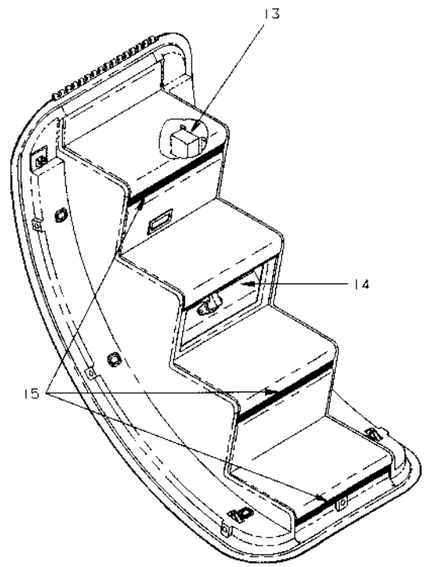
ENTRY LIGHT SWITCH PANEL

DETAIL B



OVERHEAD PANEL

DETAIL C



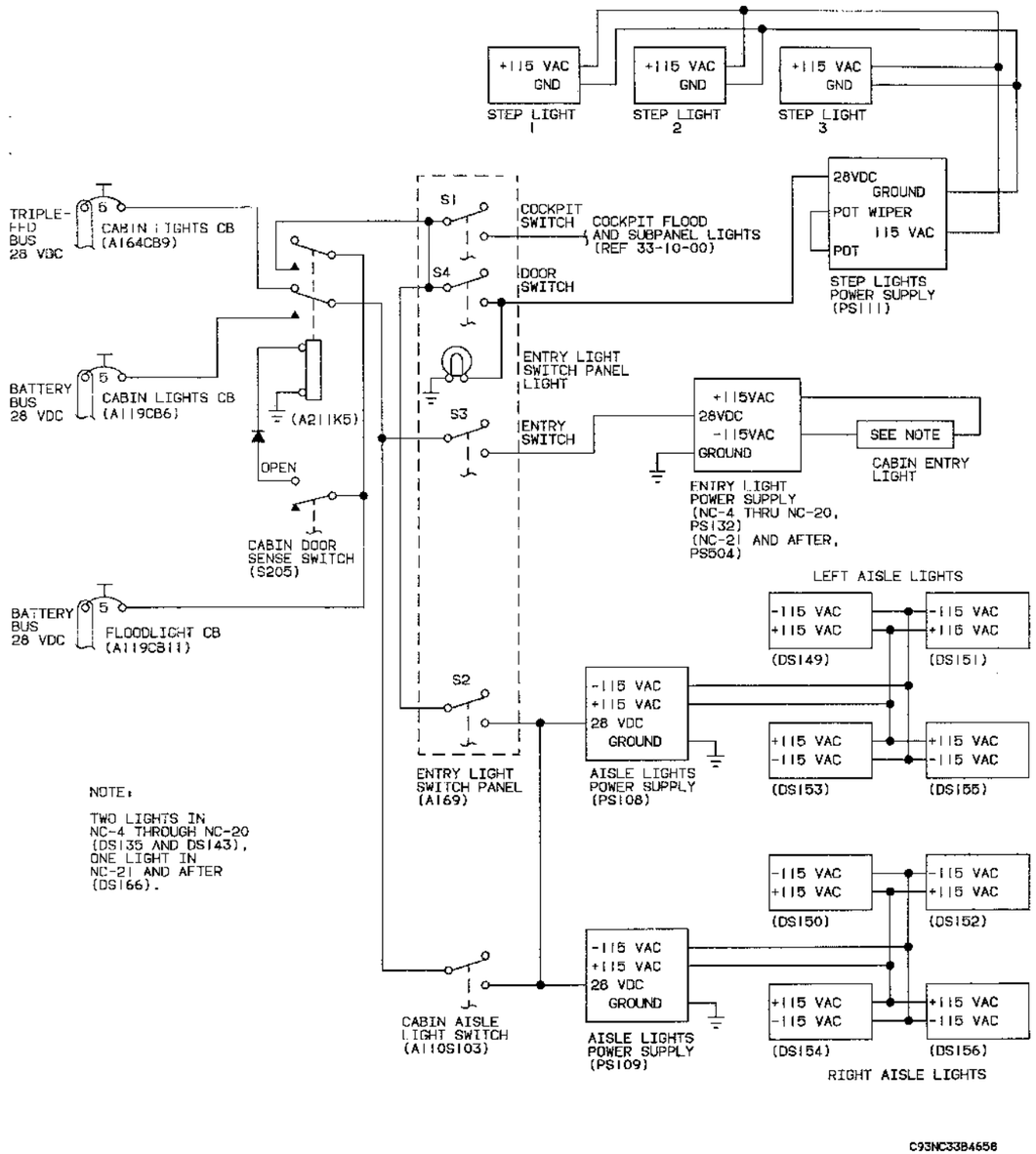
CABIN DOOR

DETAIL A

1. CABIN ENTRY LIGHT
2. NO SMOKING/FSB SIGNS AND READING LIGHT SWITCHES
3. EXIT/PULL SIGN
4. EXIT SIGN
5. AFT TOILET LIGHT
6. TOILET
7. AISLE LIGHTS (RH)
8. NO SMOKING/FSB SIGNS AND READING LIGHT SWITCHES
9. LIGHT, SWITCH AND AIR
10. ELECTRIC TOILET
11. AFT STEP ELECTRICAL EQUIPMENT PANEL
12. AFT BAGGAGE COMPARTMENT
13. STEP LIGHT POWER SUPPLY (PS111)
14. DOOR INSTRUCTION LIGHT PANEL
15. CABIN DOOR STEP LIGHTS
16. LIGHT, SWITCH AND CLOTHES HANGER
17. NO SMOKING/FSB SIGNS AND READING LIGHT SWITCHES
18. EXIT SIGN

C93NC33B4395

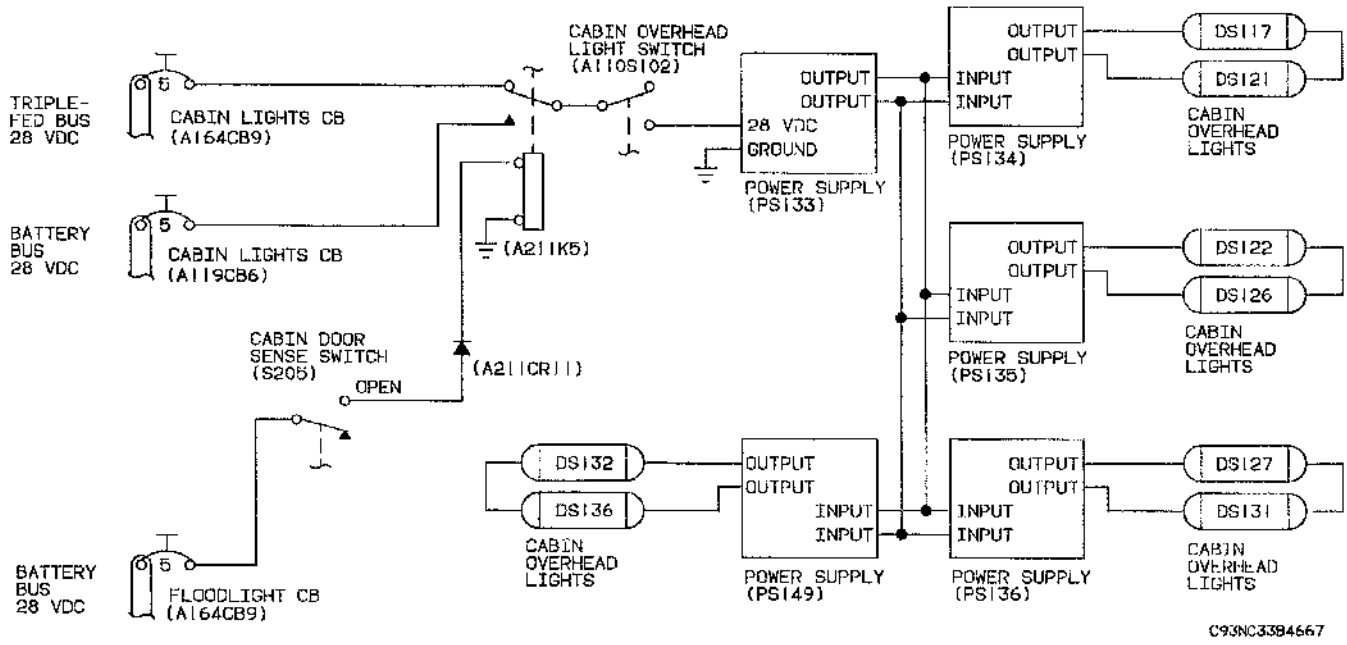
Cabin Lights (Effectivity: NC-21 and after)
Figure 2



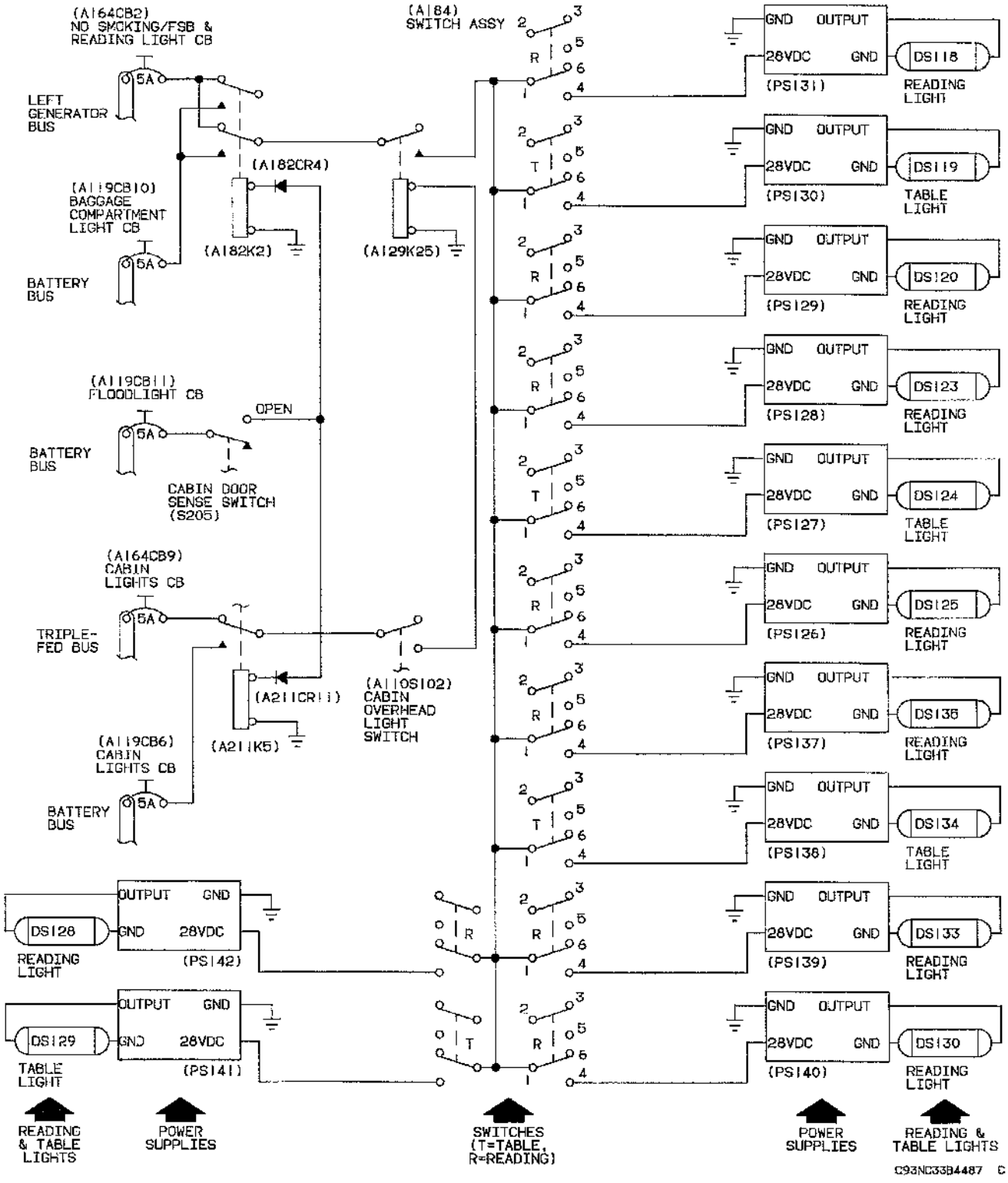
Cabin Entry, Step and Aisle Lights Schematic (Effectivity: All)
Figure 3

CABIN OVERHEAD LIGHTS (Effectivity: NC-4 through NC-20) (Figures 1, 4 and 5)

There are ten fluorescent lights located on each outboard edge of the cabin headliner (DS117 through DS126, on the RH side and DS128 through DS136 on the LH side). They are mounted in recessed reflector assemblies covered by plastic grills. When the cabin door is locked, the overhead lights are powered by the triple-fed and left generator buses. If the cabin door is unlocked, the "hot" battery bus is the voltage source. All twenty overhead lights are energized from the cockpit by the pushbutton switch (A110S102) placarded CABIN OHEAD on the overhead panel, but only eight of these (all shown in Figure 4) are controlled solely from the cockpit. These eight are positioned two forward, two aft, and four in the center of the cabin. The twelve remaining overhead lights (shown in Figure 5) are classified as either reading or table lights. There is a table light (DS119, DS124, DS129 and DS134) above each table for a total of four, and a reading light (DS118, DS120, DS123, DS125, DS128, DS130, DS133 and DS135) above each seat for a total of eight. There is a reading light switch on each console adjacent to the seat it serves. Each table light switch is mounted on the table itself, and activates as the table is unstowed.



Cabin Overhead Lights Schematic (Effectivity: NC-4 through NC-20)
Figure 4



Cabin Reading and Table Lights Schematic (Effectivity: NC-4 through NC-20)
Figure 5

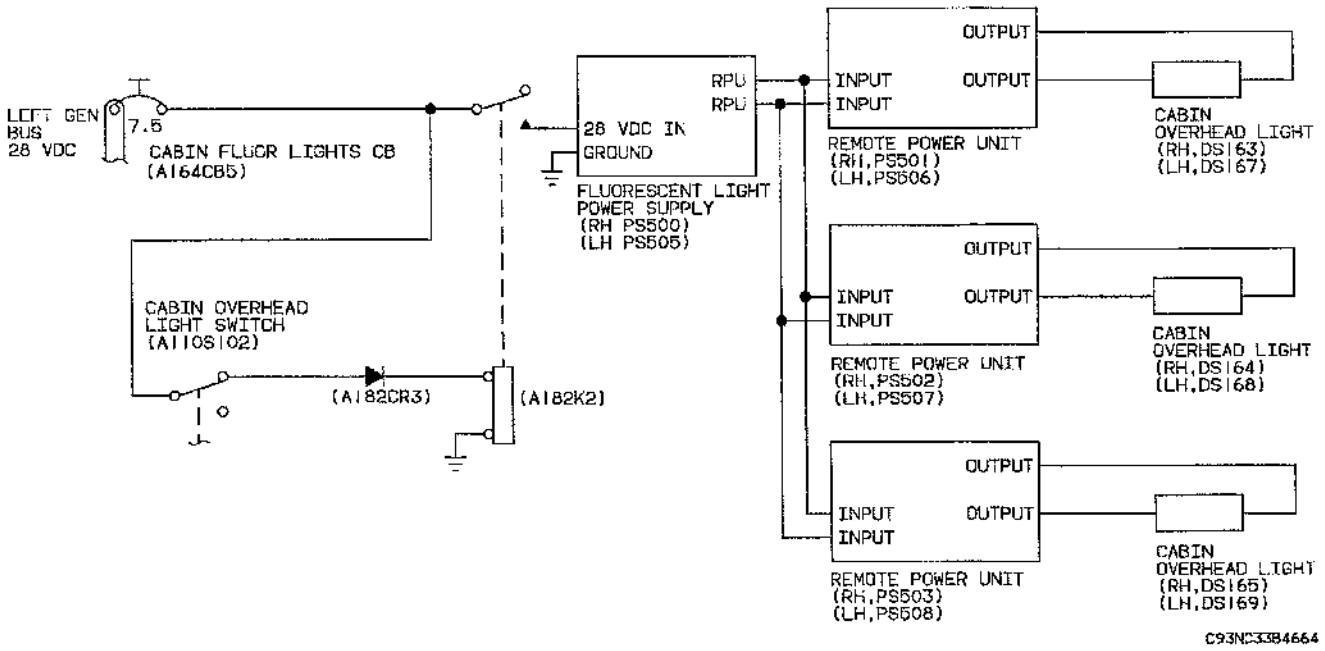
CABIN OVERHEAD LIGHTS (Effectivity: NC-21 and after) (Figures 2, 6 and 7)

There are seven fluorescent lights located on each outboard edge of the cabin headliner for a total of fourteen. They are mounted in recessed reflector assemblies protected by plastic grills. The overhead cabin lights cannot be powered from the battery. Six of these fluorescent lights provide bright cabin illumination, as shown in Figure 6. The push-button switch (A110S102) placarded CABIN OHEAD energizes the cabin overhead light relay (A182K2). Twenty eight vdc from the left generator bus through the contacts of this relay powers the two inverters (power supplies PS500 and PS505) and the six remote power units (PS501, PS502, PS503, PS506, PS507 and PS508) illuminating the six cabin overhead lights (DS163, DS164, DS165, DS167, DS168 and DS169).

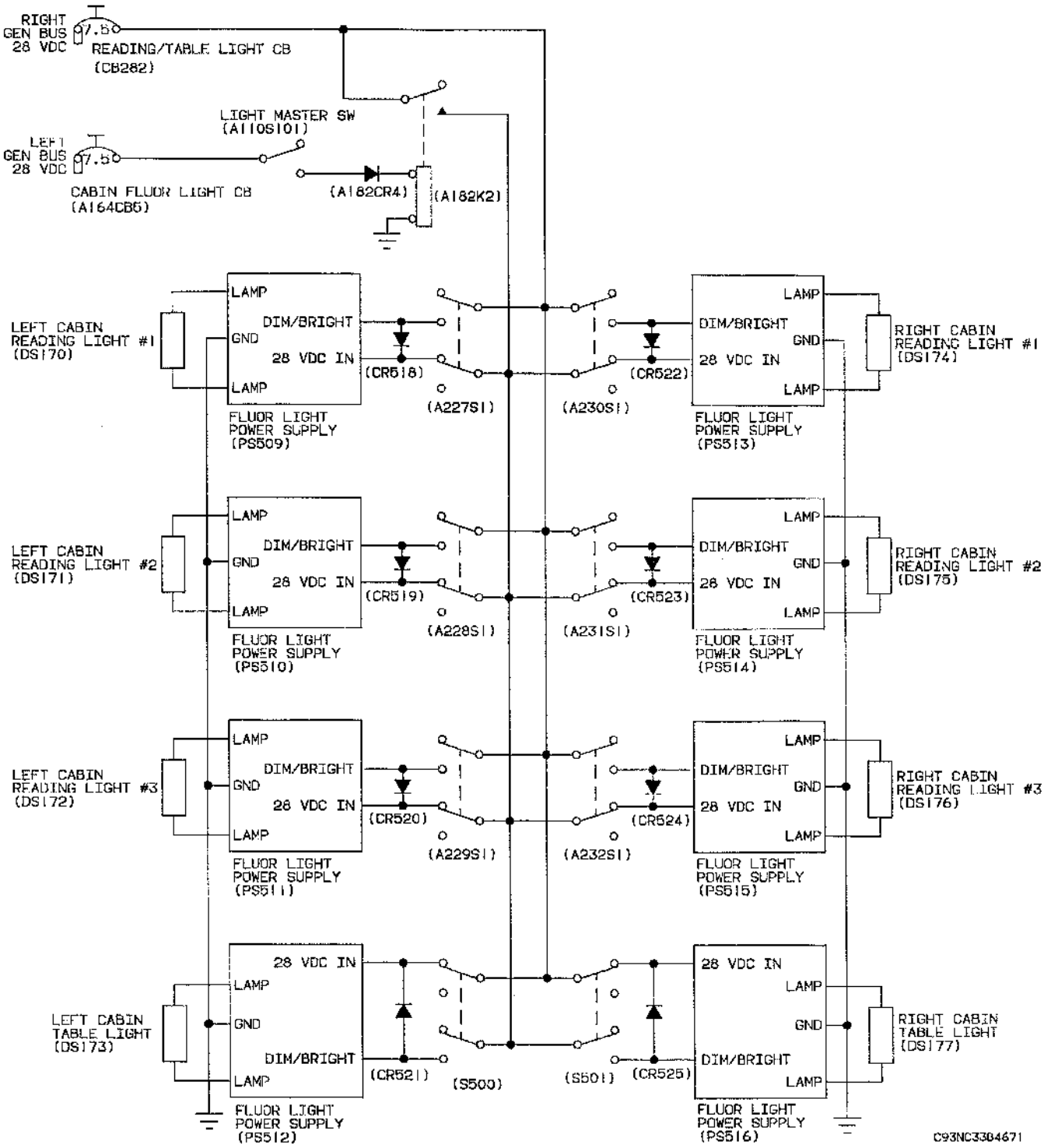
Reading lights (DS170, DS171, DS172, DS174, DS175 and DS176) and table lights (DS173 and DS177) are powered from the right generator bus, as shown in Figure 7. When the reading light switches are OFF and the tables are stowed, these lights are dimmed. This light is adequate for passenger comfort and safety during flight. When the light master switch (A110S101) on the overhead panel is ON, pushing the reading light switch or unstowing the table will turn ON the appropriate light, at full intensity. Each reading light switch is located in the console adjacent to the seat it serves. The table light switches are on the table assemblies and are activated when the table is unstowed.

WARNING AND INSTRUCTION LIGHTED SIGNS (Effectivity: NC-4 through NC-20) (Figures 1 and 8)

There are nine no-smoking, one return-to-seat and eight fasten-seatbelt signs in the cabin. Eight of the no-smoking and all eight fasten-seatbelt signs are mounted in pairs (one each) in each console forward of the reading light and window shading switches. The ninth NO SMOKING sign and the RETURN TO SEAT sign are mounted at the top of the RH forward baggage compartment. The two signs face inward and are visible only when the forward baggage compartment



Cabin Overhead Lights Schematic (Effectivity: NC-21 and after)
Figure 6

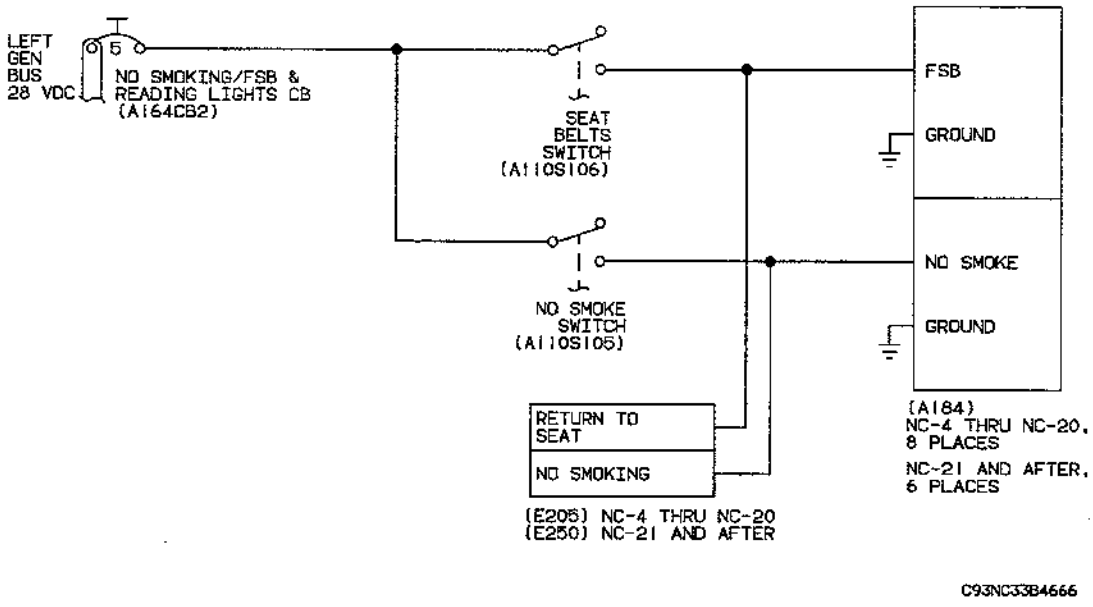


Cabin Reading and Table Lights Schematic (Effectivity: NC-21 and after)
Figure 7

doors are open. The push-button switch (A110S105) placarded NO SMOKE on the overhead panel in the cockpit illuminates all nine no-smoking signs, and the push-button switch (A110S106) placarded SEAT-BELTS on the overhead panel in the cockpit illuminates the RETURN TO SEAT and all eight fasten-seatbelt signs. All of the warning lights are powered by the left generator bus through the no smoking/FSB and reading light circuit breaker (A164CB2).

WARNING AND INSTRUCTION LIGHTED SIGNS (Effectivity: NC-21 and after) (Figures 2 and 8)

There are seven no-smoking, one return-to-seat and six fasten-seatbelt signs in the cabin. Six of the no-smoking and all six fasten-seatbelt signs are mounted in pairs (one each NO SMOKING and FSB) above the reading light switches, in the console adjacent to each seat. The seventh NO SMOKING sign and the RETURN TO SEAT sign are mounted at the top of the RH forward partition in the aft toilet compartment. The two signs are visible to a passenger inside the aft toilet compartment. The pushbutton switch (A110S105) placarded NO SMOKE on the overhead panel in the cockpit illuminates all seven no-smoking signs, and the push-button switch (A110S106) placarded SEATBELTS on the overhead panel in the cockpit illuminates the RETURN TO SEAT and all six fasten-seatbelt signs. All of the warning lights are powered by the left generator bus through the no smoking/FSB and reading light circuit breaker (A164CB2).



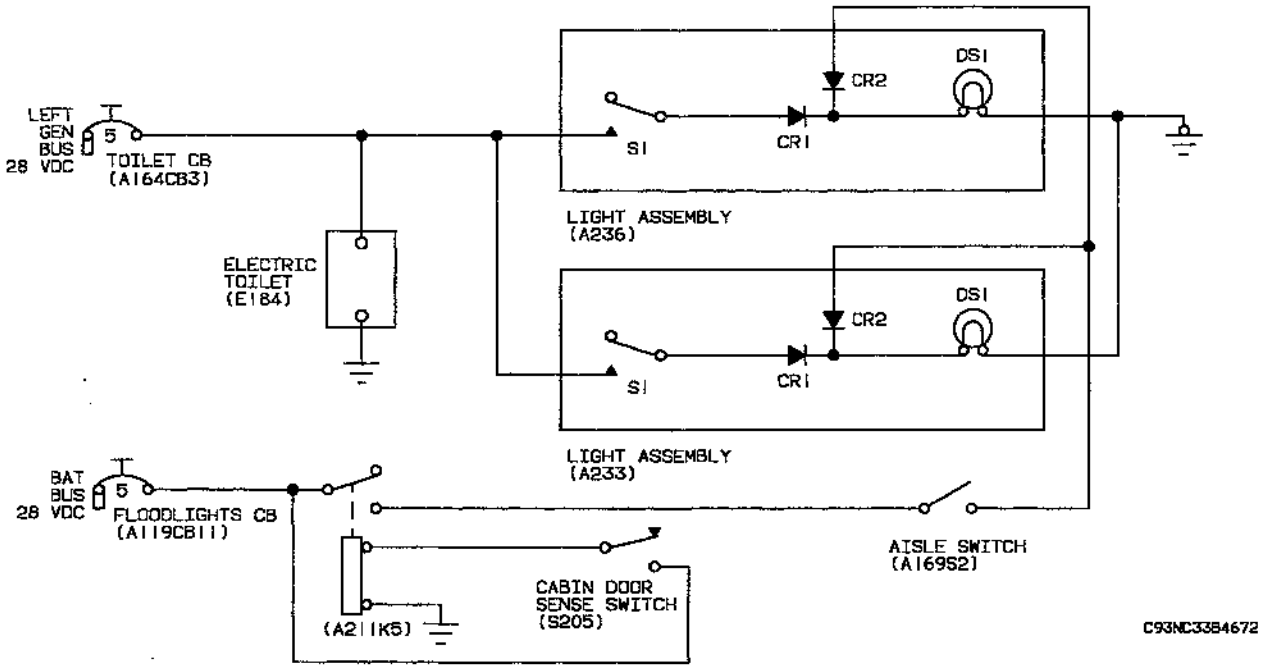
No Smoking/FSB Lights Schematic (Effectivity: All)
Figure 8

AFT TOILET COMPARTMENT LIGHTS (Effectivity: NC-21 and after) (Figures 2 and 9)

Two light fixtures (A233 and A236) near the outboard edges of the toilet compartment headliner provide air, illumination and a hook for hanging clothes hangers. The lights are turned ON or OFF by a push-button switch (S1) in the center of each light assembly. Power from the left generator bus is supplied to both light assemblies (A233 and A236) and to the toilet (E184), through the toilet circuit breaker (A164CB3). For servicing, the lights (only) can receive power from the "hot" battery bus through the floodlights circuit breaker (A119CB11), when the cabin door sense switch (S205) is in the OPEN position (closed) and the aisle lights switch (A169S2) is in the ON position.

SELF-LUMINOUS SIGNS (Effectivity: All) (Figures 1 and 2)

An EXIT sign is mounted in the vestibule headliner directly above the cabin door. Another EXIT sign is mounted above the emergency exit door. A sign placarded EXIT-PULL is mounted on the emergency exit door handle, and a cabin door instruction sign is mounted adjacent to the cabin door handle. All of these signs are self-luminous. FAA regulations require a minimum operational brightness of 160 microlamberts. Periodic brightness checks must be conducted to ensure that the brightness level remains at or above 160 microlamberts. Refer to Chapter 4 for inspection interval information.



Aft Toilet Compartment Lights Schematic (Effectivity: NC-21 and after)
Figure 9

CABIN LIGHTS - TROUBLESHOOTING (Effectivity: All)

Most troubleshooting of the lighting system will consist of lamp replacement. If a substitution does not solve the problem, a check of all connections in the wiring to the lamp holder should be made. A loose, corroded, or damaged pin connection is a common cause of lighting failure.

When troubleshooting electroluminescent panels note the following:

a. A short in one electroluminescent panel or a faulty power supply will prevent all of the electroluminescent panels in that circuit from illuminating.

b. Disconnect the output connector to the power supply to check power supply output. Use a potentiometer to vary output of the power supply.

c. If the power supply output is good, check each individual electroluminescent panel by isolating it from the other panels in the circuit and applying the correct operating voltage.

d. Electroluminescent panels cannot be repaired; therefore, replacement is the only option if a panel fails to illuminate.

CABIN LIGHTS - MAINTENANCE PRACTICES (Effectivity: All)

NOTE

Ensure that the proper replacement lamp is used. The lamp number can usually be obtained from the base of the lamp. If not, reference the LAMP REPLACEMENT GUIDE in this chapter. Visually inspect the new lamp for a properly formed glass envelope. If it is distorted or misaligned with the lamp base, discard the lamp.

CHART 1
LAMP REPLACEMENT GUIDE CABIN LIGHTS
(Effectivity: NC-4 through NC-20)

Location	Lamp #
Overhead Lights	6635-380.2S
Entry Lights	6635-360.2S
Entry Light Switch Panel	MS15670-303
Return-To-Seat Sign	1202-200

CHART 1
LAMP REPLACEMENT GUIDE CABIN LIGHTS
(Effectivity: NC-4 through NC-20) *(Continued)*

Location	Lamp #
Console No-Smoking Lights	85
Console FSB Lights	85
No-Smoking Sign	1202-200
Aft Toilet Lights	2520-10

CHART 2
LAMP REPLACEMENT GUIDE CABIN LIGHTS
(Effectivity: NC-21 and after)

Location	Lamp #
RH Forward Overhead Lights (DS163)	6335-560
LH Forward Overhead Lights (DS167)	6335-510
All Other Overhead Lights	6335-430
Entry Lights	6335-750
Entry Light Switch Panel	MS15670-303
Return-To-Seat Sign	1202-200
Console No-Smoking Lights	85
Console FSB Lights	85
No-Smoking Sign	1202-200
Aft Toilet Lights	2520-10

EXIT SIGN REPLACEMENT (Effectivity: All)

NOTE

The date of manufacture is stamped on each exit sign. This date is useful in determining the remaining life of each sign.

a. Remove the mounting screws and remove the sign.

b. Align the new sign with the mounting holes.

c. Install the mounting screws.

NO-SMOKING/RETURN-TO-SEAT-SIGN LAMP TRAY REPLACEMENT (Effectivity: All)

a. Pull the no smoking/FSB and reading lights circuit breaker.

b. Snap the bezel off the sign.

c. Remove the two screws retaining the lamp tray and remove the tray.

d. Align a new lamp tray with the edge and install the two screws.

e. Snap the bezel back onto the sign.

f. Close the no smoking/FSB and reading lights circuit breaker.

NO-SMOKING AND FASTEN-SEAT-BELT-SIGN LAMP REPLACEMENT (CONSOLE) (Effectivity: All) (Figure 10)

a. Pull the no smoking/FSB and reading lights circuit breaker.

b. Snap the indicator cap (3) off the lamp holder (1).

c. Remove the faulty lamp (4) and install a new one.

d. Snap the indicator cap (3) back onto the lamp holder (1).

e. Close the no-smoking/FSB and reading lights circuit breaker.

ENTRY LIGHT SWITCH PANEL LAMP REPLACEMENT (Effectivity: All) (Figure 11)

a. Pull the floodlight circuit breaker and both cabin lights circuit breakers.

b. Locate the actuator access hole (6) in the door entry light switch panel lens (7). Use a small pointed object to press the actuator and release the lens.

c. Remove the lens (7).

d. Reach inside the panel (3), push in on the lamp (8), and rotate it counterclockwise until the bayonets release.

e. Remove the lamp (8), then install a new one by pressing inward and rotating clockwise to engage the bayonets.

f. Snap the lens (7) back onto the panel face (3).

g. Close the floodlight circuit breaker and both cabin lights circuit breakers.

CABIN DOOR STEP LIGHT PANEL REPLACEMENT (Effectivity: All) (Figure 12)

a. Pull the cabin floodlight circuit breaker and both cabin lights circuit breakers.

b. Remove the three screws (6), washers (8), and nuts (9) securing the retainer (5) and step light panel (4) to the door structure (7).

c. Remove the retainer (5) and pull the panel (4) out far enough to disconnect the electrical connector.

d. Connect the electrical connector to the new step light panel (4).

e. Align the step light panel (4) and retainer (5) with the nutplates on the cabin door structure (7).

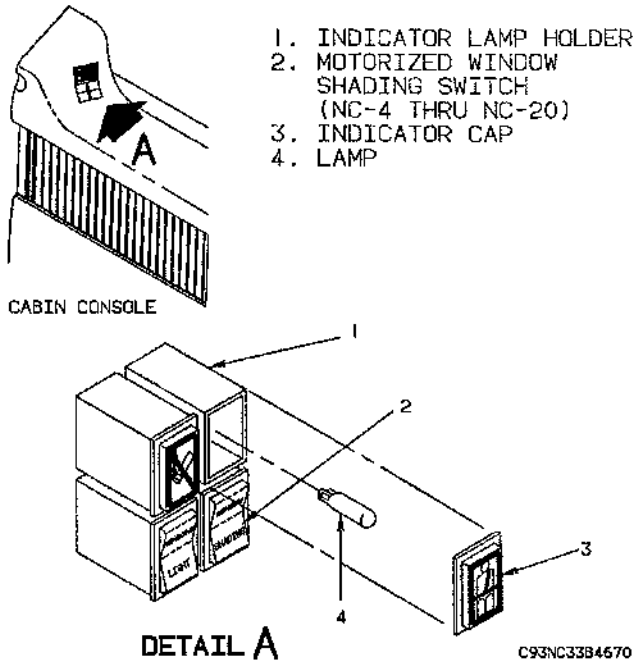
f. Install the three mounting screws (6), washers (8) and nuts (9).

g. Close the floodlight circuit breaker and both cabin lights circuit breakers.

CABIN DOOR INSTRUCTION LIGHT PANEL REPLACEMENT (Effectivity: All) (Figure 12)

NOTE

A manufacture date is stamped on the door instruction light panel. This date is useful in determining the remaining life of the light panel.



No-Smoking and Fasten-Seat-Belt-Sign Lamp Replacement (Console) (Effectivity: All) Figure 10

a. Open the cabin door and leave the handle in the OPEN position.

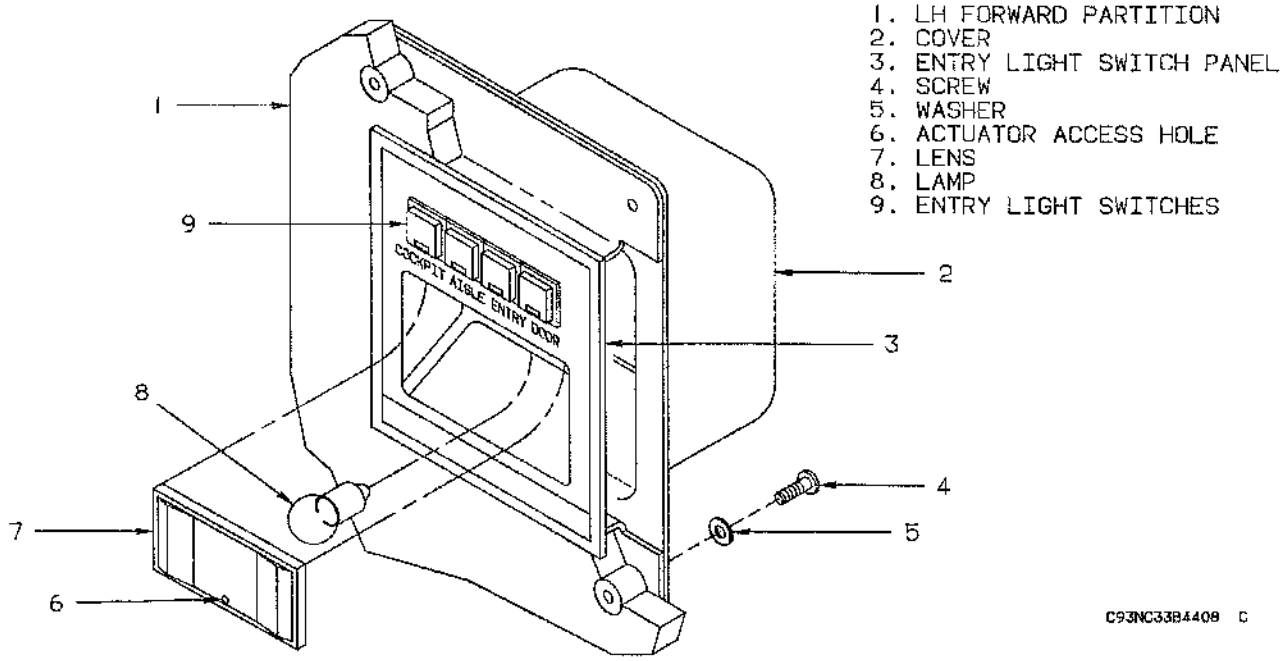
b. Remove the four mounting screws (3).

c. Remove the door instruction light panel (2).

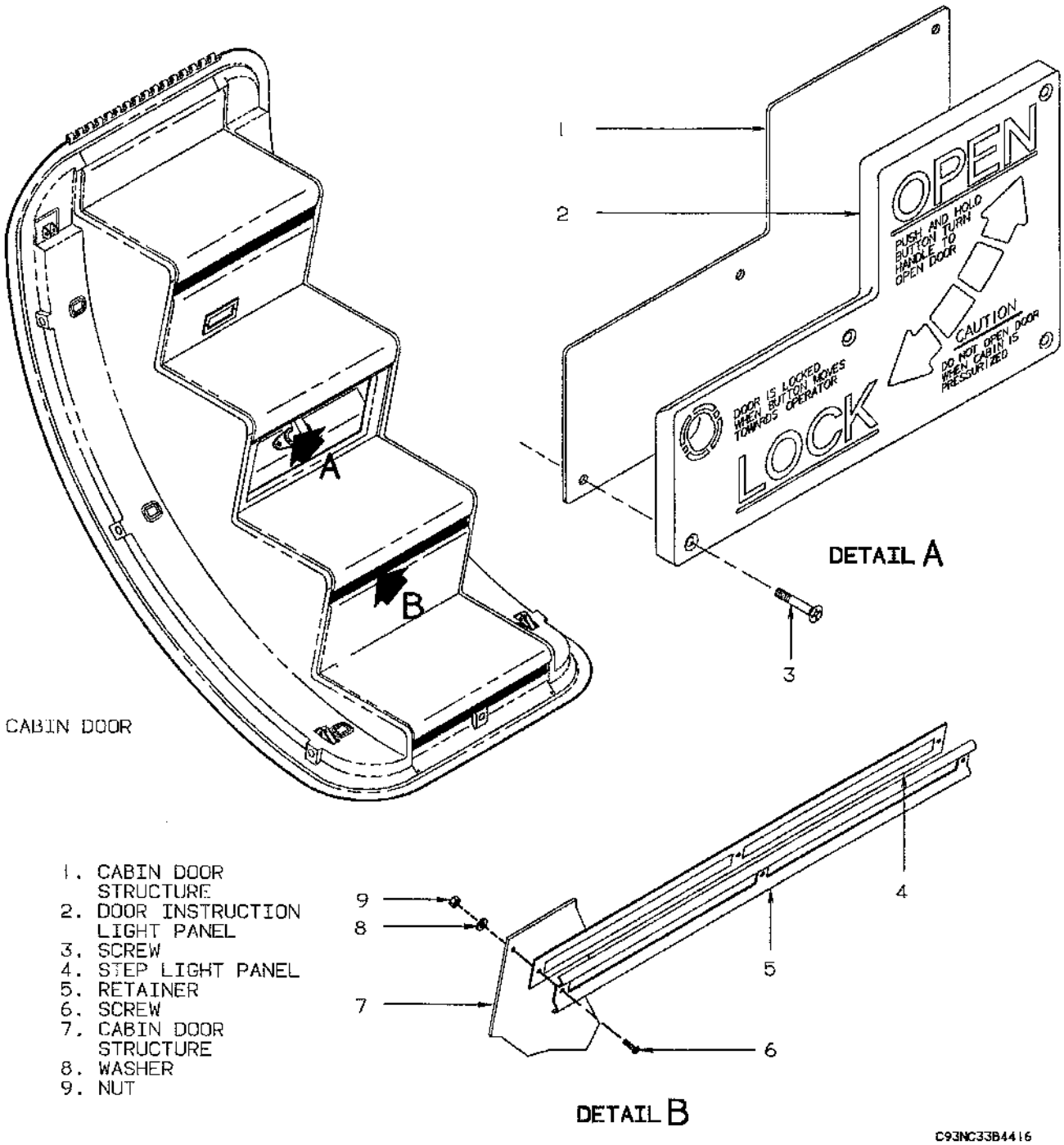
CAUTION

Use care not to damage the face of the door instruction light panel (2), as it cannot be repaired.

d. Align the new door instruction light panel (2) with the nutplates on the cabin door structure (1) and install the four mounting screws (3).



Entry Light Switch Panel Lamp Replacement (Effectivity: All)
Figure 11



1. CABIN DOOR STRUCTURE
2. DOOR INSTRUCTION LIGHT PANEL
3. SCREW
4. STEP LIGHT PANEL
5. RETAINER
6. SCREW
7. CABIN DOOR STRUCTURE
8. WASHER
9. NUT

Cabin Door Step Light and Instruction Panel Replacement
(Effectivity: All)
Figure 12

AISLE LIGHT PANEL REPLACEMENT (Effectivity: All) (Figure 13)

a. Pull the floodlight circuit breaker and both cabin lights circuit breakers.

b. Remove the mounting screws (2) from the faulty aisle aisle light panel (4).

c. Pull the aisle light panel (4) away from the keel (1) and disconnect the electrical connector (3).

d. Connect the electrical connector (3) to the new aisle light panel (4).

e. Align the new aisle light panel (4) with the inserts in the keel (1) and install the eight mounting screws (2).

f. Close the floodlight circuit breaker and both cabin lights circuit breakers.

ENTRY LIGHTS AND OVERHEAD LIGHTS LAMP REPLACEMENT (Effectivity: All) (Figure 14)

a. Remove all electrical power from the airplane.

b. To gain access to the lamp, proceed as follows:

1. On enrty lights, squeeze the sides of the plastic lamp cover (6) until one edge clears the vestibule headliner (7), then lower the lamp cover until it is removed.

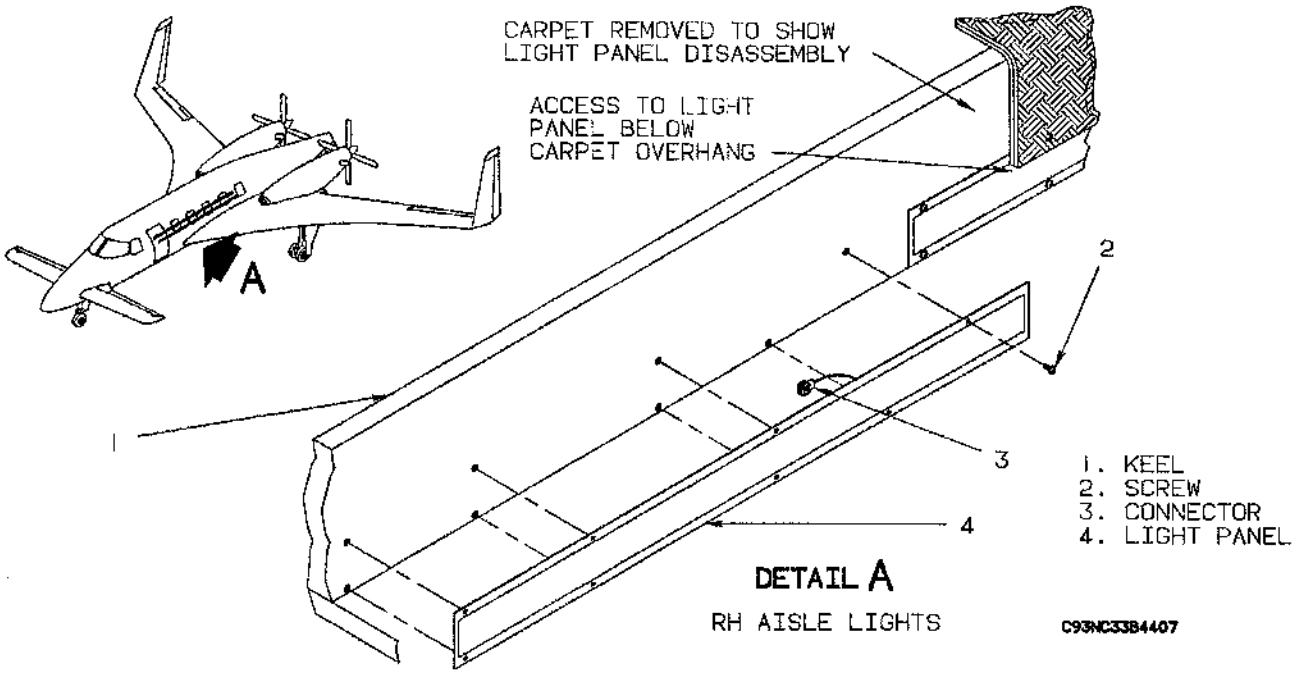
2. On overhead lights, remove the six screws (12) from the plastic grill (11) over the faulty lamp (10).

c. Refer to Detail A and proceed as follows:

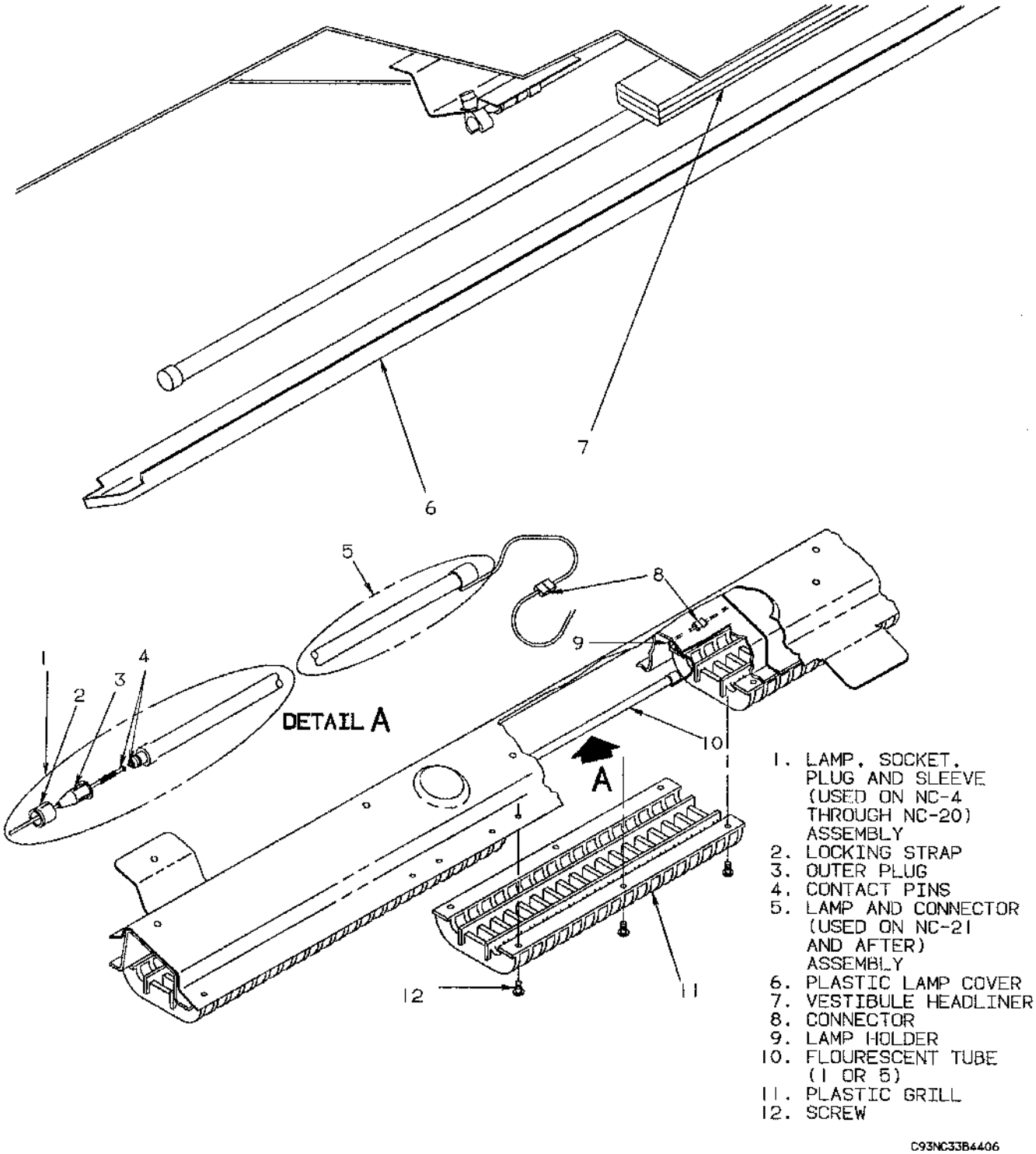
1. On NC-4 through NC-20, pull the locking strap (2) from each outer plug (3) on the ends of the lamp (1) to disengage the contact pins (4).

2. On NC-21 and after, disconnect the connector (8) from each end of the lamp (5).

d. Grasp the lamp (1 or 5) at both ends and remove it from the holders (9).



Aisle Light Panel Replacement (Effectivity: All)
Figure 13



Entry Lights and Overhead Lights Fluorescent Lamp Replacement (Effectivity: All)
Figure 14

e. Install the new lamp (1 or 5) in the holders (9).

f. Refer to Detail A and proceed as follows:

1. On NC-4 through NC-20, slide an outer plug (3) over each end of the new lamp (1) until both sets of contact pins (4) are fully engaged, then slide the locking strap (2) over each outer plug to secure them.

2. On NC-21 and after, attach the connectors (8) to the wiring harness connectors.

g. To complete the lamp replacement, proceed as follows:

1. On entry lights, squeeze the plastic lamp cover (6) at the sides and insert it in the vestibule headliner (7).

2. On overhead lights, align the plastic grill (11) mounting holes with the reflector assembly nutplates and install the six screws (12).

h. Restore electrical power to the airplane.

OVERHEAD LIGHTS POWER SUPPLY REMOVAL (AFT) (Effectivity: NC-4 through NC-20) (Figure 15)

a. Remove all electrical power from the airplane.

b. Fold down the back of the aft couch and open the aft baggage compartment door (the door is hinged at the bottom and middle, and opens forward).

c. Pull the upholstery panels free from the Velcro fasteners on the side and top of the aft baggage compartment.

d. Remove the screws, washers, and spacers securing the air return cover to the aft false bulkhead.

e. Remove the screws and washers from around the air return opening and pull the upholstery panel free from the Velcro fasteners along the bottom edge of the panel.

f. Locate the desired power supply (see detail A) by designator.

g. Remove the two mounting screws (4) and washers (3) from the desired power supply (7).

h. Disconnect the electrical connector and remove the power supply (7).

OVERHEAD LIGHTS POWER SUPPLY INSTALLATION (AFT) (Effectivity: NC-4 through NC-20) (Figure 15)

CAUTION

The power supplies must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Connect the electrical connector to the new power supply (7).

b. Align the power supply (7) with the holes in the mounting plate (2).

c. Install the two mounting screws (4), washers (3), and nuts (5).

d. Align the upholstery panel with the Velcro fasteners and air return openings on the aft false bulkhead.

e. Install the screws, washers and spacers securing the air return cover to the aft false bulkhead.

f. Install the side and top upholstery panels by aligning the Velcro fasteners on the panels with those on the aft baggage compartment.

g. Fold up the hinged aft baggage compartment door and aft couch back.

h. Restore electrical power to the airplane.

OVERHEAD LIGHTS POWER SUPPLY REMOVAL (FORWARD) (Effectivity: NC-4 through NC-20) (Figure 16)

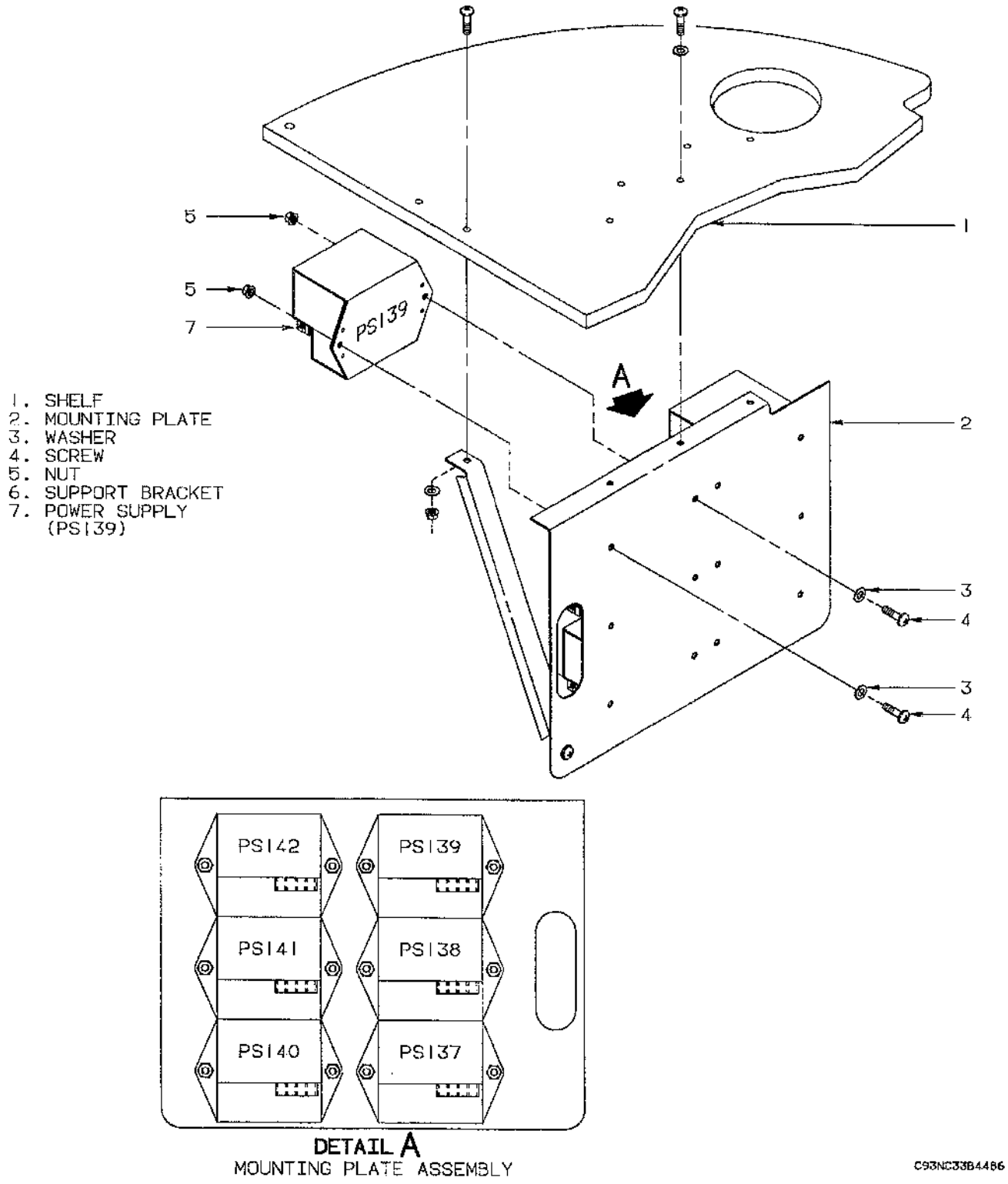
a. Remove all electrical power from the airplane.

b. Open the doors to the RH forward baggage compartment.

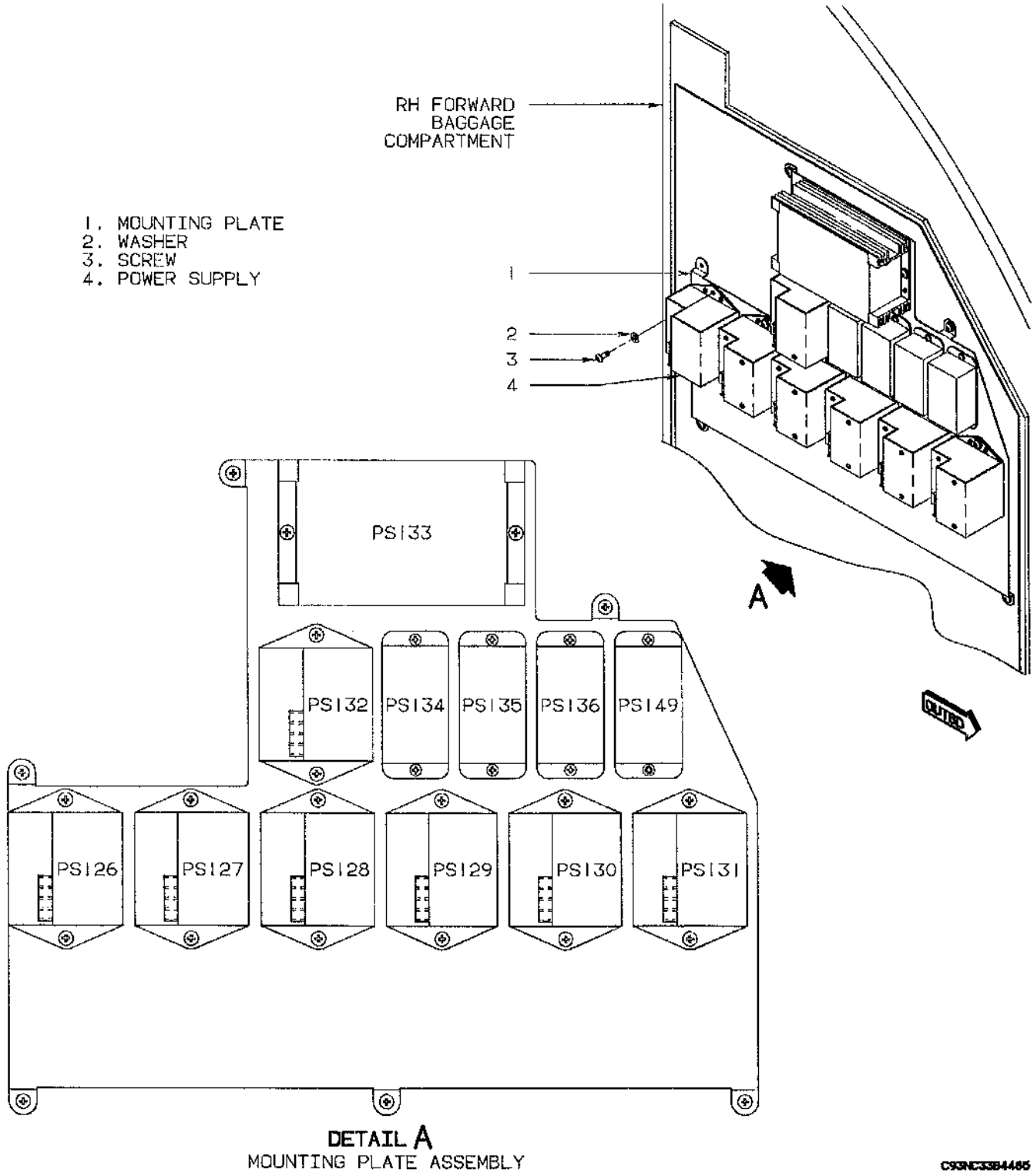
c. Reach inside the baggage compartment and remove the upper left sidewall to expose the mounting plate assembly (detail A).

d. Locate the desired power supply by designator and remove the two mounting screws (3) and washers (2).

e. Disconnect the electrical connector and remove the power supply (4) from the mounting plate (1).



Overhead Lights Power Supply Installation (Aft) (Effectivity: NC-4 through NC-20)
Figure 15



Overhead Lights Power Supply Installation (Forward) (Effectivity: NC-4 through NC-20)
Figure 16

OVERHEAD LIGHTS POWER SUPPLY INSTALLATION (FORWARD) (Effectivity: NC-4 through NC-20) (Figure 16)

CAUTION

The power supplies must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Connect the electrical connector to the power supply (4).

b. Align it with the nutplates on the mounting plate (1) and install the two mounting screws (3) and washers (2).

c. Install the upper left sidewall in the RH forward baggage compartment.

d. Close the door on the RH forward baggage compartment.

e. Restore electrical power to the airplane.

OVERHEAD LIGHTS POWER SUPPLY REMOVAL (Effectivity: NC-21 and after) (Figure 17)

a. Remove electrical power from the airplane.

b. Refer to Figure 2 of this chapter and identify the location of the appropriate power supply by use of its designator. Refer to Chapter 25-20-00 and remove the necessary side upholstery panels to gain access to the power supply.

c. Disconnect the electrical connector between the power supply and the wiring harness.

d. Refer to Detail A of Figure 17 to remove the following power supplies: PS501, PS502. PS503, PS506, PS507 and PS508.

1. Remove the two screws (3) from the bracket (2) and remove the power supply (1) from the sidewall.

e. Refer to Detail B of Figure 17 to remove the following power supplies: PS500 and PS505.

1. Remove the wiring connected to the ground stud (6).

2. Remove the two screws (8) and washers (7) from the bracket (5) and remove the power supply (4) from the sidewall.

OVERHEAD LIGHTS POWER SUPPLY INSTALLATION (Effectivity: NC-21 and after) (Figure 17)

CAUTION

The power supplies must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Refer to Figure 2 of this chapter and identify the location of the appropriate power supply by use of its designator.

b. Refer to Detail A of Figure 17 to install the following power supplies: PS501, PS502. PS503, PS506, PS507 and PS508.

1. Place the power supply (1) into position on the bracket (2) and secure with the two screws (3).

c. Refer to Detail B of Figure 17 to install the following power supplies: PS500 and PS505.

1. Place the power supply (4) into position on the bracket (5) and secure with the two screws (8) and washers (7).

2. Connect the wiring to the ground stud (6).

d. Connect the electrical connector to the power supply.

e. Check the electrical bond of the power supply (1 or 4).

f. Refer to Chapter 25-20-00 and install the upholstery panels.

g. Restore electrical power to the airplane.

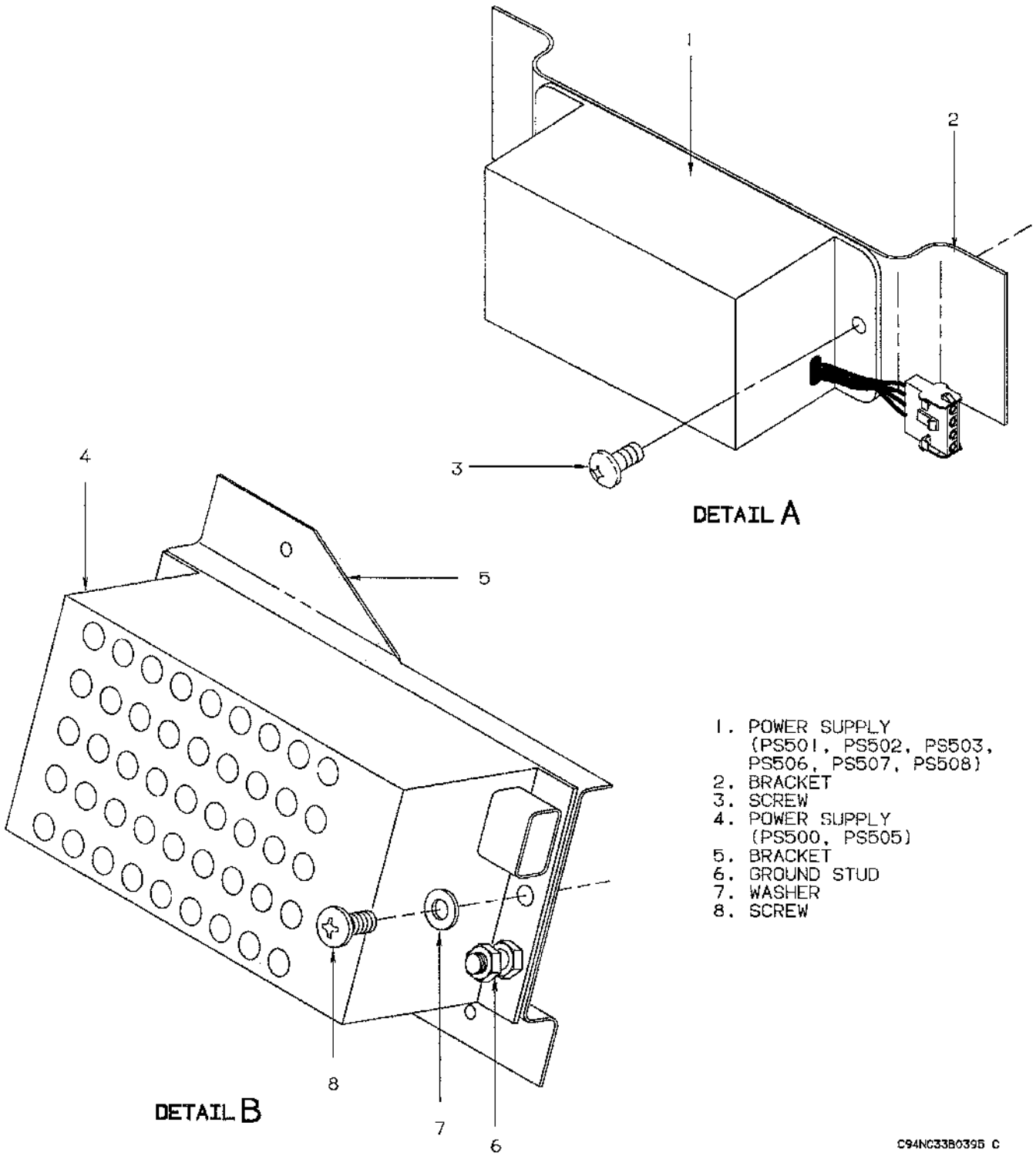
AISLE LIGHTS POWER SUPPLY REMOVAL (Effectivity: All) (Figure 18)

a. Pull the floodlight circuit breaker and both cabin lights circuit breakers.

NOTE

On airplanes NC-4 through NC-20 only, refer to Chapter 25 and remove the couch over the aft step.

b. Refer to Chapter 53-20-00 and remove the six attaching screws securing the aft step to the keel and electrical equipment panel.



Overhead Lights Power Supply Installation (Effectivity: NC-21 and after)
Figure 17

c. Remove the four screws (1), washers (3), and nuts (4) from the desired power supply (7 or 8).

d. Disconnect the electrical connector and remove the power supply (7 or 8).

AISLE LIGHTS POWER SUPPLY INSTALLATION (Effectivity: All) (Figure 18)

CAUTION

The aisle lights power supplies (7 or 8) must meet electrical bonding requirements as specified in Chapter 20-00-00.

a. Connect the electrical connector to the power supply (7 or 8).

b. Align the power supply (7 or 8) with the panel (2) mounting holes.

c. Install the four mounting screws (1), washers (3) and nuts (4).

d. Align the aft step with the electrical equipment panel (2) and keel (6) and install the six attaching screws.

NOTE

On airplanes NC-4 through NC-20 only, refer to Chapter 25 and install the couch over the aft step.

e. Close the floodlight circuit breaker and both cabin lights circuit breakers.

AFT TOILET COMPARTMENT LIGHT REPLACEMENT (Effectivity: NC-21 and after) (Figure 19)

a. Pull the toilet circuit breaker and the floodlights circuit breaker.

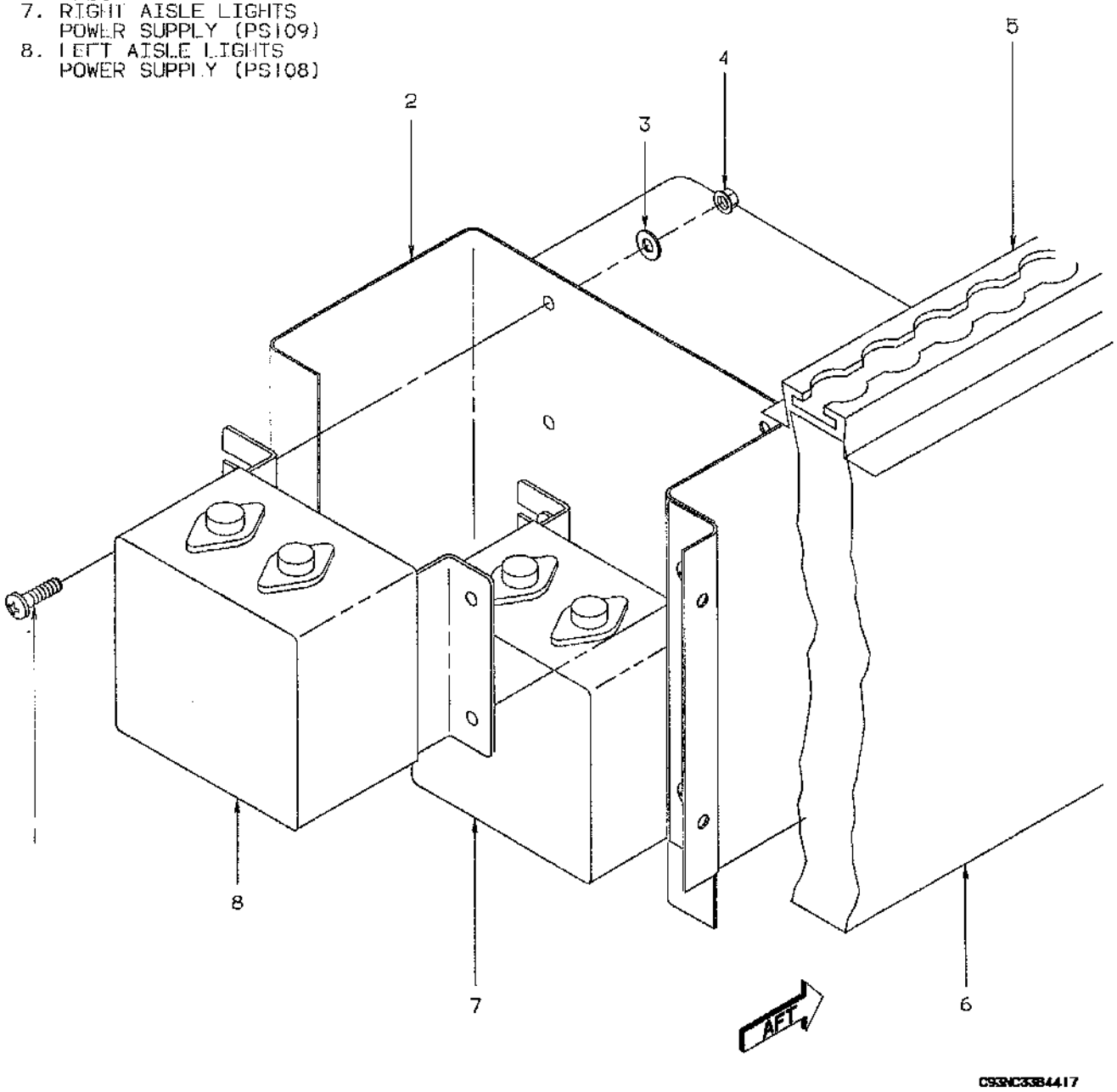
b. Pull the press-fit lens (4) out of the light fixture.

c. Remove the old lamp (5) from the lamp holder (1) and install a new one.

d. Replace the lens (4) back into the light fixture.

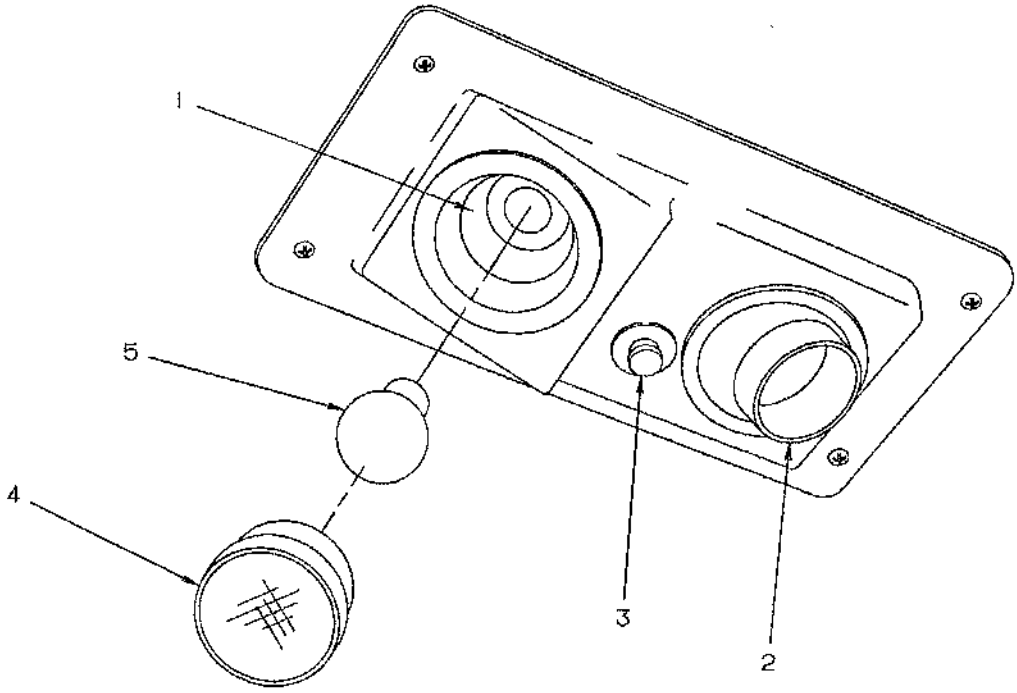
e. Close the toilet and floodlights circuit breakers.

1. SCREW
2. AFT STEP ELECTRICAL EQUIPMENT PANEL
3. WASHER
4. NUT
5. LH INBOARD SEAT TRACK (ON NC-4 THROUGH NC-20 ONLY)
6. KEEL
7. RIGHT AISLE LIGHTS POWER SUPPLY (PS109)
8. LEFT AISLE LIGHTS POWER SUPPLY (PS108)



Aisle Lights Power Supply Installation (Effectivity: All)
Figure 18

1. LAMP HOLDER
2. AIR NOZZLE
3. SWITCH
4. LENS ASSEMBLY
5. LAMP



RH TOILET COMPARTMENT
LIGHT ASSEMBLY SHOWN

Aft Toilet Compartment Light Replacement (Effectivity: NC-21 and after)
Figure 19

SELF-ILLUMINATED EXIT BRIGHTNESS CHECK (Effectivity: All) (Figure 20)

Self-illuminated exit signs can be inspected on the airplane under all types of lighting conditions without the need for removing the sign. Checking for brightness requires the use of a T-light Comparator (1, Chart 2, 33-00-00). The test is performed as follows:

Self-illuminated exit signs can be inspected on the airplane

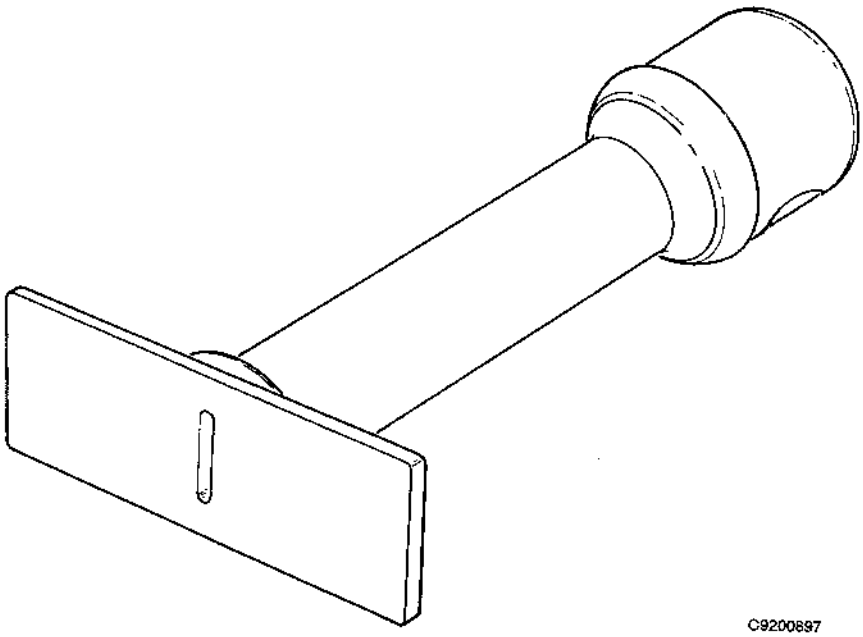
a. Compare the brightness of the sign portion seen through the aperture and the calibrated source; if the sign is as bright or brighter than the calibrated source, the sign brightness is equivalent to, or better than, the 160 microlamberts required. If the sign brightness is less than that of the comparator, the sign must be replaced.

b. Move the comparator over the illuminated portions of the sign to check minimum brightness at all points.

c. If the sign at any point appears less bright than the comparator, the sign must be replaced.

NOTE

T-light Comparators used for checking illuminated sign brightness must have had a new calibrated reference source installed within the preceding 12 months for the brightness test to be considered valid.



T-light Comparator (Effectivity: All)
Figure 20

BAGGAGE COMPARTMENT LIGHTS - DESCRIPTION AND OPERATION (Effectivity: All) (Figures 1 and 2)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

Three light assemblies illuminate the baggage compartments. Two lights (DS133 and DS142) are mounted in the aft baggage compartment headliner; another light (DS134) is mounted in the RH forward baggage compartment.

The aft baggage compartment lights are powered by the "hot" battery bus through the baggage compartment light circuit breaker. The light switch is located on RH side upholstery panel of the aft baggage compartment.

The forward baggage compartment light is mounted in the top forward area of the compartment. It is also powered by the "hot" battery bus through the baggage compartment lights circuit breaker (A119CB10). The light switch is located at the top of the compartment behind the forward door. The light switch contains a light that illuminates whenever the switch is ON.

BAGGAGE COMPARTMENT LIGHTS - TROUBLESHOOTING (Effectivity: All)

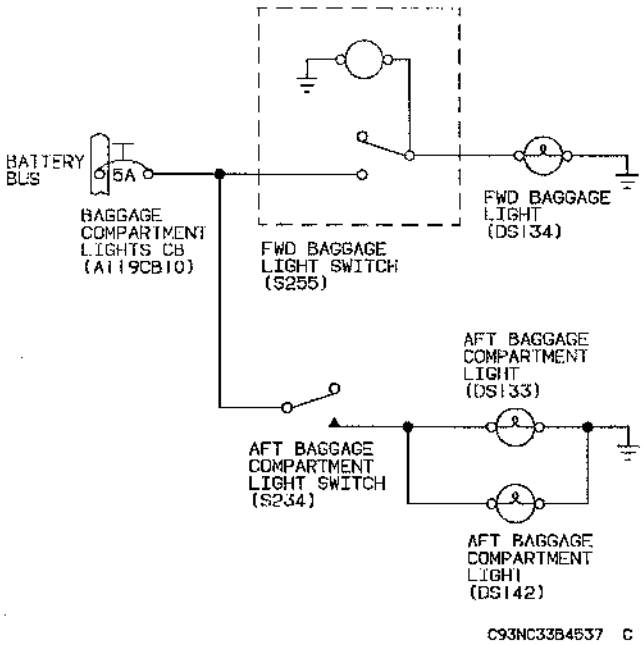
Most troubleshooting of the lighting system will consist of lamp replacement. If a substitution does not solve the problem, a check of all connections in the wiring to the lamp holder should be made. A loose, corroded, or damaged pin connection is a common cause of lighting failure.

BAGGAGE COMPARTMENT LIGHTS - MAINTENANCE PRACTICES (Effectivity: All)

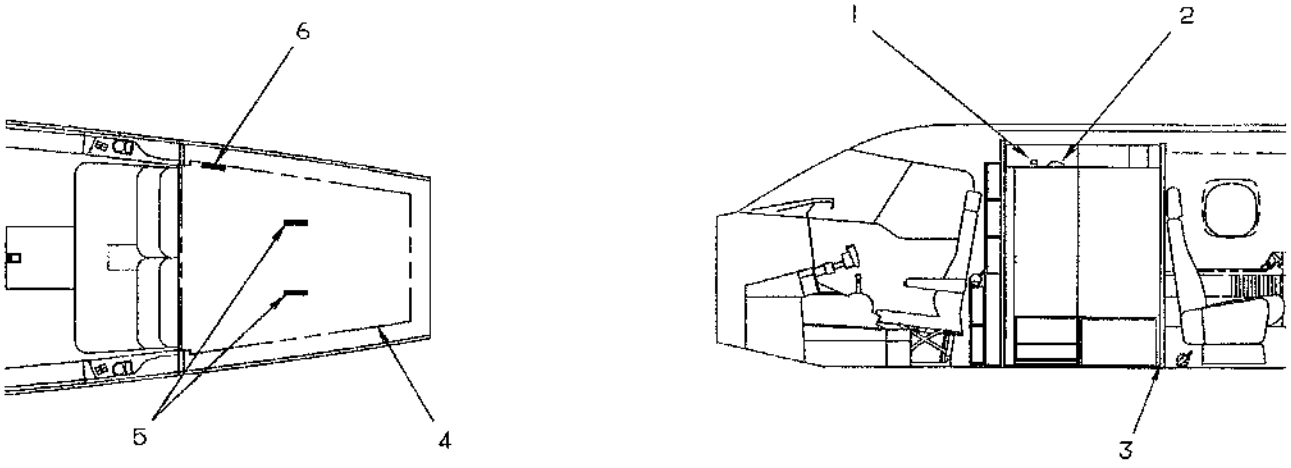
Ensure that the proper replacement lamp is used. The lamp number can usually be obtained from the base of the lamp. If not, reference the LAMP REPLACEMENT GUIDE in this chapter. Visually inspect the new lamp for a properly formed glass envelope. If it is distorted or misaligned with the lamp base, discard the lamp.

CHART 1
LAMP REPLACEMENT GUIDE
(Effectivity: All)

Location	Lamp #
Aft Baggage Compt. Light	303
Fwd. Baggage Compt. Light	305



Baggage Compartment Lights Schematic (Effectivity: All) Figure 1

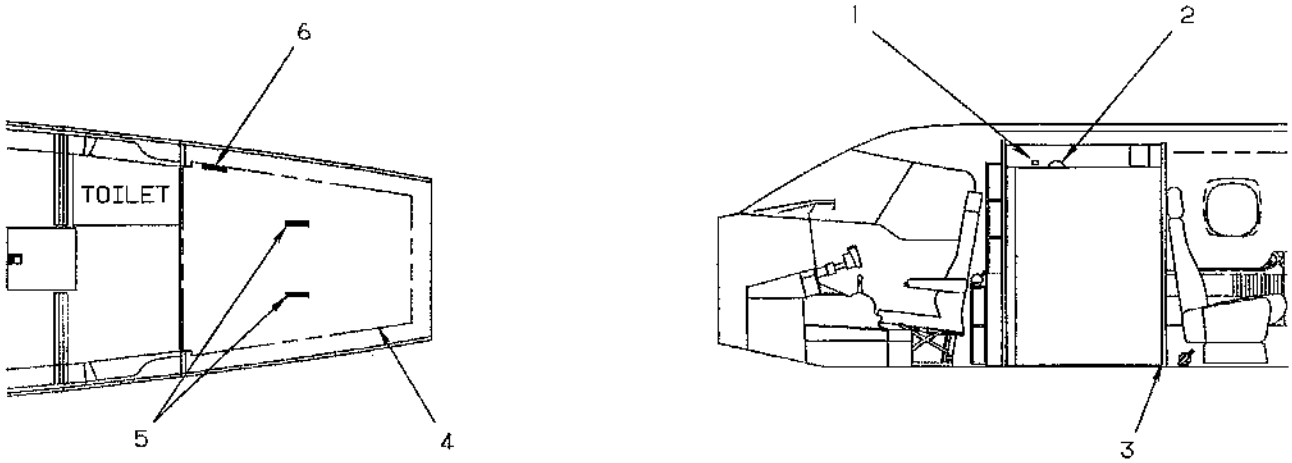


AFT CABIN
TOP VIEW

NC-4 THROUGH NC-20

FORWARD CABIN
RH SIDE VIEW

1. FORWARD BAGGAGE COMPARTMENT LIGHT SWITCH
2. FORWARD BAGGAGE COMPARTMENT LIGHT
3. FORWARD BAGGAGE COMPARTMENT (DOORS OPEN)
4. AFT BAGGAGE COMPARTMENT
5. AFT BAGGAGE COMPARTMENT LIGHTS
6. AFT BAGGAGE COMPARTMENT LIGHT SWITCH



AFT CABIN
TOP VIEW

NC-21 AND AFTER

FORWARD CABIN
RH SIDE VIEW

C93NC33B4403

Baggage Compartment Lights (Effectivity: All)
Figure 2

AFT BAGGAGE COMPARTMENT LIGHT LAMP REPLACEMENT (Effectivity: NC-4 through NC-20)
(Figure 3)

a. Pull the baggage compartment lights circuit breaker.

b. Fold down the back of the aft couch.

c. Fold down the hinged door of the aft baggage compartment.

d. Remove the two lens attaching screws (5) and remove the lens (6).

e. Remove the old lamp (4) and install a new one.

f. Align the lens (6) with the light assembly (3).

g. Install the two lens attaching screws (5).

h. Lift, unfold and close the hinged door to the aft baggage compartment.

i. Lift the back of the aft couch to the upright position.

j. Close the baggage compartment light circuit breaker.

AFT BAGGAGE COMPARTMENT LIGHT LAMP REPLACEMENT (Effectivity: NC-21 and after)
(Figure 3)

a. Pull the baggage compartment lights circuit breaker.

b. Unlatch and open the aft baggage compartment door.

c. Remove the two lens attaching screws (5) and remove the lens (6).

d. Remove the old lamp (4) and install a new one.

e. Align the lens (6) with the light assembly (3).

f. Install the two lens attaching screws (5).

g. Close and latch the aft baggage compartment door.

h. Close the baggage compartment lights circuit breaker.

FORWARD BAGGAGE COMPARTMENT LIGHT LAMP REPLACEMENT
(Effectivity: All)

a. Pull the baggage compartment lights circuit breaker.

b. Open the forward baggage compartment doors.

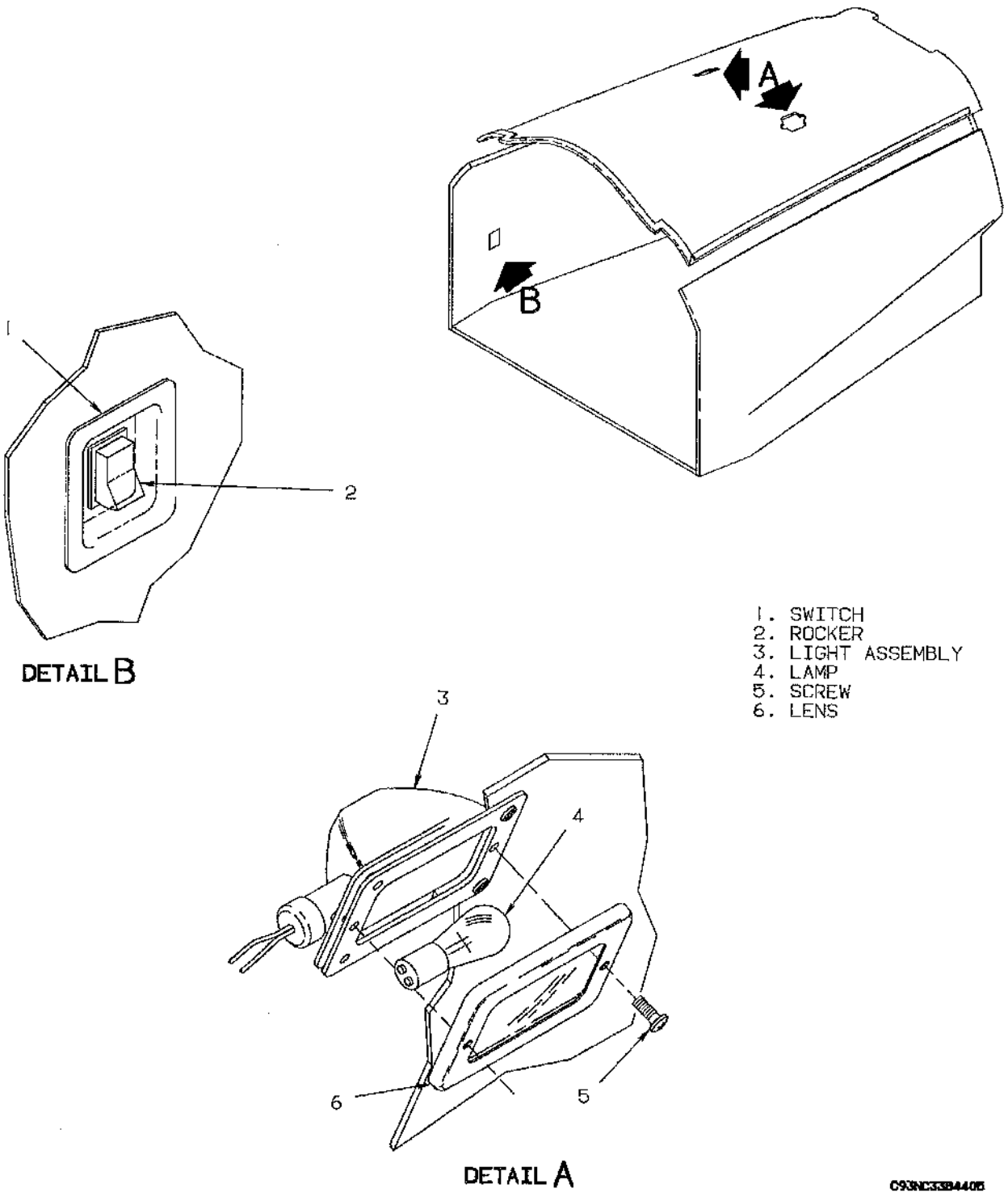
c. Rotate the lens retainer to the right and pull the lens off the light.

d. Remove the faulty lamp and install a new one.

e. Align the lens with the light assembly and rotate the lens retainer to the left to secure the lens in place.

f. Close the forward baggage compartment doors.

g. Close the baggage compartment light circuit breaker.



Aft Baggage Compartment Light Replacement (Effectivity: All)
Figure 3

EXTERIOR LIGHTS - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The Starship 1 exterior lights consist of wing inspection lights, a nose taxi light, nose and wing landing lights, wing and tail navigation lights, wing recognition lights and strobe lights. All of the exterior lights are controlled from the cockpit by the bottom row of push-button switches on the overhead panel. The switches are double-pole single-throw, and have an LED which illuminates when the switch is in the ON position.

NOSE LANDING AND TAXI LIGHTS (Effectivity: All) (Figures 1 and 2)

The nose landing and taxi lights are mounted with brackets to the nose landing gear brace for illumination of the runway during landing or taxiing. Both light assemblies are identical with the exception of the sealed beam lamp ratings. These lights are powered only when the nose landing gear is down and locked. The taxi light is mounted on the right side.

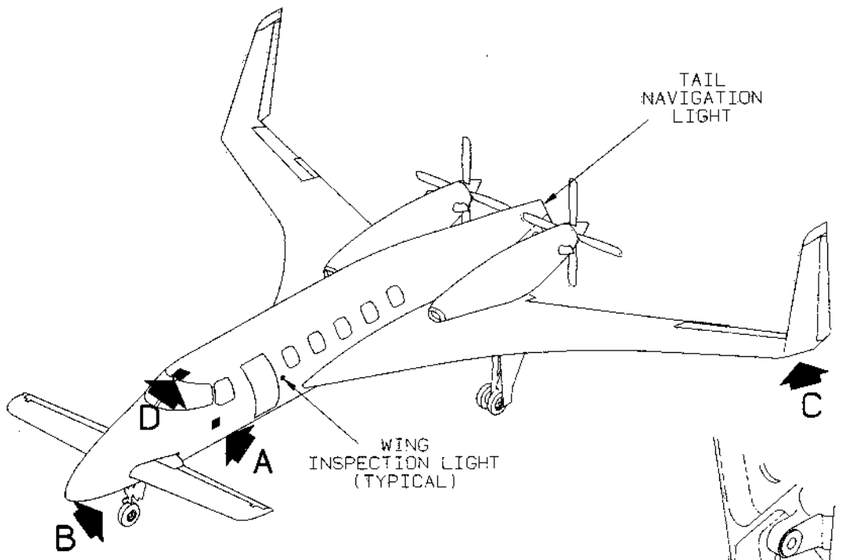
The TAXI light switch (A110S109) is powered by the center bus through the TAXI light circuit breaker (CB171). When the nose gear is down and locked, pushing the taxi light switch energizes the taxi light relay (K104) and illuminates the nose taxi light (DS108). The LANDING-NOSE light switch (A110S108) is powered by the triple-fed bus through the landing light control circuit breaker (CB141). The LANDING-NOSE light switch (A110S108) connects voltage to the nose landing gear relay (A182K4), closes the landing light relay (K128) and energizes the nose landing light (DS108), if the nose gear is down and locked. Voltage from the triple-fed bus through the landing gear control circuit breaker (CB133) closes the nose gear relays (A134K5 and A184K4), whenever the nose gear down-and-locked switch (S190) is actuated.

WING LANDING LIGHTS (Effectivity: All) (Figures 1 and 3)

A wing landing light is mounted in each aft wing tip forward of the tipsail fairing, for illumination of the runway during landing. The right landing light (DS106) is powered through the right landing light circuit breaker (CB139) from the right generator bus; and the left generator bus powers the left landing light (DS107) through the left landing light circuit breaker (CB236). Both wing landing lights are energized when the landing light power relays are actuated. The LANDING-WING light switch (A110S107), on the overhead panel, applies power from the triple-fed bus through the landing light control circuit breaker (CB141) to close the left landing light relay (K121) and the right landing light relay (K105).

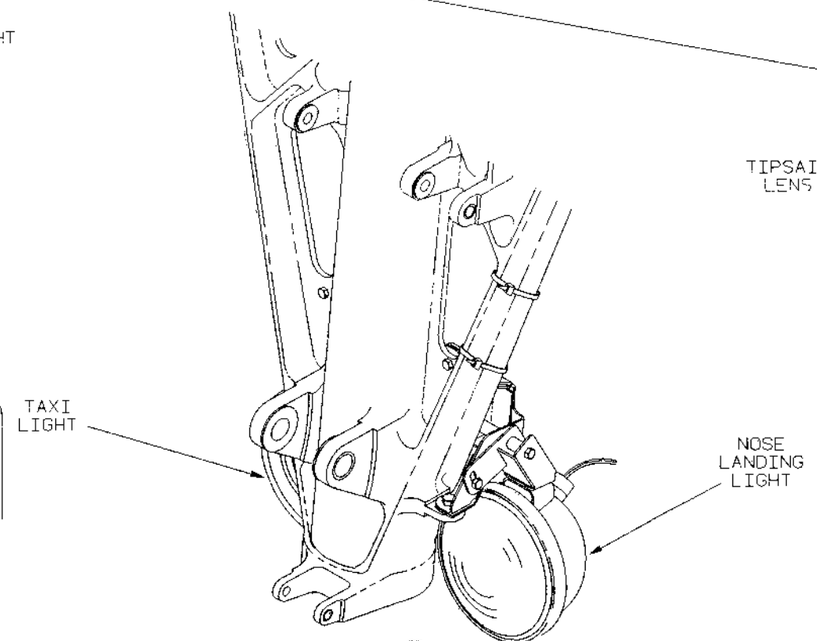
STROBE LIGHTS (Effectivity: All) (Figures 1 and 4)

The strobe lights are capable of operating at either high or low intensity, and increase the visibility of the airplane at night. An anticollision light assembly containing a strobe light is mounted on each tipsail fairing, and a strobe light power supply for each strobe light is mounted on the inside of each tipsail fairing. The left strobe light is slaved to the right, and both are controlled by the STROBE-LOW, STROBE-ANTI-COLL light switches on the overhead panel in the cockpit. When the STROBE-LOW switch (A110S113) is in the ON position, voltage from the center bus powers the strobe light power supplies through the strobe light circuit breaker (CB160). When the STROBE-LOW switch (A110S113) is actuated, the STROBE-ANTI-COL switch (A110S114) is not powered and does not function. When the STROBE-ANTI-COLL light switch (A110S114) is in the ON position and the STROBE-LOW switch is in the OFF position, voltage from the center bus through the strobe light circuit breaker



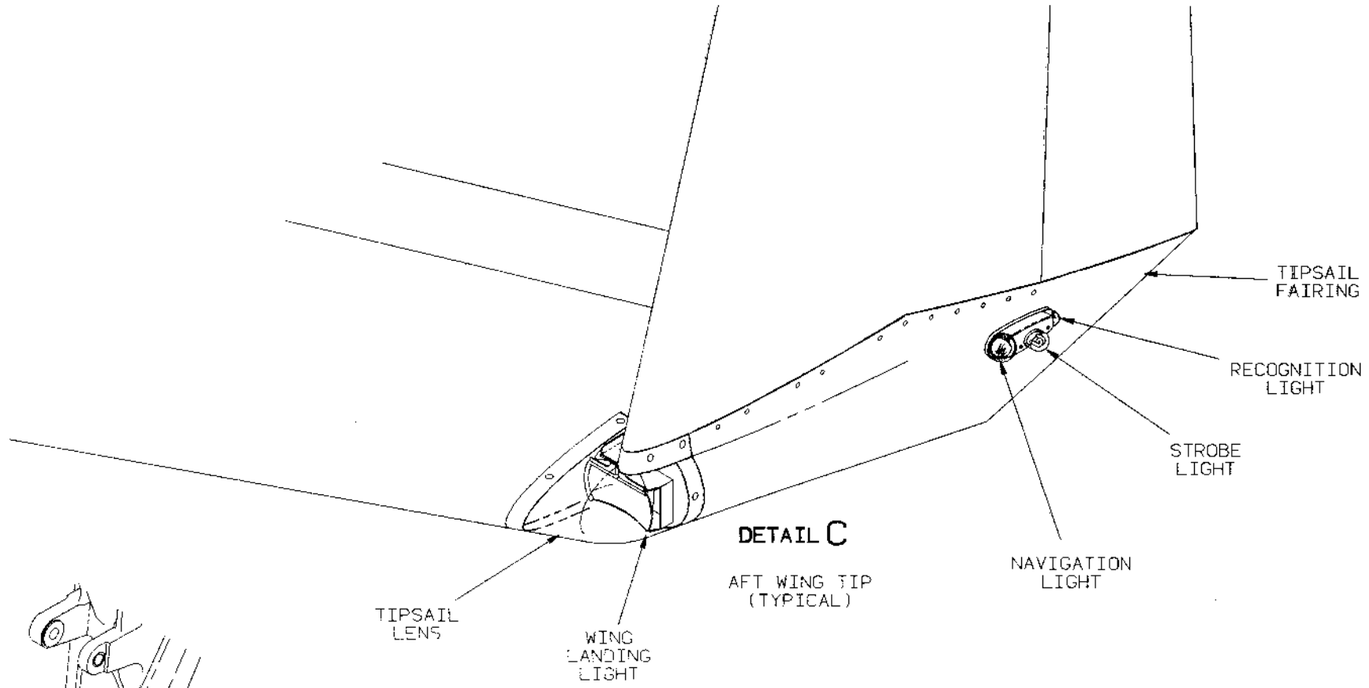
DETAIL A

AUXILIARY CIRCUIT
BREAKER PANEL



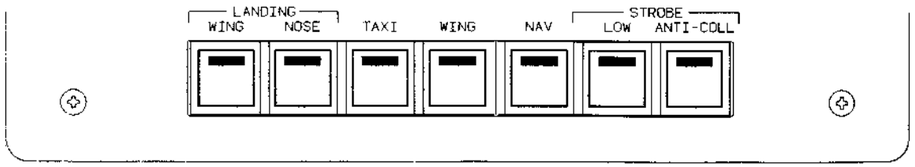
DETAIL B

NOSE LANDING
GEAR BRACE



DETAIL C

AFT WING TIP
(TYPICAL)



DETAIL D

OVERHEAD PANEL

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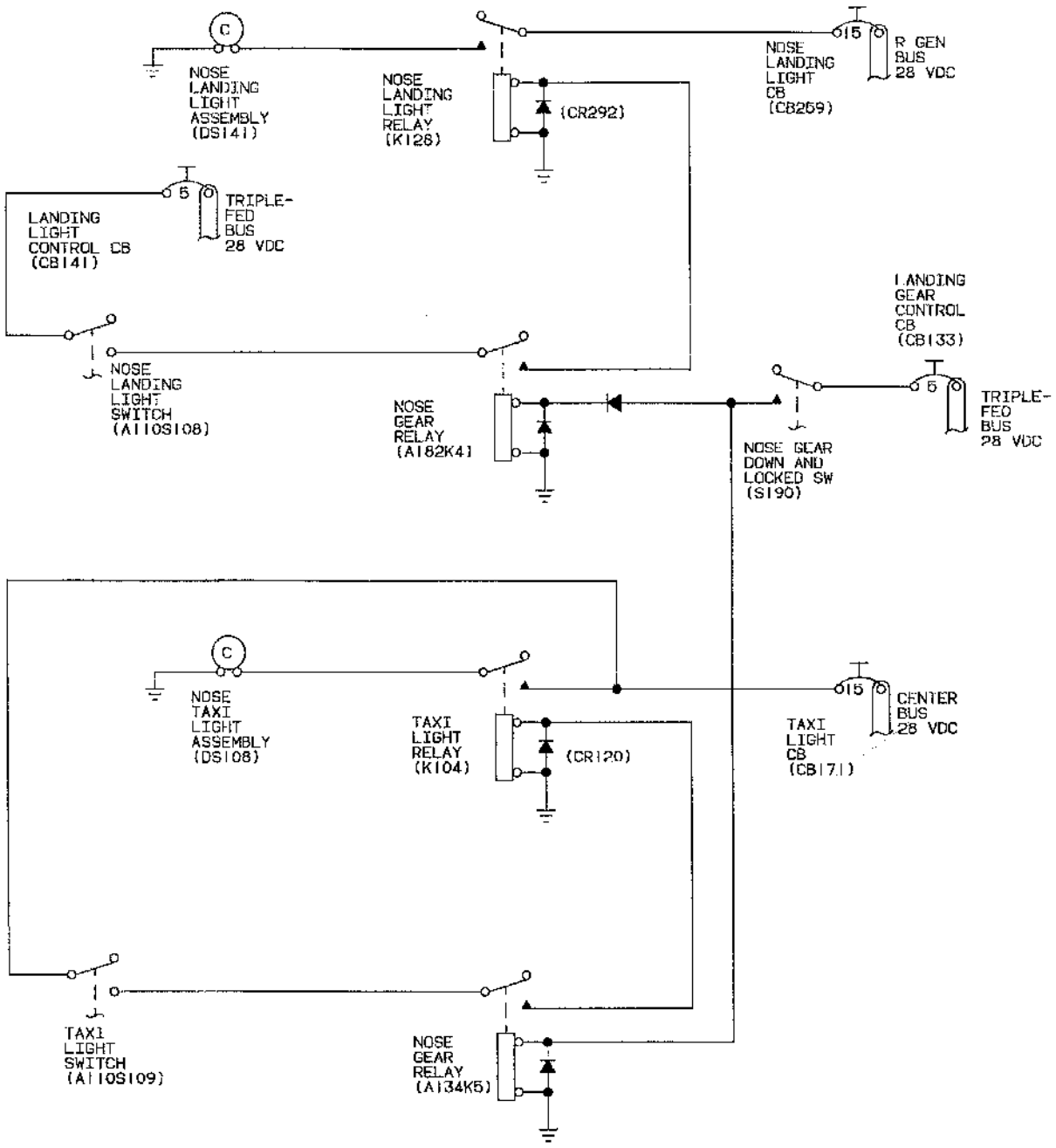
Exterior Lights (Effectivity: All)
Figure 1



C93NC33B4656

Nose Landing and Taxi Lights Schematic (Effectivity: All)
Figure 2

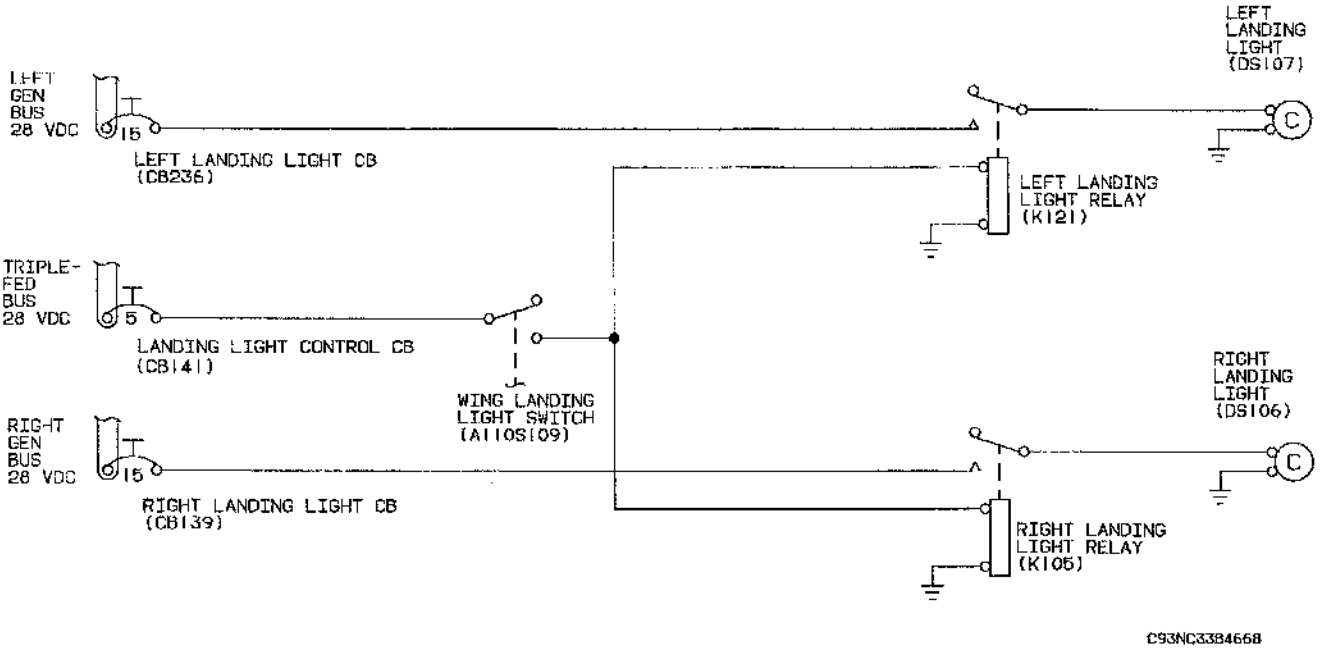
(CB160) will close the strobe light relay (A129K31). The strobe light relay (A129K31) connects center bus voltage to each strobe light power supply and also supplies a ground to the HI/LOW input, initiating anti-collision operation of the strobe lights.

NAVIGATION AND RECOGNITION LIGHTS
(Effectivity: All)
(Figures 1 and 4)

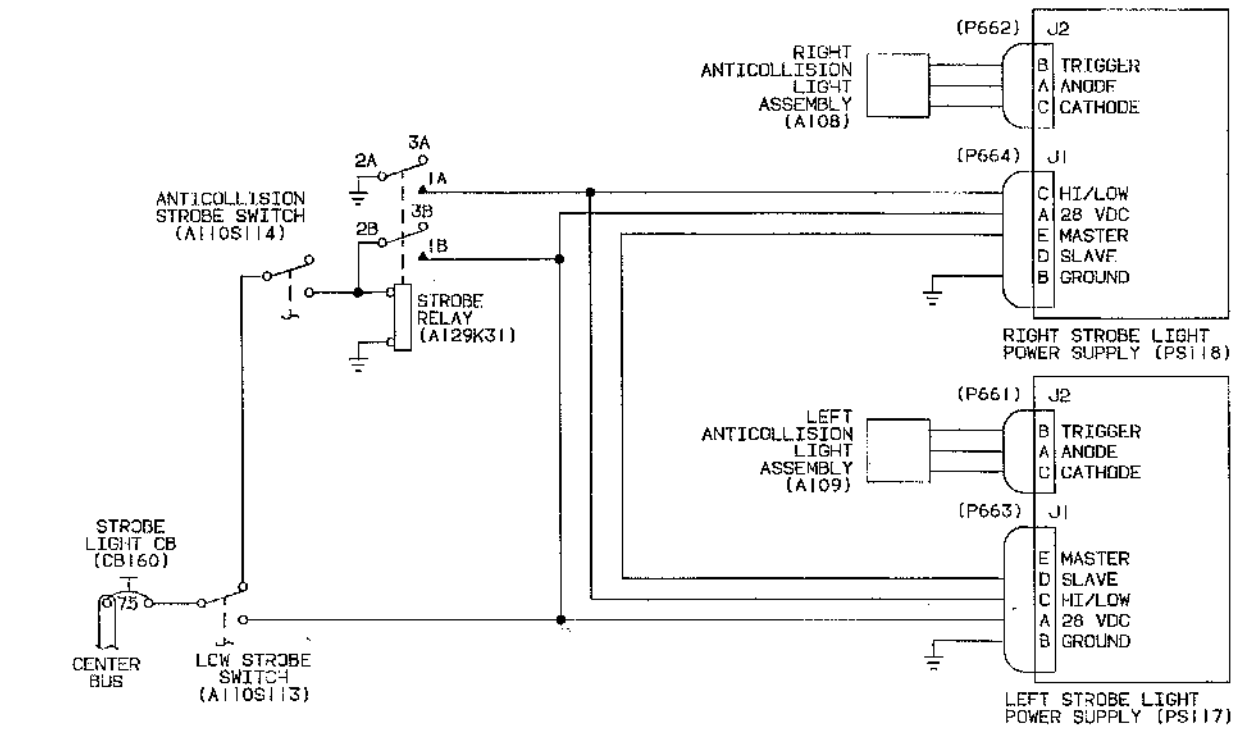
There are three navigation lights mounted on the exterior of the airplane. The left navigation light is mounted under a red glass dome in the forward part of the anti-collision light assembly on the left tipsail fairing. The right navigation light is mounted under a green glass dome in the forward part of the anticollision light assembly on the right tipsail fairing. The third navigation light is mounted under a clear glass dome in the tailcone facing aft. Each anticollision light assembly also contains a recognition light. Each recognition light is covered by a clear glass dome on the aft side of the assembly. When the NAV switch (A110S112) is in the ON position, voltage is supplied from the left generator bus through the navigation light circuit breaker (CB196) and switch (A110S112) illuminating both recognition lights (A108DS2 and A109DS2) and all three navigation lights (A170DS1, A108DS1 and A109DS1).

WING INSPECTION LIGHTS
(Effectivity: All)
(Figures 1 and 5)

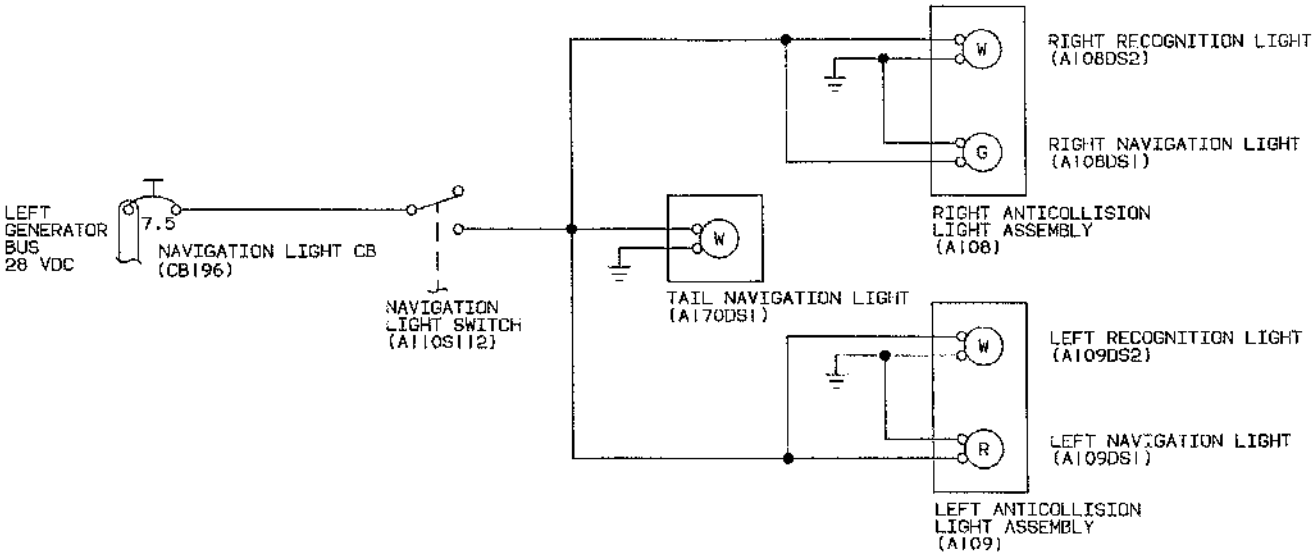
A wing inspection light is installed on the airplane to aid the pilot in detecting ice on the left side of the aft wing. A reflector inside the wing inspection light allows illumination of the ground area near the cabin door. The wing inspection light is vertically centered on the side of the fuselage just aft of the cabin door. An optional wing inspection light may also be installed, and is mounted opposite of the left wing inspection light, on the right side of the fuselage. This light does not contain a reflector. When the WING inspection light switch (A110S110) is in the ON position, voltage is applied from the center bus through the wing inspection light circuit breaker (CB172) and switch (A110S110), illuminating each wing inspection light assembly (A171 and A177).



Wing Landing Lights Schematic (Effectivity: All)
Figure 3



STROBE LIGHTS



NAVIGATION & RECOGNITION LIGHTS

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Strobe, Navigation and Recognition Lights Schematic (Effectivity: All)
Figure 4

EXTERIOR LIGHTS - TROUBLESHOOTING (Effectivity: All)

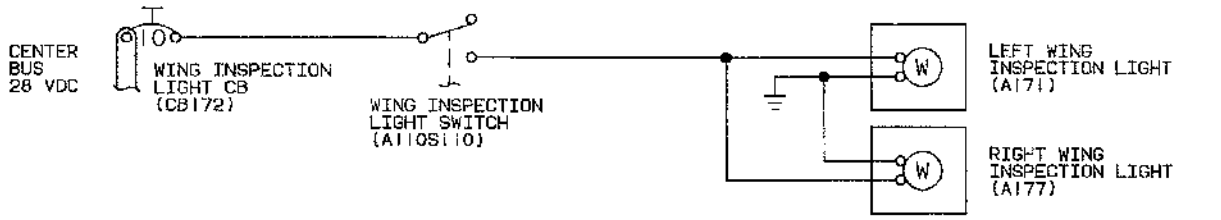
Most troubleshooting of the lighting system will consist of lamp replacement. If a substitution does not solve the problem, a check of all connections in the wiring to the lamp holder should be made. A loose, corroded, or damaged pin connection is a common cause of lighting failure.

STROBE LIGHT SYSTEM TROUBLESHOOTING (Effectivity: All)

If the strobe light is inoperative, troubleshoot the system as outlined in the following steps.

WARNING

High voltage may be present in the wiring between the strobe light power supply and flashtube. Always allow at least ten minutes after both STROBE switches are set to OFF before attempting any maintenance. Disconnecting or touching the wiring before the ten minutes has elapsed may result in electrical shock and/or injury to personnel.



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Wing Inspection Lights Schematic (Effectivity: All)
Figure 5

WARNING

When checking for illumination of the strobe lights, never look directly at the strobe flashtube.

a. Ensure that the strobe light circuit breaker is closed.

b. Check the strobe flashtube for any signs of damage such as a broken flashtube. If damaged, refer to STROBE FLASH TUBE REPLACEMENT in this chapter and replace the strobe flashtube.

c. Ensure that all connectors are properly secured to each other.

d. Check all connectors and wiring for nicks, cuts, burns or other damage that may cause a short or wiring break.

e. If the flashtube still fails to illuminate, connect a new flashtube to the power supply. If it fails to illuminate, check for battery voltage at the strobe light power supply. If the strobe light power supply is receiving battery voltage, the problem is most likely one of the following:

1. Moisture in the connectors to the strobe flashtube assembly.

2. Shorts in the power supply or the strobe flashtube assembly.

3. Shorts caused by accidental contact of the power supply to a foreign object.

4. Excessive heat build-up in the strobe light power supply.

CAUTION

Never connect the wiring of a good strobe light power supply to the circuit of an inoperative light. A short in the circuit of the inoperative light will damage the strobe light power supply.

f. If strobe light power supply replacement is required, remove the power supply using STROBE LIGHT POWER SUPPLY REMOVAL procedures outlined in this chapter.

EXTERIOR LIGHTS - MAINTENANCE PRACTICES (Effectivity: All)

NOTE

Ensure that the proper replacement lamp is used. The lamp number can usually be obtained from the base of the lamp. If not, reference the LAMP REPLACEMENT GUIDE in this chapter. Visually inspect the new lamp for a properly formed glass envelope. If it is distorted or misaligned with the lamp base, discard the lamp.

CHART 1
LAMP REPLACEMENT GUIDE - EXTERIOR LIGHTS (Effectivity: All)

LOCATION	LAMP #
Wing Landing Light	4182271
Tail Navigation Light	MS35478-1683
Wing Navigation Light	A508-28
Wing Inspection Light	MS25338-7079
Strobe Light	A-610
Nose Taxi Light	4587
Nose Landing Light	4596
Wing Recognition Light	W-1290-28

STROBE LIGHT POWER SUPPLY REMOVAL (Effectivity: All) (Figure 6)

a. Ensure that the STROBE-ANTI-COLL and STROBE-LOW switches on the overhead panel in the cockpit are OFF and pull the strobe light circuit breaker.

WARNING

High voltage may be present in the wiring between the strobe light power supply and strobe flashtube. Always allow at least ten minutes after both STROBE switches are set to OFF before attempting any maintenance. Disconnecting or touching the wiring before ten minutes have elapsed may result in electrical shock and/or injury to personnel.

b. Remove the nine screws attaching the tipsail lens to the the tipsail fairing (4), and remove the lens.

NOTE

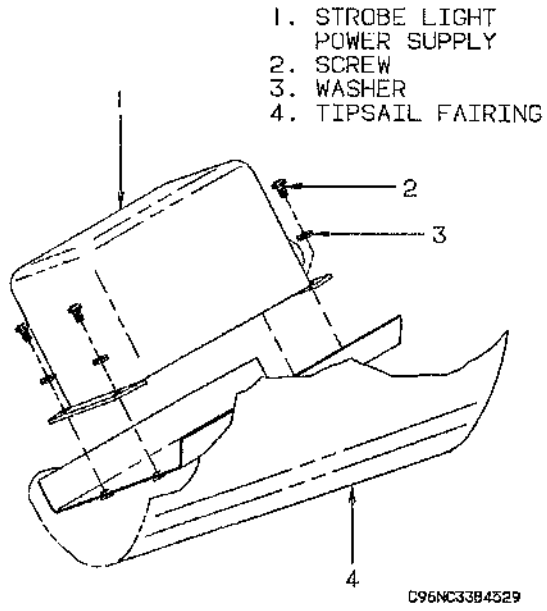
Do not allow wire harness to support the weight of the tipsail fairing (4).

c. Remove the screws attaching the tipsail fairing (4) to the tipsail, then lower the tipsail fairing (4) and disconnect the electrical connector at the tipsail fairing wire harness break.

d. Disconnect both strobe light power supply connectors at the power supply (1).

e. Remove the four strobe light power supply mounting screws (2) and washers (3).

f. Remove the strobe light power supply (1).



Strobe Light Power Supply Installation (Effectivity: All) Figure 6

STROBE LIGHT POWER SUPPLY INSTALLATION (Effectivity: All) (Figure 6)

CAUTION

The strobe light power supply (1) must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Align the strobe light power supply (1) mounting holes with the nutplates on the mounting bracket inside the tipsail fairing (4).

b. Install the four mounting screws (2) and washers (3).

c. Reconnect both electrical connectors to the strobe light power supply (1).

NOTE

Do not allow wire harness to support the weight of the tipsail fairing (4).

d. Reconnect the tipsail fairing wire harness break electrical connector.

e. Align the tipsail fairing mounting holes with the tipsail nutplates.

f. Install the tipsail fairing mounting screws.

g. Refer to TIPSAIL LENS INSTALLATION procedures in this chapter and install the tipsail lens.

h. Close the strobe light circuit breaker.

STROBE FLASHTUBE REPLACEMENT (Effectivity: All) (Figure 7)

a. Ensure that the STROBE-ANTI-COLL and STROBE-LOW switches on the overhead panel in the cockpit are OFF and pull the strobe light circuit breaker.

WARNING

High voltage may be present in the wiring between the strobe light power supply and strobe flashtube. Always allow at least ten minutes after both STROBE switches are set to OFF before attempting any maintenance. Disconnecting or touching the wiring before ten minutes have elapsed may result in electrical shock and/or injury to personnel.

b. Remove the nine screws attaching the tipsail lens to the the tipsail fairing and remove the lens.

NOTE

Do not allow wire harness to support the weight of the tipsail fairing.

c. Remove the screws attaching the tipsail fairing to the tipsail, then lower the tipsail fairing and disconnect the electrical connector at the tipsail fairing wire harness break.

d. Remove the lens retainer screws (6) attaching the lens retainer (7) to the base plate (12).

e. While removing the lens retainer (7) and flashtube dome (5), remove the glass dome (8) and the lens (3) from the base plate (12).

f. Remove tape from wire shielding over the strobe flashtube wiring.

g. Pull the wire shielding back to expose the flashtube wire splices.

h. Clip the flashtube wiring at each splice and pull it through the base plate wiring slot (13) while removing the flashtube assembly (4).

CAUTION

Use care not to contact the flashtube (4) with bare fingers. If contact occurs, clean the strobe flashtube (4) with a clean, lint-free cloth and cleaner (1, Chart 1, 33-00-00) and allow to air dry. Even a small amount of oil or grease will cause failure of the flashtube (4) when it is illuminated.

i. Feed the new strobe flashtube wiring through the base plate wiring slot (13) and determine proper wire connections with the BEECHCRAFT Starship 1 Wiring Diagram Manual. Splice the flashtube wiring using the procedures referenced in Chapter 24-00-00.

j. Pull the wire shielding over the flashtube wiring.

k. Wrap the shielding with tape to hold it in place.

NOTE

If the seal around the strobe light (4) base has been broken, reseal the periphery of the base with sealant (2, Chart 1, 33-00-00).

l. Position the flashtube (4) over its gasket (14) on the base plate (12).

m. Align the glass dome (8) and lens (3) over their gaskets (11 and 2) and install the lens retainer (7) and flashtube dome (5) over the strobe flashtube (4), gasket (14) and base plate (12).

n. Install both lens retainer screws (6).

WARNING

When checking for illumination of the strobe lights, never look directly at the strobe flashtube (4).

o. Close the strobe light circuit breaker.

WING NAVIGATION LAMP REPLACEMENT (Effectivity: All) (Figure 7)

a. Pull the navigation light circuit breaker.

b. Remove the two lens retainer screws (6) from the lens retainer (7).

c. While removing the lens retainer (7) remove both glass domes (5 and 8) and lens (3) from the base plate (12).

d. Remove the defective lamp (9) and install a new one.

NOTE

If the seal around the strobe light base has been broken, reseal the periphery of the base with sealant (2, Chart 1, 33-00-00).

e. Align the glass dome (8) and the lens (3) over the gaskets (11and 2) and install the lens retainer (7) and flashtube dome (5) over the strobe flashtube (4) and base plate (12).

NOTE

The wing navigation lamp (9) on the LH side has a red glass dome (8), while the RH side has a green glass dome (8).

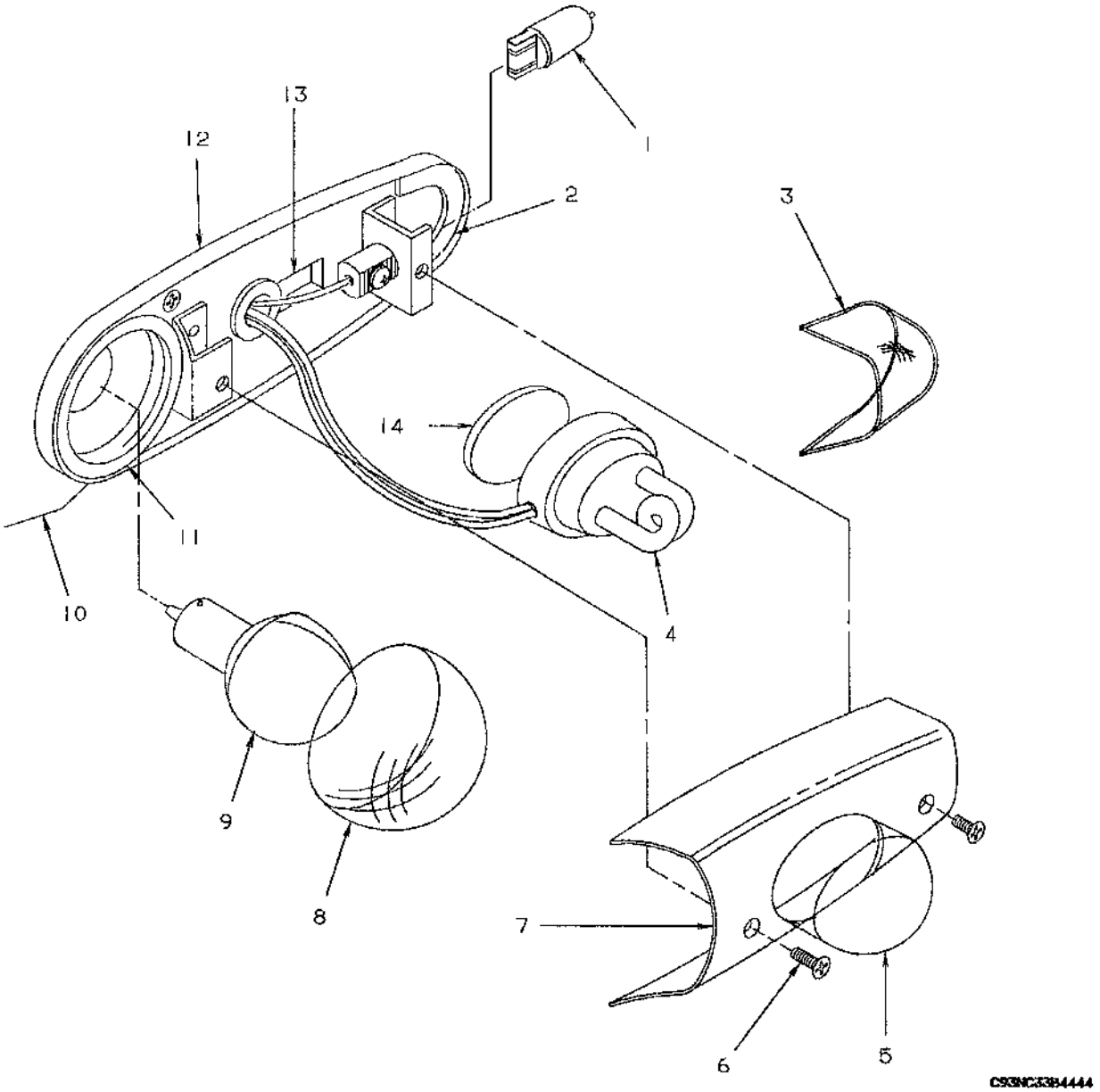
f. Install the two lens retainer screws (6).

g. Close the navigation light circuit breaker.

WING RECOGNITION LIGHT REPLACEMENT (Effectivity: All) (Figure 7)

a. Pull the navigation light circuit breaker.

1. RECOGNITION LAMP
2. GASKET
3. CLEAR GLASS LENS
4. STROBE FLASHTUBE
5. FLASHTUBE DOME
6. SCREW
7. LENS RETAINER
8. COLORED GLASS DOME (LH-RED, RH-GREEN)
9. NAVIGATION LAMP
10. TIPSAIL FAIRING
11. GASKET
12. BASE PLATE
13. WIRING SLOT
14. GASKET



AntiCollision Light Assembly Installation (Effectivity: All)
Figure 7

b. Remove the two lens retainer screws (6) attaching the lens retainer to the base plate (12).

c. Remove the lens retainer (7), lens (3) and both glass domes (5 and 8).

d. Pull the recognition lamp (1) from the socket.

CAUTION

Use care not to contact the new halogen lamp (1) with bare fingers. If contact occurs, clean the new lamp (1) with a clean, lint-free cloth and cleaner (1, Chart 1, 33-00-00) and allow to air dry. Even a small amount of oil or grease will cause failure of the lamp (1) when it is illuminated.

e. Press the new recognition lamp (1) into the socket.

NOTE

If the seal around the strobe light base has been broken, reseal the periphery of the base with sealant (2, Chart 1, 33-00-00).

f. Align the glass dome (8) over the gasket (11) and lens (3) over gasket (2) and install the lens retainer (7) and flashtube dome (5) over the strobe flashtube (4) and base plate (12).

g. Install the two lens retainer screws (6).

h. Close the navigation light circuit breaker.

TAIL NAVIGATION LIGHT REPLACEMENT (Effectivity: All) (Figure 8)

a. Pull the navigation light circuit breaker.

b. Remove the two retainer screws (6) securing the bonding ring (7) to the tailcone (1).

c. Remove the glass dome (5) and gasket (4).

d. Remove the faulty lamp (3) and install a new one.

e. Remove any old sealant from the glass dome (5) and gasket (4).

f. Apply a thin bead of sealant (3, Chart 1, 33-00-00) to the gasket surface and align the gasket (4) with the tailcone (1).

g. Position the bonding ring (7) over the glass dome (5) and align the edge of the glass dome (5) with the gasket (4) surface, ensuring a watertight seal.

h. Install both retainer screws (6).

i. Wipe off any excess sealant.

j. Close the navigation light circuit breaker.

NOSE LANDING AND TAXI LIGHT LAMP REPLACEMENT (Effectivity: All) (Figure 9)

a. Pull the nose landing or the nose taxi light circuit breaker.

b. Loosen the screw (6) and nut (9) on the retainer ring (8) enough to remove the retainer ring (8) and gasket (7).

c. Pull the sealed beam lamp(10) out of the lamp housing (4) enough to access the lamp terminals.

d. Loosen the terminal screws and tag the lamp leads as they are removed.

e. Install the lamp leads on the new lamp (10) as tagged, and tighten the terminal screws.

f. Align the new lamp (10) with the lamp housing (4).

g. Install the retainer ring (8) over the gasket (7) and edge of the lamp housing (4).

h. Tighten the retainer ring (8) screw (6) and nut (9) until the new lamp (10) is secure in the lamp housing (4).

i. Close either the nose landing or taxi light circuit breaker.

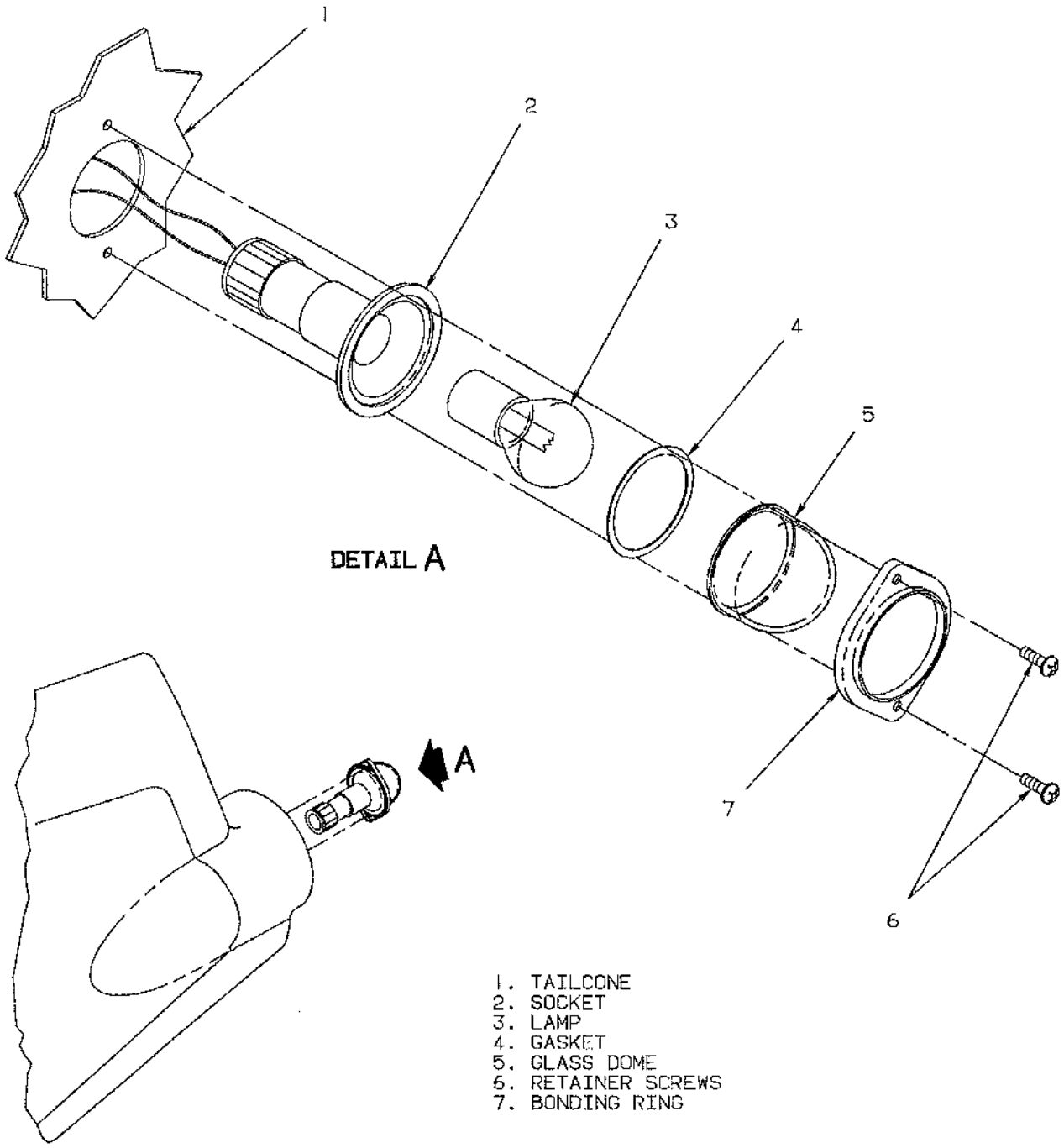
WING LANDING LIGHT REMOVAL (Effectivity: All) (Figure 10)

a. Pull the left or right landing light circuit breaker.

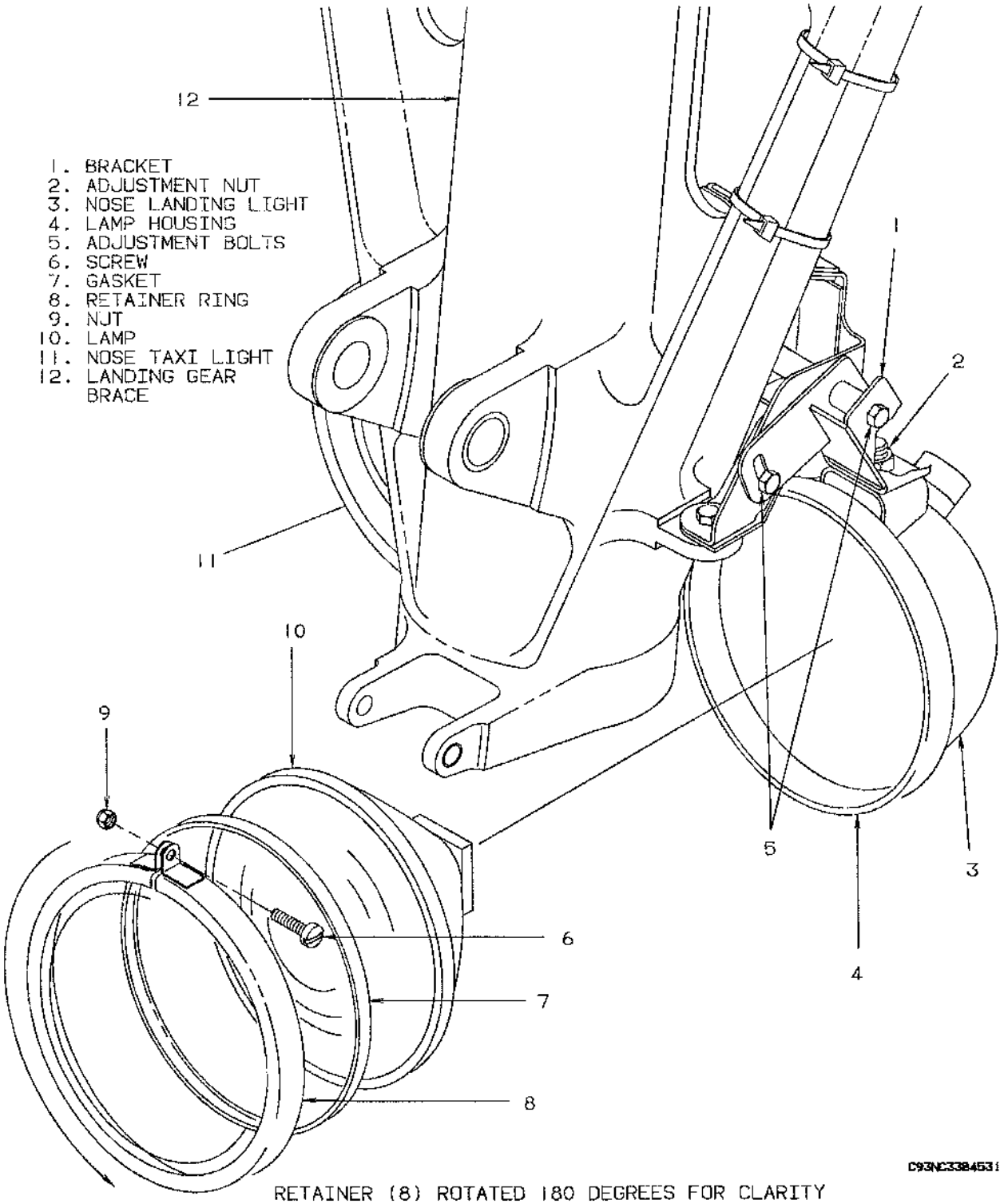
b. Remove the nine attaching screws (12) and washers (11) from the tipsail lens (13) and remove it from the tipsail fairing.

c. Remove the two bolts (2) and washers (1) attaching the landing light (9) to the adjustable hanger (3), and pull the landing light (9) out to access the wiring.

d. Tag the wiring, remove the two terminal screws (4) and washers (5) securing the wire terminal and nipples (6) to the landing light (9) and remove the lamp (9).



Tail Navigation Light Replacement (Effectivity: All)
Figure 8



Nose Landing and Taxi Light Replacement (Effectivity: All)
Figure 9

WING LANDING LIGHT INSTALLATION (Effectivity: All) (Figure 10)

CAUTION

The wing landing lights (9) must meet electrical bonding requirements specified in Chapter 20-00-00.

a. Attach the wire terminals to the landing light terminals according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual; then loosely install two terminal screws (4) and washers (5).

b. Torque the terminal screws (4) to 13.25 inch-pounds, or one full revolution after the washer (5) face contacts the wire terminal, whichever comes first.

c. Slide the nipples (6) over the wire terminals.

d. Install the two bolts (2) and washers (5) attaching the landing light (9) to the adjustable hanger (3).

e. Refer to TIPSAIL LENS INSTALLATION procedures in this chapter and install the tipsail lens.

f. Close the left or right landing light circuit breaker.

TIPSAIL LENS REMOVAL (Effectivity: All) (Figure 10)

a. Remove the nine attaching screws (12) securing the tipsail lens (13) to the tipsail fairing and remove the lens (13).

TIPSAIL LENS INSTALLATION (Effectivity: All) (Figure 10)

a. Remove any old sealant from the tipsail lens (13) mating edges.

NOTE

If any of the tipsail lens washers (11) are loose or become unbonded, bond them in place using TIPSAIL LENS WASHER BONDING procedures in this chapter.

b. Apply a thin bead of sealant (3, Chart 1, 33-00-00) to all edges of the tipsail lens (13) that contact the wing skin or tipsail fairing.

c. Align the tipsail lens (13) with the wing skin and tipsail fairing, then loosely install the nine tipsail lens attaching screws (12).

CAUTION

To avoid cracking of the tipsail lens (13), do not overtorque the tipsail lens attaching screws (12).

d. Torque the tipsail lens attaching screws (12) to 25 inch-lbs..

e. Wipe off any excess sealant around the tipsail lens (13).

f. Use water to check for leaks. If any are present apply more sealant or repeat installation steps as necessary.

TIPSAIL LENS WASHER BONDING (Effectivity: All) (Figure 10)

a. Remove any old adhesive from the washer (11) bonding surfaces by sanding with sandpaper or cleaning with naptha (4, Chart 1, 33-00-00).

b. Apply the adhesive (5, Chart 1, 33-00-00) to the washer bonding surface on the tipsail lens (13).

NOTE

When bonding, use care to prevent entrapment of air bubbles between bonding surfaces. The presence of air bubbles has an adverse effect upon bond strength.

c. Align the washer (11) with the tipsail lens (13) and press into place.

d. Apply enough pressure to the washer (11) for a tight bond, but use care not to crack the tipsail lens (13).

e. Allow the adhesive 25 minutes to set. Cure time is 6 to 8 hours.

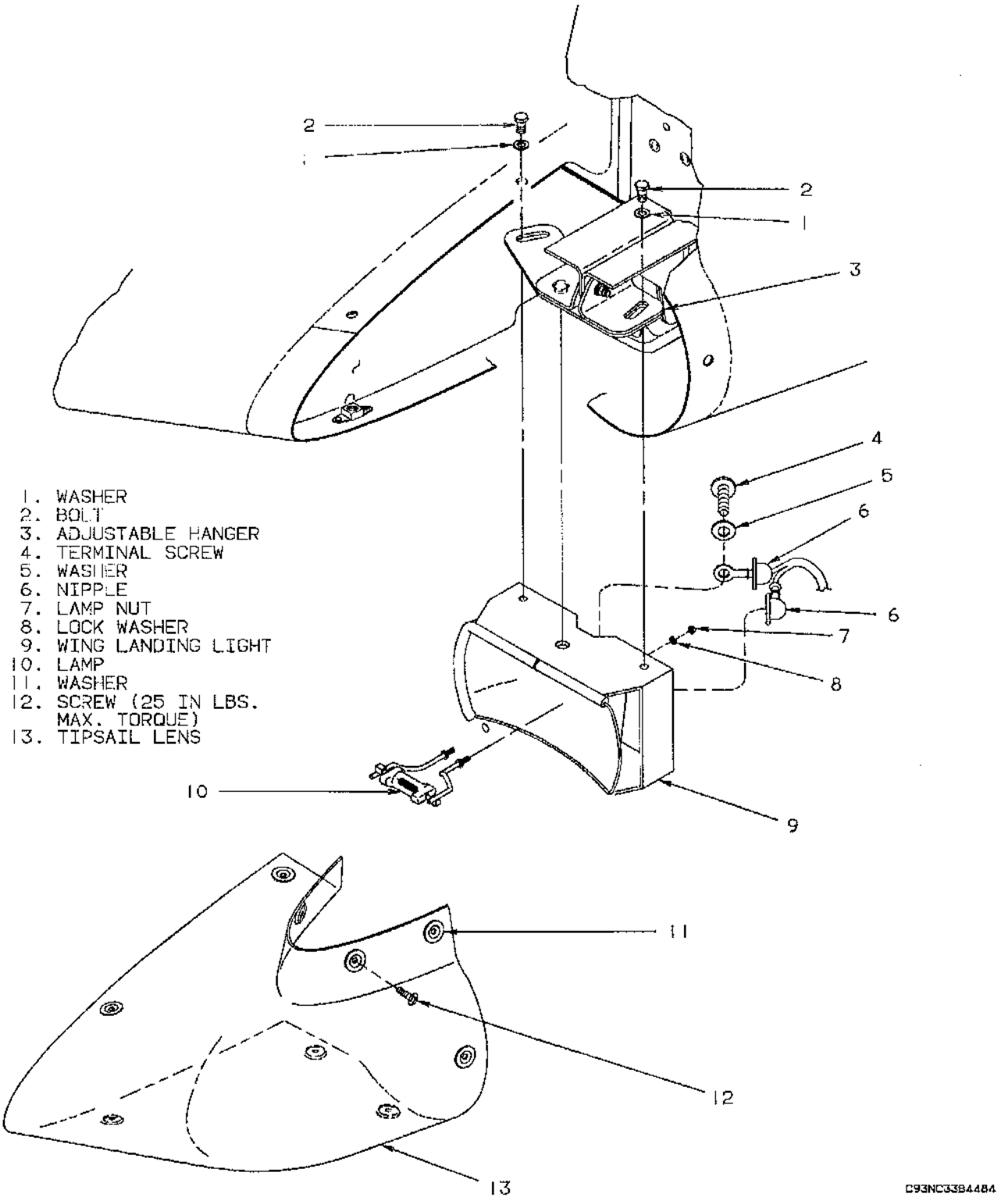
f. Refer to TIPSAIL LENS INSTALLATION procedures in this chapter and install the tipsail lens.

WING LANDING LIGHT LAMP REPLACEMENT (Effectivity: All) (Figure 10)

a. Pull the left or right landing light circuit breaker.

b. Refer to TIPSAIL LENS REMOVAL in this chapter, and remove the tipsail lens (13).

c. Remove the two lamp nuts (7) and lock washers (8) from the wing landing light lamp leads.



Wing Landing Light Installation (Effectivity: All)
Figure 10

d. Pull the lamp out the front of the wing landing light.

CAUTION

Use care not to contact the new halogen lamp (10) with bare fingers. If contact occurs, clean the new lamp (1) with a clean, lint-free cloth and cleaner (1, Chart 1, 33-00-00) and allow to air dry. Even a small amount of oil or grease will cause failure of the lamp (10) when it is illuminated.

e. Install the new lamp (10) in a horizontal position.

f. Loosely install two lock washers (8) and lamp nuts (7) on the lamp leads.

g. Torque the lamp nuts (7) to 4.5 ±.5 inch-pounds or 3/8 of a revolution after the washer face contacts the base, whichever comes first.

h. Refer to TIPSAIL LENS INSTALLATION in this chapter, and install the tipsail lens (13).

i. Close the left or right landing light circuit breaker.

WING INSPECTION LIGHT LAMP REPLACEMENT (Effectivity: All) (Figure 11)

a. Pull the wing inspection light circuit breaker.

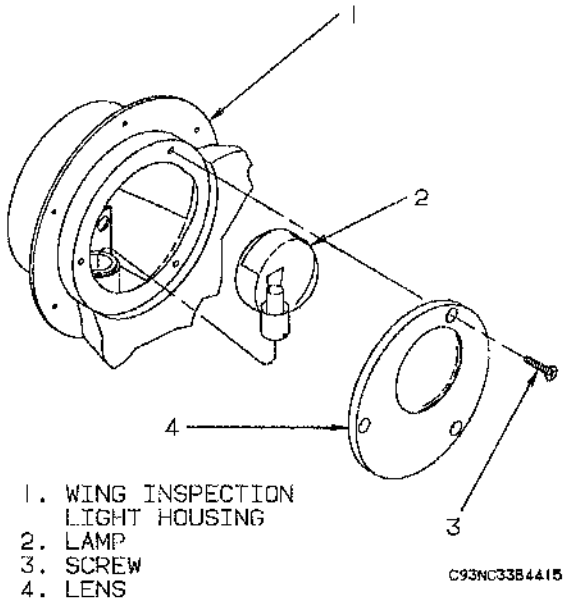
b. Remove the three screws (3) from the lens face (4).

c. Remove the wing inspection light lens (4).

d. Remove the old lamp (2) and install a new one.

e. Secure the lens (4) to the wing inspection light housing (1) with three screws (4).

f. Close the wing inspection light circuit breaker.



Wing Inspection Light Lamp Replacement (Effectivity: All) Figure 11

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Beechcraft
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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapter 32-10-00 and 32-20-00 for the proper pin installation procedures.

The navigation system for Starship 1 is divided into five major categories:

a. Flight Environmental Data

b. Attitude and Direction

c. Landing System

d. Independent Position Determining

e. Dependent Position Determining

FLIGHT ENVIRONMENTAL DATA (Effectivity: All)

The flight environmental data system senses the ambient pressures and temperatures to provide airspeed, altitude, and air temperature. This data is used with other systems for flight navigation.

a. The pitot and static system is the primary sensor for environmental data. The major components for the environmental system are:

1. The air data computer which feeds data to the airspeed display, altitude/vertical speed display and the various avionics systems.

2. The standby airspeed and altimeter indicators that are standard barometric/mechanical indicators, which take data from ambient pitot/static pressures.

3. The differential pressure switch which also takes data from ambient pitot/static pressures.

b. The outside air temperature which is displayed on the airspeed display as indicated outside air temperature (IOAT), corrected outside air temperature (OAT) and international standard atmosphere temperature (ISA). These are provided through an external sensor probe and the air data computers.

ATTITUDE AND DIRECTION (Effectivity: All)

The attitude and direction system provides data from the earth's magnetic field and gyroscopic sensors.

a. The primary airplane attitude and directional information is provided by the attitude heading system (AHS). The AHS is a dual system that senses the airplane attitude, heading, and 3-axis rate/accelerations. The pilot system is identical to the copilot system. Each system contains a computer (with a compensation module) and a flux detector. The AHC computer is remote mounted in a precisely leveled mount. The AHC replaces a conventional vertical gyro, directional gyro, three rate gyros, and three linear accelerometers. A Compensation Unit is mounted in the front end of the AHC Computer. A Flux Detector Unit is mounted in the wing of the airplane and provides magnetic flux measurements to the AHC for use in computing the airplane heading.

b. The standby compass is a standard magnetic compass mounted to the windshield centerpost. It indicates direction in degrees from magnetic north only.

c. The standby gyro is a standard mechanical/electrical instrument that indicates the attitude of the airplane against a simulated background of earth and sky.

LANDING SYSTEM (Effectivity: All)

The landing system provides data from ground station transmitters which include the glideslope and the marker beacons.

a. The glideslope indicates to the crew the position of the airplane relative to the glidepath on final approach to landing. Its data is indicated on the primary flight display and navigation display.

b. The marker beacons indicate to the crew the location of the airplane relative to the distance from the end of the runway when on final approach to landing. Its data will be annunciated on the primary flight display by the illumination of a different colored light for each marker accompanied by an audible tone (when MKR AUDIO is selected) as each of the three markers are passed over.

INDEPENDENT POSITION DETERMINING (Effectivity: All)

Independent position determining is a means of establishing position mainly independent of the ground installations. The weather radar will detect and display precipitation and turbulence associated with precipitation in the path of the airplane. The radar will also map prominent landmarks along the flight path.

a. The radio altimeter system provides data in distance of vertical feet above the terrain when the airplane is at or below 2,500 feet above ground level. A signal is beamed to the ground and back to the airplane receiver.

DEPENDENT POSITIONING DETERMINING (Effectivity: All)

Dependent position determining is made possible by various radio frequency transmitters and receivers that depend on signal data from both ground stations and airborn stations to include the following;

a. The VOR navigation system provides data to determine the position of the airplane from the ground-based VOR transmitter.

b. The DME (distance measuring equipment) system provides data on ground speed, time to station, and on distance in miles from the station.

c. The ADF (automatic direction finding) system provides direction in degrees from the airplane heading to the ground station.

d. The VLF (very low frequency) system and Omega system provides data to establish geographic position at any time and at any point along the flight path.

e. The ATC (air traffic control) transponder system provides data to the air traffic controller.

SPECIAL TOOLS AND RECOMMENDED MATERIALS (Effectivity: All)

The special tools and recommended materials listed in Charts 1 and 2 as meeting federal, military or supplier specifications are provided for reference only and are not specifically prescribed by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the supplier, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the supplier of the product to be used.

CHART 1
SPECIAL TOOLS (Effectivity: All)

TOOL NAME	PART NO.	SUPPLIER	USE
1. Pitot/Static Test Box	TK 1783-1/939 TK 2101/939	Beech Aircraft Corp. Wichita, KS Beech Aircraft Corp. Wichita, KS	To perform pitot/static system leakage test. To perform pitot/static system leakage test.
2. Mast Adapters	TK1783-3/939	Beech Aircraft Corp. Wichita, KS	To perform pitot/static system leakage test.
3. Transponder Ramp Test Set	ATC-601	IFR Systems Inc., 10200 W. York Wichita, KS 67201	To perform transponder ramp test.
4. DME Ramp Test Set	ATC-600	IFR Systems Inc., 10200 W. York Wichita, KS 67201	To perform DME ramp test.

CHART 2
RECOMMENDED MATERIALS (Effectivity: All)

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
1. Sealant	MIL-S-8802, Type II	Pro Seal 890	Pro-Seal Corp., Div. of Essex Chemical Corp., Aerospace Products Compton, CA 90220
2. Sealer	MIL-S-8802, Type I	PR1422	Products Research and Chemical Corp., 5426 San Fernando Road Glendale, CA 91203
3. Corrosion Preventive	MIL-C-5541	Alodine 1200	Amchem Products Inc., Spring Garden Street Amber, PA 19002
4. Sealant	MIL-A-46146	RTV 738	P/N of Dow Corning Corp. Midland, MI. CAGE 71984

FLIGHT ENVIRONMENTAL DATA - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lockpin installation procedures.

PITOT/STATIC PRESSURE SYSTEM (Effectivity: All) (Figures 1 and 2)

The pitot/static pressure system provides a source of impact air pressure and static air for operation of the standby pneumatic airspeed indicator and the standby pneumatic altimeter. The pitot/static pressure system also provides a source of impact air pressure and static air to the Air Data Computers (ADC) which provide air data information for operation of the Airspeed Indicators (ASI) and Altimeter/Vertical Speed Indicators (ALI). An Outside Air Temperature (OAT) sensor located on the nose of the airplane provides outside air temperature data to the ADC's. For detailed information on the Air Data System (ADS) refer to the Collins Starship 1 Avionics System Manual (523-0774076-00111A or latest revision).

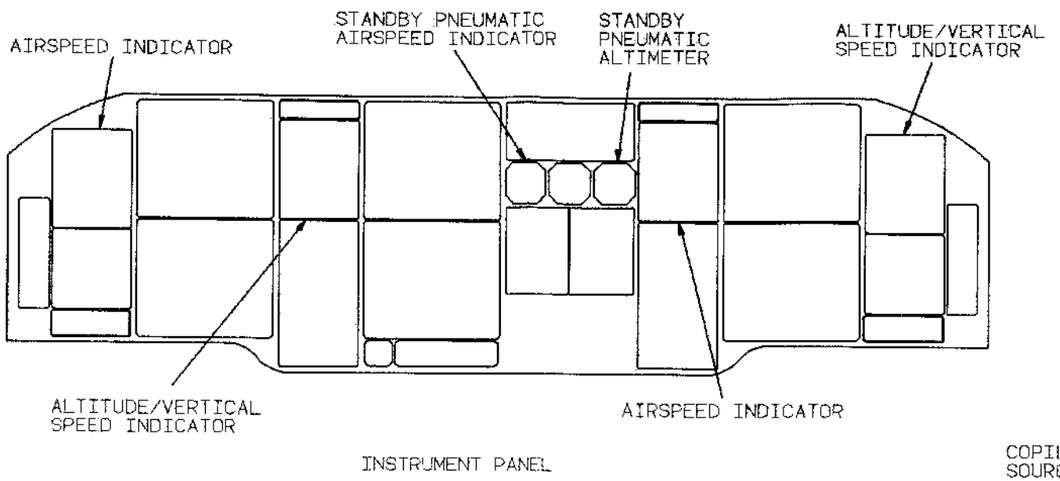
The pitot/static pressure system is comprised of two pitot/static masts (pilot's and copilot's), a STATIC SOURCE select switch, two static system isolation solenoids, an airspeed pressure differential switch (part of the landing gear position/warning circuit) and the associated drain valves, plumbing and connectors. There are two drain valves located in the cockpit behind the instrument panel and two drain valves adjacent to each ADC. The pitot/static masts are mounted just aft of the radome on either side of the nose and each one contains a heating element which prevents the formation of ice over the pitot/static mast openings. The pitot/static plumbing is routed from the pitot/static masts through the forward nose avionics compartments aft to the RH upper avionics compartment where it connects to each ADC. It also penetrates the forward pressure bulkhead into the cockpit where it connects to the standby pneumatic airspeed indicator and standby pneumatic altimeter. The plumbing is sloped and contains no water traps to prevent moisture from settling in the system. The two static isolation solenoids are installed in the static plumbing at the forward nose avionics compartments and will isolate each mast of the static system from the other when the switch placarded STATIC SOURCE, ISOLATED-NORM on the pilot's outboard subpanel is placed in the ISOLATED position.

The standby pneumatic airspeed indicator and standby pneumatic altimeter are two-inch mechanical instruments mounted in the upper center area of the instrument panel. The static pressure source for both standby pneumatic instruments is the copilot's pitot/static mast when the STATIC SOURCE select switch is in the ISOLATED position. During normal operation, the STATIC SOURCE select switch is in the NORM position; therefore, the static source to the standby instruments is the average of the static ports on the pilot's and the copilot's pitot/static probes. Electrical power is provided for illumination of the standby instruments and to operate the internal vibrator on the standby pneumatic altimeter. Refer to Chapter 24-32-00 for detailed information on standby instrument electrical power.

AIRSPEED INDICATOR (ASI) (Effectivity: All)

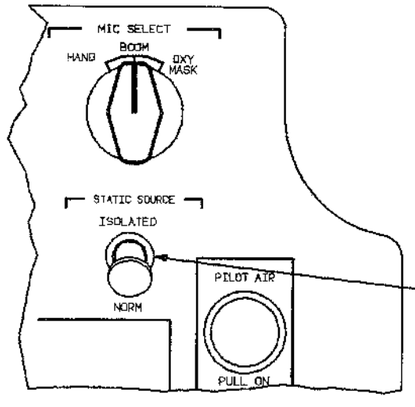
The airspeed display exhibits indicated airspeed and maximum operating airspeed information in a conventional format, with a full-size, circular display, and a large numerical readout of airspeed in the center. Provisions are made to display numeric (only) readout of true airspeed and air temperature. A front panel control is provided to enable the selection of IOAT, OAT, or ISA.

The airspeed displays are to the left of both primary flight displays and receive their data from the air data computer (ADC).

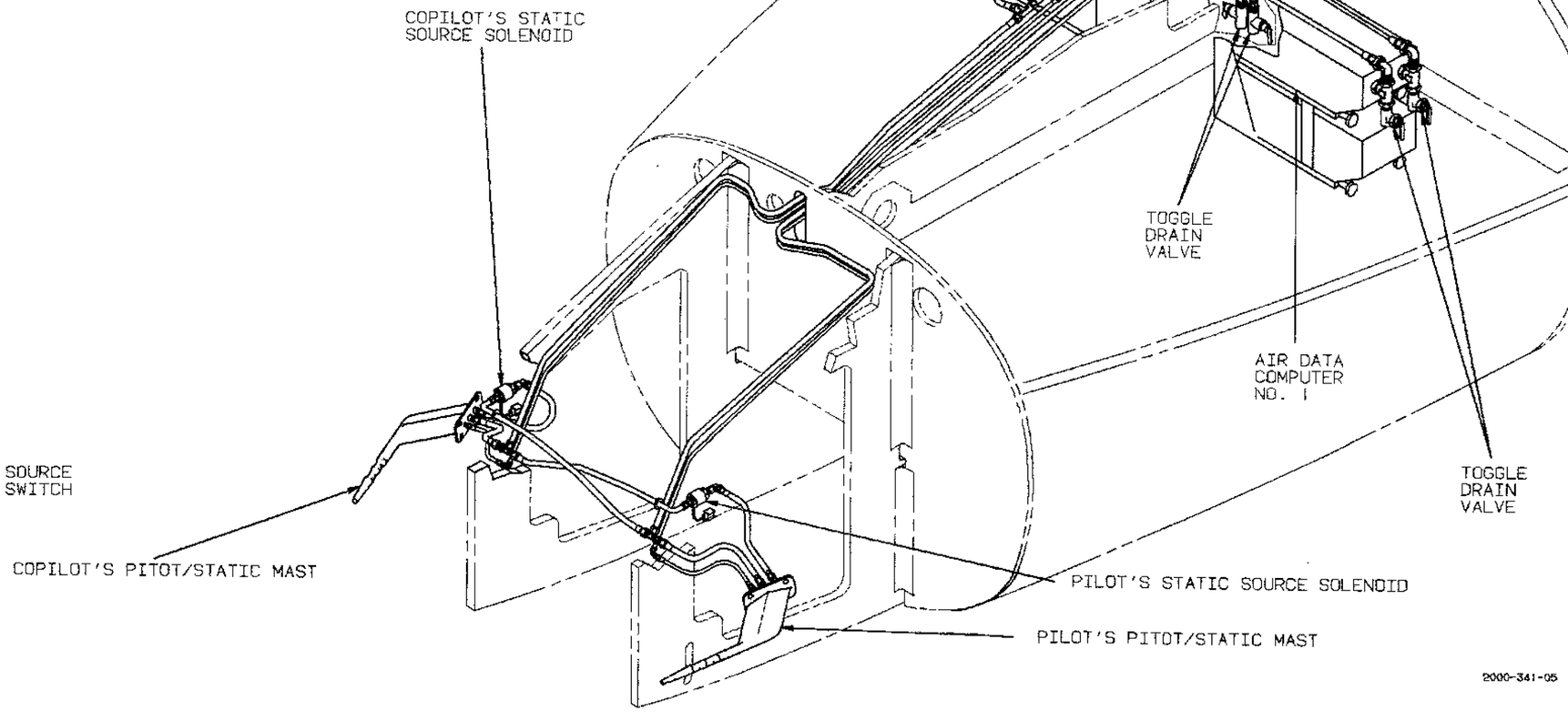


INSTRUMENT PANEL

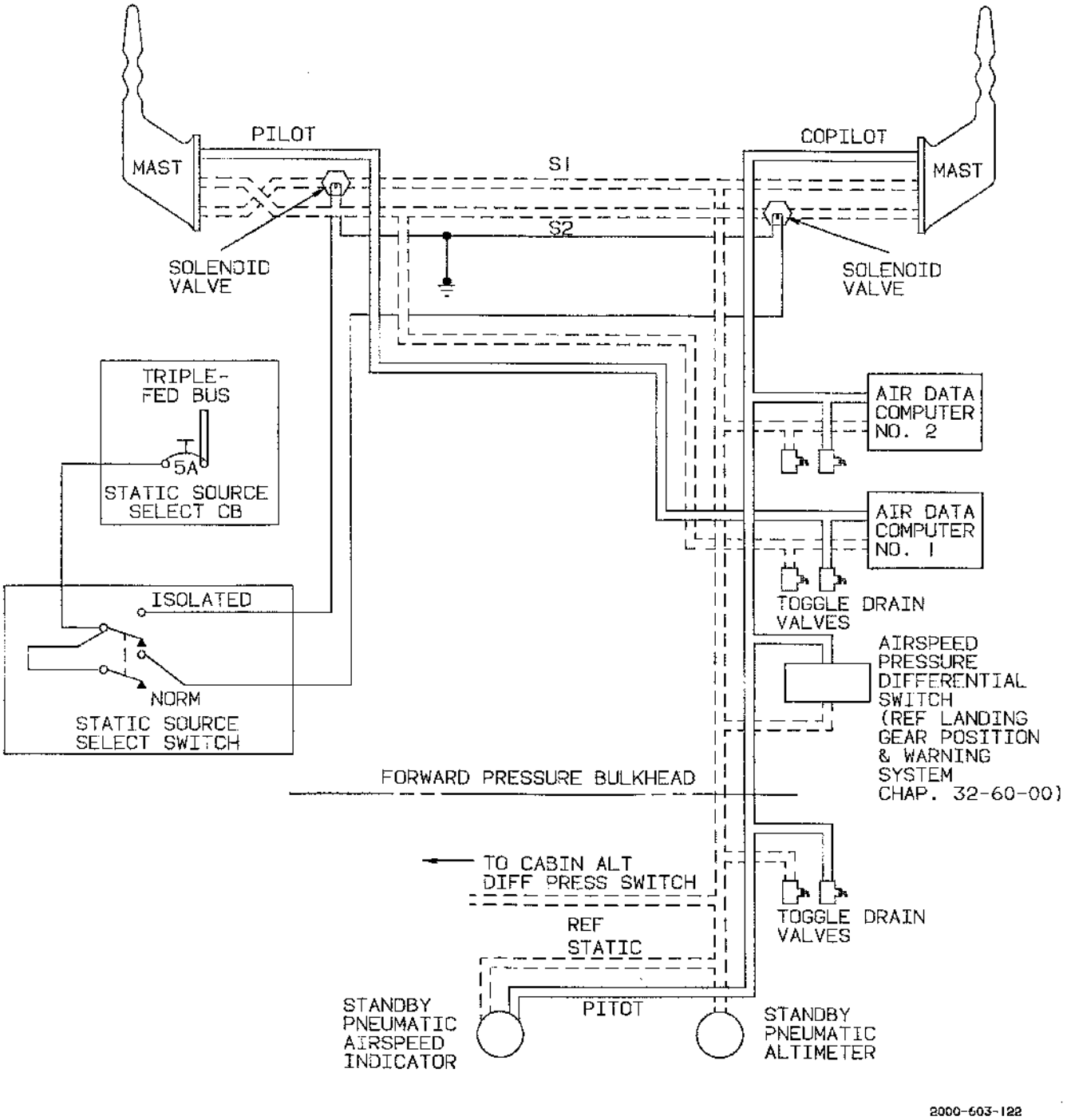
DETAIL A



PILOT'S
OUTBOARD SUBPANEL



Pitot/Static Pressure System (Effectivity: All)
Figure 1



Pitot/Static Pressure System Functional Schematic
(Effectivity: All)
Figure 2

STANDBY PNEUMATIC AIRSPEED INDICATOR (Effectivity: All)

The pneumatic airspeed indicator is one of three standby indicators. The standby airspeed indicator is used as a backup if the electrical system or the ASI should fail, and as a cross-check if the ASI display system is suspected to be in error.

The indicated airspeed displayed is from uncorrected pitot/static inputs.

ALTIMETER/VERTICAL SPEED (ALI) (Effectivity: All)

The ALI displays barometrically derived altitude from the air data computer on a conventional full-sized circular display with a numerical altitude readout in the center. An analog display of vertical speed is also displayed.

The ALI receives data from the air data computer and is located to the right of the primary flight displays.

STANDBY PNEUMATIC ALTIMETER (Effectivity: All)

The pneumatic altimeter is one of three standby indicators. The standby altimeter is used as a backup if the electrical system or the ALI should fail, and as a cross-check if the ALI display system is suspected to be in error.

AIR DATA COMPUTER (Effectivity: All)

The ADC air data computer interfaces with electronic air data display instruments and various other avionic systems. For detailed information on the Air Data System (ADS) refer to the Collins Starship 1 Avionics System Manual (523-0774076-00111A or latest revision).

OUTSIDE AIR TEMP SENSOR (OAT) (Effectivity: All)

Temperature information is fed to the air data computer, and is read out on the airspeed indicators as indicated outside air temperature. Provisions are made to display OAT and IOAT.

FLIGHT ENVIRONMENTAL DATA - TROUBLESHOOTING (Effectivity: All)

PITOT/STATIC PRESSURE SYSTEM TROUBLESHOOTING (Effectivity: All)

If a fault is detected in the pitot/static pressure system, perform the procedures of CHART 1 in this section to troubleshoot the system.

CHART 1
PITOT/STATIC AIR PRESSURE SYSTEM TROUBLESHOOTING (Effectivity: All)

Problem	Possible Cause	Corrective Action
1. Heating element inoperative.	a. Static Source Switch fails to operate properly.	a. Verify operation and replace the switch.
	b. Grounded or open circuit.	b. Check for continuity and repair.
	c. Heating element in pitot does not operate.	c. Replace the pitot/static mast.
2. Circuit breaker keeps tripping.	a. Grounded wire.	a. Check for continuity and repair.
3. Instruments inoperative or erratic in operation.	a. Clogged lines.	a. Disconnect lines at instruments and air data computers, then blow out with low pressure air.
	b. Line leaks.	b. Check lines for loose fittings at all connection points.

AIRSPEED INDICATOR (ASI) TROUBLESHOOTING (Effectivity: All)

NOTE

The two ASI's may be swapped with each other to verify airplane wiring and to isolate a failed unit.

Apply power to all avionics systems. Wait 5 minutes for the diagnostic system to stabilize. Refer to the level 2 troubleshooting procedure in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

STANDBY PNEUMATIC AIRSPEED INDICATOR TROUBLESHOOTING (Effectivity: All)

If the standby airspeed indicator is suspected to be inaccurate, perform a PITOT/STATIC TEST as described in this chapter.

ALTIMETER/VERTICAL SPEED INDICATOR (ALI) TROUBLESHOOTING (Effectivity: All)

NOTE

The two ALI's may be swapped with each other to verify the airplane wiring and to isolate a failed unt.

Apply power to all avionics systems. Wait 5 minutes for the diagnostic system to stabilize. Refer to the the level 2 troubleshooting procedure in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

STANDBY PNEUMATIC ALTIMETER TROUBLESHOOTING (Effectivity: All)

If the standby altimeter is suspected to be inaccurate, perform a PITOT/STATIC TEST as described in this chapter.

AIR DATA COMPUTER TROUBLESHOOTING (Effectivity: All)

NOTE

The two ADC's may be swapped with each other to verify airplane wiring and to isolate a failed unit. The two ADM's may be swapped to isolate a failed module.

NOTE

An airplane static check MUST be performed if the pitot/static lines are removed from the computer.

Apply power to all avionics systems. Wait 5 minutes for the diagnostic system to stabilize. Refer to the the level 2 troubleshooting procedure in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

OUTSIDE AIR TEMP SENSOR (OAT) TROUBLESHOOTING (Effectivity: All)

If the outside air temperature sensor is suspected of being inaccurate, verify the displayed temperature on the airspeed indicator against the airfield air temperature standard. There are two temperature sensing elements in the probe. One element supplies information to the No.1 ADC and the other supplies information to the No. 2 ADC, so the two sides may be compared.

FLIGHT ENVIRONMENTAL DATA - MAINTENANCE PRACTICES (Effectivity: All)

PITOT/STATIC MAST REMOVAL (Effectivity: All) (Figure 3)

The pitot/static masts are located on the right and left side of the airplane nose. These masts are accessed through the right and left nose wheel well avionics compartments. Removal and installation are identical for either mast.

a. Remove all electrical power from the airplane.

b. Operate the alternate extension hand pump in the cockpit until the forward nose landing gear doors are fully open. Refer to Chapter 32-20-00 and install the lockpin in the landing gear nose gear door bellcrank.

c. Remove the appropriate nose wheelwell access panel (710AZ or 710BZ).

d. From inside the avionics compartment, disconnect and tag the three flexible air and static hoses (2, 3, and 4). Disconnect and tag the electrical connection (1).

e. Remove the screws (6) retaining the mast.

f. Gently scrape the sealer away from the mast plate and remove the pitot/static mast.

g. Remove the pitot/static mast to the workbench or a safe storage area.

PITOT/STATIC MAST INSTALLATION (Effectivity: All) (Figure 3)

a. Remove all sealer from the pitot mast and the nose mounting area using a nonmetallic scraper and a nonmetallic cleaner pad.

b. Position the mast (5) on the airplane and secure with screws (6). Install the screws wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

c. From inside the avionics compartment, connect the electrical connector (1) and the three flexible air hoses (2, 3, and 4) to the pitot mast.

d. Apply sealer (1, Chart 2, Chapter 34-00-00) around the periphery of the pitot mast where it contacts the fuselage skin.

e. Install the appropriate nose wheelwell access panel (710AZ or 710BZ).

f. Remove the landing gear door locking pin and push the doors closed.

g. After the sealer has cured, touch up the airplane with paint.

h. Perform a PITOT/STATIC PRESSURE SYSTEM INSPECTION AND LEAKAGE TEST as described in this chapter.

AIRSPEED DISPLAY REMOVAL/ INSTALLATION (Effectivity: All)

Refer to Chapter 31-10-00 for removal/installation procedures.

STANDBY AIRSPEED INDICATOR REMOVAL (Effectivity: All) (Figure 4)

a. Loosen, but do not remove the screws (6) at the corners of the indicator. The screws are needed to retain the clamp (4) behind the instrument panel.

b. Pull indicator (5) out from the panel until it clears the mounting hole.

c. Identify, tag, and disconnect the electrical connector (2), pitot hose (1), and the static hose (3).

NOTE

If the electrical or pitot/static line connections are not long enough to allow the indicator to clear the panel, remove the center reversionary panel directly above the indicator to access and disconnect the lines from the indicator. Refer to Chapter 31-10-00 for reversionary panel removal procedures.

d. Install caps on all disconnected hoses.

e. Cap the fittings on the back of the indicator.

STANDBY AIRSPEED INDICATOR INSTALLATION (Effectivity: All) (Figure 4)

a. Remove the caps from the back of the indicator.

b. Connect the pitot and static hoses (1 and 3) and the electrical connector (2) to the indicator (5).

c. Position the indicator (5) in the instrument panel.

d. Secure the indicator (5) with the screws (6).

e. Install the center reversionary panel if it was removed. Refer to Chapter 31-10-00 for installation procedures.

f. Perform a pitot/static leak check. Refer to PITOT/ STATIC PRESSURE SYSTEM INSPECTION AND LEAKAGE TEST in this chapter

ALTIMETER/VERTICAL SPEED REMOVAL/INSTALLATION (Effectivity: All)

Refer to Chapter 31-10-00 for removal/installation procedures.

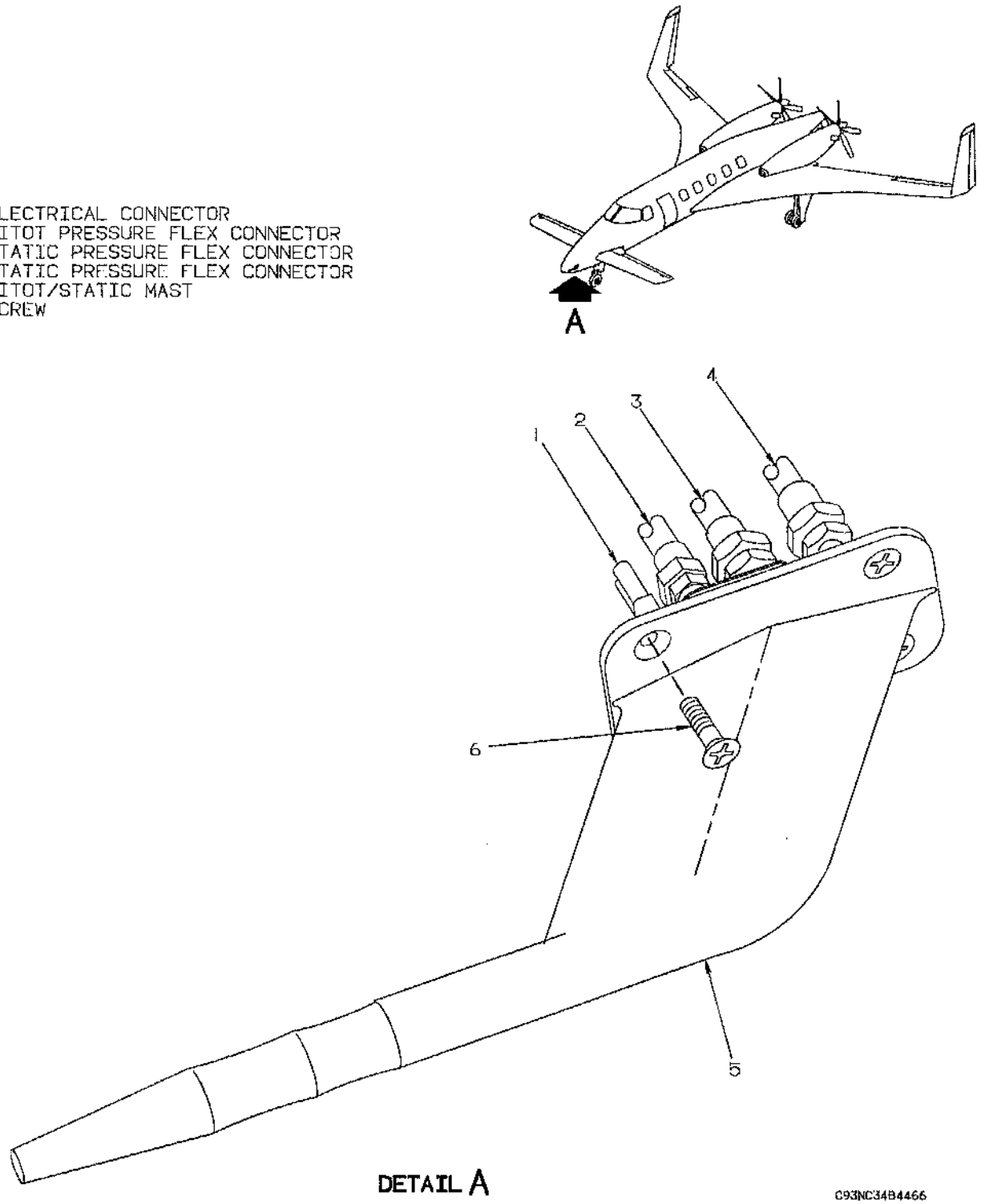
STANDBY ALTIMETER REMOVAL (Effectivity: All) (Figure 5)

a. Loosen, but do not remove the screws (2) at the four corners of the indicator (1). The screws are required to retain the clamp (5) in place.

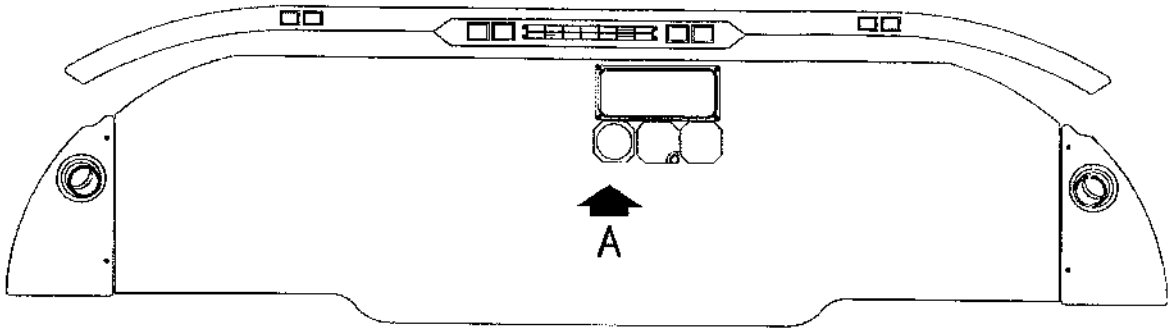
b. Pull the indicator out of the instrument panel until it clears the mounting hole.

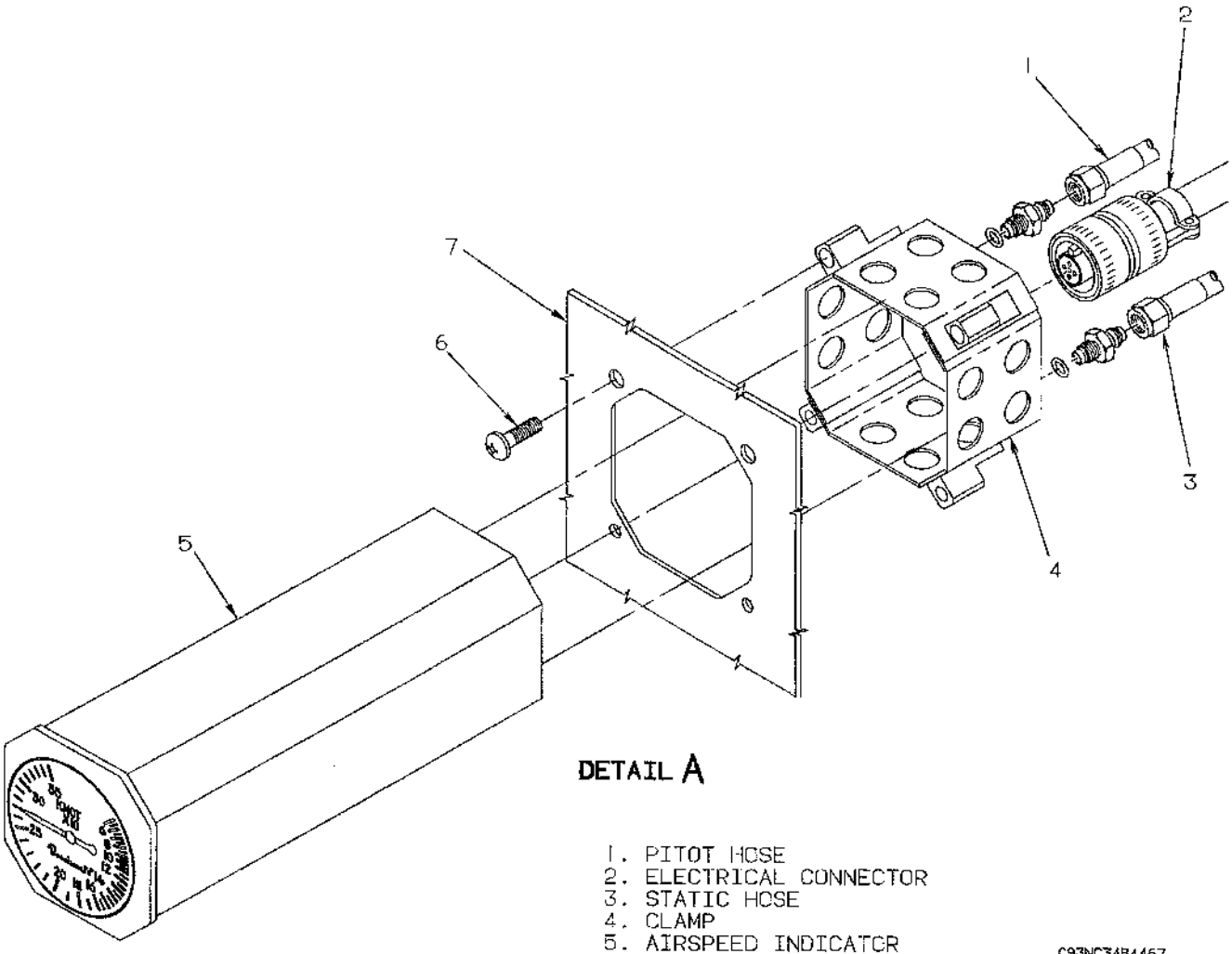
c. Identify, tag, and disconnect the electrical connector (3) and the static hose (4).

1. ELECTRICAL CONNECTOR
2. PITOT PRESSURE FLEX CONNECTOR
3. STATIC PRESSURE FLEX CONNECTOR
4. STATIC PRESSURE FLEX CONNECTOR
5. PITOT/STATIC MAST
6. SCREW



Pitot/Static Mast Installation (Effectivity: All)
Figure 3





Standby Airspeed Indicator Installation (Effectivity: All)
Figure 4

NOTE

If the electrical or pitot/static line connections are not long enough to allow the indicator to clear the panel, remove the center reversionary panel directly above the indicator to access and disconnect the lines from the indicator. Refer to Chapter 31-10-00 for reversionary panel removal procedures.

d. Cap, and stow all disconnected lines.

e. Cap the fittings on the back of the indicator (1). Remove the indicator to the workbench or a safe storage area.

STANDBY ALTIMETER INSTALLATION (Effectivity: All) (Figure 5)

a. Remove the caps from the back of the indicator (1).

b. Connect the static hose (4) and electrical connector (3) to the indicator (1).

c. Position the indicator (1) in the instrument panel (6) and secure with the screws (2).

d. Install the center reversionary panel if it was removed. Refer to Chapter 31-10-00 for installation procedures.

e. Perform a pitot/static leak check. Refer to PITOT/STATIC PRESSURE SYSTEM INSPECTION AND LEAK TEST in this chapter.

AIR DATA COMPUTER AND AIR DATA MODULE REMOVAL (Effectivity: All) (Figure 6)

Determine which unit is to be removed from the airplane. The No. 1 Air Data Computer (ADC) is on the upper left avionics equipment shelf. The No. 2 ADC is on the upper right avionics equipment shelf. The Air Data Modules are mounted on top of the respective ADC's.

a. Remove all electric power from the airplane.

b. Refer to Chapter 53-30-00 and remove the appropriate nose avionics access door (214AR or 214AL).

c. Loosen the retaining knobs (6) on the front of the air data computer (7).

d. Disconnect the pitot and static hose connectors (3 and 4) from the air data computer (7) and remove the drain line plumbing (5) from the ADC.

e. Pull the unit from the self-aligning connector. Remove the unit to the workbench or a safe storage area.

NOTE

If the air data computer (7) is to be replaced, unplug the air data module (8) from the top of the air data computer case and plug it into the new unit.

AIR DATA COMPUTER AND AIR DATA MODULE INSTALLATION (Effectivity: All) (Figure 6)

NOTE

If the air data module (8) was removed, install by pluging it into the top of the air data computer (7).

a. Place the unit in the rack and seat in the connector.

b. Latch the retaining knobs (6) to the unit, and hand tighten.

c. Connect the pitot and static hose connectors (3 and 4) to the air data computer (7).

d. Install the drain line plumbing (5) on the ADC.

e. Restore electric power to the airplane.

f. Test the system for proper operation and perform the pitot/static leak test. Refer to the Collins Starship 1 Avionics System Manual and perform the level 2 troubleshooting procedure.

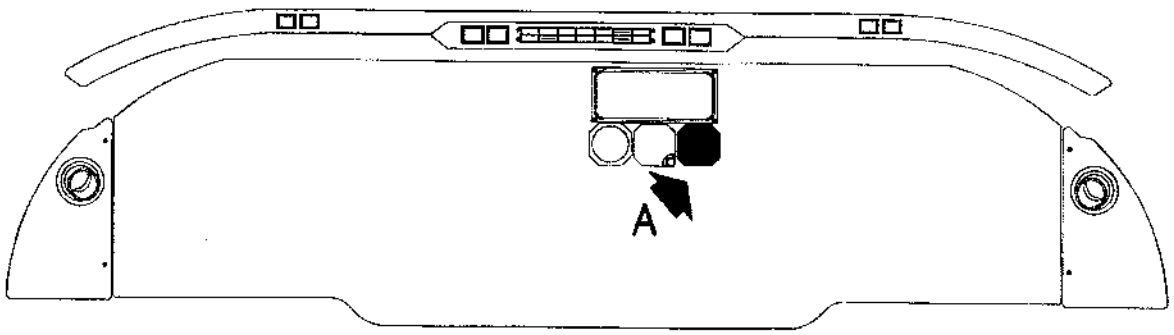
g. Refer to Chapter 53-30-00 and install the appropriate nose avionics access door (214AR or 214AL).

OUTSIDE AIR TEMPERATURE SENSOR REMOVAL (Effectivity: All) (Figure 7)

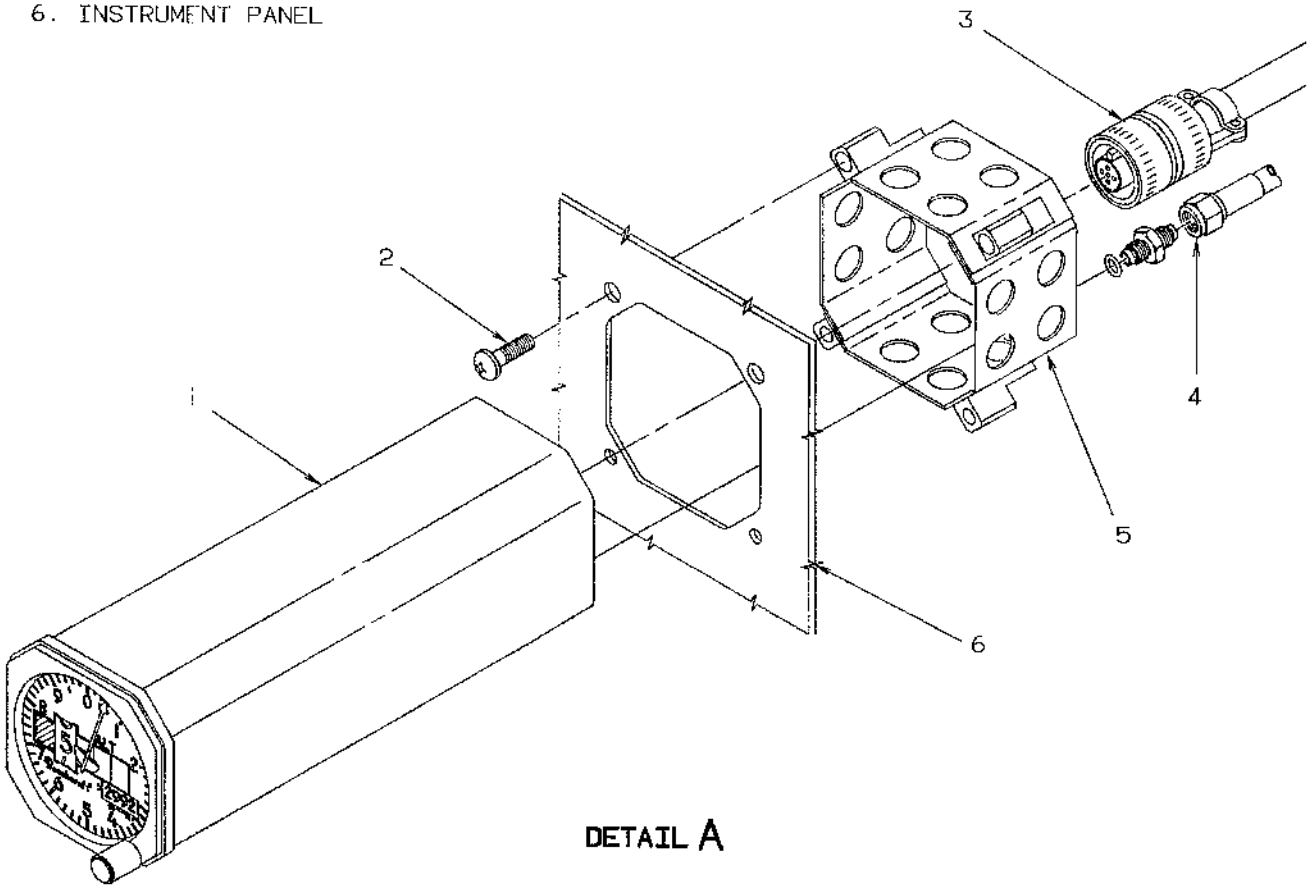
a. Operate the alternate extension hand pump in the cockpit until the forward nose landing gear doors are fully open. Refer to Chapter 32-20-00 and install the lockpin in the landing gear nose gear door bellcrank.

b. Remove the appropriate nose wheelwell access panel (710AZ or 710BZ).

c. Remove the electrical connector (5) from the temperature sensor (3).



1. ALTIMETER
2. SCREW
3. ELECTRICAL CONNECTOR
4. STATIC HOSE
5. CLAMP
6. INSTRUMENT PANEL



DETAIL A

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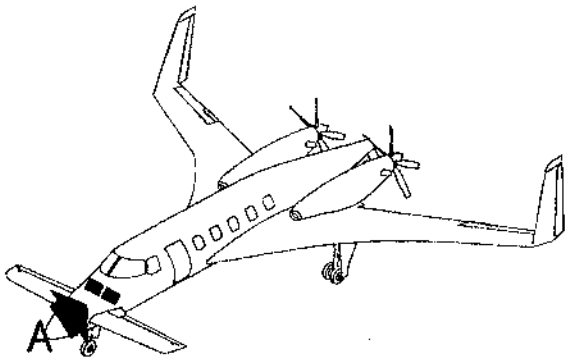
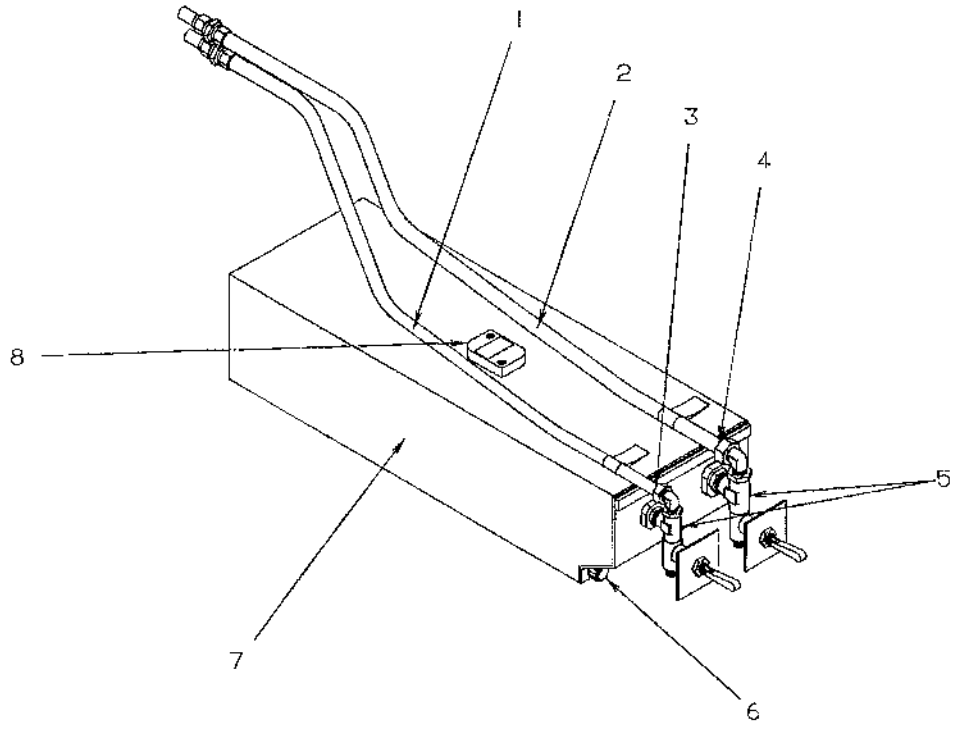
Standby Altimeter Installation
(Effectivity: All)
Figure 5

1. PITOT AIR HOSE
2. STATIC AIR HOSE
3. PITOT HOSE CONNECTOR
4. STATIC HOSE CONNECTOR
5. DRAIN PLUMBING
6. RETAINING KNOB
7. AIR DATA COMPUTER
8. AIR DATA MODULE





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Air Data Computer And Air Data Module Installation (Effectivity: All)
Figure 6

d. Remove the screws (4) that secure the sensor (3).

e. Remove the sensor (3) and cap the electrical connector.

OUTSIDE AIR TEMPERATURE SENSOR INSTALLATION (Effectivity: All) (Figure 7)

a. Remove all sealer from the OAT probe flange (2) and the nose (1) installation area using a nonmetallic scraper and nonmetallic cleaner pad.

b. After positioning the air temperature sensor (3) in the airplane, ensure that the leading edge sensor holes are facing forward then apply sealer (2, Chart 2, 34-00-00) to the screws (4). Install the screws (4) to secure the sensor (3).

c. Connect the electrical connector (5) to the sensor (3).

d. Test the circuit by comparing the temperature readout on both airspeed indicators with the air data computers on.

e. Seal around the periphery of the probe base with sealer, (1, Chart 2, Chapter 34-00-00).

f. Install the appropriate nose wheelwell access panel (710AZ or 710BZ).

g. Remove the landing gear door lockpin and push the doors closed.

PITOT/STATIC PRESSURE SYSTEM INSPECTION AND LEAKAGE TEST (Effectivity: All)

The purpose of this inspection and test is to verify that no leakage exists in the pitot/static system that could cause a significant error in indications. The test operator must be familiar with the system before making the test. Do not disconnect or make a break in the plumbing unless a leak in the system or instrument case requires disconnection. Doing so will result in an untested point and require a retest of the system.

CAUTION

When removing any components of the pitot/static system, note the location of all bonding jumpers to aid in reinstallation.

TEST PREPARATION (Effectivity: All) (Figure 8)

a. Refer to chapter 24-40-00 and connect and apply external electrical power.

b. Turn on the PILOT, COPILOT and EICAS integrated avionics switches.

c. Ensure that the STATIC SOURCE select switch is in the NORM position.

d. Inspect the pitot/static system plumbing to ensure that all fittings are secure and that all hoses and lines are free from deterioration or damage.

e. Place the pitot/static test box (1, Chart 1, 34-00-00) in the cockpit.

f. Connect a compressed air source to the test box and adjust the air pressure for 90 to 110 psi.

g. On the mast adapters (2, Chart 1, 34-00-00) turn the compression nuts counterclockwise against the stops.

h. Install a mast adapter on the pilot's pitot/static mast by using a slight back and forth motion to position the mast adapter on the mast until the tip of the mast contacts the soft snubber on the mast adapter.

NOTE

Each mast adapter must be installed with the test fittings facing downward as far back on its respective pitot/static mast as possible.

i. Hold the mast adapter in position and tighten the compression nut clockwise.

j. Install the other mast adapter on the copilots pitot/static mast using the same method.

PITOT TEST (Effectivity: All) (Figure 8)

a. Connect the test hoses to the pilot's mast adapter and to the test box.

b. Cap the ports on the copilot's mast adapter.

c. Open all of the static drain valves for this test.

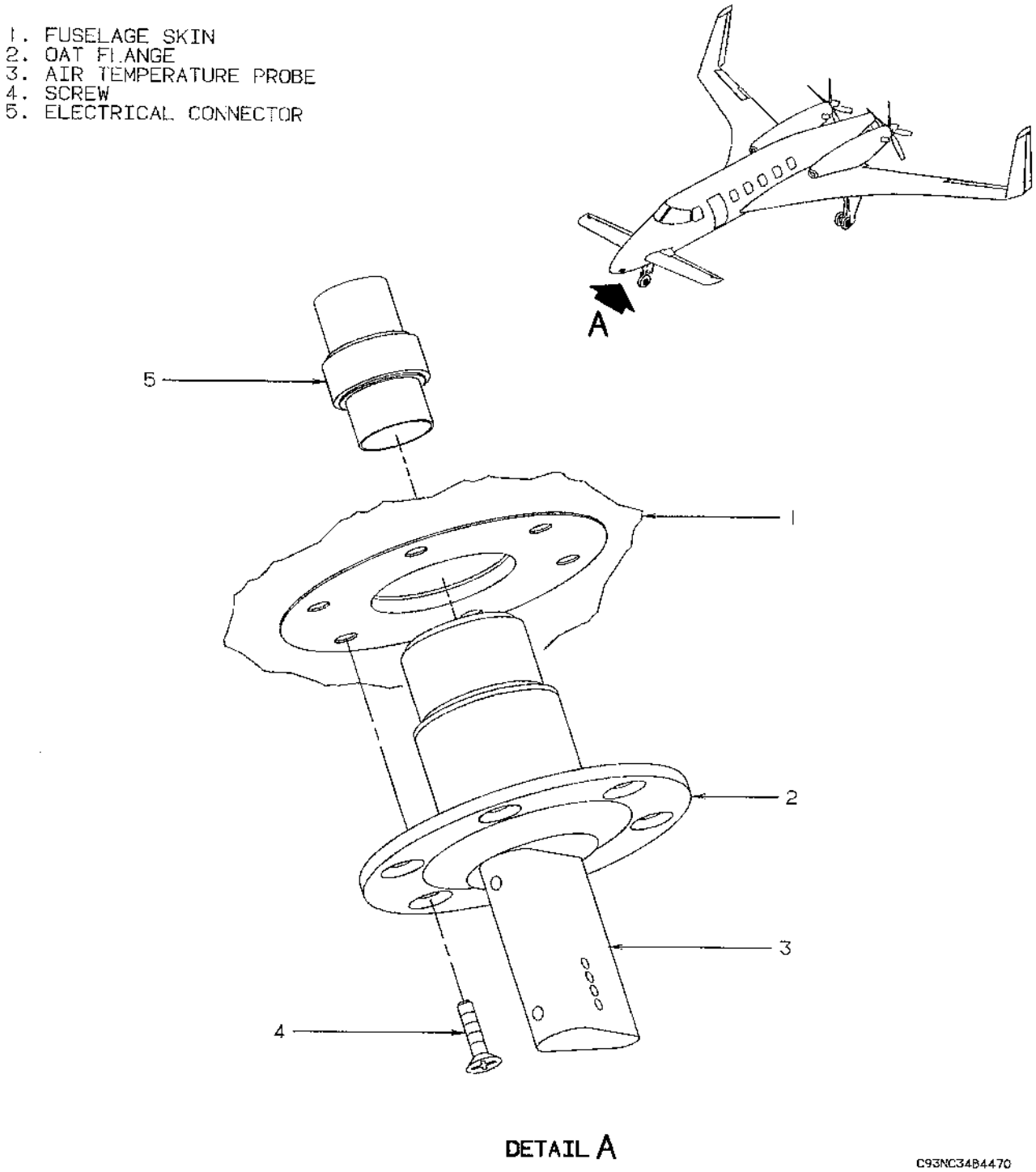
d. Close the coupler on the test box.

e. Slowly open the pressure needle valve to increase pressure in the pilot's pitot system until the pilot's ASI display reads 90% of full scale.

f. Close the pressure needle valve on the test box to isolate the pitot system and observe the pilot's ASI display. There shall be no evidence of leakage over a five-minute period.

CAUTION

Bleed the pressure slowly to avoid damage to the airspeed indication system.



Outside Air Temperature Sensor Installation (Effectivity: All)
Figure 7

g. Open the pressure bleed valve and slowly reduce the pressure.

h. Unplug the ports on the copilot's mast adapter.

i. Disconnect the test hoses from the pilot's mast adapter and connect them to the copilot's mast adapter.

j. Cap the ports on the pilot's mast adapter.

k. Repeat the test for the copilot's pitot system using the copilot's ASI display and the standby pneumatic airspeed indicator.

l. If there is any evidence of leakage in either system within the five-minute period, check the instrument and test connections and perform the test again. If leakage is still evident, suspect instrument case leakage.

CAUTION

When installing components of the pitot/ static system, ensure that all electrical bonding jumpers are returned to thier original locations and that the electrical bonding requirements specified in Chapter 20-00-03 are met.

NOTE

When any component is replaced, or there is any break in the system, the test must be repeated for that portion of the system so that there are no untested points in the system.

PITOT/STATIC TEST (Effectivity: All)
(Figures 8 and 9)

a. Connect the test box hoses to the pilot's and copilot's mast adapter pitot test ports as shown on the test schematic in Figure 8.

b. Connect the test box hoses to both of the copilot's mast adapter static tests ports as shown on the test schematic in Figure 8.

c. Cap the static test ports on the pilot's mast adapter.

d. Set the standby pneumatic alltimeter to 29.92 inches of hg. and note the indicated field elevation. Refer to the conversion graph in Figure 9 to determine the maximum acceptable leakage rate (maximum loss in feet) for the maximum pressure differential altitude.

NOTE

The coupler valve on the test box can be adjusted to delay the pressure change in the pitot system, causing both ASI displays and the standby pneumatic airspeed indicator to upscale to nearly half scale.

e. Slowly open the suction valve on the test box while observing the pilot's and copilot's ALI displays for response.

CAUTION

Excessive leakage in the static system will cause the standby pneumatic airspeed indicator to run backwards and damage the indicator.

f. Make a preliminary leakage-rate check to detect any major leakage. When 1000 feet above indicated field elevation is attained, close the suction valve and observe the standby pneumatic altimeter for 30 seconds. Verify that the system does not leak.

g. After verifying there are no leaks, begin increasing suction to attain the maximum pressure differential altitude noted in step d.

h. When the maximum pressure-differential altitude is attained, the pitot and static pressures will become equal and the airspeed will return to zero.

i. Close the suction valve on the test box.

j. Observe the standby pneumatic altimeter for one minute, and verify leakage does not exceed the maximum altitude loss in feet as specified by the conversion graph in Figure 9.

CAUTION

Bleed the pressure slowly to avoid damage to the indication system

Never disconnect any plumbing until the standby pneumatic altimeter has returned to the indicated field elevation and the pressure has equalized.

k. Use the pressure bleed valve to allow ambient air to slowly enter the system. Bleed very slowly while constantly monitoring the standby pneumatic instruments to avoid damage.

CAUTION

When reinstalling components of the pitot/static system, ensure that all electrical bonding jumpers are returned to their original locations and that the electrical bonding requirements specified in Chapter 20-00-03 are met.

If leakage in either system exceeds the limits specified in Figure 9, recheck the test connections and retest the system. If leakage is still evident, suspect instrument case leakage. In this case, remove the instruments one at a time, cap the associated plumbing and perform the leakage test again using the instruments provided on the face of the test box until the leak is isolated.

NOTE

If any components have been removed, perform another test of the system after reinstallation so that there are no untested points.

ISOLATED STATIC SOURCE TEST
(Effectivity: All)

a. Connect the test hoses from the pitot/static tester to both the pitot and static ports on the pilot's mast adapter.

b. Close the pitot/static system solenoid valves by placing the STATIC SOURCE select switch in the ISOLATED position.

c. Open the copilot's static drain ports and repeat steps e thru l of the PITOT/STATIC TEST outlined in this chapter, with the exception of observing the pilot's ALI display only.

d. Verify that the pilot's static drain valves shut off completly.

e. After the test has been completed and the pilot's ALI has returned to indicated field elevation, connect the test hoses to the pitot and static ports on the copilot's mast adapter.

f. Close the copilot's static drain ports.

g. Open the pilot's static drain ports and repeat steps e thru l of the PITOT/STATIC TEST outlined in this chapter, with the exception of observing both copilot's ALI display and the standby pneumatic altimeter only.

h. Verify that the copilot's static drain valves shutoff completely.

i. After the test is complete and both the standby pneumatic altimeter and the copilot's ALI display have returned to indicated field elevation, open the solenoid valves by placing the STATIC SOURCE switch in the NORM position.

j. Close the pilot's static drains.

k. Disconnect the test equipment.

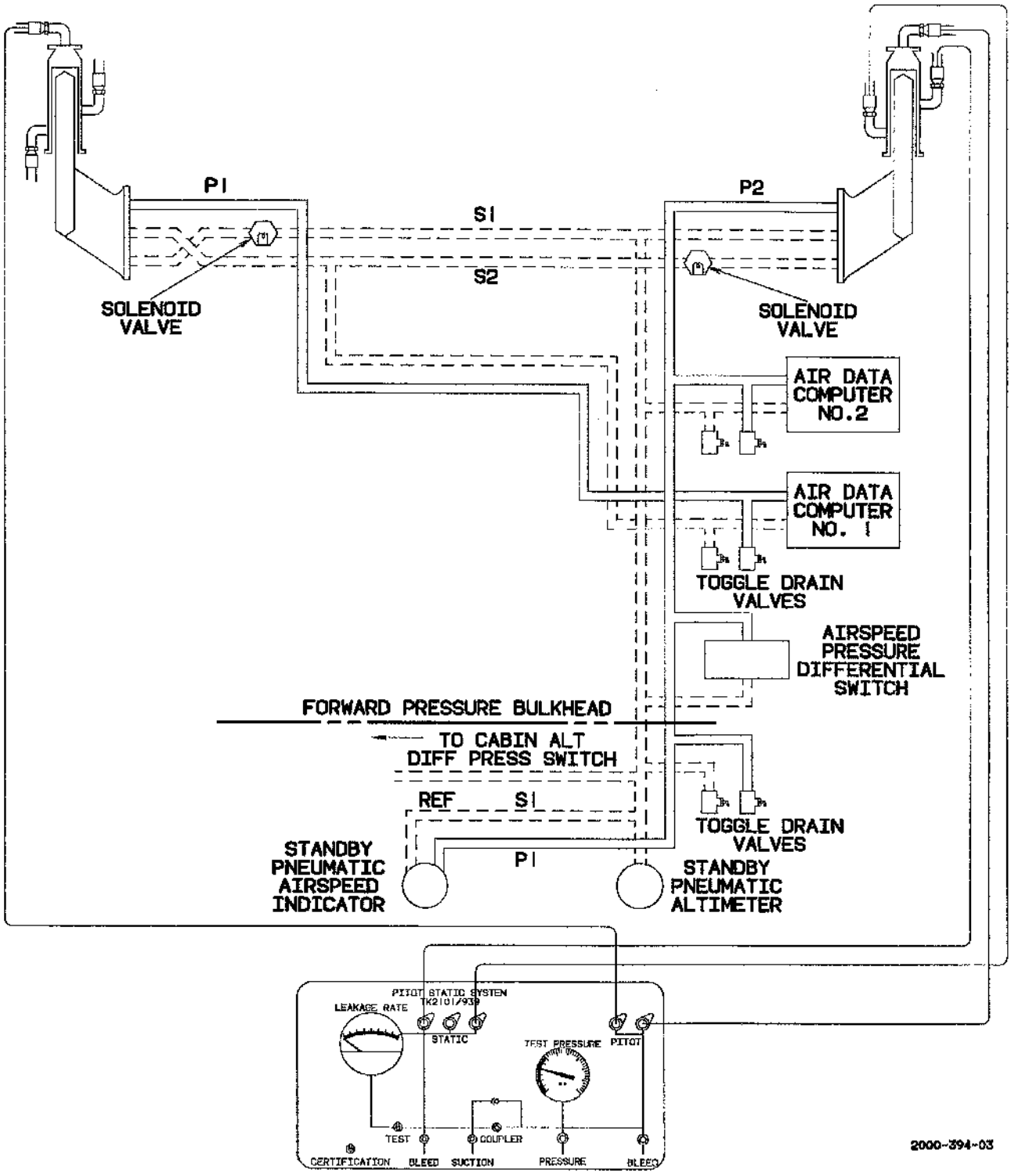
CAUTION

When reinstalling components of the pitot/static system, ensure that all electrical bonding jumpers are returned to their original locations and that the electrical bonding requirements specified in Chapter 20-00-03 are met.

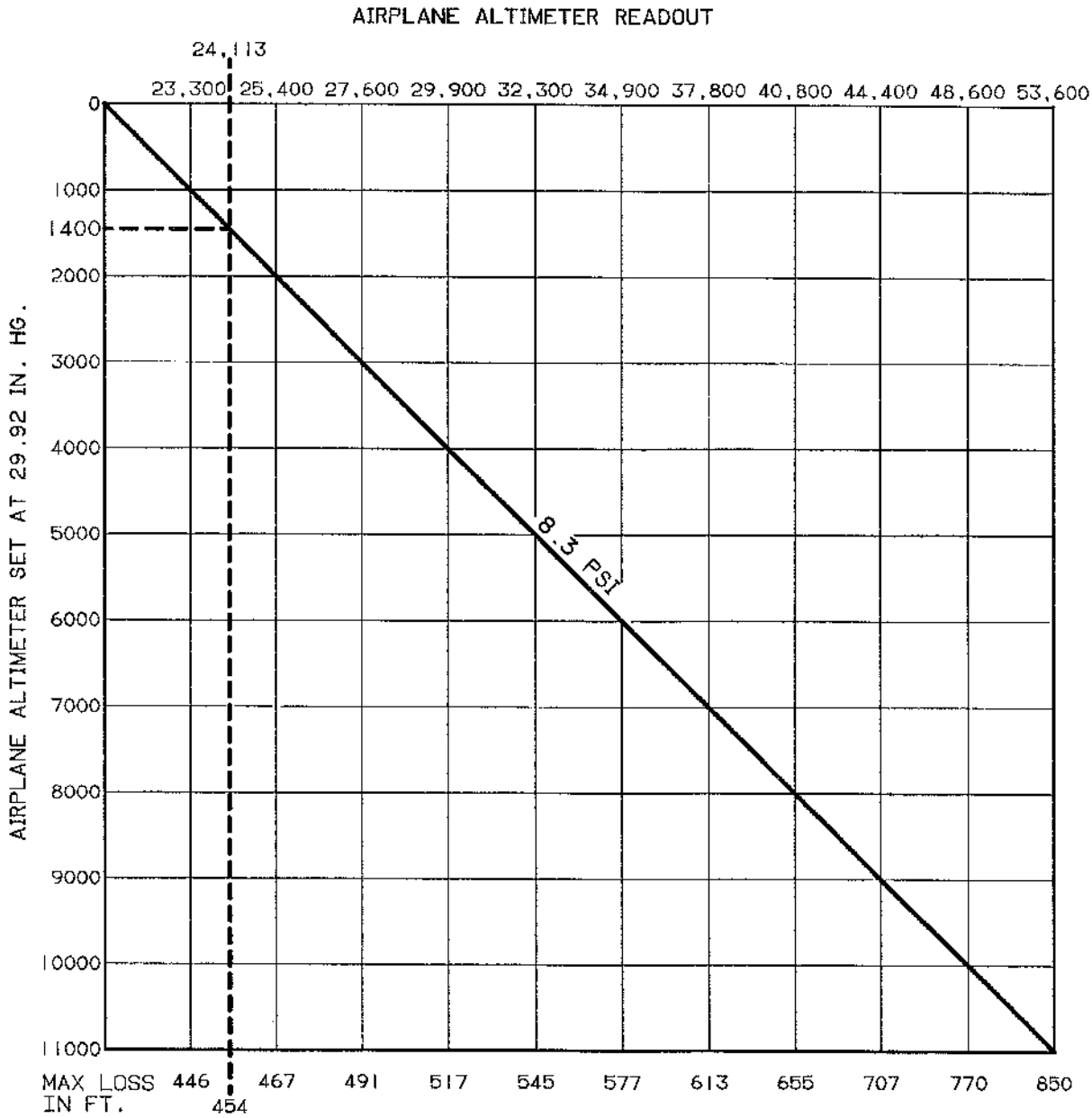
If leakage in either system exceeds the limits specified in Figure 9, isolate the faulty drain valve and repair or replace as required.

NOTE

If any components are removed, perform another test after reinstallation so that there are no untested points in the system.



Pitot/Static Test Schematic (Effectivity: All)
Figure 8



Maximum Pressure Differential Altitude Conversion Graph (Effectivity: All)
Figure 9

ATTITUDE & DIRECTION - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only execptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position.Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

MAGNETIC COMPASS (STANDBY) (Effectivity: All)

The magnetic compass is a simple "whiskey" compass that provides a backup for the electronic heading indications. The magnetic compass is located on top of the escutcheon at the center of the windshield.

ATTITUDE-HEADING COMPUTER (AHC-85D) (Effectivity: All)

The attitude-heading computer provides the roll, pitch, and stabilized magnetic heading data for the displays and the Automatic Flight Control System (AFCS). Instead of the traditional high speed gyros and synchro-transmitters, the computer uses relatively slow turning motion detectors with piezoelectric transducers. Airplane roll and pitch motions cause the motion detectors to exert gyroscopic forces on the transducers. The transducers convert these forces to voltages. The computer continuously sums the voltages to produce current attitude information. Magnetic heading inputs come from remotely mounted flux detectors. IAS and VS inputs come from the air data computer system. The two attitude heading computers are located in the nose avionics compartments.

INTERNAL COMPENSATION UNIT (ICU-85) (Effectivity: All)

The ICU calibrates the flux detector outputs for an individual airplane. It is mounted on the attitude heading computer case and should remain with the airplane if the attitude heading computer is replaced.

FLUX DETECTOR UNIT (FDU-70) (Effectivity: All)

The flux detectors are located on the underside of the aft wing outboard of each main landing gear. The flux detectors sense the horizontal component of the earth's magnetic field and generate output signals proportional to the airplane's magnetic heading.

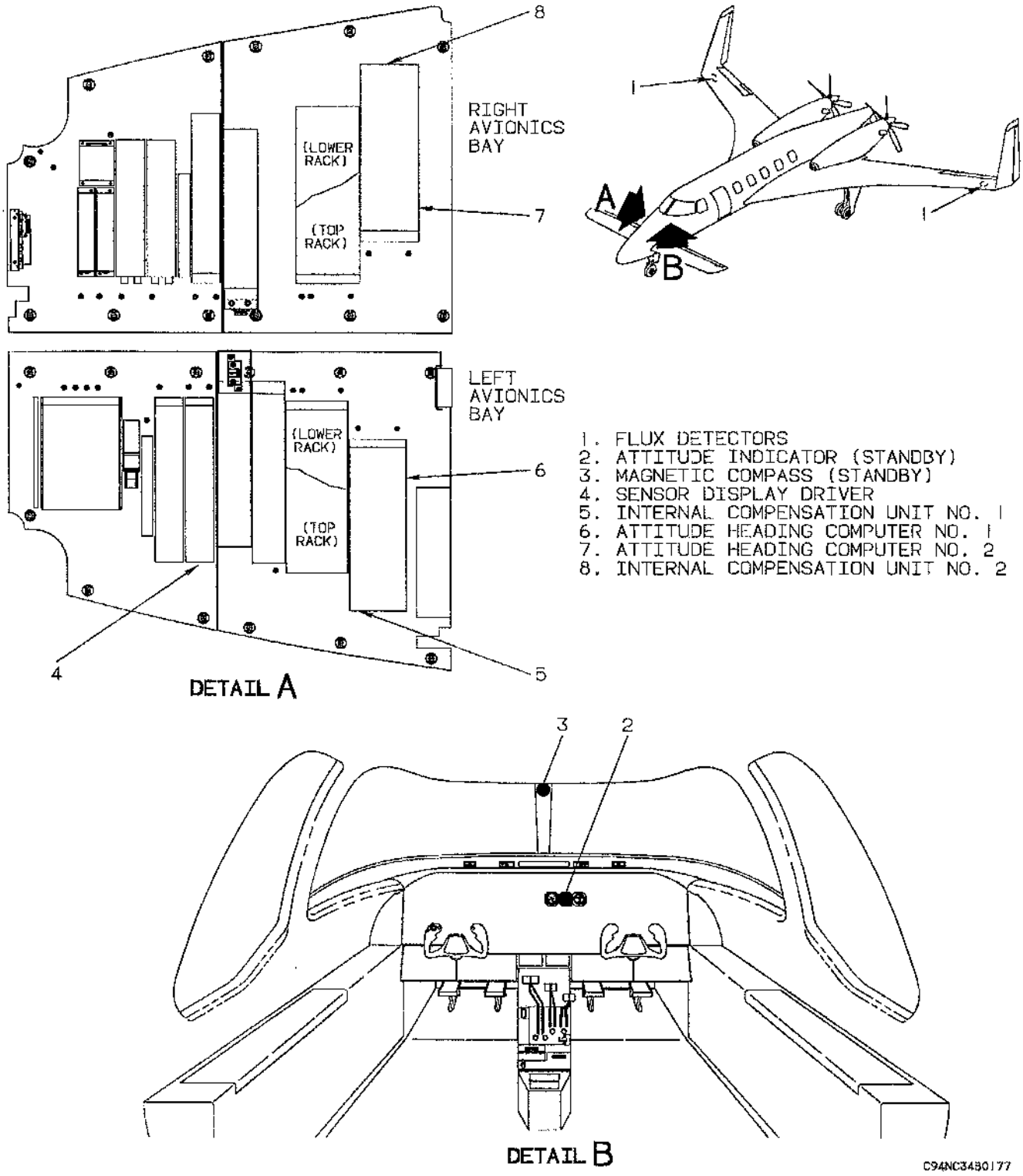
SENSOR DISPLAY DRIVER (SDD-640) (Effectivity: All)

The sensor display driver provides information to both of the sensor display units. It provides selection control from the available raw inputs that are provided from the instrumentation on the airplane. These inputs include: AHRS, VOR/ILS, DME, ADF, VLF, and the Integrated Avionics Processing System (IAPS). The IAPS provides flight management system data.

The sensor display driver is located in the left nose avionics compartment.

ATTITUDE INDICATOR (STANDBY) (Effectivity: All)

The standby attitude indicator is a two-inch internal gyroscopic indicator. It can operate on the standby attitude gyro auxiliary battery during a complete electrical failure, and maintain ±6° accuracy for approximately nine minutes after removal of all electrical power including the ship's battery. Refer to Chapter 24-32-00 for more information regarding the auxiliary battery.



Attitude And Direction System (Effectivity: All)
Figure 1

ATTITUDE & DIRECTION - TROUBLESHOOTING PROCEDURES (Effectivity: All)

ATTITUDE-HEADING COMPUTER (AHD-85D) AND INTERNAL COMPENSATION UNIT (ICU-85) (Effectivity: All)

NOTE

The two AHC's may be swapped with each other to verify airplane wiring and to isolate a failed unit. If the AHC is swapped or replaced, the ICU remains with the mount. If the ICU is replaced, a compass swing MUST be performed. Refer to the Attitude Heading Computer installation section of the Collins Manual 523-0772305 for compass swing procedures.

Refer to the level-2 AHC/ICU troubleshooting procedure in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section to troubleshoot the AHC1, AHC2, ICU1, and ICU2 units.

FLUX DETECTOR UNIT (FDU-70) (Effectivity: All)

NOTE

If the FDU is swapped or replaced, a compass swing MUST be performed. Refer to the Attitude Heading Computer installation section of the Collins Manual 523-0772305 for compass swing procedures. If any FDU mounting hardware is replaced, ensure that new hardware is made of nonmagnetic material.

Refer to the level-2 FDU troubleshooting procedure in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

SENSOR DISPLAY DRIVER (SDD-640) (Effectivity: All)

NOTE

The SDD does not report to the AVIONICS STATUS page; however, if a faulty SDD is suspected, refer to the level-2 SDD troubleshooting procedure in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

Further testing of the SDD may be indirectly accomplished by performing the SDU test procedures.

ATTITUDE & DIRECTION - MAINTENANCE PRACTICES (Effectivity: All)

STANDBY COMPASS CALIBRATION (Effectivity: All)

Correction of the compass card for magnetic error is accomplished by removing the correction card plate and adjusting the compensator with a nonmagnetic screwdriver. Calibrate the standby compass to a compass rose. Compass calibration is performed under the following conditions:

- Both engines are running.
- All avionics are ON.
- Forward blower is on HIGH.
- Both generators are ON.
- Windshield electric heat is OFF.
- Windshield wipers are OFF.

a. Position the airplane to magnetic headings at 30° intervals.

b. In each position, the standby magnetic compass should read within ±10° of actual airplane magnetic heading. Replace the magnetic compass if the deviation exceeds 10° on any of the headings.

c. For each position, record the standby magnetic compass reading under the actual heading on the compass correction plate by using a steel stamp.

ELECTROMAGNETIC INTERFERENCE CHECK (Effectivity: All)

CAUTION

Care should be taken to prevent overheating the windshield during this check.

a. Position airplane to north or south magnetic heading. Switch right windshield heat to high.

b. The standby compass deflection should not be more than 5°.

c. Switch right heat OFF and switch left windshield heat to HIGH.

d. The standby compass deflection should not be more than 5°.

e. Operate right and left windshield switches in the HIGH position. The switches are on the pilot's inboard subpanel.

f. The standby compass deflection should not be more than 10°.

STANDBY MAGNETIC COMPASS REMOVAL (Effectivity: All) (Figure 2)

a. Remove the glareshield from the airplane. Refer to Chapter 25-10-00 for removal procedures.

b. Remove screws (7) securing the compass cover (escutcheon) (6) the center post (1).

c. Disconnect electrical connector (8).

d. Remove the screws (3) from the mounting bracket (2) and remove the compass (5).

CAUTION

To prevent permanent damage to the compass, handle with care, and do not place near a strong magnetic source.

STANDBY MAGNETIC COMPASS INSTALLATION (Effectivity: All) (Figure 2)

a. Position the compass (5) on the mounting bracket (2) and install the mounting screws (3).

b. Connect the electrical connector (8).

c. Position the escutcheon (6) to the center post (1) and secure with the screws (7).

d. Install the glareshield. Refer to Chapter 25-10-00 for installation procedures.

e. Perform the standby compass calibration procedure.

ATTITUDE HEADING COMPUTER REMOVAL (Effectivity: All) (Figure 3)

When the Attitude Heading Computer (AHC) is removed for maintenance, the Internal Compensator Unit (ICU) must remain with the mount. AHC 1 is located in the left avionics compartment and AHC 2 is located in the right avionics compartment.

a. Remove all electric power from the airplane.

b. Refer to Chapter 53-30-00 and remove the appropriate nose avionics compartment door (213AL or 214AR).

c. Remove the two screws (3) that attach the ICU (2) to the AHC (1).

d. Pull the ICU (2) away from the AHC (1), and remove the electrical connector from the ICU.

CAUTION

Care should be taken to avoid damaging the exposed pins on the ICU.

e. If the AHC is to be replaced, remove the plastic pin connector cover from the replacement unit and remove the ribbon cable from the plastic dummy connector.

f. Attach the AHC cable from the unit to be removed to the dummy connector on the plastic panel that is taken from the replacement AHC.

g. Stow the ribbon cable in the AHC cavity and install the plastic panel with screws to the AHC unit.

h. Move the ICU (2) to one side.

i. Loosen the retaining knobs (4) on the latches and swing the latches away from the AHC (1).

j. Pull the AHC (1) free from the connector.

k. Remove the AHC (1) to the workbench or a safe storage area.

ATTITUDE HEADING COMPUTER INSTALLATION (Effectivity: All) (Figure 3)

a. Place the AHC (1) in the rack (6) and firmly seat in the connector.

b. Attach the ribbon cable to the connector on the ICU (2).

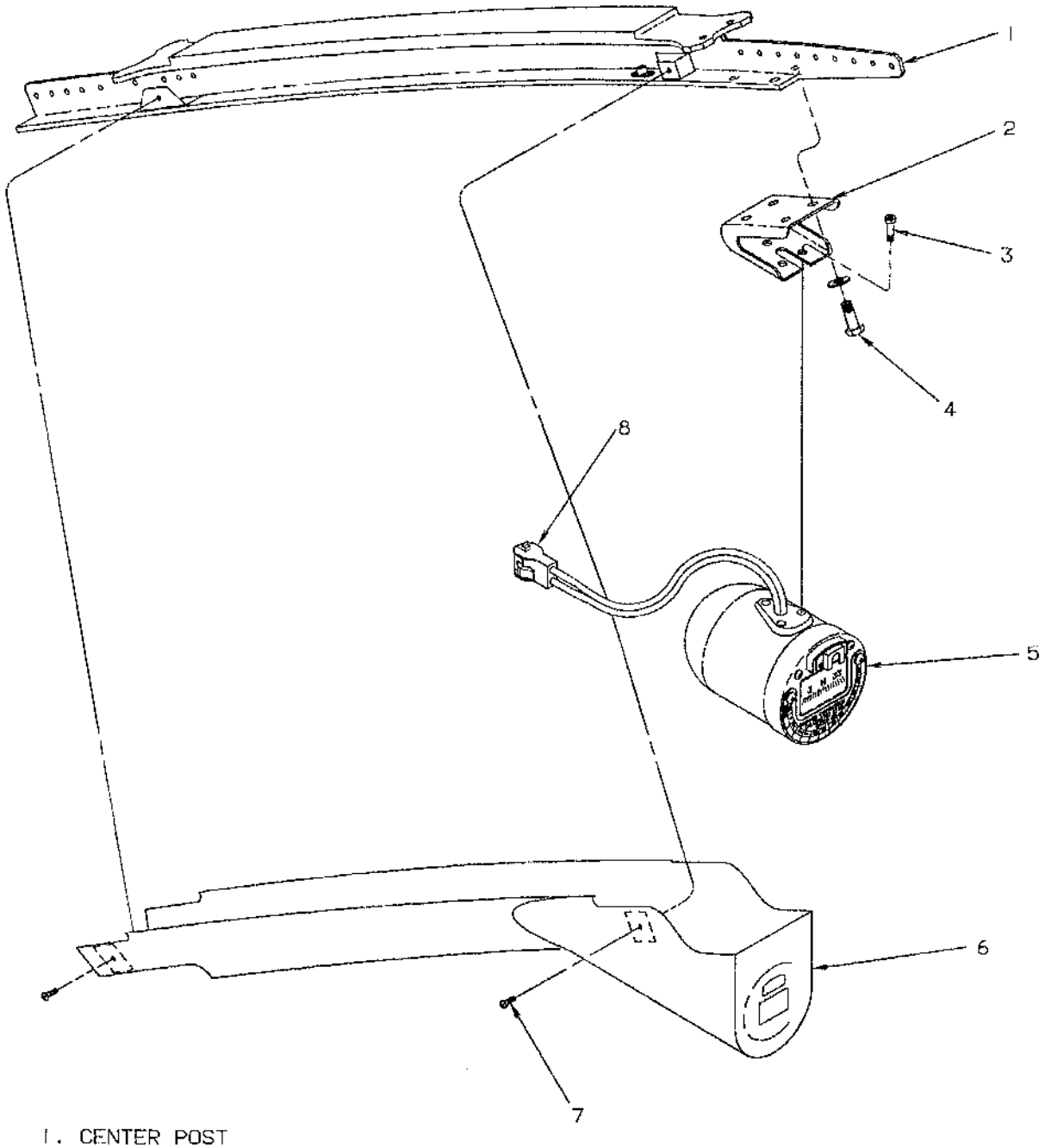
c. Secure the ICU (2) to the AHC (1).

d. Secure the latch retaining knobs (4) on the front of the AHC (1) and hand tighten.

e. Restore electical power to the airplane.

f. Apply power to all the avionics systems and wait five minutes for the diagnostics system to stabilize.

g. Refer to the level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section and perform the level-2 test procedures.



1. CENTER POST
2. BRACKET
3. SCREW
4. BRACKET SCREW
5. COMPASS
6. ESCUTCHEON
7. SCREW
8. ELECTRICAL CONNECTOR

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Standby Compass Installation (Effectivity: All)
Figure 2

h. Refer to Chapter 53-30-00 and install the appropriate nose avionics compartment door (213AL or 214AR).

INTERNAL COMPENSATION UNIT (ICU) REMOVAL (Effectivity: All) (Figure 3)

The ICU is mounted on the Attitude Heading Computer (AHC).

a. Remove all electric power from the airplane.

b. Refer to Chapter 53-30-00 and remove the appropriate nose avionics compartment door (213AL or 214AR).

c. Remove the ICU (2) from the front cavity of the AHC (1) by removing two screws (3) from the front panel of the computer.

d. Disconnect the ribbon cable from the connector on the ICU and remove the retaining strap from the ICU.

e. Remove the ICU (2) to the workbench or a safe storage area.

INTERNAL COMPENSATOR UNIT INSTALLATION (Effectivity: All) (Figure 3)

a. Attach the ribbon cable from the AHC to the ICU connector.

b. Attach the retaining strap under the bottom screw.

c. Install the ICU (2) into the cavity in the front of the AHC and install the screws (3).

d. Restore electrical power to the airplane.

e. Apply power to all the avionics systems and wait 5 minutes for the diagnostics system to stabilize. Refer to the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section and perform the level-2 test procedures.

NOTE

If the ICU is replaced, a compass swing MUST be performed. Refer to the Attitude Heading Computer installation section of the Collins Manual 523-0772305 for compass swing procedures.

f. Refer to Chapter 53-30-00 and install the appropriate nose avionics compartment door (213AL or 214AR).

FLUX DETECTOR UNIT (FDU) REMOVAL (Effectivity: All) (Figure 4)

CAUTION

When working on or near the flux detector, always use nonmagnetic tools to prevent interference with the calibration of the flux detector transmitter. To prevent damage to the sensor, do not use an ohmmeter on the flux detector.

Use only approved screws for installation of the flux detector. Using improper fasteners may damage the flux detector. The correct nonferrous screw is MS35214-27 and washer AN960-B6.

NOTE

The No.1 flux detector is located in the left wing and the No.2 flux detector is located in the right wing.

a. Refer to Chapter 57-00-00 and remove the access panel 546AB (left wing) or 646AB (right wing) to gain access to the flux detector.

b. Disconnect and cap the electrical connector (4).

c. With a nonmagnetic tool, remove the screws (5) and washers (6) that secure the flux detector (7) to the bracket (1).

d. Lower the flux detector (7) from the bracket (1).

CAUTION

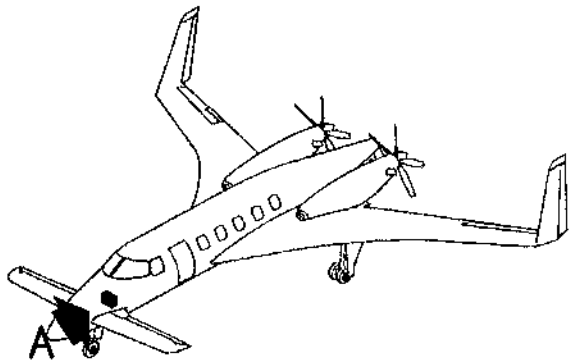
The flux detector is a very sensitive device. Handle the transmitter with care to prevent damage to the internal mechanism.

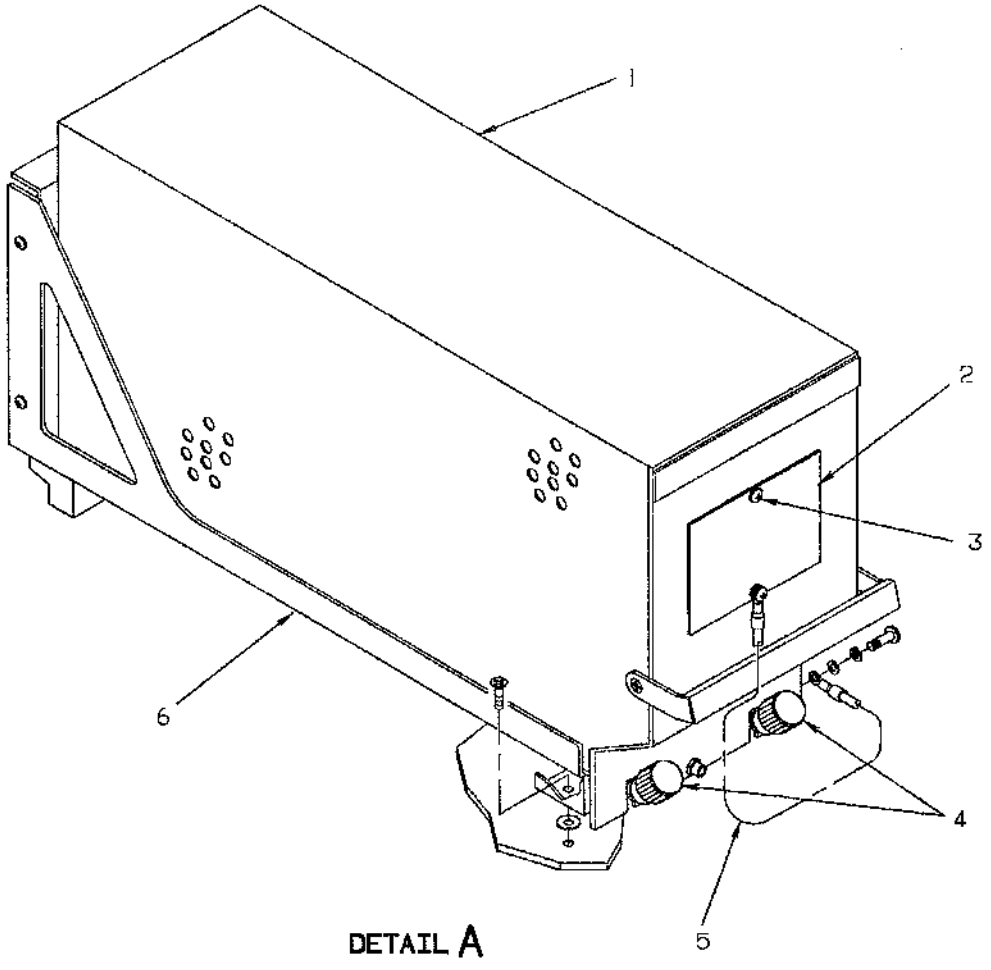
FLUX DETECTOR UNIT (FDU) INSTALLATION (Effectivity: All) (Figure 4)

a. Thread the flux detector wire harness (3) through the hole in the bracket (1) and secure the flux detector (7) to the bracket (1) with approved nonferrous screws (5) and washers (6).

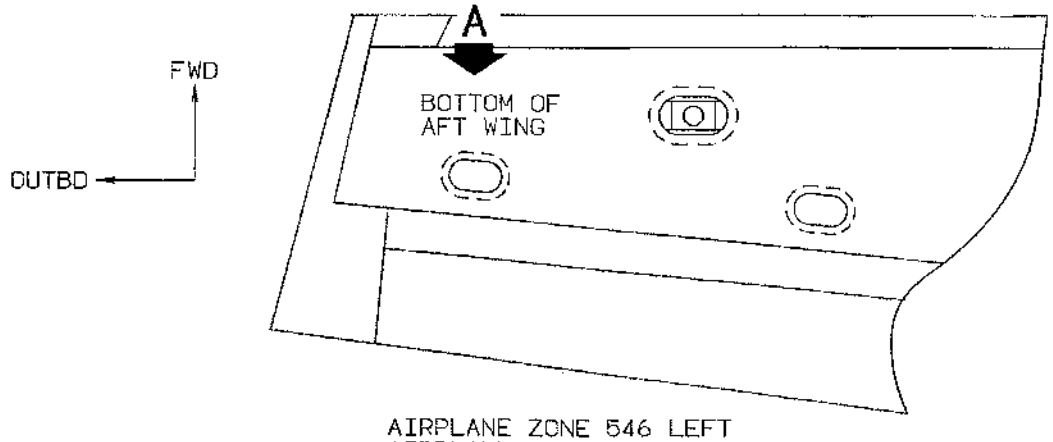
b. Connect the electrical connector (4).

1. ATTITUDE HEADING COMPUTER
2. INTERNAL COMPENSATION UNIT
3. SCREWS
4. RETAINING KNOBS
5. ICU RETAINING STRAP
6. RACK

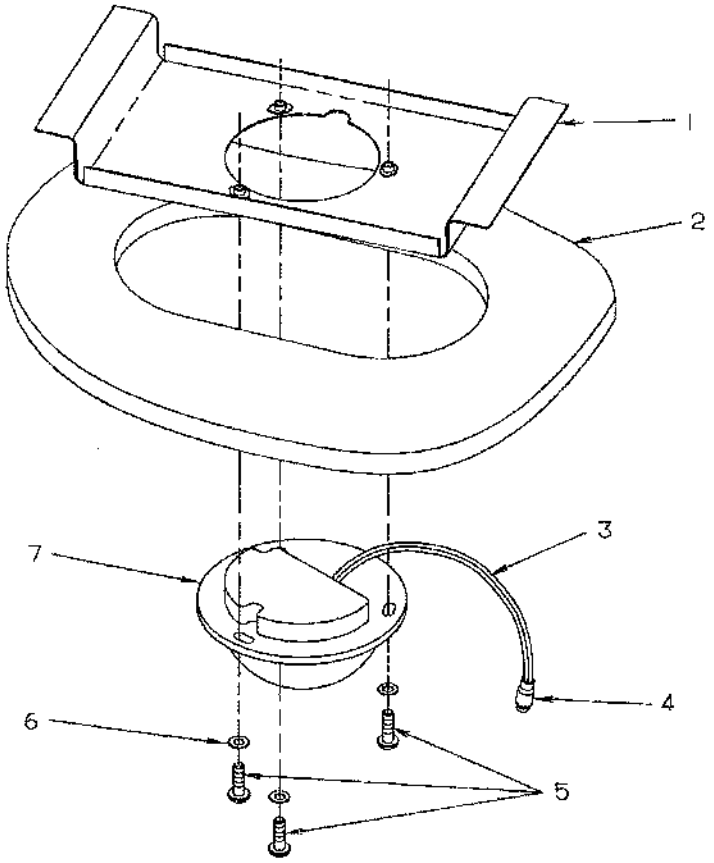




Attitude Heading Computer and Internal Compensation Unit Installation
(Effectivity: All)
Figure 3



AIRPLANE ZONE 546 LEFT
AIRPLANE ZONE 646 RIGHT



1. MOUNTING BRACKET
2. WING SKIN
3. WIRE HARNESS
4. CONNECTOR
5. SCREWS
6. WASHER
7. FLUX DETECTOR

DETAIL A

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Flux Detector Installation
(Effectivity: All)
Figure 4

NOTE

Test the system using the level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section. If the FDU flux detector unit is replaced, a compass swing must be performed. Refer to the AHC installation section of the Collins manual 523-0772305 for the compass swing procedure.

c. Refer to Chapter 57-00-00 and install the left (546AB) or right (646AB) wing access panel.

SENSOR DISPLAY DRIVER (SDD) REMOVAL (Effectivity: All) (Figure 5)

The sensor display driver (SDD) drives both sensor display units. It is mounted in the right side avionics compartment.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the left nose avionics compartment panel (213AL).

c. Loosen the retaining knob (2) that secures the sensor unit (1) in the rack (4).

d. Pull the sensor unit (1) out of the rack (ensure that it clears the connector before lifting).

e. Remove the unit to the workbench or a safe storage area.

SENSOR DISPLAY DRIVER (SDD) INSTALLATION (Effectivity: All) (Figure 5)

a. Place the sensor unit (1) in the rack (4) and firmly seat in the connector.

b. Place the retaining knob (2) on the sensor unit (1) and hand-tighten.

c. Restore electric power to the airplane.

d. Test the SDD for proper operation as follows:

NOTE

Refer to the level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

e. Install the left nose avionics compartment panel (213AL). Refer to Chapter 53-30-00 for installation procedure.

STANDBY ATTITUDE INDICATOR REMOVAL (Effectivity: All) (Figure 6)

a. Remove the center reversionary panel directly above the standby attitude indicator. Refer to Chapter 31-10-00 for removal procedures.

b. Loosen the screws (3) at the corners of the standby attitude indicator (4).

c. Reach through the opening above the indicator (4) and support it as the screws are removed.

d. Push the indicator (4) into the panel until it clears the mounting hole.

e. Move the indicator (4) up and out of the access opening above it.

f. Disconnect the electrical connector (1), cap, and identify it.

g. Cap the back of the indicator to prevent damage.

STANDBY ATTITUDE INDICATOR INSTALLATION (Effectivity: All) (Figure 6)

a. Remove the protective cap from the indicator.

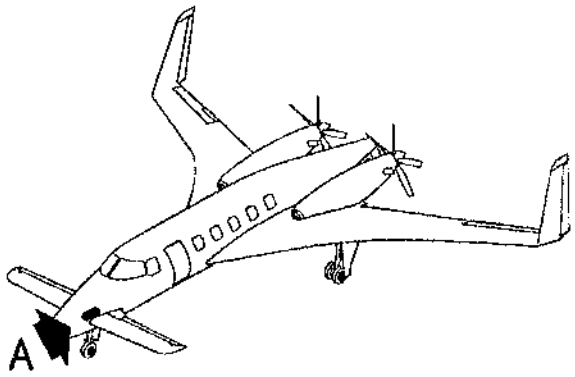
b. Attach the electrical connector (1) to the standby attitude indicator (4)

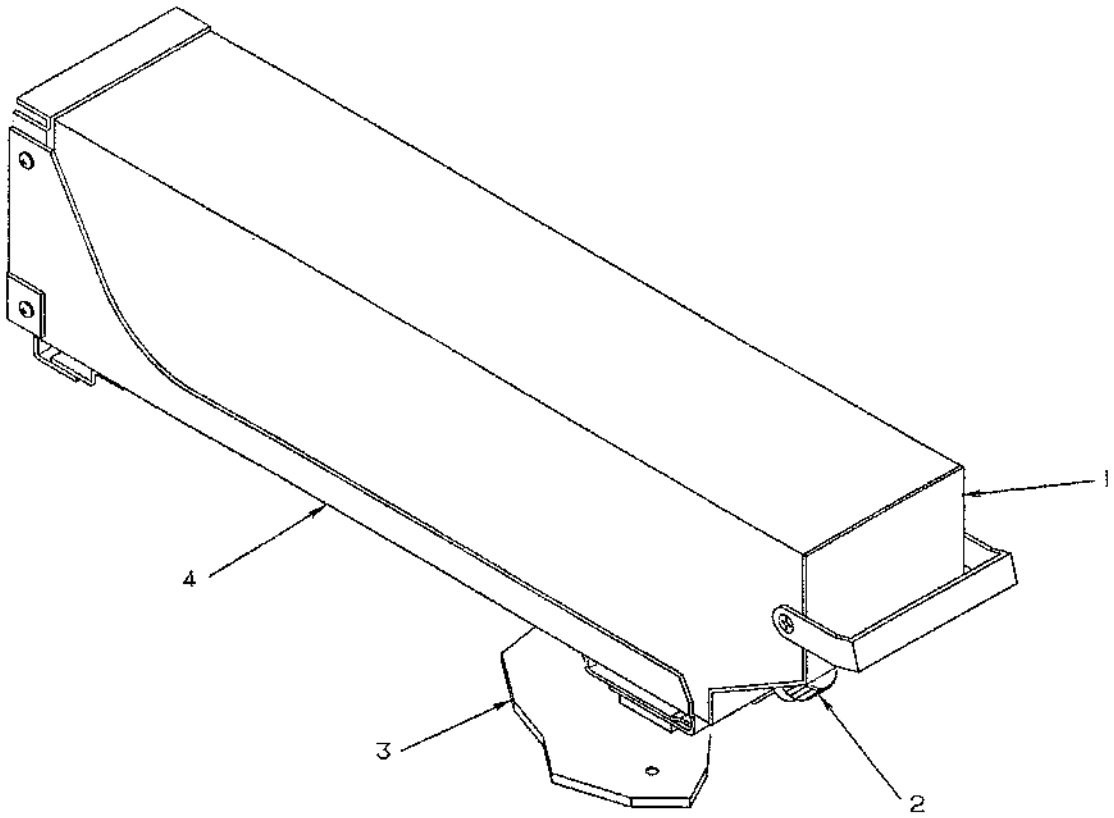
c. Push the standby attitude indicator through the access opening, and place in position on the instrument panel.

d. Support the indicator (4) and insert the screws (3) through the panel holes to secure the indicator.

e. Install the center reversionary panel. Refer to Chapter 31-10-00 for installation procedures.

1. SENSOR DISPLAY DRIVER
2. RETAINING KNOB
3. AVIONICS SHELF
4. RACK





DETAIL A

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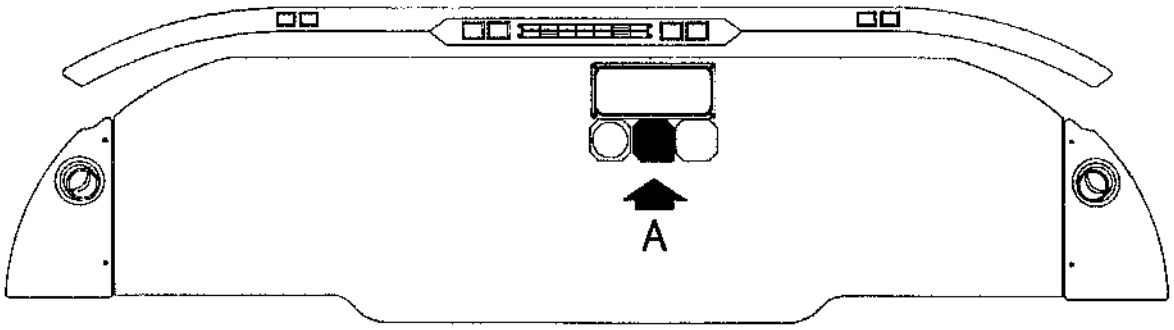
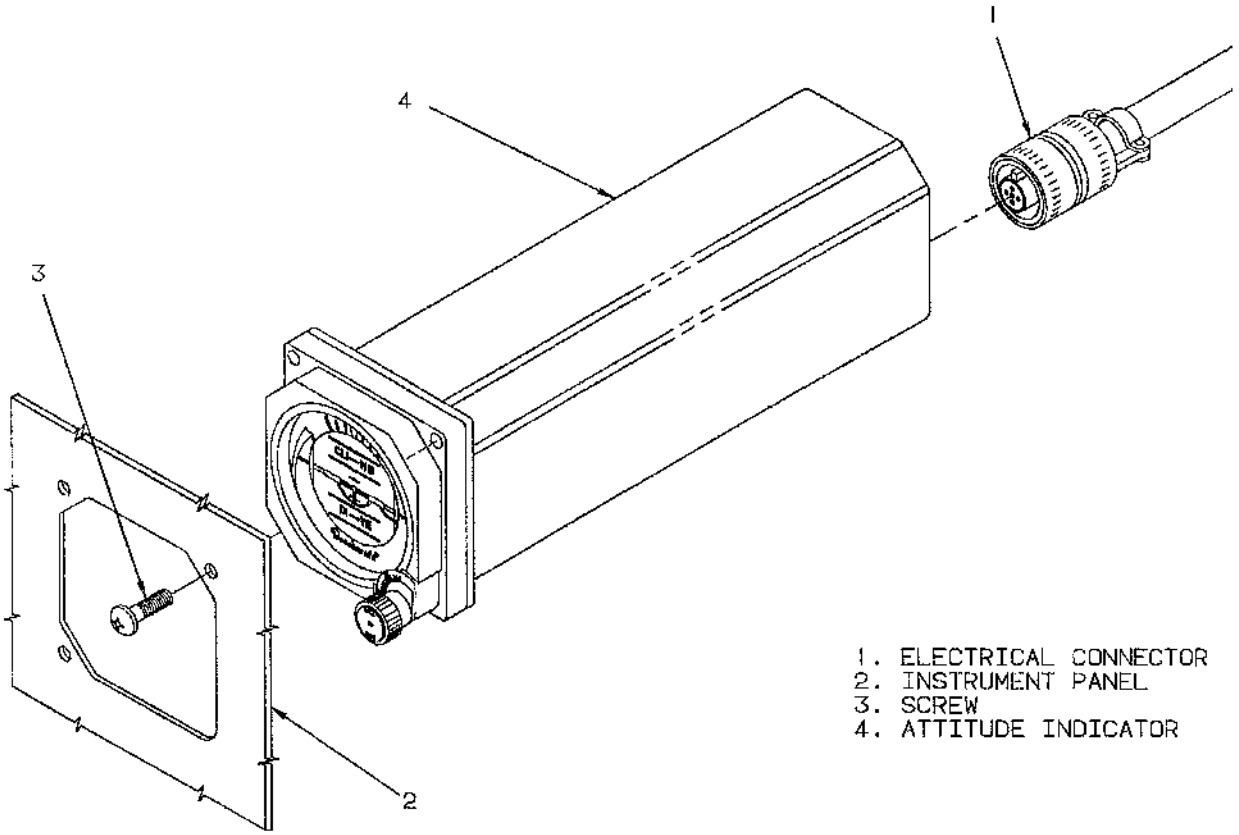
Sensor Display Driver Installation
(Effectivity: All)
Figure 5





Standby Attitude Indicator Installation
(Effectivity: All)
Figure 6

LANDING SYSTEMS DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exception would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapter 32-10-00 and 32-20-00 for the proper pin installation procedures.

GLIDESLOPE (Effectivity: All)

The glideslope receiver is integrated into the Collins VIR-432 (VOR/LOC) navigation receivers located in the forward avionics wheel well compartments. The glideslope indicates the position of the airplane relative to the correct glide path on final approach to landing. Glideslope deviation data is displayed on the Primary Flight Display (PFD) and the Navigation Display (ND). The glideslope antenna is mounted on the radome bulkhead below the weather radar transceiver. Refer to Chapter 34-50-00 for information regarding the VOR/LOC navigation receivers.

MARKER BEACON (Effectivity: All)

The marker beacons indicate the position of the airplane along the approach to the runway. Marker beacon passage is indicated on the Primary Flight Display (PFD) by the annunciation in the top right corner of the display in a box with "OM" or "MM" in the box for outer marker or middle marker. The inner or airway beacon is annunciated by only a box.

The marker beacon electronics are integrated into the Collins VIR-432 (VOR/LOC) navigation receivers which are located in the forward avionics nose gear wheel well compartments. The marker beacon antenna is located on the lower fuselage. Refer to Chapter 34-50-00 for information regarding the VOR/LOC navigation receivers.

LANDING SYSTEMS - MAINTENANCE PRACTICES (Effectivity: All)

GLIDESLOPE/MARKER BEACON RECEIVER REMOVAL/INSTALLATION (Effectivity: All)

The glideslope/marker beacon receivers are integrated into the VOR/LOC navigation receivers located in the forward avionics wheel well compartments. The Nav 1 receiver is in the left compartment mounted on the bottom of the lower shelf. The NAV 2 receiver is in the right compartment on top of the middle shelf. Refer to Chapter 34-50-00 for maintenance procedures for the VOR/LOC navigation receivers.

GLIDESLOPE COAXIAL COUPLER REMOVAL (Effectivity: All) (Figure 2)

The glideslope coaxial coupler connects both navigation receivers to one antenna.

a. Remove all electrical power from the airplane.

b. Operate the alternate extension hand pump in the cockpit until the forward nose landing gear doors are fully open.

c. Refer to Chapter 32-20-00 and install the lockpin in the nose landing gear door bellcrank.

d. Remove the access panel for the right avionics compartment (710BZ). Refer to Chapter 53-30-00 for removal procedure.

e. Remove the coaxial cable connectors from the coupler and tag.

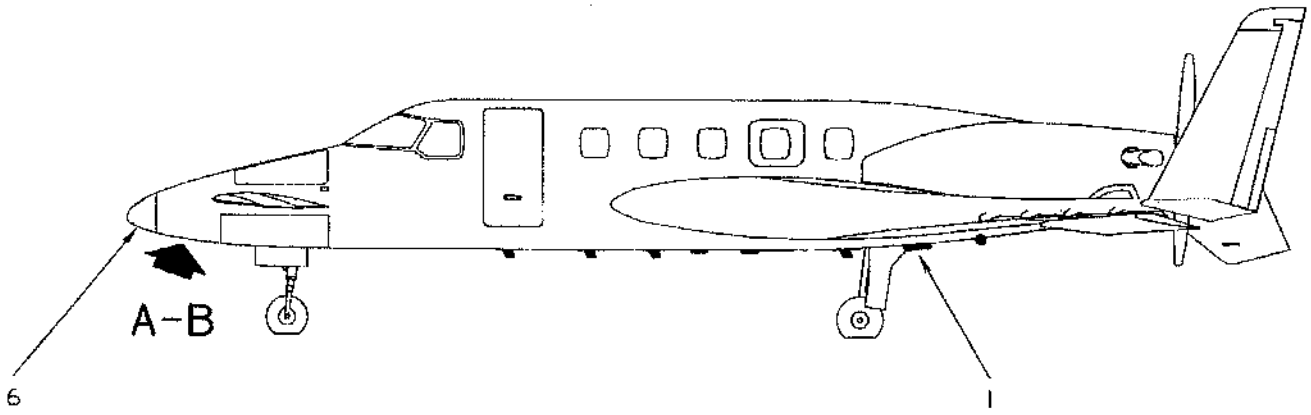
f. Remove the mounting screw (11) and washers (6, 7, 9 and 10) that secure the coupler (8) to the avionics shelf (4).

g. Remove the coupler to the workbench or a safe storage area.

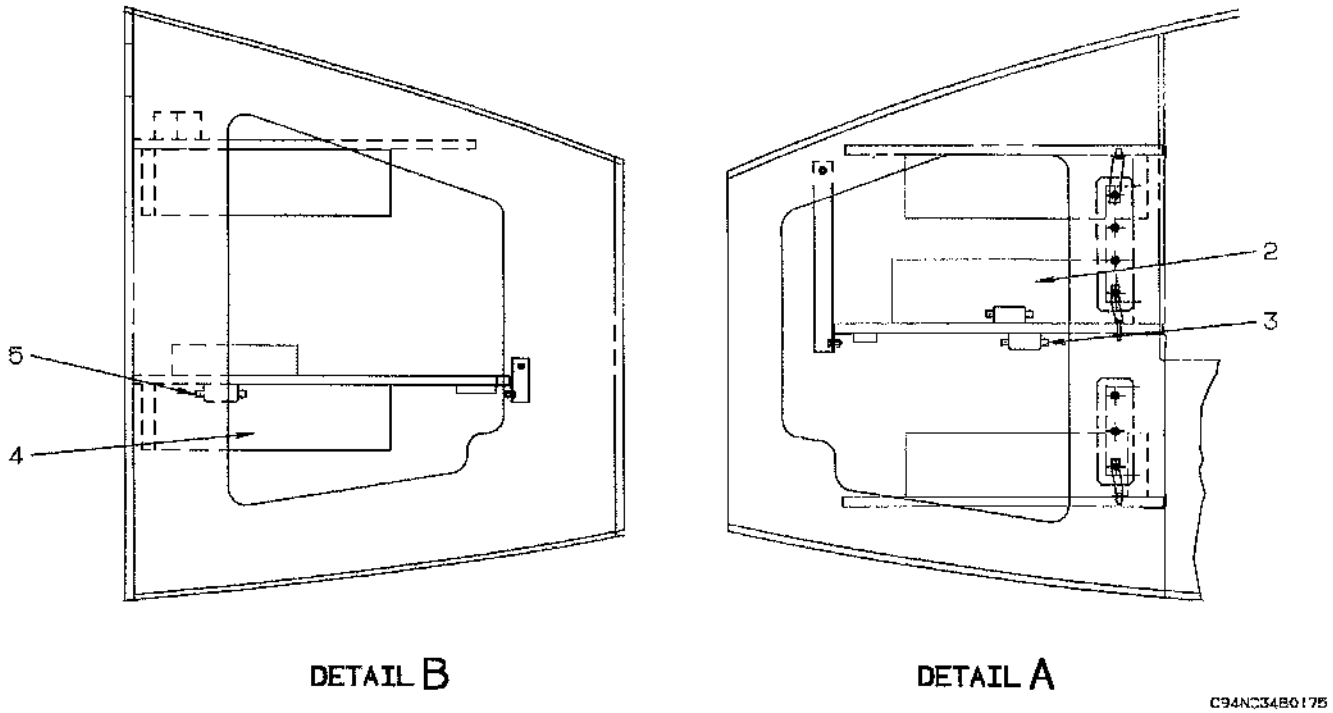
GLIDESLOPE COAXIAL COUPLER INSTALLATION (Effectivity: All) (Figure 2)

a. Position the coupler (8) on the shelf (4) and secure with the screw (11) and washers (6, 7, 9 and 10).

b. Identify the coaxial cable connectors and attach them to the coupler.



1. MARKER BEACON ANTENNA
2. VOR NAV 2 RECEIVER
3. GLIDE SLOPE COAXIAL COUPLER
4. VOR NAV 1 RECEIVER
5. MARKER BEACON COAXIAL COUPLER
6. GLIDE SLOPE ANTENNA



Landing Systems Location
(Effectivity: All)
Figure 1

c. Restore electrical power to the airplane.

d. Test the glideslope system as follows:

1. Refer to Chapter 24-40-00 and connect external power to the airplane.

2. Turn on the pilot's and copilot's integrated avionic switches on the center subpanel.

3. Ramp test the glideslope system with the IFR NAV-401 or an equivalent avionics ramp test set (3, Chart 1, 34-00-00). Perform the level 2 VOR NAV test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A.

4. Check for proper GS readout on both primary flight (PFD) displays and navigation displays (ND).

e. Disconnect the external power unit.

f. Install the right nose wheel avionics access panel (710BZ). Refer to Chapter 53-30-00 for installation procedure.

g. Refer to Chapter 32-20-00 and remove the nose landing gear door lockpin and push the doors closed.

GLIDESLOPE ANTENNA REMOVAL (Effectivity: All) (Figure 3)

a. Remove all electrical power from the airplane.

b. Operate the alternate extension hand pump in the cockpit until the forward nose landing gear doors are fully open.

c. Refer to chapter 32-20-00 and install the lockpin in the nose landing gear door bellcrank.

d. Loosen the five captive fasteners holding the radome (1) to the nose bulkhead (2) and remove the radome to a safe storage area. Refer to Chapter 53-50-00 for removal procedures.

e. Disconnect the antenna cable (4) from the antenna connector (9) and tag the antenna cable for identification.

f. Remove the four screws (8) and washers (7) that secure the glideslope antenna (6) to the nose bulkhead (2) and pull the antenna away from the bulkhead.

g. Remove the antenna (6), plate (5), and nutplate assembly (3) from the airplane to the workbench or a safe storage area.

GLIDESLOPE ANTENNA INSTALLATION (Effectivity: All) (Figure 3)

a. Remove all sealer from the antenna (6), plate (5), nutplate assembly (3), and the nose bulkhead area using a nonmetallic scraper and nonmetallic cleaner pad.

b. Thoroughly clean the front and back surfaces of the nose bulkhead (2) where the glideslope antenna is to be installed. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for the cleaning and bonding procedures.

c. Thoroughly clean the antenna to base metal and chemically treat the antenna with alodine (3, Chart 2, Chapter 34-00-00) to prevent corrosion. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for bonding procedures.

d. With the help of an assistant, place the glideslope antenna (6) and plate (5), on the forward side of the nose bulkhead and the nutplate assembly (3) on the aft side of the nose bulkhead. Secure in place with the four screws (8) and washers (7). Install the screws wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

e. Check bond resistance for the antenna as indicated in Chapter 20-00-03.

f. Attach the antenna cable (4) to the antenna base connector (9) on the aft side of the bulkhead.

g. Restore power to the airplane.

h. Test the glideslope antenna for proper operation as follows:

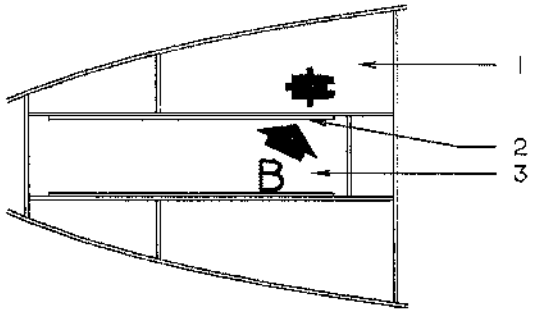
1. Refer to Chapter 24-40-00 and connect the external power to the airplane.

2. Turn on the pilot and copilot integrated avionic switches on the center subpanel.

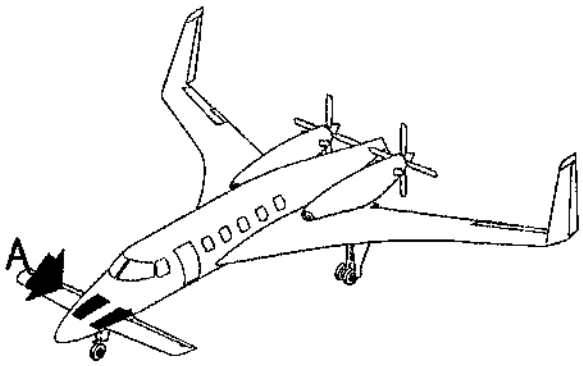
3. Ramp test the glideslope antenna with the IFR NAV-401 or an equivalent avionics ramp test set (3, Chart 1, 34-00-00). Perform the level 2 VOR NAV test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

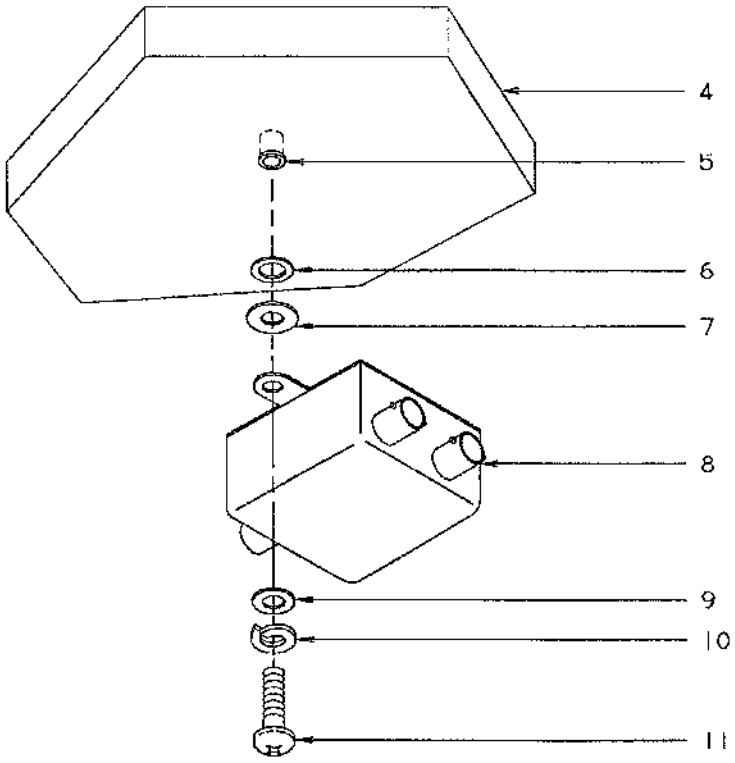
4. Check for proper GS readout on the primary flight display (PFD) unit and the navigation display (ND).

i. From the aft side of the bulkhead, seal around the periphery of the nut plate assembly (3) with sealer (1, Chart 2, Chapter 34-00-00). Refer to Detail B of Figure 3.



DETAIL A





DETAIL B

1. RIGHT NOSE WHEEL WELL AVIONICS COMPARTMENT
2. RIGHT AVIONICS ACCESS DOOR
3. NOSE GEAR WHEELWELL
4. AVIONICS SHELF
5. RIVNUT
6. WASHER
7. WASHER
8. GLIDESLOPE COAX COUPLER
9. WASHER
10. LOCK WASHER
11. SCREW

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Glideslope Coaxial Coupler Installation
(Effectivity: All)
Figure 2

j. Refer to Chapter 53-50-00 and install the radome (1) and secure with the five fasteners.

k. Refer to Chapter 32-20-00 and remove the nose landing gear door lockpin and push the doors closed.

MARKER BEACON COAXIAL COUPLER REMOVAL (Effectivity: All) (Figure 4)

The marker beacon coaxial coupler connects both marker beacon receivers to one antenna and is located in the left nose avionics compartment.

a. Remove all electrical power from the airplane.

b. Operate the alternate extension hand pump in the cockpit until the forward nose landing gear doors are fully open.

c. Refer to Chapter 32-20-00 and install the lockpin in the nose landing gear door bellcrank.

d. Refer to Chapter 53-30-00 and remove the left nose avionics compartment access panel (213AL).

e. Remove the coaxial cable connectors from the coupler and tag.

f. Remove the mounting screw (11) and washers (6, 7, 9 and 10) that secure the coupler (8) to the avionics shelf (4).

g. Remove the coupler to the workbench or a safe storage area.

MARKER BEACON COAXIAL COUPLER INSTALLATION (Effectivity: All) (Figure 4)

a. Position the coupler (8) on the shelf (4) and secure with the screw (11) and washers (6, 7, 9 and 10).

b. Identify the coaxial cable connectors and attach them to the coupler.

c. Test the marker beacon system as follows:

1. Refer to Chapter 24-40-00 and connect external power to the airplane.

2. Ramp test the marker beacon system with the IFR NAV-401 or an equivalent avionics ramp test set (3, Chart 1, 34-00-00). Perform the level 2 VOR NAV test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

3. Check for marker beacon readout on both primary flight displays (PFD).

4. Check for marker beacon audio signal.

d. Disconnect the external power unit.

e. Refer to Chapter 53-30-00 and install the left nose wheel avionics access panel (213AL).

f. Refer to Chapter 32-30-00 and remove the nose landing gear door lockpin and push the doors closed.

MARKER BEACON ANTENNA REMOVAL (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 5)

a. Remove all electrical power from the airplane.

b. Disconnect the coaxial connector (2) from the marker beacon antenna (4).

c. Refer to Chapter 53-30-00 and remove the aft wing box shear panel (528AB).

d. Remove the four screws (6) and washers (5) that secure the marker beacon antenna (4) to the aft wing box shear panel (3).

e. Remove the bead of sealant from around the antenna with a nonmetallic scraper to loosen the marker beacon antenna.

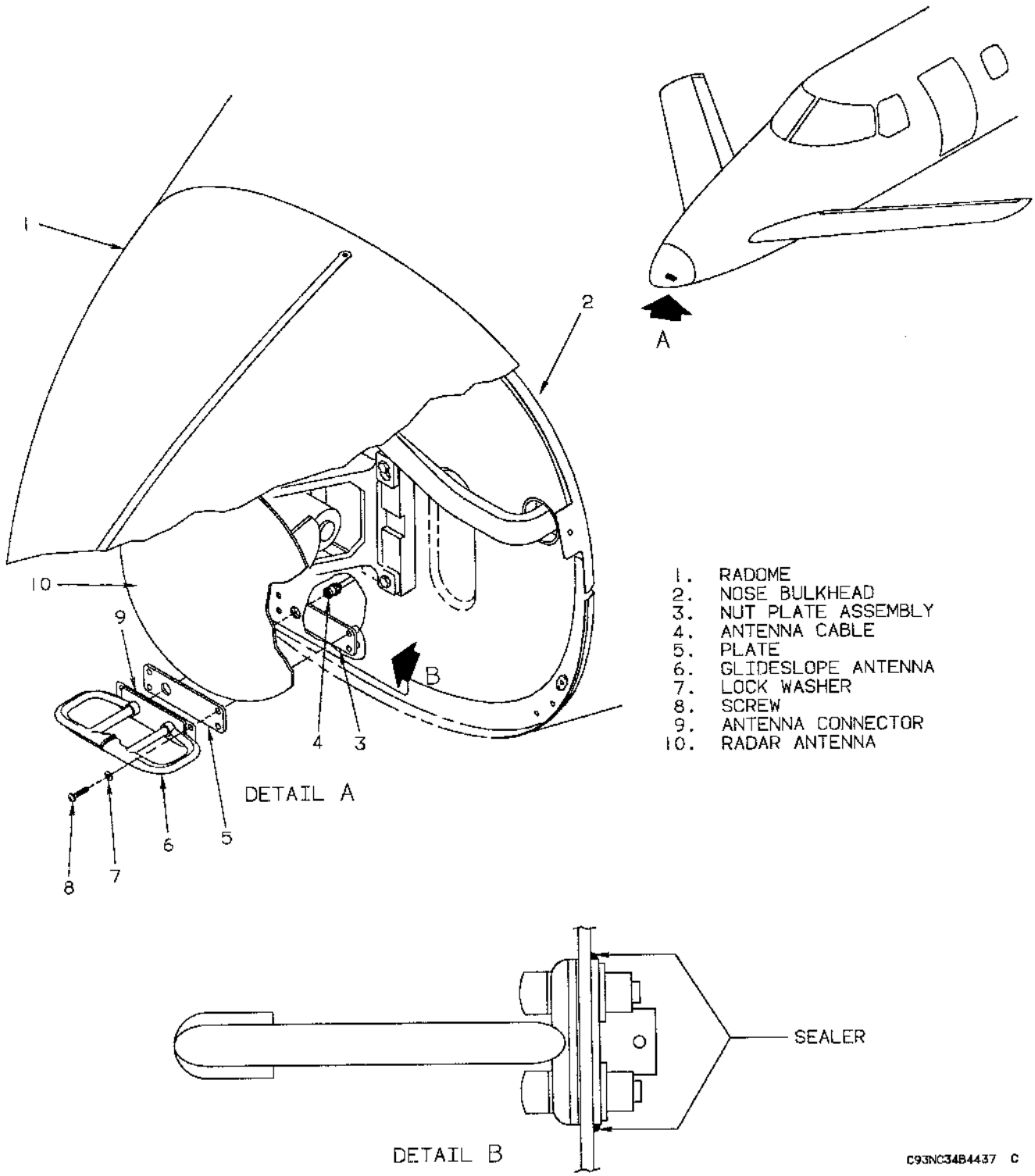
f. Remove the antenna from the wing box shear panel.

MARKER BEACON ANTENNA INSTALLATION (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 5)

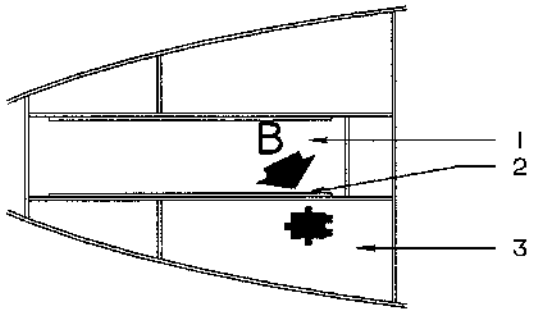
a. Remove all old sealer from the surfaces of the marker beacon antenna (4), reinforcement assembly (1), and wing box shear panel (3) contact surfaces (top and bottom) using a nonmetallic scraper and nonmetallic scratch pad.

b. Thoroughly clean the mounting surfaces of the wing box shear panel where the antenna is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

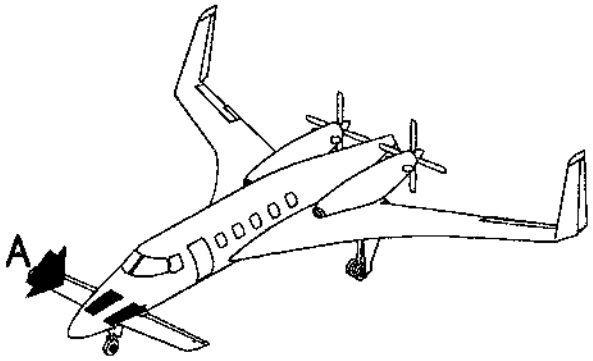
c. Thoroughly clean the antenna and reinforcement plate assembly to base metal and chemically treat the antenna with alodine (3, Chart 2, Chapter 34-00-00) to

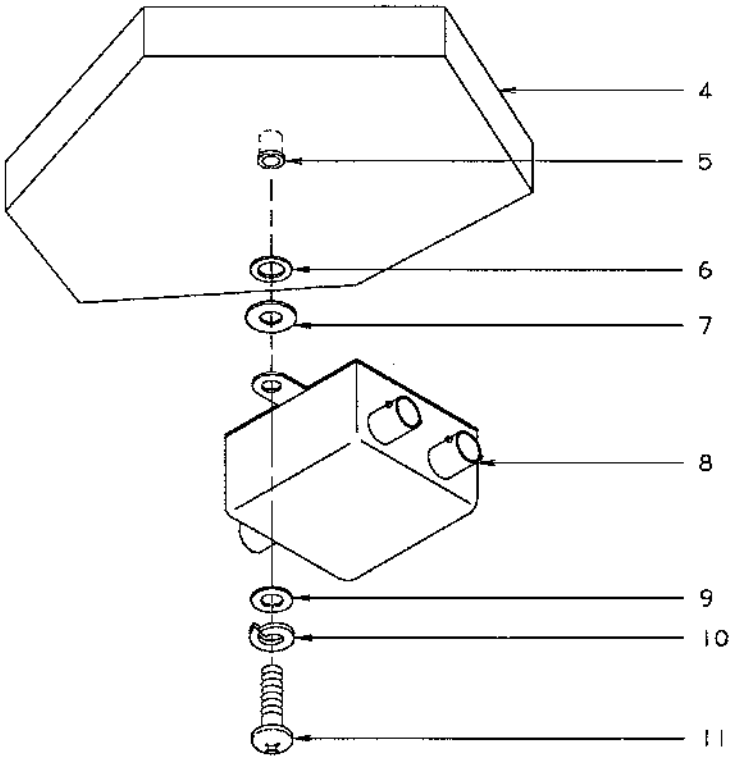


Glideslope Antenna Installation (Effectivity: All)
Figure 3



DETAIL A





DETAIL B

1. NOSE GEAR WHEELWELL
2. LEFT AVIONICS ACCESS DOOR
3. LEFT NOSE WHEEL WELL AVIONICS COMPARTMENT
4. AVIONICS SHELF
5. RIVNUT
6. WASHER
7. WASHER
8. MARKER BEACON COUPLER
9. WASHER
10. LOCK WASHER
11. SCREW

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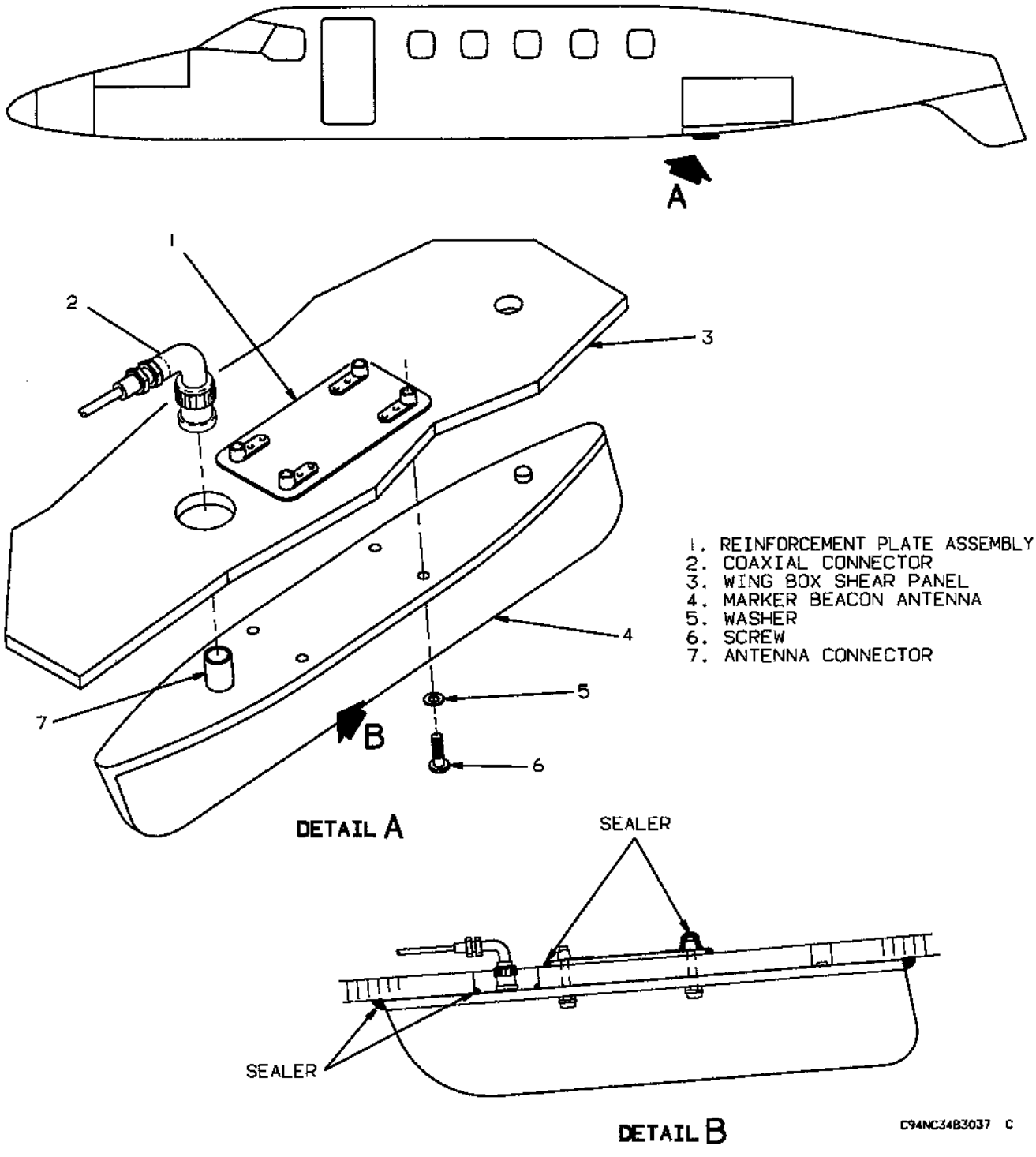
Marker Beacon Coaxial Coupler Installation
(Effectivity: All)
Figure 4



Marker Beacon Antenna Installation (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 5

prevent corrosion. Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for bonding procedures.

d. Install the marker beacon antenna (4) on the aft wing shear panel (3) and reinforcement plate (1) and secure with the four screws (6) and washers (5).

e. Check the bond resistance for the antenna as indicated in Chapter 20-00-03.

f. After the resistance check, apply sealer (1, Chart 2, 34-00-00) around periphery of the antenna where it contacts the wing-box shear panel as shown by Detail B of Figure 5.

g. Apply sealer (1, Chart 2, 34-00-00) around periphery of the reinforcement plate assembly on the upper side of the shear panel. Apply sealer also to cover the nutplates on the reinforcement plate.

h. Apply sealer (1, Chart 2, 34-00-00) around periphery of the antenna connector base where the antenna contacts the shear panel.

i. Refer to Chapter 53-50-00 and install the aft wing box shear panel (528AB) over the wing box.

j. Connect the coaxial connector (2) to the marker beacon antenna (4). Ensure that the bonding strap from the shear panel is connected to the antenna coaxial connectors.

k. Restore electrical power to the airplane.

l. Test the marker beacon antenna for proper operation as follows:

1. Refer to Chapter 24-40-00 and connect external power to the airplane.

2. Ramp test the marker beacon and NAV receiver with the IFR NAV-401, or an equivalent avionics ramp test set (3, Chart 1, 34-00-00). Perform the level 2 VOR NAV test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

m. When the sealer has cured, touch up the paint as necessary.

MARKER BEACON ANTENNA REMOVAL (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 6)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the aft wing box shear panel (528AB).

c. Remove the four screws (8) and washers (7) that secure the marker beacon antenna (6) to the aft wing box shear panel (5).

d. Remove the bead of sealant from around the antenna with a nonmetallic scraper to loosen the marker beacon antenna.

e. Cut and remove the tie-wrap (2), then remove the sealant from the boot opening.

CAUTION

Use care to avoid damaging the coaxial cable (1), connector (9) and surrounding surface of the reinforcement plate assembly (4).

f. Slice open the antenna boot (3) as necessary to gain access to the antenna connector (9).

g. Disconnect the coaxial cable (1) from the antenna (6).

h. Remove the antenna from the wing box shear panel.

NOTE

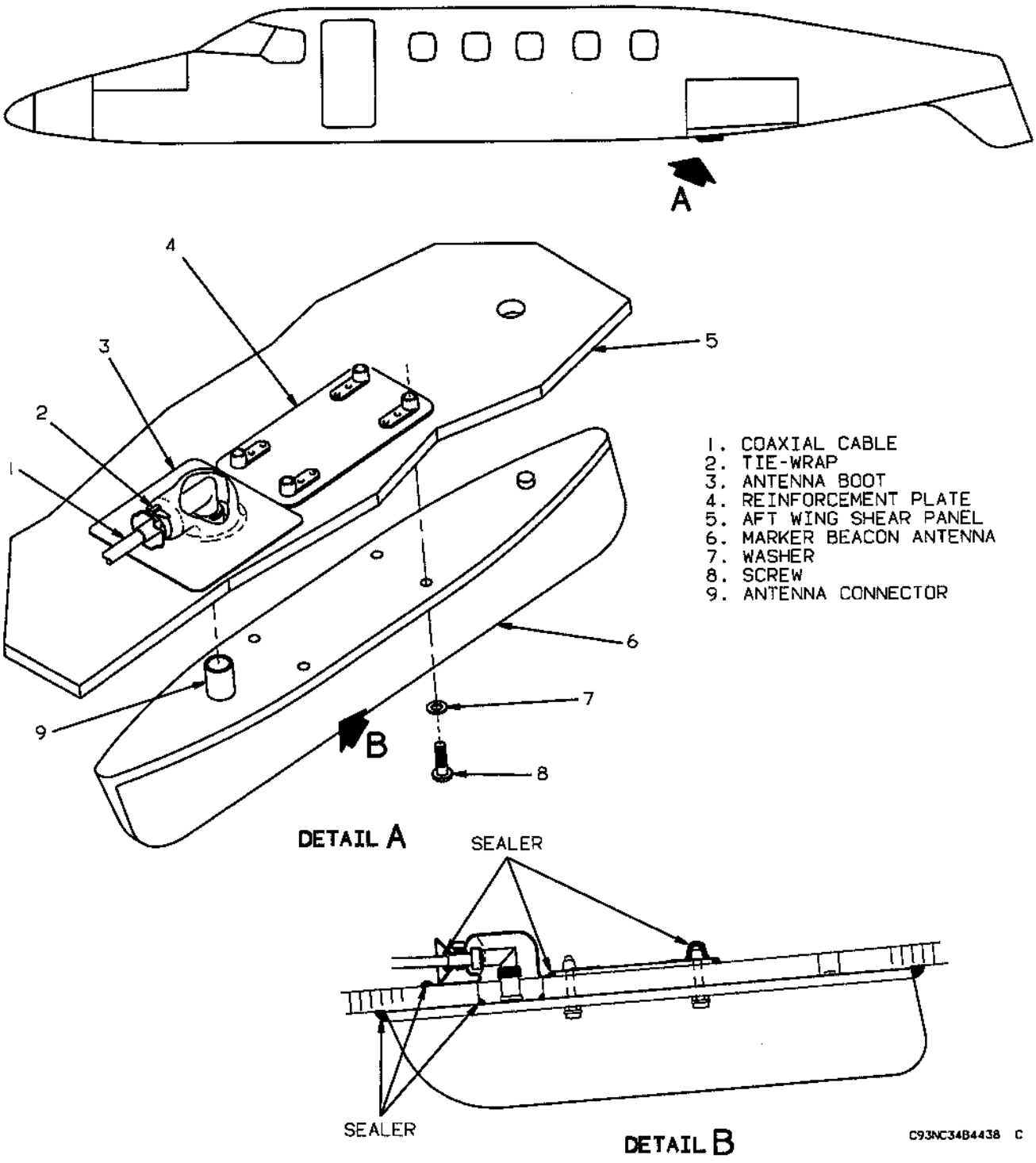
If the antenna boot is not damaged it may be reused. However, a cut or torn boot must be replaced as it is not repairable.

MARKER BEACON ANTENNA INSTALLATION (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 6)

a. Remove all old sealer from the marker beacon antenna (6), reinforcement assembly (4) and wing box shear panel (5) contact surfaces (top and bottom) using a nonmetallic scraper and nonmetallic scratch pad.

b. Thoroughly clean the mounting surfaces of the aft wing box shear panel where the antenna is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the antenna and reinforcement plate assembly to base metal and chemically treat the antenna with alodine (3, Chart 2, Chapter 34-00-00) to



Marker Beacon Antenna Installation (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 6

prevent corrosion. Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for bonding procedures.

d. Install the marker beacon antenna (6) on the aft wing box shear panel (5) and reinforcement plate (4), then secure with four screws (8) and washers (7).

e. Check the bond resistance for the antenna as specified in Chapter 20-00-03.

1. Refer to Chapter 24-40-00 and connect external power to the airplane.

2. Ramp test the marker beacon and NAV receiver with the IFR NAV-401, or an equivalent avionics ramp test set (3, Chart 1, 34-00-00). Perform the level 2 VOR NAV test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

f. Remove electrical power from the airplane.

g. Disconnect the coaxial connector (1) from the marker beacon antenna (6).

h. From inside the cabin, apply sealant (1, Chart 2, 34-00-00) around the inside of the fuselage antenna opening where the marker beacon antenna (6) and fuselage are joined together, as shown by Detail B of Figure 6.

NOTE

If a new antenna boot (3) is to be installed, carefully trim the boot to match the reinforcement plate (4) and nut plates before performing the remaining installation procedures.

i. Insert the coaxial cable (1) into the antenna boot (3) opening and carefully push the antenna connector (9) out the bottom.

j. Connect the coaxial cable (1) to the marker beacon antenna (6).

k. Bond the antenna boot (3) to the reinforcement plate assembly (4) with sealant (4, Chart 2, 34-00-00).

l. Apply sealant (4, Chart 2, 34-00-00) to the periphery of the reinforcement plate assembly (4) and antenna boot (3) to provide a good moisture barrier between the antenna mounting assembly and fuselage skin, as shown by Detail B of Figure 6.

m. Completely cover the nut plates on the reinforcement plate assembly (4) with sealant (4, Chart 2, 34-00-00).

n. Apply sealant (4, Chart 4, 34-00-00) approximately 1/2 inch into the opening where the coaxial cable (1) exits the antenna boot (3).

o. Install a tie-wrap (2) around the boot neck and coaxial cable (1).

p. Apply sealer (1, Chart 2, 34-00-00) around the periphery of the antenna where it contacts the aft wing box shear panel as shown by Detail B of Figure 6.

q. Refer to Chapter 53-50-00 and install the aft wing box shear panel (528AB) over the wing box. Connect the bonding strap from the panel to the antenna coaxial connectors (1).

r. Restore electrical power to the airplane.

s. When the sealer has cured, touch up the paint as necessary.

INDEPENDENT POSITION DETERMINING - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapter 32-10-00 and 32-20-00 for the proper pin installation procedures.

WEATHER RADAR (Effectivity: All) (Figure 1)

The weather radar system provides a picture of the precipitation and any related turbulence ahead of the airplane.

The Collins Avionics radar system is located on the radome bulkhead. It is a one-piece design and does not require a separate transceiver and the associated waveguide.

INDEPENDENT POSITION DETERMINING - TROUBLESHOOTING (Effectivity: All)

a. Apply power to all avionics systems. Wait 5 minutes for the diagnostic system to stabilize.

b. Refer to the level-2 troubleshooting procedures in the Collins Starship 1 Avionics System Manual, 523-0774076-0011A, Maintenance Section when troubleshooting the independent position determining system.

INDEPENDENT POSITION DETERMINING - MAINTENANCE PRACTICES (Effectivity: All)

RADAR TRANSCEIVER REMOVAL (Effectivity: All) (Figure 1)

The weather radar transceiver is an integral part of the radar antenna installation. Access to the radar transceiver is made through the radome.

a. Remove all electrical power from the airplane.

b. Remove the radome from the airplane. Refer to Chapter 53-50-00 for removal procedures.

c. Loosen the four screws (2) on the weather radar unit (3).

CAUTION

To prevent damaging the gears or stops, do not manually move the antenna when working with or removing the weather radar.

d. Disconnect the wire harness from the radar unit.

e. Remove the two bottom screws (2) and with the top screws loose, lift the unit up and out of the slots at the tops of the screw holes. Ensure that any leveling shims (4) in place on the screws are retained for installation.

NOTE

Do not lose the leveling shims (4) used on the screws holding the radar mounting plate. These shims MUST be replaced on the same side that they were removed from. The LOSS OF SHIMS will require releveling the radar transceiver/antenna unit. Refer to the leveling section of this chapter for procedures.

f. Remove the radar unit (3) to the workbench or a safe storage area.

RADAR TRANSCEIVER INSTALLATION (Effectivity: All) (Figure 1)

CAUTION

To prevent damaging the gears or stops, do not manually move the antenna when working with or removing the weather radar.

a. Place the radar unit (3) on the two top screws on the bulkhead and lower in place.

b. Replace the two lower screws (2) wet with sealer (2, Chart 2, 34-00-00). Ensure that any shims (4) that were originally installed are replaced. Tighten the lower screws (2).

c. Remove the top screws (2) and wet with sealer (2, Chart 2, Chapter 34-00-00). Install screws in top slots and tighten.

d. Reconnect the wire harness to the radar unit.

e. Test the radar system for proper operation as follows:

WARNING

The area within the scan arc and within 0.65 meters (2 feet) of an operating RTA can be a hazardous area. Do not operate the RTA in any mode other than STANDBY or TEST when the antenna might scan over personnel within that range. NEVER turn the transmitter on inside the hanger. FAA Advisory Circular 20-68B provides additional details on the radiation hazards associated with ground operation of airbourne weather radar.

1. Tow the airplane outside and away from personnel activity.

2. Refer to chapter 24-40-00 and connect external power.

3. Perform the level-2 RADAR test procedures contained in the Collins Avionics System Manual, 523-0774076-00111A, Maintenance Section.

4. Refer to chapter 24-40-00 and disconnect external power.

5. Tow the airplane back to the hanger.

f. Install the radome and secure with the five fasteners. Refer to Chapter 53-50-00 for installation procedures.

WEATHER RADAR LEVELING PROCEDURES (Effectivity: All) (Figure 2)

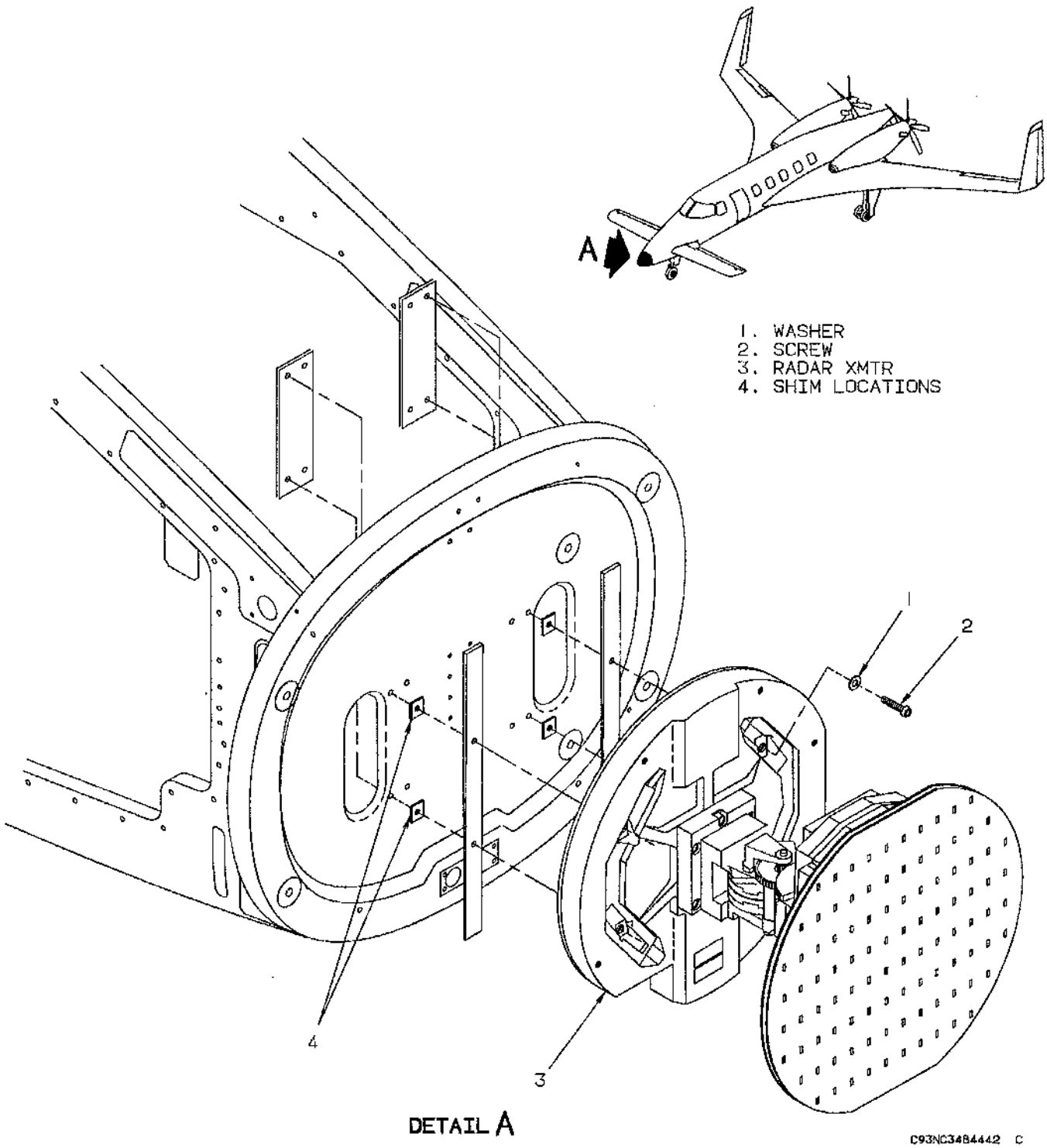
a. Refer to Chapter 8-10-00 and level the airplane.

b. Apply power to the radar and place control in STBY. The antenna should boresight (zero pitch and zero scan). Turn radar power off.

c. Remove all electrical power from the airplane.

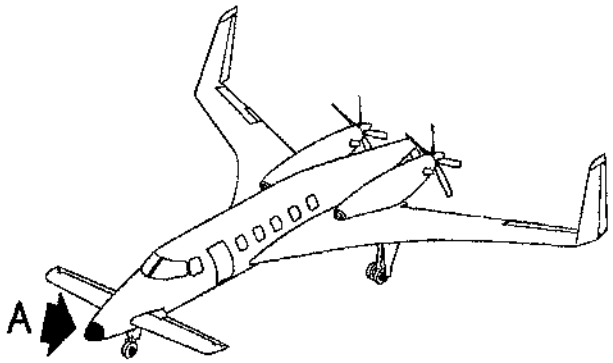
d. Place inclinometer on antenna flatplate as shown in Figure 2. Be sure to get the edge of the inclinometer between the vertical rows of folder tabs on the flatplate surface. Try to make all measurements at the same place.

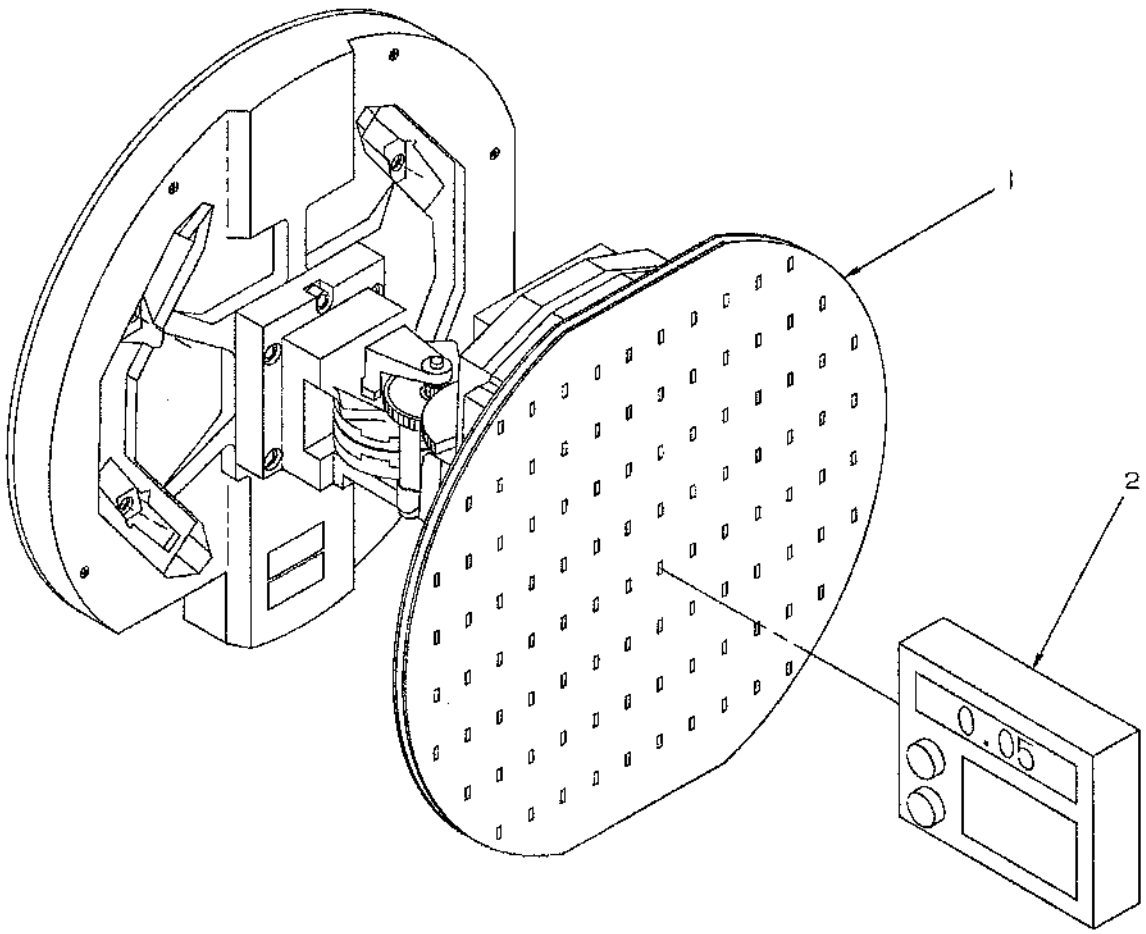
e. Ensure that the flatplate is at 0.0° ±0.25°. If not, shim antenna mounts. Refer to Figure 2 for shim location. Shims may be placed under both upper and lower mounting pads. Shims are supplied in a shim kit, Collins part number 687-9728-001.



Weather Radar Installation (Effectivity: All)
Figure 1

1. ANTENNA FLATPLATE
2. DIGITAL INCLINOMETER





DETAIL A

C93NC34B4441 C

Radar Antenna Leveling
(Effectivity: All)
Figure 2

DEPENDENT POSITION DETERMINING - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapter 32-10-00 and 32-20-00 for the proper pin installation procedures.

RADIO NAVIGATION SENSOR SYSTEMS (Effectivity: All) (Figure 1)

The navigation sensor systems include all radio units that determine the airplane's position.

The radio navigation sensor systems include:

a. VHF Navigation Receivers (VOR/LOC)

b. Distance Measuring Equipment (DME)

c. Automatic Direction Finder (ADF)

d. VLF/OMEGA Receiver (VLF)

e. Radio Altimeter System (RAD ALT)

f. Transponder System

VHF OMNIDIRECTIONAL RANGING NAVIGATION RECEIVERS (VOR) (Effectivity: All) (Figure 1)

The VOR navigation receivers perform VOR/LOC, glideslope (GL), and marker beacon (MB) functions. The omnidirectional range station provides the pilot with courses from any point within its service range. Operation is in the Very High Frequency (VHF) portion of the radio spectrum. The VHF spectrum provides clear signal reception so atmospheric and precipitation static is negligible.

The glideslope (GS) receiver is integrated into the navigation receivers. The glideslope indicates the position of the airplane relative to the correct glide path on final approach to landing. Glideslope deviation data is displayed on the Primary Flight Display (PFD) and the navigation display (ND). The glideslope antenna is mounted on the radome bulkhead below the weather radar transceiver. Refer to Chapter 34-30-00 for information regarding the glideslope antenna.

The marker beacon (MB) indicates the position of the airplane along the approach to the runway. Marker beacon passage is indicated on the Primary Flight Display (PFD) by the annunciation in the top right corner of the display in a box with "OM" or "MM" in the box for outer moarker or middle marker. The inner or airway beacon is annunciated by a box.

The marker beacon electronics are also integrated into the navigation receivers. Refer to Chapter 34-30-00 for information regarding the marker beacon antenna.

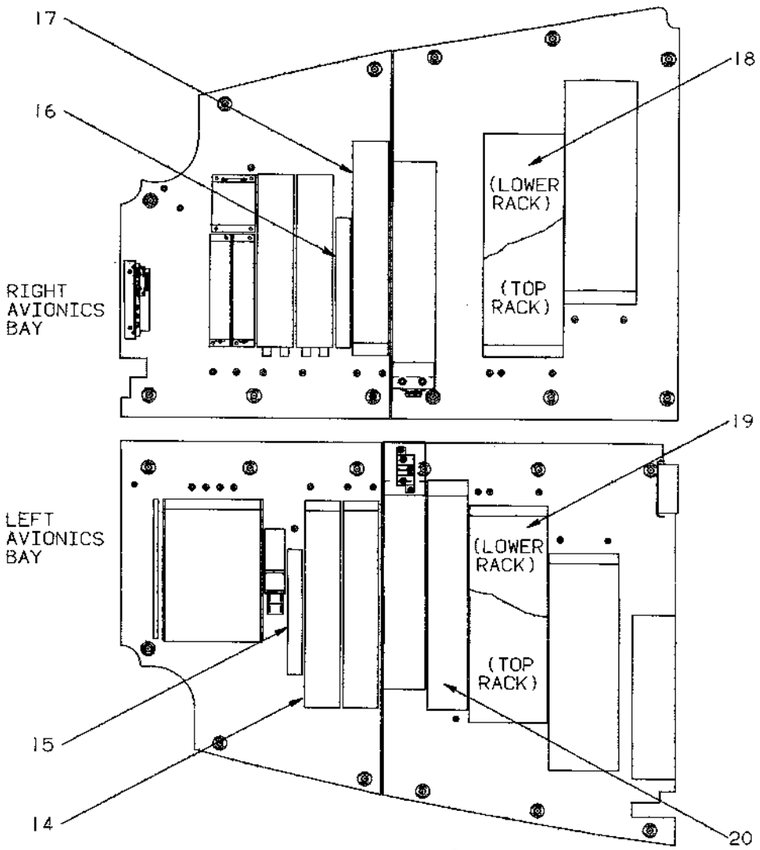
The navigation receivers are located in the forward avionics nose gear wheel well compartments. The nose gear must be extended and the forward nose gear doors must be opened to gain access to the nose gear wheel well. The left (710AZ) or right (710BZ) nose wheel well panel must be removed to gain access to the navigation receivers.

DISTANCE MEASURING EQUIPMENT (DME) (Effectivity: All) (Figure 1)

Distance Measuring Equipment (DME) provides a constant visual indication of the distance the airplane is from a ground station. A DME reading is not a true indication of point-to-point distance as measured over the ground. DME indicates the slant range between the airplane and the ground station. Slant range error increases as the airplane approaches the station.

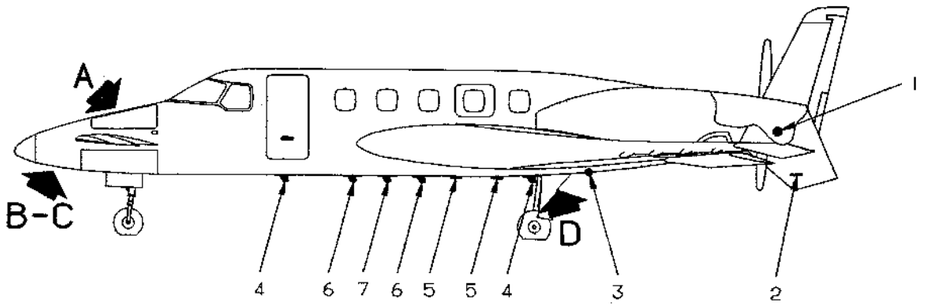
The DME operates in the UHF range of the radio frequency spectrum. Transmitting and receiving frequencies are given a channel number paired with a VOR channel. The stations are usually located with a VOR facility.

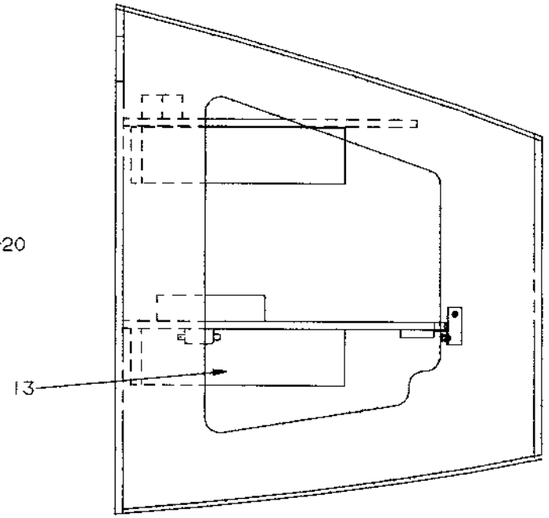
The DME units are mounted in the forward nose avionics compartments.



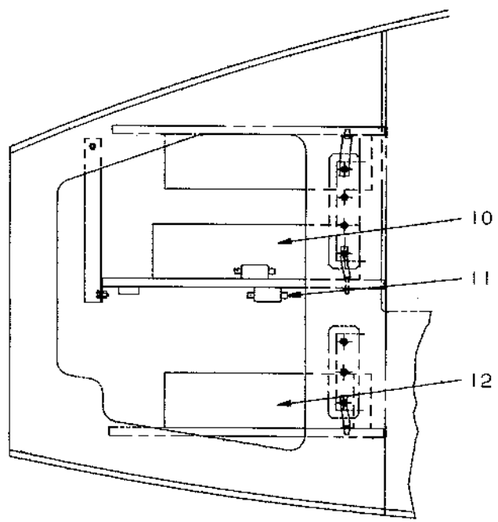
DETAIL A

1. VOR ANTENNA COUPLER
2. VOR ANTENNA
3. VLF ANTENNA
4. DME ANTENNA
5. RADIO ALTIMETER ANTENNA
6. TRANSPONDER ANTENNA
7. ADF ANTENNA
8. RADIO ALTIMETER TRANSMITTER/RECEIVER
9. RADIO ALTIMETER CONVERTER
10. VOR NAV 2 RECEIVER
11. VOR NAV COUPLER
12. ADF RECEIVER
13. VOR NAV 1 RECEIVER
14. TRANSPONDER TRANSCEIVER #1
15. TRANSPONDER CONTROL ADAPTER #1 (NC-4 THRU NC-25)
16. TRANSPONDER CONTROL ADAPTER #2 (NC-4 THRU NC-25)
17. TRANSPONDER TRANSCEIVER #2
18. DME TRANSCEIVER #2
19. DME TRANSCEIVER #1
20. VLF RECEIVER

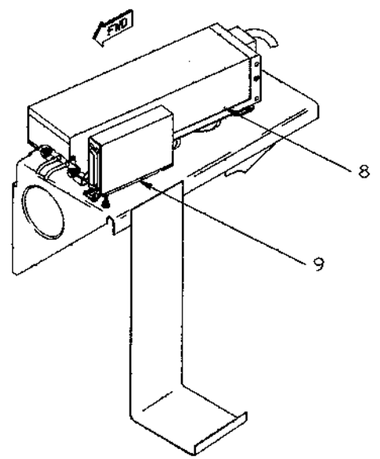




DETAIL B



DETAIL C



DETAIL D

C94NC34B0176

Radio Navigation Sensor Systems
(Effectivity: All)
Figure 1

AUTOMATIC DIRECTION FINDER (ADF) (Effectivity: All) (Figure 1)

Automatic Direction Finder (ADF) is a radio receiver equipped with a directional antenna used to determine the direction from which signals are received. The ADF operates in the low and medium frequency spectrum.

The ADF receiver is located in the right forward avionics compartment in the nose wheel well.

VLF/OMEGA NAV SYSTEM (VLF) (Effectivity: All) (Figure 1)

VLF/Omega navigation equipment uses signals from eight Omega transmitting stations and signals from the U.S. Navy VLF communications network to provide global enroute-navigation capabilities. The system on the airplane automatically selects the stations that provide the most accurate navigation. The system operates in the Very Low Frequency (VLF) band of the radio spectrum.

The VLF/Omega receiver is located in the left nose avionics compartment.

RADIO ALTIMETER (Effectivity: All) (Figure 1)

Radio altimeters are used to measure the distance from the airplane to the ground. This is accomplished by transmitting radio frequency energy to the ground and receiving the reflected energy at the airplane. Altitude is determined by measuring the time required for the transmitted pulse to hit the ground and return. The radio altimeter indicates the true altitude of the airplane. True altitude is the height above water, mountains, or other objects on the surface of the earth.

The radio altimeter transceiver is located in the aft wing box section in the forward manhole.

ATC TRANSPONDER (Effectivity: All)

The airplane is equipped with dual altitude encoding transponders to provide air traffic control with location and identification information.

The No. 1 transponder transceiver is mounted in the left avionics compartment, and the No. 2 transponder transceiver is mounted in the right avionics compartment.

DEPENDENT POSITION DETERMINING - TROUBLESHOOTING (Effectivity: All)

When troubleshooting the VOR navigation receivers, DME receivers, ADF receiver, VLF receiver, radio altimeter transceiver, and the transponder transceiver, refer to the electrical schematic diagrams shown in the BEECHCRAFT Starship 1 Avionics Wiring Diagrams ("R" drawings) and the level-2 troubleshooting procedures in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

DEPENDENT POSITION DETERMINING - MAINTENANCE PRACTICES (Effectivity: All)

VOR NAV RECEIVER REMOVAL (Effectivity: All) (Figure 2)

The two VOR receivers are located in the forward avionics nose gear wheel well compartments. NAV No. 1 is in the left compartment mounted on the bottom of the lower shelf. NAV No. 2 is in the right compartment mounted on top of the middle shelf.

a. Determine the unit to be removed.

b. Remove all electrical power from the airplane.

c. Operate the alternate extension hand pump in the cockpit until the forward nose landing gear doors are fully open.

d. Install the locking pin in the nose gear door bellcrank. Refer to Chapter 32-20-00, for pin location.

e. Remove the left (710AZ) or right (710BZ) nose wheel well access panel for the appropriate VOR unit.

f. Loosen the retaining knobs (5) that secure the receiver unit (1 or 2) to the rack assembly (3).

g. Release the latches and pull the receiver unit (1 or 2) forward and out of the connector.

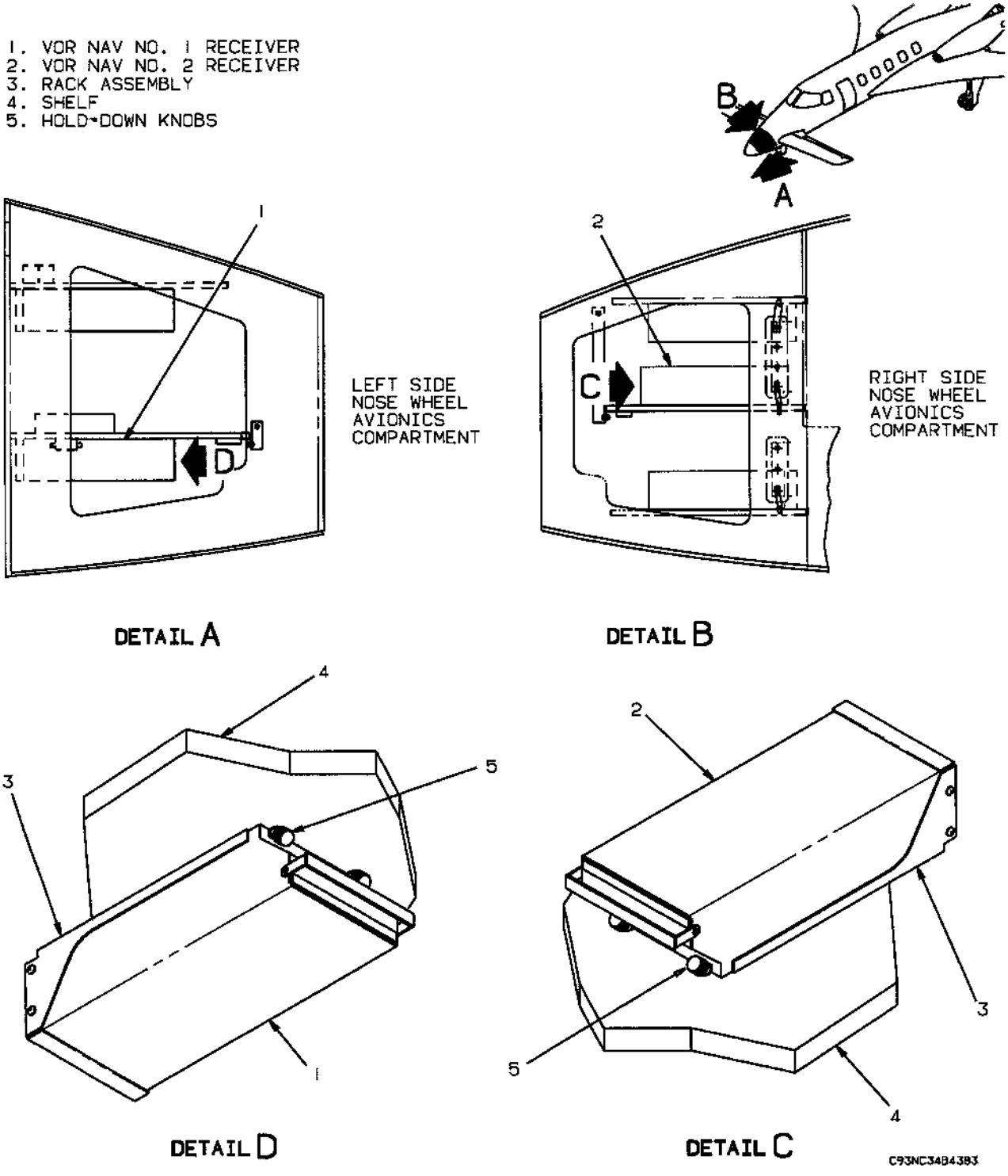
h. Remove the VOR receiver to the workbench or a safe storage area.

VOR NAV RECEIVER INSTALLATION (Effectivity: All) (Figure 2)

a. Place the receiver unit (1 or 2) in the rack assembly (3) and firmly set it in the connector.

b. Swing the latches into place and hand-tighten the retaining knobs (5).

1. VOR NAV NO. 1 RECEIVER
2. VOR NAV NO. 2 RECEIVER
3. RACK ASSEMBLY
4. SHELF
5. HOLD-DOWN KNOBS



VOR NAV Receiver Installation (Effectivity: All)
Figure 2

c. Test the NAV system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28-vdc external power to the airplane.

2. Ramp test the NAV system using the IFR NAV-401, or an equivalent ramp test set (3, Chart 1, 34-00-00). Perform the level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

d. Install the left (710AZ) or right (710BZ) nose wheel well avionics access panel.

e. Remove the landing gear door locking pin and push the doors closed.

VOR COAXIAL COUPLER REMOVAL (Effectivity: All) (Figure 3)

The coaxial coupler connects both navigation receivers to one antenna coaxial cable. The coupler is mounted in the right avionics nose gear wheel well compartment on top of the center shelf.

a. Remove all electrical power from the airplane.

b. Operate the alternate extension hand pump in the cockpit until the forward nose landing gear doors are fully open.

c. Install the locking pin in the landing gear door bellcrank. Refer to Chapter 32-20-00, for pin location.

d. Remove the right nose wheel well access panel (710BZ).

e. Remove the coaxial cable connectors from the coupler and tag.

f. Remove the mounting screws (1) and washers (2, 3, 5 and 6) that secure the coupler (4) to the avionics shelf.

g. Remove the coupler (4) to the workbench or a safe storage area.

VOR COAXIAL COUPLER INSTALLATION (Effectivity: All) (Figure 3)

a. Position the coupler (4) on the shelf and secure with the screw (1) and washers (2, 3, 5, and 6).

b. Identify the coaxial cable connectors and attach them to the coupler.

c. Test the NAV system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28-vdc external power to the airplane.

2. Ramp test the NAV system using the IFR NAV-401, or an equivalent ramp test set (3, Chart 1, 34-00-00). Perform the level-2 VOR test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

d. Install the right nose wheel well avionics access panel (710BZ).

e. Refer to Chapter 32-20-00 and remove the nose landing gear door lockpin, then push the doors closed.

VOR NAV ANTENNA REMOVAL (Effectivity: NC-4 thru NC-24 without BEECHCRAFT Kit No. 122-3020-1 installed, and NC-25 and after without Kit No. 122-3020-3 installed) (Figure 4)

NOTE

Maintenance procedures for the left and right VOR NAV antenna blades are typical. Only the procedures for the left antenna blade are described.

a. Remove all electrical power from the airplane.

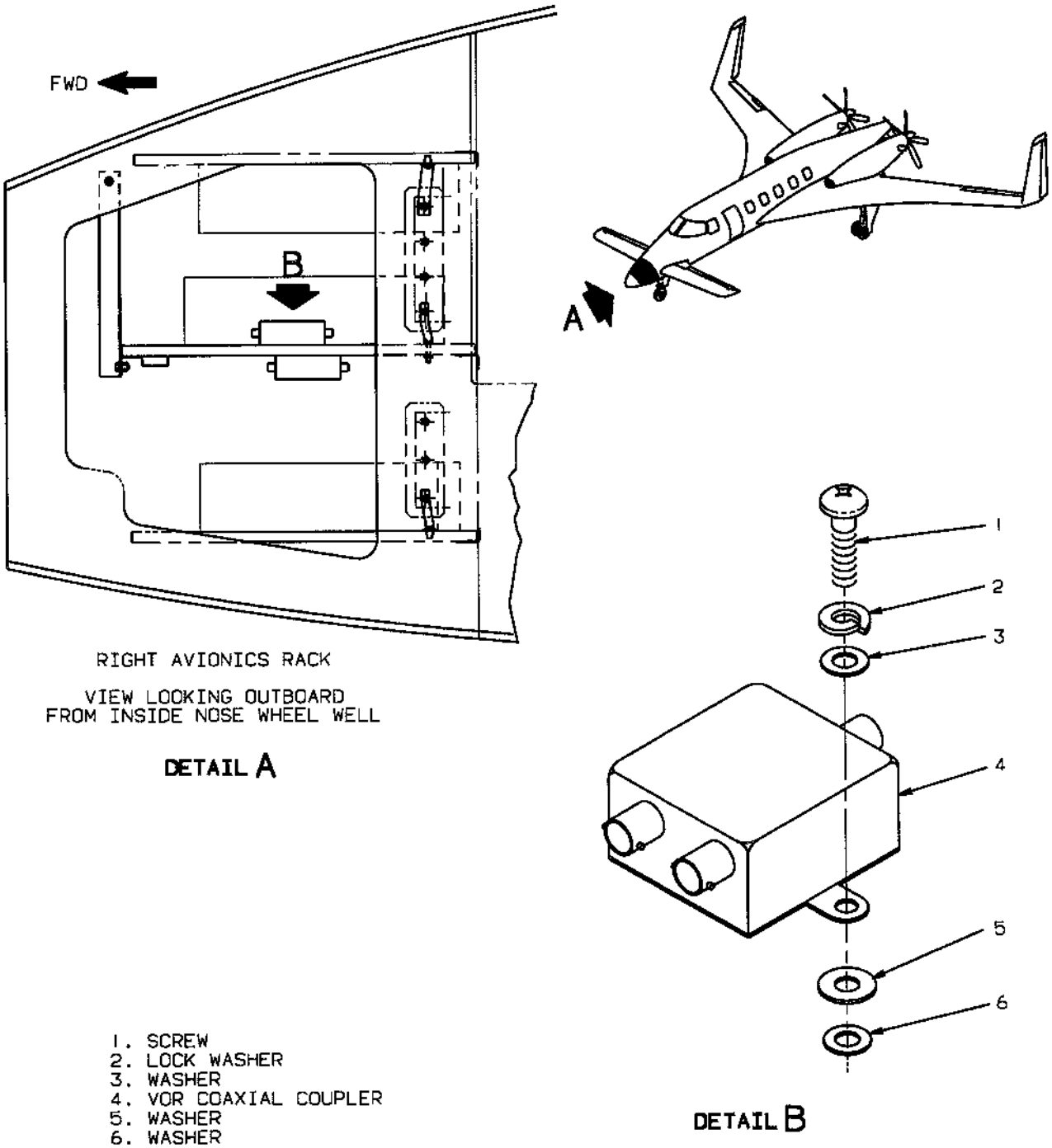
b. Remove the four screws (7) and lockwashers (8) that secure the antenna (6) to the side of the ventral fin (10).

c. Carefully strip the sealer from around the edge of the antenna (6), ventral fin (10) and filler (5) with a nonmetalic scraper.

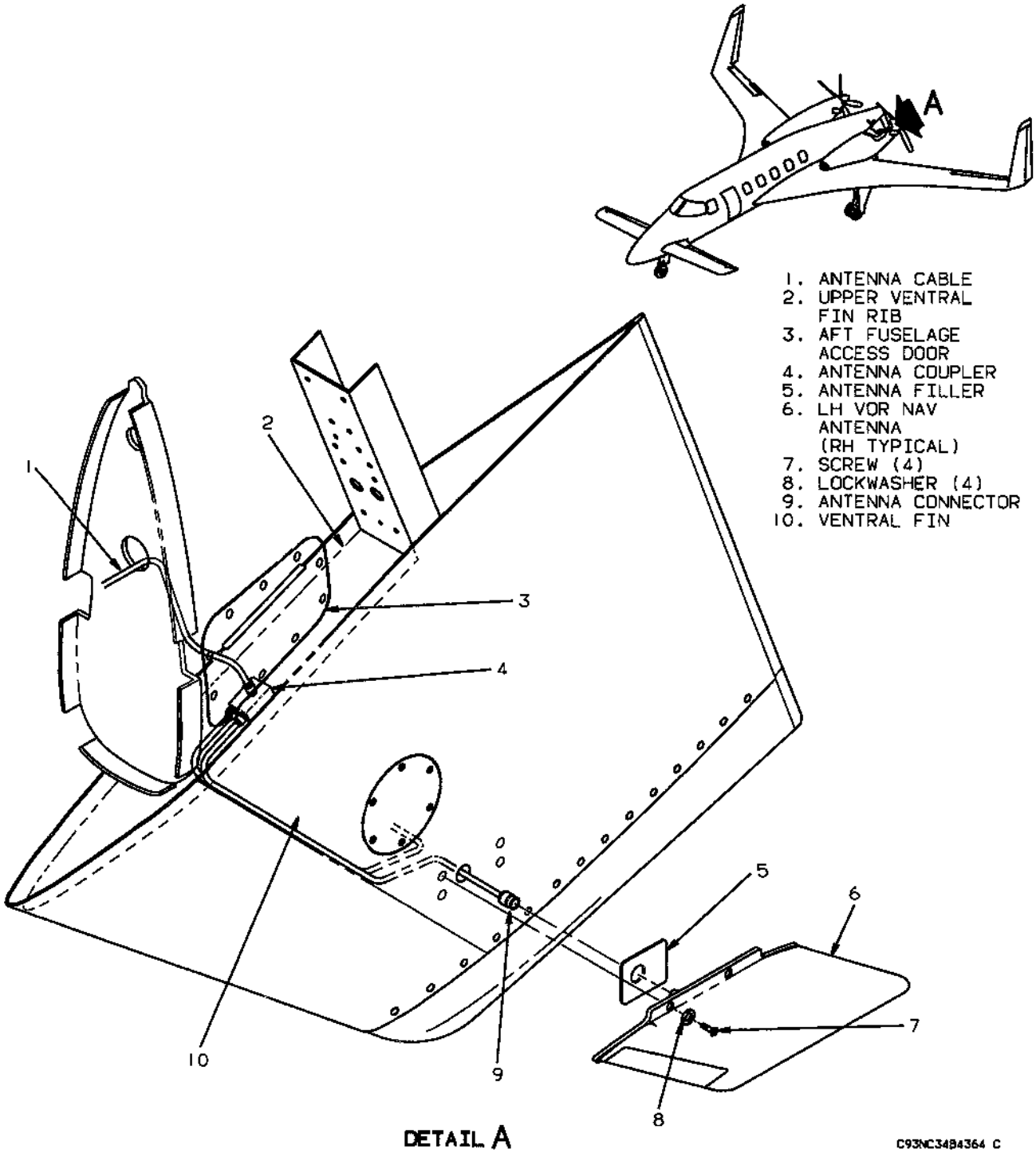
d. Pull the antenna away from the ventral fin to expose the antenna cable and connector (9).

e. Disconnect the antenna connector (9) from the antenna (6) and tag or tape the antenna connector (9) to keep it from falling back into the ventral fin (10).

f. Remove the VOR NAV antenna (6) from the airplane.



VOR Coaxial Coupler Installation
(Effectivity: All)
Figure 3



VOR NAV Antenna Installation (Effectivity: NC-4 thru NC-24 without BEECHCRAFT Kit No. 122-3020-1 installed, and NC-25 and after without BEECHCRAFT Kit No. 122-3020-3 installed)
Figure 4

VOR NAV ANTENNA INSTALLATION (Effectivity: NC-4 thru NC-24 without BEECHCRAFT Kit No. 122-3020-1 installed, and NC-25 and after without BEECHCRAFT Kit No. 122-3020-3 installed) (Figure 4)

NOTE

Maintenance procedures for the left and right VOR NAV antenna blades are typical. Only the procedures for the left antenna blade are described.

a. Remove all the old sealer from the antenna (6), filler (5), and ventral fin (10). Use a nonmetallic scraper and a nonmetallic cleaner pad.

b. Thoroughly clean the contact surface of the fuselage where the VOR antenna is to be attached. Refer to Chapter 20-00-03 COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the mounting surfaces of the antenna (6) and filler (5) to base metal and chemically treat with alodine (3, Chart 2, Chapter 34-00-00) to prevent corrosion. Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

d. Place the filler (5) over the antenna connector (9) and connect the antenna connector (9) to the antenna (6).

e. Position the antenna (6) and filler (5) on the ventral fin (10) and install the four screws (7) and lockwashers (8). Install the screws (7) wet with sealer (2, Chart 2, Chapter 34-00-00).

f. Refer to Chapter 53-30-00 and loosen the captive screws around the aft fuselage access door (3) (330AL) located above the ventral fin and remove the panel (330AL) (3).

g. Check bond resistance for the antenna (6) to an adjacent ground plane per Chapter 20-00-03.

h. Refer to Chapter 53-30-00 and install the fuselage access door (3) (330AL).

i. Test the NAV system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28-vdc external power to the airplane.

2. Ramp test the NAV system using the IFR NAV-401, or an equivalent ramp test set (3, Chart 1, 34-00-00). Perform the level-2 VOR test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

j. Seal around the base of the antenna (6) where it attaches to the ventral fin (10) with sealer (1, Chart 2, Chapter 34-00-00).

k. When the sealer is dry, touch up antenna, filler, and sealer with paint where necessary.

VOR NAV ANTENNA REMOVAL (Effectivity: NC-4 thru NC-24 with BEECHCRAFT Kit No. 122-3020-1 installed, and NC-25 and after with BEECHRAFT Kit No. 122-3020-3 installed) (Figure 5)

NOTE

Maintenance procedures for the left and right VOR NAV antenna blades are typical. Only the procedures for the left antenna blade are described.

a. Remove all electrical power from the airplane.

b. Remove the four screws (8) and lockwashers (9) that secure the antenna (7) to the side of the ventral fin (11).

c. Carefully strip the sealer from around the edge of the antenna (7), ventral fin (11), filler (5) and gasket (6) with a nonmetalic scraper.

d. Pull the antenna (7) away from the ventral fin to expose the antenna cable and connector (10).

e. Disconnect the antenna connector (10) from the antenna (7), then remove the antenna filler (5) and gasket (6). Tape the antenna connector (10) to the side of the ventral fin (11) to prevent the cable from falling inside.

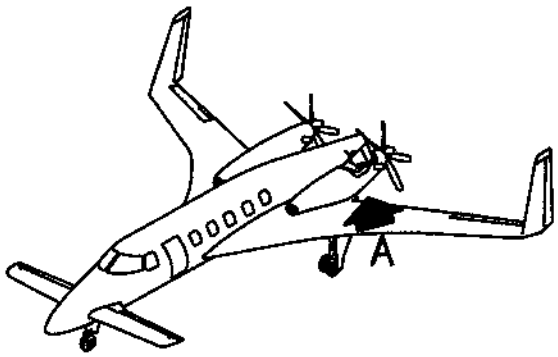
f. Remove the VOR NAV antenna (7) from the airplane.

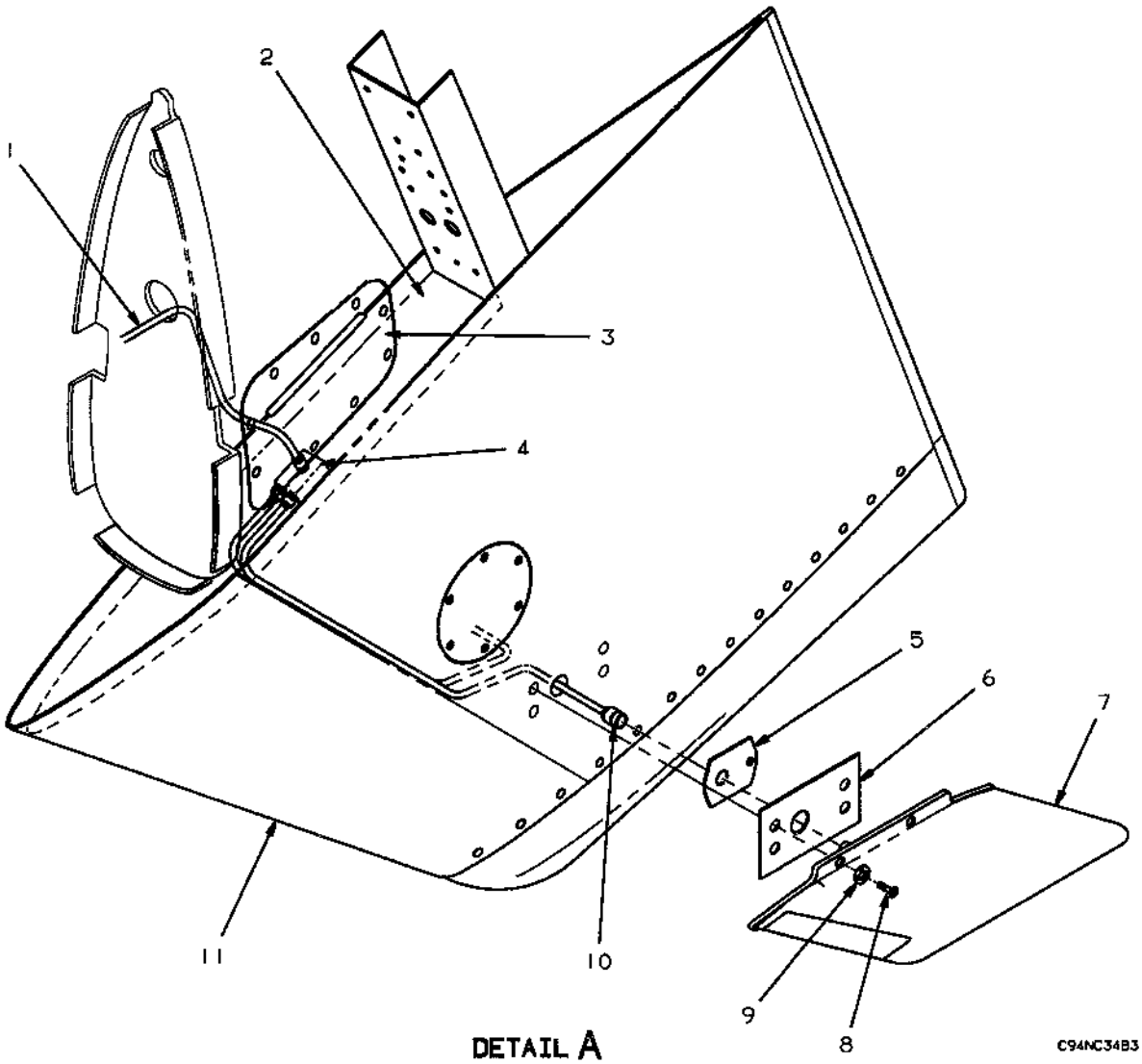
VOR NAV ANTENNA INSTALLATION (Effectivity: NC-4 thru NC-24 with BEECHCRAFT Kit No. 122-3020-1 installed, and NC-25 and after with Kit No. 122-3020-3 installed) (Figure 5)

NOTE

Maintenance procedures for the left and right VOR NAV antenna blades are typical. Only the procedures for the left antenna blade are described.

1. ANTENNA CABLE
2. UPPER VENTRAL FIN RIB
3. AFT FUSELAGE ACCESS DOOR
4. ANTENNA COUPLER
5. ANTENNA FILLER
6. GASKET
7. LH VOR NAV ANTENNA (RH TYPICAL)
8. SCREW (4)
9. LOCKWASHER (4)
10. ANTENNA CONNECTOR
11. VENTRAL FIN





DETAIL A

C94NC34B3109 C

VOR NAV Antenna Installation (Effectivity: NC-4 thru NC-24 with BEECHCRAFT Kit No. 122-3020-1, and NC-25 and after with BEECHCRAFT Kit No. 122-3020-3 installed)
Figure 5

a. Remove all the old sealer from the antenna (7), filler (5), gasket (6) and ventral fin (11). Use a nonmetallic scraper and a nonmetallic cleaner pad.

b. Thoroughly clean the contact surface of the fuselage where the VOR antenna (7) is to be attached. Refer to Chapter 20-00-03 COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the mounting surfaces of the antenna (7) and filler (5) to base metal and chemically treat with alodine (3, Chart 2, Chapter 34-00-00) to prevent corrosion. Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

d. Place the filler (5) and gasket (6) over the antenna connector (10) and connect it to the antenna (7).

e. Align the filler (5), gasket (6) and antenna (7) with the mounting holes in the ventral fin (11), then secure with four screws (8) and lock washers (9). Install the screws (8) wet with sealer (2, Chart 2, Chapter 34-00-00).

f. Refer to Chapter 53-30-00 and loosen the captive screws around the aft fuselage access door (3) (330AL) located above the ventral fin and remove the panel (330AL) (3).

g. Check bond resistance from the antenna (7) to an adjacent ground plane per instructions in Chapter 20-00-03.

h. Refer to Chapter 53-30-00 and install the fuselage access door (3) (330AL).

i. Test the NAV system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28-vdc external power to the airplane.

2. Ramp test the NAV system using the IFR NAV-401, or an equivalent ramp test set (3, Chart 1, 34-00-00). Perform the level-2 VOR test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

j. Seal around the base of the antenna (7) where it attaches to the ventral fin (11) with sealer (1, Chart 2, Chapter 34-00-00).

k. When the sealer is dry, touch up antenna, filler, and sealer with paint where necessary.

VOR ANTENNA COUPLER REMOVAL
(Effectivity: All)
(Figure 6)

NOTE

The antenna coupler is mounted inside the aft fuselage on the top ventral fin rib.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the access door (2) (330AL) on the left side of the tail cone above the ventral fin (7).

c. Reach inside the tail cone and disconnect the three coaxial connectors from the coupler (6). Identify and tag each coaxial connector as it is removed.

d. Remove the two screws (3) and washers (4, and 5) that secure the antenna coupler (6) to the top ventral fin rib (1).

e. Remove the VOR antenna coupler (6) to the workbench or a safe storage area.

VOR ANTENNA COUPLER INSTALLATION
(Effectivity: All)
(Figure 6)

a. Position the antenna coupler (6) on the upper ventral fin rib (1).

b. Secure the antenna coupler (6) to the rib with the two screws (3), washers (5) and lockwashers (4).

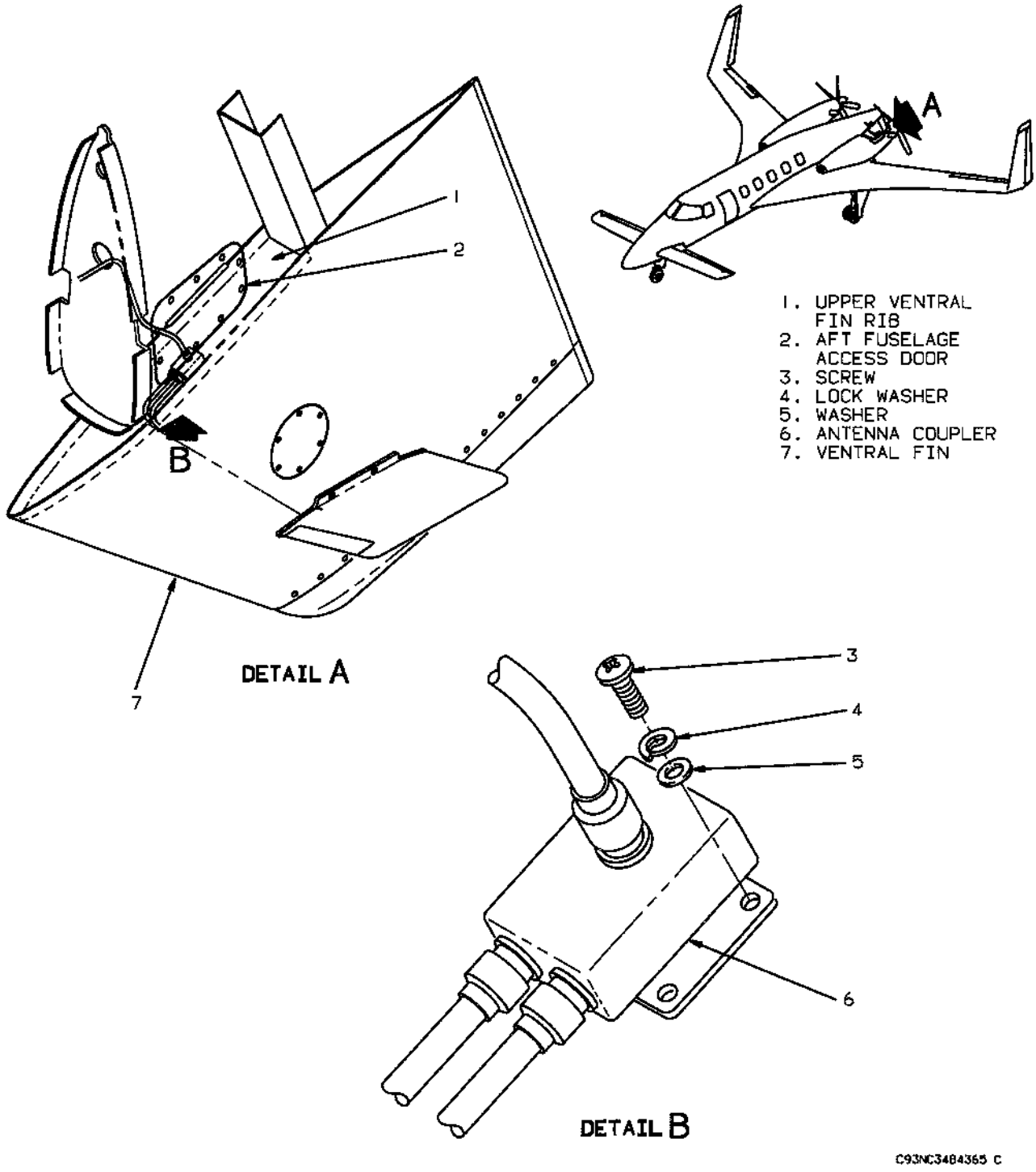
c. Attach the coaxial connectors to the antenna coupler (6) as indicated by the identification tags.

d. Test the NAV system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28-vdc of external power to the airplane.

2. Ramp test the NAV system using the IFR NAV-401, or an equivalent ramp test set (3, Chart 1, 34-00-00). Perform the level-2 VOR test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

e. Refer to Chapter 53-30-00 and install the the aft fuselage access door (330AL) (2) on the fuselage.



VOR Antenna Coupler Installation (Effectivity: All)
Figure 6

DME TRANSCEIVER REMOVAL (Effectivity: All) (Figure 7)

NOTE

Maintenance procedures are typical for the two DME transceivers. The DME 1 transceiver is located in the left avionics compartment and DME 2 transceiver is located in the right avionics compartment.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the the left (213AL) or right (214AR) nose avionics compartment door.

c. Disconnect the antenna coaxial connector from the transceiver (3).

d. Loosen the retainer knobs (1) on the latches and swing the knobs (1) away from the transceiver (3).

e. Pull out on the transceiver (3) to disengage the rack connector and slide the transceiver out of the rack (2).

f. Remove DME transceiver (3) from the airplane.

DME TRANSCEIVER INSTALLATION (Effectivity: All) (Figure 7)

a. Slide the DME Transceiver (3) into the rack (2) and make sure the connectors are engaged.

b. Install the two retainer knobs (1) on the latches and hand-tighten the retainer knobs (1).

c. Reconnect the antenna coaxial connector.

d. Test the DME for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the DME Transceivers with the ATC-600 (3, Chart 1, 34-00-00) or an equivalent ramp test set. Perform the level-2 DME test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

e. Refer to Chapter 53-30-00 and install the left (213AL) or right (214AR) nose avionics compartment door.

DME ANTENNA REMOVAL (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 8)

NOTE

Maintenance procedures for both the forward and aft DME antennas are typical.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-20-00 and remove the center aisle carpet.

c. Refer to Chapter 53-20-00 and remove the forward center aisle floorboard (163ATC) to gain access to the forward DME antenna cable (2), or remove the aft center aisle floorboard (173ATC) to gain access to the aft DME antenna cable (2).

d. Disconnect the antenna cable (2) from the DME antenna (4) and tag the cable (2) for identification.

e. Locate the DME antenna (4) on the underside of the fuselage and remove the six screws (5) that secure the DME antenna (4) to the fuselage.

f. Carefully strip sealer from around the edge of the antenna with a nonmetallic scraper. Avoid damage to the fuselage, paint, or antenna.

g. Remove the antenna from the airplane.

DME ANTENNA INSTALLATION (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 8)

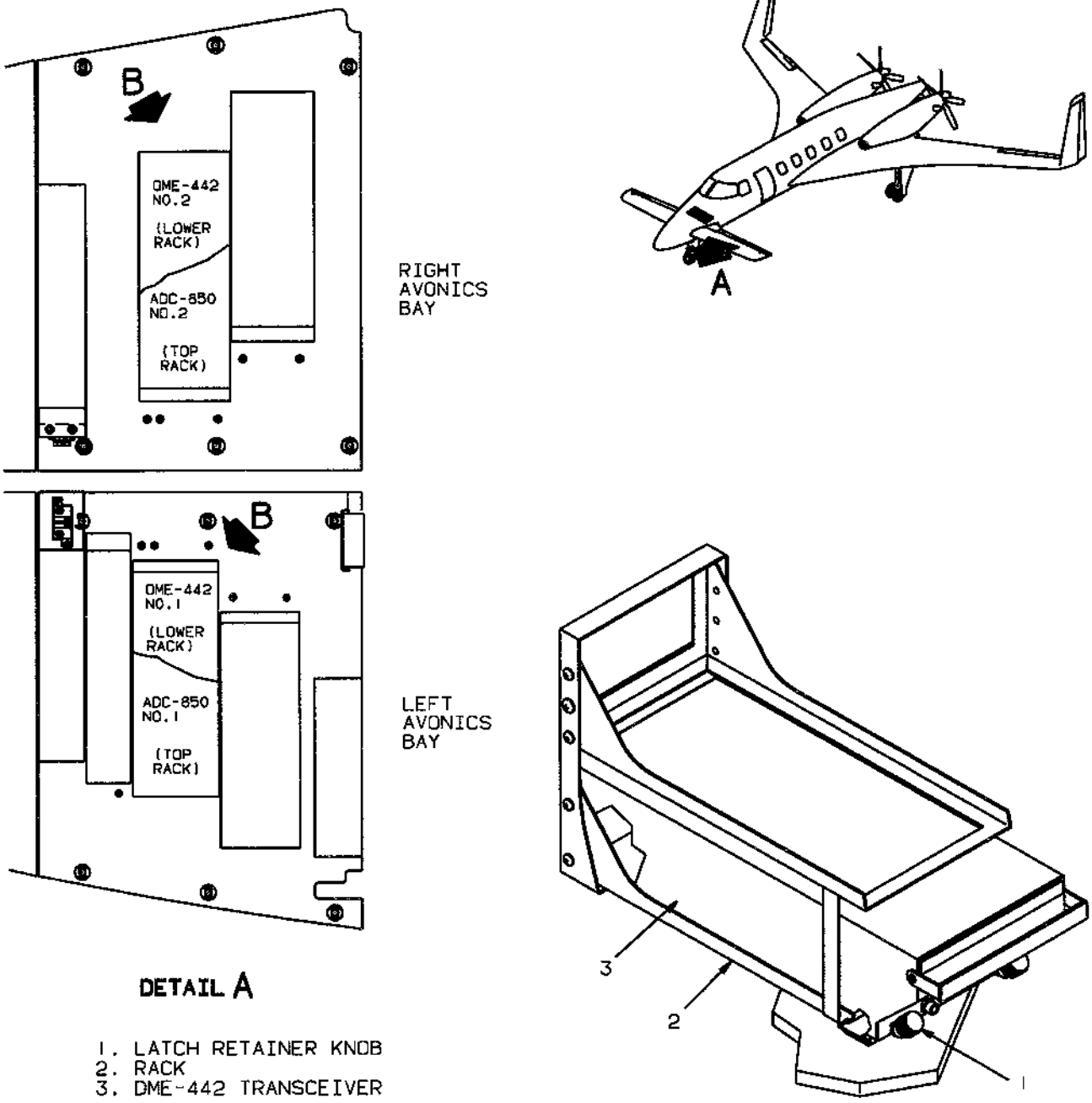
a. Remove all the old sealer from the fuselage (3), antenna (4), and reinforcement assembly (1) contact area with a nonmetallic scraper and nonmetallic cleaner pad.

b. Thoroughly clean the contact surface of the fuselage (3) where the DME antenna is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the inside of the fuselage skin where the reinforcment assembly is to be placed.

d. Have an assistant place the antenna reinforcement assembly (1) on the inside of the fuselage where the DME antenna is to be attached.

e. Position the DME antenna (4) on the outside of the fuselage skin and secure with the six screws (5).



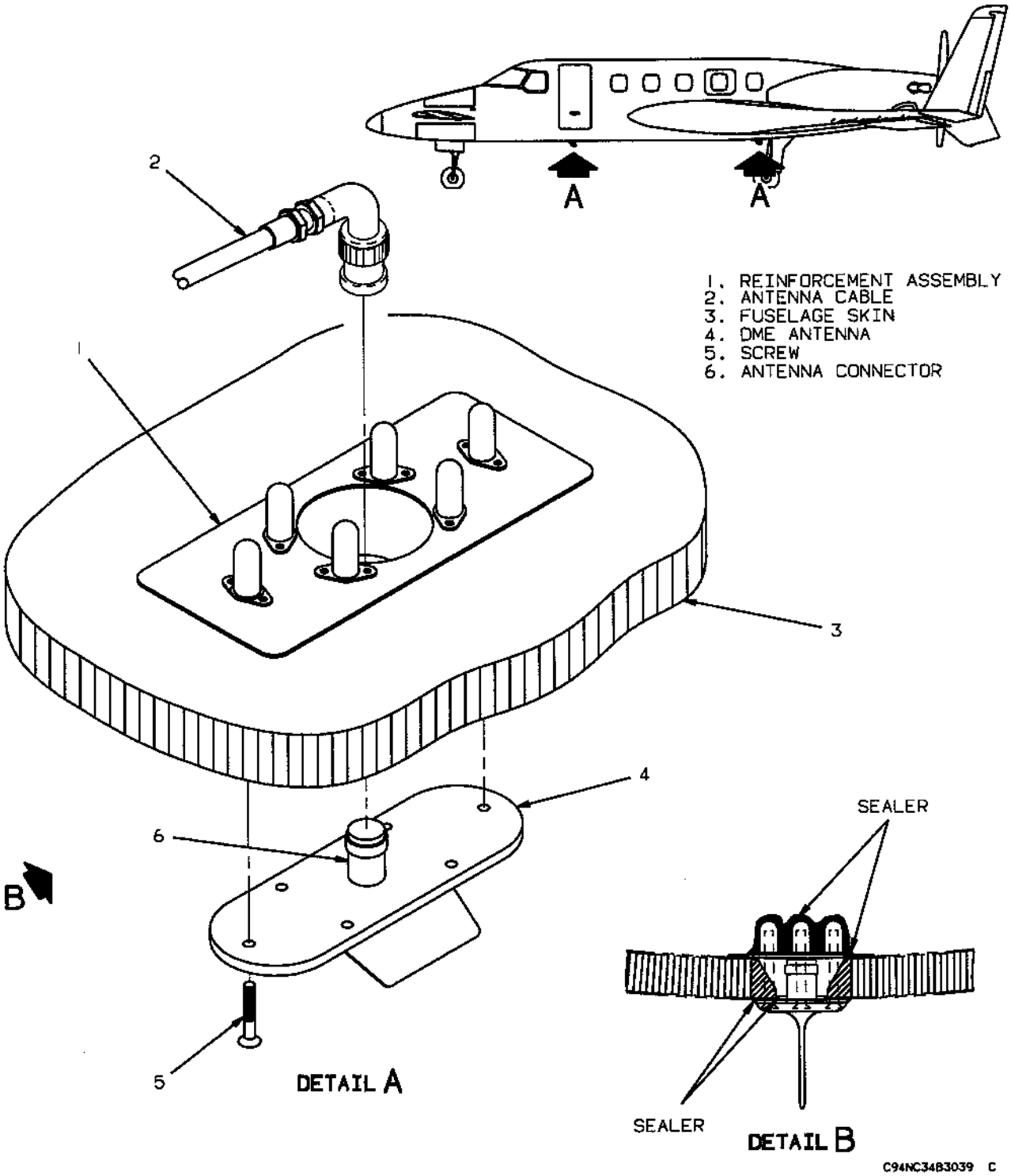
DETAIL A

1. LATCH RETAINER KNOB
2. RACK
3. DME-442 TRANSCEIVER

DETAIL B

C93NC34B4371 C

DME Transceiver Installation
(Effectivity: All)
Figure 7



DME Antenna Installation (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 8

Install the screws wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

f. Check the bond resistance for the antenna as indicated in Chapter 20-00-03.

g. Connect the antenna cable (2) to the DME antenna connector (6).

h. Test the DME antenna for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the DME system with the ATC-601 (3, Chart 1, 34-00-00), or an equivalent ramp test set. Perform the level-2 DME test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

i. From the inside of the cabin, apply sealant (1, Chart 2, Chapter 34-00-00) around the inside of the antenna fuselage opening where the antenna joins the fuselage skin as shown in Detail B, Figure 9. Apply sealant around the periphery of the reinforcement plate assembly and completely cover each nut plate with sealant.

j. Refer to Chapter 53-20-00 and install the forward center aisle floorboard (163ATC) or the aft center aisle floorboard (173ATC)

k. Refer to Chapter 25-20-00 and install the center aisle carpet.

l. From the outside of the airplane, apply sealant (1, Chart 2, Chapter 34-00-00) around the periphery of the DME antenna base.

m. When the sealer has cured, touch up the antenna and fuselage with paint as necessary.

DME ANTENNA REMOVAL (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 9)

NOTE

Maintenance procedures for both the forward and aft DME antennas are typical.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-20-00 and remove the center aisle carpet.

c. Refer to Chapter 53-20-00 and remove the forward center aisle floorboard (163ATC) to gain access to the forward DME coaxial cable (1), or remove the aft center aisle floorboard (173ATC) to gain access to the aft DME coaxial cable (1).

d. Cut and remove tie-wrap (2), then remove sealant from the boot opening.

CAUTION

Use care to avoid damaging the coaxial cable (1), connector (8) and surrounding surface of the reinforcement plate assembly (4).

e. Slice open the antenna boot (3) as necessary to gain access to the antenna connector (8).

f. Disconnect the coaxial cable (1) from the DME antenna (6) and tag the coaxial cable (1) for identification.

g. Locate the DME antenna (6) on the underside of the fuselage and remove the six screws (7) that secure it to the fuselage.

h. Carefully strip sealer from around the edge of the antenna with a nonmetallic scraper. Use care not to damage the fuselage, paint or antenna.

i. Remove the antenna from the airplane.

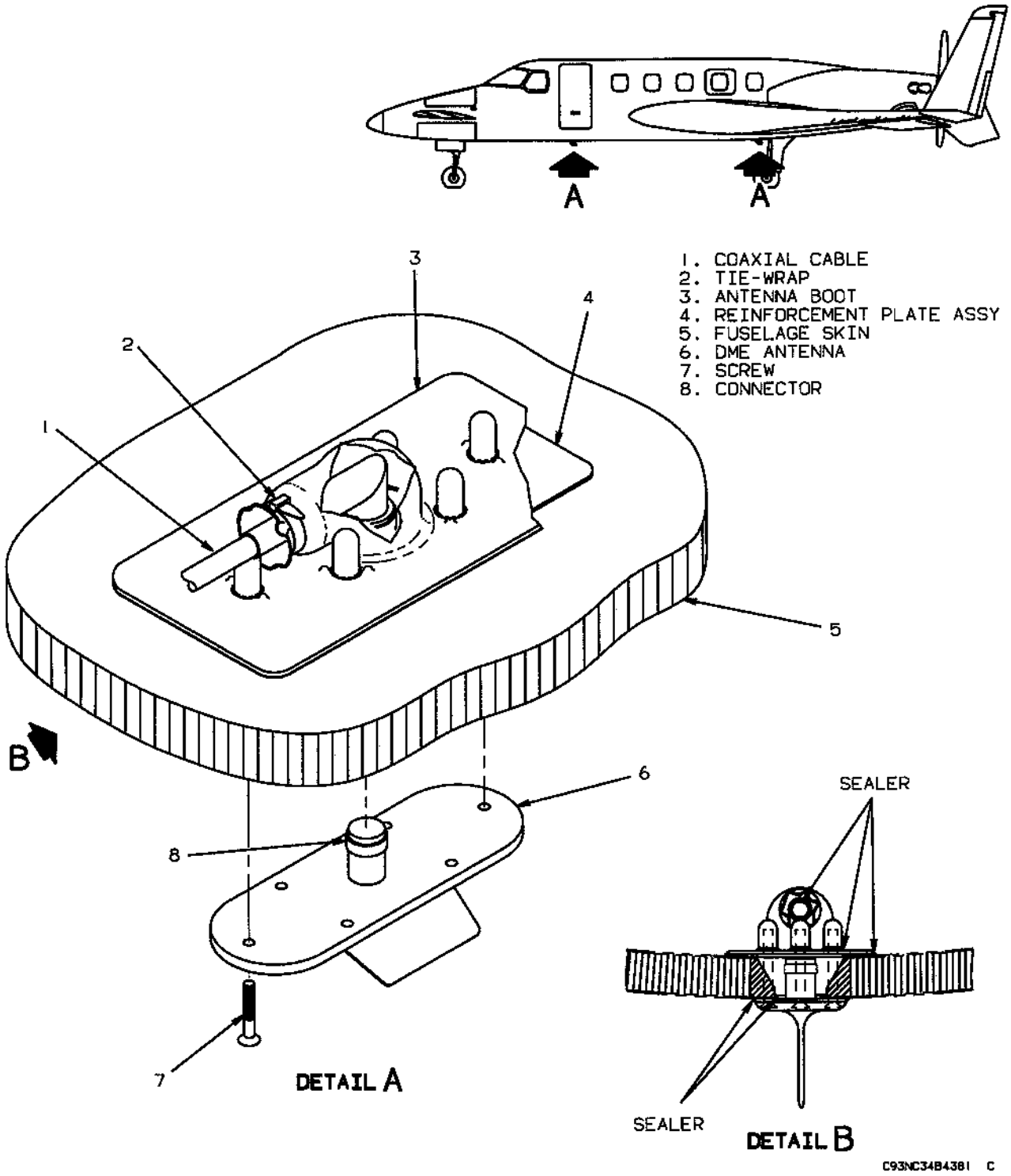
NOTE

If the antenna boot is not damaged it may be reused. However, a cut or torn boot must be replaced, as they are not repairable.

DME ANTENNA INSTALLATION (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 9)

a. Remove all the old sealer from the fuselage skin (5), DME antenna (6) and reinforcement plate assembly (4) contact area with a nonmetallic scraper and nonmetallic cleaner pad.

b. Thoroughly clean the contact surface of the fuselage skin (5) where the DME antenna (6) is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.



DME Antenna Installation (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 9

c. Thoroughly clean the inside of the fuselage skin (5) where the reinforcment plate assembly (4) will be positioned.

d. Have an assistant place the reinforcement plate assembly (4) on the inside of the fuselage where the DME antenna (6) is to be attached.

e. Position the DME antenna (6) on the outside of the fuselage skin (5) and secure with the six screws (7). Install the screws wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

f. Check the bond resistance of the antenna installation as outlined in Chapter 20-00-03.

g. Connect the coaxial cable (1) to the antenna (6) to perform an operational test of the antenna as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the DME system with the ATC-601 (3, Chart 1, 34-00-00), or an equivalent ramp test set. Perform the level-2 DME test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

h. Remove electrical power from the airplane.

i. Disconnect the coaxial cable (1) from the DME antenna (6).

j. From inside the cabin, apply sealant (1, Chart 2, 34-00-00) around the inside of the fuselage antenna opening where the DME antenna (6) and fuselage are joined together, as shown by Detail B of Figure 9.

NOTE

If a new antenna boot (3) is to be installed, carefully trim the new boot to match the reinforcement plate assembly (4) and nut plates before performing the remaining installation procedures.

k. Insert the coaxial cable (1) into the antenna boot (3) opening and carefully push the connector (8) out the bottom.

l. Connect the coaxial cable (1) to the DME antenna (6).

m. Bond the antenna boot (3) to the reinforcement plate assembly (4) with sealant (4, Chart 2, 34-00-00).

n. Apply sealant (4, Chart 2, 34-00-00) to the periphery of the reinforcement plate assembly (4) and antenna boot (3) to provide a good moisture barrier between the antenna mounting assembly and fuselage skin (5).

o. Completely cover the nut plates on the reinforcement plate assembly (4) with sealant (4, Chart 2, 34-00-00).

p. Apply sealant (4, Chart 2, 34-00-00) approximately 1/2 inch into the opening where the coaxial cable (1) exits the antenna boot (3).

q. Install a tie-wrap (2) around the boot neck and coaxial cable (1).

r. Refer to Chapter 53-20-00 and install the forward center aisle floorboard (163ATC) or the aft center aisle floorboard (173ATC) as applicable.

s. Refer to Chapter 25-20-00 and install the center aisle carpet.

t. From the outside of the airplane, apply sealant (1, Chart 2, Chapter 34-00-00) around the periphery of the DME antenna base.

u. When the sealer has cured, touch up the antenna and fuselage with paint as necessary.

v. Restore electrical power to the airplane.

ADF RECEIVER REMOVAL (Effectivity: All) (Figure 10)

NOTE

The ADF receiver is mounted in the right nose wheel well avionics compartment.

a. Remove all electrical power from the airplane.

b. Operate the alternate extension hand pump in the cockpit until the forward nose landing gear doors are fully open.

c. Install the locking pin in the nose gear door bellcrank. Refer to Chapter 32-20-00 for lockpin procedure.

d. Remove the right nose wheel well access panel (710BZ).

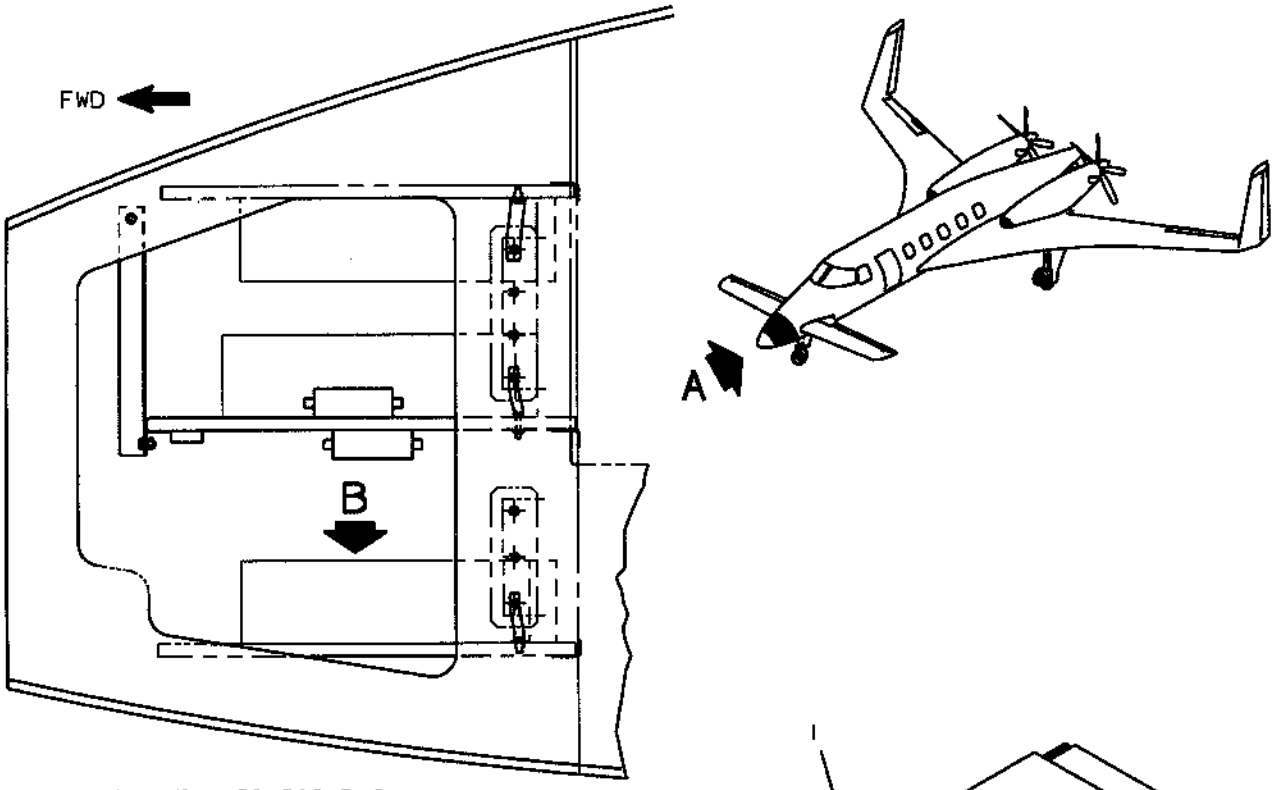
e. Loosen the retainer knobs (3) that secure the receiver (1), and swing the knobs (3) away from the latches.

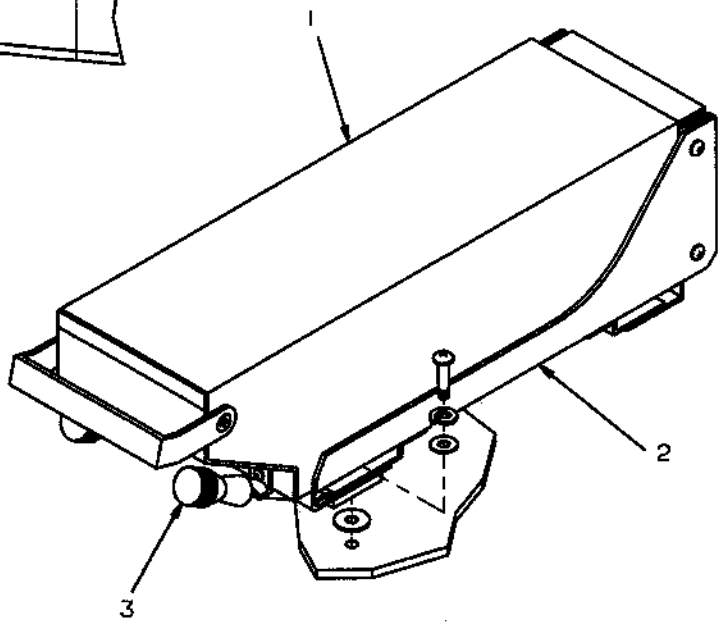
f. Pull the receiver (1) away from the connector.

g. Lift the receiver (1) from the rack (2) and remove it from the airplane.

ADF RECEIVER INSTALLATION (Effectivity: All) (Figure 10)

a. Position the receiver (1) in the rack (2) and make sure the connector is firmly seated.





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ADF Receiver Installation (Effectivity: All)
Figure 10

b. Install the knobs (3) on the latches and hand-tighten the retainer knobs (3).

c. Test the ADF system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the ADF for proper operation by tuning to a local AM radio station or local NDB. The pilot radio tuning unit (RTU) controls the ADF. Check for bearing pointer movement and check for audio reception.

NOTE

For an alternate method of testing, Refer to the level-2 test procedures contained in latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

d. Install the right nose wheel well access panel (710BZ).

e. Refer to Chapter 32-20-00 and remove the nose landing gear door lockpin and push the doors closed.

ADF ANTENNA REMOVAL (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 11)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-20-00 and remove the center aisle carpet.

c. Refer to Chapter 53-20-00 and remove the forward center aisle floorboard (163ATC) to gain access to the ADF antenna connector (5), antenna cable (2) and doubler assembly (1 and 13).

d. Disconnect the antenna cable (2) from the antenna connector (5) and tag the antenna cable (2) for identification.

e. From the underside of the fuselage, remove the four screws (7 and 10), washers (8), and lock washers (9), that secure the ADF antenna (6) to the fuselage skin (3).

f. Carefully strip the sealer from around the edge of the antenna (6) with a nonmetallic scraper. Avoid damage to the fuselage skin (3), paint, gasket, or antenna (6).

g. Remove the ADF antenna (6), spacers (4, and 11), gasket (12), and doubler assemblies (1 and 13) from the airplane.

ADF ANTENNA INSTALLATION (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-0314-1 or -3 installed) (Figure 11)

a. Remove all the old sealer from the ADF antenna (6), gasket (12), doubler assemblies (1 and 13), and fuselage skin (3) using a nonmetallic scraper and nonmetallic cleaner pad.

b. Clean the contact surfaces of the fuselage skin (3) thoroughly where the ADF antenna (6) is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the ADF antenna (6), spacers (4 and 11) contact surfaces and chemically treat with alodine (3, Chart 2, Chapter 34-00-00). Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

d. Thoroughly clean the doubler assemblies (1 and 13). Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

e. Have an assistant hold the doubler assemblies (1 and 13) on the inside of the fuselage skin (3).

f. Place the antenna (6), spacers (4 and 11), and gasket (12) in place on the outside of the fuselage skin (3) as shown in Figure 11. Secure with the four screws (7 and 10), washers (8), and lock washers (9). Install the screws (7 and 10) wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

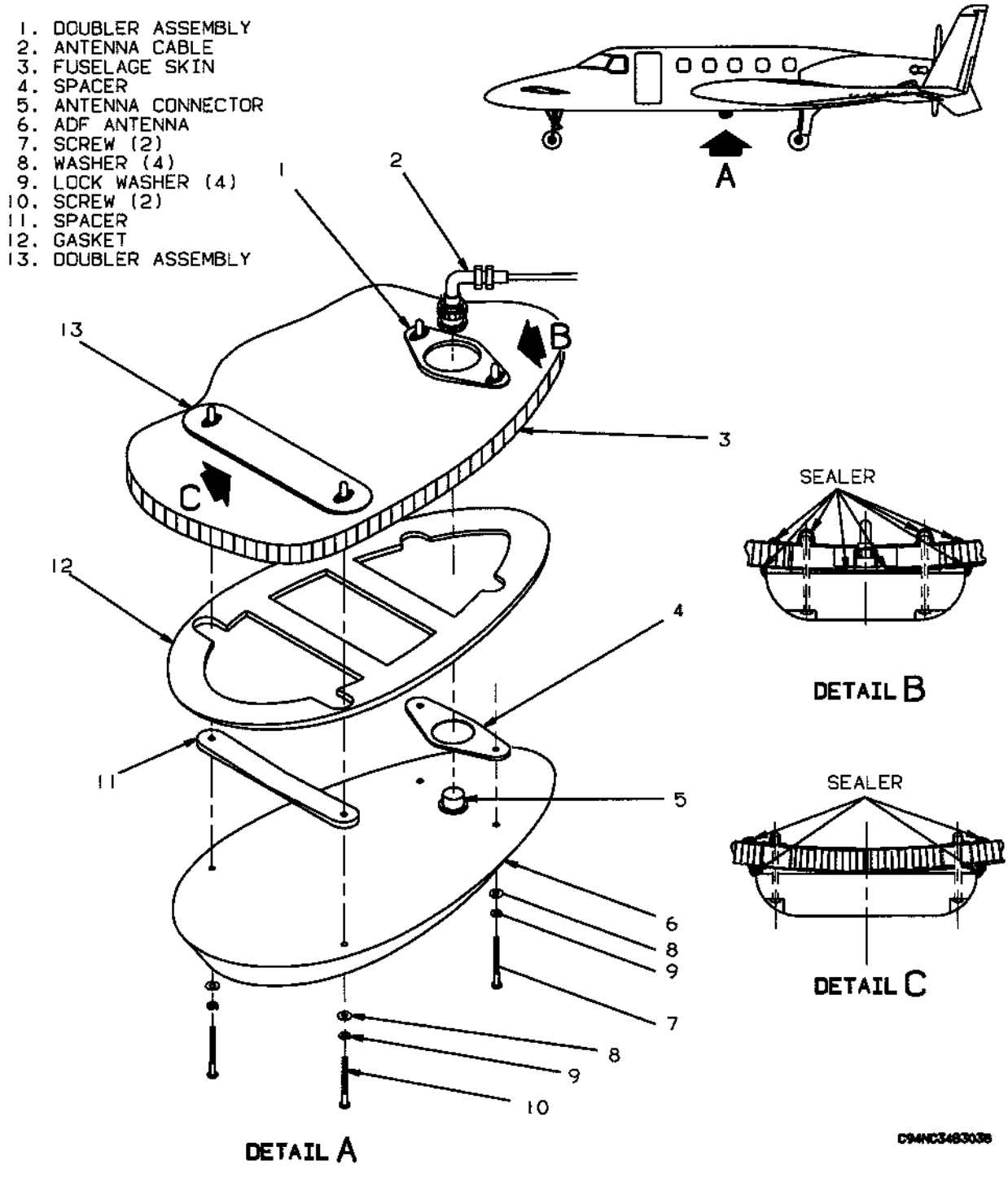
g. Check the bond resistance for the antenna as indicated in Chapter 20-00-03.

h. After the resistance check, apply sealer (1, Chart 2, chapter 34-00-00) from inside the cabin around the inside antenna fuselage opening where the antenna base contacts the fuselage skin (3). Apply sealer around the periphery of the doubler assemblies (1 and 13) as shown in Detail B and Detail C of Figure 11. Completely cover the nut plates on both doubler assemblies (1 and 13) with sealer.

i. Connect the antenna cable (2) to the ADF antenna connector (5).

j. Test the ADF system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.



ADF Antenna Installation (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 11

2. Ramp test the ADF antenna for proper operation by tuning to a local AM radio station or local NDB. The pilot's radio tuning unit (RTU) controls the ADF. Check for bearing pointer movement and check for audio reception.

NOTE

For an alternate method of testing, refer to the level-2 test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

k. Refer to Chapter 53-20-00 and install the forward center aisle floorboard (163ATC).

l. Refer to Chapter 25-20-00 and install the center aisle carpet.

m. On the outside of the fuselage skin (3), apply sealer (1, Chart 2, Chapter 34-00-00) around the periphery of the ADF antenna (6).

n. When the sealer has cured, touch up the antenna and fuselage area with paint as required.

ADF ANTENNA REMOVAL (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 12)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-20-00 and remove the center aisle carpet.

c. Refer to Chapter 53-20-00 and remove the forward center aisle floorboard (163ATC) to gain access to the ADF antenna connector (5), antenna cable (2) and doubler assembly (1 and 13).

d. Cut and remove the tie-wrap (15), then remove sealant from the boot opening.

e. Slice open the antenna boot (14) as necessary to gain access to the antenna connector (5)

CAUTION

Use care to avoid damaging the antenna cable (2), connector (5) and surrounding surface of the doubler assembly (1).

f. Disconnect the antenna cable (2) from the antenna connector (5) and tag the antenna cable (2) for identification.

g. From the underside of the fuselage, remove the four screws (7 and 10), washers (8), and lock washers (9) that secure the ADF antenna (6) to the fuselage skin (3).

h. Carefully strip the sealer from around the edge of the antenna (6) with a nonmetallic scraper. Avoid damage to the fuselage skin (3), paint, gasket or antenna (6).

i. Remove the ADF antenna (6), spacers (4, and 11), gasket (12), and doubler assemblies (1 and 13) from the airplane.

NOTE

If the antenna boot is not damaged it may be reused. However, a cut or torn boot must be replaced as they are not repairable.

ADF ANTENNA INSTALLATION (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-0314-1 or -3 installed) (Figure 12)

a. Remove all the old sealer from the ADF antenna (6), gasket (12), doubler assemblies (1 and 13) and fuselage skin (3) using a nonmetallic scraper and nonmetallic cleaner pad.

b. Clean the contact surfaces of the fuselage skin (3) thoroughly where the ADF antenna (6) is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the ADF antenna (6), spacers (4 and 11) contact surfaces and chemically treat with alodine (3, Chart 2, Chapter 34-00-00). Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

d. Thoroughly clean the doubler assemblies (1 and 13). Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

e. Have an assistant hold the doubler assemblies (1 and 13) on the inside of the fuselage skin (3).

f. Position the antenna (6), spacers (4 and 11), and gasket (12) in place on the outside of the fuselage skin (3) as shown in Figure 12. Secure with the four screws (7 and 10), washers (8) and lock washers (9). Install the screws (7 and 10) wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

g. Check the bond resistance for the antenna as indicated in Chapter 20-00-03.

h. Connect the antenna cable (2) to the ADF antenna (6) to perform a ramp test of the antenna. Test the ADF antenna for proper operation as follows:

1. Tune to a local AM radio station or local NDB. The pilot's radio tuning unit (RTU) controls the ADF.

2. Verify bearing pointer movement and check for audio reception.

NOTE

For an alternate method of testing, refer to the level-2 test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

i. Remove electrical power from the airplane.

j. Disconnect the antenna cable (2) from the ADF antenna (6).

k. From inside the cabin, apply sealer (1, Chart 2, Chapter 34-00-00) around the inside antenna fuselage opening where the antenna base contacts the fuselage skin (3) as shown by Detail B of Figure 12.

l. Connect the antenna cable (2) to the ADF antenna (6).

NOTE

If a new antenna boot (14) is to be installed, carefully trim the new boot to match the doubler assembly (1) and nut plates before performing the remaining installation procedures.

m. Insert the antenna cable (2) into the antenna boot (14) opening and carefully push the connector out the bottom.

n. Connect the antenna cable (2) to the antenna connector (5).

o. Bond the antenna boot (14) to the doubler assembly (13) with sealant (4, Chart 2, 34-00-00).

p. Apply sealant (4, Chart 2, 34-00-00) to the periphery of the doubler assembly (1) and antenna boot (14) to provide a good moisture barrier between the antenna mounting assembly and fuselage skin (3).

q. Completely cover the nut plates on the doubler assembly (1) with sealant (4, Chart 2, 34-00-00).

r. Apply sealant (4, Chart 2, 34-00-00) approximately 1/2 inch into the opening where the antenna cable (2) exits the antenna boot.

s. Install a tie-wrap (15) around the boot neck and antenna cable (2).

t. Refer to Chapter 53-20-00 and install the forward center aisle floorboard (163ATC).

u. Refer to Chapter 25-20-00 and install the center aisle carpet.

v. On the outside of the fuselage skin (3), apply sealer (1, Chart 2, Chapter 34-00-00) around the periphery of the ADF antenna (6).

w. When the sealer has cured, touch up the antenna and fuselage area with paint as required.

x. Restore electrical power to the airplane.

VLF RECEIVER REMOVAL (Effectivity: All) (Figure 13)

NOTE

The VLF receiver is mounted in the left avionics compartment in the nose.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the left nose avionics compartment panel (213AL).

c. Remove and tag the electrical connector.

d. Loosen the retainer knob (2) on the latch and swing the retainer knob (2) away from the latch.

e. Pull out on the VLF receiver (1) away from the connector and slide the receiver (1) out from the rack (3).

f. Remove the VLF receiver (1) from the airplane.

VLF RECEIVER INSTALLATION (Effectivity: All) (Figure 13)

a. Slide the VLF receiver (1) into the rack (3) until it is firmly seated in the connector.

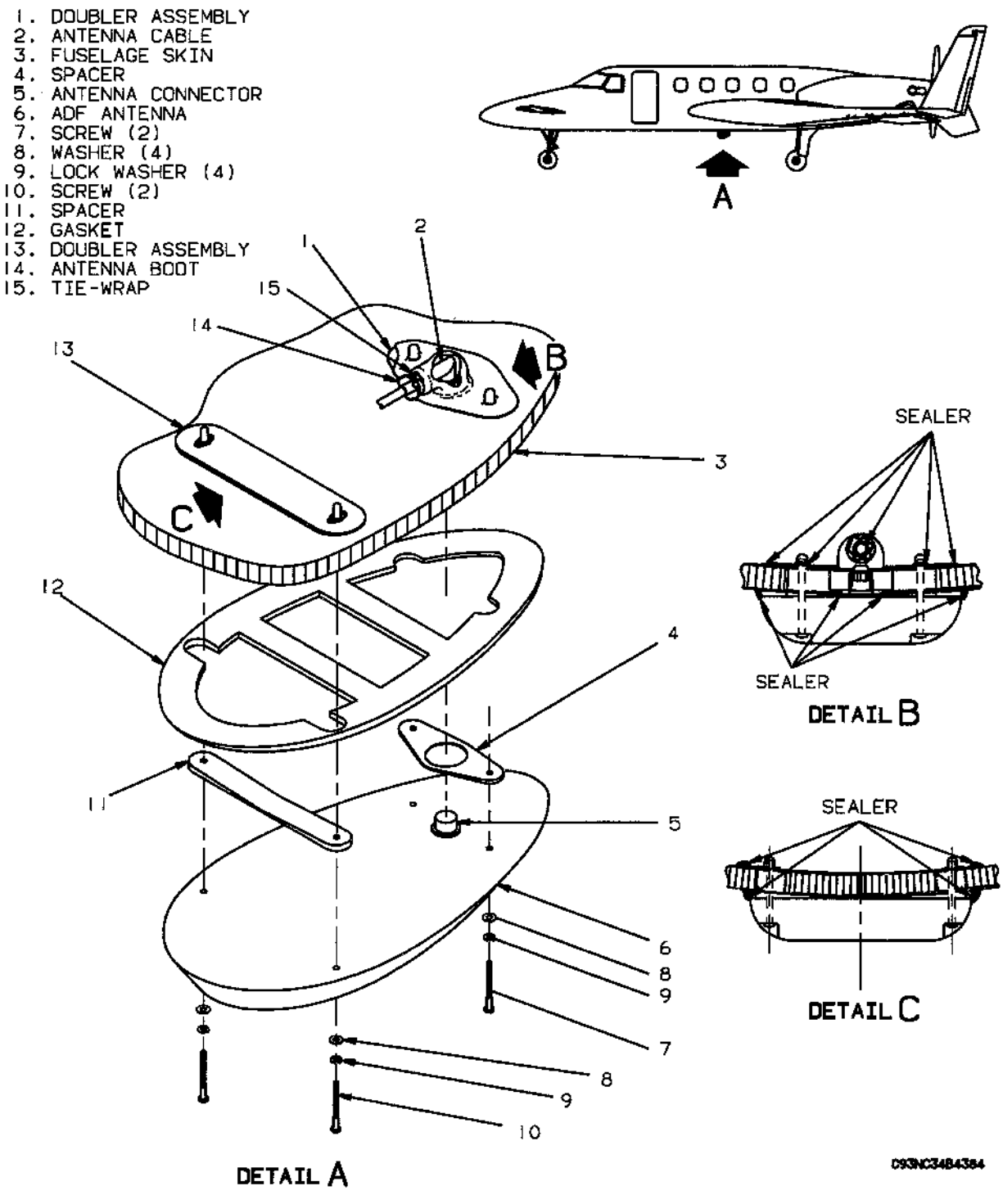
b. Install the retainer knob (2) on the latch and hand-tighten the retainer knob (2).

c. Connect the electrical connector.

d. Test the VLF system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the VLF receiver using the level-2 test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.



ADF Antenna Installation (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 12

e. Refer to Chapter 53-30-00 and install the left nose avionics compartment panel (213AL).

VLF ANTENNA REMOVAL (Effectivity: All) (Figure 14)

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-50-00 and remove the aft wing box shear panel (528AB).

c. Remove the eight screws (6 and 7) that secure the VLF antenna (3) to the aft wing box shear panel (5).

d. Carefully strip the sealer from around the edges of the VLF antenna (3) and wing box shear panel area of contact with a nonmetallic scraper. Avoid damage to the antenna or the wing-box shear panel.

e. Remove the VLF antenna (3) from the wing box shear panel (528AB) (5).

VLF ANTENNA INSTALLATION (Effectivity: All) (Figure 14)

a. Remove all the old sealer from the VLF antenna (3), doubler assembly (4), and wing box shear panel (528AB) (5) using a nonmetallic scraper and nonmetallic cleaner pad.

b. Thoroughly clean the contact surface of the wing box shear panel (5) where the VLF antenna (3) is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the metal contact surfaces of the wing box shear-panel ground plate, the antenna doubler assembly (4), and the VLF antenna-to-base metal, then chemically treat with alodine (3, Chart 2, Chapter 34-00-00). Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

d. Have an assistant hold the doubler assembly (4) on the upper side of the wing box shear panel (5), and place the antenna on the bottom side of the shear panel (5) and secure with the seven longer screws (6) around the VLF antenna (3), with the one shorter screw (7) installed under the antenna connector.

e. Check the bond resistance for the VLF antenna (3) as indicated in Chapter 20-00-03.

f. After the resistance check, apply sealant (1, Chart 2, Chapter 34-00-00) around the periphery of the doubler assembly (4) and around the inside of the doubler assembly (4) where the doubler meets the antenna housing (Detail B of Figure 13). Apply sealant (1, Chart 2, Chapter 34-00-00) on each nut plate of the doubler assembly (4) to completely cover the nut plate.

g. Refer to Chapter 53-30-00 and install the aft wing box shear panel (528AB) on the fuselage. Ensure that the bonding strap is connected to the panel.

h. Seal around the periphery of the VLF antenna (3) with sealant (1, Chart 2, Chapter 34-00-00). Refer to Detail B, Figure 14.

i. Test the VLF antenna system for proper operation as follows:

1. Refer to Chapter 24-40-00, and apply 28 vdc of external power to the airplane.

2. Ramp test the VLF antenna system by referring to the level-2 test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

j. When the sealer has cured, touch up the VLF antenna (3) and wing box shear panel (5) with paint as required.

RADIO ALTIMETER CONVERTER REMOVAL (Effectivity: All) (Figure 15)

NOTE

The radio altimeter converter is mounted in the aft wing box in the forward manhole on the right side of the airplane.

a. Remove all electrical power from the airplane.

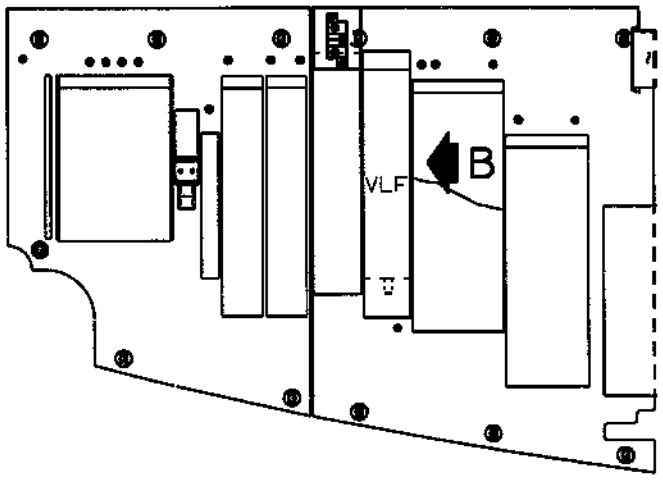
b. Refer to Chapter 53-50-00, and remove the aft wing shear panel (528AB) below the wing.

c. There are two manholes inside the wing box shear panel (528AB). The forward manhole (6) provides access to the radio altimeter converter (9).

d. Loosen the screw jacks that secure the power connector and remove it from the radio altimeter converter (9).

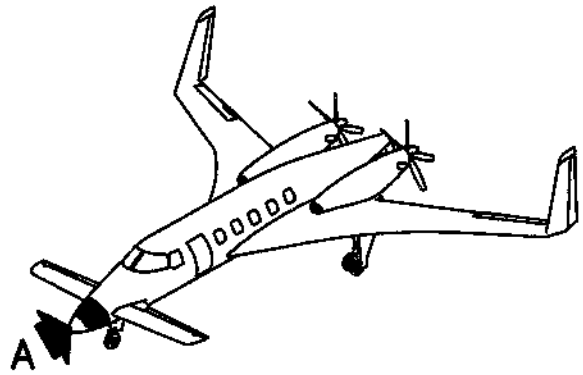
e. Loosen and remove the screw through the clip that secures the radio altimeter converter (9) to the equipment shelf (8).

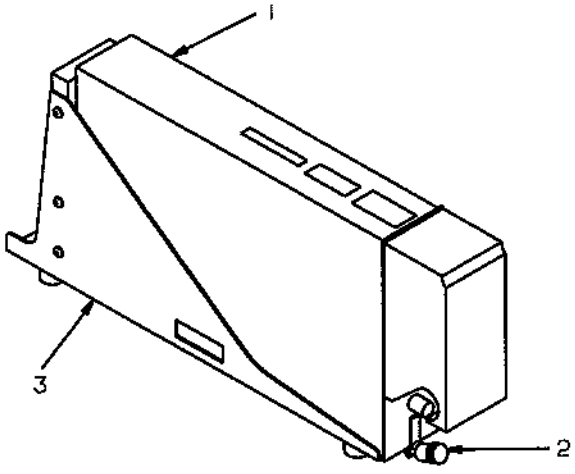
f. Lift the altimeter converter (9) away from the tab that hooks into the detent at the rear of the unit and remove the converter from the airplane.



LEFT AVIONICS COMPARTMENT

DETAIL A





1. VLF/OMEGA RECEIVER
2. LATCH RETAINER KNOB
3. RACK

DETAIL B

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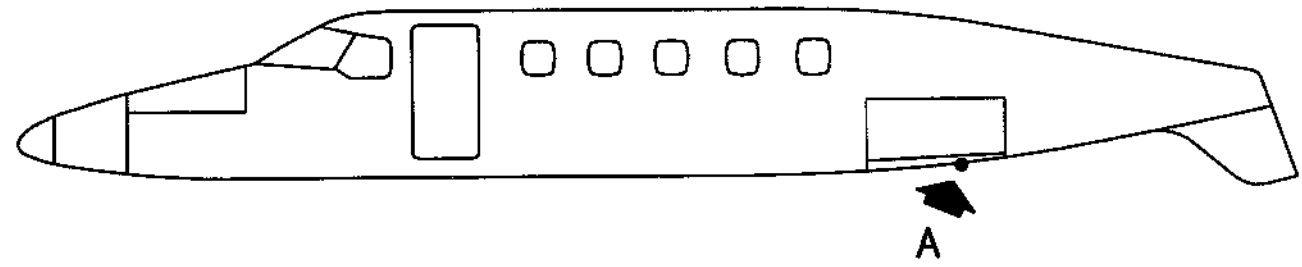
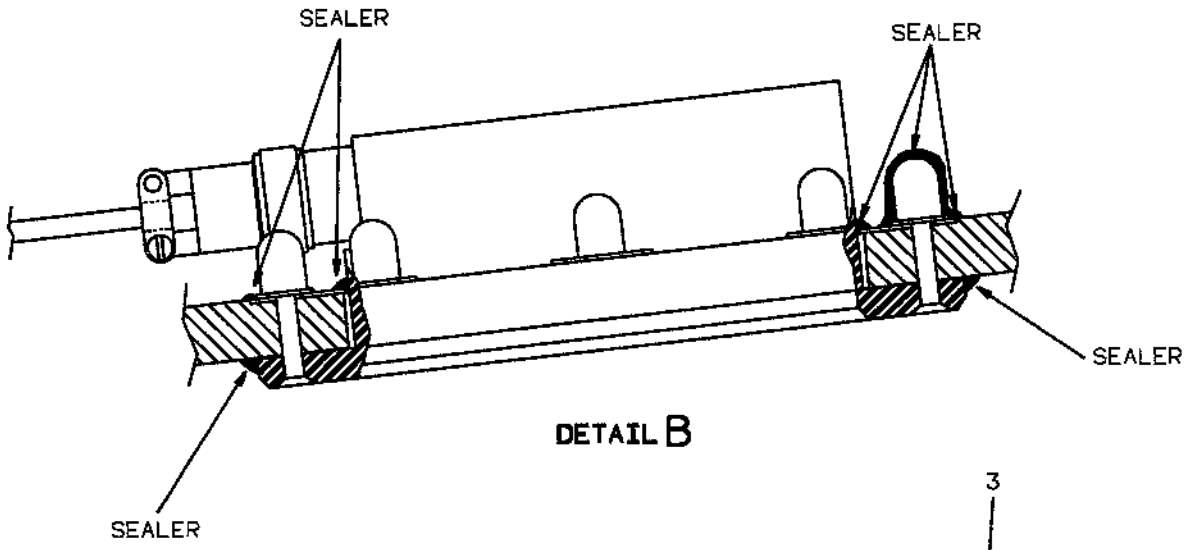
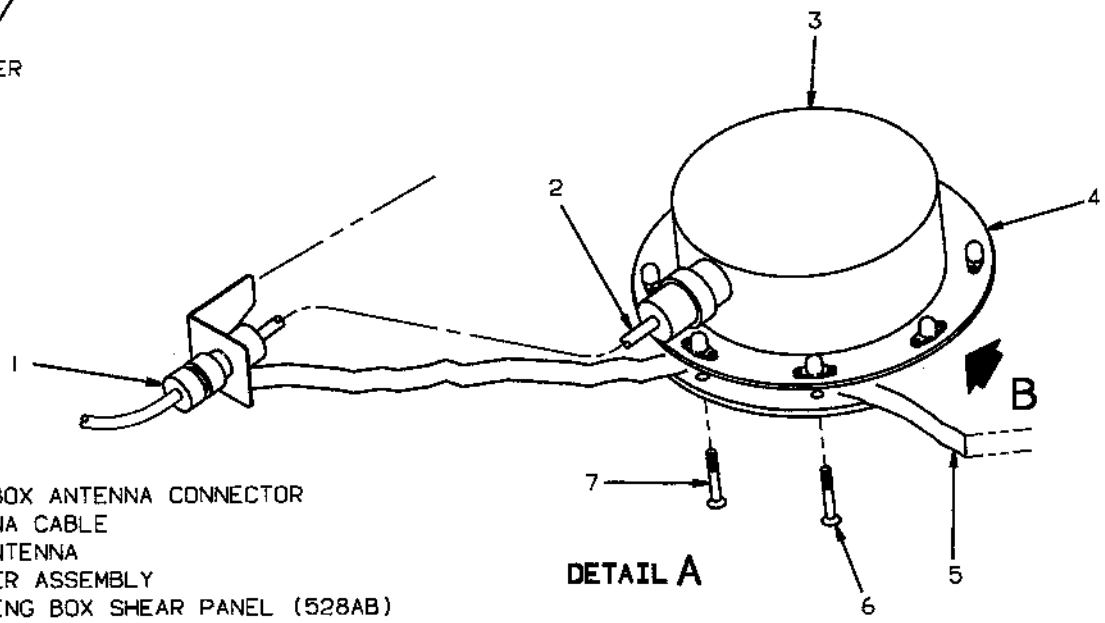
VLF Receiver Installation (Effectivity: All)
Figure 13







1. WING BOX ANTENNA CONNECTOR
2. ANTENNA CABLE
3. VLF ANTENNA
4. DOUBLER ASSEMBLY
5. AFT WING BOX SHEAR PANEL (528AB)
6. SCREW (7)
7. SCREW (1)

C9100159 C

VLF Antenna Installation (Effectivity: All)
Figure 14

RADIO ALTIMETER CONVERTER INSTALLATION (Effectivity: All) (Figure 15)

a. Position the converter (9) in the mount so that the tab at the rear of the mount hooks into the detent at the rear of the unit.

b. Install the screw and clip on the front of the converter (9) and secure to the equipment shelf (8).

c. Position the power connector on the radio altimeter converter (9) and tighten the screw jacks.

d. Refer to Chapter 53-50-00 and install the wing box shear panel on the fuselage.

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

e. Refer to the level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section and test the radio altimeter for proper operation.

RADIO ALTIMETER TRANSCEIVER REMOVAL (Effectivity: All) (Figure 15)

NOTE

The radio altimeter transceiver is mounted in the aft wing-box in the forward manhole on the right side of the airplane.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-50-00 and remove the aft wing box shear panel (528AB).

c. Inside the wing box there are two manholes for access. The forward manhole provides access to the radar altimeter transceiver (3).

d. Refer to the RADIO ALTIMETER CONVERTER REMOVAL section of this chapter and remove the radio altimeter converter.

e. Loosen the retainer knobs (5), and lift the latches away from the transceiver (3).

f. Pull the altimeter transceiver unit (3) out of the rack connector (1) and tag and remove the antenna cables (4).

g. Remove the altimeter transceiver (3) from the airplane.

RADIO ALTIMETER TRANSCEIVER INSTALLATION (Effectivity: All) (Figure 15)

a. Identify and connect the antenna cables (4) before placing the transceiver (3) in the rack (2).

b. Place the radio altimeter transceiver (3) in the rack (2) and firmly seat it into the rack connector (1).

c. Install the retainer knobs (5) on the latches, and hand-tighten the retaining knobs (5).

d. Refer to the RADIO ALTIMETER CONVERTER INSTALLATION procedure in this chapter and install the radio altimeter converter.

e. Refer to Chapter 53-50-00 and install the wing-box shear panel (528AB).

f. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

g. Refer to the level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section and test the radio altimeter for proper operation.

RADIO ALTIMETER ANTENNA REMOVAL (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 16)

NOTE

Maintenance procedures for both the radio altimeter transmitter antenna and receiver antenna are typical.

a. Remove all electrical power from the airplane.

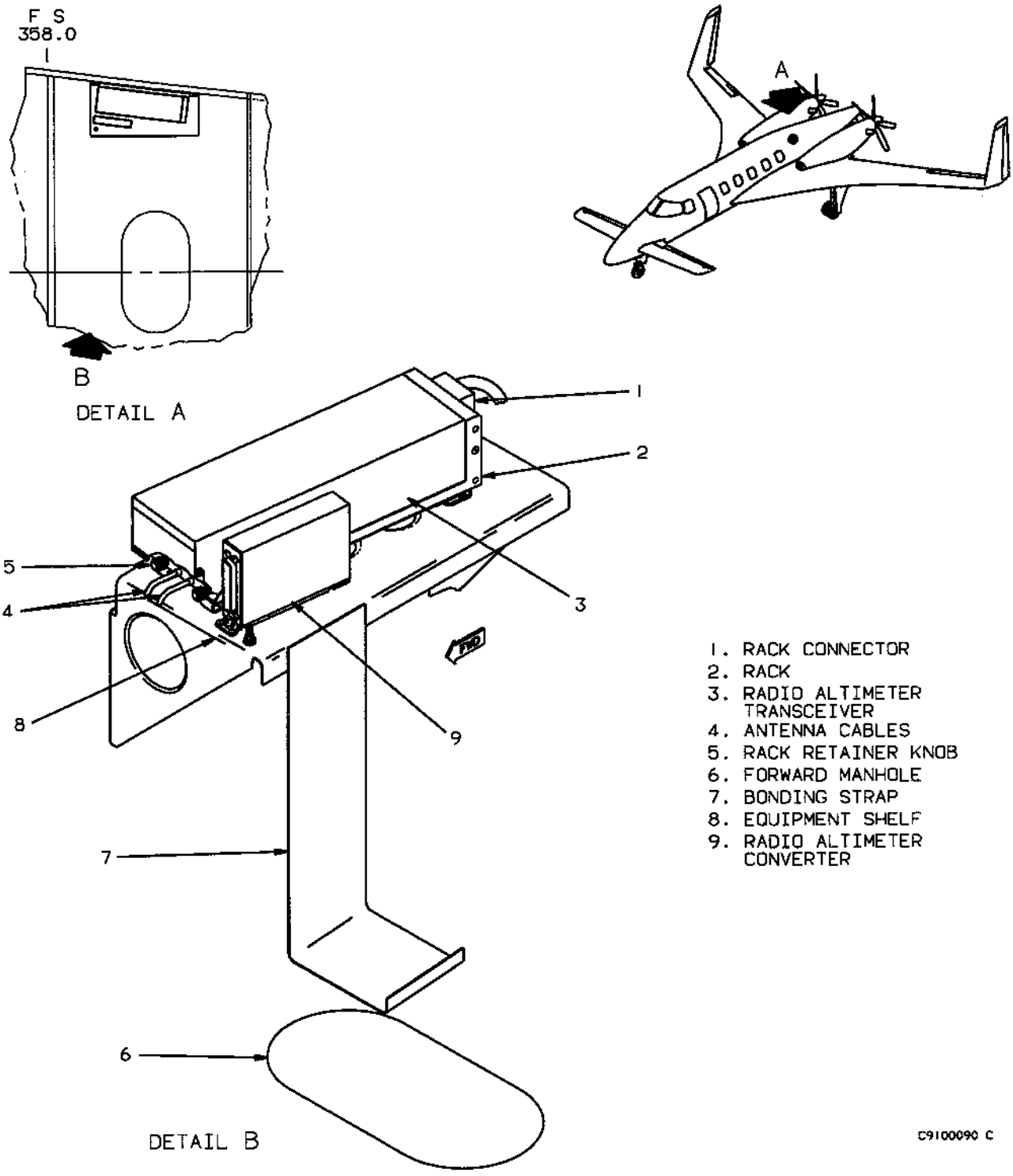
b. Locate the antenna to be removed (the transmitter antenna is located at F.S. 295.00 forward of the receiver radio altimeter antenna at F.S. 323.00).

c. Refer to Chapter 25-20-00 and remove the center aisle carpet.

d. Refer to Chapter 53-30-00 and remove the forward center aisle floorboard (163ATC) and the aft center aisle floorboard (173ATC) to gain access to the radio altimeter antenna connector, cable (2), and reinforcement assembly (1).

e. Tag and disconnect the antenna cable (2) from the radio altimeter antenna (4).

f. Remove the four screws (5) that secure the radio altimeter antenna (4) to the underside of the fuselage.



Radio Altimeter Converter and Transceiver Installation
(Effectivity: All)
Figure 15

g. Carefully strip the bead of sealer from around the antenna (4) with a nonmetallic scraper. Avoid damage to the fuselage or the antenna.

h. Remove the antenna (4) and reinforcement assembly (1) from the airplane.

RADIO ALTIMETER ANTENNA INSTALLATION (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
(Figure 16)

NOTE

Maintenance procedures for both the radio altimeter transmitter antenna and receiver antenna are typical.

a. Remove all the old sealer from the radio altimeter antenna (4), reinforcement assembly (1), and fuselage skin (3) with a nonmetallic scraper and nonmetallic cleaner pad.

b. Clean the contact surfaces of the fuselage skin (3) thoroughly where the antenna is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the antenna (4) and contact surfaces of the reinforcement plate (1) and the fuselage skin (3), then chemically treat the antenna with alodine (3, Chart 2, Chapter 34-00-00) to prevent corrosion. Refer to Chapter 20-00-03, METAL SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

d. With the help of an assistant, place the reinforcement assembly (1) on the inside of the fuselage skin (3) and position the radio altimeter antenna (4) on the outside of the fuselage skin (3). Make sure the arrow on the antenna face is pointing in the forward direction.

e. Secure the radio altimeter antenna (4) on the fuselage with the four screws (5) into the reinforcement plate assembly (1). Install the four screws wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

f. Check the bond resistance for the antenna as indicated in Chapter 20-00-03.

g. Connect the antenna cable (2) to the radio altimeter antenna (4).

h. Test the radio altimeter antenna for proper operation as follows.

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Refer to the level-2 test procedures contained in latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

i. Apply sealer (1, Chart 2, Chapter 34-00-00) around the inside of the fuselage antenna opening where the antenna touches the fuselage skin and around the antenna reinforcement assembly as shown in Detail B of Figure 16. Apply sealer on each nut plate of the reinforcement assembly to completely cover the nut plate.

j. Refer to Chapter 53-20-00 and install the forward (163ATC) and aft (173ATC) floorboards.

k. Refer to Chapter 25-20-00 and install the center aisle carpet and seats.

l. On the outside of the airplane, and after the bonding resistance checks, apply sealer (1, Chart 2, Chapter 34-00-00) around the periphery of the altimeter antenna base. Refer to Detail B, figure 16.

m. When the sealer has cured, touch up the antenna and fuselage with paint as necessary.

RADIO ALTIMETER ANTENNA REMOVAL (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
(Figure 17)

NOTE

Maintenance procedures for both the radio altimeter transmitter antenna and receiver antenna are typical.

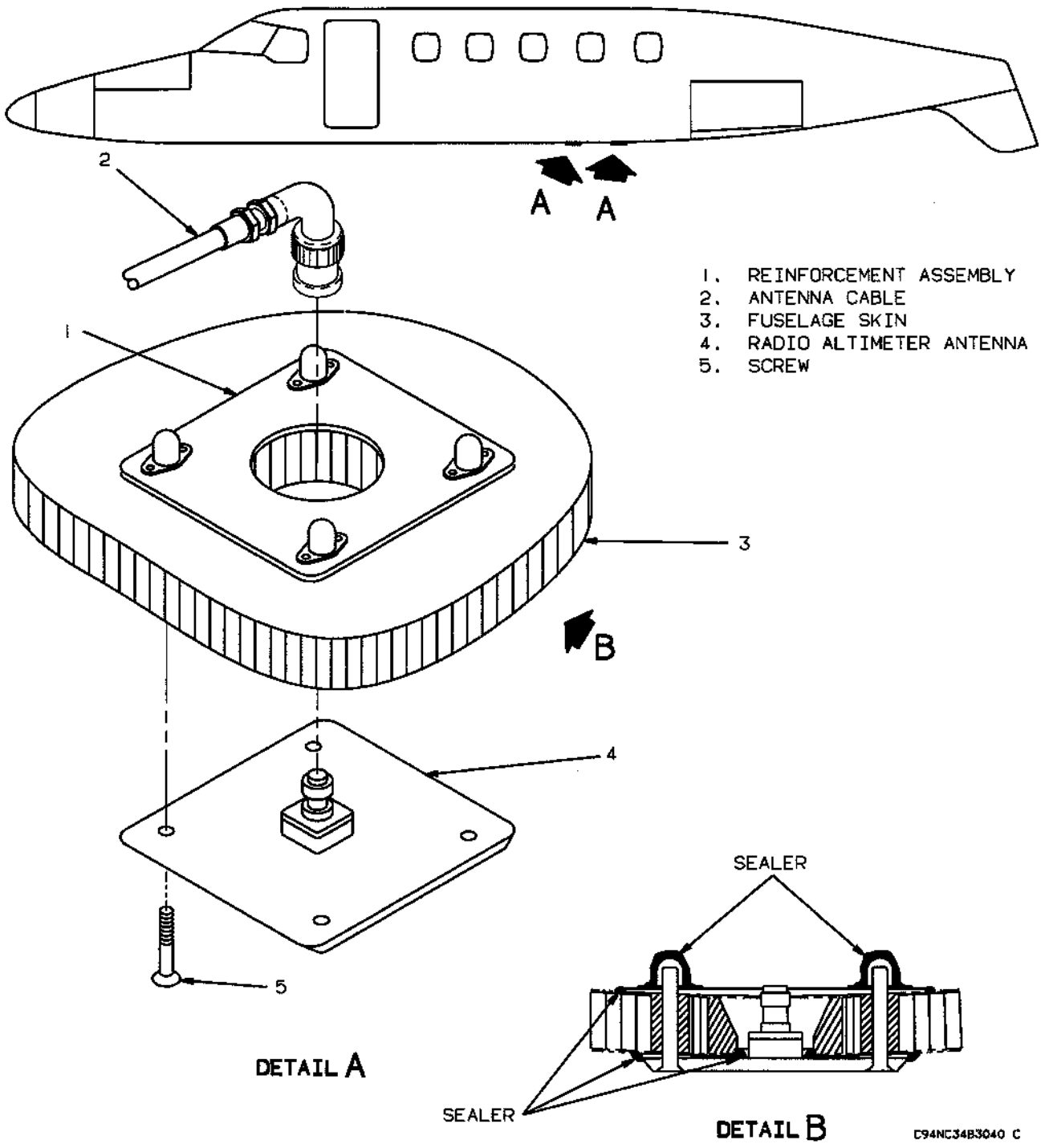
a. Remove all electrical power from the airplane.

b. Locate the antenna to be removed (the transmitter antenna is located at F.S. 295.00 forward of the receiver radio altimeter antenna at F.S. 323.00).

c. Refer to Chapter 25-20-00 and remove the center aisle carpet.

d. Refer to Chapter 53-30-00 and remove the forward center aisle floorboard (163ATC) and the aft center aisle floorboard (173ATC) to gain access to the radio altimeter antenna connector (8), coaxial cable (1) and reinforcement plate assembly (3).

e. Cut and remove the tie-wrap (2), then remove sealant from the boot opening.



Radio Altimeter Antenna Installation (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 Installed)
Figure 16

CAUTION

Use care to avoid damaging the coaxial cable (1), connector (8) and surrounding surface of the reinforcement plate assembly (3).

f. Slice open the antenna boot (4) as necessary to gain access to the antenna connector (8).

g. Tag and disconnect the coaxial cable (1) from the radio altimeter antenna (6).

h. Remove the four screws (7) that secure the radio altimeter antenna (6) to the underside of the fuselage.

i. Carefully strip the bead of sealer from around the antenna (6) with a nonmetallic scraper. Avoid damage to the fuselage skin (5) or the antenna.

j. Remove the antenna (6) and reinforcement plate assembly (3) from the airplane.

NOTE

If the antenna boot is not damaged it may be reused. However, a cut or torn boot must be replaced, as it is not repairable.

RADIO ALTIMETER ANTENNA INSTALLATION (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
(Figure 17)

NOTE

Maintenance procedures for both the radio altimeter transmitter antenna and receiver antenna are typical.

a. Remove all the old sealer from the radio altimeter antenna (6), reinforcement plate assembly (3), and fuselage skin (5) with a nonmetallic scraper and nonmetallic cleaner pad.

b. Thoroughly clean the contact surfaces of the fuselage skin (5) where the antenna (6) is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the antenna (6) and contact surfaces of the reinforcement plate assembly (3) and the fuselage skin (5), then chemically treat the antenna with alodine (3, Chart 2, Chapter 34-00-00) to prevent corrosion. Refer to Chapter 20-00-03, METAL SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

d. With the help of an assistant, position the reinforcement plate assembly (3) on the inside of the fuselage skin (5) and align the radio altimeter antenna (6) with the outside of the fuselage skin (5). Ensure that the arrow on the antenna face is pointing forward.

e. Secure the radio altimeter antenna (6) to the fuselage and reinforcement plate assembly (3) by installing the four screws (7). Install the four screws wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

f. Check the bond resistance for the antenna as indicated in Chapter 20-00-03.

g. Connect the coaxial cable (1) to the radio altimeter antenna (6) to perform an operational check of the antenna. Test the radio altimeter antenna for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Refer to the level-2 test procedures contained in latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section and verify proper operation of the antenna.

h. Remove electrical power from the airplane.

i. Disconnect the coaxial cable (1) from the radio altimeter antenna (6).

j. From inside the cabin apply sealant (1, Chart 2, 34-00-00) around the inside of the fuselage antenna opening where the radio altimeter antanna (6) and fuselage are joined together, as shown by Detail B of Figure 17.

NOTE

If a new antenna boot (4) is to be installed, carefully trim the new boot to match the reinforcement plate assembly (3) and nut plates before performing the remaining installation procedures.

k. Insert the coaxial cable (1) into the antenna boot (4) opening and carefully push the connector (8) out the bottom end.

l. Connect the coaxial cable (1) to the radio altimeter antenna (6)

m. Bond the antenna boot (4) to the reinforcement plate assembly (3) with sealant (4, Chart 2, 34-00-00).

n. Apply sealant (4, Chart 2, 34-00-00) to the periphery of the reinforcement plate assembly (3) and

antenna boot (4) to provide a good moisture barrier between the antenna mounting assembly and fuselage skin (5).

o. Completely cover the nut plates on the reinforcement plate assembly (3) with sealant (4, Chart 2, 34-00-00).

p. Apply sealant (4, Chart 2, 34-00-00) approximately 1/2 inch into the opening where the coaxial cable (1) exits the antenna boot (4).

q. Install a tie-wrap (2) around the the boot neck and coaxial cable (1).

r. Refer to Chapter 53-20-00 and install the forward (163ATC) and aft (173ATC) floorboards.

s. Refer to Chapter 25-20-00 and install the center aisle carpet and seats.

t. On the outside of the airplane, after all checks are complete, apply sealer (1, Chart 2, Chapter 34-00-00) around the periphery of the altimeter antenna base as shown by Detail B of Figure 17.

u. When the sealer has cured, touch up the antenna and fuselage with paint as necessary.

v. Restore electrical power to the airplane.

TRANSPONDER TRANSCEIVER REMOVAL (Effectivity: All) (Figure 18)

NOTE

The No. 1 transponder is mounted in the left avionics compartment, and the No. 2 transponder is mounted in the right avionics compartment.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the right (214AR) or left (213AL) nose avionics compartment panel.

c. Tag and remove the coaxial connector.

d. Loosen the retainer knob(s) (2) and remove the retainer knob(s) (2) from the latch(es).

e. Pull the transceiver (1) out from the connector. (Do not lift the unit until it is free of the connectors).

f. Lift the transceiver (1) from the rack (3) and remove it from the airplane.

TRANSPONDER TRANSCEIVER INSTALLATION (Effectivity: All) (Figure 18)

a. Place the transceiver (1) in the rack (3) and firmly seat it in the connector.

b. Install the retainer knob(s) (2) on the latches and hand-tighten the retainer knob(s) (2).

c. Connect the coaxial connector to the transceiver (1).

d. Test the transponder system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the transponder system for proper operation using the ATC-601 (3, Chart 1, 34-00-00) or an equivalent ramp test set. Perform the level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

e. Refer to Chapter 53-30-00 and install the left (213AL) and right (214AR) nose avionics compartment panel.

TRANSPONDER CONTROL ADAPTER REMOVAL (Effectivity: NC-4 thru NC-25) (Figure 19)

NOTE

The No. 1 Control Adapter is mounted in the left avionics compartment, and the No. 2 Control Adapter is mounted in the right avionics compartment.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the right (214AR) or left (213AL) nose avionics compartment door.

c. Remove the electrical connector (6) and tag for identification.

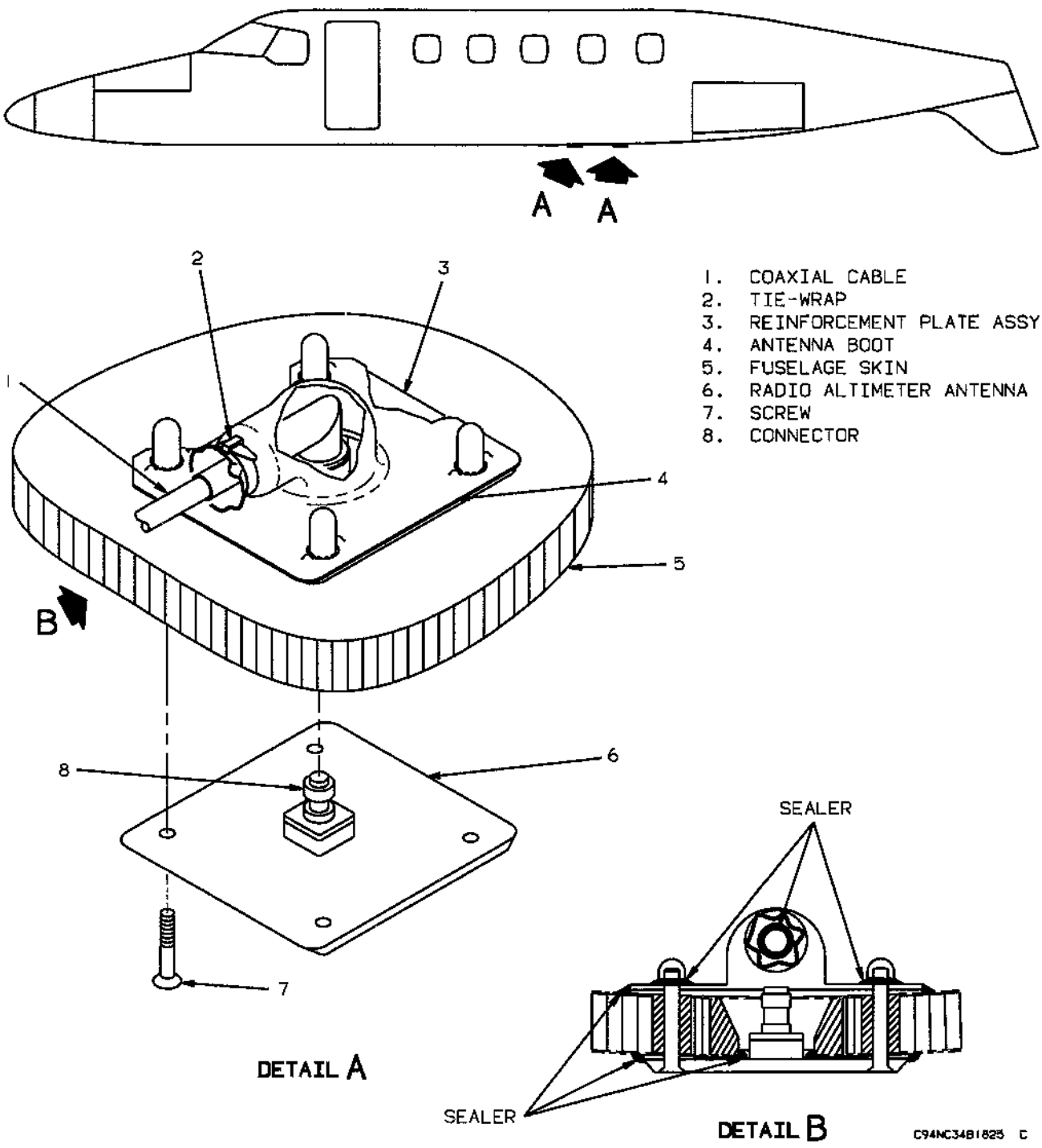
d. Loosen the screw (4) until the clamp (5) is released from the transponder control adapter (1).

e. Lift and remove the transponder control adapter (1) from the mount (3).

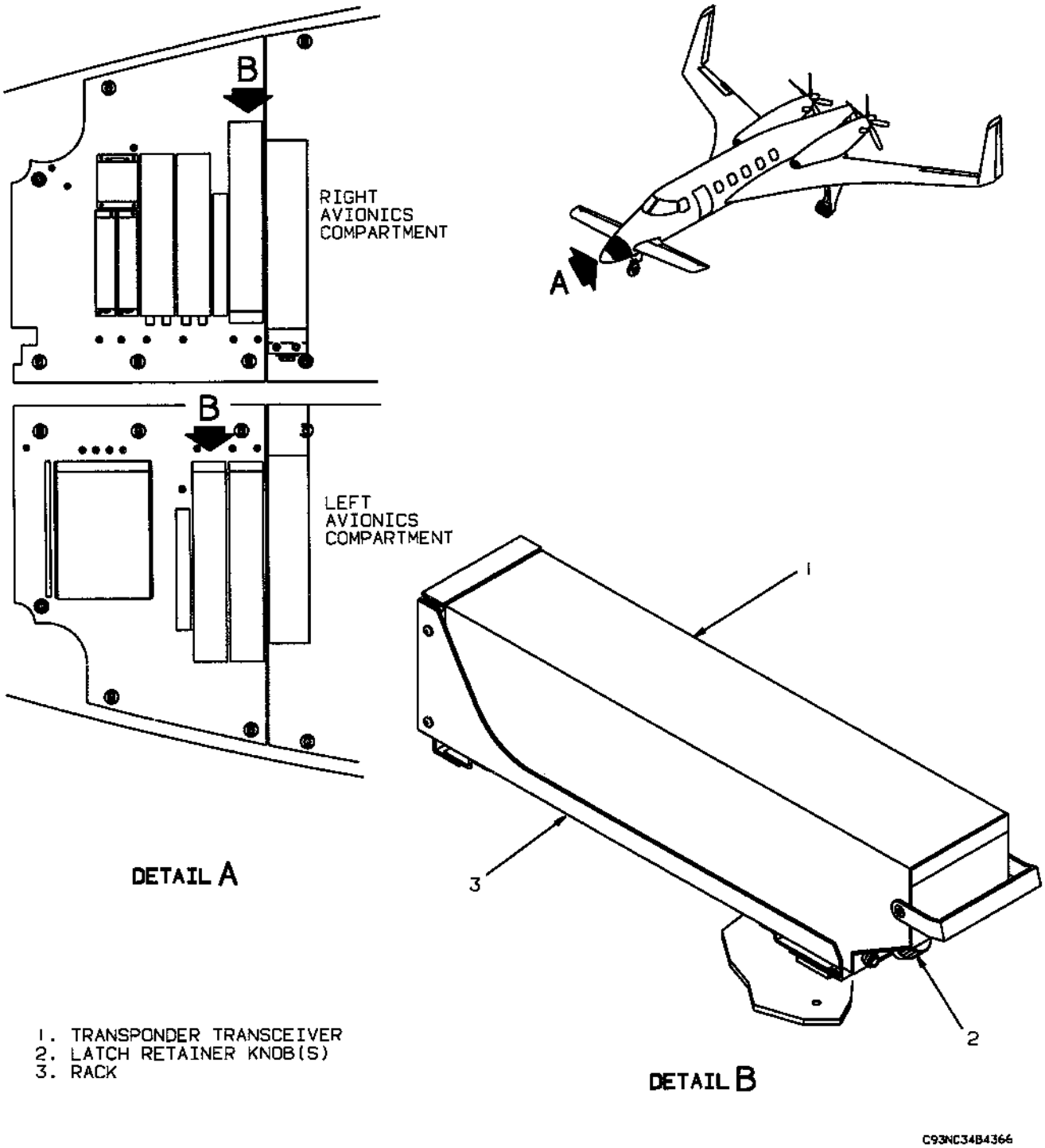
TRANSPONDER CONTROL ADAPTER INSTALLATION (Effectivity: NC-4 thru NC-25) (Figure 19)

a. Align the slot (2) in the transponder control adapter (1) with the mount (3) and position the transponder control adapter (1) on the mount (3).

b. Tighten the screw (4) until the clamp (5) secures the transponder to the mount (3).



Radio Altimeter Antenna Installation (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 17



1. TRANSPONDER TRANSCEIVER
2. LATCH RETAINER KNOB(S)
3. RACK

Transponder Transceiver Installation (Effectivity: All)
Figure 18

c. Connect the electrical connector (6) to the transponder control adapter (1).

d. Test the transponder system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the transponder antenna system for proper operation using the ATC-601 (3, Chart 1, 34-00-00), or an equivalent ramp test set. Perform the level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

e. Refer to Chapter 53-30-00 and install the left (213AL) or right (214AR) nose avionics compartment panel.

TRANSPONDER ANTENNA REMOVAL (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 20)

NOTE

Maintenance procedures for both the forward and aft transponder antennas are typical.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-20-00 and remove the center aisle carpet.

c. Refer to Chapter 53-20-00 and remove the center aisle floorboard (163ATC) to gain access to the transponder antenna cable and antenna mounting area.

d. Disconnect the antenna cable (2) from the antenna connector (5).

e. Remove the six screws (6) that secure the transponder antenna (4) to the fuselage skin (3).

f. Carefully strip the sealer from around the transponder antenna (4) and fuselage contact area with a nonmetallic scraper. Avoid damage to the fuselage or the antenna.

g. Remove the transponder antenna (4) and reinforcement plate assembly (1) from the airplane.

TRANSPONDER ANTENNA INSTALLATION (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 20)

a. Remove all the old sealer from the fuselage skin (3), reinforcement plate assembly (1), and transponder antenna (4) using a nonmetallic scraper and a nonmetallic cleaner pad.

b. Thoroughly clean the contact surfaces of the fuselage skin (3) where the antenna is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

c. Thoroughly clean the antenna (4) and the contacting surfaces of the reinforcement plate (1) and fuselage skin (3) then chemically treat the antenna with alodine (3, Chart 2, Chapter 34-00-00) to prevent corrosion. Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

d. Have an assistant hold the reinforcement assembly (1) on the inside of the fuselage skin (3) where the transponder antenna (4) is to be attached.

e. Position the transponder antenna (4) on the outside of the fuselage skin (3) and secure with the six screws (6). Install the screws wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

f. Check bond resistance for the antenna (4) as indicated in Chapter 20-00-03.

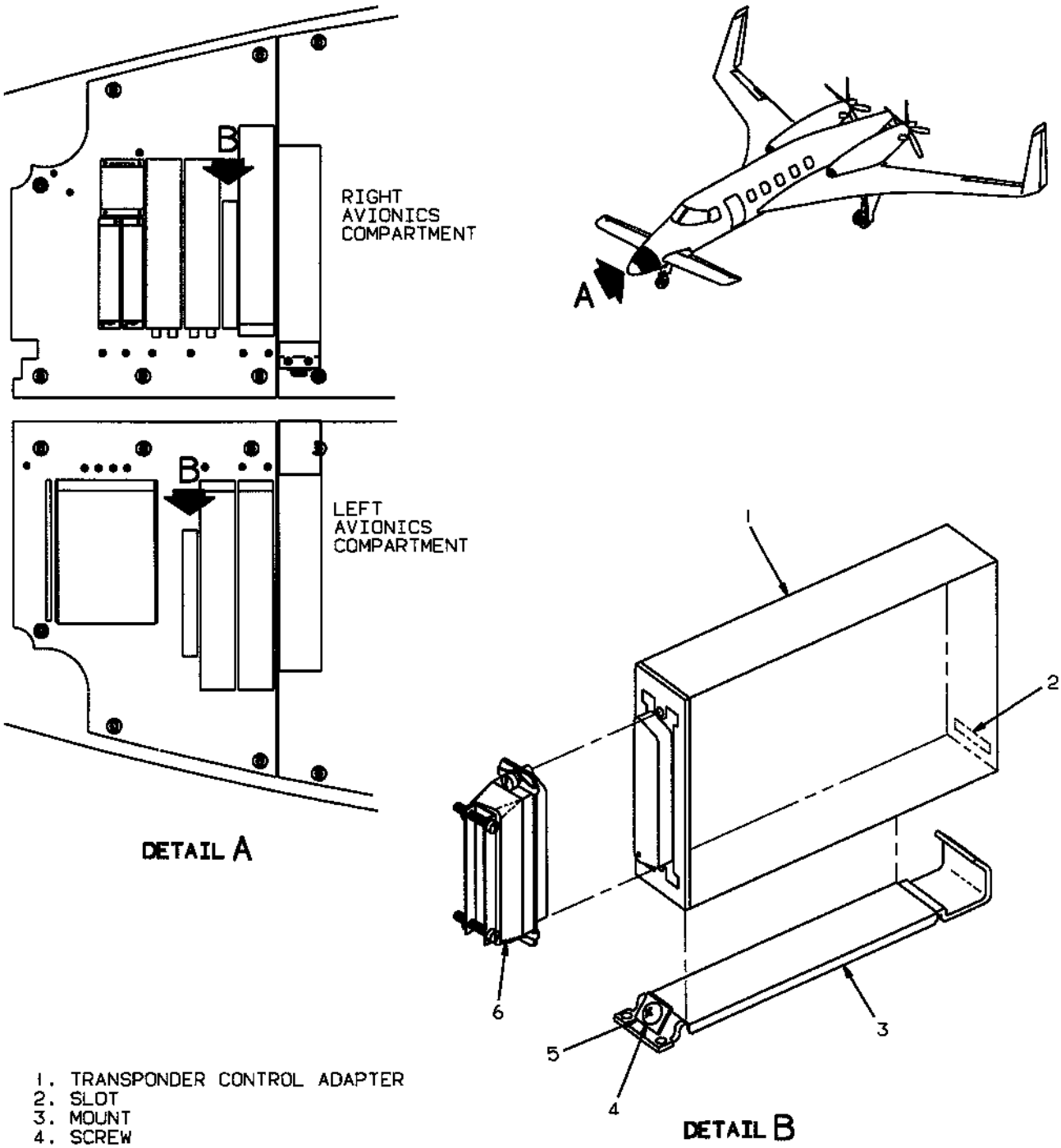
g. From the inside of the cabin, apply a bead of sealer (1, Chart 2, Chapter 34-00-00) around the area where the transponder antenna (4) contacts the fuselage and around the periphery of the reinforcement plate assembly (1). Refer to Detail B, Figure 20. Apply sealer to each of the nut plates on the reinforcement plate assembly to completely cover the nut plates.

h. Connect the antenna cable (2) to the transponder antenna connector (5).

i. Test the transponder antenna system for proper operation as follows:

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the transponder antenna (4) for proper operation using the ATC-601 (3, Chart 1, 34-00-00) , or an equivalent ramp test set. Perform the



Transponder Control Adapter Installation
(Effectivity: NC-4 thru NC-25)
Figure 19

level-2 test procedures contained in the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

j. Refer to Chapter 53-20-00 and install the center aisle floorboard (163ATC).

k. Refer to Chapter 25-20-00 and install the center aisle carpet.

l. On the underside of the fuselage apply a bead of sealer (1, Chart 2, Chapter 34-00-00) around the periphery of the antenna where the antenna contacts the fuselage. Refer to Detail B, Figure 20.

m. When the sealer has cured, touch up the fuselage and antenna with paint as necessary.

TRANSPONDER ANTENNA REMOVAL (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 21)

NOTE

Maintenance procedures for both the forward and aft transponder antennas are typical.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-20-00 and remove the center aisle carpet.

c. Refer to Chapter 53-20-00 and remove the center aisle floorboard (163ATC) to gain access to the transponder antenna cable and antenna mounting area.

d. Cut and remove the tie-wrap (1), then remove the sealant from the boot opening.

CAUTION

Use care to avoid damaging the coaxial cable (8), antenna connector (5) and surrounding surface of the reinforcement plate assembly (2).

e. Slice open the antenna boot (7) as necessary to gain access to the antenna connector (5).

f. Disconnect the coaxial cable (8) from the antenna connector (5).

g. Remove the six screws (6) that secure the transponder antenna (4) to the fuselage skin (3).

h. Carefully strip the sealer from around the transponder antenna (4) and fuselage contact area with a nonmetallic scraper. Avoid damage to the fuselage or the antenna.

i. Remove the transponder antenna (4) and reinforcement plate assembly (2) from the airplane.

NOTE

If the antenna boot is not damaged it may be reused. However, a cut or torn boot must be replaced, as it is not repairable.

TRANSPONDER ANTENNA INSTALLATION (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed) (Figure 21)

a. Remove all the old sealer from the fuselage skin (3), reinforcement plate assembly (2), and transponder antenna (4) using a nonmetallic scraper and a nonmetallic cleaner pad.

b. Thoroughly clean the contact surfaces of the fuselage skin (3) where the antenna is to be attached. Refer to Chapter 20-00-03, COMPOSITE SURFACE PREPARATION FOR ELECTRICAL BONDING, for cleaning and bonding procedures.

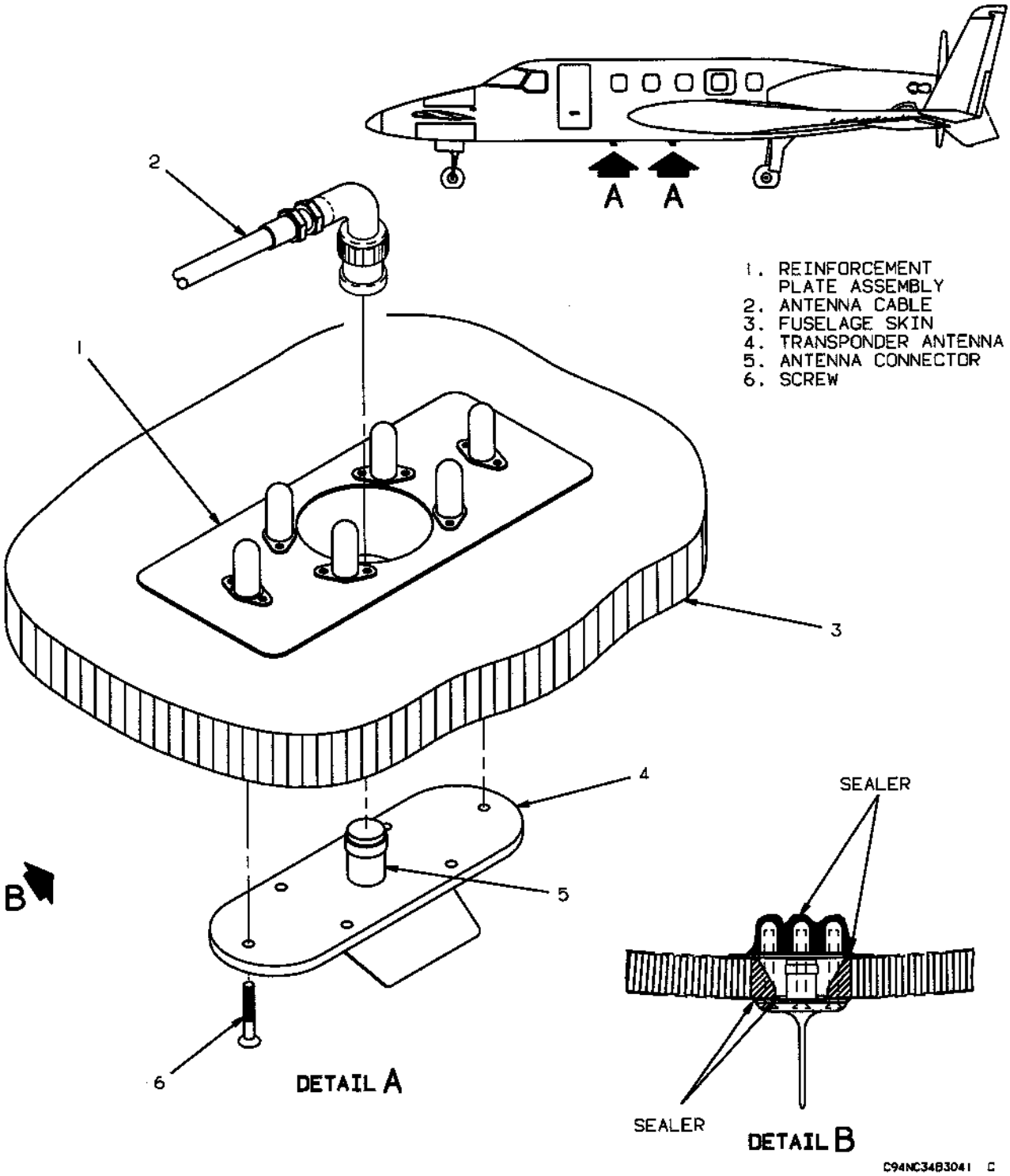
c. Thoroughly clean the transponder antenna (4) and the contacting surfaces of the reinforcement plate assembly (2) and fuselage skin (3), then chemically treat the antenna with alodine (3, Chart 2, Chapter 34-00-00) to prevent corrosion. Refer to Chapter 20-00-03, METAL SURFACE PREPARATION AND ELECTRICAL BONDING, for cleaning and bonding procedures.

d. Have an assistant hold the reinforcement plate assembly (2) on the inside of the fuselage skin (3) where the transponder antenna (4) is to be attached.

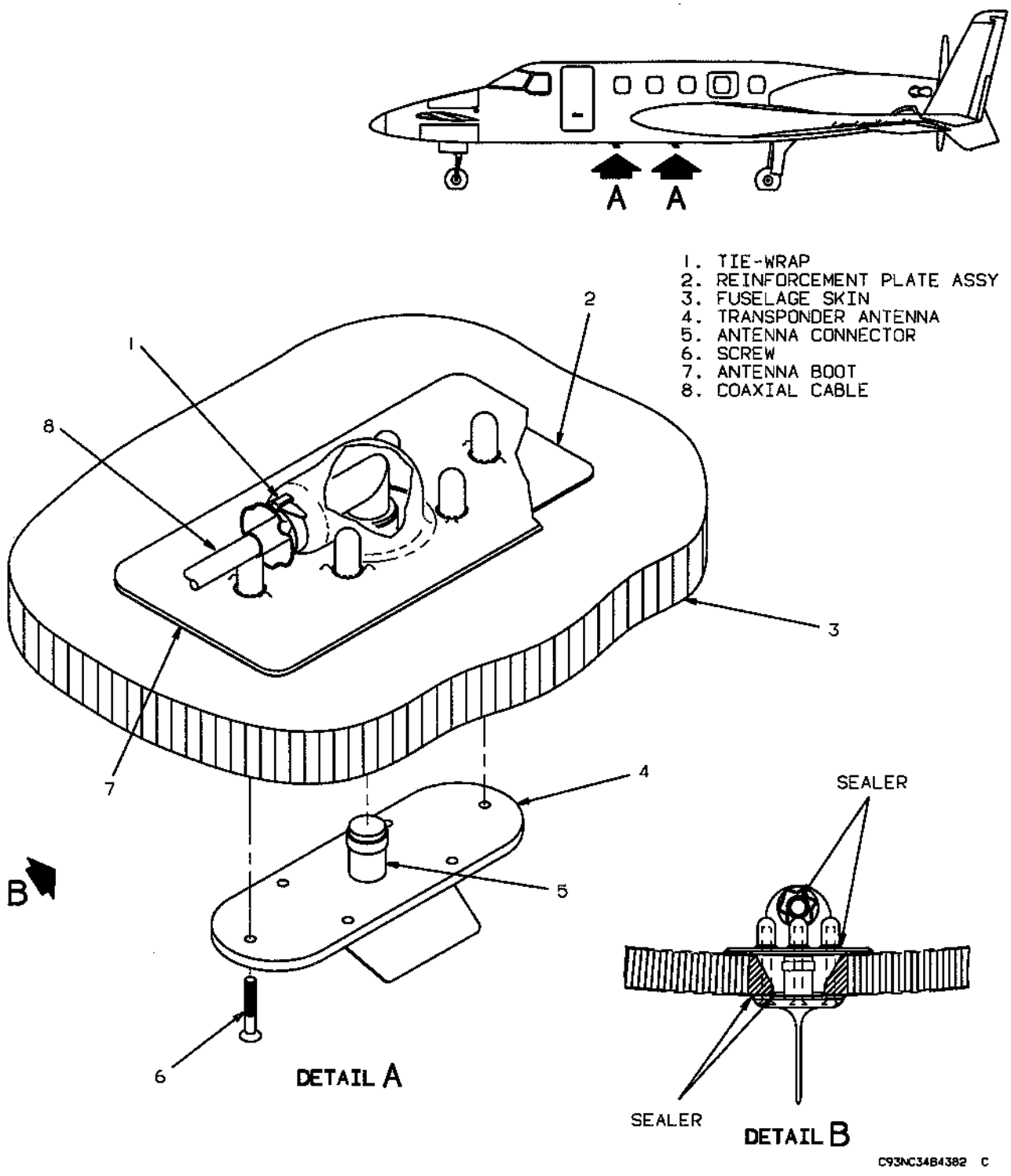
e. Position the transponder antenna (4) on the outside of the fuselage skin (3) and secure with the six screws (6). Install the screws wet with polysulfide sealer (2, Chart 2, Chapter 34-00-00).

f. Check bond resistance for the antenna (4) as indicated in Chapter 20-00-03.

g. Connect the coaxial cable (8) to the transponder antenna (4) to perform a ramp test of the antenna. Test the transponder antenna for proper operation as follows:



Transponder Antenna Installation (Effectivity: NC-4 and after without BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 20



Transponder Antenna Installation (Effectivity: NC-4 and after with BEECHCRAFT Kit No. 122-3014-1 or -3 installed)
Figure 21

1. Refer to Chapter 24-40-00 and apply 28 vdc of external power to the airplane.

2. Ramp test the transponder antenna (4) for proper operation using the ATC-601 (3, Chart 1, 34-00-00) , or an equivalent ramp test set. Perform the level-2 test procedures contained in the latest revision of the Collins Starship 1 Avionics System Manual, 523-0774076-00111A, Maintenance Section.

h. Remove all electrical power from the airplane.

i. Disconnect the coaxial cable (8) from the transponder antenna (4).

j. From the inside of the cabin, apply sealant (1, Chart 2, 34-00-00) around the inside of the fuselage antenna opening where the transponder antenna (4) and fuselage are joined together as shown by Detail B of Figure 21.

NOTE

If a new antenna boot (7) is to be installed, carefully trim the new boot to match the reinforcement plate assembly (2) and nut plates before performing the remaining installation procedures.

k. Insert the coaxial cable (8) into the antenna boot (7) opening and carefully push the antenna connector (5) out the bottom end of the boot.

l. Connect the coaxial cable (8) to the transponder antenna (4).

m. Bond the antenna boot (7) to the reinforcement plate assembly (2) with sealant (4, Chart 2, 34-00-00).

n. Apply sealant (4, Chart 2, 34-00-00) to the periphery of the reinforcement plate assembly (2) and antenna boot (7) to provide a good moisture barrier between the antenna mounting assembly and fuselage skin (3).

o. Completely cover the nut plates on the reinforcement plate assembly (2) with sealant (4, Chart 2, 34-00-00).

p. Apply sealant (4, Chart, 2, 34-00-00) approximately 1/2 inch into the opening where the coaxial cable (8) exits the antenna boot (7).

q. Install a tie-wrap (1) around the boot neck and coaxial cable (8).

r. Refer to Chapter 53-20-00 and install the center aisle floorboard (163ATC).

s. Refer to Chapter 25-20-00 and install the center aisle carpet.

t. On the underside of the fuselage apply a bead of sealer (1, Chart 2, Chapter 34-00-00) around the periphery of the antenna where the antenna contacts the fuselage, as shown by Detail B of Figure 21.

u. When the sealer has cured, touch up the fuselage and antenna with paint as necessary.

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GENERAL - DESCRIPTION AND OPERATION (FIGURES 1 AND 2)

WARNING: Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The oxygen system provides breathable oxygen to the flight crew and passengers in the event of a pressurization system failure and when flight altitudes require its use.

The standard system uses a 77 cubic foot supply cylinder (the optional system uses a 115 cubic foot cylinder) located in the RH lower nose compartment and filled with aviator's breathing oxygen (4, Chart 4, 35-00-00) to a pressure of 1850 ± 50 psig at 70° F. The supply cylinder's regulator is used to start and stop the flow of oxygen from the cylinder, provide high pressure oxygen for referencing and to provide low pressure oxygen (70 ± 10 psig) to the two flight crew masks and five cabin oxygen compartments.

The regulator's high pressure ports are individually opened and closed by a valve in each port. These ports are open, allowing high pressure oxygen to flow from them, anytime an oxygen line is connected to them. The regulator's control lever does not have any control over the high pressure ports.

When the low pressure port on the regulator is opened by pulling ON the oxygen system ready push-pull control knob, placarded OXYGEN SYS READY and located on the pilot's inboard subpanel, oxygen flows through the system plumbing to the flight crew masks and the cabin shutoff valve.

The flight crew masks are a quick donning type that are armed and ready to provide oxygen to the user anytime the OXYGEN SYS READY control knob is pulled ON. Four different modes of operation are available on the masks (N, 100%, EMERGENCY and PRESS TO TEST) allowing the user to choose how the masks will deliver the oxygen (normal inhalation or under pressure) and what the mixture will be (oxygen diluted with cabin air or 100 percent oxygen). When the N position is selected, the mask's mixture control provides oxygen diluted with cabin air for cabin altitudes up to 30,000 feet and 100 percent oxygen for cabin altitudes of 30,000 feet or greater. The method of delivery, when in the N position, is by normal inhalation for cabin altitudes up to 35,000 feet and under pressure for cabin altitudes of 35,000 feet or greater. Placing the mask's control in the 100% position changes the mask's mixture control to 100 percent oxygen at all cabin altitudes while maintaining the same method of delivery as in the N position. When the mask's control is in the EMERGENCY position, 100 percent oxygen is delivered under pressure at all cabin altitudes. The PRESS TO TEST position is used as a way to quickly determine whether there is a flow of oxygen from the mask. These masks also contain microphones which allow continued communication while the masks are being worn. To turn ON each mask's microphone a MIC SELECT control knob, located on both the pilot's and copilot's outboard subpanels, must be turned to the OXY MASK position. When the masks are not in use they are stored in special compartment located in the pilot's and copilot's consoles just aft of the instrument panel.

The flow of oxygen to the five cabin oxygen mask compartments is controlled by the solenoid operated cabin shutoff valve under the RH cockpit floor on the aft side of the forward pressure bulkhead. During normal operating conditions, the cabin shutoff valve solenoid is energized by electrical power from the cabin barometric pressure switch (located under the RH cockpit floor in the outboard side of the RH keel near the forward pressure bulkhead). Power to the cabin barometric switch is supplied through the oxygen control circuit breaker in the pilot's console, placarded OXY CONTROL. When increasing cabin altitude (decreasing pressure) reaches 12,500 +0, -500 feet, the cabin barometric pressure switch closes, completing the electrical circuit to the cabin shutoff valve.

When the cabin shutoff valve is opened, oxygen flows through the cabin plumbing to the cabin oxygen mask compartments located in the forward baggage cabinet (one single mask unit) and in the cabin headliner (four dual mask units). The oxygen, upon reaching the mask compartments, actuates each compartment's plunger valve (oxygen pressure of between 30 to 55 psig is required to actuate the plunger valve), opening the door and allowing the masks to drop out of the compartment. When the oxygen masks are deployed, they are suspended by lanyards. Pulling on each mask and lanyard removes the lanyard pin from the valve in each mask compartment, opening the valve and allowing oxygen to flow to the mask. Reinserting the lanyard pin in the compartment valve closes the valve stopping the flow of oxygen to its mask. The cabin masks are a constant flow type having a flow rate of 4.8 ± 0.1 lpm at 62.7 psia, a reservoir bag for economizing oxygen usage and a soft cleanable facepiece.

Oxygen to the cabin oxygen mask compartments can be shut off and the remaining oxygen isolated to the cockpit by pulling the OXY CONTROL circuit breaker and pushing in the OXYGEN PASS MAN DEPLOY control knob.

The supply cylinder is filled at a fill valve mounted in the oxygen service compartment, which is located in the RH lower nose compartment skin panel. A dial pressure gage is located adjacent to the fill valve in the oxygen service compartment and allows the supply cylinder's pressure to be checked during filling. The grounding jack for the oxygen system is located in the oxygen service compartment and must be used anytime the system is being filled.

A pressure transducer, located in the RH lower nose compartment behind the service compartment, translates the supply cylinder's pressure into an electric signal which is sent to the digital pressure indicator. The digital pressure indicator, located on the copilot's inboard subpanel, interprets this signal and provides a digital display of the supply cylinder pressure for cockpit referencing.

The plumbing for the oxygen system is separated into two distinct sections based on oxygen pressure. The high pressure side of the oxygen system plumbing (1850 ± 50 psig at 70° F.) is contained in the RH lower nose compartment with the supply cylinder and is composed of rigid, metal tubing. The low pressure side of the oxygen system plumbing (70 ± 10 psig) is routed from the supply cylinder's regulator, located in the RH lower nose compartment, up through the lower and upper RH shelves, just forward of the forward pressure bulkhead, to the RH upper nose compartment. In the RH upper nose compartment the plumbing divides, with one section proceeding to the LH upper nose compartment where it penetrates the forward pressure bulkhead and proceeds to the LH flight crew oxygen compartment and mask. The other section penetrates the RH upper forward pressure bulkhead and divides, with one section being routed to the RH flight crew oxygen compartment and mask and the other section being routed along the aft side of the forward pressure bulkhead to the inlet port of the cabin shutoff valve. From the outlet port of the cabin shutoff valve, the plumbing is routed aft through the copilot's console to the forward baggage cabinet. It then is routed behind the forward baggage cabinet to the oxygen compartment mounted in the cabinet. From the forward baggage cabinet oxygen mask compartment, plumbing is routed to the cabin headliner where it divides and proceeds along both sides of the headliner, connecting the four oxygen mask compartments mounted there (two on each side). The plumbing for the low pressure side is composed of flexible hoses except for that found in the nose section and connected to the cockpit's oxygen mask compartment which is rigid, metal tubing.

The Engine Instrument, Crew Alerting System (EICAS) display, located on the instrument panel, provides three annunciation messages to the flight crew concerning the oxygen system. The three messages are a yellow OXYGEN PRES LO caution message, a yellow OXYGEN NOT ARMED caution message and a white PASS OXYGEN ON status message.

For the OXYGEN PRESS LO message to be displayed, the following conditions must occur: (1) Airplane electrical power must be ON, (2) The EICAS display must be ON, (3) The supply cylinder low pressure switch, installed in the high pressure side of the system and located in the RH lower nose compartment behind the oxygen service compartment, must be closed with electrical continuity through the switch (switch closes with decreasing oxygen pressure at 500 psig and opens with increasing oxygen pressure at 550 ± 30 psig.) and (4) The system armed pressure switch, installed in the low pressure side of the system and located in the RH forward wing compartment just forward of the forward pressure bulkhead, must be closed with electrical continuity through the switch (switch

closes with increasing oxygen pressure at 50 ± 3 psig and opens with decreasing oxygen pressure at 40 psig). Electrical power to the supply cylinder low pressure switch and the system armed pressure switch is supplied by the No. 2 Annunciator circuit breaker, located in the cockpit's avionics circuit breaker panel.

For the OXYGEN NOT ARMED message to be displayed these conditions must occur: (1) Airplane electrical power must be ON, (2) The EICAS display must be ON and (3) The system armed pressure switch must be open with no electrical continuity through the switch.

In order for the PASS OXYGEN ON message to appear the following conditions must occur: (1) Airplane electrical power must be ON, (2) The EICAS display must be ON and (3) The cabin oxygen pressure sense switch, installed in the low pressure side of the system on the aft RH side of the cabin headliner, must close with electrical continuity through the switch (switch closes with increasing oxygen pressure at 50 ± 3 psig and opens with decreasing pressure at 40 psig). Electrical power for the cabin oxygen pressure sense switch is supplied by the No. 4 Annunciator circuit breaker, located in the cockpit's avionics circuit breaker panel.

Chart 1
OXYGEN COMPONENTS AND LOCATIONS
(Ref. Chapter 6-40-00 for zone diagrams)

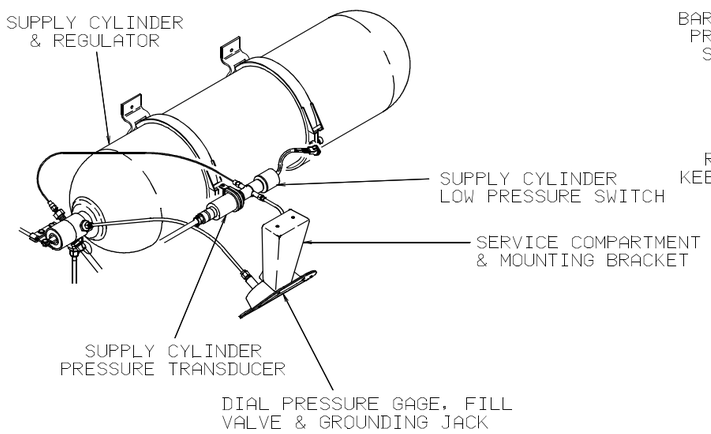
Item	Location	Zone
No. 2 Annunciator circuit breaker	Located on the cockpit's avionics circuit breaker panel.	246
No. 4 Annunciator circuit breaker	Located on the cockpit's avionics circuit breaker panel.	246
Cabin Barometric Pressure Switch	Located below the RH cockpit floor on the outboard side of the RH keel near the forward pressure bulkhead.	142
Cabin shutoff valve	Located on the aft side of the forward pressure bulkhead below the RH cockpit floor.	142
Dial Pressure Gage	Located on the oxygen service compartment.	132
Digital Pressure Indicator	Located on the copilot's inboard subpanel.	243
Fill Valve	Located on the oxygen service compartment.	132
Oxygen System Grounding Jack	Located on the oxygen service compartment.	132
LCD DSPLY LIGHTS circuit breaker	Located on the aft circuit breaker panel in the aft fuselage compartment.	320
OXY CONTROL circuit breaker	Located on the cockpit's LH circuit breaker panel.	236
Cabin Oxygen Pressure Sense Switch	Located on the aft RH side of the cabin headliner.	286
Push-Pull Control Knobs	Located on the pilot's inboard subpanel.	233
Service Compartment	Located on the RH lower nose compartment skin panel.	132
Supply Cylinder and Regulator	Located in the RH lower nose compartment.	132

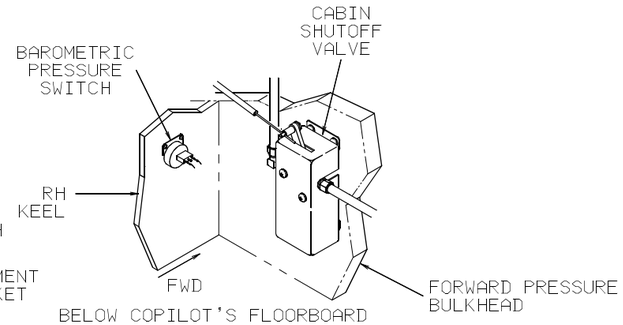
Chart 1
OXYGEN COMPONENTS AND LOCATIONS (Continued)
(Ref. Chapter 6-40-00 for zone diagrams)

Item	Location	Zone
Supply Cylinder Low Pressure Switch	Located in the RH lower nose compartment behind the oxygen service compartment.	132
Supply Cylinder Pressure Transducer	Located in the RH lower nose compartment behind the oxygen service compartment.	132
System Armed Pressure Switch	Located in the cockpit plumbing line just forward of the forward pressure bulkhead in the RH nose compartment.	212

Chart 2
OXYGEN SYSTEM SPECIFICATIONS

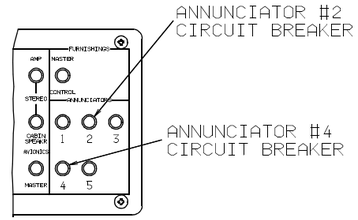
Item	Description
System Pressure (Low Side)	70 ± 10 psig.
System Pressure (High Side)	1850 ± 50 psig at 70°F.
System Armed Pressure Switch	Closes at an increasing pressure of 50 ± 3 psig. Opens at a decreasing pressure of 40 psig minimum.
Supply Cylinder Low Pressure Switch	Opens at an increasing pressure of 550 ± 30 psig. Closes at a decreasing pressure of 500 psig minimum.
Cabin Oxygen Pressure Sense Switch	Closes at an increasing pressure of 50 ± 3 psig. Opens at a decreasing pressure of 40 psig.
Cabin Barometric Pressure Switch	Closes at an increasing cabin altitude of 12,500 +0, -500 feet. Opens at a decreasing cabin altitude of 10,500 feet.
Supply Cylinder (Standard)	77 cubic foot capacity pressurized to 1850 ± 50 psid at 70°F.
Supply Cylinder (Optional)	115 cubic foot capacity pressurized to 1850 ± 50 psid at 70°F.
Cabin Oxygen Mask Compartments	Pressure of 30 - 50 psig on each mask compartment's plunger valve is required to open the compartment door.
Passenger Mask Flow Rate	4.8 ± 0.1 lpm at 62.7 psia.





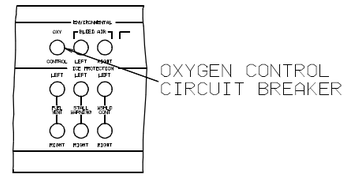
BELOW COPILOT'S FLOORBOARD

DETAIL K



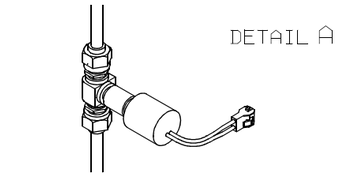
AVIONICS CIRCUIT BREAKER PANEL

DETAIL J



LH CIRCUIT BREAKER PANEL

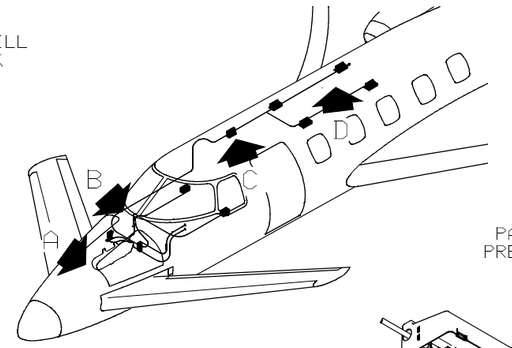
DETAIL I

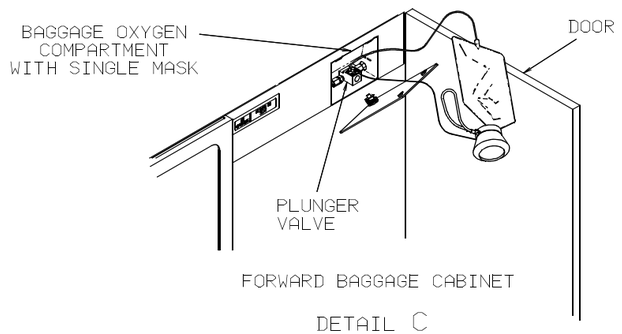


SYSTEM ARMED
PRESSURE
SWITCH
RH FORWARD WING
COMPARTMENT

DETAIL B

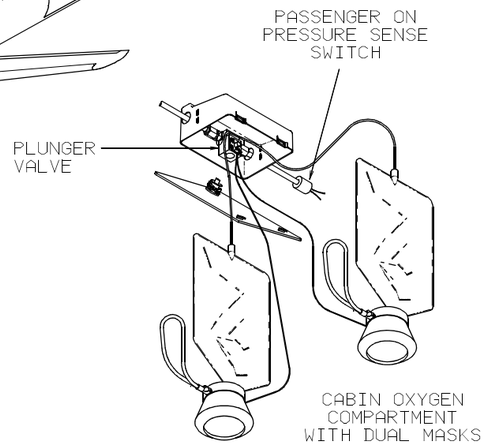
DETAIL A





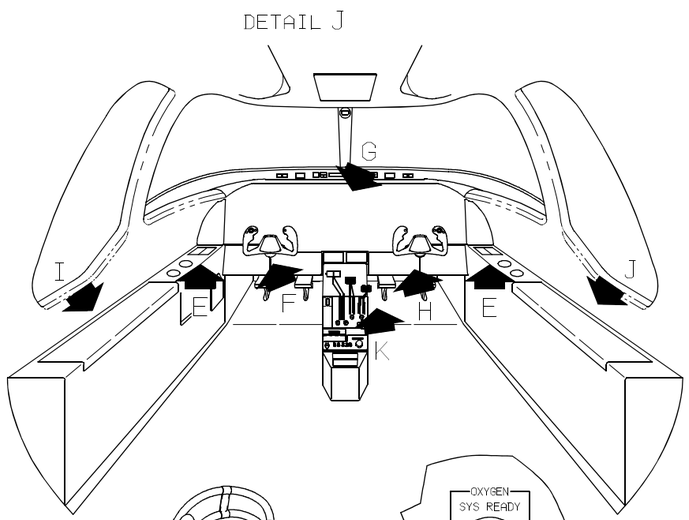
FORWARD BAGGAGE CABINET

DETAIL C



CABIN HEADLINER

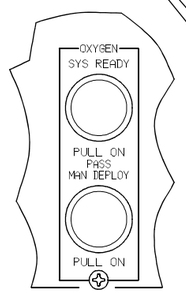
DETAIL D



COCKPIT MASK &
OUTLET VALVE

PILOT'S & COPILOT'S
SIDE CONSOLE

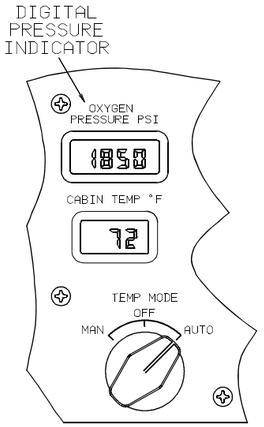
DETAIL E



PUSH/PULL CONTROLS

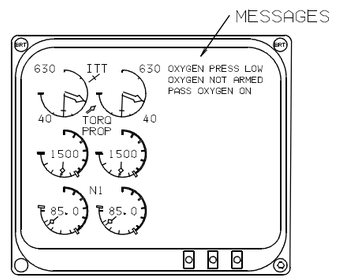
PILOT'S INBOARD SUBPANEL

DETAIL F



COPILOT'S
INBOARD SUBPANEL

DETAIL H



ENGINE INSTRUMENT, CREW
ALERTING SYSTEM (EICAS) DISPLAY
INSTRUMENT PANEL

DETAIL G

2000-341-18

Oxygen System
Figure 1

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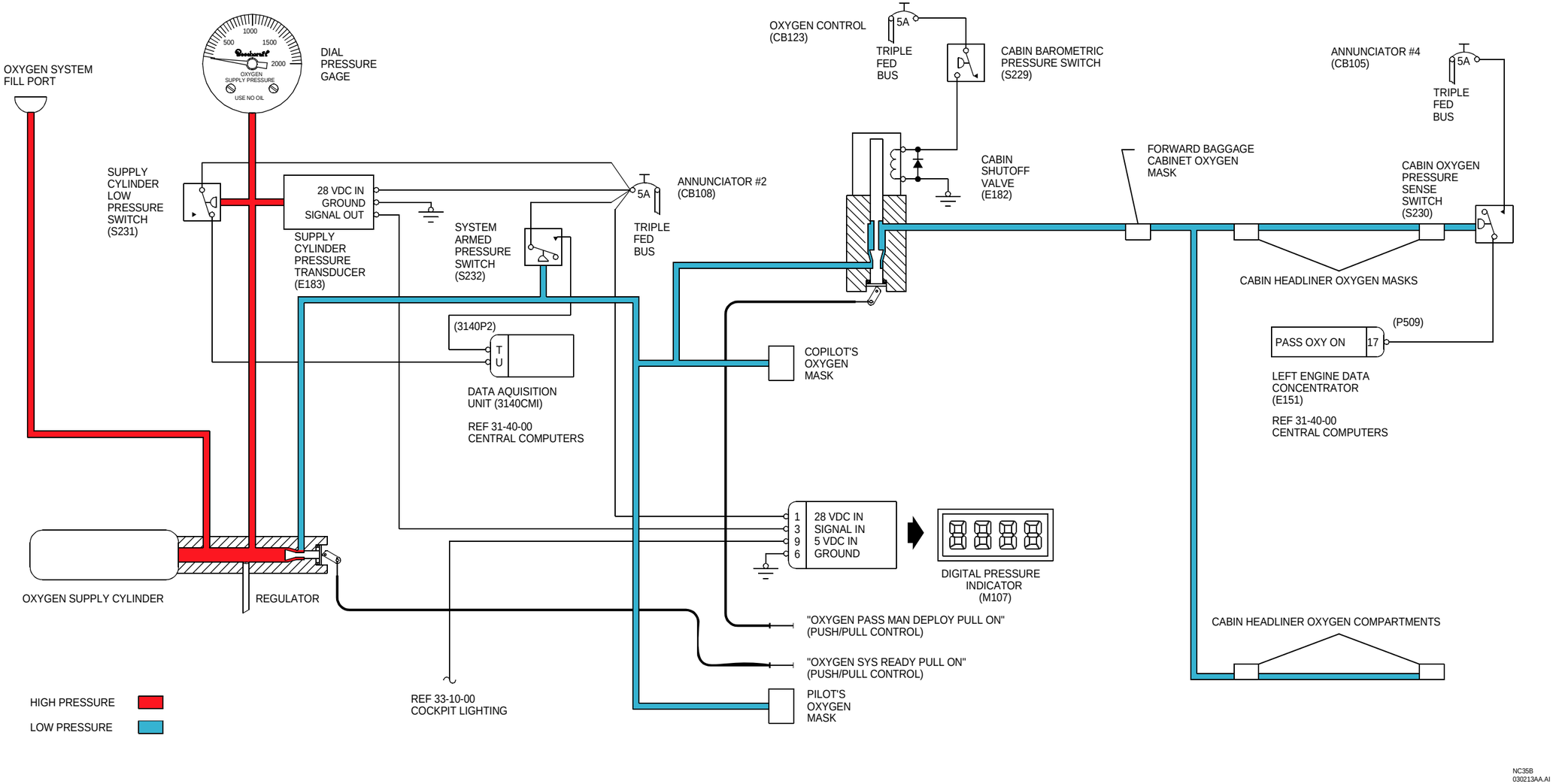

Oxygen System Functional Schematic
Figure 2

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SPECIAL TOOLS AND RECOMMENDED MATERIALS

The special tools and recommended materials listed in Charts 3 and 4 as meeting federal, military or supplier specifications are provided for reference only and are not specifically prescribed by Raytheon Aircraft Company. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Raytheon Aircraft Company, by the supplier, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the supplier of the product to be used.

Chart 3
SPECIAL TOOLS AND EQUIPMENT

TOOL NAME	PART NO.	SUPPLIER	USE
1. Cabin shutoff valve tool.	4460102-2	Raytheon Aircraft Co., Wichita, KS	Replace O-rings in the cabin oxygen shutoff valve.

Chart 4
RECOMMENDED MATERIALS

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
1. Cleaner		Anhydrous Ethyl Alcohol	Obtain Locally
2. Cleaner	TT-I-735	Isopropyl Alcohol	Obtain Locally
3. Polytetrafluourethylene Anti-Sieze Tape	MIL-T-27730		Obtain Locally
4. Oxygen, Aviator's Breathing	MIL-O-27210		Obtain Locally
5. Cleaner		Methyl Propyl Ketone	Obtain Locally
6. Sealant	MIL-S-8802, Type II	Pro Seal 890	Pro-Seal Corp., Div. of Essex Chemical Corp., Aerospace Products, Compton, CA 90220
7. Adhesive	MMM-A-121	EC1300L	Minnesota Mining and Manufacturing, 900 Bush Ave., St. Paul, MN 55144
8. Cleaner	O-T-634	1,1,1, Trichloroethane	B.F. Goodrich, 500 S. Main Street, Akron, OH 44318
9. Corrosion Preventive	MIL-C-5541	Alodine 1200	Amchem Products Inc., Spring Garden Street, Ambler, PA 19002

Chart 4
RECOMMENDED MATERIALS (Continued)

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
10. Sealant	MIL-S-8802	Pro-Seal 890	Pro-Seal Corp., Div. of Essex Chemical Corp., Aerospace Products, Compton, CA 90220
11. Leak Detector	MIL-L-25567	Leak Detecting Fluid	Obtain Locally
12. Disinfectant		Zephirah Chloride	Sierra Engineering Co., Div. of Captech Inc., Sierra Madre, CA 91024
13. Powder		Neo-Noracite	Sierra Engineering Co., Div. of Captech Inc., Sierra Madre, CA 91024

GENERAL - TROUBLESHOOTING

The following charts are provided as an aid in troubleshooting the oxygen system.

The steps in these troubleshooting charts are to be followed sequentially unless otherwise specified.

When a NO condition is encountered, without any directions immediately following, proceed to the next step.

Before any troubleshooting or maintenance procedures are attempted, personnel must review SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM found in this chapter, for general procedures and safety guidelines that pertain to handling pressurized oxygen equipment.

Chart 5
Troubleshooting - Oxygen System

Condition	Probable Cause	Remedy
1. Oxygen system loses pressure when turned on or oxygen system's use duration is too short.	a. Leaks in system's plumbing.	a. Check the entire system for leaks and repair.
2. Oxygen does not flow from crew masks when the system is turned on.	a. Oxygen cylinder empty.	a. Fill cylinder.
	b. Defective oxygen cylinder regulator.	b. Replace the oxygen cylinder and regulator assembly.
	c. Crew mask defect.	c. Overhaul or replace.
	d. OXYGEN SYS READY control cable broken.	d. Replace control cable.
	e. OXYGEN SYS READY control cable not attached to the supply cylinder's cable retainer.	e. Secure the cable to the cable retainer.
	f. Crew compartment outlet valve defective.	f. Replace outlet valve.
3. Not all cabin oxygen mask compartments deploy (Including the forward baggage cabinet's oxygen mask compartment.)	a. Defective oxygen mask compartment.	a. Replace the mask compartment.
	b. Oxygen mask not stowed properly.	b. Stow oxygen mask as described in CABIN OXYGEN MASK STOWAGE.
4. Cabin oxygen masks fail to deploy.	a. OXY CONTROL circuit breaker is tripped.	a. Reset the circuit breaker.
	b. Oxygen supply cylinder is empty.	b. Fill the oxygen cylinder.
	c. OXYGEN SYS READY control cable broken.	c. Replace the control cable.

Chart 5
Troubleshooting - Oxygen System (Continued)

Condition	Probable Cause	Remedy
	d. OXYGEN SYS READY control cable not attached to the supply cylinder.	d. Secure the cable to the supply cylinder cable retainer.
	e. OXYGEN PASS MAN DEPLOY control cable broken.	e. Replace the control cable.
	f. OXYGEN PASS MAN DEPLOY control cable not attached to the cabin shutoff valve.	f. Secure the cable to the shutoff valve.
	g. Cabin Barometric Pressure Switch malfunctioning.	g. Replace the barometric pressure switch.
	h. Cabin shutoff valve malfunctioning.	h. Replace the shutoff valve.
5. Control cables will not move (in or out) or are very hard to move.	a. Control cable is bent or broken.	a. Replace the control cable in question.

Chart 6
OXYGEN SYSTEM DIGITAL PRESSURE INDICATOR

OXYGEN DIGITAL PRESSURE INDICATOR DISPLAY WILL NOT ILLUMINATE.

Step 1	Is the LCD DSPLY LIGHTS circuit breaker tripped?	**YES**	Reset the circuit breaker.
	NO		
Step 2	Is a 5 vdc input reading available at the indicator?	**YES**	Replace the indicator.
	NO		
Step 3	Is a 5 vdc output reading available at the display light's power supply?	**YES**	Check the wire harness between the power supply and the indicator for breaks.
	NO		
Step 4	Is a 28 vdc input reading available at the display light's power supply?	**YES**	Replace the power supply.
	NO		
	Check the wire harness between the power supply and the LCD DSPLY LIGHTS circuit breaker for breaks.		

Chart 7
OXYGEN SYSTEM DIGITAL PRESSURE INDICATOR

OXYGEN DIGITAL PRESSURE INDICATOR DOES NOT DISPLAY THE CORRECT READING

Step 1	Is the No. 2 Annunciator circuit breaker tripped?	**YES**	Reset the circuit breaker.
	NO		
Step 2	Is a 28 vdc input reading available at the quantity pressure transducer?	**YES**	Go to Step 3.
	NO		
	Check the wire harness between the quantity pressure transducer and the No. 2 Annunciator circuit breaker for breaks.		
Step 3	Is a 28 vdc input reading available at the indicator?	**YES**	Go to Step 4.
	NO		
	Check the wire harness between the quantity pressure transducer and the indicator for breaks.		
Step 4	Is the voltage at the indicator's signal-in correct? (Ref. DIGITAL PRESSURE INDICATOR FUNCTIONAL TEST procedure found in this chapter.)	**YES**	Replace the indicator.
	NO		
Step 5	Is the voltage at the supply cylinder pressure transducer's signal-out correct? (Ref. SUPPLY CYLINDER PRESSURE TRANSDUCER FUNCTIONAL TEST procedure found in this chapter.)	**YES**	Check the wire harness between the transducer and the indicator for breaks.
	NO		
	Replace the transducer.		

Chart 8
OXYGEN SYSTEM ANNUNCIATION

THE "OXYGEN PRES LO" MESSAGE IS NOT DISPLAYED. (1)

Step 1	Is the No. 2 Annunciator circuit breaker tripped?	**YES**	Reset the circuit breaker.

Chart 8
OXYGEN SYSTEM ANNUNCIATION (Continued)

THE "OXYGEN PRES LO" MESSAGE IS NOT DISPLAYED. (1)

	NO		
Step 2	Is the supply cylinder low pressure switch malfunctioning? (Ref. SUPPLY CYLINDER LOW PRESSURE SWITCH FUNCTIONAL TEST procedure found in this chapter.)	**YES**	Replace the switch.
	NO		
Step 3	Is the system armed pressure switch malfunctioning? (Ref. SYSTEM ARMED PRESSURE SWITCH FUNCTIONAL TEST procedure found in this chapter.)	**YES**	Replace the switch.
	NO		
Step 4	Is a 28 vdc reading available at pin "U" of Data Acquisition Unit plug (3140P2).	**YES**	Refer to Chapter 31 for troubleshooting information on the Data Acquisition Unit and EICAS display.
	NO		
	Check the wire harness between the Data Acquisition Unit and the supply cylinder low pressure switch for breaks.		

1. The following conditions must be present. The oxygen cylinder pressure is below 500 psig, the OXYGEN SYS READY control knob is pulled ON, the airplane's electrical power is ON and the EICAS display is ON.

Chart 9
OXYGEN SYSTEM ANNUNCIATION

THE "OXYGEN NOT ARMED" MESSAGE IS NOT DISPLAYED. (1)

Step 1	Is the No. 2 Annunciator circuit breaker tripped?	**YES**	Reset the circuit breaker.
	NO		
Step 2	Is the system armed pressure switch malfunctioning? (Ref. SYSTEM ARMED SWITCH FUNCTIONAL TEST procedure found in this chapter.)	**YES**	Replace the switch.
	NO		
Step 3	Is a 28 vdc reading available at pin "T" of Data Acquisition Unit plug (3140P2).	**YES**	Refer to Chapter 31 for troubleshooting information on the Data Acquisition Unit and EICAS display.

Chart 9
OXYGEN SYSTEM ANNUNCIATION (Continued)

THE "OXYGEN NOT ARMED" MESSAGE IS NOT DISPLAYED. (1)

NO

Check the wire harness between the Data Acquisition Unit and the system armed pressure switch for breaks.

1. The following conditions must be present. The OXYGEN SYS READY control knob is pushed in, the airplane's electrical power is ON and the EICAS display is ON.

Chart 10
OXYGEN SYSTEM ANNUNCIATION

THE "PASS OXYGEN ON" MESSAGE IS NOT DISPLAYED. (1)

Step 1	Is the No. 4 Annunciator circuit breaker tripped?	**YES**	Reset the circuit breaker.
	NO		
Step 2	Is the cabin oxygen pressure sense switch malfunctioning? (Ref. CABIN OXYGEN PRESSURE SENSE SWITCH FUNCTIONAL TEST procedure found in this chapter.)	**YES**	Replace the switch.
	NO		
Step 3	Is a 28 vdc reading available at pin "17" of Left Engine Data Concentrator plug (P509)?	**YES**	Refer to Chapter 31 for troubleshooting information on the Left Engine Data Concentrator and the EICAS display.
	NO		
	Check the wire harness between the Left Engine Data Concentrator and the cabin oxygen pressure sense switch for breaks.		

1. The following conditions must be present. Both the OXYGEN SYS READY and the OXYGEN PASS MAN DEPLOY control knobs are pulled ON, the airplane's electrical power is ON, the EICAS display is ON and the passenger masks are deployed.

GENERAL - MAINTENANCE PRACTICES

SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM

The following information provides general servicing procedures and safety guidelines that must be followed when handling pressurized oxygen equipment.

Prior to servicing the airplane's oxygen system, servicing personnel must always refer to OXYGEN SYSTEM SERVICING found in this chapter and FAA Advisory Circular 43.13-1A/3 or subsequent.

The greatest safety hazard associated with handling oxygen equipment is the possibility of fire and explosion. Gaseous oxygen, under the right conditions, will cause oil and grease to ignite spontaneously and either burn or explode. It is extremely important to keep all types of oil and grease away from oxygen equipment. Oxygen servicing tools and equipment must be cleaned regularly with anhydrous ethyl alcohol (1, Chart 4, 35-00-00), isopropyl alcohol (2, Chart 5, 35-00-00) or other approved cleaning agents. After cleaning, they must be dried and kept in a clean storage area, protected with plugs, caps and covers when applicable.

Before servicing the oxygen system, all clothing and oxygen equipment contaminated with oil, grease, fuel, hydraulic fluid, etc. must be cleaned or changed; all lip creams and suntan oils must also be removed. It is highly recommended that a thorough washing of the hands and face be adopted as a standard procedure before handling any kind of oxygen equipment.

When removing components from the oxygen system, note electrical grounding and bonding hardware types and locations. Be sure that all electrical grounding and bonding hardware are installed in the same locations per instructions given under ELECTRICAL BONDING AND GROUNDING OF THE OXYGEN SYSTEM found in this chapter.

If in the process of servicing and purging the system, replacing the oxygen cylinder or any component of the system, it becomes necessary to disconnect a tapered fitting, the threads must be lubricated with anti-seize tape (3, Chart 4, 35-00-00). Do not use any lubrication on straight fittings. A swage-lock fitting should be tightened one and a quarter turns from the finger tight position.

The following list contains additional general safety precautions that must be adhered to during the servicing process.

- Always ground the system and the servicing equipment before connecting the fill adapter.
- Close the shutoff valve on the oxygen supply cylinder regulator by pushing in the OXYGEN SYS READY control knob and securing the regulator's control lever in the off position.
- Remove all electrical power from the airplane. Do not operate electrical switches, or connect or disconnect the ground auxiliary power unit during the oxygen charging operation.
- Do not service the oxygen system if fueling or other flammable fluid servicing is in process.
- Do not charge the system too fast. Rapid charging can create a dangerous overheating condition.

OXYGEN SYSTEM SERVICING

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

For information on servicing the oxygen system (Ref. Chapter 12-10-00, OXYGEN SYSTEM SERVICING).

OXYGEN SYSTEM PURGING

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

The oxygen system should be purged anytime the system's pressure drops below 50 psig or the lines are left open. Offensive odors may also be removed from the oxygen system by purging. Purging is accomplished by connecting a recharging cart to the system and permitting oxygen to flow through the lines and outlets until the offensive odors have been carried away. The following steps outline the procedures recommended for purging the oxygen system.

WARNING: Avoid making sparks and keep fire and all burning cigarettes away from the vicinity of the airplane when the outlets are in use. Inspect the fill connection for cleanliness before attaching it to the fill valve. Make certain that your hands, tools and clothing are clean, particularly of grease or oil. These contaminants, under the right conditions, will ignite upon contact with pure oxygen. As a further precaution against fire, open and close all oxygen valves slowly during filling.

When purging the oxygen system, use only MIL-0-27210 aviator's breathing oxygen (4, Chart 4, 35-00-00).

WARNING: Do not use oxygen intended for medical purposes or industrial uses such as welding. Such oxygen may contain excessive moisture that could cause the oxygen system's valves and lines to freeze up.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Ground the airplane's oxygen system by connecting a grounding cable from the oxygen recharging cart to the oxygen grounding jack, located in the airplane's oxygen service compartment, making sure the recharging cart is properly grounded.

d. Remove the fill valve protective cap and attach the hose from an oxygen recharging cart to the fill valve.

e. PULL ON both the OXYGEN SYS READY and the OXYGEN PASS MAN DEPLOY control knobs to turn on the system.

f. Open the cabin door to provide air circulation through the airplane.

g. With the flight crew masks installed, turn each mask's control knob clockwise to the EMERGENCY position (indicated by a full circle).

h. Pull the lanyard pin(s) from the valve in each cabin oxygen compartment (including the forward baggage cabinet oxygen mask compartment).

i. Set the recharging cart pressure regulator to deliver 50 psig of pressurized oxygen to the system.

j. Allow the system to purge for one hour.

k. If any offensive odor still lingers after purging for a hour, continue purging the system for an additional hour. If the odors still remain, replace the supply cylinder and repurge the system.

l. After purging, turn the flight crew mask's control knob counter-clockwise to the normal operating position (indicated by a half circle).

m. Insert the lanyard pins into the cabin oxygen mask compartment valves (including the forward baggage cabinet oxygen mask compartment).

n. Push in both the OXYGEN SYS READY and the OXYGEN PASS MAN DEPLOY control knobs to turn off the system.

o. Turn off the recharging cart.

p. Stow all oxygen masks (cockpit and cabin) in their compartments.

q. After the system has been adequately purged, service the system as instructed under OXYGEN SYSTEM SERVICING found in this chapter.

r. Restore electrical power to the airplane.

OXYGEN SUPPLY CYLINDER AND REGULATOR REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Push in the OXYGEN SYS READY control knob.

f. Secure the regulator's control lever in the off position.

g. Loosen the regulator's cable clamp bolts and the regulator control lever's cable retainer and then remove the control cable from the regulator (Ref. Figure 3).

h. Label, disconnect and cap all lines from the regulator.

i. Unfasten the T-bolts on the supply cylinder mounting clamps and remove the supply cylinder from the mounting clamps.

OXYGEN SUPPLY CYLINDER AND REGULATOR INSTALLATION

a. Position the oxygen supply cylinder in its mounting clamps and secure using the T-bolts. Be sure to orient the cylinder so the control lever (secured in the off position) is aligned with the control cable (Ref. Figure 3).

b. Connect all the lines to the regulator.

c. With the OXYGEN SYS READY control knob pushed in, install the control cable in the regulator's cable clamp and the control lever's cable retainer and tighten the retainer and cable clamp bolts.

WARNING: If the cable retainer has two different size cable attach holes, use the smaller of the two for attaching the control cable. This will ensure that the control cable is attached securely to the regulator's control lever, enabling supply cylinder operation.

d. Release the regulator's control lever.

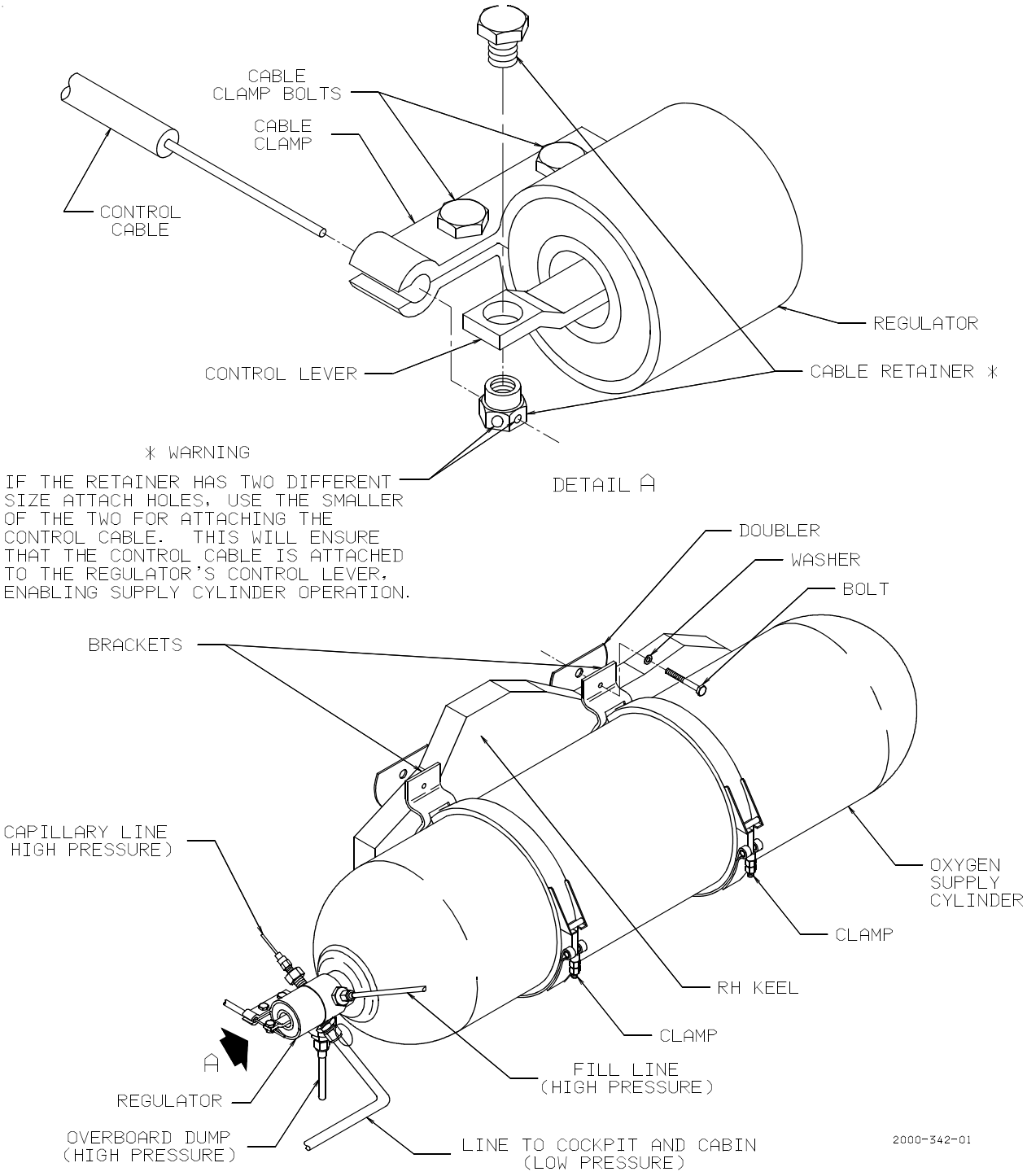
e. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

f. Install the RH lower nose compartment access panel and secure with the captive screws around the edge of the access panel.

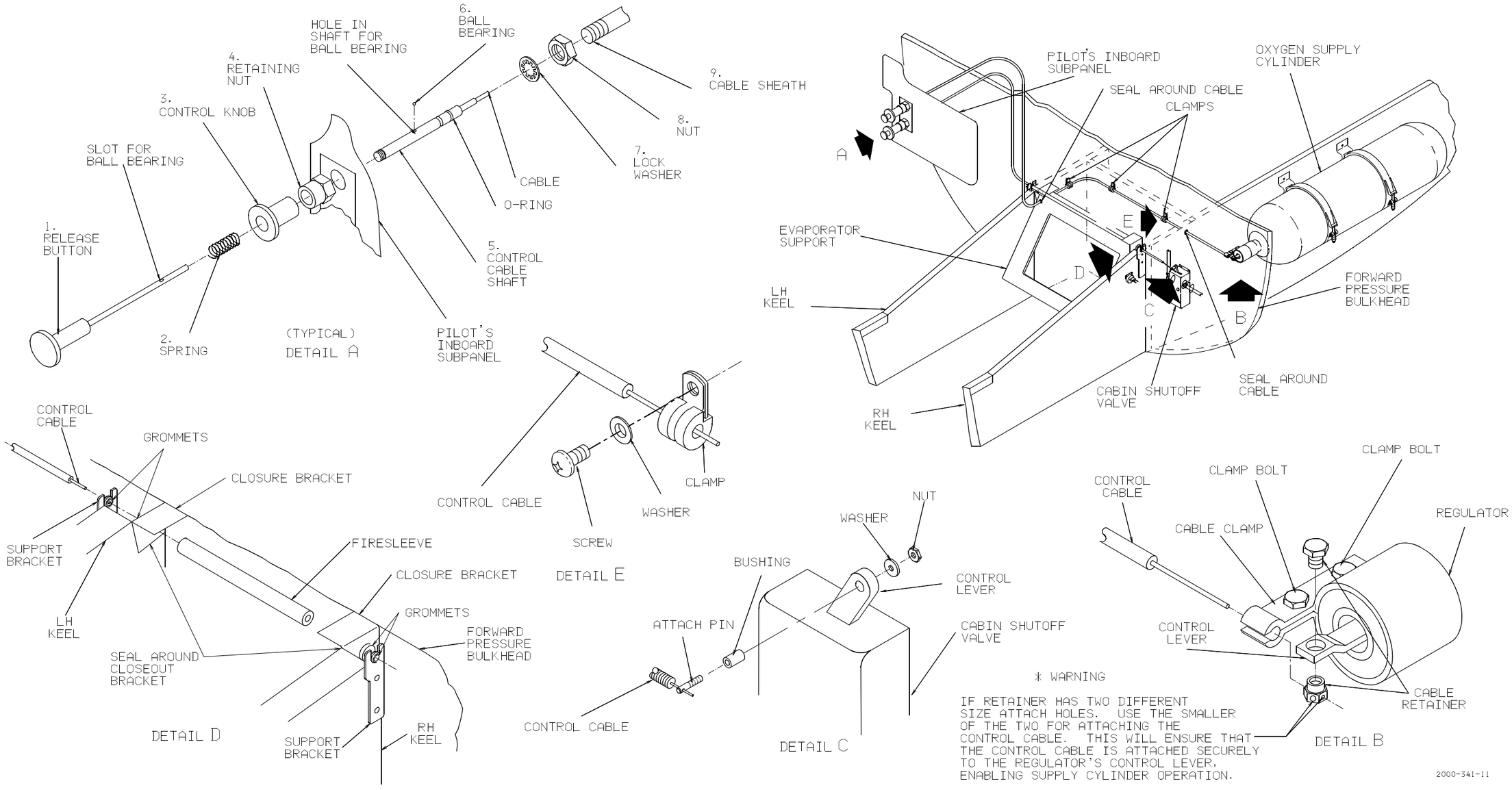
g. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

h. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

i. Restore electrical power to the airplane.



Oxygen Supply Cylinder and Regulator
Figure 3



Oxygen System Control Cables
Figure 4

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OXYGEN SUPPLY CYLINDER MOUNTING BRACKETS REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Remove the oxygen supply cylinder as instructed under OXYGEN SUPPLY CYLINDER AND REGULATOR REMOVAL found in this chapter.

c. Remove the bolts and washers that secure the mounting brackets to the keel and remove the mounting brackets (Ref. Figure 3).

OXYGEN SUPPLY CYLINDER MOUNTING BRACKETS INSTALLATION

a. Position the mounting brackets to the keels and secure with the bolts and washers (Ref. Figure 3).

b. Install the oxygen supply cylinder as instructed under OXYGEN SUPPLY CYLINDER AND REGULATOR INSTALLATION found in this chapter.

c. Restore electrical power to the airplane.

OXYGEN SYSTEM READY CONTROL CABLE REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Remove the pilot's seat and carpeting (Ref. Chapter 25-10-00).

c. Loosen the captive screws around the edge of the pilot's forward cockpit floorboard and remove the floorboard.

d. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

e. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

f. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

g. Push in the OXYGEN SYS READY control knob.

h. Secure the regulator's control lever in the off position.

i. Remove the sealant from around the control cable where it penetrates the forward evaporator support and the RH keel using a plastic scraper (NEVER USE A METAL SCRAPER) and cleaning solvent (5, Chart 4, 35-00-00).

j. Loosen the regulator's cable clamp bolts and the regulator control lever's cable retainer and then remove the control cable from the regulator.

k. Depress the release button (1) and pull out the control knob (3) (Ref. Figure 4).

l. Loosen the retaining nut (4).

m. With the release button (1) still depressed, continue pulling out the control knob (3) until the hole in the control cable shaft (5) is exposed. Remove the ball bearing (6) from the hole.

n. Remove the release button (1) and spring (2).

o. Unscrew and remove the control knob (3) from the control cable shaft (5).

p. Remove the retaining nut (4).

q. Remove the control cable from the back side of the pilot's inboard subpanel and remove the lockwasher (7) and nut (8) from the cable sheath (9).

r. Remove the pedestal side upholstery panels (Ref. Chapter 25-10-00).

s. Remove the screws located around the edge of the pressurization exit ramp and remove the ramp.

t. Remove the forward blower shroud (Ref. Chapter 21-20-00).

u. Remove the outflow valve mount panel (Ref. Chapter 21-30-00).

v. Reaching through the outflow valve mount opening, remove the clamps, screws and washers securing the control cable to the compartment walls and then remove the control cable.

OXYGEN SYSTEM READY CONTROL CABLE INSTALLATION

a. Route the OXYGEN SYS READY control cable through the hole in the RH keel forward of the forward pressure bulkhead, the hole in the forward evaporator support and up the aft side of the forward pressure bulkhead to the pilot's inboard subpanel.

b. Install the nut (8) and lockwasher (7) onto the cable sheath (9) and then insert the control cable into the opening in the backside of the pilot's inboard subpanel labeled OXYGEN SYS READY (Ref. Figure 4).

c. Install the retaining nut (4) on the cable sheath (9).

d. Screw the control knob (3) on the control cable shaft (5) and then pull out the control cable shaft (5) until the hole in it is visible.

e. Assemble the release button (1) and spring (2) and insert them into the control cable shaft (5).

f. With the release button (1) depressed place the ball bearing (6) into the hole in the control cable shaft (5) then insert the cable into the cable sheath.

g. Tighten the retaining nut (4).

h. With the OXYGEN SYS READY control knob pushed in, install the control cable in the regulator's cable clamp and the control lever's cable retainer and tighten the retainer and clamp bolts.

WARNING: If the retainer has two different size attach holes, use the smaller of the two for attaching the control cable. This will ensure that the control cable is attached securely to the regulator's control lever, enabling supply cylinder operation.

i. Reaching through the outflow valve mounting opening, secure the control cable to the compartment walls with the clamps, screws and washers.

j. Seal the holes where the control cable penetrates the RH keel and the forward evaporator support with sealant (6, Chart 4, 35-00-00).

k. Install the pilot's forward cockpit floorboard and secure with the captive screws around the edge of the floorboard.

l. Install the forward blower shroud (Ref. Chapter 21-20-00).

m. Install the pedestal side upholstery panel (Ref. Chapter 25-10-00).

n. Install the outflow valve mount panel (Ref. Chapter 21-30-00).

o. Install the pressurization exit ramp and secure with the screws around the edge of the ramp.

p. Release the regulator's control lever.

q. Install the RH lower nose compartment access panel and secure with the captive screws around the edge of the panel.

r. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

s. Install the oxygen service compartment access panel and secure with the captive screws around the edge of the panel.

t. Install the pilot's seat and carpeting (Ref. Chapter 25-10-00).

u. Restore electrical power to the airplane.

OXYGEN PASSENGER MANUAL DEPLOY CONTROL CABLE REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Remove the cockpit seats and carpeting (Ref. Chapter 25-10-00).

c. Loosen the captive screws from around the edge of the pilot's and copilot's forward cockpit floorboards and remove the floorboards.

d. Remove the pedestal side upholstery panels (Ref. Chapter 25-10-00).

e. Remove the forward blower shroud (Ref. Chapter 21-20-00).

f. Push in the OXYGEN PASS MAN DEPLOY control knob.

g. Secure the cabin shutoff valve's control lever in the off position.

h. Detach the control cable from the cabin shutoff valve control lever's attach pin by loosening the nut.

i. Depress the release button (1) and pull out the control knob (3) (Ref. Figure 4).

j. Loosen the retaining nut (4).

k. With the release button (1) still depressed, continue pulling out the control knob (3) until the hole in the control cable shaft (5) is exposed. Remove the ball bearing (6) from the hole.

l. Remove the release button (1) and spring (2).

m. Unscrew and remove the control knob (3) from the control cable shaft (5).

n. Remove the retaining nut (4).

o. Remove the control cable from the backside of the pilot's inboard subpanel and remove the lockwasher (7) and nut (8) from the cable sheath (9).

p. Remove the control cable from the LH support bracket and closure bracket, the firesleeve and the RH closure bracket and support bracket.

OXYGEN PASSENGER MANUAL DEPLOY CONTROL CABLE INSTALLATION

a. Route the OXYGEN PASS MAN DEPLOY control cable through the RH support bracket and closure bracket, the firesleeve and the LH closure bracket and support bracket.

NOTE: Replace grommets in the closure brackets and the support brackets if they show signs of wear or age (brittleness and cracking).

b. Install the nut (8) and lockwasher (7) onto the cable sheath (9) and then insert the control cable into the opening in the backside of the pilot's inboard subpanel labeled OXYGEN PASS MAN DEPLOY (Ref. Figure 4).

c. Install the retaining nut (4) on the cable sheath (9).

d. Screw the control knob (3) on the control cable shaft (5) and then pull out the control cable shaft (5) until the hole in it is visible.

e. Assemble the release button (1) and spring (2) and insert them into the control cable shaft (5).

f. With the release button (1) depressed, place the ball bearing (6) into the hole in the control cable shaft (5) then insert the shaft into the cable sheath.

g. Tighten the retaining nut (4).

h. Connect the control cable to the cabin shutoff valve control lever's attach pin and secure by tightening the nut.

i. Release the shutoff valve's control lever.

j. Install the forward blower shroud (Ref. Chapter 21-20-00).

k. Install the pedestal side upholstery panel (Ref. Chapter 25-10-00).

l. Install the pilot's and copilot's forward floorboards and secure with the captive screws located around the edge of the floorboard.

m. Install the cockpit seats and carpeting (Ref. Chapter 25-10-00).

n. Restore electrical power to the airplane.

OXYGEN SYSTEM FILL VALVE REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Disconnect the bonding jumper from the oxygen fill line bonding clamp by removing the screw, washers and nut that secure the bonding jumper to the clamp (Ref. Figure 5).

f. Disconnect the oxygen fill line from the regulator and cap the fill line.

WARNING: Always disconnect the oxygen fill line from the regulator before removing it from the fill valve to insure the flow of high pressure oxygen from the supply cylinder is shut off.

g. Disconnect the oxygen fill line from the fill valve and cap the fill line.

h. Remove the fill valve cap from the fill valve.

i. Disconnect the screws, washers and nuts securing the fill valve to the service compartment and remove the fill valve from the backside of the compartment.

OXYGEN SYSTEM FILL VALVE INSTALLATION

a. Position the fill valve in the service compartment and secure with its screws, washers and nuts (Ref. Figure 5).

b. Install the fill valve cap on the fill valve.

c. Connect the oxygen fill line to the fill valve.

WARNING: Always connect the oxygen fill line to the fill valve before connecting it to the regulator to prevent a premature discharge of high pressure oxygen from the supply cylinder.

d. Connect the oxygen fill line to the regulator.

e. Connect the bonding jumper to the oxygen fill line using the bonding clamp, screw, washers and nut as instructed under ELECTRICAL BONDING AND GROUNDING OF THE OXYGEN SYSTEM found in this chapter.

f. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

g. Install the RH lower nose compartment access panel and secure with the captive screws around the edge of the panel.

h. Install the screws attaching the RH lower nose compartment access panel to the service compartment.

i. Install the oxygen service compartment access panel and secure with the captive screws around the edge of the panel.

j. Restore electrical power to the airplane.

DIAL PRESSURE GAGE REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Disconnect the capillary line from the regulator and cap the capillary line (Ref. Figure 5).

WARNING: Always disconnect the capillary line from the regulator before removing the oxygen line from the dial pressure gage to insure the flow of high pressure oxygen from the supply cylinder is shut off.

f. Disconnect the oxygen line from the dial pressure gage and cap the oxygen line.

g. Remove the nuts and retainer securing the dial pressure gage to the service compartment and remove the gage from the compartment.

DIAL PRESSURE GAGE INSTALLATION

a. Position the dial pressure gage in the service compartment and secure with the retainer and nuts (Ref. Figure 5).

b. Connect the oxygen line to the dial pressure gage.

WARNING: Always connect the oxygen line to the dial pressure gage before connecting the capillary line to the regulator to prevent a premature discharge of high pressure oxygen from the supply cylinder.

c. Connect the capillary line to the regulator.

d. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

e. Install the RH lower nose compartment access panel and secure with the captive screws around the edge of the panel.

f. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

g. Install the oxygen service compartment access panel and secure with the captive screws around the edge of the panel.

h. Restore electrical power to the airplane.

OXYGEN SYSTEM GROUNDING JACK REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Remove the nut and washer securing the bonding jumper to the grounding jack (Ref. Figure 5).

f. Remove the nut securing the grounding jack to the service compartment and remove the grounding jack from the compartment.

OXYGEN SYSTEM GROUNDING JACK INSTALLATION

a. Position the grounding jack in the service compartment and secure with the nut (Ref. Figure 5).

b. Place the bonding jumper on the grounding jack and secure with the washer and nut.

c. Install the RH lower nose compartment access panel and secure with the captive screws around the edge of the panel.

d. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

e. Install the oxygen service compartment access panel and secure with the captive screws around the edge of the panel.

f. Restore electrical power to the airplane.

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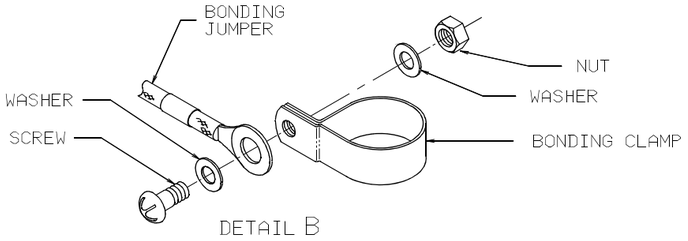

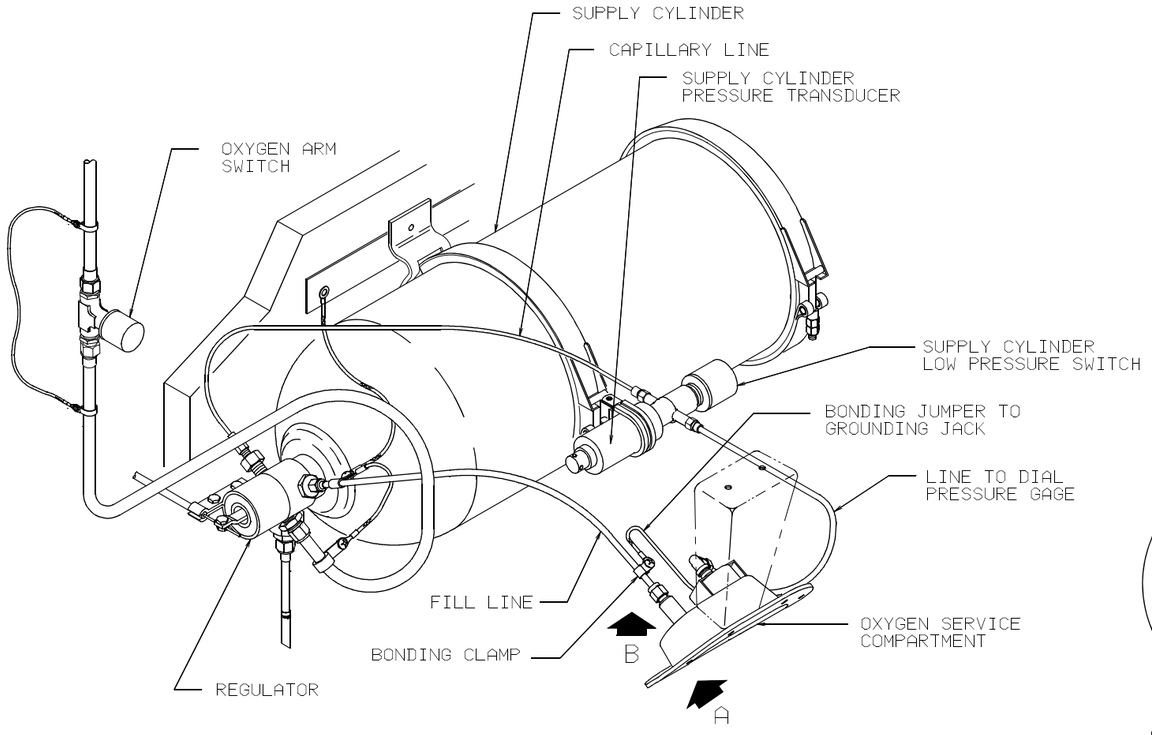

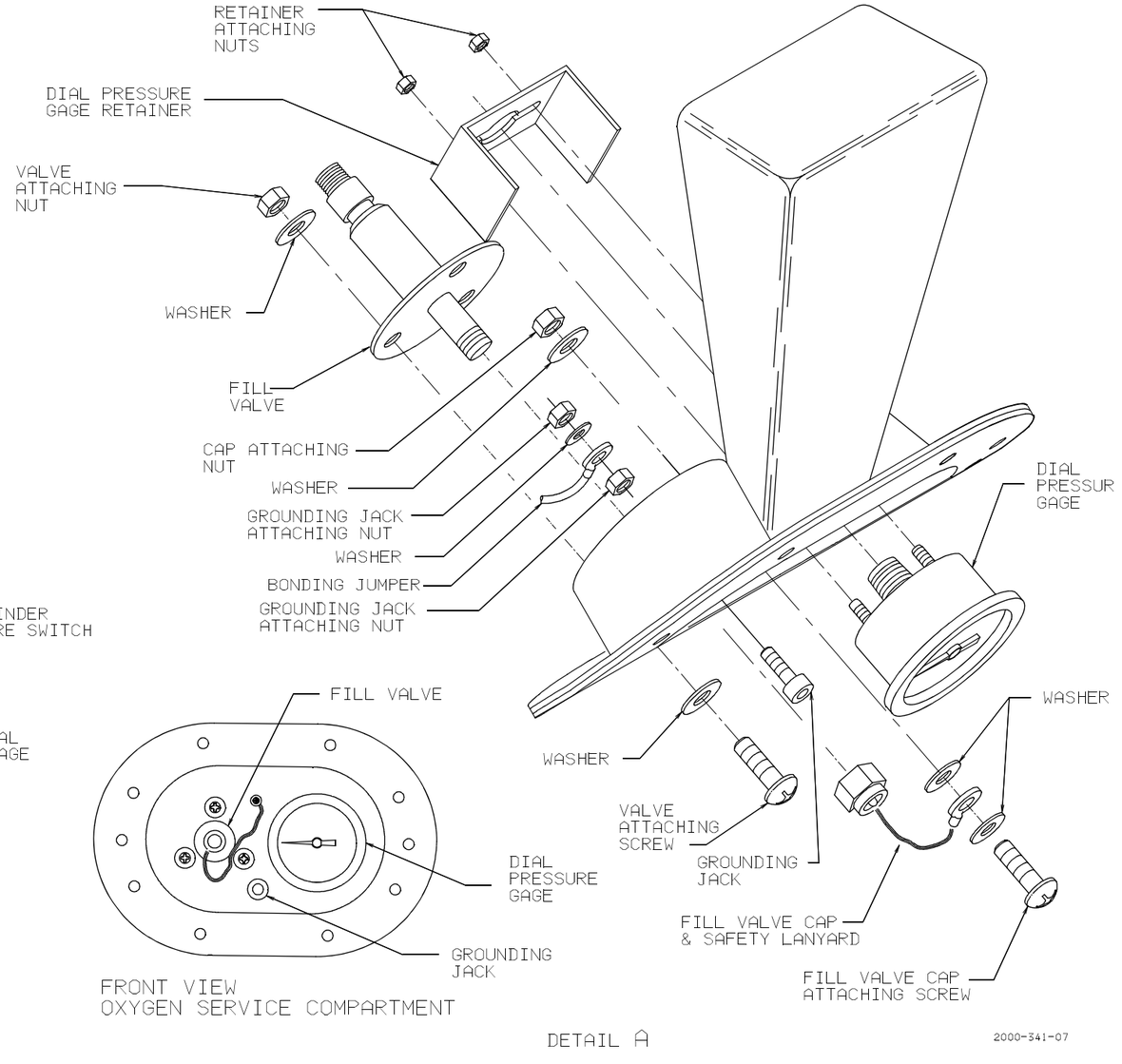

Oxygen System Service Compartment Components
Figure 5

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OXYGEN SERVICE COMPARTMENT REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Remove the fill valve, dial pressure gage and grounding jack as instructed under OXYGEN SYSTEM FILL VALVE REMOVAL, DIAL PRESSURE GAGE REMOVAL and OXYGEN SYSTEM GROUNDING JACK REMOVAL found in this chapter.

f. Remove the bolts and washers securing the service compartment mounting bracket to the RH lower nose compartment shelf and remove the service compartment (Ref. Figure 6).

OXYGEN SERVICE COMPARTMENT INSTALLATION

a. Position the oxygen service compartment to the RH lower nose compartment shelf and secure the service compartment mounting bracket with the bolts and washers (Ref. Figure 6).

b. Install the grounding jack, dial pressure gage and fill valve as instructed under OXYGEN SYSTEM GROUNDING JACK INSTALLATION, DIAL PRESSURE GAGE INSTALLATION and OXYGEN SYSTEM FILL VALVE INSTALLATION found in this chapter.

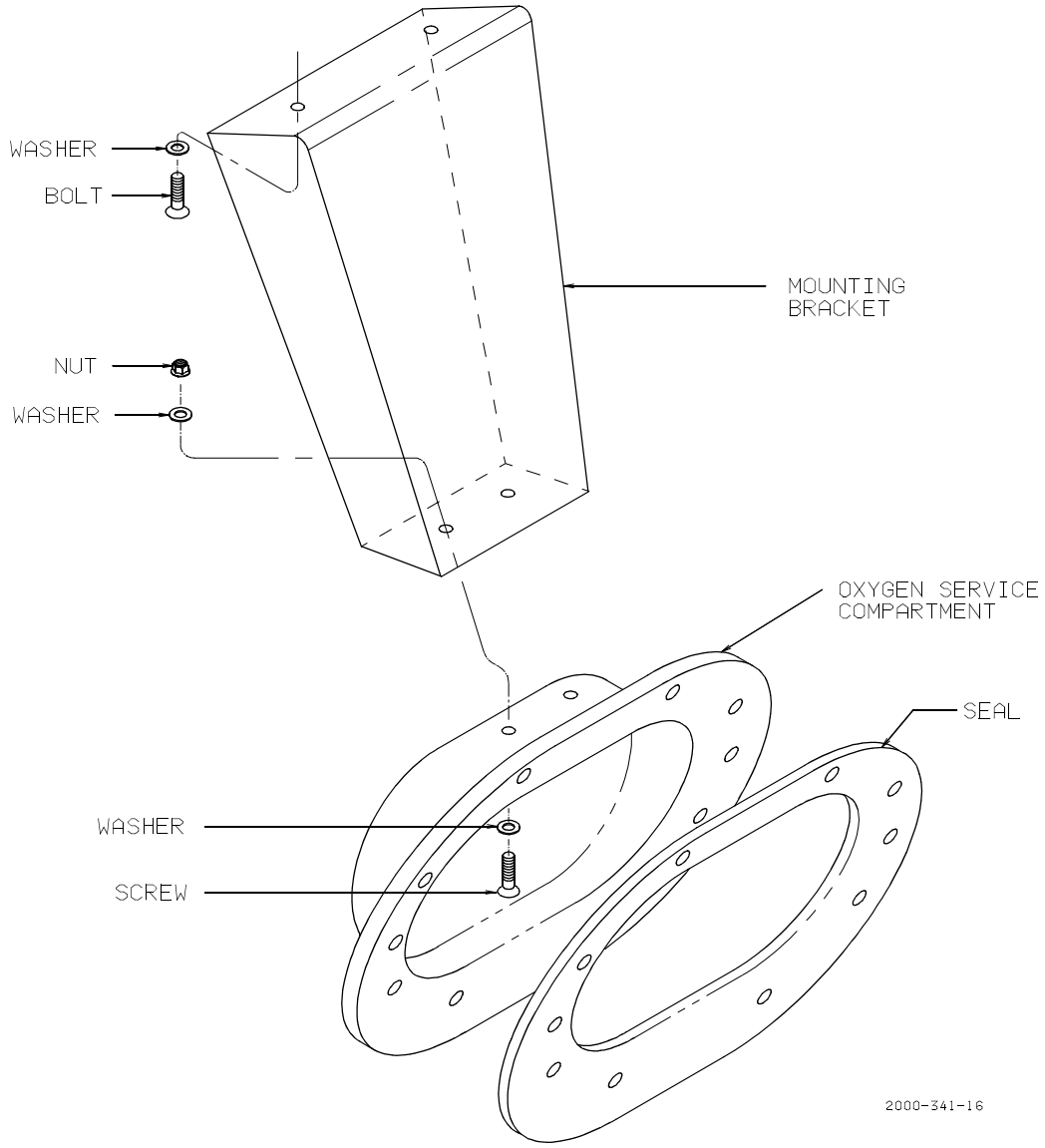
c. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

d. Install the RH lower nose compartment access panel and secure with the captive screws around the edge of the panel.

e. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

f. Install the oxygen service compartment access panel and secure with the captive screws around the edge of the panel.

g. Restore electrical power to the airplane.



Oxygen Service Compartment Mounting Bracket and Seal
Figure 6

OXYGEN SERVICE COMPARTMENT SEAL REPLACEMENT

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Remove the seal from the oxygen service compartment using a plastic scraper (NEVER USE A METAL SCRAPER) and cleaning solvent (5, Chart 4, 35-00-00) (Ref. Figure 6).

f. Prepare the oxygen service compartment surface and bond the seal to it with adhesive (7, Chart 4, 35-00-00) as follows:

   1. Surfaces to be bonded must be clean, dry and free from oil and/or grease.

   2. Apply a thin, even coat of adhesive to each surface to be bonded.

   3. Allow the adhesive to dry until it is quite tacky and will not transfer when touched.

   4. Press mating surfaces firmly together.

   5. Part is ready to use immediately and will be fully cured in 3 to 7 days.

g. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

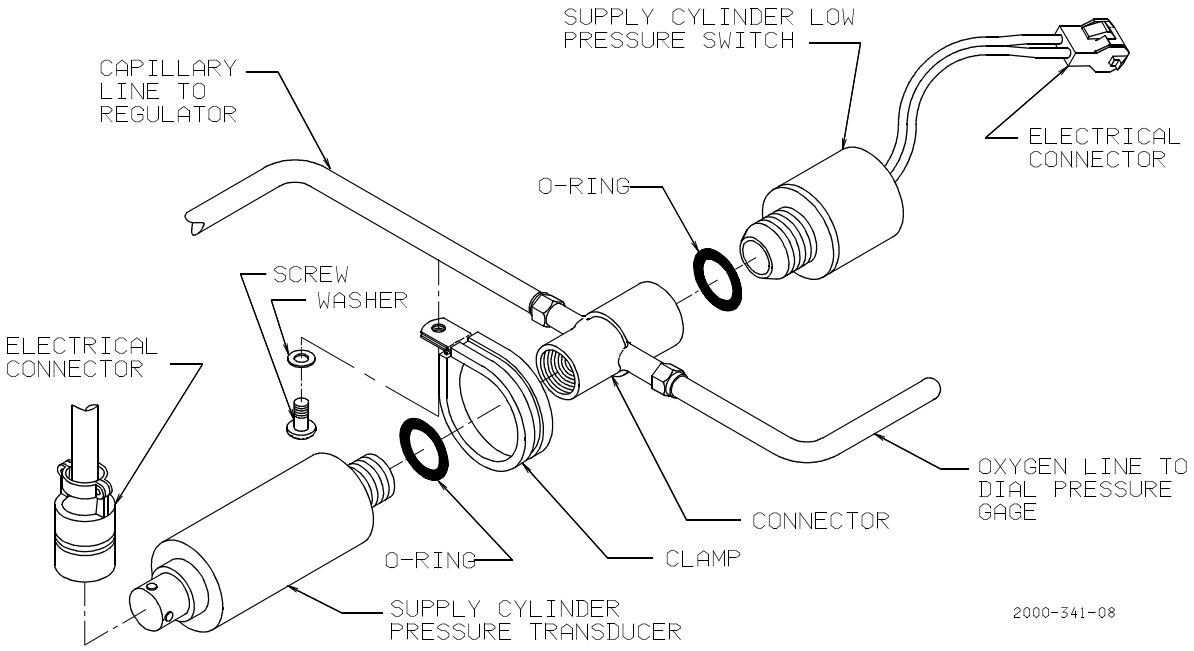
h. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

i. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

j. Restore electrical power to the airplane.

OXYGEN DIGITAL PRESSURE INDICATOR

Refer to Chapter 31-10-00 for removal and installation instructions.



Supply Cylinder Pressure Transducer & Low Pressure Switch
Figure 7

SUPPLY CYLINDER PRESSURE TRANSDUCER REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Disconnect the electrical connector from the supply cylinder pressure transducer (Ref. Figure 7).

f. Remove the screw and washer securing the clamp to the underside of the RH lower nose compartment shelf and remove the clamp.

g. Disconnect the capillary line from the regulator and cap the capillary line.

WARNING: Always disconnect the capillary line from the regulator before removing the transducer from its connector to insure the flow of high pressure oxygen from the supply cylinder is shut off.

h. Remove the safety wire from the transducer, remove the transducer from the connector and cover the connector's opening. Provide support for the connector, plumbing and low pressure switch.

i. Remove and discard the O-ring from the transducer.

SUPPLY CYLINDER PRESSURE TRANSDUCER INSTALLATION

a. Install a new O-ring on the supply cylinder pressure transducer (Ref. Figure 7).

b. With the plumbing, connector and supply cylinder low pressure switch supported, install the transducer in the connector. Torque to 50 in-lbs. and safety.

WARNING: Always install the transducer in its connector before connecting the capillary line to the regulator to prevent a premature discharge of high pressure oxygen from the supply cylinder.

c. Connect the capillary line to the regulator.

d. Secure the transducer to the shelf with the clamp, washer and screw.

e. Connect the electrical connector to the transducer.

f. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

g. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

h. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

i. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

j. Restore electrical power to the airplane.

SUPPLY CYLINDER LOW PRESSURE SWITCH REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Disconnect the supply cylinder low pressure switch's electrical connector (Ref. Figure 7).

f. Disconnect the capillary line from the regulator and cap the capillary line.

WARNING: Always disconnect the capillary line from the regulator before removing the pressure switch from the connector to insure the flow of high pressure oxygen from the supply cylinder is shut off.

g. Remove the safety wire from the supply cylinder low pressure switch and remove the switch from the connector. Cover the connector opening.

h. Remove and discard the O-ring from the switch.

SUPPLY CYLINDER LOW PRESSURE SWITCH INSTALLATION

a. Install a new O-ring on the supply cylinder low pressure switch (Ref. Figure 7).

b. Install the switch in the connector and safety wire.

WARNING: Always install the pressure switch in the connector before connecting the capillary line to the regulator to prevent a premature discharge of high pressure oxygen from the supply cylinder.

c. Connect the capillary line to the regulator.

d. Connect the supply cylinder low pressure switch's electrical connector.

e. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

f. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

g. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

h. Install the oxygen service compartment access panel and secure with the captive screws around the edge of the panel.

i. Restore electrical power to the airplane.

CABIN SHUTOFF VALVE REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws found around the edge of the oxygen service.

d. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

e. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

f. Push in the OXYGEN SYS READY control knob.

g. Secure the regulator's control lever in the off position.

h. Remove the copilot's seat and carpeting (Ref. Chapter 25-10-00).

i. Loosen the captive screws located around the edge of the copilot's forward cockpit floorboard and remove the floorboard.

j. Disconnect the electrical wiring from the cabin shutoff valve (Ref. Figure 8).

k. Push in the OXYGEN PASS MAN DEPLOY control knob.

l. Disconnect the control cable from the shutoff valve control lever's attach pin by loosening the nut and pulling the cable out of the attach pin.

m. Remove the screws and washers attaching the cover, shutoff valve

n. Remove the screws and washers attaching the cover, shutoff valve and spacers to the mounting bracket and remove the cover and spacers.

o. Disconnect the oxygen hoses and fitting from the shutoff valve's inboard and outboard unions. Cap the oxygen hoses.

p. Remove the unions from the shutoff valve. Remove and discard the O-rings from the unions.

CABIN SHUTOFF VALVE INSTALLATION

a. Install new O-rings on the unions. Install the unions in the cabin shutoff valve (Ref. Figure 8).

b. Connect the oxygen hose and fitting from the regulator to the shutoff valve's inboard union and the oxygen hose from the cabin oxygen compartments to the shutoff valve's outboard union.

c. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

d. Position the cover, shutoff valve and spacers to the mounting bracket and secure with the screws and washers.

e. With the OXYGEN PASS MAN DEPLOY control knob pushed in and the shutoff valve's control lever in the off position (lever pointing toward the out port of the shutoff valve), connect the control cable to the shutoff valve control lever attach pin by sliding the cable into the attach pin hole and then tightening the nut.

f. Connect the electrical wiring to the shutoff valve.

g. Install the copilot's forward cockpit floorboard and secure with the captive screws located around the edge of the floorboard.

h. Install the copilot's seat and chair (Ref. Chapter 25-10-00).

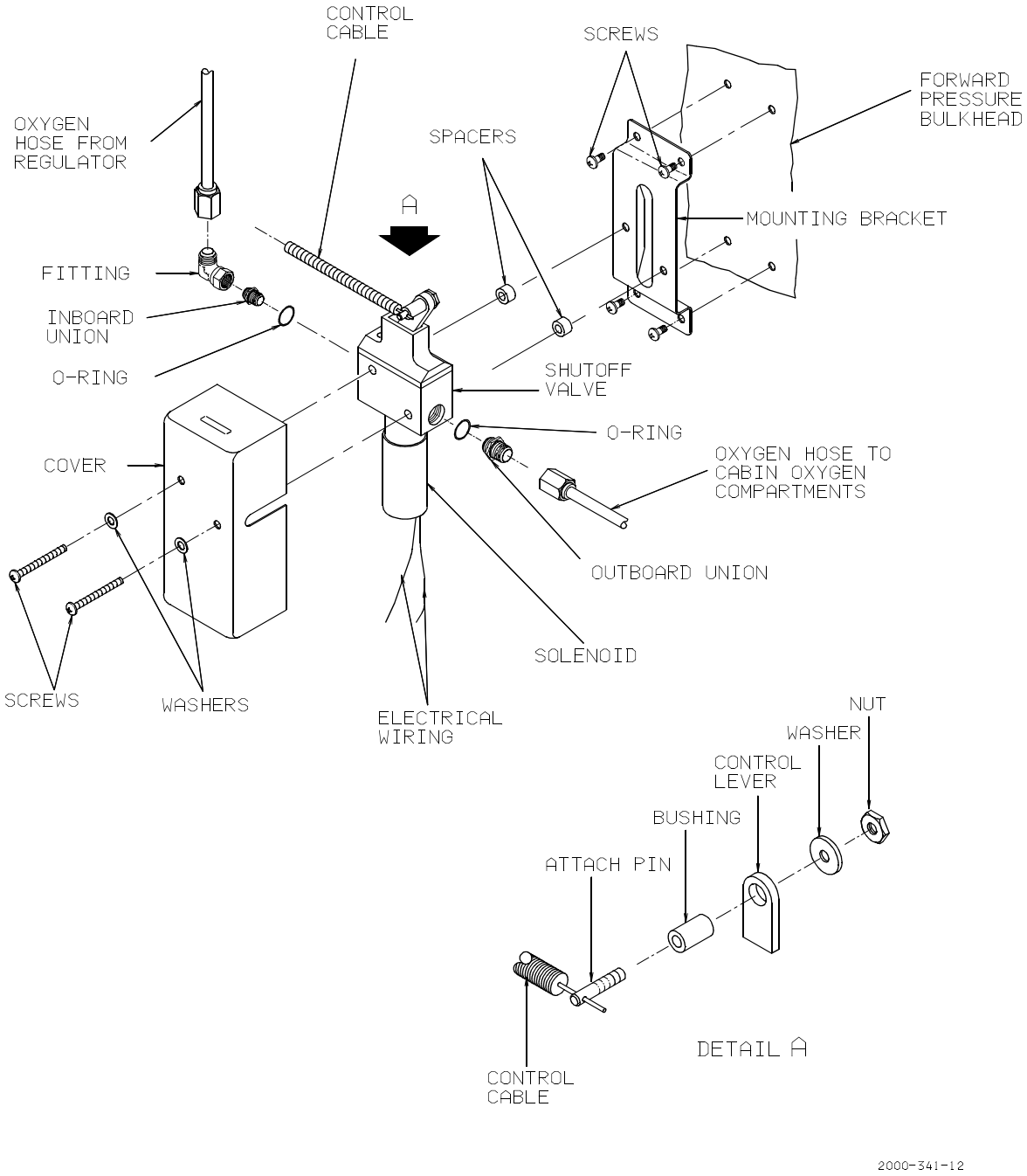
i. Release the regulator's control lever.

j. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

k. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

l. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

m. Restore electrical power to the airplane.



Cabin Shutoff Valve Installation
Figure 8

CABIN SHUTOFF VALVE O-RING REPLACEMENT

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

Every 24 months, disassemble the cabin shutoff valve and replace the O-rings. Because this valve is actuated so infrequently in normal service there is the possibility of the O-rings deteriorating, which could cause the valve to leak oxygen (O-ring deterioration does not hinder valve operation). Parts and instructions for replacing the O-rings are contained in the following Kits: 4460102-1 (O-rings and instructions), 4460102-2 (1, Chart 3, 35-00-00) (reusable tool) and 4460102-5 (lubricant) which may be ordered through Raytheon Aircraft Service Centers.

SYSTEM ARMED PRESSURE SWITCH REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Push in the OXYGEN SYS READY control knob.

f. Secure the regulator's control lever in the off position.

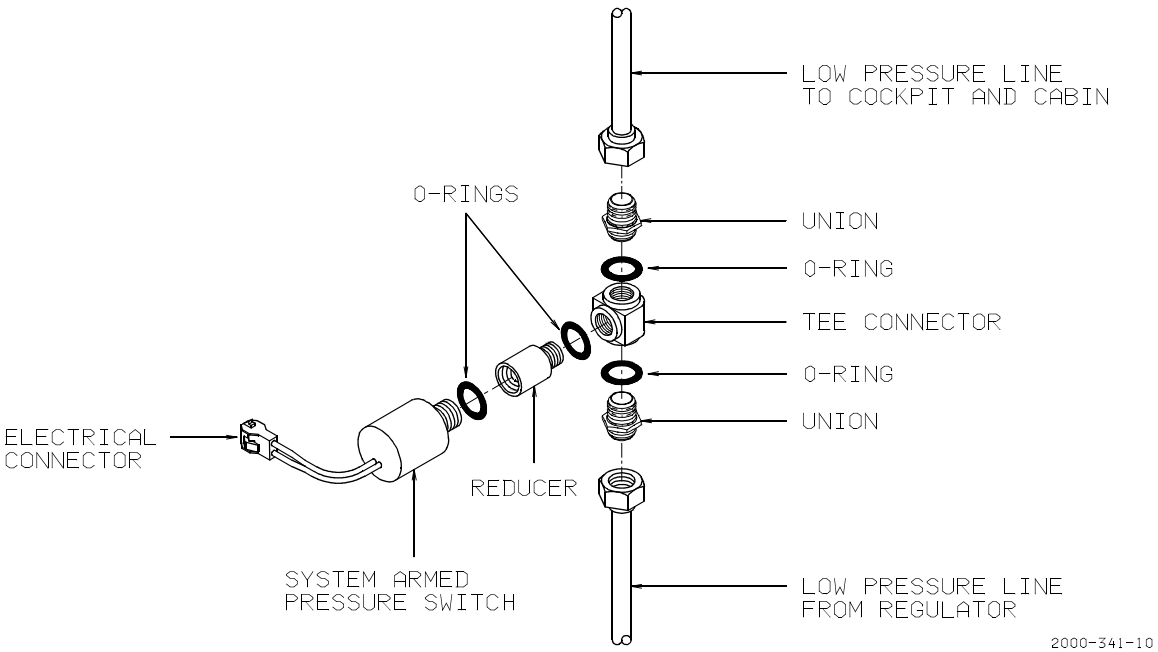
g. Loosen the captive screws located around the edge of the RH nose forward wing fairings and remove the fairings.

h. Disconnect the system armed pressure switch's electrical connector (Ref. Figure 9).

i. Remove the safety wire from the system armed pressure switch and remove the switch and the reducer from the Tee connector. Cover the Tee connector's opening.

j. Remove the switch from the reducer.

k. Remove the O-rings from the switch and the reducer.



System Armed Pressure Switch
Figure 9

SYSTEM ARMED PRESSURE SWITCH INSTALLATION

a. Install new O-rings on the system armed pressure switch and the reducer (Ref. Figure 9).

b. Install the switch in the reducer.

c. Install the switch and the reducer in the Tee connector and safety wire.

d. Connect the switch's electrical connector.

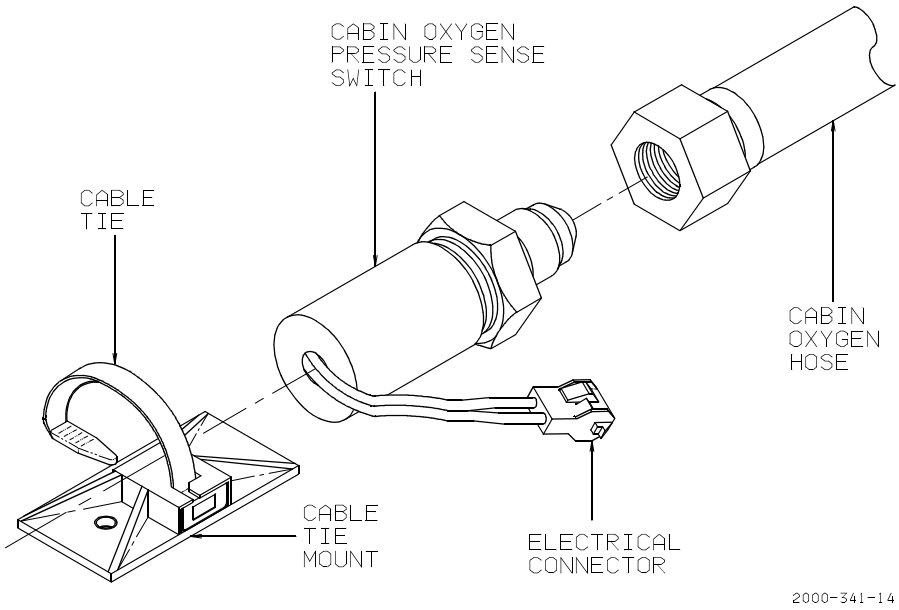
e. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

f. Install the RH nose forward wing fairings and secure with the captive screws located around the edge of the fairings.

g. Release the regulator's control lever.

h. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

i. Install the screws which attach the RH lower nose compartment access panel to the service compartment.



Cabin Oxygen Pressure Sense Switch
Figure 10

j. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

k. Restore electrical power to the airplane.

CABIN OXYGEN PRESSURE SENSE SWITCH REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Push in the OXYGEN SYS READY control knob.

f. Secure the regulator's control lever in the off position.

g. Remove the cabin headliner (Ref. Chapter 25-20-00).

h. Disconnect the cabin oxygen pressure sense switch's electrical connector (Ref. Figure 10).

i. Cut the cable tie securing the switch to the headliner.

j. Remove the switch from the cabin oxygen hose and cover the hose's opening.

CABIN OXYGEN PRESSURE SENSE SWITCH INSTALLATION

a. Install the cabin oxygen pressure sense switch in cabin oxygen hose (Ref. Figure 10).

b. Install a new cable tie in the headliner mount and secure the switch to the headliner with it.

c. Connect the switch's electrical connector.

d. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

e. Install the cabin headliner (Ref. Chapter 25-20-00).

f. Release the regulator's control lever.

g. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

h. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

i. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

j. Restore electrical power to the airplane.

CABIN BAROMETRIC PRESSURE SWITCH REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

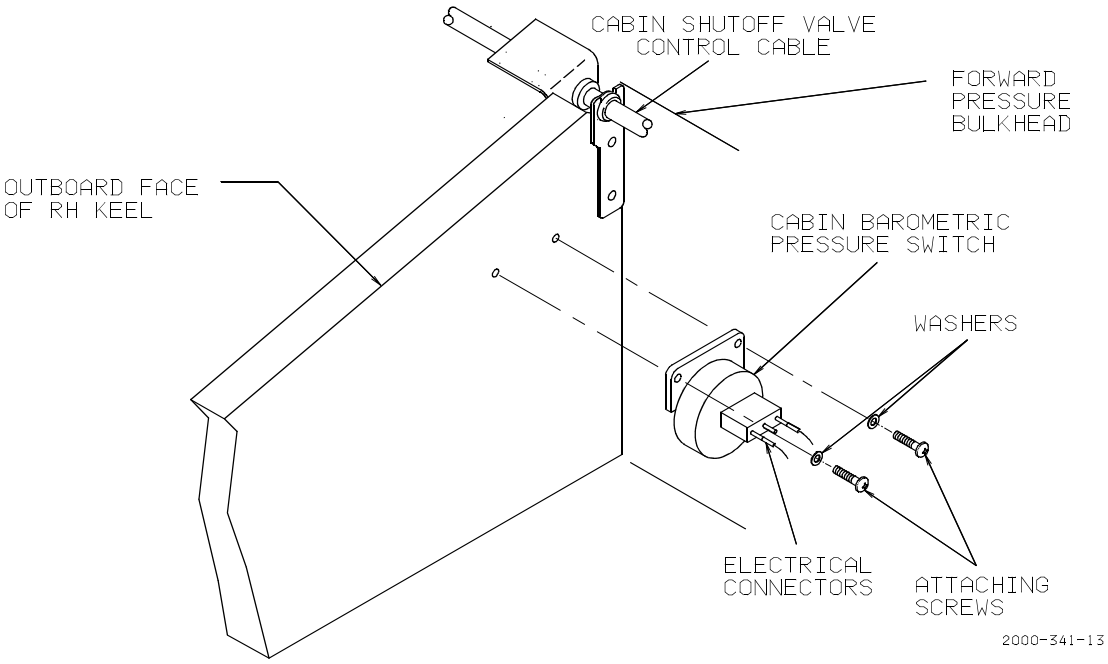
a. Remove all electrical power from the airplane.

b. Remove the copilot's seat and carpet (Ref. Chapter 25-10-00).

c. Loosen the captive screws located around the edge of the copilot's forward cockpit floorboard and remove the floorboard.

d. Disconnect the electrical connectors from the cabin barometric pressure switch (Ref. Figure 11).

e. Remove the screws and washers securing the switch to the outboard side of the RH keel and remove the switch.



Cabin Barometric Pressure Switch
Figure 11

CABIN BAROMETRIC PRESSURE SWITCH INSTALLATION

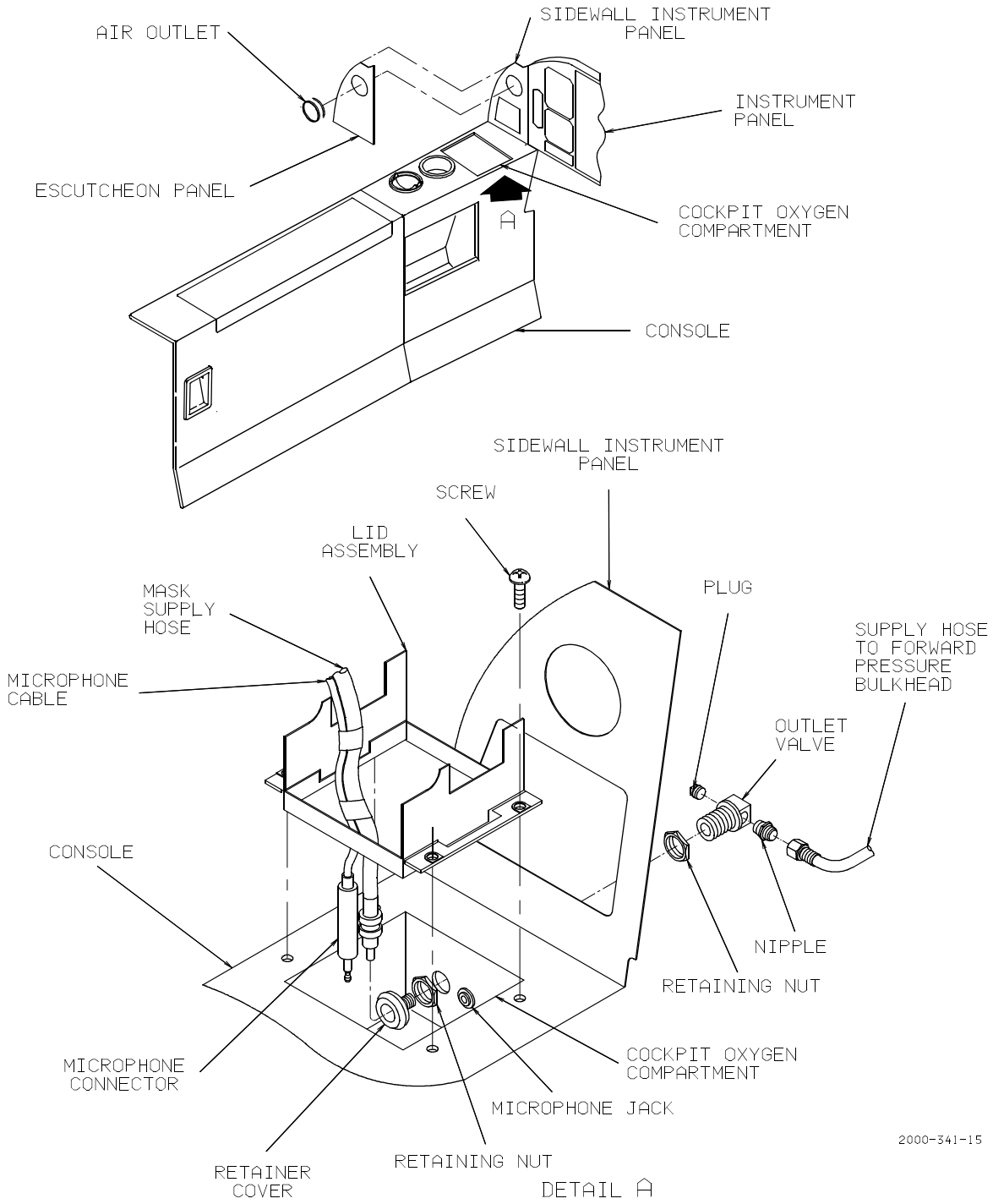
a. Position the cabin barometric pressure switch to the outboard side of the RH keel and secure with the screws and washers (Ref. Figure 11).

b. Connect the electrical connectors to the switch.

c. Install the copilot's forward cockpit floorboard and secure with the captive screws located around the edge of the floorboard.

d. Install the copilot's seat and carpet (Ref. Chapter 25-10-00).

e. Restore electrical power to the airplane.



Cockpit Oxygen Mask, Compartment and Outlet Valve
Figure 12

COCKPIT OXYGEN MASK COMPARTMENT LID ASSEMBLY AND OUTLET VALVE REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose access panel and remove the panel.

e. Push in the OXYGEN SYS READY control knob.

f. Secure the regulator's control lever in the off position.

g. Remove the mask from the cockpit oxygen compartment.

h. Unplug the mask's microphone cable plug from the oxygen mask compartment microphone jack (Ref. Figure 12).

i. Rotate the mask's supply hose bayonet plug and pull it out of the oxygen mask compartment's outlet valve.

j. Unfasten the screws that secure the lid assembly to the console and remove the lid assembly.

k. Remove the console upholstery panel (Ref. Chapter 25-10-00).

l. Unscrew and remove the cockpit air outlet from the escutcheon panel and sidewall panel.

m. Remove the escutcheon panel from the sidewall instrument panel (the escutcheon panel is held in place by Velcro fasteners).

n. Reaching through the opening in the sidewall instrument panel, remove the supply hose from the outlet valve and cap the supply hose.

o. Remove the retainer cover, retaining nuts and the outlet valve from the compartment.

COCKPIT OXYGEN MASK COMPARTMENT LID ASSEMBLY AND OUTLET VALVE INSTALLATION

a. Position the outlet valve in the cockpit oxygen mask compartment and secure with the retaining nuts and retainer cover (Ref. Figure 12).

b. Reaching through the opening in the sidewall instrument panel, install the supply hose on the outlet valve.

c. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

d. Install the escutcheon panel on the sidewall instrument panel (held in place with Velcro fasteners).

e. Install the cockpit air outlet on the escutcheon panel opening by screwing the outlet into the attach nut mounted on the forward side of the sidewall instrument panel.

f. Install the console upholstery panel (Ref. Chapter 25-10-00).

g. Position the oxygen mask compartment lid assembly in the console and secure with the screws.

h. Insert the mask's supply hose bayonet plug into the oxygen mask compartment outlet valve and rotate to secure.

i. Insert the mask's microphone cable plug into the oxygen mask compartment microphone jack.

j. Stow the mask in the compartment.

k. Release the regulator's control lever.

l. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

m. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

n. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

o. Restore electrical power to the airplane.

FORWARD BAGGAGE CABINET OXYGEN MASK COMPARTMENT REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Open the forward baggage cabinet doors.

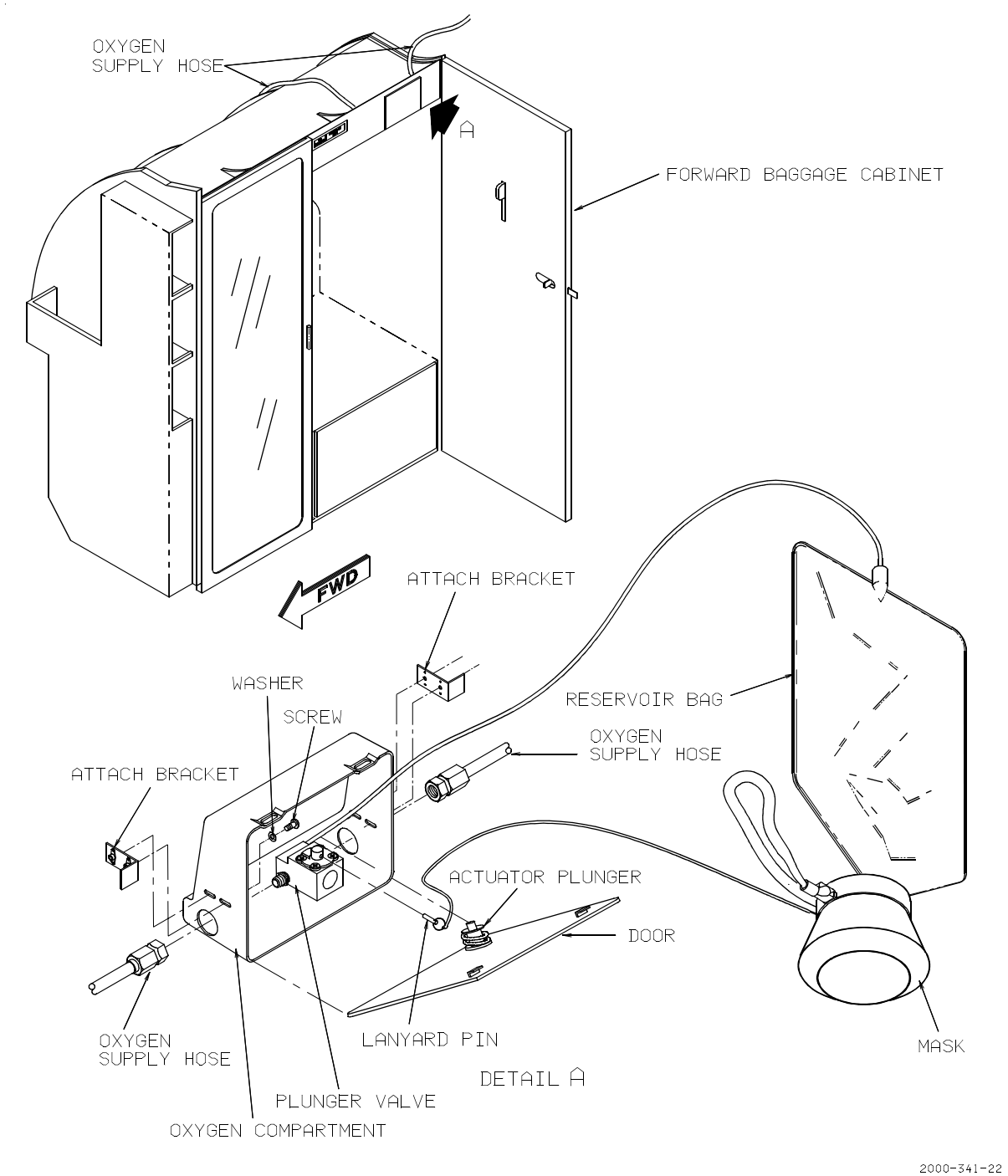
f. Open the oxygen mask compartment door by pulling ON both the OXYGEN SYS READY and the OXYGEN PASS MAN DEPLOY control knobs. This will cause all cabin oxygen mask compartment doors to open (including the cabin headliner oxygen mask compartments).

g. After the doors have opened, turn off the oxygen system by pushing in both control knobs.

h. Secure the regulator's control lever in the off position.

i. Disconnect the oxygen supply hoses from the forward baggage cabinet oxygen mask compartment plunger valve and cap the supply hoses (Ref. Figure 13).

j. Remove the screws and washers, securing the mask compartment to the attach brackets in the baggage cabinet and remove the compartment.



Forward Baggage Cabinet Oxygen Mask Compartment
Figure 13

FORWARD BAGGAGE CABINET OXYGEN MASK COMPARTMENT INSTALLATION

a. Position the oxygen mask compartment in the forward baggage cabinet opening and route the oxygen supply hoses into the compartment (Ref. Figure 13).

b. Secure the compartment to the attach brackets in the cabinet with the screws and washers.

c. Connect the supply hoses to the compartment plunger valve.

d. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

e. Stow all cabin oxygen compartment masks (including the cabin headliner oxygen compartment masks) as instructed under CABIN OXYGEN MASK STOWAGE found in this chapter and install the oxygen mask compartment doors.

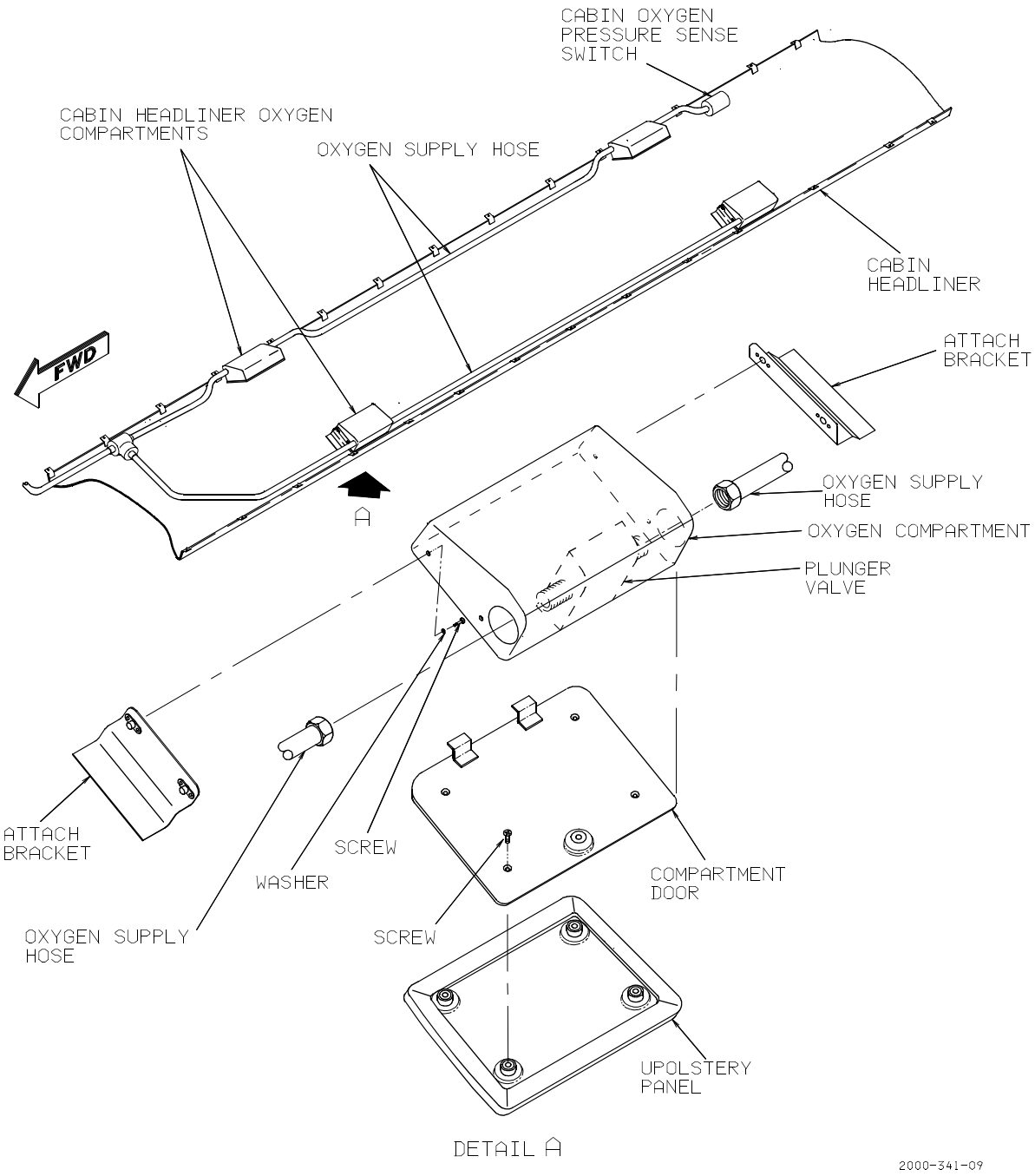
f. Release the regulator's control lever.

g. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

h. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

i. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

j. Restore electrical power to the airplane.



Cabin Headliner Oxygen Mask Compartments
Figure 14

CABIN HEADLINER OXYGEN MASK COMPARTMENT REMOVAL

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Remove all electrical power from the airplane.

b. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

c. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

d. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

e. Open the oxygen mask compartment doors by pulling ON both the OXYGEN SYS READY and the OXYGEN PASS MAN DEPLOY control knobs. This will cause all passenger oxygen mask compartment doors to open (including the forward baggage cabinet oxygen mask compartment).

f. After the doors have opened, turn off the oxygen system by pushing in both control knobs.

g. Secure the regulator's control lever in the off position.

h. Disconnect the oxygen supply hoses from the cabin oxygen mask compartment plunger valve and cap the supply hoses (Ref. Figure 14).

i. Remove the screws and washers, securing the mask compartment to the attach brackets in the headliner and remove the compartment.

j. Remove the screws securing the upholstery panel to the oxygen mask compartment door and remove the panel.

CABIN HEADLINER OXYGEN MASK COMPARTMENT INSTALLATION

a. Position the oxygen mask compartment in the opening of the headliner and route the oxygen supply hoses into the compartment (Ref. Figure 14).

b. Secure the mask compartment to the attach brackets in the headliner with the screws and washers.

c. Connect the supply hoses to the cabin oxygen mask compartment plunger valve.

d. Perform a leak check on the installation as instructed under OXYGEN SYSTEM LEAK DETECTION TEST found in this chapter.

e. Attach the upholstery panel to the compartment door with the screws.

f. Stow all cabin oxygen compartment masks (including the forward baggage oxygen compartment mask) as instructed under CABIN OXYGEN MASK STOWAGE found in this chapter and install the oxygen mask compartment doors.

g. Release the regulator's control lever.

h. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

i. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

j. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

k. Restore electrical power to the airplane.

CABIN OXYGEN MASK STOWAGE

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

a. Clean the mask if necessary. Reference CABIN OXYGEN MASK CLEANING found in this chapter.

b. Disinfect the mask if it has been used. Reference CABIN OXYGEN COMPARTMENT found in this chapter.

c. Fold the reservoir bag in thirds lengthwise as shown in Steps 1 and 2 (Ref. Figure 15).

d. Fold the reservoir bag in thirds again per Step 3.

e. Gather the headstrap and place it in the mask's facecup as shown in Steps 4 and 5.

f. Fold two sides of the mask's facecup (opposite each other) over the headstrap as shown in Steps 6 and 7.

g. Fold the folded reservoir bag over the folded mask per Step 8.

h. Fold the two remaining sides of the mask's facecup over the reservoir bag as shown in Steps 9 and 10.

i. Coil each mask's supply tube and place in the compartment under the folded mask and reservoir bag. Insert the lanyard pin in the compartment's valve and install the compartment's door as shown in Step 11.

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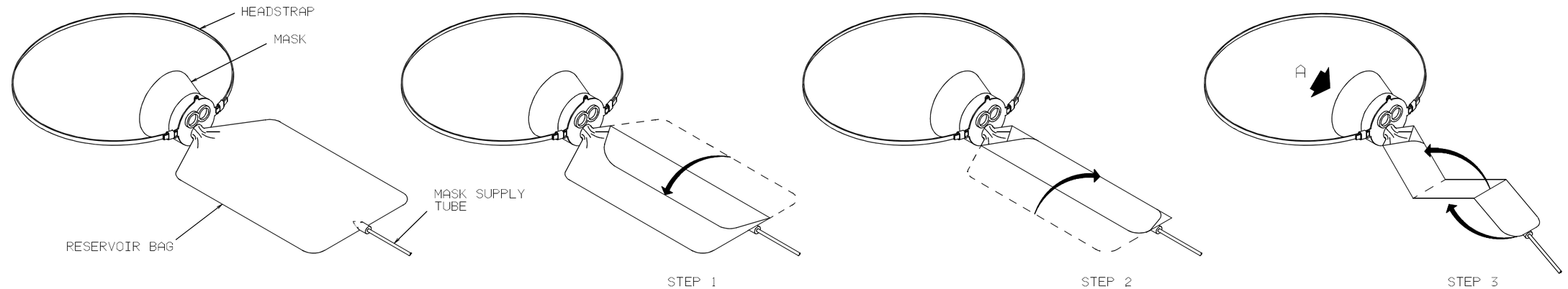

Cabin Oxygen Mask Stowage
Figure 15

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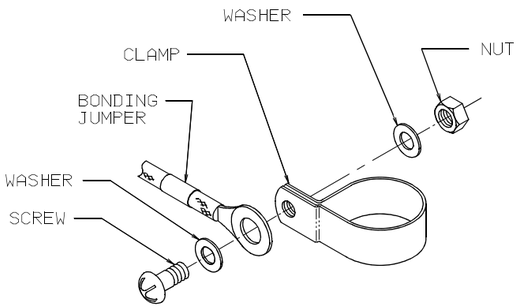

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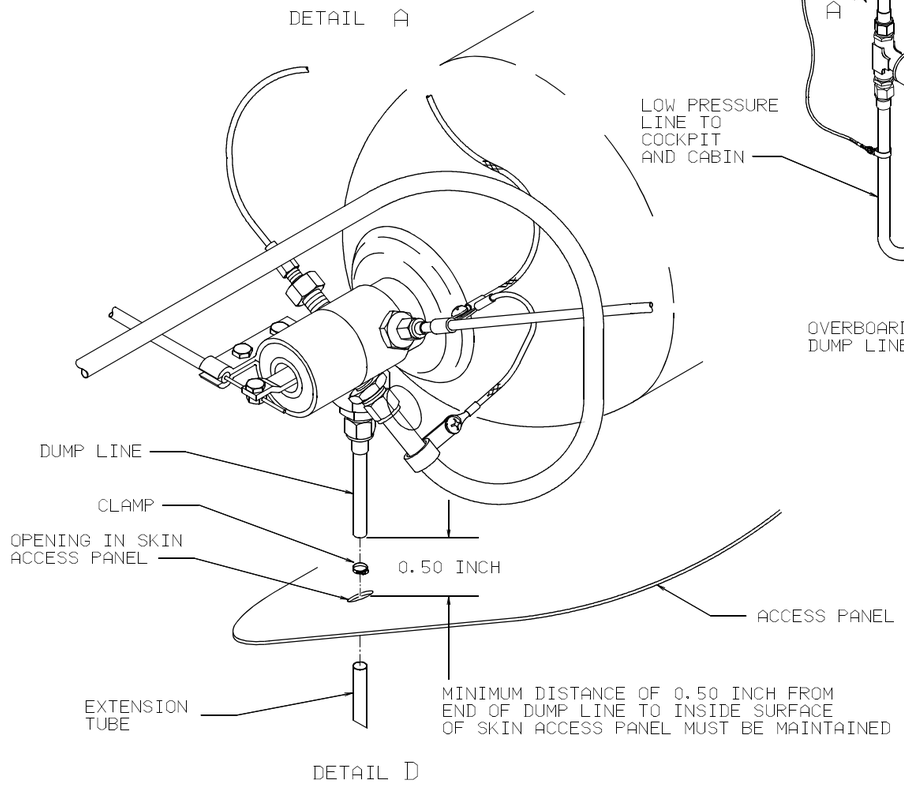
DETAIL A



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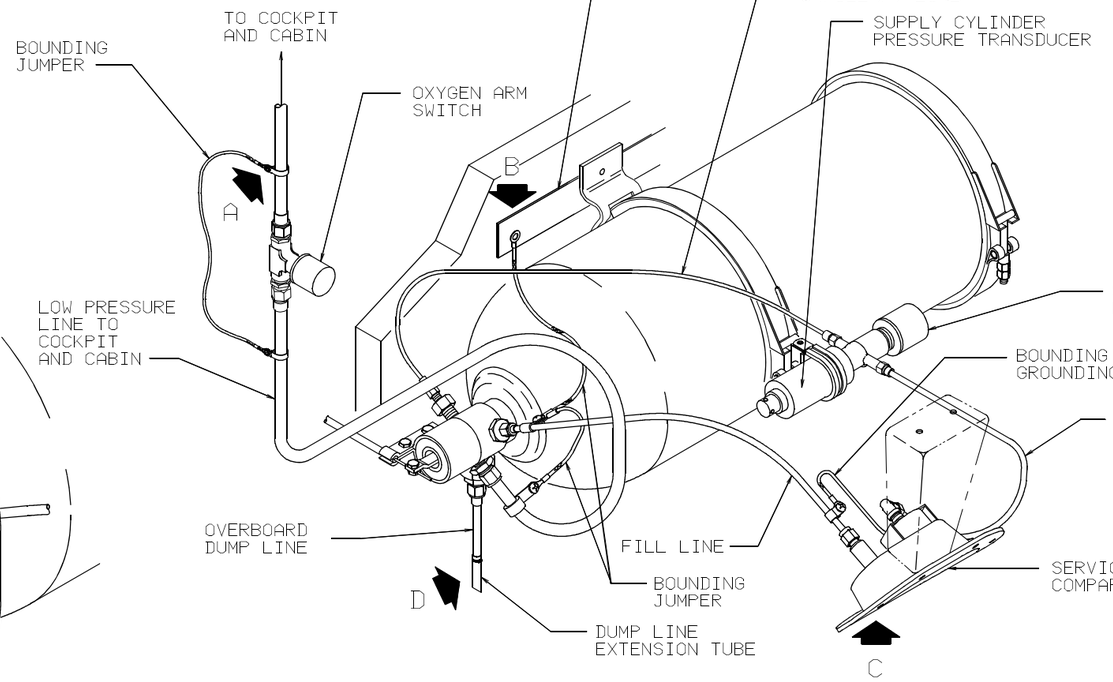
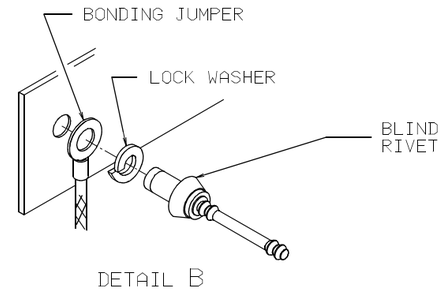
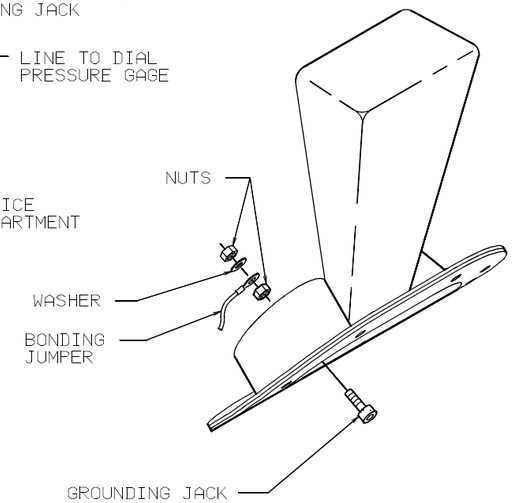




DETAIL B



DETAIL C

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Oxygen System Electrical Bonding and Grounding
Figure 16

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ELECTRICAL BONDING AND GROUNDING OF THE OXYGEN SYSTEM

WARNING: Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing on the oxygen system.

Electrical Bonding of the high pressure side of the oxygen system is very important and must be performed using the following procedures.

a. Clean the surfaces that are to be electrically bonded with a cleaning solvent (5, Chart 4, 35-00-00) or (8, Chart 4, 35-00-00) (Ref. Figure 16).

NOTE: If a new vinyl coated aluminum tube is installed, the vinyl coating at the point of electrical bonding must be removed to a width of 1 1/2 times the width of the clamp.

b. All nonsoluble films or substances are to be removed by either sanding or polishing with fine garnet paper (silicon carbide or aluminum oxide, 320 grit or finer) or fiberglass eraser. Use caution to not remove excessive metal.

CAUTION: *Do not use emery cloth/paper or iron oxide paper/cloth.*

c. Apply corrosion preventive compound (9, Chart 4, 35-00-00) to the cleaned area with a clean Scotch-Brite sponge or equivalent. Shake the compound vigorously before applying. Keep the area wet for 3 to 5 minutes (until a yellow color develops).

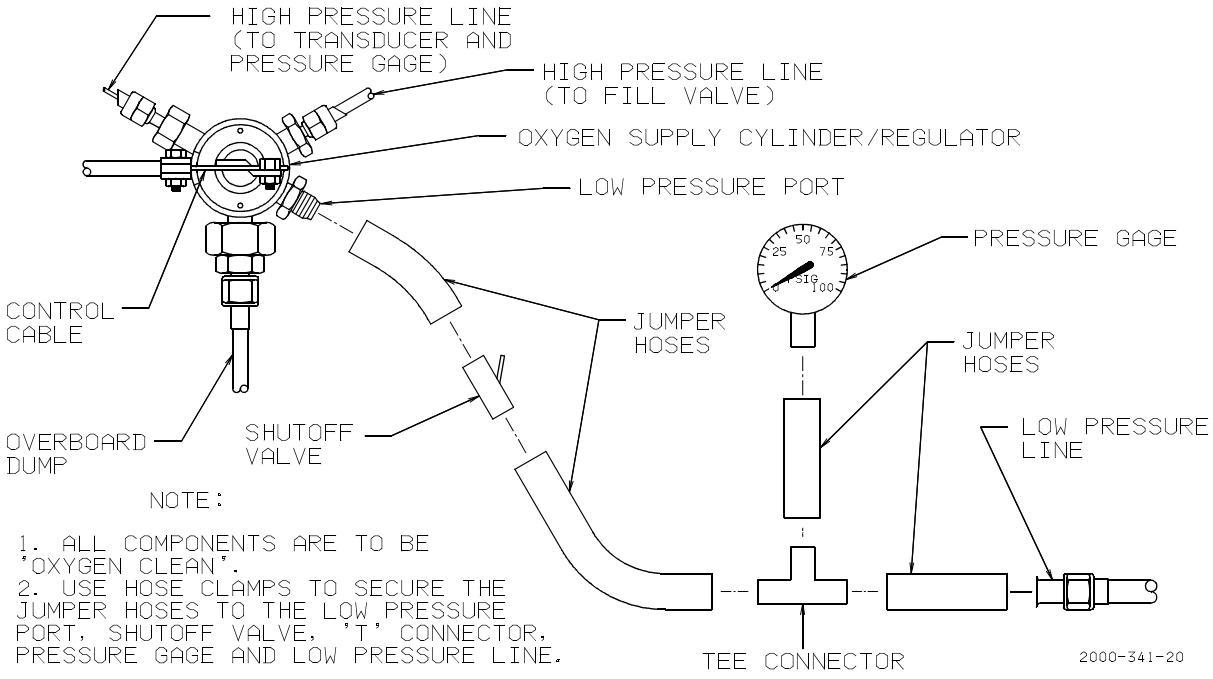
d. Clean the bond area with clean, deionized water to remove chromic oxide and allow to dry. If the bond coating is broken, retouch with corrosion preventive compound (9, Chart 4, 35-00-00). Wipe with care because the formed coating is soft while still wet.

e. Allow the area to thoroughly air dry for no longer than 1 hour maximum before installing the mating parts and hardware.

NOTE: If the maximum drying time limit is exceeded, retouch the surface with corrosion preventive compound (9, Chart 4, 35-00-00).

f. Install the clamp, bonding jumper and attaching hardware. The mating surfaces must be smooth and contoured to the surface area that it is in actual contact with.

g. Seal the entire connection and 1 1/2 times the cleaned area with sealant (10, Chart 4, 35-00-00).



Oxygen System Leak Detection Test Equipment
Figure 17

OXYGEN SYSTEM LEAK DETECTION TEST

a. The following equipment is needed to perform this test:

   1. 0 to 100 psig pressure gage (1 psig increments and oxygen clean)
   2. Tee connector (oxygen clean)
   3. Shutoff valve (oxygen clean)
   4. Four jumper hoses (oxygen clean)
   5. Eight hose clamps
   6. Leak detector fluid (11, Chart 4, 35-00-00)

b. Remove all electrical power from the airplane.

c. Loosen the captive screws located around the edge of the oxygen service compartment access panel and remove the panel.

d. Remove the screws securing the RH lower nose compartment access panel to the service compartment.

e. Loosen the captive screws located around the edge of the RH lower nose compartment access panel and remove the panel.

f. Perform a High Pressure Test and/or a Low Pressure Test as follows:

1. HIGH PRESSURE TEST

a) Pressurize the supply cylinder to between 1,500 and 1,850 psig at 70° F, refer to OXYGEN SYSTEM SERVICING for proper procedures.

b) Maintain the cylinder pressure without loss of pressure for a minimum of 2 hours.

c) If a pressure loss is observed, apply leak detector fluid sparingly to all supply cylinder regulator's fittings and check for leaks. Make any necessary repairs or replacements and retest. Wipe dry immediately after testing.

2. LOW PRESSURE TEST

a) Disconnect the low pressure line from the oxygen supply cylinder's regulator.

b) Connect the pressure gage, shutoff valve and Tee connector to the low pressure port on the regulator and low pressure line using the jumper hoses and hose clamps (Ref. Figure 17).

c) Pull ON both the OXYGEN SYS READY and the OXYGEN PASS MAN DEPLOY control knobs, located on the pilot's inboard subpanel, to charge the system's low pressure hoses and lines. The cabin oxygen compartment masks should deploy.

d) Let the system pressure stabilize for 3 minutes.

e) Push in the OXYGEN SYS READY control knob to turn off the system. Wait 15 minutes and check the system's pressure.

f) If the pressure loss is greater than 5 psi in 15 minutes, check for leaks by applying leak detector fluid sparingly to the fitting being checked or all oxygen system fittings when performing a general check. Make any necessary repairs and retest. Wipe dry immediately after testing.

g) If no leak is indicated, push in both the OXYGEN PASS MAN DEPLOY and the OXYGEN SYS READY control knobs, disconnect the test equipment and reconnect the low pressure line to the oxygen supply cylinder's regulator.

h) Stow all cabin oxygen compartment masks (including the forward baggage oxygen compartment mask) as instructed under CABIN OXYGEN MASK STOWAGE found in this chapter and install the oxygen mask compartment doors.

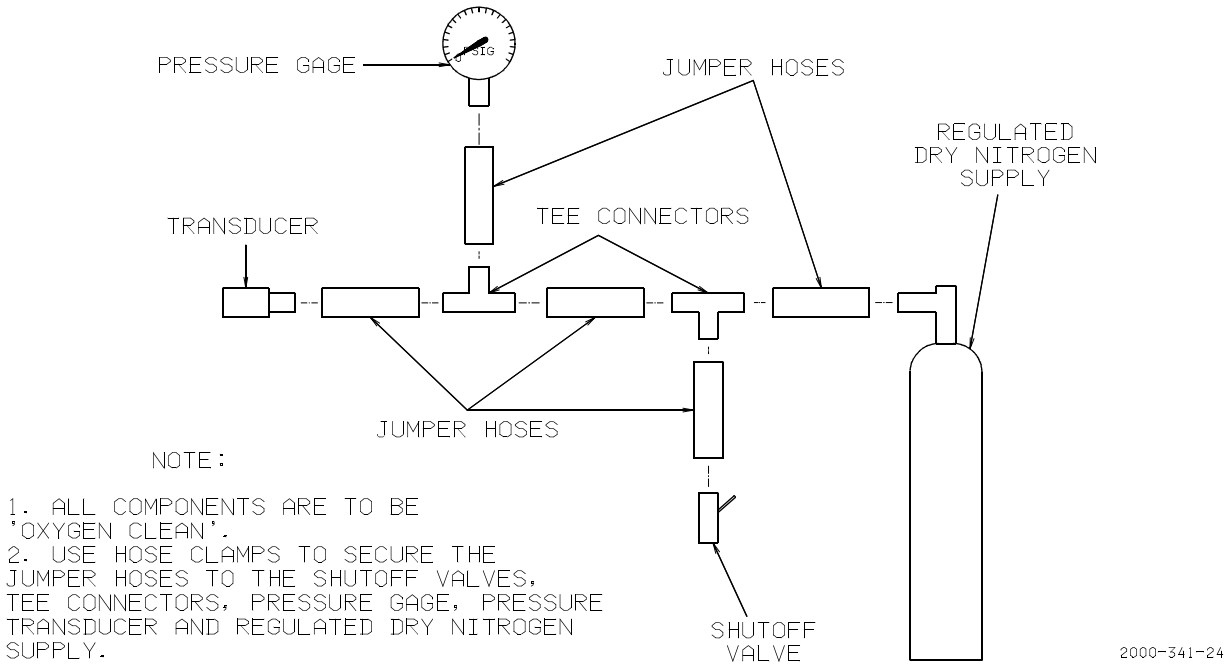
i) Pull ON the OXYGEN SYS READY control knob and apply leak detector fluid to the regulator's low pressure line connection to check for leaks. If no leaks are found push in the OXYGEN SYS READY control knob and wipe off the leak detector fluid.

g. Install the RH lower nose compartment access panel and secure with the captive screws located around the edge of the panel.

h. Install the screws which attach the RH lower nose compartment access panel to the service compartment.

i. Install the oxygen service compartment access panel and secure with the captive screws located around the edge of the panel.

j. Restore electrical power to the airplane.



Pressure Transducer Test Equipment
Figure 18

SUPPLY CYLINDER PRESSURE TRANSDUCER FUNCTIONAL TEST

a. The following equipment is needed to perform this test:

1. Regulated dry nitrogen supply
2. 28 vdc power supply
3. Digital voltmeter
4. Two Tee connectors (oxygen clean)
5. Five jumper hoses (oxygen clean)
6. Ten hose clamps
7. Shutoff valve (oxygen clean)
8. 0 to 2,000 psig pressure gage (50 psig increments and oxygen clean)

b. Remove all electrical power from the airplane.

c. Remove the supply cylinder pressure transducer from the oxygen system as instructed under SUPPLY CYLINDER PRESSURE TRANSDUCER REMOVAL found in this chapter.

d. Connect the transducer to the pressure gage, shutoff valve, Tee connectors and regulated dry nitrogen supply using the jumper hoses and hose clamps (Ref. Figure 18). Close the shutoff valve.

e. Connect the 28 vdc power supply to the transducer as follows: the positive lead to the transducer's pin "A" and the negative lead to pin "B".

f. Connect the digital voltmeter to the transducer as follows: the positive probe to pin "D" and the negative probe to pin "B".

g. Apply power from the 28 vdc power supply to the transducer.

h. Apply pressure to the transducer to obtain each input pressure value listed in Chart 11.

i. If the transducer's output voltage falls outside the corresponding voltage range for any of the pressure values, listed in Chart 11, it must be replaced.

j. Turn off the regulated dry nitrogen supply and slowly bleed off the pressure in the test equipment using the shutoff valve.

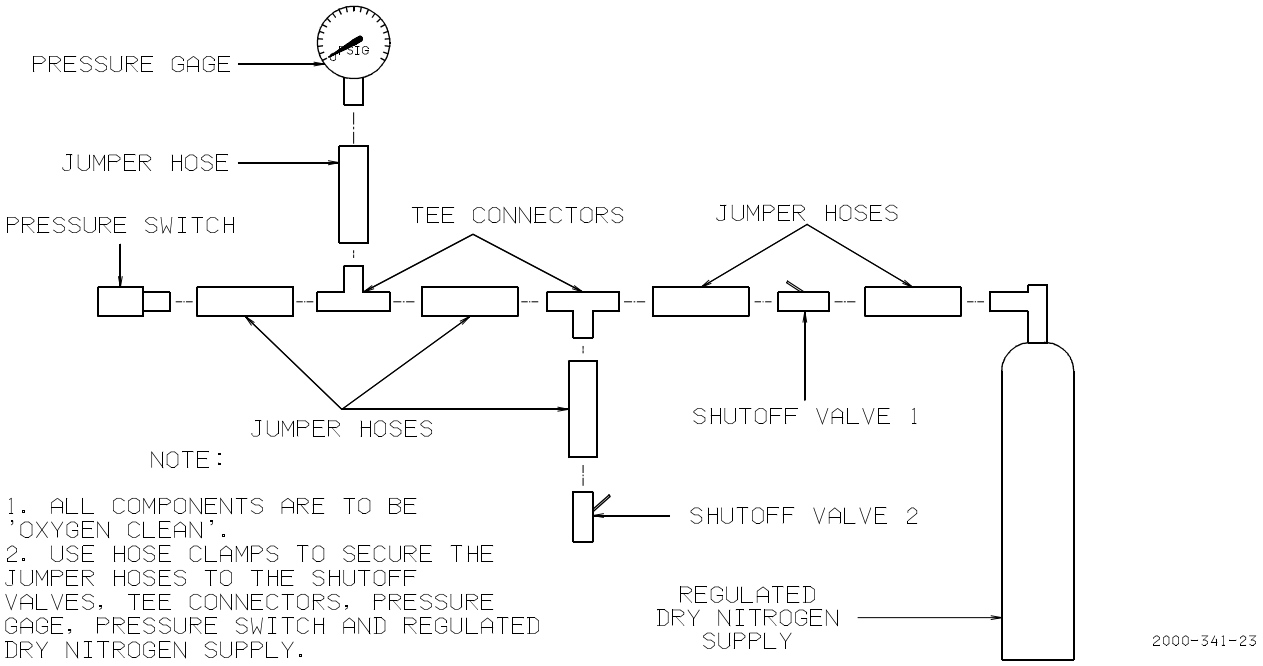
k. Disconnect the power supply and voltmeter from the transducer.

l. After the test equipment pressure has been bled off, disconnect the transducer from the test equipment and install it in the oxygen system as instructed under SUPPLY CYLINDER PRESSURE TRANSDUCER INSTALLATION found in this chapter.

m. Restore electrical power to the airplane.

Chart 11
Supply Cylinder Pressure Transducer Functional Test Requirements

Input Pressure (psig)	Output Voltage (vdc)
0	0.171 to 0.329
150	0.546 to 0.704
300	0.921 to 1.079
550	1.546 to 1.704
750	2.046 to 2.204
1,200	3.171 to 3.329
1,450	3.796 to 3.954
1,750	4.546 to 4.704
1,850	4.796 to 4.954
2,000	5.171 to 5.329



Pressure Switch Test Equipment
Figure 19

SUPPLY CYLINDER LOW PRESSURE SWITCH FUNCTIONAL TEST

a. The following equipment is needed to perform this test:

1. Regulated dry nitrogen supply
2. Continuity tester
3. Two Tee connectors (oxygen clean)
4. Two shutoff valves (oxygen clean)
5. Six jumper hoses (oxygen clean)
6. Twelve hose clamps
7. 0 to 2,000 psig pressure gage (50 psig increments and oxygen clean)

b. Remove all electrical power from the airplane.

c. Disconnect and remove the supply cylinder low pressure switch as instructed under SUPPLY CYLINDER LOW PRESSURE SWITCH REMOVAL found in this chapter.

d. Connect the pressure switch to the pressure gage, shutoff valves, Tee connectors and regulated dry nitrogen supply using the jumper hoses and hose clamps (Ref. Figure 19). Close both shutoff valves.

e. Connect the leads of the continuity tester to pins "1" and "3" of the pressure switch.

f. Open shutoff valve 1 and apply pressure to the pressure switch. Verify that the pressure switch opens (no continuity) at an increasing pressure of 550 ± 30 psig. Close shutoff valve 1 when the pressure switch opens.

g. After waiting 1 minute, decrease the pressure on the pressure switch by opening shutoff valve 2 and verifying that the pressure switch closes (continuity) at a minimum decreasing pressure of 500 psig.

h. The pressure switch must be replaced if it does not meet these specifications.

i. Disconnect the test equipment and install the pressure switch in the oxygen system as instructed under SUPPLY CYLINDER LOW PRESSURE SWITCH INSTALLATION found in this chapter.

j. Restore electrical power to the airplane.

SYSTEM ARMED PRESSURE SWITCH FUNCTIONAL TEST

a. The following equipment is needed to perform this test:

1. Regulated dry nitrogen supply
2. Continuity Tester
3. Two Tee connectors (oxygen clean)
4. Two shutoff values (oxygen clean)
5. Six jumper hoses (oxygen clean)
6. Twelve hose clamps
7. 0 to 60 psig pressure gage (1 psig increments and oxygen clean)

b. Remove all electrical power from the airplane.

c. Disconnect and remove the system armed pressure switch from the oxygen system as instructed under SYSTEM ARMED PRESSURE SWITCH REMOVAL found in this chapter.

d. Connect the pressure switch to the pressure gage, shutoff valves, Tee connectors and regulated dry nitrogen supply using the jumper hoses and hose clamps (Ref. Figure 19). Close both shutoff valves.

e. Connect the leads of the continuity tester to pins "1" and "3" of the pressure switch.

f. Open shutoff valve 1 and apply pressure to the pressure switch. Verify that the pressure switch closes (continuity) at an increasing pressure of 50 ± 3 psig. Close shutoff valve 1 when the pressure switch closes.

g. After waiting 1 minute, decrease the pressure on the pressure switch by opening shutoff valve 2 and verifying that the pressure switch opens (no continuity) at a minimum decreasing pressure of 40 psig minimum.

h. The pressure switch must be replaced if it does not meet these specifications.

i. Disconnect the test equipment and install the pressure switch in the oxygen system as instructed under SYSTEM ARMED PRESSURE SWITCH INSTALLATION found in this chapter.

j. Restore electrical power to the airplane.

CABIN OXYGEN PRESSURE SENSE SWITCH FUNCTIONAL TEST

a. The following equipment is needed to perform this test:

1. Regulated dry nitrogen supply
2. Continuity tester
3. Two Tee connectors (oxygen clean)
4. Two shutoff valves (oxygen clean)
5. Six jumper hoses (oxygen clean)
6. Twelve hose clamps
7. 0 to 60 psig pressure gage (1 psig increments and oxygen clean)

b. Remove all electrical power from the airplane.

c. Disconnect and remove the cabin oxygen pressure sense switch as instructed under CABIN OXYGEN PRESSURE SENSE SWITCH REMOVAL found in this chapter.

d. Connect the pressure switch to the pressure gage, shutoff valves, Tee connectors and regulated dry nitrogen supply using the hoses and hose clamps (Ref. Figure 19). Close the shutoff valves.

e. Connect the leads of the continuity tester to pins "1" and "3" of the pressure switch.

f. Open shutoff valve 1 and apply pressure to the pressure switch. Verify that the pressure switch closes (continuity) at an increasing pressure of 50 ± 3 psig. Close shutoff valve 1 when the pressure switch closes.

g. After waiting 1 minute, decrease the pressure on the pressure switch by opening shutoff valve 2 and verifying that the pressure switch opens (no continuity) at a minimum decreasing pressure of 40 psig.

h. The pressure switch must be replaced if it does not meet these specifications.

i. Disconnect the test equipment and install the pressure switch in the oxygen system as instructed under CABIN OXYGEN PRESSURE SENSE SWITCH INSTALLATION found in this chapter.

j. Restore electrical power to the airplane.

DIGITAL PRESSURE INDICATOR FUNCTIONAL TEST

a. The following equipment is needed to perform this test:

1. 28 vdc power supply
2. 0 to 5.25 vdc variable power supply
3. Digital voltmeter

b. Remove all electrical power from the airplane.

c. Remove the digital pressure indicator from the copilot’s subpanel (Ref. Chapter 31-10-00).

d. Connect the 28 vdc power supply to the indicator as follows: the positive lead to the indicator’s pin "1" and the negative lead to pin "6".

e. Connect the 0 to 5.25 vdc variable power supply and digital voltmeter to indicator as follows: the positive lead and probe to the indicator's pin "3" and the negative lead and probe to pin "6".

f. Apply voltage from the 28 volt power supply to the indicator.

g. Using the variable power supply, adjust the amount of voltage applied to the indicator until each display reading, listed in Chart 12, is obtained.

h. Compare the amount of voltage required to obtain each display reading with the display reading's corresponding voltage range listed in Chart 12. If it does not fall within this range, the indicator must be replaced.

i. Disconnect the power supplies and voltmeter from the indicator and install the indicator in the copilot's inboard subpanel (Ref. Chapter 31-10-00).

j. Restore electrical power to the airplane.

Chart 12
Digital Pressure Indicator Functional Test Requirements

Display Reading (psig)	Voltage Range (vdc)
0	0.200 to 0.300
110	0.475 to 0.575
150	0.575 to 0.675
220	0.750 to 0.850
360	1.100 to 1.200
530	1.525 to 1.625
670	1.875 to 1.975
740	2.050 to 2.150
880	2.400 to 2.500
1,090	2.925 to 3.025
1,250	3.325 to 3.425
1,400	3.700 to 3.800
1,500	3.950 to 4.050
1,900	4.950 to 5.050
1,999	5.200 to 5.300

BAROMETRIC PRESSURE SWITCH CHECK

NOTE: If this procedure is being performed due to a Time Limited Inspection requirement of Chapter 5-10-00, the METHOD 1 procedure MUST be performed. If the procedure is being performed solely to check the operation of the barometric switch, then either the METHOD 1 or the METHOD 2 may be performed.

If this procedure is being performed due to a Time Limited Inspection requirement of Chapter 5-10-00, perform the CABIN ALTITUDE WARNING PRESSURE SWITCH CHECK procedure in Chapter 21-30-00 in conjunction with this procedure.

METHOD 1

WARNING: Comply with standard FAA regulations for oxygen usage when performing this test.

NOTE: If the pressure switch is not connected to the airplane wiring, connect an ohmmeter across the contacts (COM and NO) of the switch. During this procedure the text implying that the annunciator illuminates is to be taken as the equivalent of the ohmmeter indicating continuity and the text implying that annunciator extinguishes is to be taken as the equivalent of the ohmmeter indicating an open circuit.

It is not necessary to exceed an altitude of 13,000 feet while performing this check.

a. Fly the **unpressurized** airplane to the altitude at which the oxygen masks deploy.

b. As airplane altitude increases, note the altitude on a hand held certified altimeter set to 29.92 inch Hg. Verify that the following events occur at an altitude of 12,000 to 12,500 feet.

1. PASS OXYGEN ON annunciator illuminates.

2. Cabin oxygen masks deploy.

c. If this procedure is being performed due to a Time Limited Inspection requirement of Chapter 5-10-00, pull the lanyard pin out and verify oxygen flow, then replace the lanyard pin of every oxygen mask.

d. The pilot shall then descend to an altitude below 10,500 feet.

e. As the airplane descends, note the cabin altitude at which the annunciator extinguishes. The annunciator shall extinguish at 10,500 feet or higher.

f. If the requirements of Steps b. and e. are met, the barometric switch is operating properly and the system is operating as designed.

CAUTION: *The mask container door could be installed 180° out of the correct position. If this happens, the plunger cannot push the door open when activated. Ensure that the door is installed so that the plunger will contact the block on the door when activated.*

g. Perform the CABIN OXYGEN COMPARTMENT AND MASK INSPECTION procedure in this chapter.

h. Repack the masks and latch the mask container doors.

i. If required, perform the OXYGEN SYSTEM SERVICING procedure (Ref. Chapter 12-10-00).

METHOD 2

a. Remove the barometric pressure switch from the airplane, if required.

b. Place the barometric pressure switch inside a vacuum chamber with an ohmmeter connected to the contacts (COM and NO) of the switch. If the vacuum chamber does not have a scale for determining degree of vacuum achieved, a certified altimeter must be used. Set the altimeter to 29.92 inch Hg. Place the altimeter in the vacuum chamber in a position so that it can be clearly read.

c. Increase the vacuum in the chamber and check for continuity at the increasing altitude activation point (12,500 +000/-500 feet).

d. After continuity is acquired, slowly decrease the vacuum in the chamber and check for termination of continuity at the decreasing altitude activation point (10,500 feet minimum). Allow the vacuum chamber to return to atmospheric pressure.

e. The switch must be replaced if it does not meet these specifications.

f. Remove the pressure switch from the vacuum chamber.

g. Install the barometric pressure switch in the airplane, if required.

CABIN SHUTOFF VALVE FUNCTIONAL TEST

a. A 28 vdc power supply is needed to perform this test.

b. Remove all electrical power from the airplane.

c. Make sure the oxygen supply cylinder has a minimum of 100 psig. If not, fill the cylinder as instructed under OXYGEN SYSTEM SERVICING found in this chapter.

d. Remove the copilot's seat and carpet (Ref. Chapter 25-10-00).

e. Loosen the captive screws located around the edge of the copilot's forward cockpit floorboard and remove the floorboard.

f. Disconnect the electrical wiring from the cabin shutoff valve.

g. Connect the leads from the 28 vdc power supply to the shutoff valve's electrical wiring.

h. Stow all cabin oxygen masks (including the forward baggage cabinet oxygen compartment mask) as instructed under CABIN OXYGEN MASK STOWAGE found in this chapter.

i. PULL ON the OXYGEN SYS READY control knob.

j. Check the shutoff valve's manual operation by pulling ON the OXYGEN PASS MAN DEPLOY control knob and observing whether the cabin oxygen mask compartment doors open and the masks drop.

NOTE: All of the cabin oxygen mask compartment doors must open and the masks drop.

k. Check each of the masks for proper oxygen flow as follows:

   1. Pull the lanyard pin from the oxygen inlet valve to initiate oxygen flow.

   2. Verify that oxygen is flowing from the mask.

   3. Replace the lanyard pin in the oxygen inlet valve.

l. Push in the OXYGEN PASS MAN DEPLOY control knob.

m. Perform the CABIN OXYGEN COMPARTMENT AND MASK INSPECTION procedure.

n. Stow all cabin oxygen masks (including the forward baggage cabinet oxygen compartment mask) as instructed under CABIN OXYGEN MASK STOWAGE found in this chapter.

o. Check the shutoff valve's automatic (electrical) operation by applying 28 vdc to the valve and observing whether the cabin oxygen mask compartment doors open and the masks drop.

p. If the cabin oxygen mask compartment doors fail to open during either of these checks the shutoff valve must be replaced as instructed under CABIN SHUTOFF VALVE REMOVAL and INSTALLATION.

q. Disconnect the 28 vdc power supply from the shutoff valve's electrical wiring and connect the airplane's wiring to the shutoff valve's.

r. Push in the OXYGEN SYS READY control knob.

s. Stow all cabin oxygen masks (including the forward baggage cabinet oxygen compartment mask) as instructed under CABIN OXYGEN MASK STOWAGE found in this chapter.

t. Install the copilot's forward cockpit floorboard and secure with the captive screws located around the edge.

u. Install the copilot's seat and carpet (Ref. Chapter 25-10-00).

v. Restore electrical power to the airplane.

CABIN OXYGEN COMPARTMENT AND MASK INSPECTION

Inspect all cabin oxygen compartments and masks (including the forward baggage cabinet oxygen compartment and mask) for the occurrence of any of the following conditions.

a. The oxygen mask sticks to the compartment or to itself.

b. The lanyard pin is not installed properly or requires excessive force to remove from the valve actuator.

c. The mask, reservoir bag and mask supply tube show signs of tears, cracking, splitting, hardening or deterioration.

d. The mask's supply tube is kinked or is not properly connected to the orifice fitting on the compartment's valve.

e. Contamination of the mask or it's compartment.

f. The donning instruction label located on the inside surface of each passenger oxygen mask compartment door is not legible or present

OXYGEN MASK CLEANING

Should the oxygen masks need cleaning, wipe the surface to be cleaned with a clean, soft, lint-free cloth that has been moistened with a mild detergent and warm water solution (not to exceed 110× F; 43× C). Rinse thoroughly with clean water and allow to completely air dry.

NOTE: Isopropyl alcohol (2, Chart 4, 35-00-00) can also be used for cleaning as well as for disinfecting. Refer to OXYGEN MASK AND COMPARTMENT DISINFECTING, this section.

OXYGEN MASK AND COMPARTMENT DISINFECTING

The masks and compartments are to be disinfected after each use, anytime the mask is cleaned and/or overhauled, and at least once a year.

a. Prepare an aqueous disinfecting solution by mixing three (3) parts disinfectant (12, Chart 4, 35-00-00) with one (1) part water.

b. Apply the disinfectant solution by wiping the mask with a lint-free cloth soaked in the solution. Allow the mask to completely dry.

c. Lightly dust the outside of the mask's facepiece with powder (13, Chart 4, 35-00-00) or talcum powder.

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CHAPTER 36 - PNEUMATIC

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The pneumatic system uses P_3 bleed air from each engine compressor. One engine can supply sufficient bleed air to operate all the systems requiring presssurized air. Engine P_3 air is routed from the engines through the firewall bleed air shutoff valves which limit the bleed air pressure to 30 to 36 psi. This pressure is supplied to the bleed air flow control valve and the pneumatic system pressure regulator valve. The pneumatic system pressure regulator valve limits the bleed air pressure to 22 to 25 psi to all downstream systems. The pneumatic system supplies regulated engine bleed air to the left and right aft wing surface deice systems, the left and right forward wing surface deice systems, the pneumatic pressure display on the pilot's inboard subpanel, the environmental systems, the cabin door seal and the jet pump vacuum ejector of the pressurization control system. Unregulated pneumatic pressure is supplied to the fuel purge system and the landing gear power pack pressure regulator valve.

A fitting on each bleed air supply tube downstream of the engine P_3 supply point furnishes pneumatic pressure to the fuel purge system. Another fitting on the left bleed air system supplies pneumatic pressure to the pressure regulator valve of the landing gear power pack. The landing gear power pack pressure regulator valve is mounted on the engine air inlet duct in the forward part of the nacelle and regulates the pneumatic pressure to 18 ± 1 psi. Refer to Chapter 36-10-00 for information on the pneumatic distribution system.

The bleed air flow control valve controls the bleed air flow rate to the environmental systems. Refer to Chapter 21-40-00 for detailed information on the bleed air flow control valve and the heating system. Bleed air used for heating is also used for cabin pressurization. Refer to Chapter 21-30-00 for detailed information on the pressurization control system.

RECOMMENDED MATERIALS (Effectivity: All)

The recommended materials listed in Chart 1 as meeting federal, military or vendor specifications are provided for reference only and are not specifically prescribed by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification may be used even though not included in the chart. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
RECOMMENDED MATERIALS (Effectivity: All)

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
1. Leak Detector	Mil-L-25567	Leak detecting fluid	Obtain locally
2. Adhesive/Sealer		RTV-106	General Electric Schenectady, NY

CHART 1
RECOMMENDED MATERIALS (Effectivity: All) *(Continued)*

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
3. Insulation		BS272-HHJ-C	Beech Aircraft Corp., Wichita, KS
4. Insulation		BS272-HH	Beech Aircraft Corp., Wichita, KS
5. Aluminum Tape		10100B-1 1A 812	Mactac, Stow, OH

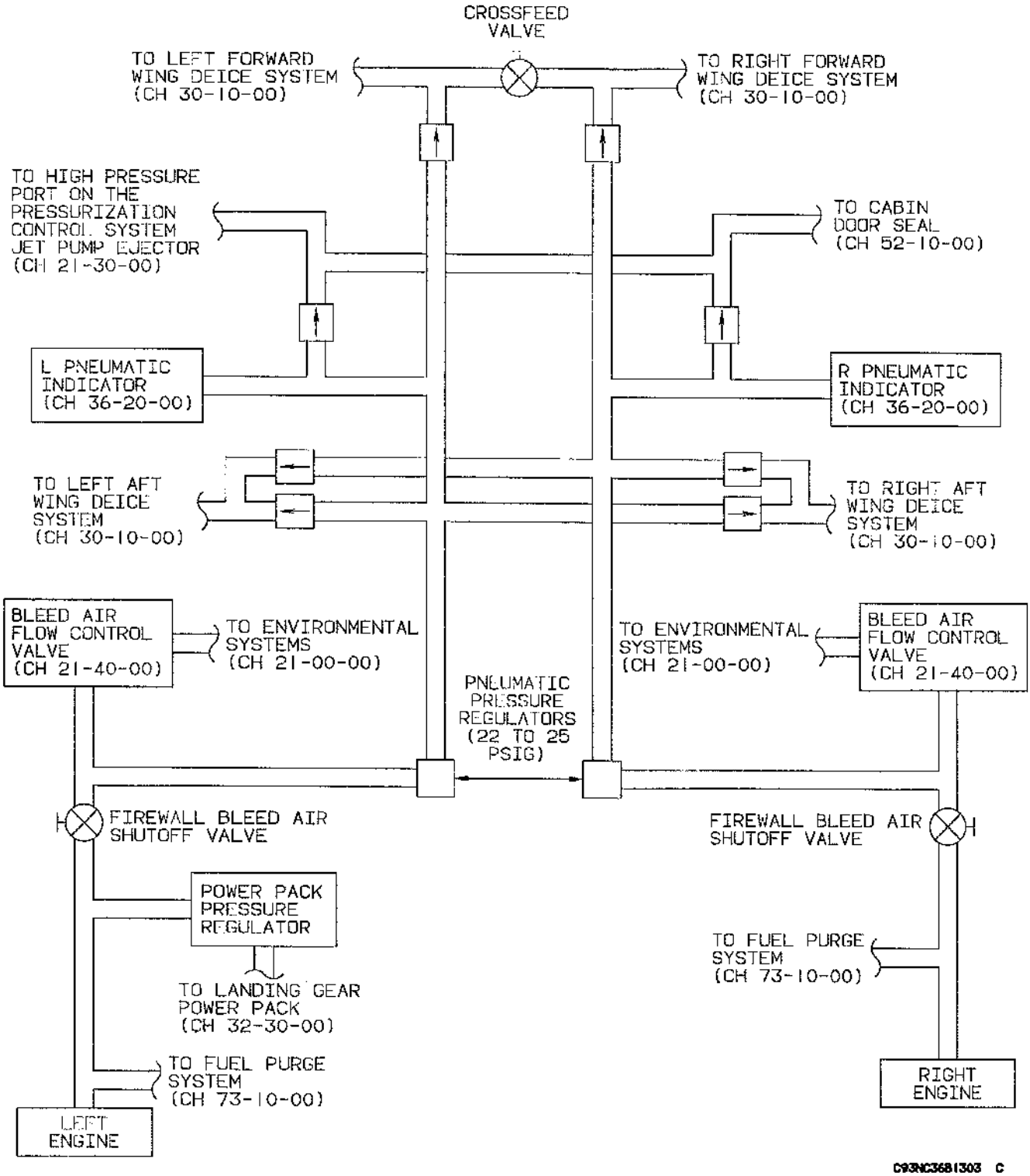
DISTRIBUTION - DESCRIPTION AND OPERATION (Effectivity: All) (Figures 1 and 2)

WARNING

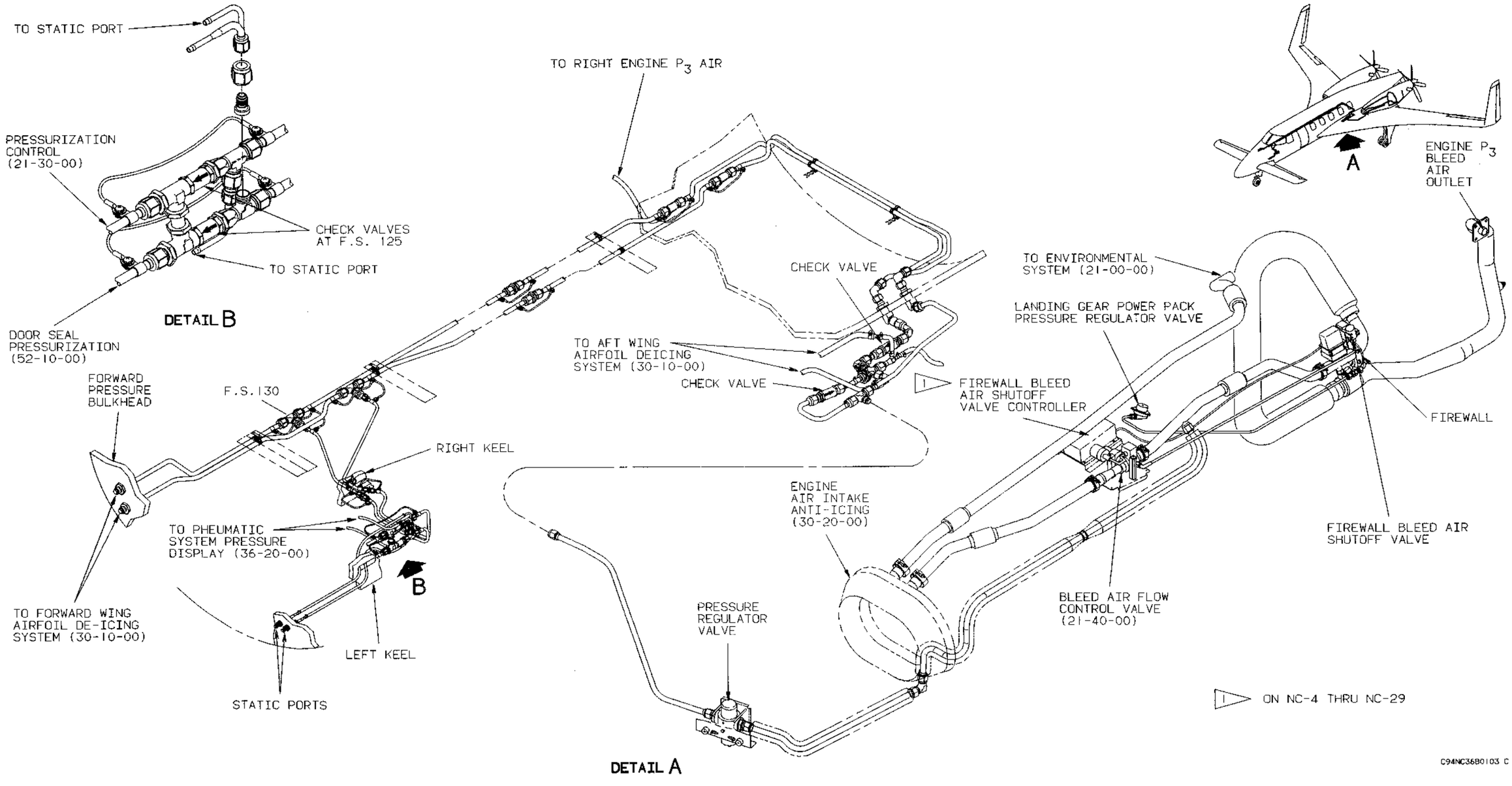
Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

Engine compressor bleed air (P_3) is routed from the engines to the firewall bleed air shutoff (regulator) valves (right-E108, left-E109) which regulates the downstream pneumatic pressure to 33 ± 3 psig. The firewall bleed air shutoff valves are installed just forward of each fire wall. On airplanes NC-4 thru NC-29, each firewall bleed air shutoff valve has a controller mounted adjacent to the flow control valve in the forward area of each nacelle. On airplanes NC-30 and after, the firewall bleed air shutoff valve (Figure 4) is used in place of the shutoff valve/controller unit installed on airplanes NC-4 thru NC-29 (Figure 2). The NC-30 and after firewall bleed air shutoff valve and the NC-4 thru NC-29 firewall bleed air shutoff valve controllers are independently controlled from the crew compartment by the rotary switch (A175) placarded BLEED AIR VALVES on the copilot's inboard subpanel. Control selections on the BLEED AIR VALVES rotary switch are: HIGH FLOW, R ENG, BOTH, L ENG, OFF, and EMER. Power is supplied to the firewall bleed air shutoff valves through the circuit breakers placarded BLEED AIR LEFT (CB152) and BLEED AIR RIGHT (CB114) on the left circuit breaker panel in the pilot's console. A small tube, which is an integral part of the firewall bleed air shutoff valve, connects the bleed air tube portion of the valve assembly to a pressure switch. When power is applied to the firewall bleed air shutoff valve, the pressure switch allows the valve to become operable when the air pressure sensed through the bleed air tube is 6 to 9 psig or above. By not allowing the firewall bleed air shutoff valve to open below this pressure, the pressure switch enables the engine to speed up more freely during starting. A pressure transducer is mounted on the firewall bleed air shutoff valves used on airplanes NC-4 thru NC-29. The pressure transducer senses the pressure of the bleed air flowing to the bleed air flow control valve. The firewall bleed air shutoff valves on airplanes NC-30 and after do not have the pressure transducer and its attendant sense line. When power is removed, the firewall bleed air shutoff valve on airplanes NC-4 thru NC-29 remains in the last selected position. The firewall bleed air shutoff valve on airplanes NC-30 and after closes when power is removed. When the firewall shutoff valves are closed, the Engine Instrument Crew Alerting System (EICAS) yellow L BLEED OFF and R BLEED OFF caution messages are displayed on the EICAS display.

A bleed air supply tube downstream of each firewall bleed air shutoff valve routes the bleed air to the forward end of the respective nacelle and through the aft wing to a pressure regulator valve in each main landing gear wheel well. The pressure regulator valves limit bleed air pressure to 22 to 25 psig. As a safety feature should the regulator fail, the valve is equipped with a pressure-relief feature that will actuate at a pressure of 30 ± 1 psig. Pneumatic regulator valve failure would be noted by the flight crew due to the higher pressure displayed on the digital pneumatic display. The bleed air lines are routed from the pressure regulator valves to two tee-fittings (one per side) located between the fuselage and the wing root rib at F.S. 325. One line from each tee is routed to the opposite side to provide a redundant pneumatic pressure source for the aft wing surface deice system. There are four check valves (two per side) installed between the tee-fittings and the surface deicer plumbing for the aft wing at F.S. 320 and F.S. 330. Two tee-fittings between F.S. 335 and F.S. 345 on the right side of the fuselage route the regulated pneumatic pressure to two lines running the length of the fuselage. These lines are routed routed under the floorboards on the right side of the fuselage to the forward pressure bulkhead. The lines penetrate the forward pressure bulkhead to provide pneumatic pressure to the forward wing surface deice system. There are also tee-fittings at F.S. 130 and F.S. 140 that split pneumatic pressure to two check valves on the inboard



Pneumatic Distribution System Schematic
(Effectivity: All)
Figure 1



Pneumatic Distribution System
(Effectivity: All)
Figure 2

side of the left keel at F.S. 125. A tee-fitting is installed in the upstream end of each check valve. The tees provide pneumatic pressure to the respective side of the pneumatic pressure display (M100) on the pilot's inboard subpanel. The display shows the pressure coming from each pressure regulator valve. To provide a redundant pressure source to the pressurization control system and the cabin door seal, tee-fittings are installed in the downstream end of the two check valves located at F.S. 125. The legs of these tee-fittings are connected with a union, thereby permitting a cross flow of pressure. The upper check valve and tee supplies pneumatic pressure to the jet pump vacuum ejector of the pressurization control system while the lower check valve and tee supplies pneumatic pressure to the cabin door seal. Refer to Chapter 21-30-00 for detailed information on the cabin pressurization control system. Refer to Chapter 36-20-00 for information on pneumatic pressure indicating. Refer to Chapter 52-10-00 for information on the cabin door seal pressurization.

DISTRIBUTION - MAINTENANCE PRACTICES (Effectivity: All)

FIREWALL BLEED AIR SHUTOFF VALVE REMOVAL (Effectivity: NC-4 thru NC-29) (Figure 3)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 54-00-00 and remove the upper aft access panel (583AT or 683AT) from the applicable nacelle.

c. Refer to Chapter 71-10-00 and remove the upper forward engine cowling.

d. Remove tape and insulation to allow access to the pneumatic connections of the firewall bleed air shutoff valve (3).

e. Tag and disconnect the electrical harness from the firewall bleed air shutoff valve (3).

f. Tag and disconnect the pneumatic pressure sense line (11) from the shutoff valve (3).

g. Remove the union (10) from the sense port. Remove and discard the O-ring (9).

h. Remove the coupling (1) and gasket (2) from the forward connection on the firewall bleed air shutoff valve (3).

i. Remove the bolts (12), washers (7) and nuts (6) securing the shutoff valve (3) to the fire wall (5).

j. Pull the firewall bleed air shutoff valve (3) away from the fire wall (5) and remove the spacer (8) and soft copper gasket (4). Retain the spacer for reinstallation.

FIREWALL BLEED AIR SHUTOFF VALVE INSTALLATION (Effectivity: NC-4 thru NC-29) (Figure 3)

CAUTION

The firewall bleed air shutoff valve must meet the electrical bonding requirements specified in Chapter 20-00-03.

a. Install a new soft copper gasket (4) in the recess in the firewall bleed air shutoff valve (3) and align the spacer (8) with the doubler on the fire wall (5).

b. Seal the flange of the firewall bleed air shutoff valve (3) to the spacer (8) and the spacer to the firewall doubler with sealant (2, Chart 1, 36-00-00). Wipe off any excess sealant.

c. Align the firewall bleed air shutoff valve (3) with the mounting holes in the spacer (8) and the doubler on the fire wall (5).

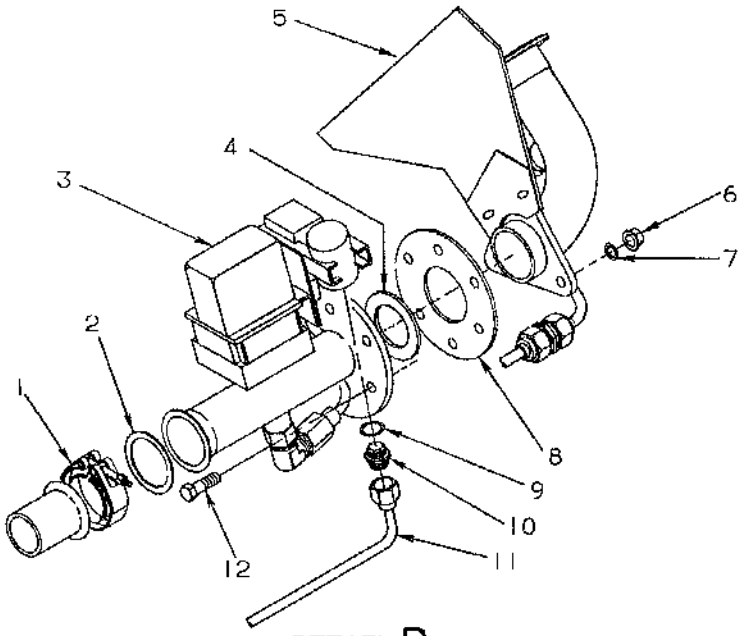
d. Install the bolts (12) , washers (7) and nuts (6) securing the firewall bleed air shutoff valve (3) to the fire wall.

e. Install a new gasket (2) at the connection between the forward bleed air tube and the fire wall bleed air shutoff valve (3) and secure with a coupling (1). Tighten the coupling to 45-55 inch-pounds.

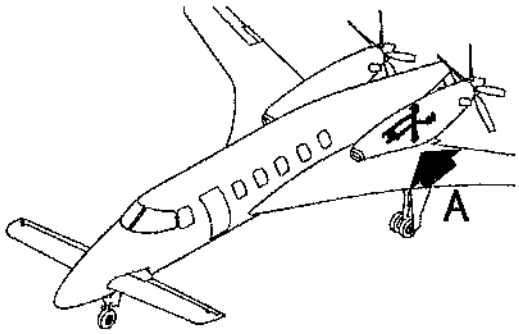
f. Install a new O-ring (9) on the union (10) and install the union in the sense port.

g. Connect the pneumatic pressure sense line (11) to the bleed air firewall shutoff valve union (10).

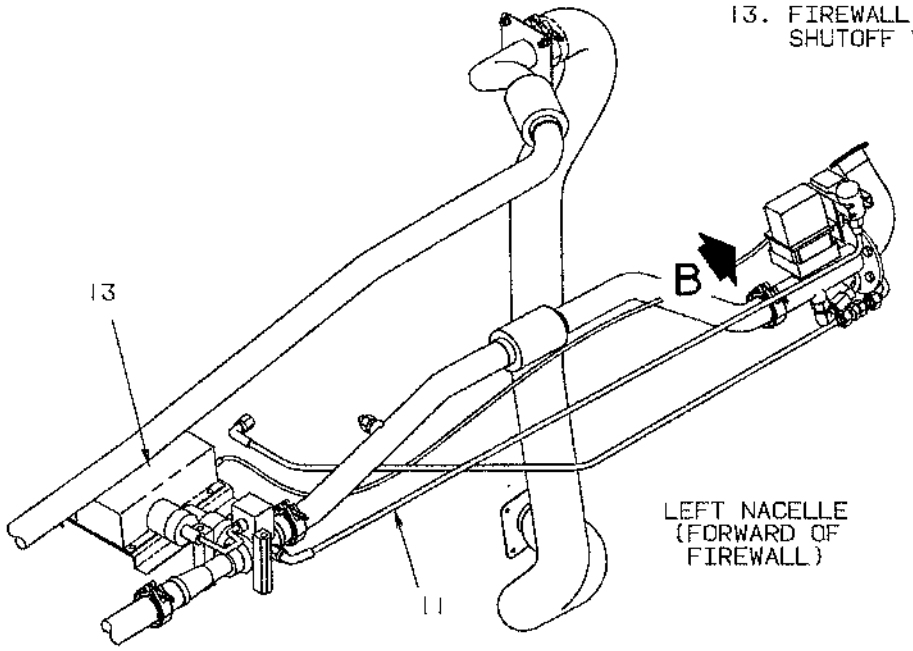
h. Connect the electrial harness and bonding jumpers to the firewall bleed air shutoff valve (3) according to tags.



DETAIL B



1. COUPLING
2. GASKET
3. FIREWALL BLEED AIR SHUTOFF VALVE
4. COPPER GASKET
5. FIRE WALL
6. NUT
7. WASHER
8. SPACER
9. O-RING
10. UNION
11. SENSE LINE
12. BOLT
13. FIREWALL BLEED AIR SHUTOFF VALVE CONTROLLER



DETAIL A

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Firewall Bleed Air Shutoff Valve Installation
(Effectivity: NC-4 thru NC-29)
Figure 3

i. Wrap the bleed air line with two layers of insulation (3, Chart 1, 36-00-00) or (4, Chart 1, 36-00-00).

j. Wrap the insulation with tape (5, Chart 1, 36-00-00).

k. Refer to Chapter 54-00-00 and install the upper aft access panel (583AT or 683AT) on the applicable nacelle.

l. Refer to Chapter 71-10-00 and install the upper forward engine cowling.

m. Restore electrical power to the airplane.

FIREWALL BLEED AIR SHUTOFF VALVE CONTROLLER REMOVAL (Effectivity: NC-4 thru NC-29) (Figure 4)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 54-00-00 and remove the upper forward (581AT or 681AT) and the upper aft (583AT or 683AT) nacelle access panels from the applicable nacelle.

c. Tag and disconnect the electrical connectors from the firewall bleed air shutoff valve controller (1).

d. Remove the screws (5) and washers (4) securing the firewall bleed air shutoff valve controller (1) to the mounting bracket (3).

e. Remove the firewall bleed air shutoff valve controller (1) from the nacelle.

FIREWALL BLEED AIR SHUTOFF VALVE CONTROLLER INSTALLATION (Effectivity: NC-4 thru NC-29) (Figure 4)

CAUTION

The firewall bleed air shutoff valve controller must meet the electrical bonding requirements specified in Chapter 20-00-03.

a. Align the firewall bleed air shutoff valve controller (1) with the mounting bracket (3) and secure with screws (5) and washers (4).

b. Connect the electrical connectors to the firewall bleed air shutoff valve controller (1) and connect all bonding jumpers according to tags.

c. Refer to Chapter 54-00-00 and install the nacelle upper forward and upper aft access panels on the applicable nacelle.

d. Restore electrical power to the airplane.

FIREWALL BLEED AIR SHUTOFF VALVE REMOVAL (Effectivity: NC-30 and after) (Figure 5)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 54-00-00 and remove the upper aft access panel (583AT or 683AT) from the applicable nacelle.

c. Refer to Chapter 71-10-00 and remove the upper forward engine cowling.

d. Remove tape and insulation to allow access to the pneumatic connections of the firewall bleed air shutoff valve (3).

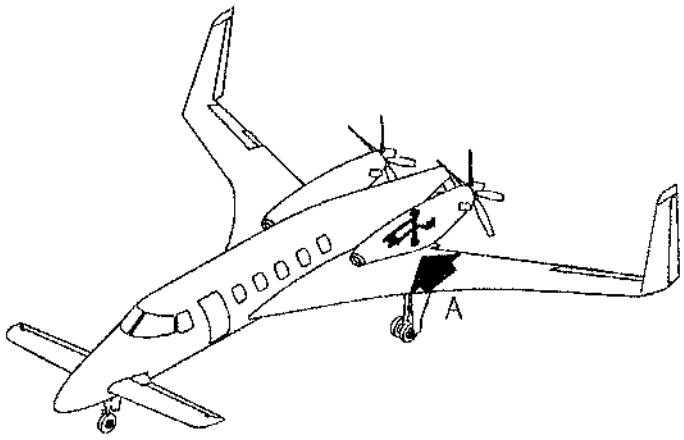
e. Disconnect the electrical harness from the electrical connector (4) of the firewall bleed air shutoff valve.

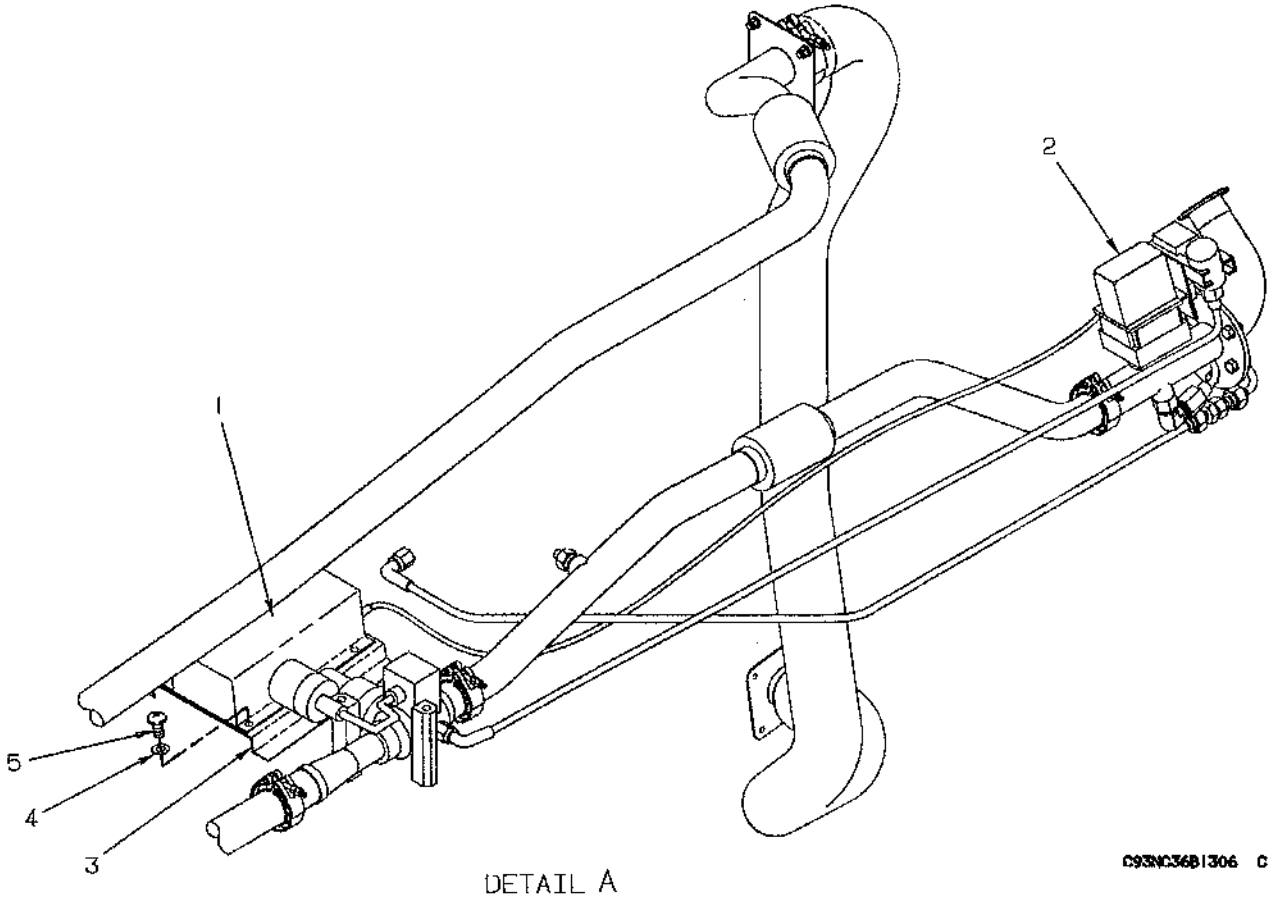
f. Remove the coupling (1) and gasket (2) from the forward connection on the firewall bleed air shutoff valve (3).

g. Remove the bolts (10), washers (7) and nuts (6) securing the shutoff valve (3) to the fire wall (5).

h. Pull the firewall bleed air shutoff valve (3) away from the fire wall (5) and remove the spacer (8) and soft copper gasket (9). Retain the spacer for reinstallation.

1. FIREWALL BLEED AIR SHUTOFF VALVE CONTROLLER
2. FIREWALL BLEED AIR SHUTOFF VALVE
3. MOUNTING BRACKET
4. WASHER
5. SCREW





DETAIL A

Firewall Bleed Air Shutoff Valve Controller Installation
(Effectivity: NC-4 thru NC-29)
Figure 4

FIREWALL BLEED AIR SHUTOFF VALVE INSTALLATION (Effectivity: NC-30 and after)
(Figure 5)

The P/N 122-382171-1 firewall bleed air shutoff valve installed on airplanes NC-30 and after replaces P/N 122-382066 firewall bleed air shutoff valve/controller used on airplanes NC-4 thru NC-29. When making this replacement, remove and discard sense line (12) and install an AN929-4J cap (14) on sense line port (13). Also remove and discard firewall bleed air shutoff valve controller (11). The electrical harness that connects to the controller on airplanes NC-4 thru NC-29 connects to the electrical connector (4) on the firewall bleed air shutoff valves on airplanes NC-30 and after.

CAUTION

The firewall bleed air shutoff valve must meet the electrical bonding requirements specified in Chapter 20-00-03.

a. Install a new soft copper gasket (9) in the recess in the firewall bleed air shutoff valve (3) and align the spacer (8) with the doubler on the fire wall (5).

b. Align the firewall bleed air shutoff valve (3) with the mounting holes in the spacer (8) and firewall doubler (5).

c. Seal the flange of the firewall bleed air shutoff valve (3) to the spacer (8) and the spacer to the firewall doubler with sealant (2, Chart 1, 36-00-00). Wipe off any excess sealant.

d. Install the bolts (10) , washers (7) and nuts (6) securing the firewall bleed air shutoff valve (3) to the fire wall.

e. Install a new gasket (2) at the connection between the forward bleed air tube and the fire wall bleed air shutoff valve (3) and secure with a coupling (1). Tighten the coupling to 45-55 inch-pounds.

f. Connect the electrial harness to the firewall bleed air shutoff valve electrical connector (4).

g. Wrap the bleed air line with two layers of insulation (3, Chart 1, 36-00-00) or (4, Chart 1, 36-00-00).

h. Wrap the insulation with tape (5, Chart 1, 36-00-00).

i. Refer to Chapter 54-00-00 and install the upper aft access panel (583AT or 683AT) on the applicable nacelle.

j. Refer to Chapter 71-10-00 and install the upper forward engine cowling.

k. Restore electrical power to the airplane.

PNEUMATIC SYSTEM PRESSURE REGULATOR VALVE REMOVAL
(Effectivity: All)
(Figure 6)

The pneumatic system pressure regulator valves are mounted on the forward wall of the main landing gear wheel wells.

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove all electrical power from the airplane.

b. Disconnect both pneumatic lines (1 and 13) from the pressure regulator valve (7).

c. Remove the four bolts (15) securing the forward bracket (16) to the forward wall of the main wheel well and remove the brackets (8 and 16) and pneumatic pressure regulator valve (7) from the airplane.

d. Remove the four bolts (10) and spacers (11) securing the brackets (8 and 16) together.

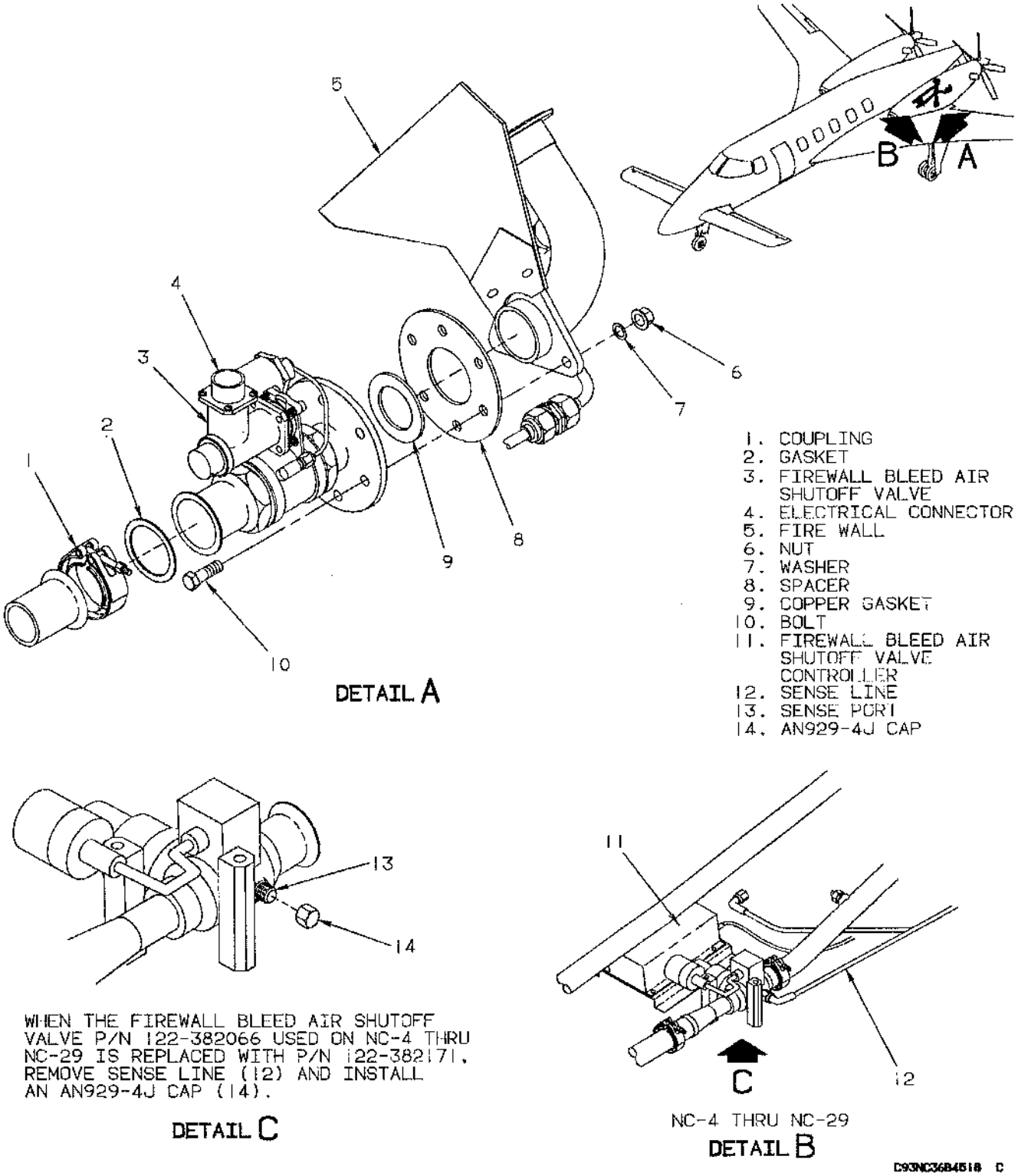
e. Remove the four bolts (9) securing the brackets to the pneumatic pressure regulator valve (7) and remove the brackets.

f. Remove the reducer (5) from the presure regulator valve (7). Remove and discard the O-ring (6) from the reducer (5).

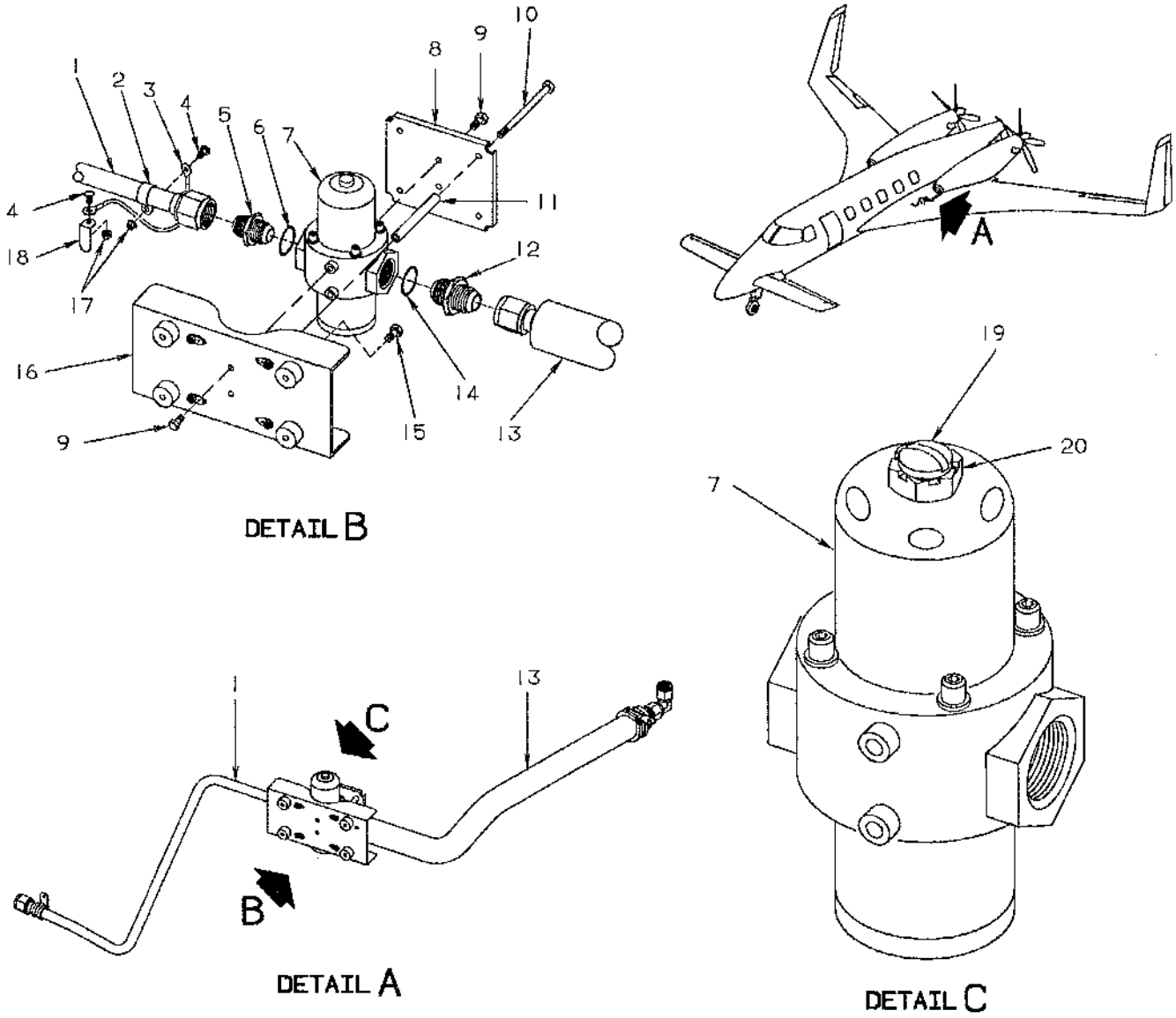
g. Remove the union (12) from the pneumatic pressure regulator valve (7). Remove and discard the O-ring (14) from the union (12).

PNEUMATIC SYSTEM PRESSURE REGULATOR VALVE INSTALLATION
(Effectivity: All)
(Figure 6)

a. Install a new O-ring (6) on the reducer (5) and install the reducer (5) in the pneumatic pressure regulator valve (7).



Firewall Bleed Air Shutoff Valve Installation
(Effectivity: NC-30 and after)
Figure 5



1. INBOARD BLEED AIR LINE
2. CLAMP
3. BONDING JUMPER
4. SCREW (2)
5. REDUCER
6. O-RING
7. PRESSURE REGULATOR VALVE
8. SUPPORT BRACKET
9. BOLT (4)
10. BOLT (4)
11. SPACER (4)
12. UNION
13. OUTBOARD BLEED AIR LINE
14. O-RING
15. BOLT
16. FORWARD BRACKET
17. NUT (2)
18. BONDING JUMPER BRACKET
19. ADJUSTING SCREW
20. LOCKNUT

LEFT SIDE SHOWN
TYPICAL FOR RIGHT SIDE

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Pneumatic System Pressure Regulator Valve Installation
(Effectivity: All)
Figure 6

b. Install a new O-ring (14) on the union (12) and install the union (12) in the pneumatic pressure regulator valve (7).

c. Install the two bolts (9), attaching the forward bracket (16) to the pneumatic pressure regulator valve (7) on the side with the flow arrow pointing inboard.

NOTE

The flow arrow on the pneumatic pressure regulator valve (7) should be on the lower side and should point inboard.

d. Attach the support bracket (8) to the opposite side of the pneumatic pressure regulator valve (7) and secure with two bolts (9).

e. Install the four bolts (10) and spacers (11) securing the two brackets together.

f. Install the regulator valve/bracket assembly on the forward wall of the main landing gear wheel well with four bolts (15).

g. Connect the pneumatic lines (1 and 13) to the pneumatic pressure regulator valve (7).

h. Connect the bonding jumper (3) with screw (4) and nut (17) to clamp (2) if disconnected.

i. Restore electrical power to the airplane.

PNEUMATIC SYSTEM PRESSURE REGULATOR VALVE ADJUSTMENT (Effectivity: All) (Figure 6)

Ordinarily, the pressure regulator valve will not require adjustment since each valve is preset at the factory; however, the pressure regulator valve is equipped with an adjusting screw should it become necessary to reset the valve in the field. Before adjusting the pressure regulator valve due to a low pressure reading on the pressure display, it might be beneficial to perform the PNEUMATIC SYSTEM LEAK CHECK procedure in this section to check for leakage in the plumbing, especially after maintenance has been performed on the system. Adjustment may be accomplished as follows:

a. Run up the engine until a reading between 70 to 80 percent N_1 is observed.

b. Check the pneumatic pressure display for a reading of 22 to 25 psig.

c. If adjustment is necessary, loosen the locknut (20) located on top of the valve (7).

d. Turn the adjusting screw (19) clockwise to increase or counterclockwise to decrease the pressure setting until the pneumatic pressure display indicates 22 to 25 psig.

CAUTION

Do not exceed the recommended 22 to 25 psig regulator setting because higher settings will result in excessive wear on the relief valve.

e. Tighten the locknut (20), then shut down the engine.

LANDING GEAR POWER PACK PRESSURE REGULATOR VALVE REMOVAL (Effectivity: All) (Figure 7)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 54-00-00 and remove the upper forward (581AT) and upper aft (583AT) access panels from the left nacelle.

c. Disconnect the pneumatic lines (9) from the landing gear power pack pressure regulator valve (3).

d. Remove both unions (5) from the pressure regulator valve (3).

e. Remove and discard the O-rings (6) from the unions (5).

f. Remove the bolts (1), washers (2 and 8) and nuts (7) securing the pressure regulator valve (3) to the mounting bracket (4) on the engine air inlet duct.

g. Remove the pressure regulator valve (3) from the airplane.

LANDING GEAR POWER PACK PRESSURE REGULATOR VALVE INSTALLATION (Effectivity: All) (Figure 7)

The power pack pressure regulator valve is factory set and cannot be readjusted in the field.

CAUTION

The power pack pressure regulator valve must meet the electrical bonding requirements specified in Chapter 20-00-03.

a. Install a new O-ring (6) on each union (5) and install the unions in the pressure regulator valve.

b. Align the pressure regulator valve (3) with the mounting bracket (4) on the engine air inlet duct and secure with bolts (1), washers (2 and 8) ands nuts (7).

c. Connect the pneumatic lines (9) to the power pack pressure regulator valve (3).

d. Restore electrical power to the airplane.

e. Refer to Chapter 54-00-00 and install the left nacelle upper forward (581AT) and upper aft (583AT) access panels.

AFT WING SURFACE DEICE SYSTEM CHECK VALVES REMOVAL (Effectivity: All) (Figure 8)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

NOTE

The check valves that distribute pneumatic pressure to the aft wing surface deice systems are located two per side between the fuselage and the wing root rib from F.S. 320 to F.S. 340. Removal is the same for all four check valves.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-50-00 and remove the center (527CB or 627CB) and aft (527DB or 627DB) panels from the applicable side.

c. Disconnect the inlet and downstream tubes (1 and 5) from the check valve (4) being removed. Note the direction of the flow arrow on the check valve.

d. Remove the check valve (4) from the airplane.

e. Remove the orifice assembly (2) from the IN port on the check valve (4).

f. Remove and discard the O-ring (3).

AFT WING SURFACE DEICE SYSTEM CHECK VALVES INSTALLATION (Effectivity: All) (Figure 8)

CAUTION

The aft wing deice system check valves must meet the electrical bonding requirements specified in Chapter 20-00-03.

NOTE

Installation procedures for the aft wing surface deice system check valves is the same.

a. Install a new O-ring (3) on the orifice assembly (2) and install the orifice assembly (2) in the IN port of the check valve (4).

b. Align the check valve (4) with the inlet and downstream tubes (1 and 5) and connect the tubes to the check valve.

NOTE

Ensure that the arrow on the check valve (4) points in the same direction noted during removal (downstream).

c. Install any bonding jumpers that may have been removed from the plumbing.

d. Restore electrical power to the airplane.

e. Perform the PNEUMATIC SYSTEM LEAK CHECK procedure outlined in this section.

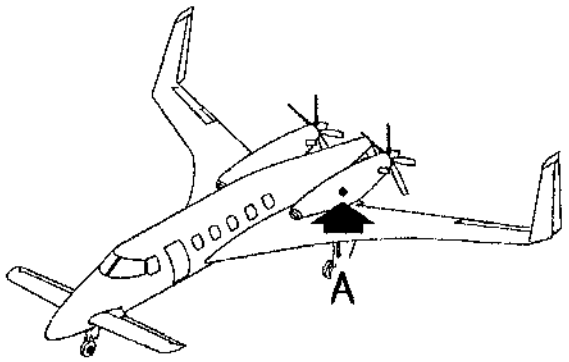
f. Refer to Chapter 53-50-00 and install the aft and center fairing panels between the fuselage and wing.

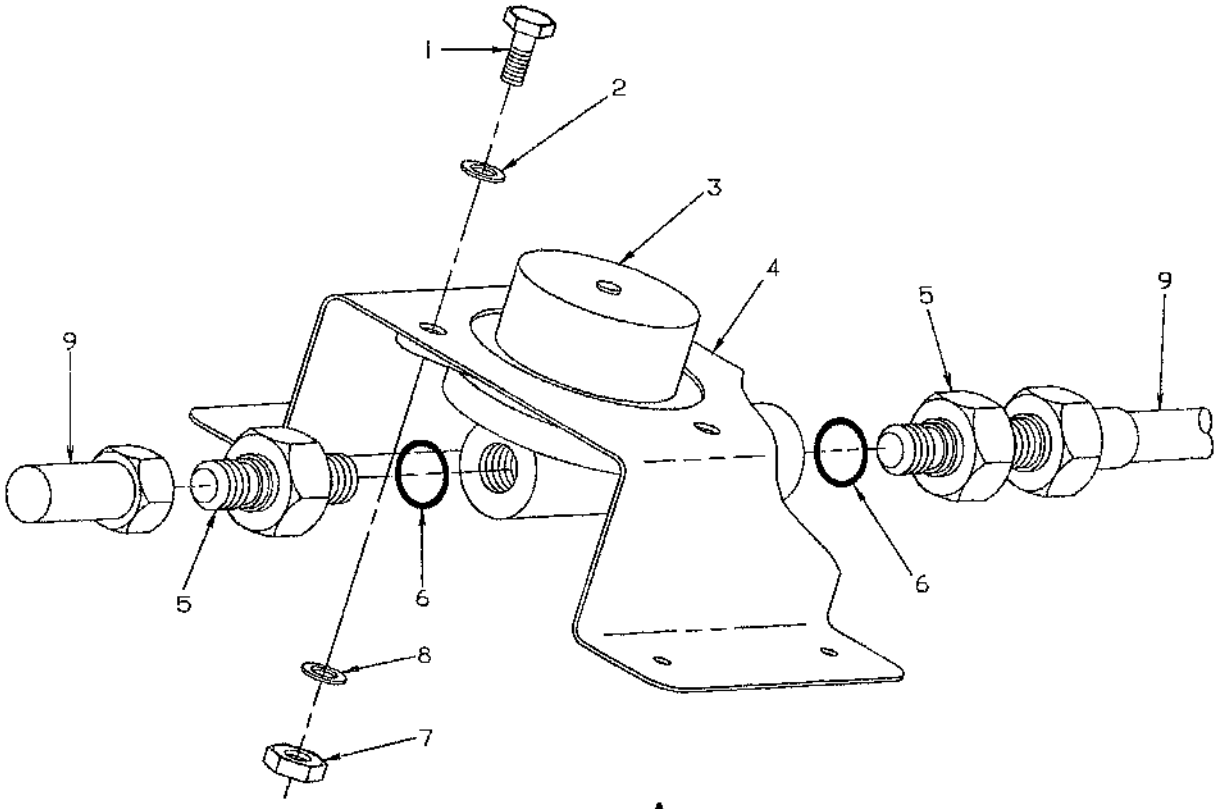
FOWARD WING SURFACE DEICE SYSTEM CHECK VALVE REMOVAL (Effectivity: All) (Figure 9)

CAUTION

When removing components, note the locations of all bonding jumpers to aid in reinstallation.

1. BOLT
2. WASHER
3. LANDING GEAR POWER PACK PRESSURE REGULATOR VALVE
4. BRACKET
5. UNION
6. O-RING
7. NUT
8. WASHER
9. PNEUMATIC LINE





DETAIL A

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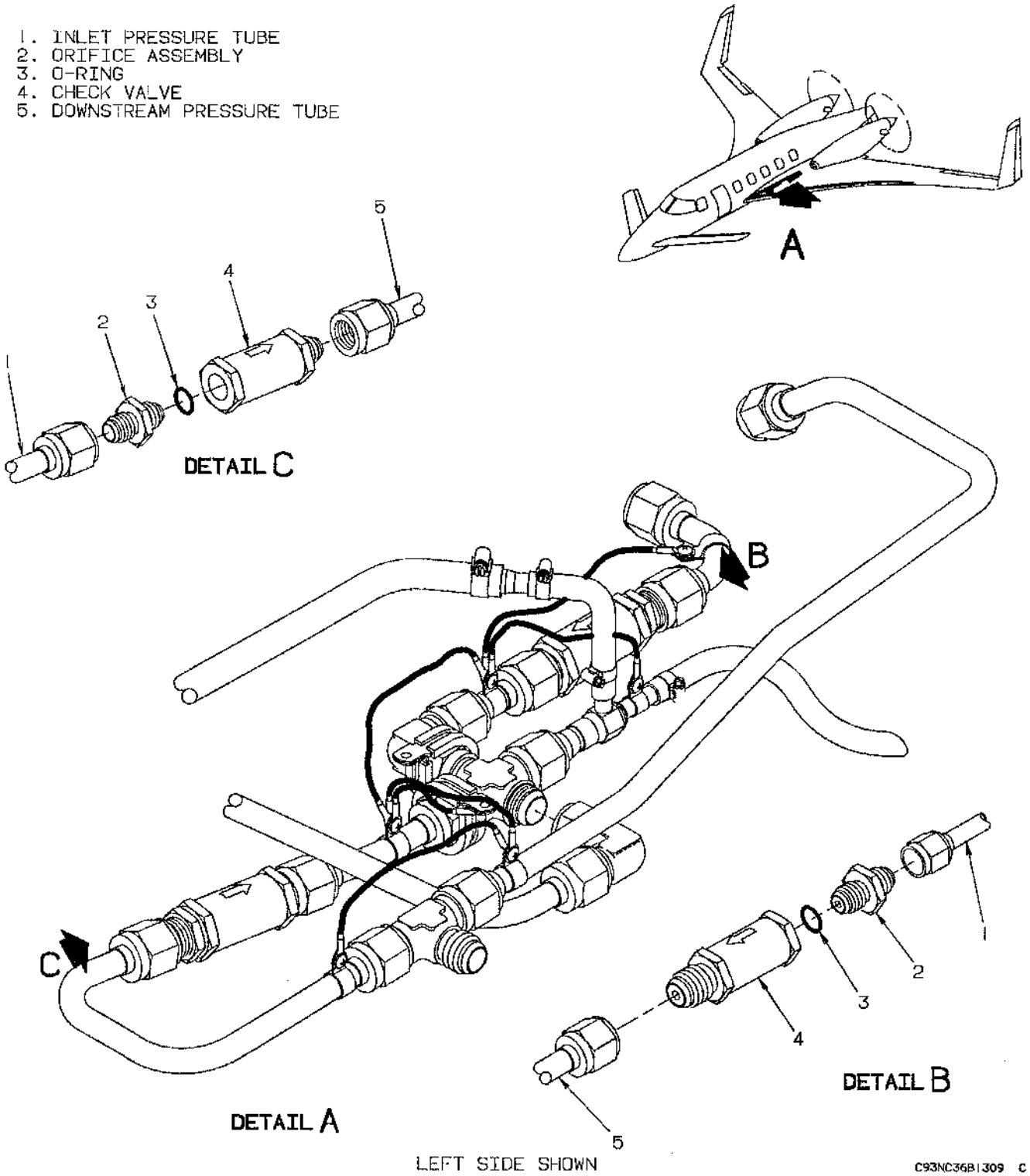
Landing Gear Power Pack Pressure Regulator Valve Installation
(Effectivity: All)
Figure 7



Aft Wing Surface Deice System Check Valve Installation
(Effectivity: All)
Figure 8

NOTE

The forward wing surface deice system check valves (2) are located in the airplane nose area where the pneumatic supply tubes penetrate the forward pressure bulkhead. Removal is the same for both check valves.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 53-30-00 and remove the right nose skin panel (132AR).

c. Disconnect the downstream pressure tube (1) from the check valve (2) being removed.

CAUTION

The orifice assembly (4) is pressure sealed where it passes through the forward pressure bulkhead (5). Hold the orifice assembly (4) stationary while removing the check valve (2) to preserve the pressure seal.

d. Remove the check valve (2) from the orifice assembly (4) at the forward pressure bulkhead (5).

e. Remove and discard the O-ring (3) from the orifice assembly (4).

FORWARD WING SURFACE DEICE SYSTEM CHECK VALVE INSTALLATION (Effectivity: All) (Figure 9)

CAUTION

The forward wing surface deice system check valves (2) must meet the electrical bonding requirements specified in Chapter 20-00-03.

a. Install a new O-ring (3) on the orifice assembly (4).

CAUTION

The orifice assembly (4) is pressure sealed where it passes through the forward pressure bulkhead (5). Hold the orifice assembly (4) stationary while installing the check valve (2) to preserve the pressure seal.

b. Install the check valve (2) on the orifice assembly (4) at the forward pressure bulkhead (5). The flow arrow should point forward (downstream).

c. Install the downstream pressure tube (1) on the other side of the check valve (2).

d. Install any bonding jumpers that may have been removed from the plumbing.

e. Restore electrical power to the airplane.

f. Perform the PNEUMATIC SYSTEM LEAK CHECK procedure outlined in this section.

g. Refer to Chapter 53-30-00 and install the right nose skin panel.

PNEUMATIC SYSTEM FORWARD CHECK VALVES REMOVAL (Effectivity: All) (Figure 10)

The pneumatic system forward check valves are mounted on the right side of the keel below the control pedestal.

a. Remove all electrical power from the airplane.

b. Remove the pilot's seat and control pedestal side upholstery panels as instructed in Chapter 25-10-00.

c. Disconnect the pressurization control tube (1) and the door seal pressurization tube (15) from the tee assembly (3).

d. Remove the nut (4) and washer (5) from the clamp stud (13).

e. Disconnect the swivel tees (10) from the unions (9) and remove the check valve/tee assembly from beneath the control pedestal. Remove the spacer (12) from the stud (13).

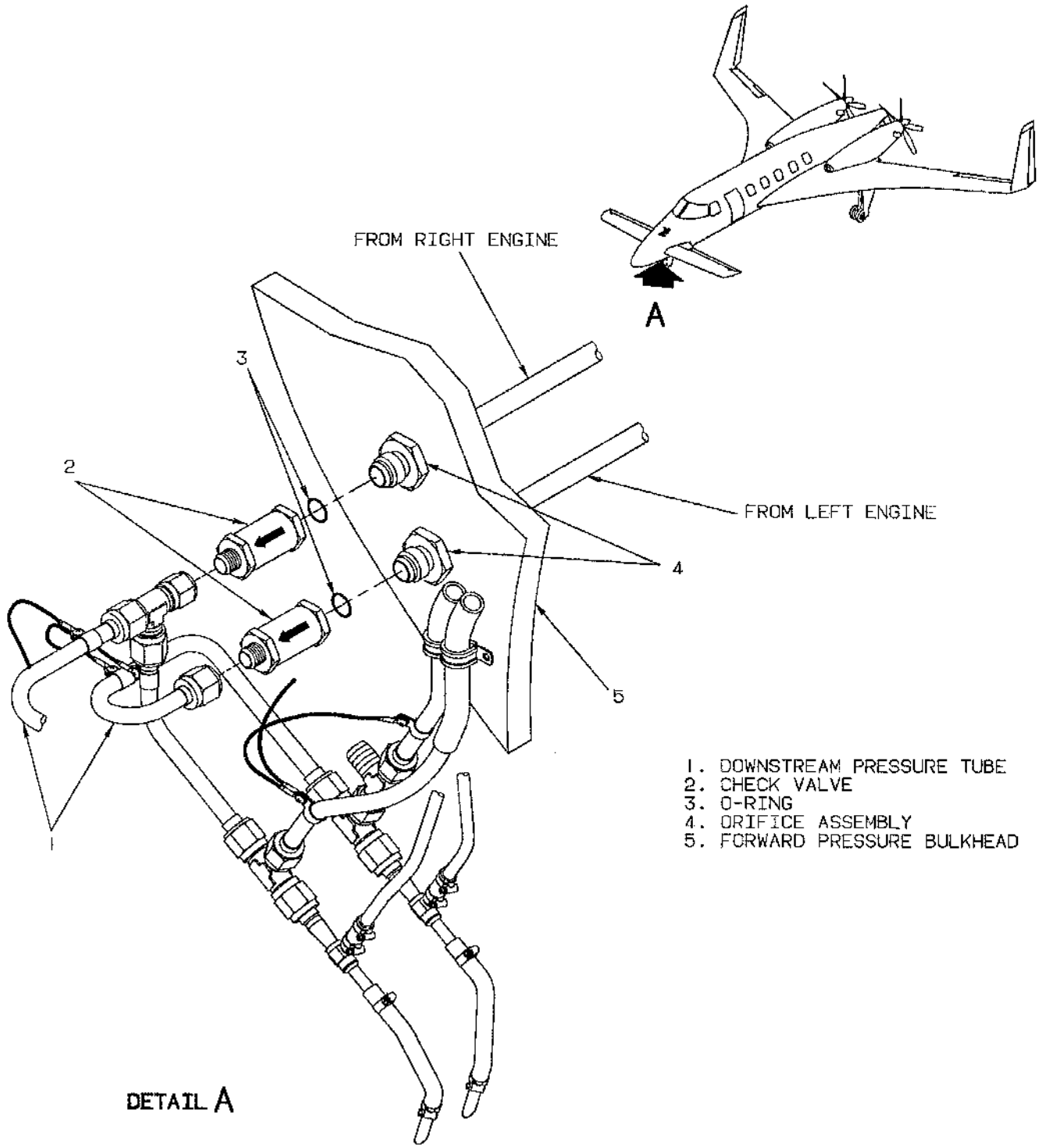
f. Remove the check valves (7 and 11) from the tee assembly (3). Remove the O-rings and discard them.

g. Remove the unions (9) from the check valves (7 and 11). Remove the O-rings and discard them.

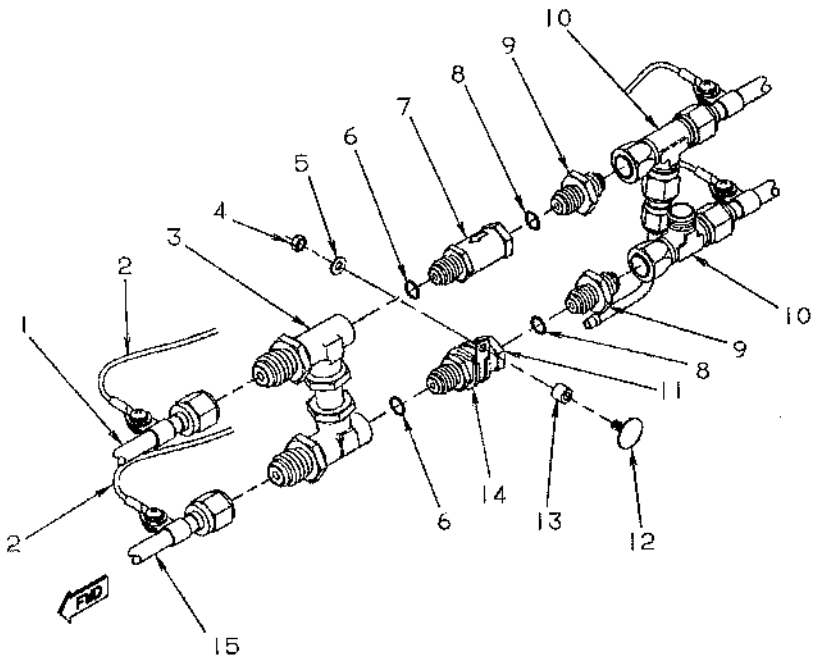
h. Remove the clamp (14) from the check valve (11).

PNEUMATIC SYSTEM FORWARD CHECK VALVES INSTALLATION (Effectivity: All) (Figure 10)

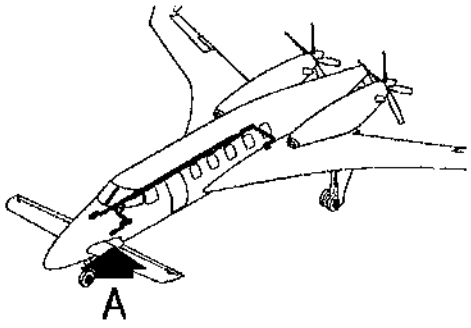
a. Install new O-rings (6) on the check valves (7 and 11) and connect the check valves to the tee assembly (3).



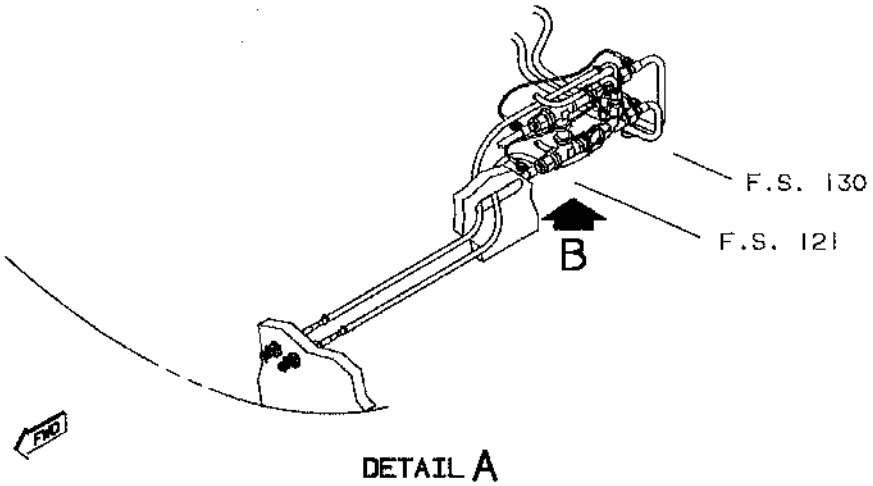
Forward Wing Surface Deice System Check Valve Installation
(Effectivity: All)
Figure 9



DETAIL B



1. PRESSURIZATION CONTROL TUBE
2. BONDING JUMPER
3. TEE ASSEMBLY
4. NUT
5. WASHER
6. O-RING
7. TOP CHECK VALVE
8. O-RING
9. UNION
10. SWIVEL TEE
11. BOTTOM CHECK VALVE
12. STUD
13. SPACER
14. CLAMP
15. DOOR SEAL PRESSURIZATION TUBE



DETAIL A

C94NC36B0104

Pneumatic System Forward Check Valves Installation
(Effectivity: All)
Figure 10

b. Install new O-rings (8) on the unions (9) and connect the unions to the check valves.

c. Install a clamp (14) around the bottom check valve (11).

d. Install a spacer (12) on the stud (13).

e. Place the check valve/tee assembly in position and connect the swivel tees (10) to the unions (9). Install the clamp (14) on the stud (13). Make sure the spacer (12) is between the clamp (14) and the stud (13). Install the washer (5) and the nut (4).

f. Connect tubes (1 and 15) to the tee assembly (3).

g. Leak test the pneumatic system as instructed in the PNEUMATIC SYSTEM LEAK CHECK procedure in this chapter.

h. Install the pedestal side upholstery panels and the pilot's seat as instructed in Chapter 25-10-00.

ENGINE P_3 AIR TUBING REMOVAL (Effectivity: All) (Figure 11)

The engine P_3 air outlet is on the right side of the engines. The bleed air plumbing is routed through the outboard side of the fire wall. Flex hose (9) on the right engine is shorter than the left hose which crosses over the top of the left engine to the outboard side. Individual pieces of the tubing can be removed without removing the other pieces.

a. When removing the left (4) or right (8) tube assembly, refer to Chapter 54-00-00 and remove the upper aft access panel (583AT or 683AT) from the applicable nacelle.

b. Refer to Chapter 71-10-00 and remove the upper forward and upper aft cowling from the engine.

c. Refer to Chapter 71-60-00 and remove the plenum top cover.

d. When removing tube (12), remove the closeout panel assembly that surrounds the tube as instructed in Chapter 71-60-00.

e. To remove tube (12), disconnect the fuel purge hose (13) and remove the coupling (1) connecting tube (12) to the flex hose (9). Remove four bolts (10) and washers (11) attaching the tube (12) to the engine. Remove the tube from the engine.

f. To remove flex hose (9), remove the coupling (1) at each end of the flex hose and remove the flex hose from the engine.

g. To remove the left (4) or right (8) tube assembly, remove the coupling (1) that connects the flex hose to the tubing assembly. On the left tube assembly (4), disconnect the landing gear power pack pressure tube (14). Remove three bolts (3), nuts (7) and washers (6) and remove the tube assembly from the engine. Remove the soft copper gasket (5) from the cavity.

ENGINE P_3 AIR TUBING INSTALLATION (Effectivity: All) (Figure 11)

a. Install the left tube assembly (4, Detail D) on the left nacelle fire wall with a soft copper gasket (5), three bolts (3), washers (6) and nuts (7). Connect the landing gear power pack pressure tube (14).

b. Install the right tube assembly (8) with a soft copper gasket (5) and three bolts (3), washers (6) and nuts (7).

c. Connect the flex hose (9) to the left (4) or right (8) tube assembly with a coupling (1) and tighten the coupling bolt to 45 to 55 inch-pounds.

d. Position the forward end of the tube (12) through the hole in the plenum and connect the aft end of the tube to the engine P_3 outlet with four bolts (10) and washers (11).

e. Connect the fuel purge hose (13) to the tube (12).

f. Connect tube (12) to the flex hose (9) with a coupling (1) and tighten the coupling bolt to 45 to 55 inch-pounds.

g. Install the closeout panel on the plenum as instructed in Chapter 71-60-00.

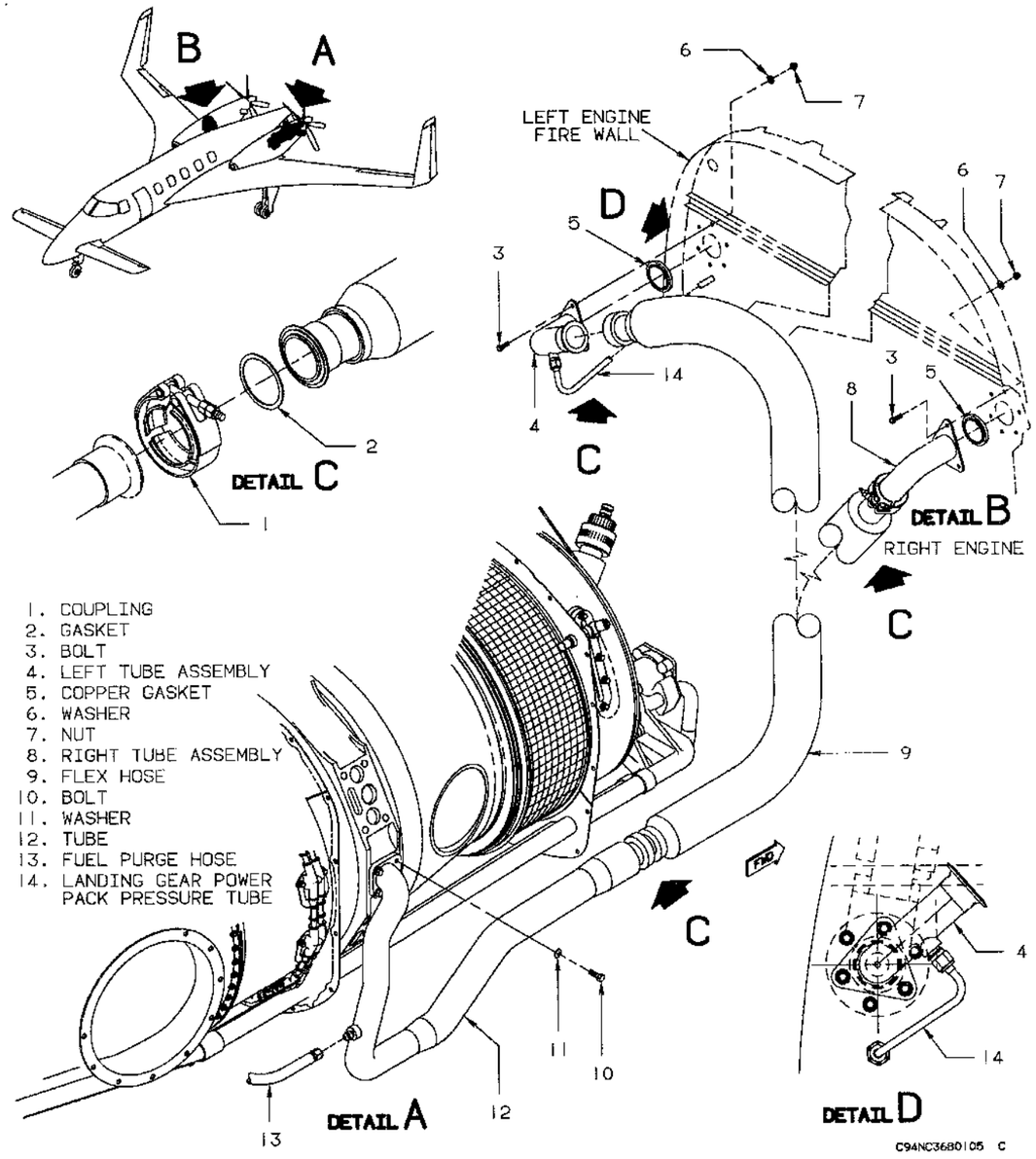
h. Refer to Chapter 71-60-00 and install the plenum top cover.

i. Refer to Chapter 71-10-00 and install the upper forward and upper aft cowlings.

j. Install the upper aft access panel (583AT or 683AT) as instructed in Chapter 54-00-00.

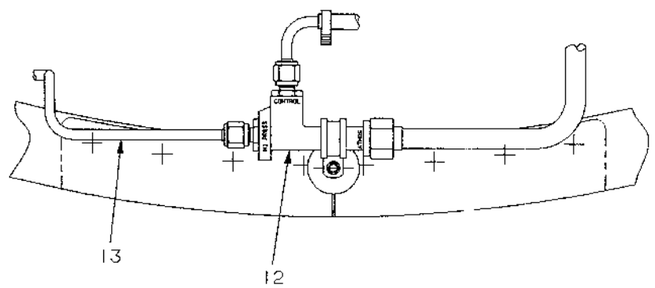
PNEUMATIC SYSTEM LEAK CHECK (Effectivity: All) (Figure 12)

This procedure checks the pneumatic system plumbing for leaks from the pneumatic system pressure regulator valves to each of the surface deice system distributor valves and the pressure display, pressurization control system and the door seal pressure tubing. This leak check should be accomplished after

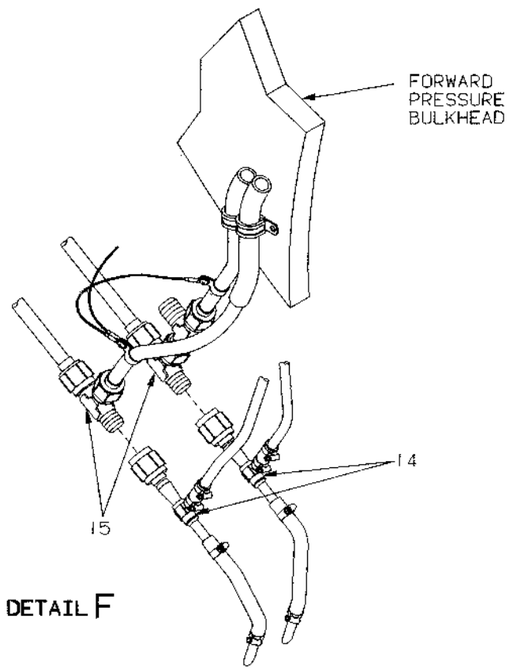


Engine P_3 Air Tubing Installation
(Effectivity: All)
Figure 11

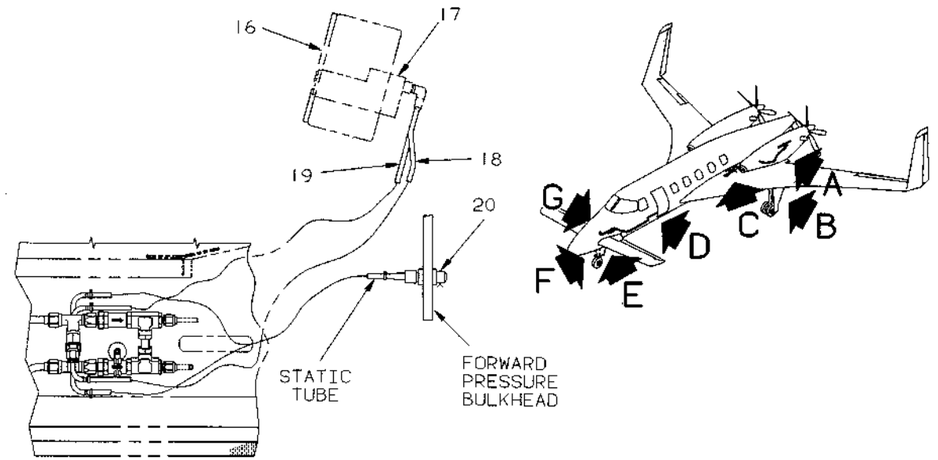
1. PNEUMATIC SYSTEM TUBE
2. BLEED AIR DUCT
3. INLET LINE (OUTBOARD)
4. PRESSURE REGULATOR VALVE
5. OUTLET LINE (INBOARD)
6. VACUUM EJECTOR
7. FOUR-WAY FITTING
8. AN 806-4 PLUG
9. DOOR SEAL SOLENOID VALVE
10. DOOR SEAL PRESSURE TUBE FITTING
11. TEE
12. PRESSURIZATION VACUUM EJECTOR
13. PRESSURIZATION CONTROL TUBE
14. VACUUM EJECTOR
15. TEE-FITTING
16. PILOT'S INBOARD SUBPANEL
17. PNEUMATIC SYSTEM PRESSURIZATION DISPLAY
18. LEFT PRESSURE TUBE
19. RIGHT PRESSURE TUBE
20. COTTER PIN



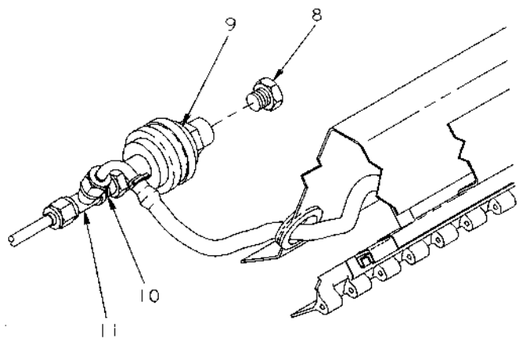
DETAIL E



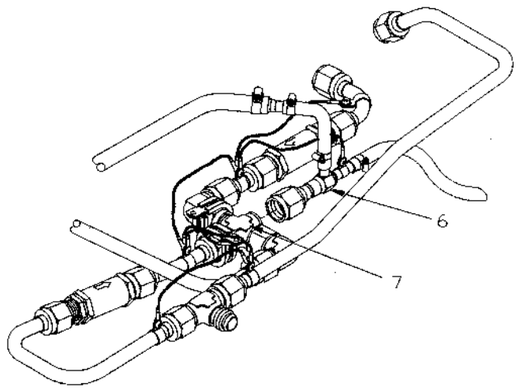
DETAIL F



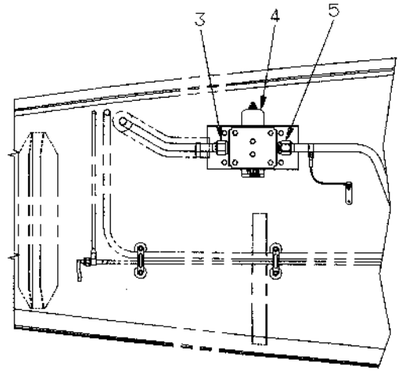
DETAIL G



DETAIL D

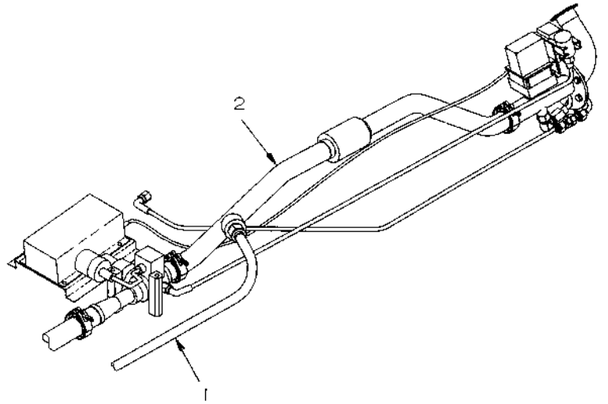


DETAIL C



LEFT SIDE SHOWN
TYPICAL FOR RIGHT SIDE

DETAIL B



DETAIL A

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Pneumatic System Leak Check
(Effectivity: All)
Figure 12

plumbing has been disconnected or when the pressure display shows consistent low pressure. The pressure leak check should be accomplished prior to adjustment of the pneumatic system pressure regulator valve.

a. Remove the upper forward nacelle access panel (581AT and 681AT) from the nacelle as instructed in Chapter 54-00-00.

b. Disconnect the pneumatic system tube (1) from the bleed air duct (2) and plug the pneumatic system tube (1) with an AN806-J10 plug.

c. Remove the wing-to-fuselage fairing panels (527BB, 527CB, 527DB, 627BB, 627CB and 627DB) from between the fuselage and wing as instructed in Chapter 53-50-00.

d. Disconnect the vacuum ejector (6) from the four-way fitting (7) and cap the four-way fitting with an AN929-10 cap.

e. Remove the right nose skin panel (132AR) as instructed in Chapter 53-30-00 and disconnect the vacuum ejectors (14) from the tees (15). Cap the tees with AN929-10 caps.

f. Remove the left nose skin panel (131AL) as instructed in Chapter 53-30-00. Remove the cotter pins (20) from the static ports on the forward side of the forward pressure bulkhead and install AN929-4 caps.

g. Remove the access panel (133AB) from the bottom of the nose and disconnect the pressurization control tube (13) from the pressurization vacuum ejector (12). Plug the pressurization control tube (13) with an AN 806-4 plug.

h. Disconnect the pressure tubes (18 and 19) from the pneumatic pressure display (17) located in the pilots inboard subpanel (16). Plug the tubes.

i. Remove the floorboard (151AT) at the door entryway as instructed in Chapter 53-20-00.

j. Disconnect the door seal pressure tube fitting (10) from the tee (11). Cap the tee (11) with an AN929-4 cap. Plug the door seal solenoid valve (8) outlet port with an AN806-4 plug (9).

k. In the main landing gear wheel wells, disconnect both tubes (3 and 5) from the right and left pneumatic pressure regulator valves (4). Connect the two tubes (3 and 5) with a tee. Connect an external air source that is regulated to 46 to 50 psig to the right and left tees.

l. Open the valve on the external air source and pressurize the system. Allow the applied pressure (46 to 50 psig) to stabilize for five minutes then close the valve.

m. Allow the system to remain pressurized for 15 minutes. The drop in pressure shall not exceed two psi.

n. If there is a drop in pressure of more than two psi, apply leak detector fluid (1, Chart 1, 36-00-00) sparingly to all connections. Wipe dry after testing. Make necessary repairs if a leak is found and retest the system.

o. Remove the air pressure source, all plugs and caps. Connect the vacuum ejectors and connect all tubing.

p. Remove the caps from the static ports on the forward side of the forward pressure bulkhead and install new cotter pins (20).

q. Replace all panels and floorboards as instructed in the referenced chapters in steps a through i.

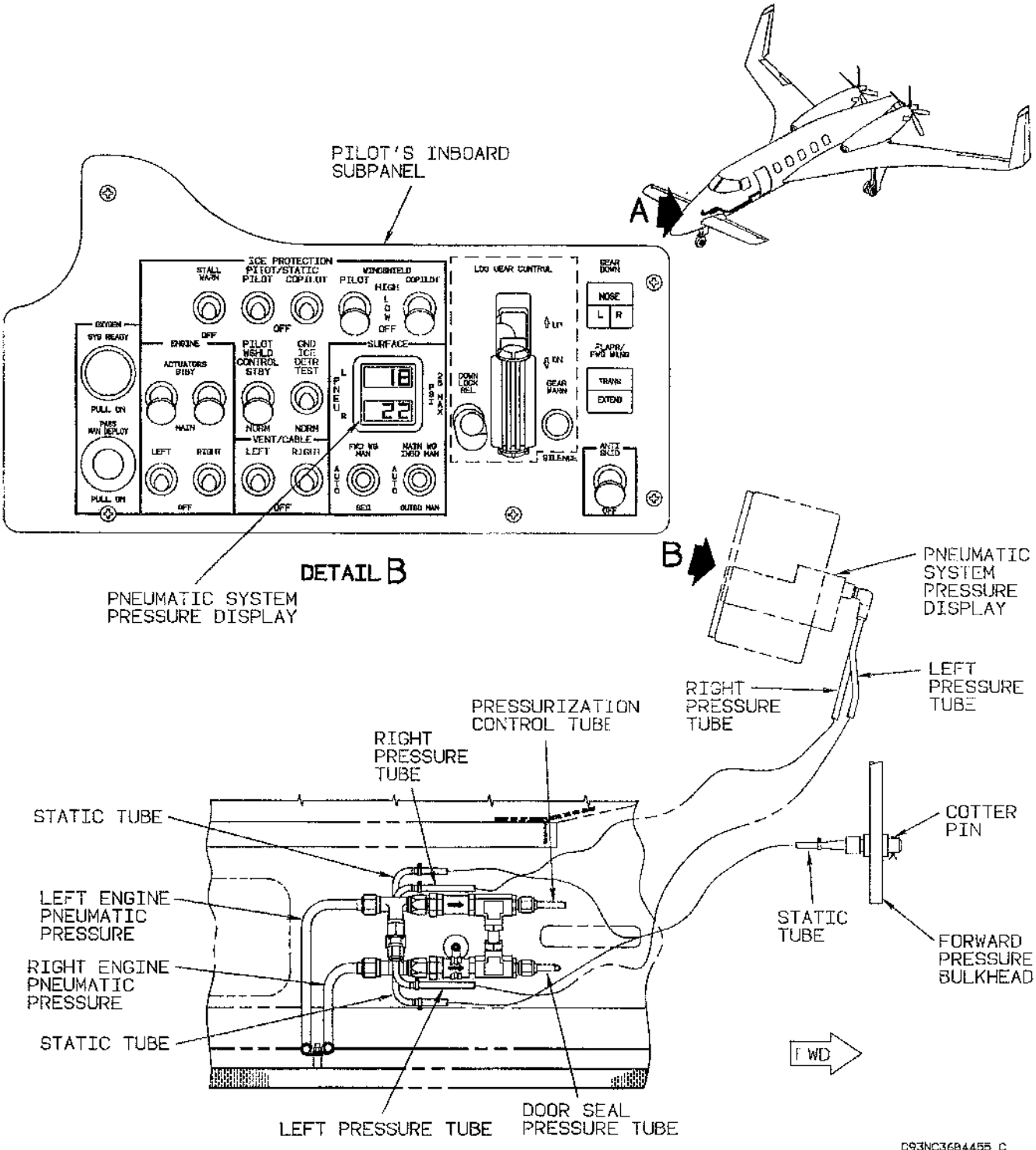
INDICATING - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The pneumatic system pressure display has a dual numeric display showing the individual pressure for each of the left and right pneumatic systems. The face of the pneumatic display is placarded PNEU, L-R, PSI 25 MAX and is located on the pilot's inboard subpanel. Pneumatic pressure is furnished through two tee-fittings at F.S. 127; one just before the cabin door seal tube check valve for the right side of the system and one just before the pressurization control tube check valve for the left side of the system. Both tee-fittings are located adjacent to the left keel under the flight compartment floor. The lower tee-fitting connects to the right pneumatic pressure display with the right pressure tube and the upper tee-fitting connects to the left pneumatic pressure display with the left pressure tube. Each pneumatic pressure tube is also connected to static pressure ports on the forward side of the forward pressure bulkhead to obtain a pressure reference.

The left and right pneumatic pressure displays receive 28 vdc operating voltage from the left and right generator buses through the circuit breakers placarded PNEU IND and LEFT-RIGHT on the circuit breaker panel in the pilot's console. If either side of the pneumatic system has a drop in pneumatic pressure below 10 ± 0.5 psi, a signal is sent to the respective Engine Data Concentrator (EDC). The EDC processes the information and causes the Engine Instrument, Crew Alerting System (EICAS) yellow PNEU PRESS LO caution message to be displayed on the EICAS display. This indicates that pneumatic pressure is too low to inflate the deice boots. The pneumatic pressure displays will indicate which side of the pneumatic system has low pressure. The display will blank out whenever the power source falls below 15 vdc. Data will be restored when adequate voltage is again available.



Pneumatic System Indicating
(Effectivity: All)
Figure 1

38 - WATER/WASTE

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Beechcraft
STARSHIP 1
MAINTENANCE MANUAL

CHAPTER 38 - WATER/WASTE

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

This chapter describes the components which provide for the storage, distribution, and removal of the water and waste. Chapter 38-30-00 describes the portion of the system used for the disposal of water and waste. This includes the toilet, relief tubes, and overboard drain system.

RECOMMENDED MATERIALS (Effectivity: All)

The recommended materials listed in Chart 1 as meeting federal, military or vendor specifications are provided for reference only and are not specifically recommended by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in this chart have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor or by compliance with the applicable specification. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included herein. Only the basic number of each product specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
RECOMMENDED MATERIALS
(Effectivity: All)

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
1. Sealant	MIL-S-8802	Proseal 890	Proseal Corporation, Division of Essex Chemical Corporation Aerospace Products Compton, CA 90220
2. Sealant	MIL-A-46146 Type I	RTV-162	General Electric Co. Hudson River Road Waterford, NY 12188

WASTE DISPOSAL - DESCRIPTION AND OPERATION (Effectivity: All)

FORWARD TOILET (Effectivity: NC-4 thru NC-20) (Figure 1)

The toilet is located in the forward right hand baggage cabinet. The cabinet is located on the right side across from the entrance door. The toilet is installed in the aft cabinet drawer. The tissue dispenser, toilet paper dispenser, relief tube, and general storage are located in the forward portion of this cabinet. The cabinet is part of an enclosure that provides privacy for the occupant. The aft cabinet door swings 90° to close off the lavatory area from the cabin. The front door slides along a track and pivots to close off the cockpit area. This tracking/pivoting motion is to insure there are no reflective surfaces facing into the cockpit area.

The toilet is a noncirculating flushing system completely self-contained with a waste container capacity of 2 gallons. The toilet is exposed by pulling out the drawer; the flushing sensor is activated and the toilet flapper valve will open. When the cover is repositioned, the flapper valve will close. The flushing reservoir pump will activate for approximately 4 seconds, drawing fluid from the reservoir and dispensing it over the bowl to deposit all waste products into the waste container.

The toilet consists of the bowl assembly which rests in the cabinet drawer. Attached to the bowl assembly is the toilet seat and bowl, the flushing reservoir, flushing sensor switch, flapper valve and gear motor. Inside the toilet is the removable waste container and flushing reservoir pump. An overboard drain tube is connected to toilet with an in-line check valve located at the back of the drawer. The toilet receives 28 vdc from the LEFT GEN BUS through the TOILET circuit breaker. The TOILET circuit breaker is rated at 5 amps.

AFT TOILET (Effectivity: NC-21 and after) (Figure 2)

The aft toilet is located on the right side of the aft cabin. The toilet is separated from the cabin by a partition with two sliding doors. The toilet/lavatory seat faces toward the aisle. Access to the toilet is gained by raising the hinged cabinet seat.

The toilet is a noncirculating flushing system completely self-contained with a waste container capacity of 2 gallons. When the hinged cabinet seat is repositioned, the flushing sensor is activated magnetically by a hall effect device located on the edge of the toilet seat cover which opens the toilet flapper valve. When the cabinet seat is repositioned to cover the toilet, the flapper valve will close. The flushing reservoir pump will activate for approximately 4 seconds to draw fluid from the reservoir and dispense it over the bowl for the removal of all waste products into the waste container.

The toilet consists of the bowl assembly which rests in the toilet cabinet. Attached to the bowl assembly is the toilet seat and bowl, the flushing reservoir, flushing sensor switch, the flapper valve and gear motor. Inside the toilet is a removeable waste container and flushing reservoir pump. The toilet receives 28 vdc from the LEFT GEN BUS through the TOILET circuit breaker. The TOILET circuit breaker is rated at 5 amps.

RELIEF TUBES (Effectivity: NC-4 thru NC-20) (Figure 1)

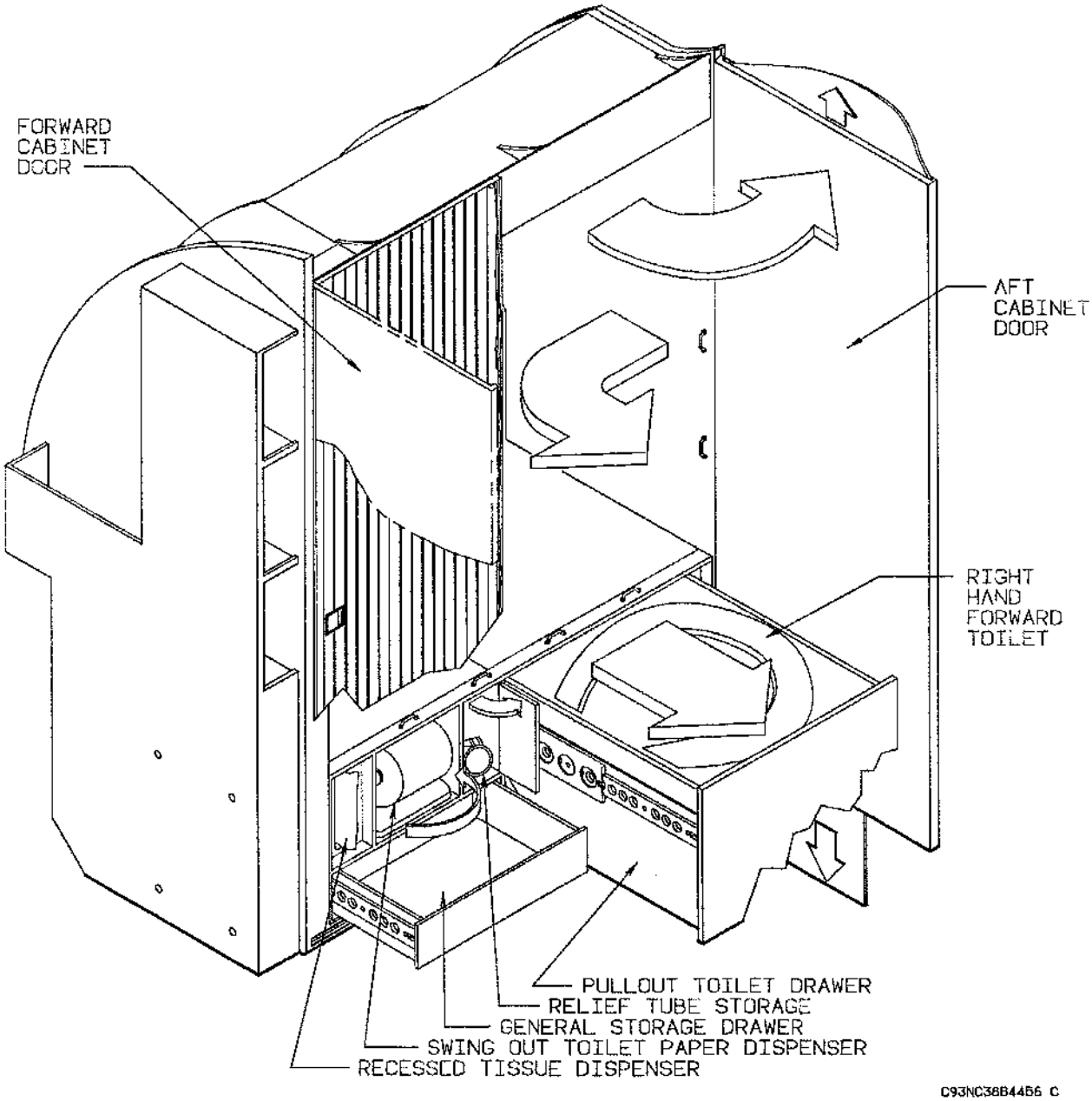
There are two relief tubes installed in the airplane. One relief tube is installed in the cockpit beneath the pilot's seat and the other next to the toilet in the forward baggage cabinet mounted on a retractable reel in the toilet module.

RELIEF TUBES (Effectivity: NC-21 and after) (Figure 2)

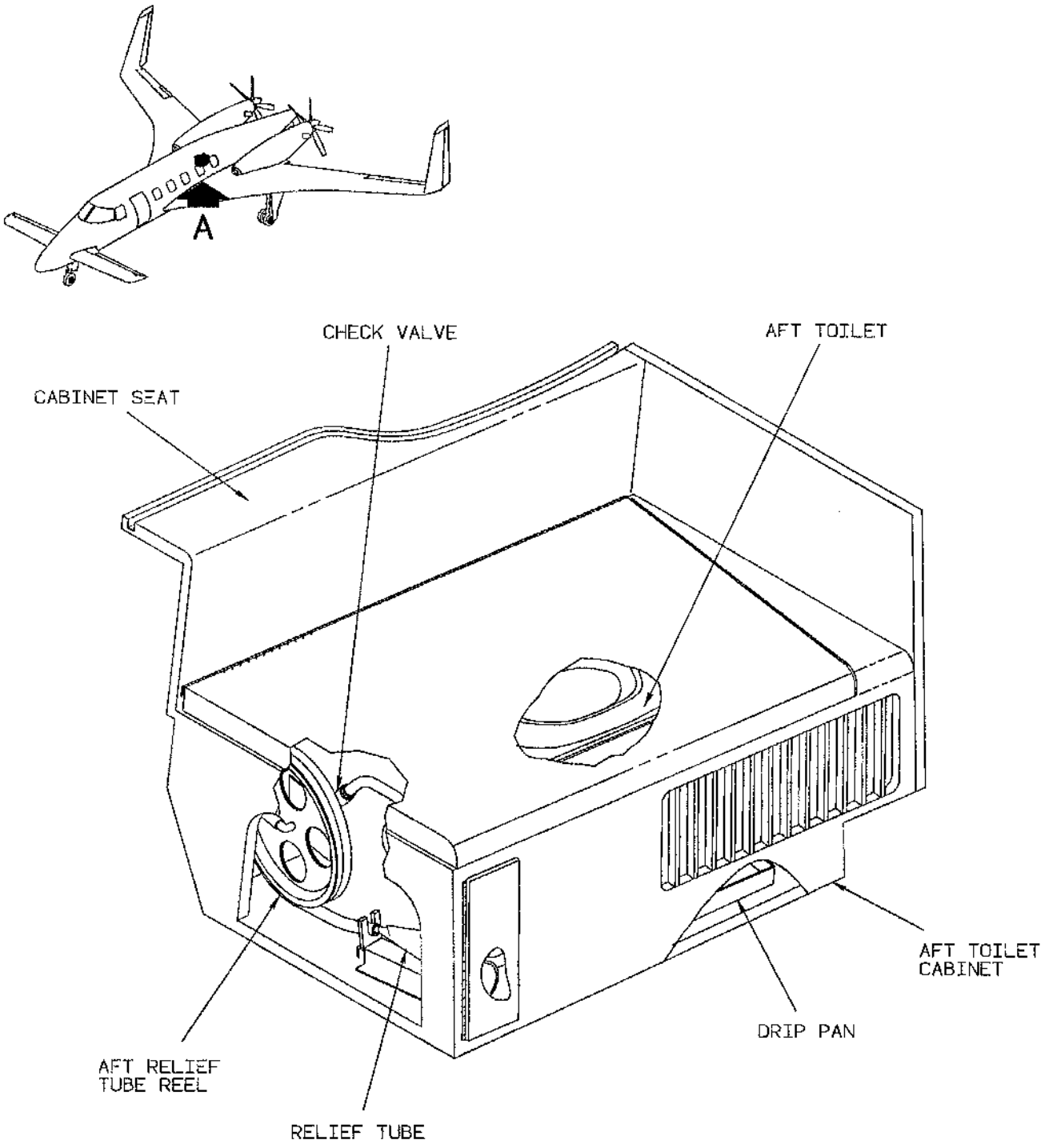
There are two relief tubes installed in the airplane. One relief tube is installed in the cockpit beneath the pilot's seat and the other is installed in the aft toilet cabinet mounted on a retractable reel.

LEFT FORWARD REFRESHMENT CABINET (Effectivity: All) (Figure 3)

The left forward refreshment cabinet is located aft of the cabin door. The cabinet contains a drain for the dispenser drip pan and a drain for the ice container. A pushbutton drain valve is located in the drip pan to allow the drip pan to be drained overboard. The dispenser drip pan is plumbed into the overboard drain system below the solenoid shutoff valve to allow the drip pan to be drained. A solenoid shutoff valve is

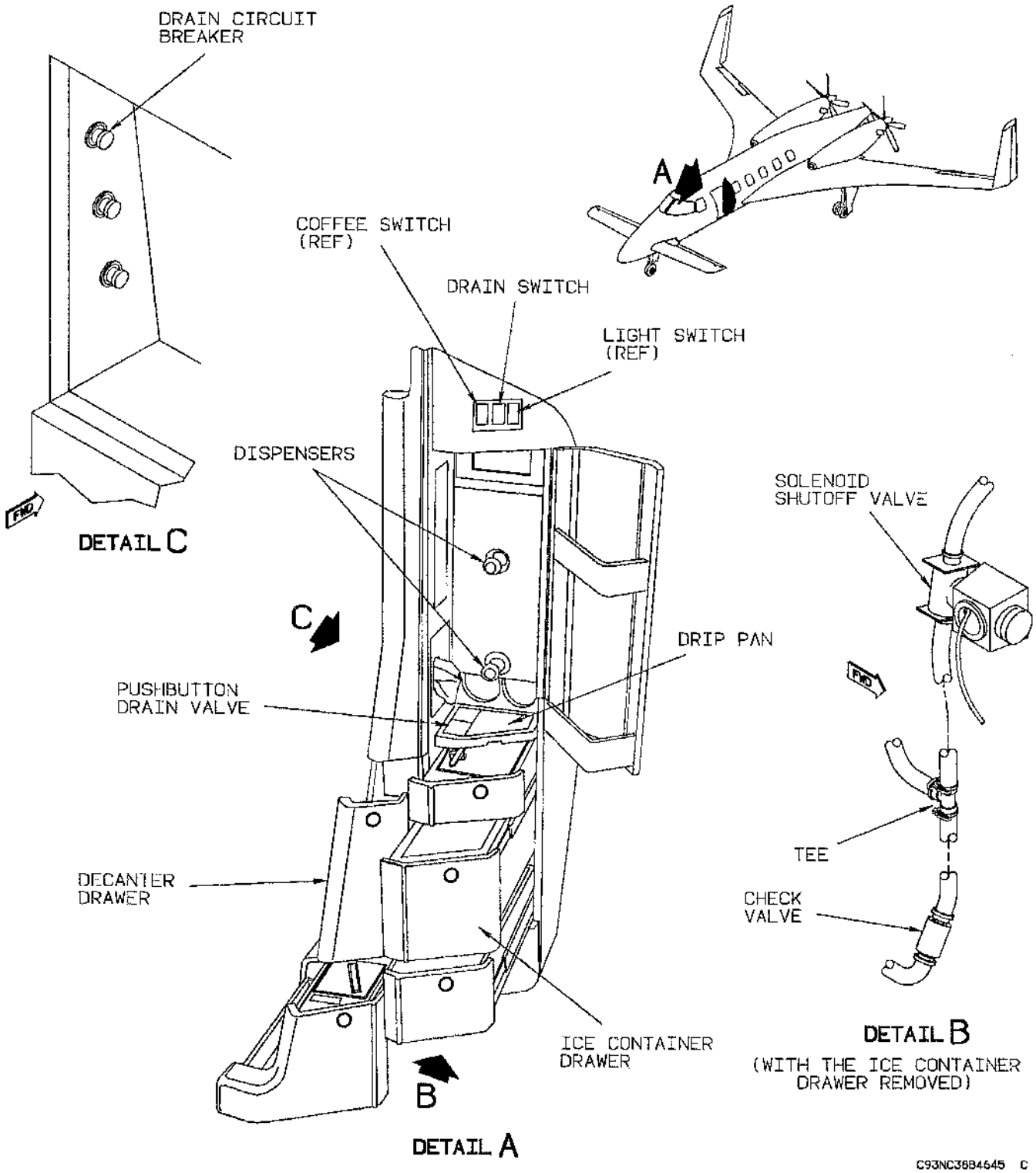


Forward Toilet Locations
(Effectivity: NC-4 thru NC-20)
Figure 1



DETAIL A

Aft Toilet Locations
(Effectivity: NC-21 and after)
Figure 2



Left Forward Refreshment Cabinet Locations
(Effectivity: All)
Figure 3

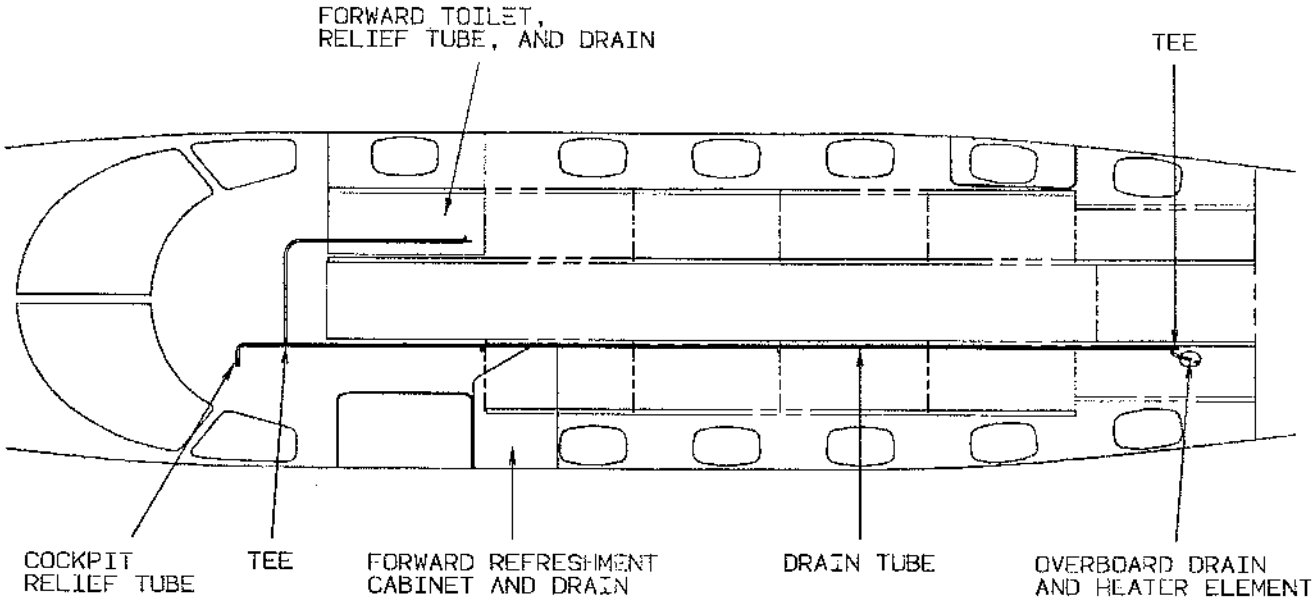
installed on the ice container drain line. On airplanes NC-4 thru NC-27, this shutoff valve is a normally closed-type valve (note Figure 7). When power is applied and the BAR switch is selected, the valve will open to allow the drip pan to drain. On airplanes NC-28 and after, the shutoff valve is a normally open-type valve (note Figure 8). This allows the ice container to drain when the airplane is on the ground. When power is applied and the BAR switch is selected, the valve will close to prevent water from being discharged from the overboard drain. A drain switch on the top of the cabinet allows the drain valve to be opened. The drain valve receives power through the DRAIN circuit breaker. The one-amp circuit breaker is located in the aft compartment of the refreshment cabinet above the decanter drawer. The three refreshment-cabinet circuit breakers are tied to the circuit breaker bus bar and receive power from the L REFRESHMENT BAR circuit breaker located on the right circuit breaker panel. Power is controlled by the BAR switch (S104) on the overhead control panel which energizes the BUS BAR relay. Refer to the LEFT FORWARD REFRESHMENT CABINET schematic diagrams in Figures 7 and 8.

OVERBOARD PLUMBING AND DRAIN (Effectivity: All) (Figures 4 and 5)

The overboard plumbing and drain is provided to drain fluid from the refreshment cabinet, relief tubes, and vent the toilet. A check valve is installed to keep fluid from backing up into the toilet in the event of a clogged or pinched drain line. A heater element is installed on the end of the overboard drain line to keep the drain line from freezing. The element is part of the airplane's ice and rain protection system. Refer to Chapter 30-70-00 for information regarding the overboard drain heater element.

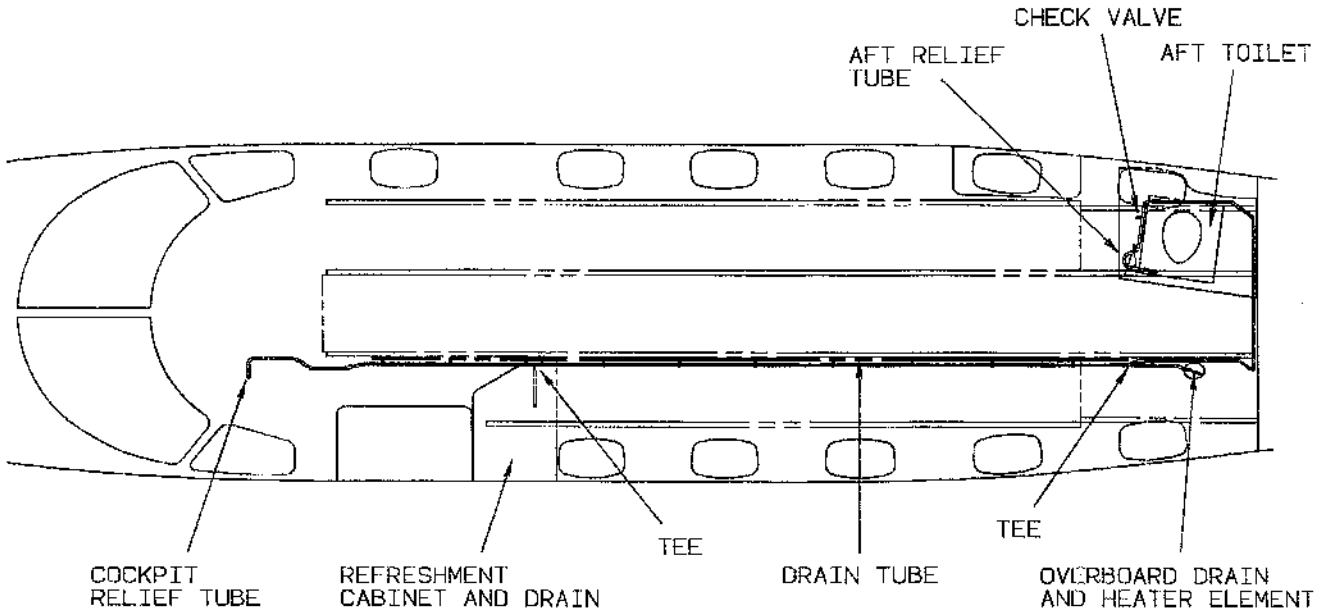
WASTE DISPOSAL - TROUBLESHOOTING (Effectivity: All)

Refer to Chart 1 to troubleshoot the toilet. Refer to Figure 6 for a schematic diagram of the forward and aft cabin toilet. Refer to Figures 7 and 8 for schematic



Overboard Plumbing and Drain Locations (Effectivity: NC-4 thru NC-20) Figure 4

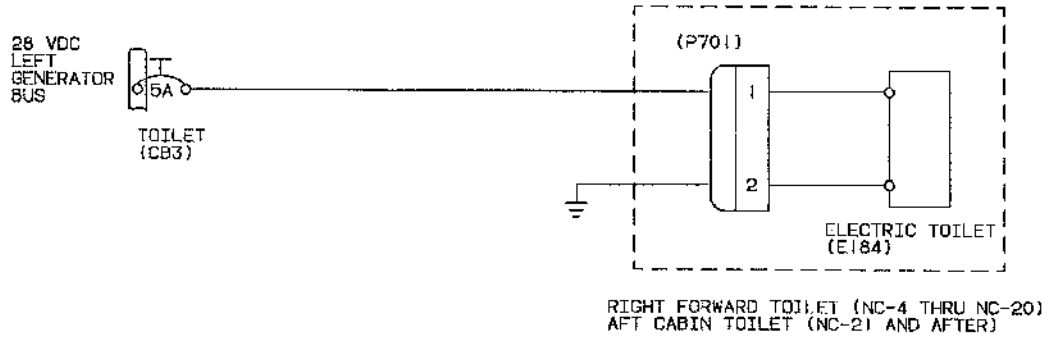
diagrams of the left forward refreshment cabinet. When troubleshooting problems with the overboard drain system, the transparent drain lines can be visually inspected to trace the flow of liquid through the system.



Overboard Plumbing and Drain Locations
(Effectivity: NC-21 and after)
Figure 5

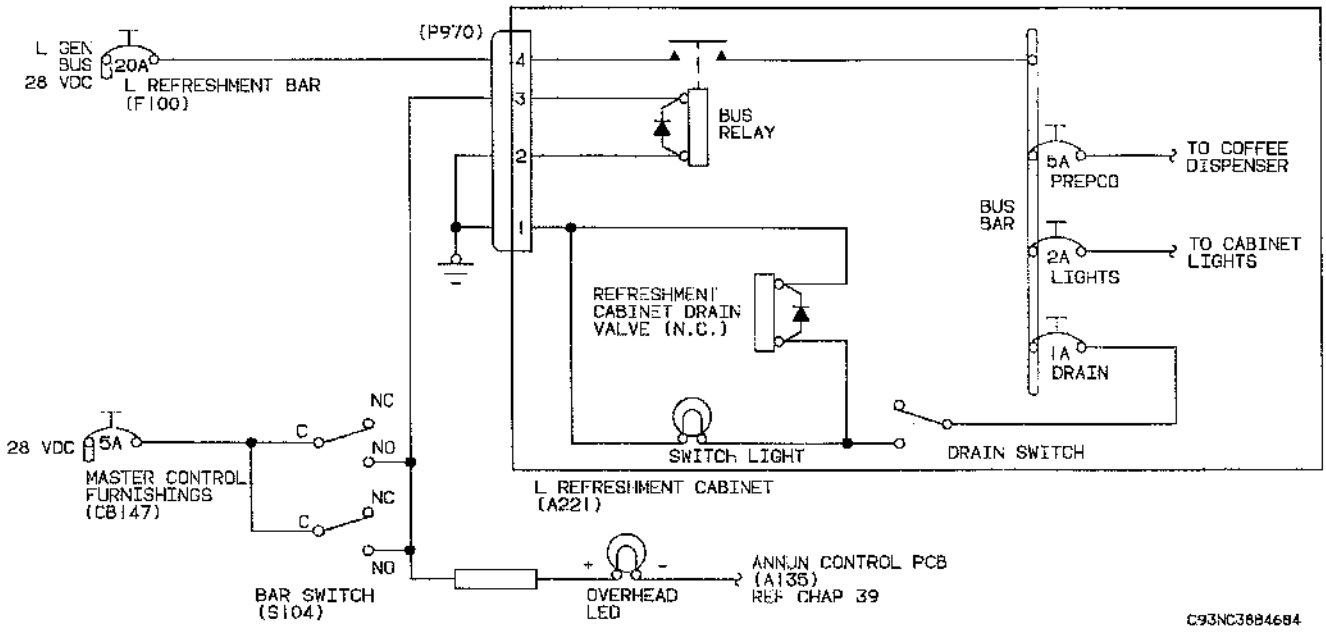
CHART 1
TROUBLESHOOTING - TOILET (Effectivity: All)

PROBLEM	PROBABLE CAUSE	CORRECTIVE ACTION
1. Toilet will not flush.	a. Circuit breaker tripped or blown fuse.	a. Check toilet electrical system for shorted circuits, defective circuit breaker or blown fuse. Repair or replace components as required.
	b. Inadequate flush fluid charge.	b. Recharge as required.
	c. Pump failed.	c. Repair or replace pump as required.
	d. Check timer.	d. Repair or replace as required.
2. Toilet flushes continuously.	a. Flush switch malfunctions.	a. Replace switch.
	b. Check timer on toilet.	b. Repair or replace as required.
3. Excessive noise and/or current drain.	a. Pump failed.	a. Repair or replace pump as required.
4. Offensive odor.	a. Inadequate or lack of chemical mix in flush fluid charge, or excessive usage between service periods.	a. Empty waste container and reservice flush fluid with correct mix.

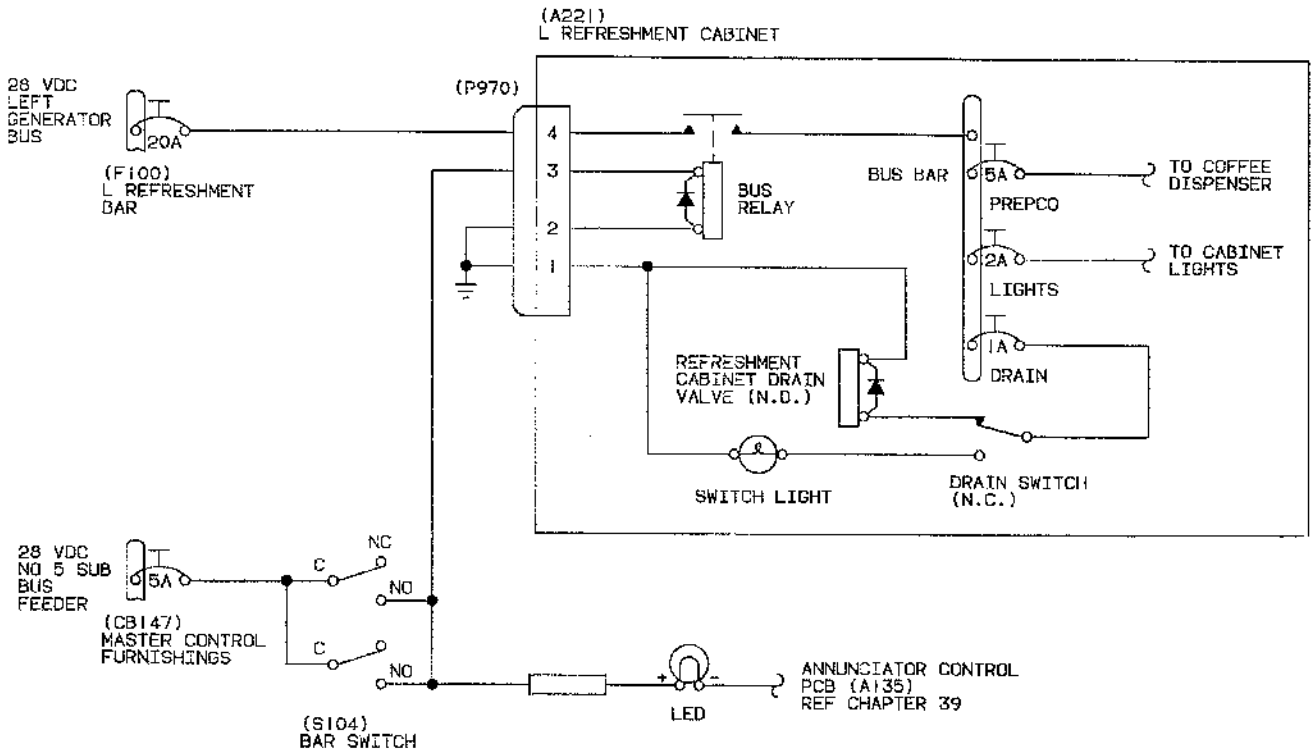


C93NC38B4467 C

Forward and Aft Cabin Toilet Schematic Diagram
(Effectivity: All)
Figure 6



Left Forward Refreshment Cabinet Schematic Diagram
(Effectivity: NC-4 thru NC-27)
Figure 7



C93NC38B4644 C

Left Forward Refreshment Cabinet Schematic Diagram
(Effectivity: NC-28 and after)
Figure 8

WASTE DISPOSAL - MAINTENANCE PRACTICES

SERVICING THE TOILET (Effectivity: All)

Refer to Chapter 12-10-00 for servicing information for all toilets which may be used in the airplane.

FORWARD TOILET REMOVAL (Effectivity: NC-4 thru NC-20) (Figure 9)

a. Open the cabinet doors.

b. Pull out the toilet drawer (9) to gain access to the toilet (6).

c. Raise the hinged toilet seat (1) upright.

d. Insert the snap cap (3) into the waste container (2) and remove the waste container (2).

e. Remove the four cabinet retaining screws (4).

f. Lift the toilet (6) out of the toilet drawer (9).

g. Twist the jaws in opposite directions to open the clamp (7) and disconnect the vent line tube (8) from the toilet (6).

h. Disconnect the electrical connector (5) and remove the toilet (6) from the toilet drawer (9).

FORWARD TOILET INSTALLATION (Effectivity: NC-4 thru NC-20) (Figure 9)

a. Connect the electrical connector (5).

b. Connect the vent line tube (8) to the toilet (6) and install the clamp (7) by pressing the jaws together.

c. Position the toilet (6) into the toilet drawer (9).

d. Install the four cabinet retaining screws (4).

e. Raise the hinged toilet seat (1), install the waste container (2), and remove the snap cap (3) and stow inside the toilet (6).

f. Lower the toilet seat (1) and close the toilet drawer (9).

g. Close the cabinet doors.

AFT TOILET REMOVAL (Effectivity: NC-21 and after) (Figure 10)

a. Lift the cabinet seat (3) to gain access to the toilet (6).

b. Raise the hinged toilet seat (4) upright.

c. Insert the snap cap (2) into the waste container (10) and remove the waste container (10).

d. Remove the four cabinet retaining screws (1).

e. Lift the toilet (6) out of the cabinet assembly (9).

f. Twist the jaws in opposite directions to open the clamp (7) and disconnect the vent line tube (8) from the toilet (6).

g. Disconnect the electrical connector (5).

AFT TOILET INSTALLATION (Effectivity: NC-21 and after) (Figure 10)

a. Connect the electrical connector (5).

b. Connect the vent line tube (8) to the toilet (6) and install the clamp (7) by pressing the jaws together.

c. Position the toilet (6) in the cabinet assembly (9) and install the four screws (1).

d. Install the waste container (10), remove the snap cap (2) and stow inside the toilet (6).

e. Close the hinged toilet seat (4).

f. Close the cabinet seat (3).

RELIEF TUBE REEL ASSEMBLY REMOVAL (Effectivity: NC-4 thru NC-20) (Figure 11)

a. Open the forward baggage compartment doors.

b. Remove the bottom storage drawer (6) from the toilet module (9).

c. Loosen the clamp (7) and disconnect the drain line tube (8) from the drain line tee (5).

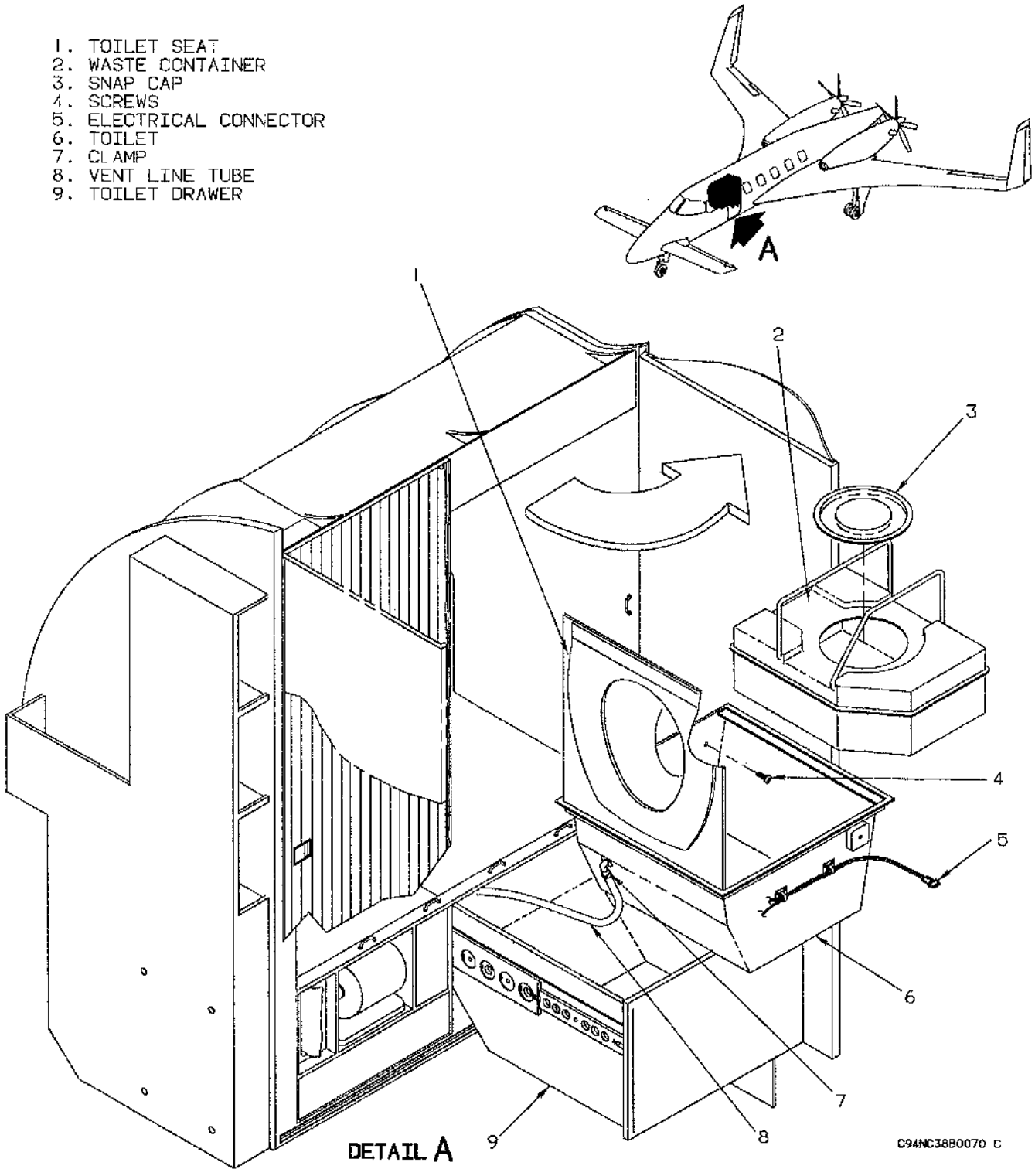
d. Remove the toilet module (9) from the forward baggage cabinet.

CAUTION

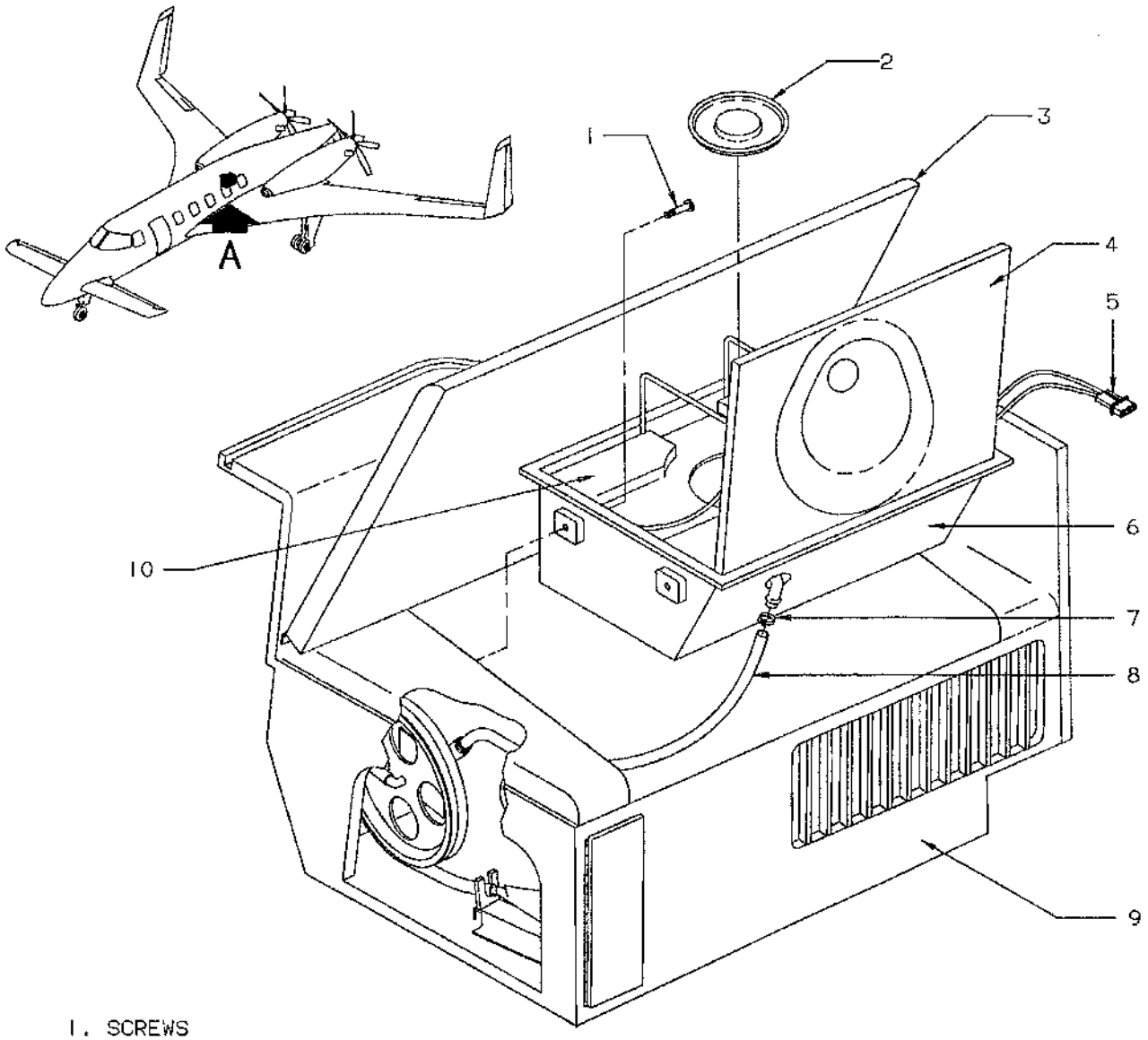
The relief tube reel is under tension from a spring-loaded retraction mechanism.

e. Hold the relief tube reel (1), remove the clamp (3) and disconnect the drain tube (2) from the relief tube horn (4).

f. Slowly release the relief tube reel assembly (1) to remove the tension from the spring.



Forward Toilet Installation
(Effectivity: NC-4 thru NC-20)
Figure 9



1. SCREWS
2. SNAP CAP
3. CABINET SEAT
4. TOILET SEAT
5. ELECTRICAL CONNECTOR
6. TOILET
7. CLAMP
8. VENT LINE TUBE
9. CABINET ASSEMBLY
10. WASTE CONTAINER

DETAIL A

C94NC38B0071 C

Aft Toilet Installation
(Effectivity: NC-21 and after)
Figure 10

g. Remove three screws (10) and the reel assembly (1) from the toilet module (9).

RELIEF TUBE REEL ASSEMBLY INSTALLATION (Effectivity: NC-4 thru NC-20) (Figure 11)

a. Position the relief tube reel assembly (1) in the toilet module (9) and install three screws (10).

b. Route the drain tube (2) through the access hole to the front of the cabinet, connect the relief tube horn (4) to the tube (2) and tighten the clamp (3).

c. Install the toilet module (9) in the forward baggage cabinet.

d. Connect the drain tube (8) to the drain line tee (5) and tighten the clamp (7).

e. Install the bottom storage drawer (6) in the toilet module (9).

f. Close the baggage compartment doors.

AFT RELIEF TUBE REEL ASSEMBLY REMOVAL (Effectivity: NC-21 and after) (Figure 12)

a. Remove the right aft partition.

b. Refer to AFT CABIN TOILET REMOVAL and remove the aft cabin toilet.

c. Open the relief tube cabinet door (2).

CAUTION

The relief tube reel assembly is under tension from a spring-loaded retraction mechanism.

d. Hold the relief tube reel assembly (6), loosen the clamp (4), and disconnect the relief tube (5) from the relief tube horn (3).

e. Slowly release the relief tube reel assembly (6) to remove the tension from the spring.

f. Twist and remove the clamp (9) from the drain tube (8) and disconnect the drain tube (8) from the tee (10).

g. Remove the three screws (7) from the forward side of the toilet cabinet (1) and remove the reel assembly (6) from the toilet cabinet (1).

AFT RELIEF TUBE REEL ASSEMBLY INSTALLATION (Effectivity: NC-21 and after) (Figure 12)

a. Position the relief tube reel assembly (6) inside the toilet cabinet (1) and install the three screws (7) through the forward side of the cabinet (1).

b. Connect the drain tube (8) to the tee (10) and tighten the clamp (9).

c. Pull the end of the relief tube (5) out through the relief tube cabinet door (2), connect the relief tube horn (3) to the tube (5) and tighten the clamp (4).

d. Close the relief tube cabinet door (2).

e. Refer to the AFT CABIN TOILET INSTALLATION and install the aft cabin toilet.

f. Install the aft right partition.

REFRESHMENT CABINET DRAIN VALVE REMOVAL (Effectivity: All) (Figure 13)

a. Open the cabinet door (2) and open the DRAIN circuit breaker (1).

b. Remove the ice container drawer (4) and the forward lower cabinet drawer (5) from the refreshment cabinet (3).

c. Disconnect the electrical connector (10) from the drain valve (9).

d. Remove the two clamps (7 and 12) from the the drain line tubes (6 and 11).

e. Disconnect the drain line tubes (6 and 11) from the fittings (8 and 13).

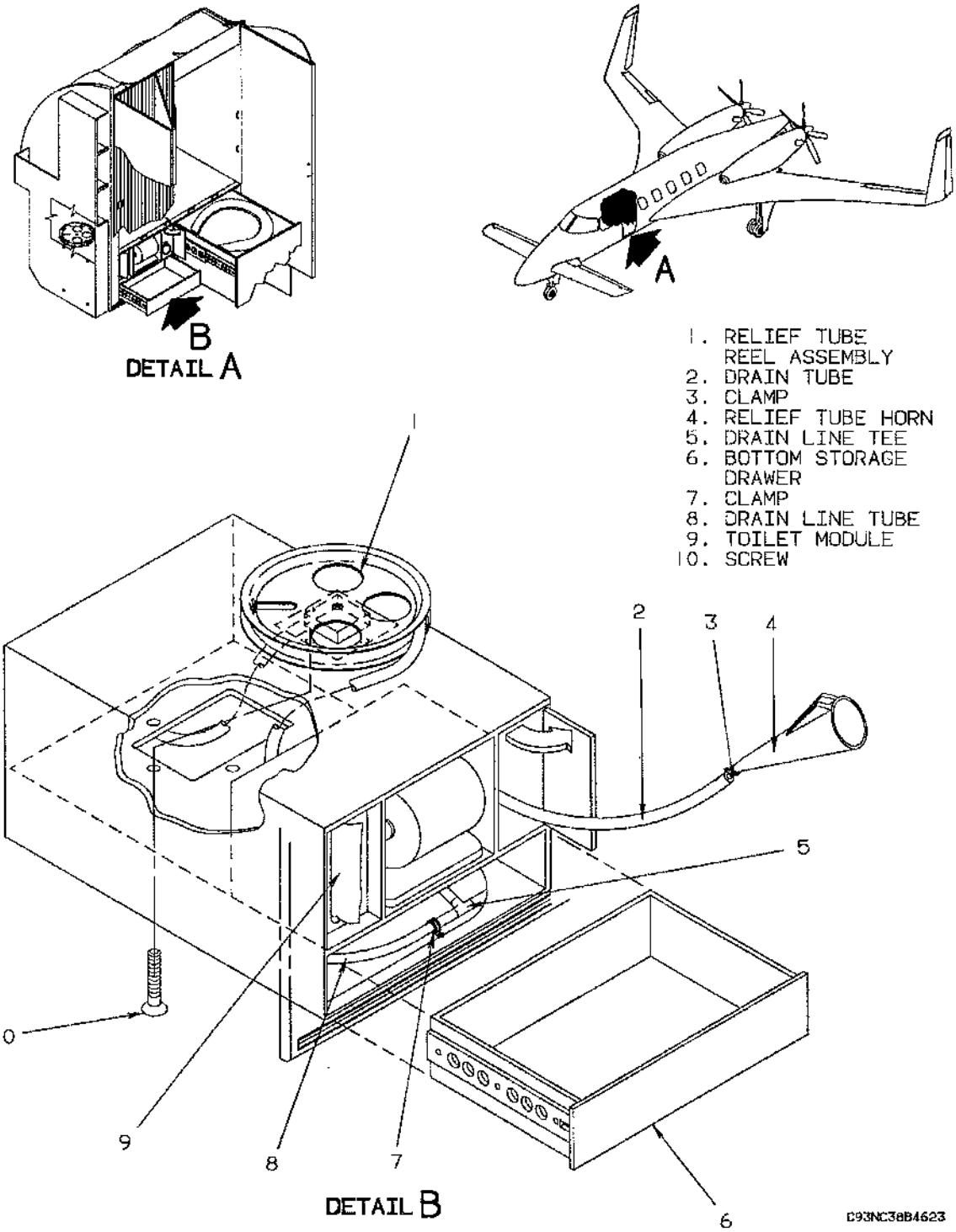
f. Remove the fittings (8 and 13) from the drain valve (9) and remove the valve (9) from the bracket (14).

REFRESHMENT CABINET DRAIN VALVE INSTALLATION (Effectivity: All) (Figure 13)

a. Position the drain valve (9) in the bracket (14) and install the two fittings (8 and 13) in the drain valve (9).

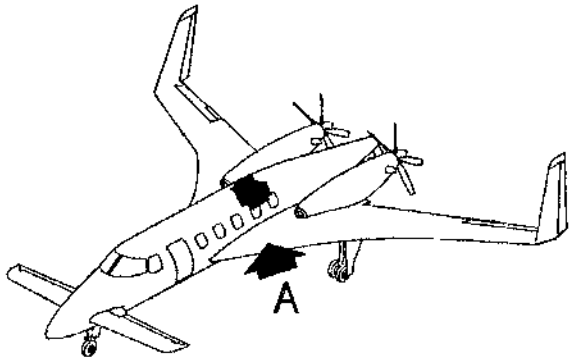
b. Connect the drain line tubes (6 and 11) to the fittings (8 and 13) and install the two clamps (7 and 12).

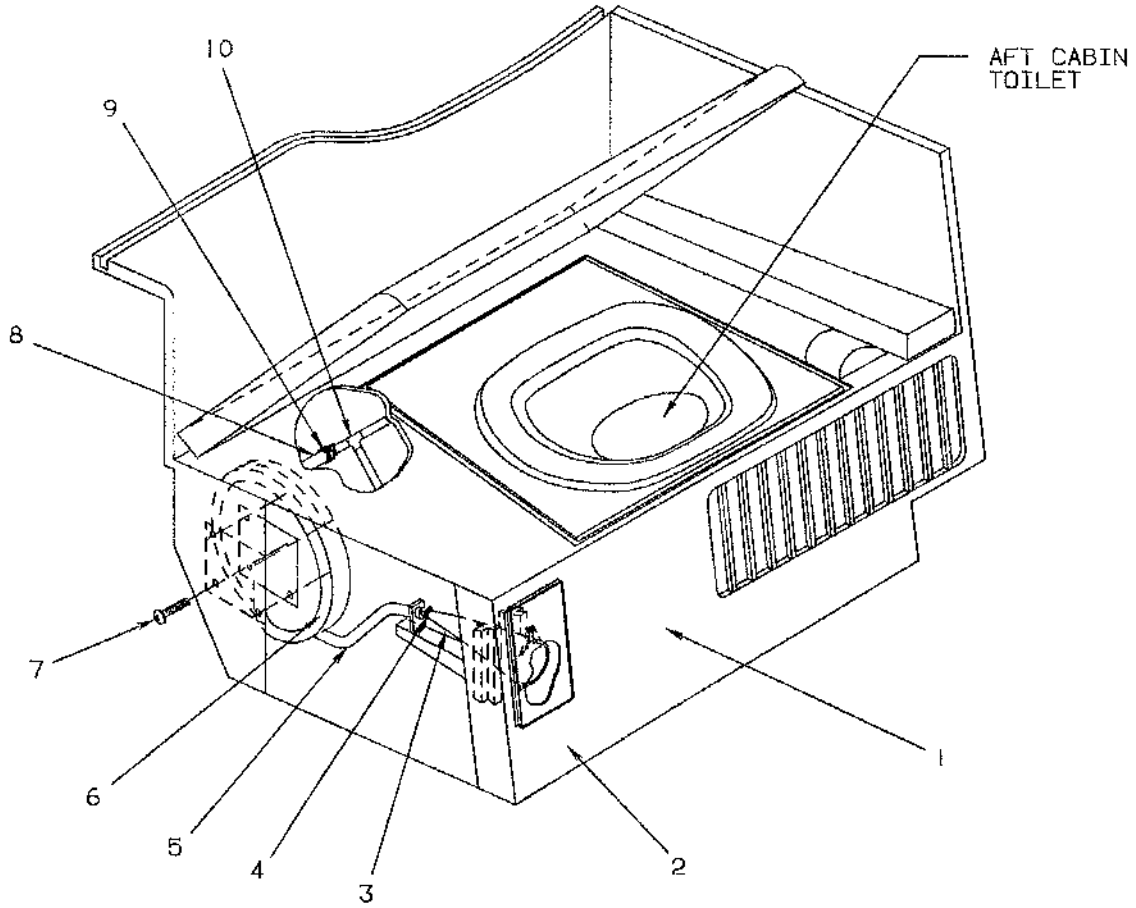
c. Connect the electrical connector (10) to the plug.



Relief Tube Reel Assembly Installation
(Effectivity: NC-4 thru NC-20)
Figure 11

1. TOILET CABINET
2. RELIEF TUBE CABINET DOOR
3. RELIEF TUBE HORN
4. CLAMP
5. RELIEF TUBE
6. RELIEF TUBE REEL ASSEMBLY
7. SCREW
8. DRAIN TUBE
9. CLAMP
10. TEE

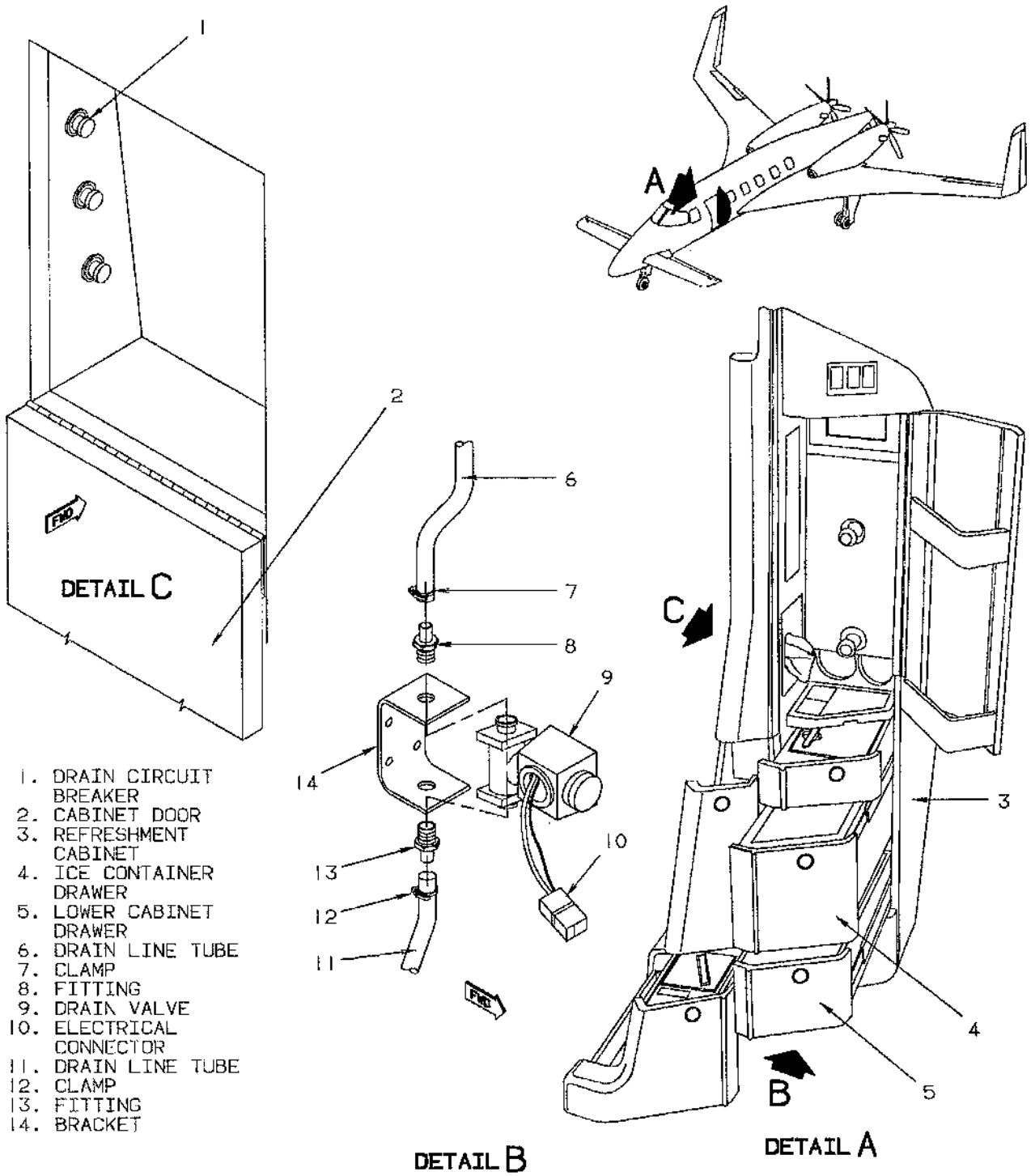




DETAIL A

C93NC38B4622

Aft Relief Tube Reel Assembly Installation
(Effectivity: NC-21 and after)
Figure 12



Refreshment Cabinet Drain Valve Installation
(Effectivity: All)
Figure 13

d. Close the DRAIN circuit breaker (1) and close the cabinet door (2).

e. Apply power to the airplane, engage the BAR switch on the overhead control panel, and make sure the valve opens (NC-4 thru NC-27), or closes (NC-28 and after).

f. Fill the ice container drawer (4) with enough water to check the valve fittings (8 and 13) for leaks and proper operation.

OVERBOARD DRAIN REMOVAL
(Effectivity: All)
(Figure 14)

a. Refer to Chaper 25-20-00 and remove the carpet from the floorboard (171BT).

b. Refer to Chapter 53-20-00 and remove the left aft floorboard (171BT).

c. Loosen the clamp (5) and disconnect the drain tube (6) from the overboard drain (7).

d. Disconnect the electrical connector (1) to the heater element (4).

e. Remove the sealant between the overboard drain (7) and the shield assembly (3).

f. Remove the six Hi-lok fasteners (9) by holding the collars with a wrench, inserting the a hex head wrench in the end of the threaded pins (8), and unscrewing the threaded pins (8) from the collars.

g. Remove the drain assembly from inside the fuselage (2) and the shield assembly (3) from outside the fuselage (2).

h. Refer to Chapter 30-70-00 and remove the heater element (4) from the overboard drain (7).

OVERBOARD DRAIN INSTALLATION
(Effectivity: All)
(Figure 14)

a. Refer to Chapter 30-70-00 and install the heater element (4) on the overboard drain (7).

b. Position the overboard drain (7) inside the fuselage (2) and the shield assembly (3) on the outside of the fuselage (2).

c. Install the six Hi-lok fasteners (9) by inserting the threaded pins (8) through the shield assembly (3), fuselage (2), and overboard drain (7), then install the collars on the threaded pins (8).

d. Seal around the inside and outside perimeter of the plates and Hi-lok fasteners (9) with sealant (1, Chart 1, 38-00-00).

e. Fill the gap between the shield assembly (3) and the overboard drain (7) with sealant (2, Chart 1, 38-00-00) to a depth of less than .25 inch.

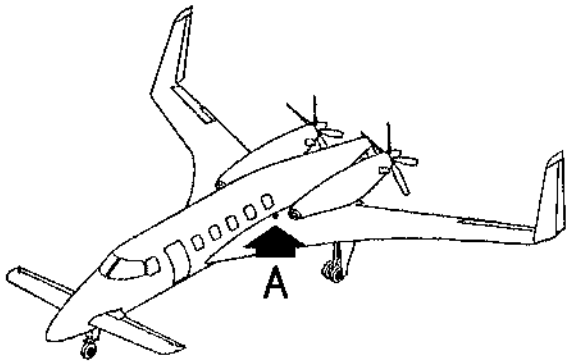
f. Connect the electrical connector (1).

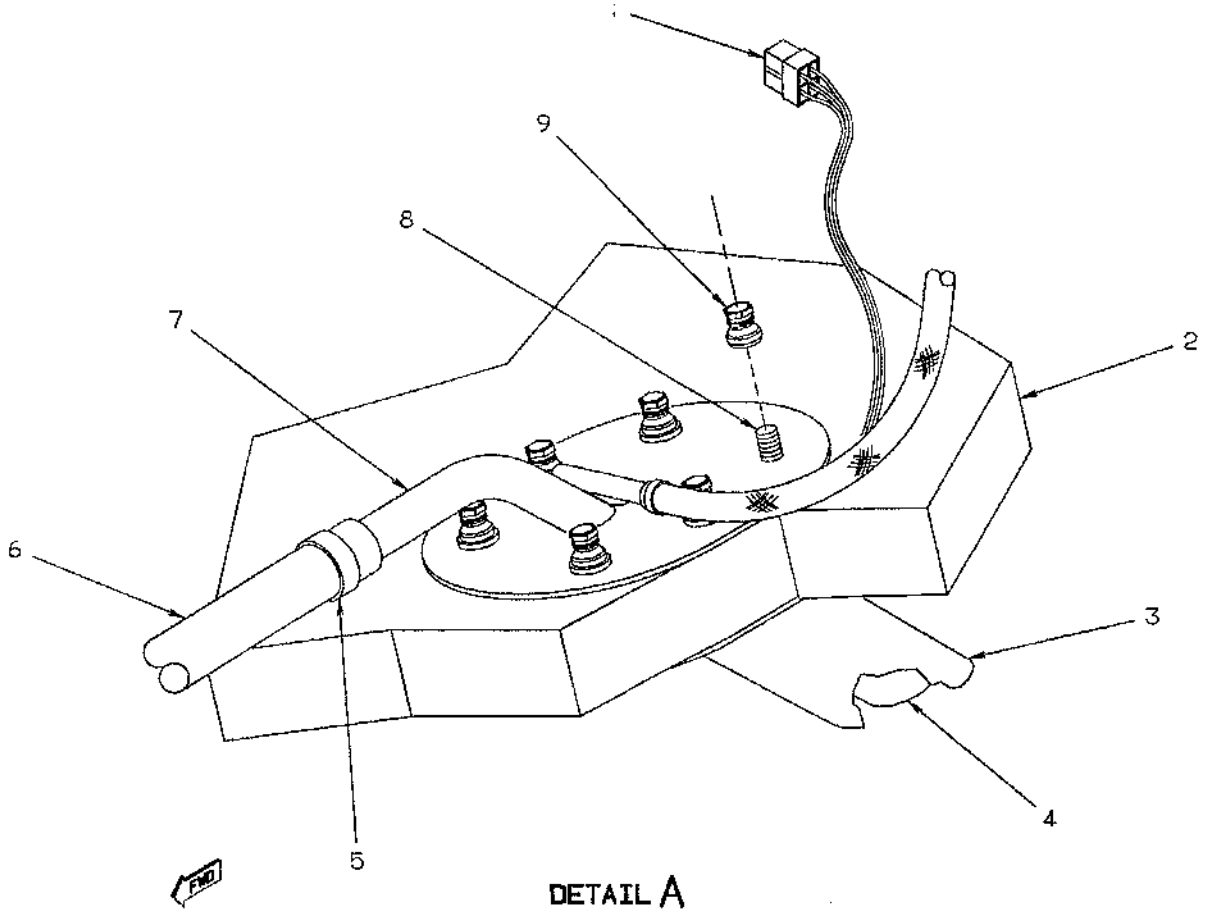
g. Connect the drain tube (6) to the overboard drain (7) and install the clamp (5) by pressing the jaws together.

h. Refer to Chapter 53-20-00 and install the left aft floorboard (171BT).

i. Refer to Chapter 25-20-00 and install the carpet on the floorboard.

1. ELECTRICAL CONNECTOR
2. FUSELAGE
3. SHIELD ASSEMBLY
4. HEATER ELEMENT
5. CLAMP
6. DRAIN TUBE
7. OVERBOARD DRAIN
8. THREADED PINS
9. HI-LOK FASTENER





DETAIL A

C93NC38B4561 C

Overboard Drain Installation
(Effectivity: All)
Figure 14

CHAPTER 39 - ELECTRICAL PANELS & MULTIPURPOSE PARTS

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CHAPTER 39 - ELECTRICAL PANELS & MULTIPURPOSE PARTS

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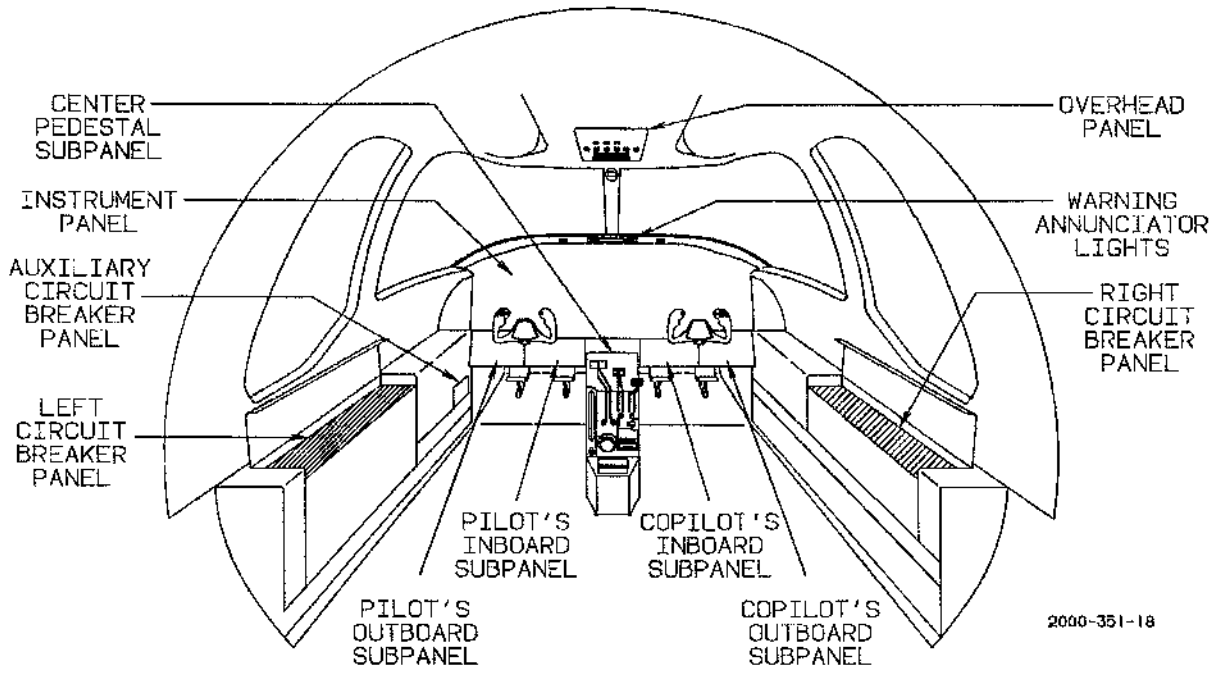
INSTRUMENT AND CONTROL PANELS - DESCRIPTION AND OPERATION (Effectivity: All)

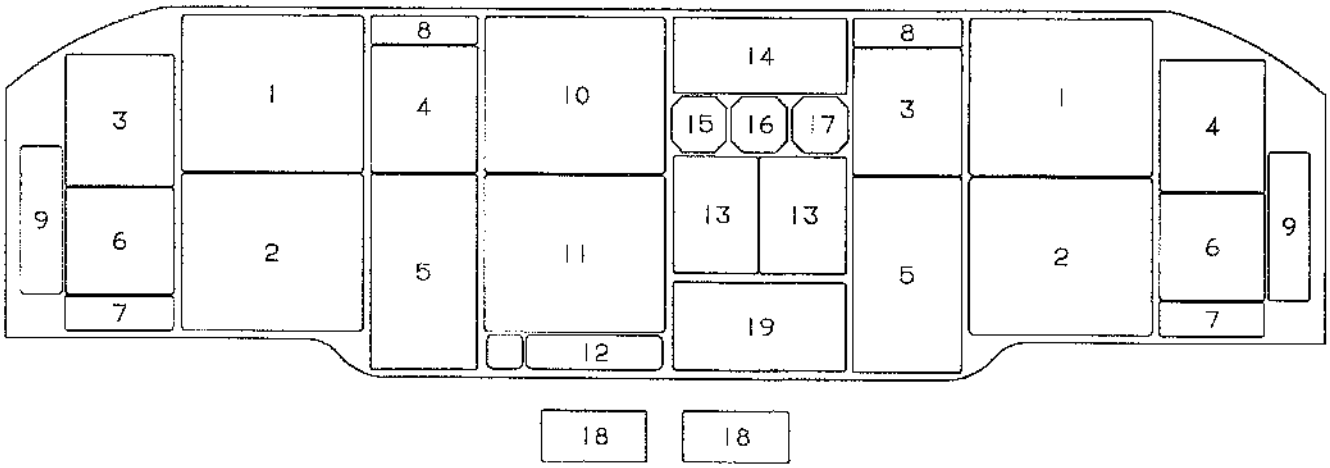
WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

This chapter, through the use of illustrations, locates and identifies the electrical equipment installed on the instrument and control panels in the cockpit. Figure 1 identifies the panels covered in this chapter. The remaining illustrations locate and identify the various components installed on the panels. The panels are located by the airplane zone numbers; refer to Chapter 6-40-00 for the locations of the zones. The components are identified by their reference designator numbers; these same reference designators are used in the BEECHCRAFT Starship 1 Wiring Diagram Manual and the avionics wiring diagrams.



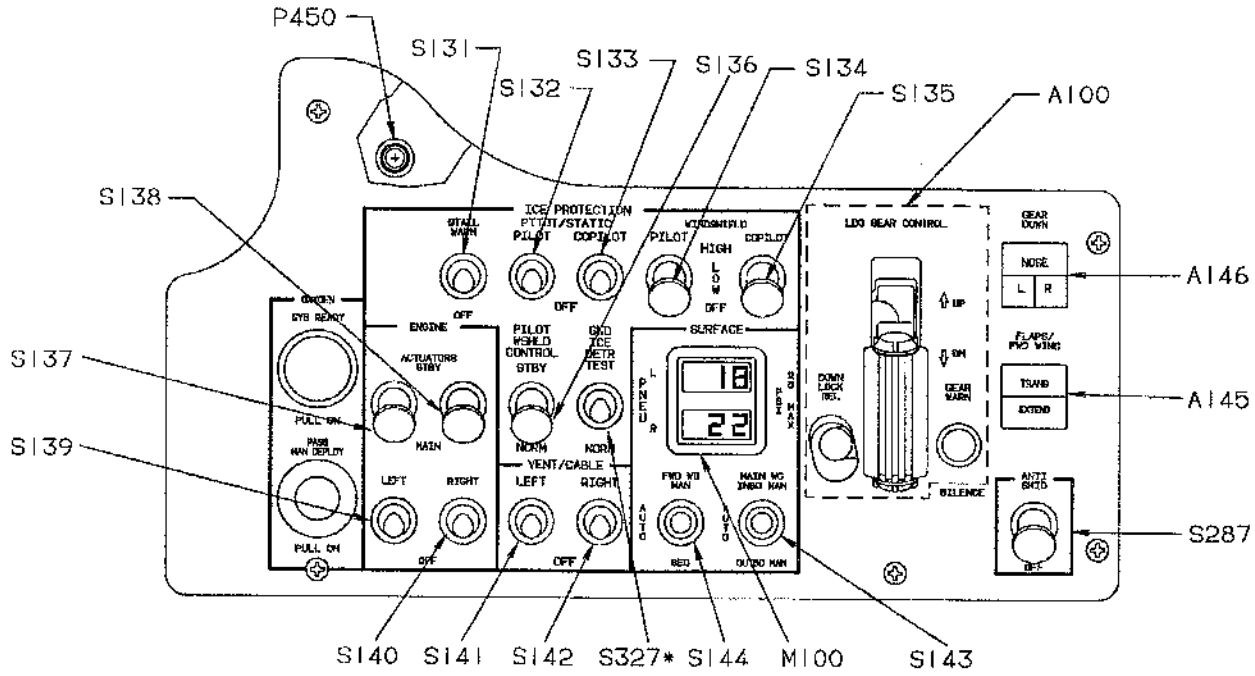
Instrument and Control Panels (Effectivity: All)
Figure 1



1. PRIMARY FLIGHT DISPLAYS (PFD)
2. NAVIGATION DISPLAYS (ND)
3. AIRSPEED INDICATORS (ASI)
4. ALTITUDE INDICATORS (ALI)
5. CONTROL DISPLAY UNITS (CDU)
6. SENSOR DISPLAY UNITS (SDU)
7. CLOCKS
8. ALTITUDE AWARENESS PANELS (AAP)
9. REVERSIONARY PANELS
10. ENGINE INSTRUMENT, CREW ALERTING SYSTEM (EICAS) DISPLAY
11. MULTIFUNCTION DISPLAY (MFD)
12. COURSE HEADING PANEL (MFD)
13. RADIO TUNING UNIT (RTU)
14. CENTER REVERSIONARY PANEL
15. STANDBY AIRSPEED INDICATOR
16. STANDBY ATTITUDE INDICATOR
17. STANDBY ALTIMETER
18. MODE SELECT PANEL (MSP)
19. PRESSURIZATION CONTROL PANEL

2000-392-016

Instrument Panel (Zone 256) (Effectivity: All)
Figure 2



A100 - LANDING GEAR CONTROL SWITCH ASSEMBLY
DS1 - LANDING GEAR IN-TRANSIT LIGHT
DS2 - LANDING GEAR IN-TRANSIT LIGHT

A145 - FLAP/FORWARD WING POSITION INDICATOR

A146 - LANDING GEAR POSITION INDICATOR
CR12 - DIODE
CR13 - DIODE
CR14 - DIODE

M100 - PNEUMATIC PRESSURE INDICATOR

S131 - ANGLE-OF-ATTACK TRANSMITTER HEAT SWITCH

S132 - PILOTS'S PITOT/STATIC TUBE HEAT SWITCH

S133 - COPILOTS'S PITOT/STATIC TUBE HEAT SWITCH

S134 - PILOT'S WINDSHIELD ANTI-ICE SWITCH

S135 - COPILOT'S WINDSHIELD ANTI-ICE SWITCH

S136 - PILOT'S WINDSHIELD REDUNDANCY ANTI-ICE SWITCH

S137 - LH ENGINE ANTI-ICE POWER SELECTION SWITCH

S138 - RH ENGINE ANTI-ICE POWER SELECTION SWITCH

S139 - LH ENGINE ANTI-ICE CONTROL SWITCH

S140 - RH ENGINE ANTI-ICE CONTROL SWITCH

S141 - LH FUEL VENT/POWER CABLE HEAT SWITCH

S142 - RH FUEL VENT/POWER CABLE HEAT SWITCH

S143 - AFT WING DEICE SWITCH

S144 - FORWARD WING SURFACE DEICE SWITCH

S287 - ANTI-SKID BRAKE SYSTEM SWITCH

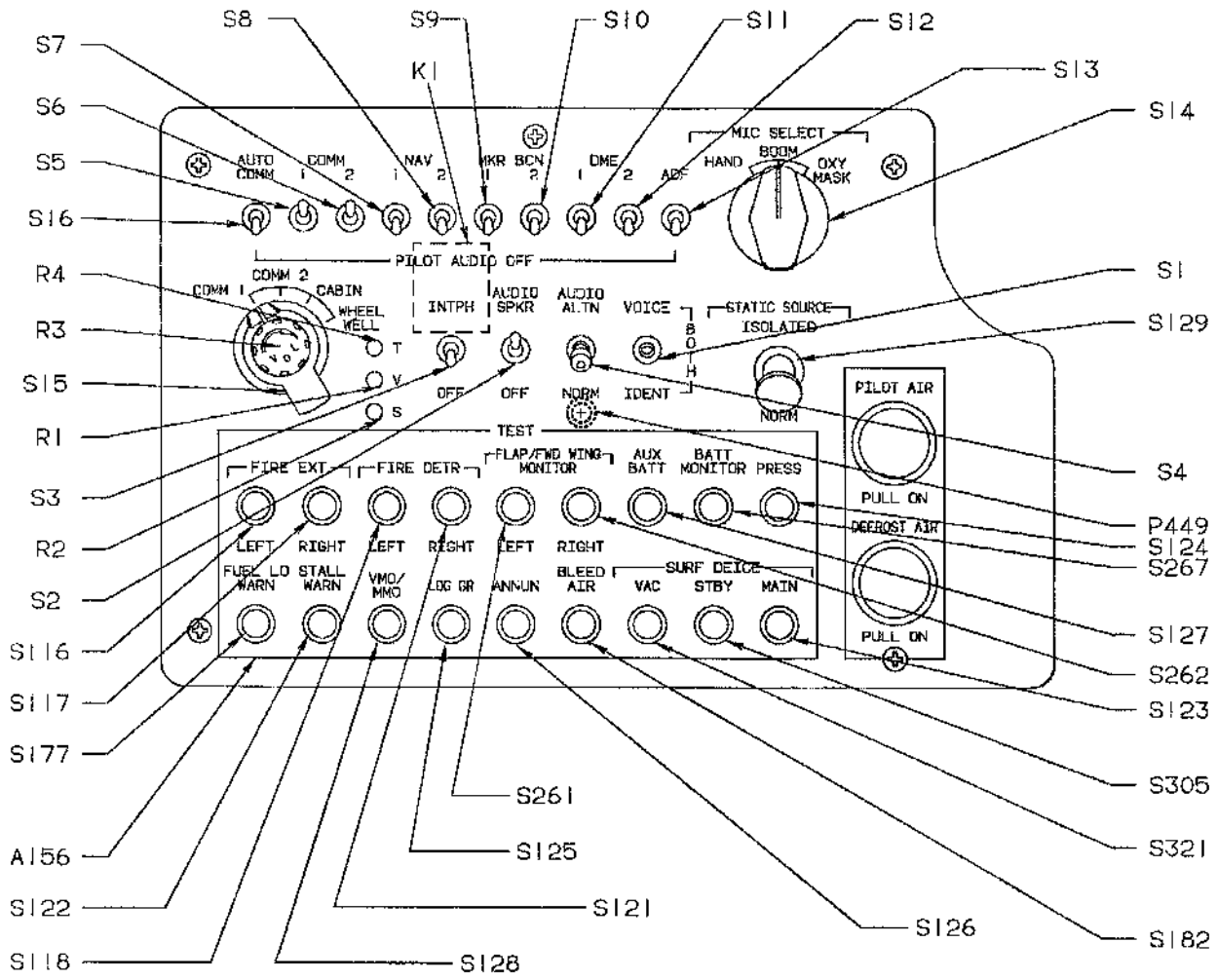
*S327 - GROUND ICE DETECTOR SWITCH

P450 - ELECTROLUMINESCENT PANEL CONNECTOR

* NC-4 THRU NC-9 AND NC-11 THRU NC-43 WITH BEECHCRAFT KIT NO. 122-5024 INSTALLED, NC-10 AND NC-44 AND AFTER

C94NC39B0521

Pilot's Inboard Subpanel (Zone 233) (Effectivity: All)
Figure 3



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Pilot's Outboard Subpanel (Zone 234) (Effectivity: All)
(Sheet 1 of 2)
Figure 4

S16 - AUTO COMM SWITCH

S5 - COMM 1 SWITCH

S6 - COMM 2 SWITCH

S7 - NAV 1 SWITCH

S8 - NAV 2 SWITCH

S9 - MARKER BEACON 1 SWITCH

S10 - MARKER BEACON 2 SWITCH

S11 - DME 1 SWITCH

S12 - DME 2 SWITCH

S13 - ADF SWITCH

S14 - MICROPHONE SELECT SWITCH

S15 - MICROPHONE OUTPUT SELECT SWITCH

R3 - MASTER VOLUME CONTROL SWITCH

R4 - SIDETONE TRIMPOT

R1 - INTERPHONE VOLUME TRIMPOT

R2 - INTERPHONE SQUELCH TRIMPOT

S3 - INTERPHONE SWITCH

S2 - AUDIO SPEAKER SWITCH

S4 - AUDIO ALTERNATE SWITCH

S1 - VOICE/IDENT SWITCH

S129 - STATIC SOURCE SELECT SWITCH

K1 - BOOM AND OXYGEN MASK MICROPHONE RELAY

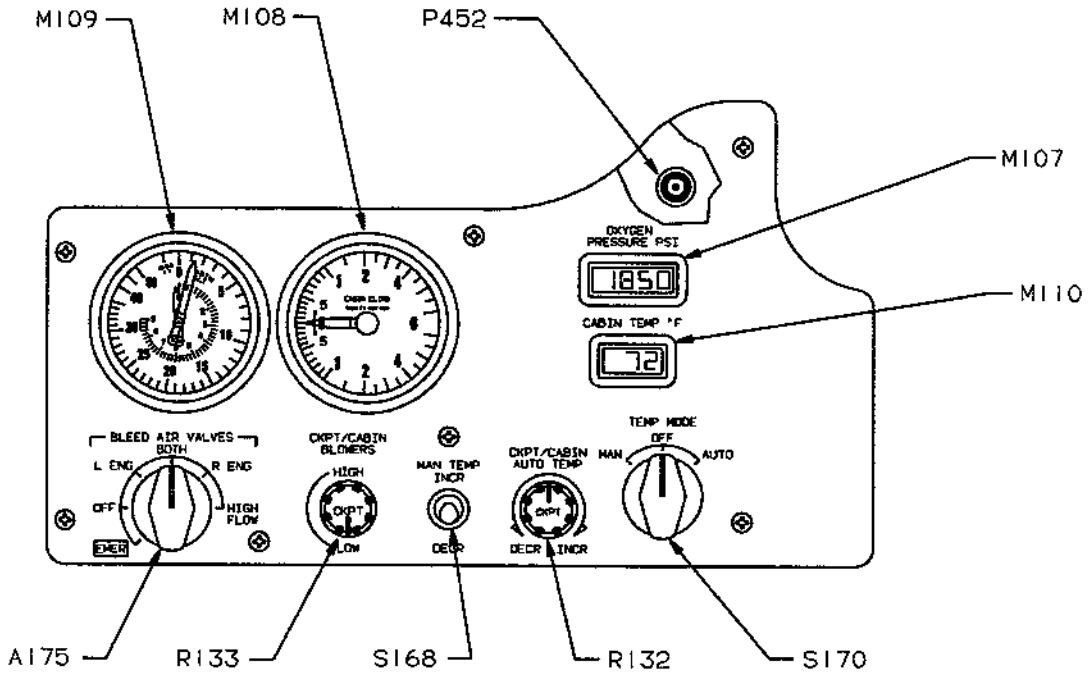
P449 - ELECTROLUMINESCENT PANEL CONNECTOR

A156 - TEST SWITCH MODULE

- S116 - LH FIRE EXTINGUISHER TEST SWITCH
- S117 - RH FIRE EXTINGUISHER TEST SWITCH
- S118 - LH FIRE DETECTOR TEST SWITCH
- S121 - RH FIRE DETECTOR TEST SWITCH
- S261 - LH FLAP/FORWARD WING MONITOR TEST SWITCH
- S262 - RH FLAP/FORWARD WING MONITOR TEST SWITCH
- S127 - AUXILIARY BATTERY TEST SWITCH
- S267 - BATTERY MONITOR TEST SWITCH
- S124 - CABIN PRESSURE TEST SWITCH
- S177 - LH AND RH FUEL QUANTITY WARNING TEST SWITCH
- S122 - STALL TEST SWITCH
- S128 - VMO/MMO TEST SWITCH
- S125 - LANDING GEAR TEST SWITCH
- S126 - ANNUNCIATOR TEST SWITCH
- S182 - BLEED AIR TEST SWITCH
- S321 - SURFACE DEICE VACUUM SWITCH
- S305 - SURFACE DEICE STANDBY SWITCH
- S123 - SURFACE DEICE MAIN SWITCH

2000-351-079

Pilot's Outboard Subpanel (Zone 234) (Effectivity: All)
(Sheet 2 of 2)
Figure 4



M109 - CABIN ALTITUDE AND DIFFERENTIAL PRESSURE INDICATOR

M108 - CABIN RATE-OF-CLIMB INDICATOR

P452 - ELECTROLUMINESCENT PANEL CONNECTOR

M107 - OXYGEN PRESSURE DISPLAY

M110 - CABIN TEMPERATURE DISPLAY

A175 - BLEED AIR VALVE SWITCH
S166 - BLEED AIR CONTROL SWITCH
CR1 THRU CR7 - DIODES

R133 - CABIN/COCKPIT BLOWER SPEED CONTROL SWITCH

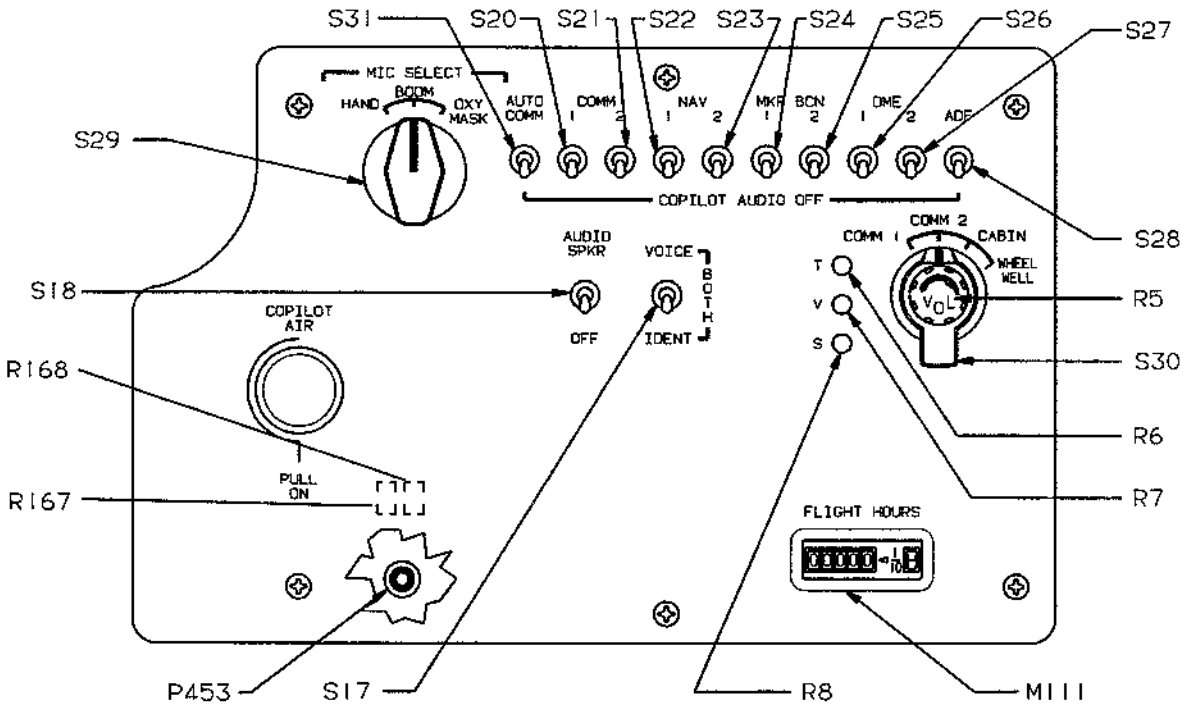
S168 - MANUAL TEMPERATURE CONTROL SWITCH

R132 - AUTO TEMPERATURE SWITCH

S170 - TEMPERATURE MODE SWITCH

2000-351-010

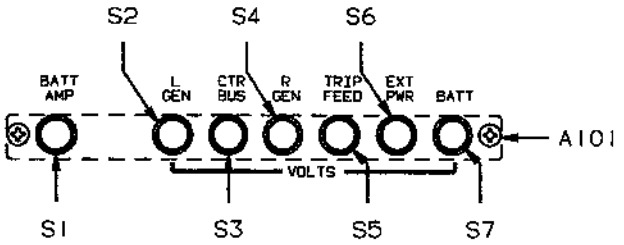
Copilot's Inboard Subpanel (Zone 243) (Effectivity: All)
Figure 5



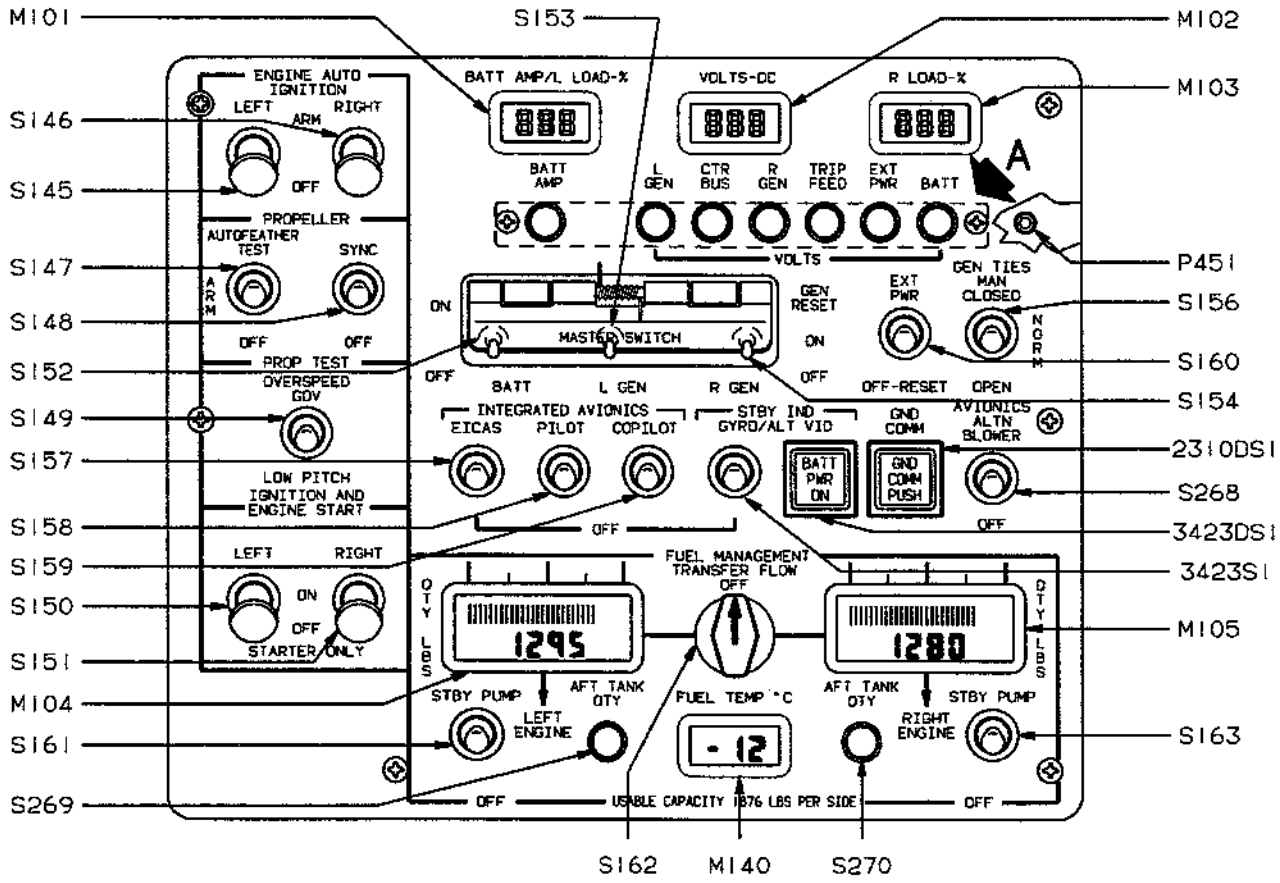
S31 - AUTO COMM SWITCH
S20 - COMM 1 SWITCH
S21 - COMM 2 SWITCH
S22 - NAV 1 SWITCH
S23 - NAV 2 SWITCH
S24 - MARKER BEACON 1 SWITCH
S25 - MARKER BEACON 2 SWITCH
S26 - DME 1 SWITCH
S27 - DME 2 SWITCH
S28 - ADF SWITCH
S29 - MICROPHONE SELECT SWITCH
S30 - MICROPHONE OUTPUT SELECT SWITCH
R5 - MASTER VOLUME CONTROL SWITCH
R6 - SIDETONE TRIMPOT
R7 - INTERPHONE VOLUME TRIMPOT
R8 - INTERPHONE SQUELCH TRIMPOT
S18 - AUDIO SPEAKER SWITCH
S17 - VOICE/IDENT SWITCH
M111 - FLIGHT HOUR METER
R167 - LEFT ELEVATOR POSITION STALL WARNING TRIMPOT
R168 - RIGHT ELEVATOR POSITION STALL WARNING TRIMPOT
P453 - ELECTROLUMINESCENT PANEL CONNECTOR

2000-351-009

Copilot's Outboard Subpanel (Zone 244) (Effectivity: All)
Figure 6



DETAIL A



C9200114 C

Center Pedestal Subpanel (Zone 242) (Effectivity: All)
(Sheet 1 of 2)
Figure 7

A101 - VOLTAGE SELECT SWITCH ASSEMBLY
- S1 - BATTERY AMP SWITCH
- S2 - LH GENERATOR VOLTAGE SWITCH
- S3 - CENTER BUS VOLTAGE SWITCH
- S4 - RH GENERATOR VOLTAGE SWITCH
- S5 - TRIPLE FED BUS VOLTAGE SWITCH
- S6 - EXTERNAL POWER VOLTAGE SWITCH
- S7 - BATTERY VOLTAGE SWITCH

M101 - BATTERY AMMETER/LEFT DC LOADMETER DISPLAY

M102 - DC VOLTAGE DISPLAY

M103 - RIGHT DC LOADMETER DISPLAY

M104 - LH FUEL QUANTITY DISPLAY

M105 - RH FUEL QUANTITY DISPLAY

M140 - FUEL TEMPERATURE DISPLAY

S145 - LH ENGINE AUTO IGNITION SWITCH

S146 - RH ENGINE AUTO IGNITION SWITCH

S147 - PROPELLER AUTOFEATHER SWITCH

S148 - PROPELLER SYNCHROPHASER CONTROL SWITCH

S149 - PROPELLER OVERSPEED GOVERNOR/ LOW PITCH TEST SWITCH

S150 - LH ENGINE IGNITION AND START SWITCH

S151 - RH ENGINE IGNITION AND START SWITCH

S152 - BATTERY SWITCH

S153 - LH GENERATOR CONTROL SWITCH

S154 - RH GENERATOR CONTROL SWITCH

S156 - GENERATOR BUS TIE RELAY SWITCH
- CR149 - DIODE
- CR150 - DIODE

S157 - EICAS DISPLAY SWITCH

S158 - PILOT'S AVIONICS SWITCH

S159 - COPILOT'S AVIONICS SWITCH

S160 - EXTERNAL POWER SWITCH

S161 - LH STANDBY FUEL PUMP SWITCH

S162 - FUEL TRANSFER FLOW SWITCH

S163 - RH STANDBY FUEL PUMP SWITCH

S268 - AVIONICS ALTERNATE BLOWER SWITCH

S269 - LH AFT FUEL QUANTITY SWITCH

S270 - RH AFT FUEL QUANTITY SWITCH

3423S1 - STANDBY ATTITUDE GYRO/ALTIMETER VIBRATOR SWITCH

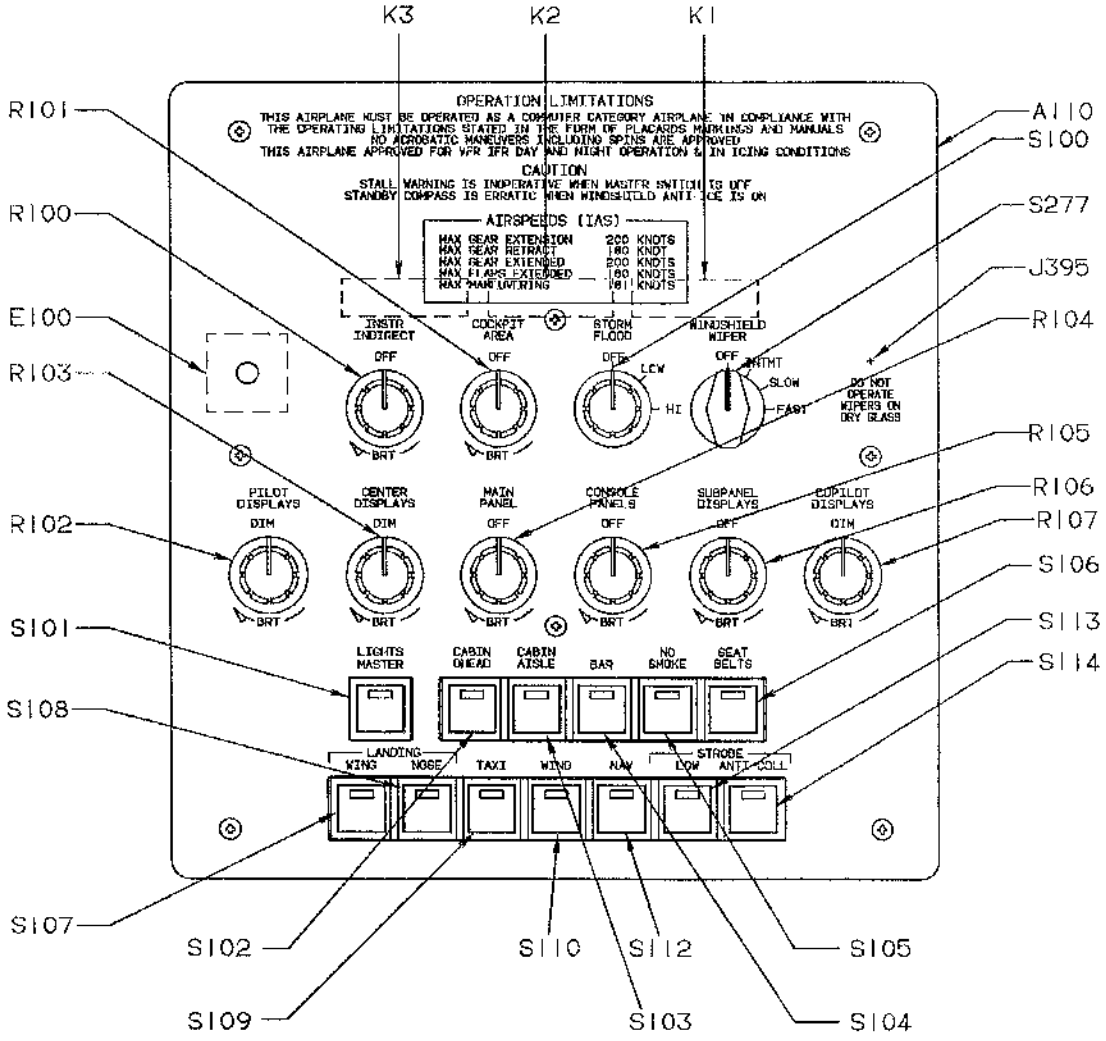
3423DS1 - STANDBY ATTITUDE GYRO AUXILIARY BATTERY ON LIGHT

2310DS1 - GROUND COMM LIGHT/SWITCH

P451 - ELECTROLUMINESCENT PANEL CONNECTOR

C9200199 C

Center Pedestal Subpanel (Zone 242) (Effectivity: All)
(Sheet 2 of 2)
Figure 7



2000-368-010

Overhead Switch Panel (Zone 253) (Effectivity: All)
(Sheet 1 of 2)
Figure 8

A110 - OVERHEAD SWITCH PANEL ASSEMBLY

K1 - PILOT AND CENTER DISPLAY POTENTIOMETERS RELAY

K2 - MAIN AND CONSOLE PANELS POTENTIOMETERS RELAY

K3 - COPILOT AND SUBPANEL DISPLAY POTENTIOMETERS RELAY

E100 - AMBIENT LIGHT SENSOR

R100 - INSTRUMENT INDIRECT LIGHTING POTENTIOMETER

R101 - COCKPIT AREA LIGHT POTENTIOMETER

R102 - PILOT DISPLAYS DIMMING POTENTIOMETER

R103 - CENTER DISPLAYS DIMMING POTENTIOMETER

R104 - MAIN PANEL POTENTIOMETER

R105 - CONSOLE PANEL POTENTIOMETER

R106 - SUBPANEL DISPLAYS POTENTIOMETER

R107 - COKPILOT DISPLAYS DIMMING POTENTIOMETER

S100 - STORM/FLOODLIGHT SWITCH
CR166 - SWITCH DIODE

S101 - COCKPIT LIGHTS MASTER SWITCH

S102 - CABIN OVERHEAD LIGHT SWITCH

S103 - CABIN AISLE LIGHT SWITCH

S104 - BAR LIGHT SWITCH

S105 - CABIN NO SMOKING LIGHT SWITCH

S106 - CABIN FASTEN SEAT BELT LIGHT SWITCH

S107 - WING LANDING LIGHTS SWITCH

S108 - NOSE LANDING LIGHT SWITCH

S109 - NOSE TAXI LIGHT SWITCH

S110 - WING INSPECTION LIGHTS SWITCH

S112 - NAVIGATION LIGHT SWITCH

S113 - LOW STROBE LIGHTS SWITCH

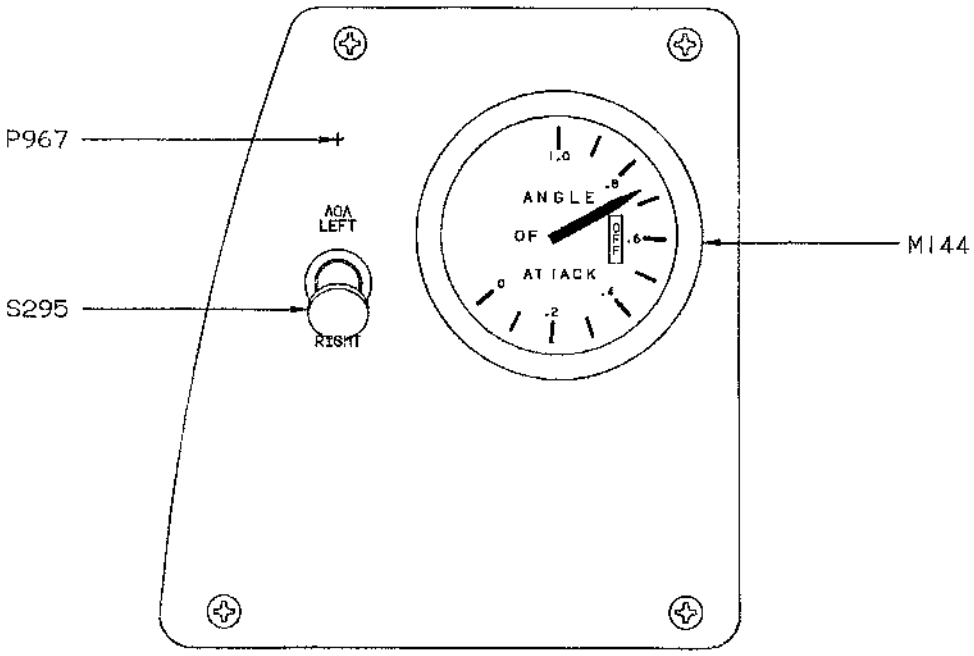
S114 - ANTI-COLLISION LIGHTS SWITCH

S277 - WINDSHIELD WIPER SWITCH

J395 - ELECTROLUMINESCENT PANEL CONNECTOR

2000-368-011

Overhead Switch Panel (Zone 253) (Effectivity: All)
(Sheet 2 of 2)
Figure 8



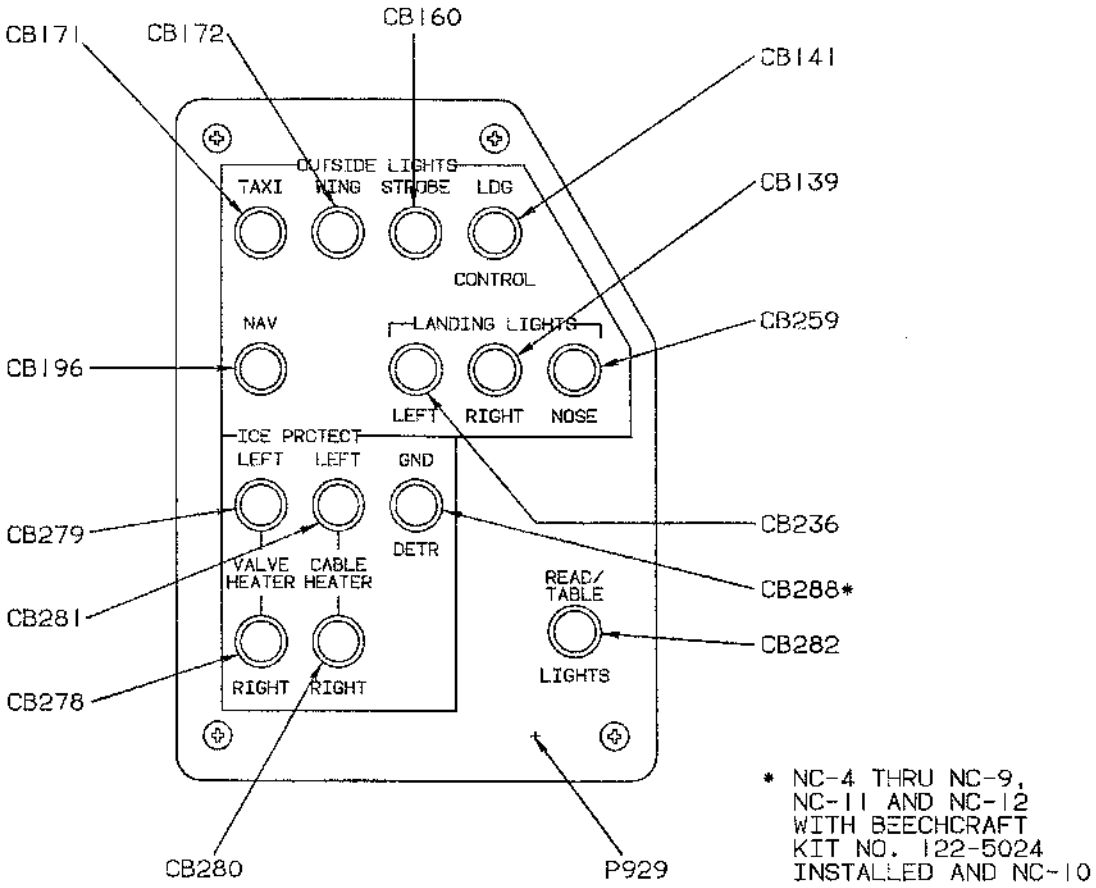
S295 - ANGLE-OF-ATTACK INDICATOR TRANSFER SWITCH

M144 - ANGLE-OF-ATTACK INDICATOR

P967 - ELECTROLUMINESCENT PANEL CONNECTOR

2000-351-078

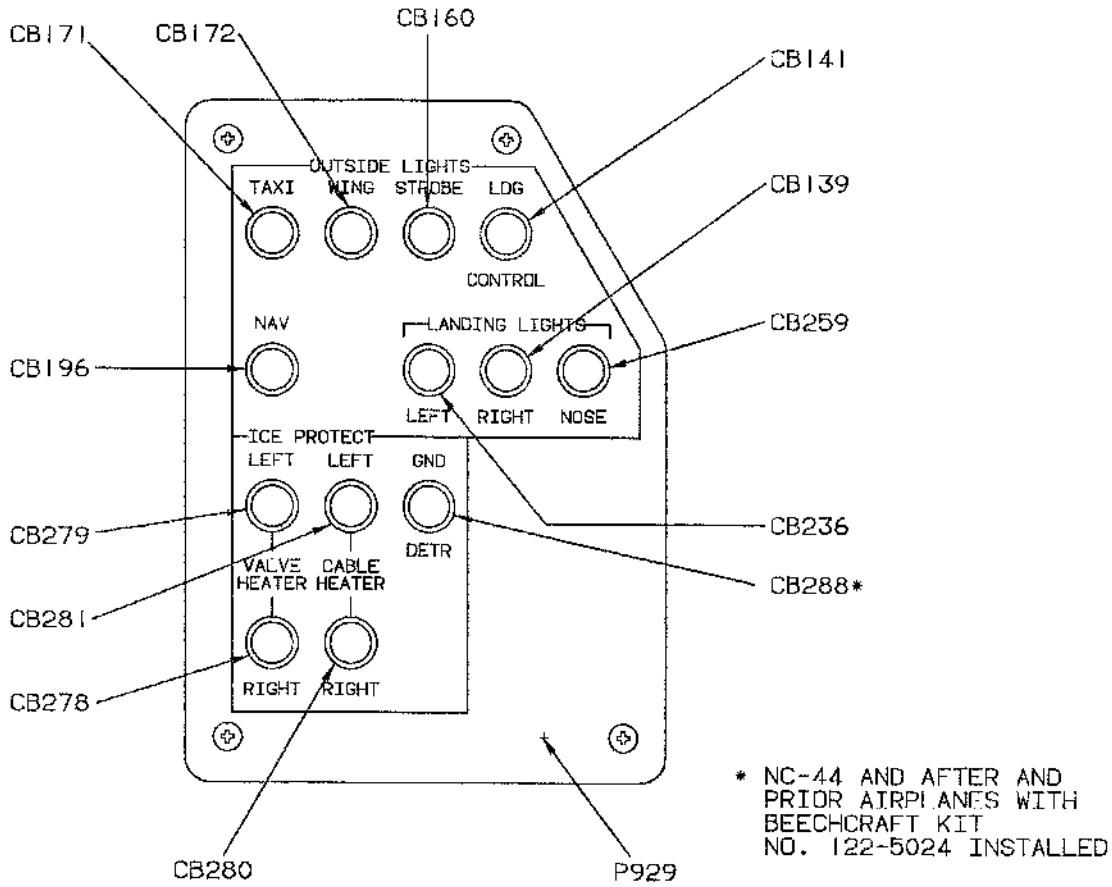
LH Upper Console Panel (Zone 259) (Effectivity: All)
Figure 9



CB171 - NOSE TAXI LIGHT CIRCUIT BREAKER
CB172 - WING INSPECTION LIGHTS CIRCUIT BREAKER
CB160 - STROBE LIGHT CIRCUIT BREAKER
CB141 - NOSE AND WING LANDING LIGHTS CONTROL CIRCUIT BREAKER
CB196 - NAVIGATION LIGHTS CIRCUIT BREAKER
CB236 - LH WING LANDING LIGHT POWER CIRCUIT BREAKER
CB139 - RH WING LANDING LIGHT POWER CIRCUIT BREAKER
CB259 - NOSE LANDING LIGHT POWER CIRCUIT BREAKER
CB279 - LEFT SURFACE DEICE VALVES HEAT CIRCUIT BREAKER
CB281 - LH ENGINE POWER CABLE HEAT CIRCUIT BREAKER
CB278 - RIGHT SURFACE DEICE VALVES HEAT CIRCUIT BREAKER
CB280 - RH ENGINE POWER CABLE HEAT CIRCUIT BREAKER
CB282 - CABIN/READING LIGHTS CIRCUIT BREAKER
*CB288 - GROUND ICE DETECTOR CIRCUIT BREAKER
P929 - ELECTROLUMINESCENT PANEL CONNECTOR

C94NC39B0520

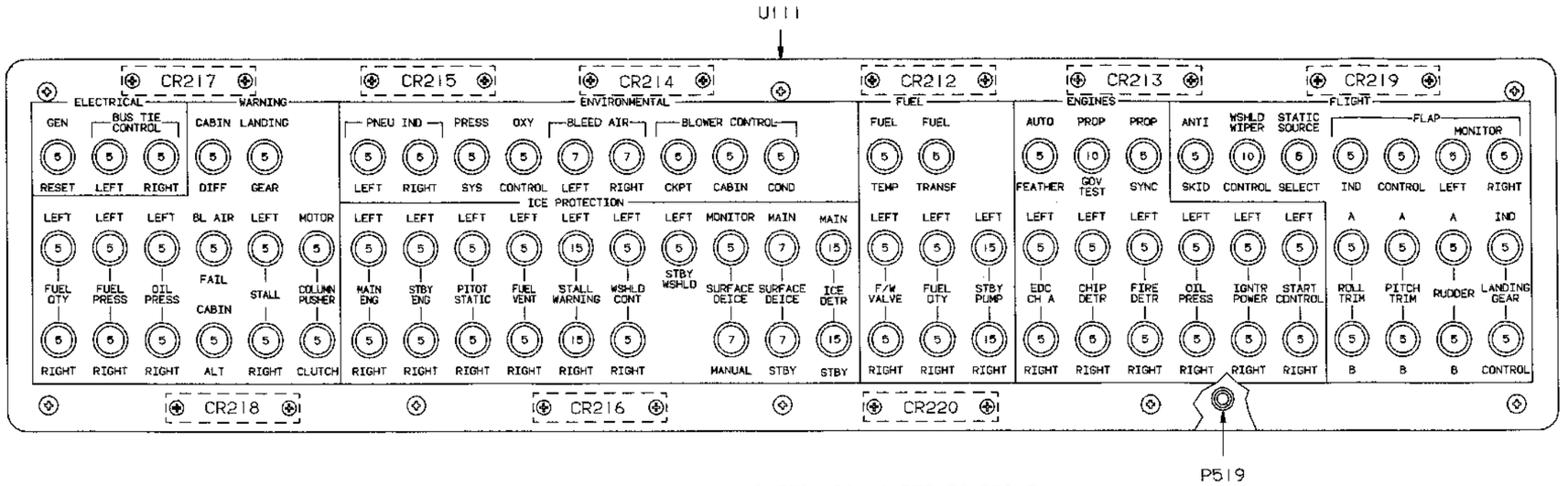
Auxiliary Circuit Breaker Panel (Zone 235) (Effectivity: NC-4 thru NC-12)
Figure 10



CB171 - NOSE TAXI LIGHT CIRCUIT BREAKER
CB172 - WING INSPECTION LIGHTS CIRCUIT BREAKER
CB160 - STROBE LIGHT CIRCUIT BREAKER
CB141 - NOSE AND WING LANDING LIGHTS CONTROL CIRCUIT BREAKER
CB196 - NAVIGATION LIGHTS CIRCUIT BREAKER
CB236 - LH WING LANDING LIGHT POWER CIRCUIT BREAKER
CB139 - RH WING LANDING LIGHT POWER CIRCUIT BREAKER
CB259 - NOSE LANDING LIGHT POWER CIRCUIT BREAKER
CB279 - LEFT SURFACE DEICE VALVES HEAT CIRCUIT BREAKER
CB281 - LH ENGINE POWER CABLE HEAT CIRCUIT BREAKER
CB278 - RIGHT SURFACE DEICE VALVES HEAT CIRCUIT BREAKER
CB280 - RH ENGINE POWER CABLE HEAT CIRCUIT BREAKER
*CB288 - GROUND ICE DETECTOR CIRCUIT BREAKER
P929 - ELECTROLUMINESCENT PANEL CONNECTOR

C94NC39B0519

Auxiliary Circuit Breaker Panel (Zone 235) (Effectivity: NC-13 and After)
Figure 11



TOP ROW (LEFT TO RIGHT)
- CB130 - GENERATOR RESET
- CB131 - BUS TIE CONTROL LEFT
- CB129 - BUS TIE CONTROL RIGHT
- CB251 - CABIN DIFF PRESSURE WARNING
- CB102 - LANDING GEAR WARNING HORN
- CB270 - PNEUMATIC INDICATOR LEFT
- CB274 - PNEUMATIC INDICATOR RIGHT
- CB122 - CABIN PRESSURE SYSTEM
- CB123 - OXYGEN CONTROL
- CB152 - LEFT BLEED AIR CONTROL
- CB114 - RIGHT BLEED AIR CONTROL
- CB262 - BLOWER CONTROL COCKPIT
- CB263 - BLOWER CONTROL CABIN
- CB264 - BLOWER CONTROL CONDENSER
- CB247 - FUEL TEMPERATURE
- CB156 - FUEL TRANSFER
- CB111 - PROP AUTO FEATHER
- CB137 - PROP GOV TEST
- CB151 - PROP SYNC
- CB265 - ANTI-SKID
- CB250 - WINDSHIELD WIPER CONTROL
- CB168 - STATIC SOURCE SELECT
- CB182 - FLAP INDICATOR
- CB134 - FLAP CONTROL
- CB248 - LEFT FLAP MONITOR
- CB249 - RIGHT FLAP MONITOR

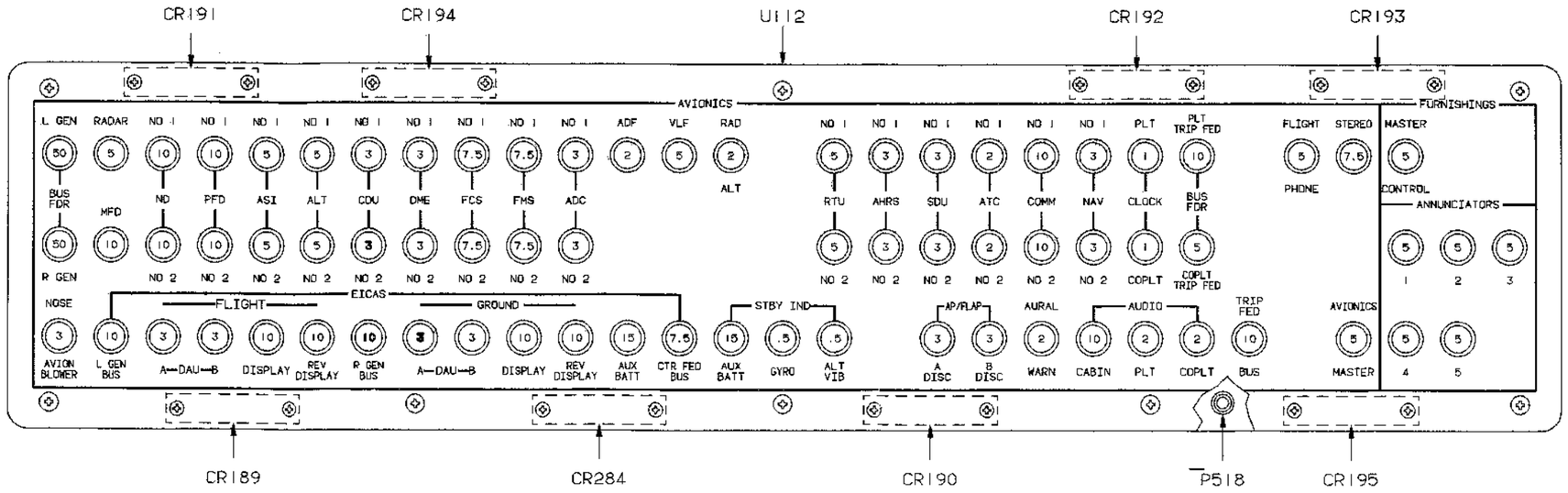
MIDDLE ROW (LEFT TO RIGHT)
- CB145 - LEFT FUEL QUANTITY WARNING
- CB179 - LEFT LOW FUEL PRESSURE WARNING
- CB191 - LEFT ENGINE OIL PRESSURE WARNING
- CB198 - BLEED AIR FAIL WARNING
- CB227 - LEFT STALL WARNING
- CB267 - COLUMN PUSHER MOTOR
- CB110 - LEFT MAIN ENGINE ANTI-ICE
- CB144 - LEFT STANDBY ENGINE ANTI-ICE
- CB101 - LEFT PITOT/STATIC HEAT
- CB174 - LEFT FUEL VENT HEATER
- CB234 - LEFT STALL WARNING HEAT
- CB175 - PILOT WINDSHIELD ANTI-ICE CONTROL
- CB177 - PILOT STANDBY WINDSHIELD ANTI-ICE
- CB277 - SURFACE DEICE MONITOR
- CB225 - SURFACE DEICE MAIN
- CB226 - ICE DETECTOR MAIN
- CB165 - LH FIREWALL FUEL SHUTOFF VALVE
- CB140 - LEFT FUEL QUANTITY
- CB164 - LEFT STANDBY FUEL PUMP
- CB219 - LEFT EDC (CHANNEL A)
- CB178 - LEFT CHIP DETECTOR
- CB189 - LEFT FIRE DETECTOR
- CB115 - LEFT ENGINE OIL PRESSURE
- CB167 - LEFT IGNITOR POWER
- CB113 - LEFT START CONTROL
- CB256 - ROLL TRIM A
- CB254 - PITCH TRIM A
- CB157 - RUDDER TRIM A
- CB103 - LANDING GEAR POSITION INDICATOR

BOTTOM ROW (LEFT TO RIGHT)
- CB158 - RIGHT FUEL QUANTITY WARNING
- CB181 - RIGHT LOW FUEL PRESSURE WARNING
- CB190 - RIGHT ENGINE OIL PRESSURE WARNING
- CB125 - CABIN ALTITUDE WARNING
- CB112 - RIGHT STALL WARNING
- CB267 - COLUMN PUSHER CLUTCH
- CB128 - RIGHT MAIN ENGINE ANTI-ICE
- CB153 - RIGHT STANDBY ENGINE ANTI-ICE
- CB148 - RIGHT PITOT/STATIC HEAT
- CB170 - RIGHT FUEL VENT HEATER
- CB173 - RIGHT STALL WARNING HEAT
- CB176 - COPILOT WINDSHIELD ANTI-ICE CONTROL
- CB276 - SURFACE DEICE MANUAL
- CB260 - SURFACE DEICE STANDBY
- CB272 - ICE DETECTOR STANDBY
- CB163 - RIGHT FIREWALL FUEL SHUTOFF VALVE
- CB150 - RIGHT FUEL QUANTITY WARNING
- CB162 - RIGHT STANDBY FUEL PUMP
- CB220 - RIGHT EDC (CHANNEL A)
- CB180 - RIGHT CHIP DETECTOR
- CB188 - RIGHT FIRE DETECTOR
- CB117 - RIGHT ENGINE OIL PRESSURE
- CB197 - RIGHT IGNITOR POWER
- CB120 - RIGHT START CONTROL
- CB257 - ROLL TRIM B
- CB255 - PITCH TRIM B
- CB138 - RUDDER TRIM
- CB133 - LANDING GEAR CONTROL

- CR212 - METAL OXIDE VARISTOR
- CR213 - METAL OXIDE VARISTOR
- CR214 - METAL OXIDE VARISTOR
- CR215 - METAL OXIDE VARISTOR
- CR216 - METAL OXIDE VARISTOR
- CR217 - METAL OXIDE VARISTOR
- CR218 - METAL OXIDE VARISTOR
- CR219 - METAL OXIDE VARISTOR
- CR220 - METAL OXIDE VARISTOR
- U111 - ELECTROLUMINESCENT PANEL
- P519 - CONNECTOR

2000-351-016

LH Circuit Breaker Panel (Zone 236)
(Effectivity: All)
Figure 12



TOP ROW (LEFT TO RIGHT)
CB214 - LEFT GENERATOR AVIONICS BUS FEEDER
CB18 - RADAR
CB15 - NAVIGATION DISPLAY NO. 1
CB14 - PRIMARY FLIGHT DISPLAY NO. 1
CB13 - AIRSPEED INDICATOR NO. 1
CB16 - ALTITUDE INDICATOR NO. 1
CB12 - CONTROL DISPLAY UNIT NO. 1
CB8 - DISTANCE MEASURING EQUIPMENT NO. 1
CB7 - FLIGHT CONTROL SYSTEM NO. 1
CB6 - FLIGHT MANAGEMENT SYSTEM NO. 1
CB5 - AIR DATA COMPUTER NO. 1
CB10 - ADF
CB19 - VLF
CB20 - RADIO ALTIMETER
CB2 - RADIO TUNING UNIT NO. 1
CB61 - AHRS NO. 1
CB60 - SENSOR DISPLAY UNIT NO.1
CB1 - ATC NO. 1
CB4 - COMM NO. 1
CB3 - NAV NO. 1
CB59 - PILOT'S CLOCK
CB216 - PILOT'S AVIONICS TRIPLE FED BUS FEEDER
CB56 - FLIGHT PHONE
CB57 - STEREO
CB147 - FURNISHINGS MASTER CONTROL

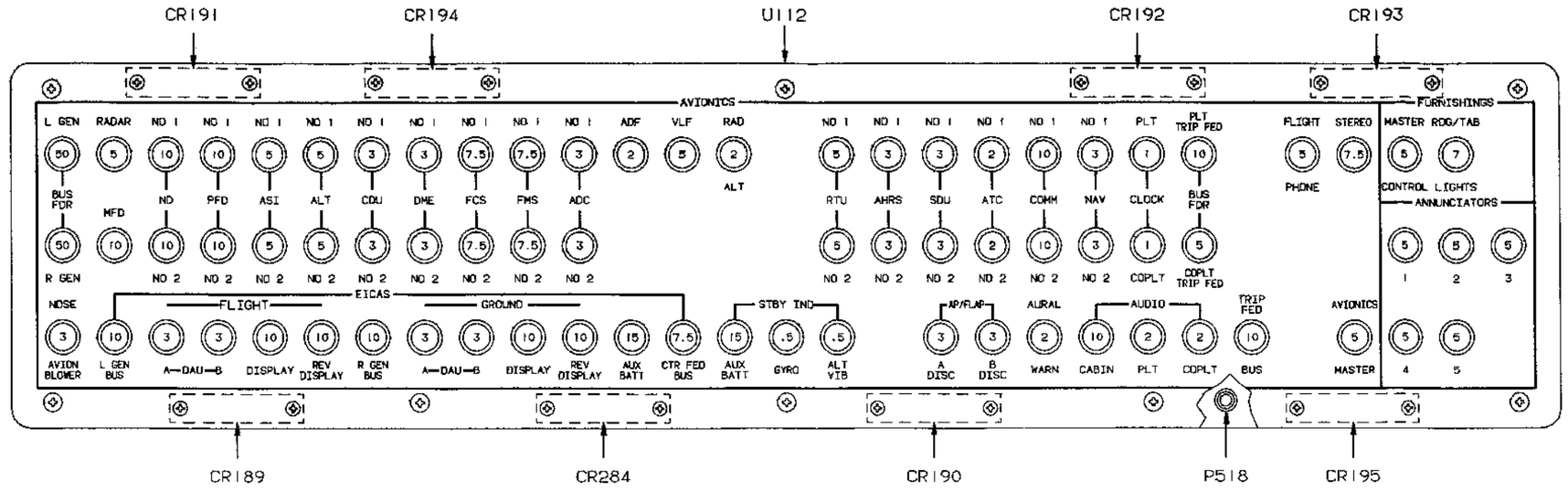
MIDDLE ROW (LEFT TO RIGHT)
CB217 - RIGHT GENERATOR AVIONICS BUS FEEDER
CB37 - MULTIFUNCTION DISPLAY
CB34 - NAVIGATION DISPLAY UNIT NO. 2
CB33 - PRIMARY FLIGHT DISPLAY UNIT NO. 2
CB32 - AIRSPEED INDICATOR NO. 2
CB35 - ALTITUDE INDICATOR NO. 2
CB31 - CONTROL DISPLAY UNIT NO. 2
CB27 - DISTANCE MEASURING EQUIPMENT NO. 2
CB26 - FLIGHT CONTROL SYSTEM NO. 2
CB30 - FLIGHT MANAGEMENT SYSTEM NO. 2
CB29 - AIR DATA COMPUTER NO. 2
CB22 - RADIO TUNING UNIT NO. 2
CB28 - AHRS NO. 2
CB11 - SENSOR DISPLAY UNIT NO. 2
CB25 - ATC NO. 2
CB23 - COMM NO. 2
CB24 - NAV NO. 2
CB21 - COPILOT'S CLOCK
CB215 - COPILOT'S AVIONICS TRIPLE FED BUS FEEDER
CB104 - NO.1 ANNUNCIATOR
CB108 - NO.2 ANNUNCIATOR
CB107 - NO.3 ANNUNCIATOR

BOTTOM ROW (LEFT TO RIGHT)
CB48 - NOSE AVIONICS BLOWERS
CB210 - LEFT GENERATOR EICAS BUS FEEDER
CB47 - FLIGHT DATA ACQUISTION UNIT A
CB49 - FLIGHT DATA ACQUISTION UNIT B
CB46 - FLIGHT DISPLAY
CB50 - FLIGHT REV DISPLAY
CB211 - RIGHT GENERATOR EICAS BUS FEEDER
CB54 - GROUND DATA ACQUISTION UNIT A
CB55 - GROUND DATA ACQUISTION UNIT B
CB52 - GROUND DISPLAY
CB53 - GROUND REV DISPLAY
CB43 - EICAS AUXILIARY BATTERY
CB258 - AVIONICS CENTER FED BUS FEEDER
CB42 - ATTITUDE GYRO AUXILIARY BATTERY
CB51 - STANDBY ATTITUDE GYRO
CB62 - STANDBY ALTIMETER VIBRATOR
CB45 - AUTOPILOT/FLAP DISCONNECT A
CB44 - AUTOPILOT/FLAP DISCONNECT B
CB40 - AURAL WARNING
CB41 - CABIN AUDIO
CB38 - PILOT AUDIO
CB39 - COPILOT AUDIO
CB212 - AVIONICS TRIPLE FED BUS FEEDER
CB213 - AVIONICS MASTER POWER
CB106 - NO. 4 ANNUNCIATOR
CB105 - NO. 5 ANNUNCIATOR

CR189 - METAL OXIDE VARISTOR
CR190 - METAL OXIDE VARISTOR
CR191 - METAL OXIDE VARISTOR
CR192 - METAL OXIDE VARISTOR
CR193 - METAL OXIDE VARISTOR
CR194 - METAL OXIDE VARISTOR
CR195 - METAL OXIDE VARISTOR
CR284 - METAL OXIDE VARISTOR
U112 - ELECTROLUMINESCENT PANEL
P518 - CONNECTOR

2000-351-081

RH Circuit Breaker Panel (Zone 246) (Effectivity: NC-4 thru NC-12)
Figure 13



TOP ROW (LEFT TO RIGHT)
CB214 - LEFT GENERATOR AVIONICS BUS FEEDER
CB18 - RADAR
CB15 - NAVIGATION DISPLAY NO. 1
CB14 - PRIMARY FLIGHT DISPLAY NO. 1
CB13 - AIRSPEED INDICATOR NO. 1
CB16 - ALTITUDE INDICATOR NO. 1
CB12 - CONTROL DISPLAY UNIT NO. 1
CB8 - DISTANCE MEASURING EQUIPMENT NO. 1
CB7 - FLIGHT CONTROL SYSTEM NO. 1
CB6 - FLIGHT MANAGEMENT SYSTEM NO. 1
CB5 - AIR DATA COMPUTER NO. 1
CB10 - ADF
CB19 - VLF
CB20 - RADIO ALTIMETER
CB2 - RADIO TUNING UNIT NO. 1
CB61 - AHRS NO. 1
CB60 - SENSOR DISPLAY UNIT NO.1
CB1 - ATC NO. 1
CB4 - COMM NO. 1
CB3 - NAV NO. 1
CB59 - PILOT'S CLOCK
CB216 - PILOT'S AVIONICS TRIPLE FED BUS FEEDER
CB56 - FLIGHT PHONE
CB57 - STEREO
CB147 - FURNISHINGS MASTER CONTROL
CB282 - CABIN READING AND TABLE LIGHTS

MIDDLE ROW (LEFT TO RIGHT)
CB217 - RIGHT GENERATOR AVIONICS BUS FEEDER
CB37 - MULTIFUNCTION DISPLAY
CB34 - NAVIGATION DISPLAY UNIT NO. 2
CB33 - PRIMARY FLIGHT DISPLAY UNIT NO. 2
CB32 - AIRSPEED INDICATOR NO. 2
CB35 - ALTITUDE INDICATOR NO. 2
CB31 - CONTROL DISPLAY UNIT NO. 2
CB27 - DISTANCE MEASURING EQUIPMENT NO. 2
CB26 - FLIGHT CONTROL SYSTEM NO. 2
CB30 - FLIGHT MANAGEMENT SYSTEM NO. 2
CB29 - AIR DATA COMPUTER NO. 2
CB22 - RADIO TUNING UNIT NO. 2
CB28 - AHRS NO. 2
CB11 - SENSOR DISPLAY UNIT NO. 2
CB25 - ATC NO. 2
CB23 - COMM NO. 2
CB24 - NAV NO. 2
CB21 - COPILOT'S CLOCK
CB215 - COPILOT'S AVIONICS TRIPLE FED BUS FEEDER
CB104 - NO.1 ANNUNCIATOR
CB108 - NO.2 ANNUNCIATOR
CB107 - NO.3 ANNUNCIATOR

BOTTOM ROW (LEFT TO RIGHT)
CB48 - NOSE AVIONICS BLOWERS
CB210 - LEFT GENERATOR EICAS BUS FEEDER
CB47 - FLIGHT DATA ACQUISTION UNIT A
CB49 - FLIGHT DATA ACQUISTION UNIT B
CB46 - FLIGHT DISPLAY
CB50 - FLIGHT REV DISPLAY
CB211 - RIGHT GENERATOR EICAS BUS FEEDER
CB54 - GROUND DATA ACQUISTION UNIT A
CB55 - GROUND DATA ACQUISTION UNIT B
CB52 - GROUND DISPLAY
CB53 - GROUND REV DISPLAY
CB43 - EICAS AUXILIARY BATTERY
CB258 - AVIONICS CENTER FED BUS FEEDER
CB42 - ATTITUDE GYRO AUXILIARY BATTERY
CB51 - STANDBY ATTITUDE GYRO
CB62 - STANDBY ALTIMETER VIBRATOR
CB45 - AUTOPILOT/FLAP DISCONNECT A
CB44 - AUTOPILOT/FLAP DISCONNECT B
CB40 - AURAL WARNING
CB41 - CABIN AUDIO
CB38 - PILOT AUDIO
CB39 - COPILOT AUDIO
CB212 - AVIONICS TRIPLE FED BUS FEEDER
CB213 - AVIONICS MASTER POWER
CB106 - NO. 4 ANNUNCIATOR
CB105 - NO. 5 ANNUNCIATOR

CR189 - METAL OXIDE VARISTOR
CR190 - METAL OXIDE VARISTOR
CR191 - METAL OXIDE VARISTOR
CR192 - METAL OXIDE VARISTOR
CR193 - METAL OXIDE VARISTOR
CR194 - METAL OXIDE VARISTOR
CR195 - METAL OXIDE VARISTOR
CR284 - METAL OXIDE VARISTOR
U112 - ELECTROLUMINESCENT PANEL
P518 - CONNECTOR

2000-351-080

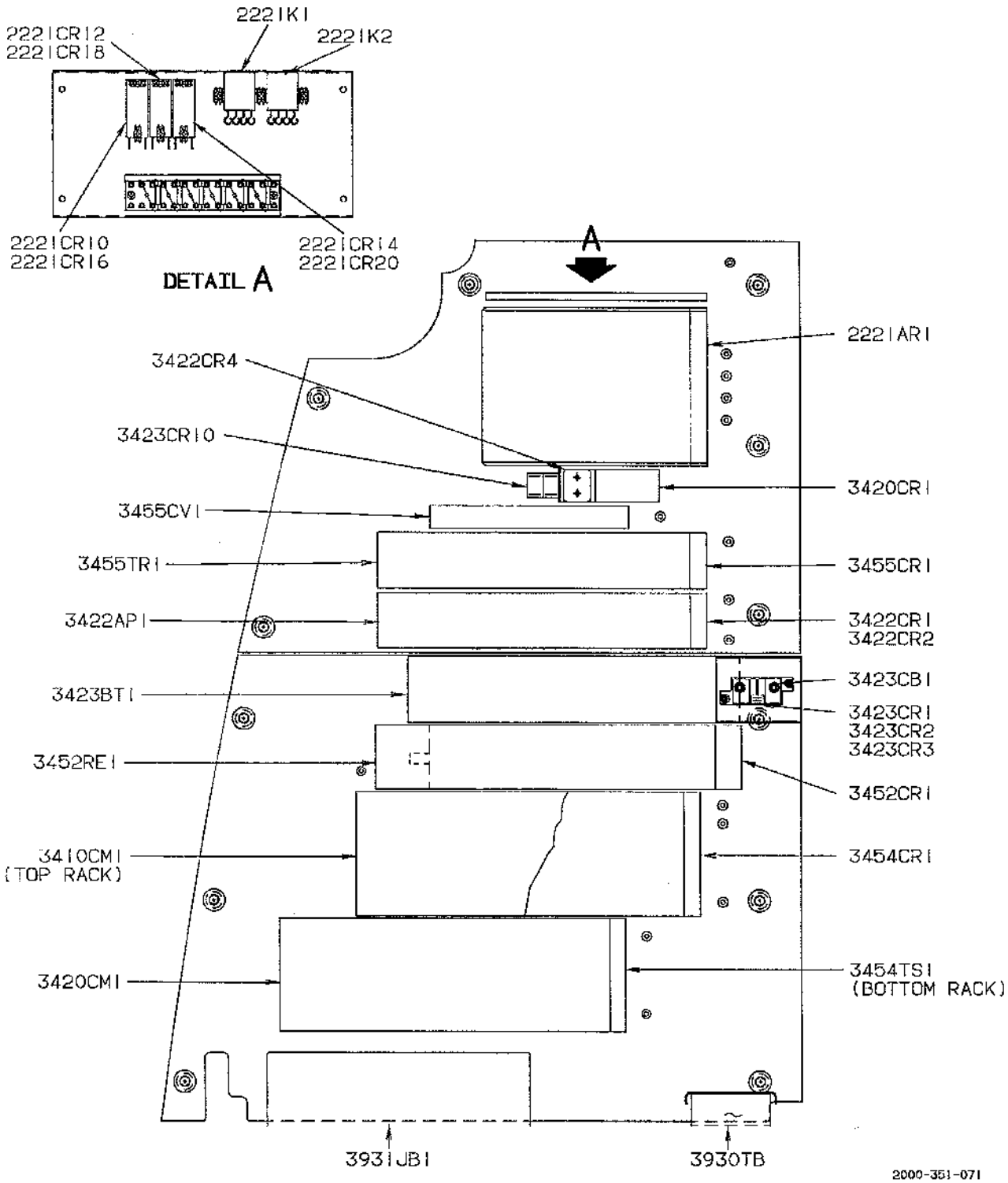
RH Circuit Breaker Panel (Zone 246)
(Effectivity: NC-13 and After)
Figure 14

ELECTRICAL AND ELECTRONIC EQUIPMENT PANELS - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

This chapter locates and identifies, through the use of illustrations, electrical equipment installed in the airplane. The equipment is located by the airplane zone numbers; refer to Chapter 6-40-00 for the locations of the zones. The components are identified by their reference designator numbers; these same reference designators are used in the BEECHCRAFT Starship 1 Wiring Diagram Manual and the avionics wiring diagrams.



LH Nose Avionics Compartment Electrical Equipment (Zone 213) (Effectivity: All)
(Sheet 1 of 2)
Figure 1

2221AR1 - ELECTRIC TRIM CONTROL BOX

3422CR4 - SENSOR DISPLAY DRIVER TRANSZORB

3423CR10 - STANDBY ATTITUDE GYRO AUXILIARY BATTERY TRANSZORB

3420CR1 - ATTITUDE HEADING COMPUTER NO. 2 TRANSZORB

3455CV1 - ATC TRANSPONDER NO. 1 CONTROL ADAPTER (NC-4 THRU NC-25)

3455TR1 - ATC TRANSPONDER NO. 1

3455CR1 - ATC TRANSPONDER NO. 1 METAL OXIDE VARISTOR

3422AP1 - SENSOR DISPLAY DRIVER

3422CR1 - SENSOR DISPLAY DRIVER METAL OXIDE VARISTOR

3422CR2 - SENSOR DISPLAY DRIVER METAL OXIDE VARISTOR

3423BT1 - STANDBY ATTITUDE GYRO AUXILIARY BATTERY

3423CB1 - STANDBY ATTITUDE GYRO AUXILIARY BATTERY REMOTE CONTROL CIRCUIT BREAKER

3423CR1 - STANDBY ATTITUDE GYRO AUXILIARY BATTERY METAL OXIDE VARISTOR

3423CR2 - STANDBY ATTITUDE GYRO AUXILIARY BATTERY METAL OXIDE VARISTOR

3423CR3 - STANDBY ATTITUDE GYRO AUXILIARY BATTERY METAL OXIDE VARISTOR

3452RE1 - VLF/OMEGA RECEIVER

3452CR1 - VLF/OMEGA RECEIVER METAL OXIDE VARISTOR

3410CM1 - AIR DATA COMPUTER NO. 1

3454TS1 - DME TRANSCEIVER NO. 1

3454CR1 - DME TRANSCEIVER NO. 1 METAL OXIDE VARISTOR

3420CM1 - ATTITUDE HEADING COMPUTER NO. 1

3931JB1 - RADIO JUNCTION BOX

3930TB - INTERFACE RACK

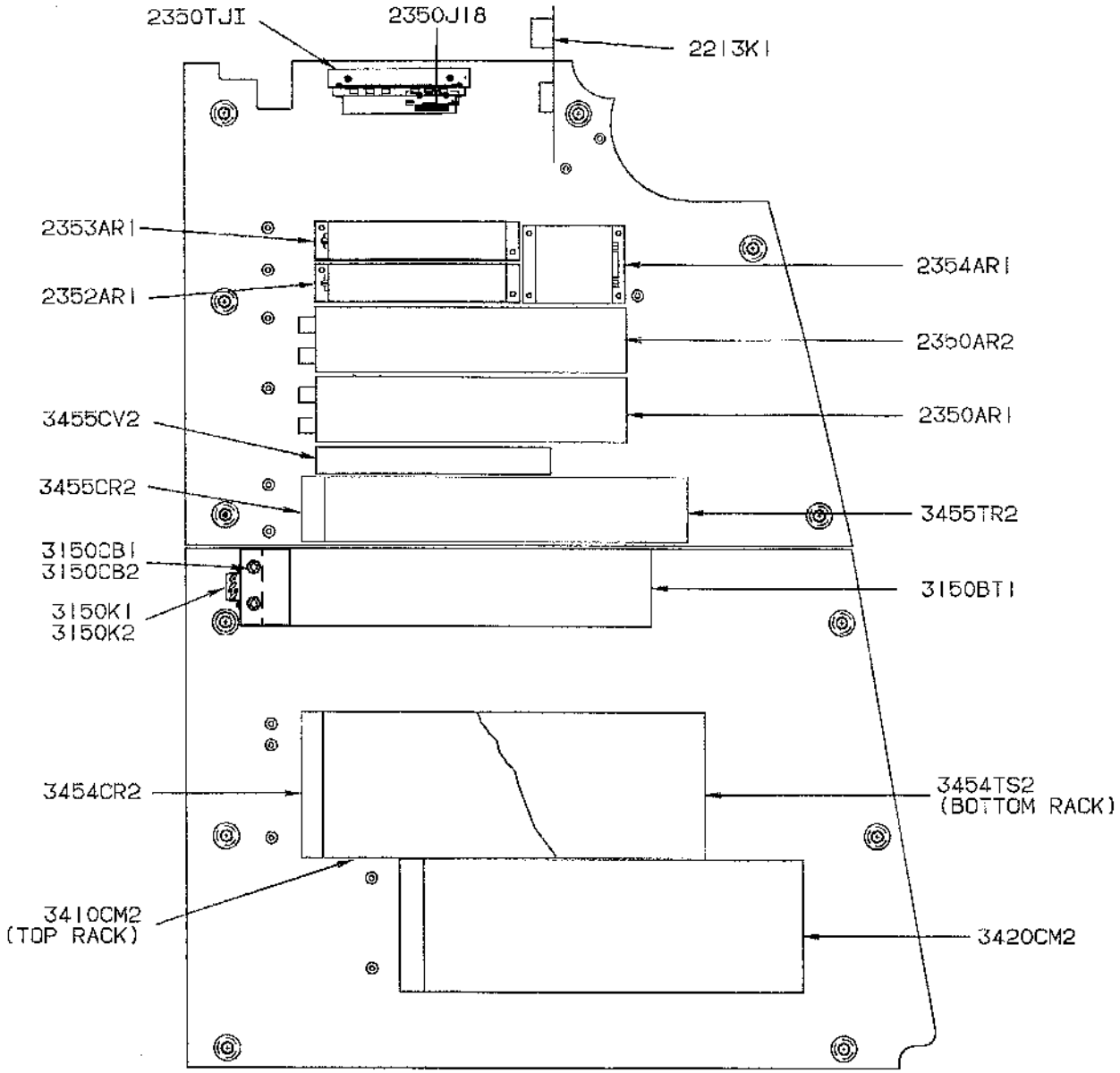
2221K1 - AUTOPILOT/TRIM SYSTEM RELAY

2221K2 - AUTOPILOT/TRIM SYSTEM RELAY

2221CR10 - AUTOPILOT/TRIM SYSTEM
2221CR12 TRANSZORBS
2221CR14
2221CR16
2221CR18
2221CR20

2000-351-072

LH Nose Avionics Compartment Electrical Equipment (Zone 213) (Effectivity: All)
(Sheet 2 of 2)
Figure 1



2000-351-073

RH Nose Avionics Compartment Electrical Equipment (Zone 214) (Effectivity: All)
(Sheet 1 of 2)
Figure 2

2350TJ1 - AUDIO BUS CONNECTOR

2350J18 - AUDIO GROUND CONNECTOR

2213K1 - AUTOPILOT PITCH SERVO/COLUMN PUSHER TRANSFER RELAY

2353AR1 - PAGING/STEREO AMPLIFIER

2352AR1 - AURAL WARNING TONE GENERATOR

2354AR1 - INTERPHONE AMPLIFIER

2350AR2 - AUDIO AMPLIFIER NO. 2

2350AR1 - AUDIO AMPLIFIER NO. 1

3455CV2 - ATC TRANSPONDER NO. 2 CONTROL ADAPTER (NC-4 THRU NC-25)

3455TR2 - ATC TRANSPONDER NO. 2

3455CR2 - ATC TRANSPONDER NO. 2 METAL OXIDE VARISTOR

3150BT1 - EICAS AUXILIARY BATTERY

3150K1 - AHRS STANDBY POWER RELAY

3150K2 - EICAS STANDBY POWER RELAY

3150CB1 - AHRS NO.1 STANDBY CIRCUIT BREAKER

3150CB2 - AHRS NO.2 STANDBY CIRCUIT BREAKER

3454TS2 - DME NO. 2 TRANSCEIVER

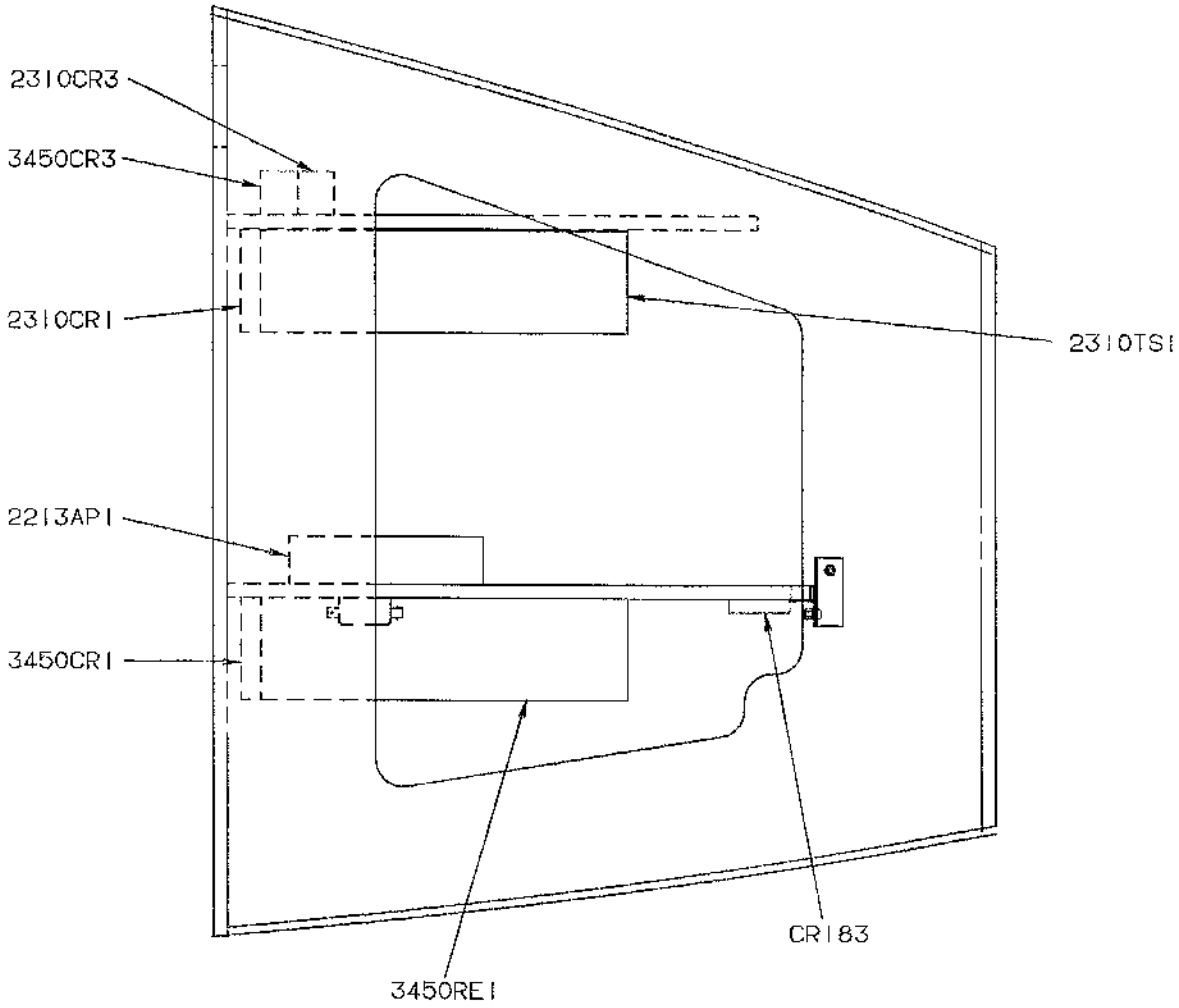
3454CR2 - DME NO. 2 TRANSCEIVER METAL OXIDE VARISTOR

3410CM2 - AIR DATA COMPUTER NO. 2

3420CM2 - ATTITUDE HEADING COMPUTER NO. 2

2000-351-074

RH Nose Avionics Compartment Electrical Equipment (Zone 214) (Effectivity: All)
(Sheet 2 of 2)
Figure 2

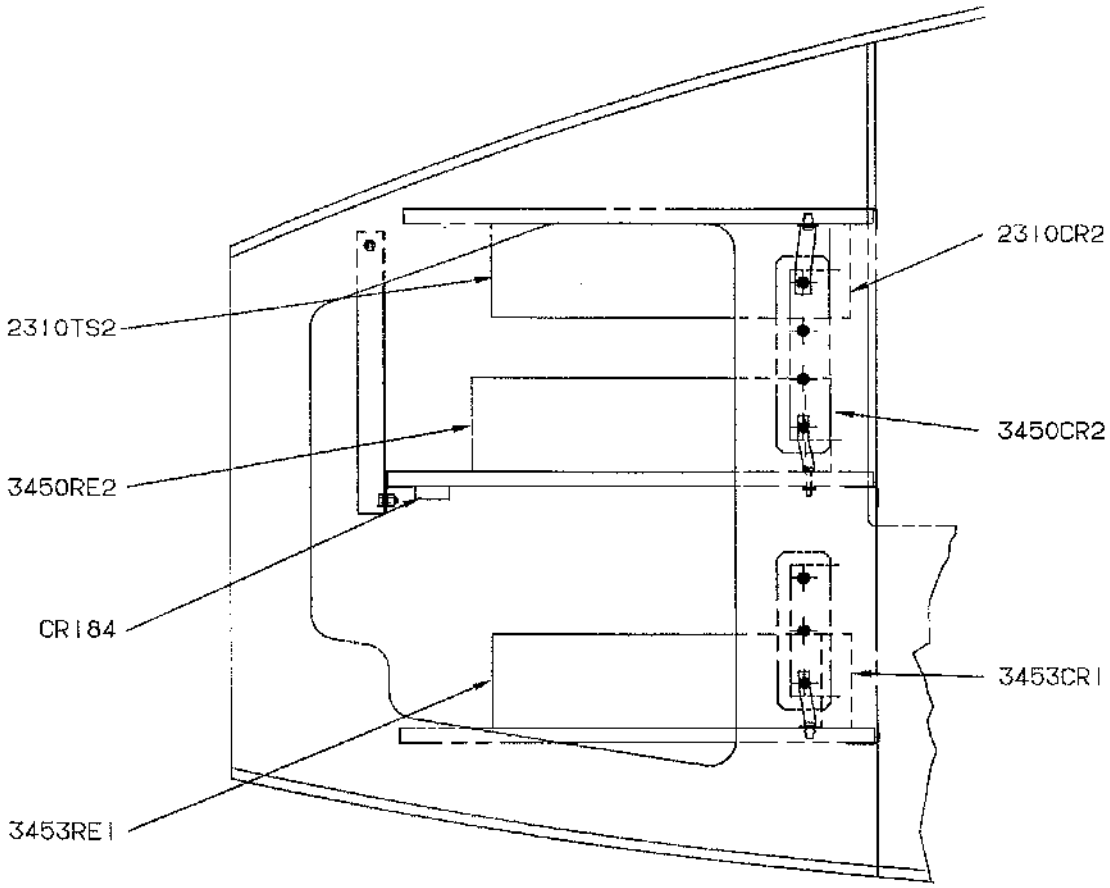


VIEW LOOKING OUTBOARD FROM INSIDE NOSE WHEEL WELL

2310CR3 - COMM NO. 1 TRANSCEIVER TRANSZORB
3450CR3 - VHF NAV NO.1 RECEIVER TRANSZORB
2310TS1 - COMM NO. 1 TRANSCEIVER
2310CR1 - COMM NO. 1 TRANSCEIVER METAL OXIDE VARISTER
2213AP1 - AUTOMATIC TRIM COUPLER
CR183 - LH PITOT TUBE HEAT METAL OXIDE VARISTOR
3450RE1 - VHF NAV NO. 1 RECEIVER
3450CR1 - VHF NAV NO. 1 RECEIVER METAL OXIDE VARISTOR

2000-351-069

LH Nose Electrical Equipment (Zone 121) (Effectivity: All)
Figure 3

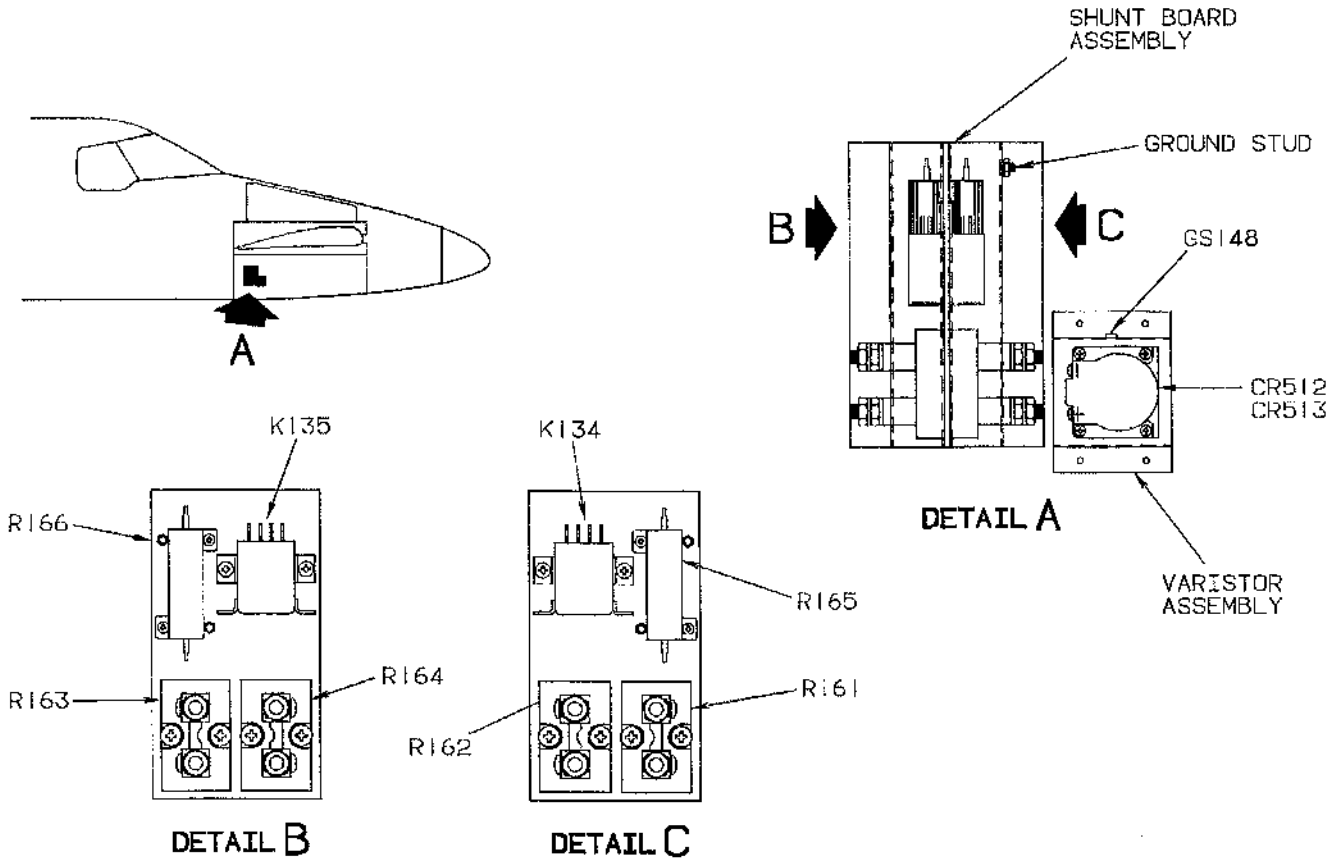


VIEW - LOOKING OUTBOARD FROM INSIDE NOSE WHEEL WELL

2310TS2 - COMM NO. 2 TRANSCEIVER
2310CR2 - COMM NO. 2 TRANSCEIVER METAL OXIDE VARISTOR
3450RE2 - VHF NAV NO. 2 RECEIVER
3450CR2 - VHF NAV NO. 2 RECEIVER METAL OXIDE VARISTOR
CR184 - RH PITOT TUBE HEAT METAL OXIDE VARISTOR
3453RE1 - ADF RECEIVER
3453CR1 - ADF RECEIVER METAL OXIDE VARISTOR

2000-351-070

RH Nose Electrical Equipment (Zone 122) (Effectivity: All)
Figure 4



SHUNT BOARD ASSEMBLY

K134 - STANDBY ICE DETECTOR TEST RELAY

K135 - MAIN ICE DETECTOR TEST RELAY

R165 - STANDBY ICE DETECTOR TEST RESISTOR

R166 - MAIN ICE DETECTOR TEST RESISTOR

R162 - STANDBY ICE DETECTOR SHUNT

R161 - STANDBY ICE DETECTOR SHUNT

R163 - MAIN ICE DETECTOR SHUNT

R164 - MAIN ICE DETECTOR SHUNT

VARISTOR ASSEMBLY

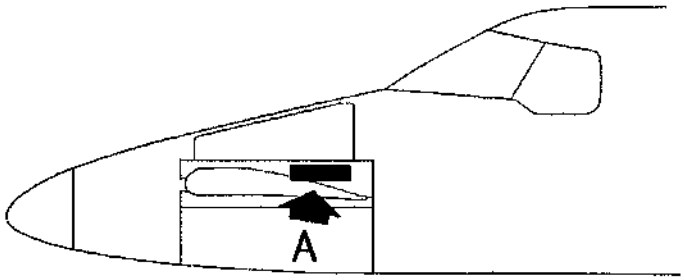
CR512 - METAL OXIDE VARISTOR (STANDBY ICE DETECTOR)

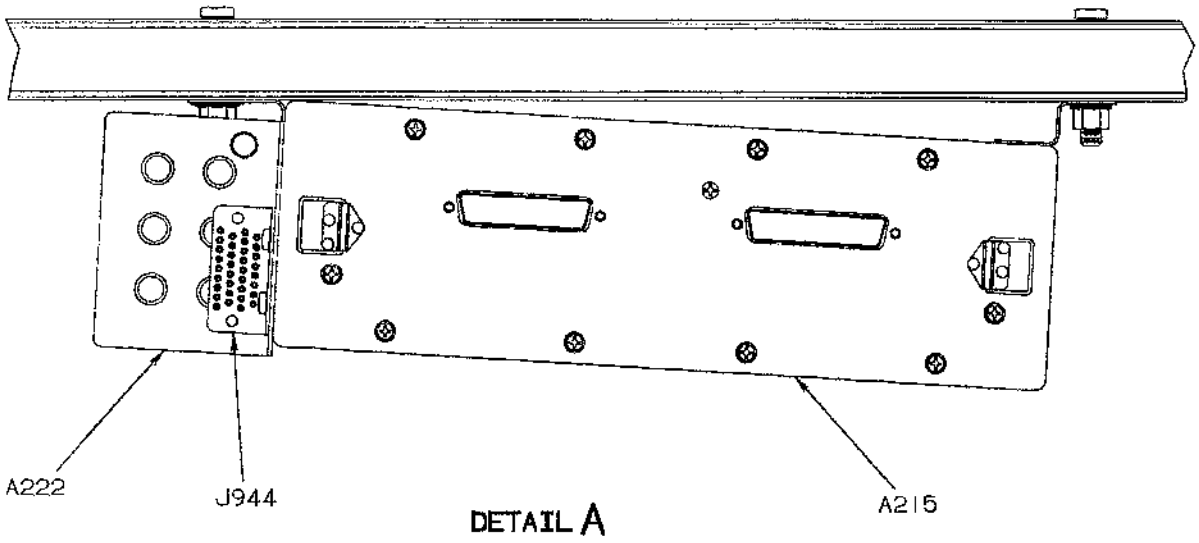
CR513 - METAL OXIDE VARISTOR (MAIN ICE DETECTOR)

GS148 - GROUND STUD

2000-351-64

Surface Deice System Electrical Equipment (Zone 132) (Effectivity: All)
Figure 5





DETAIL A

A215 - SURFACE DEICE SYSTEM PCB BOX ASSEMBLY

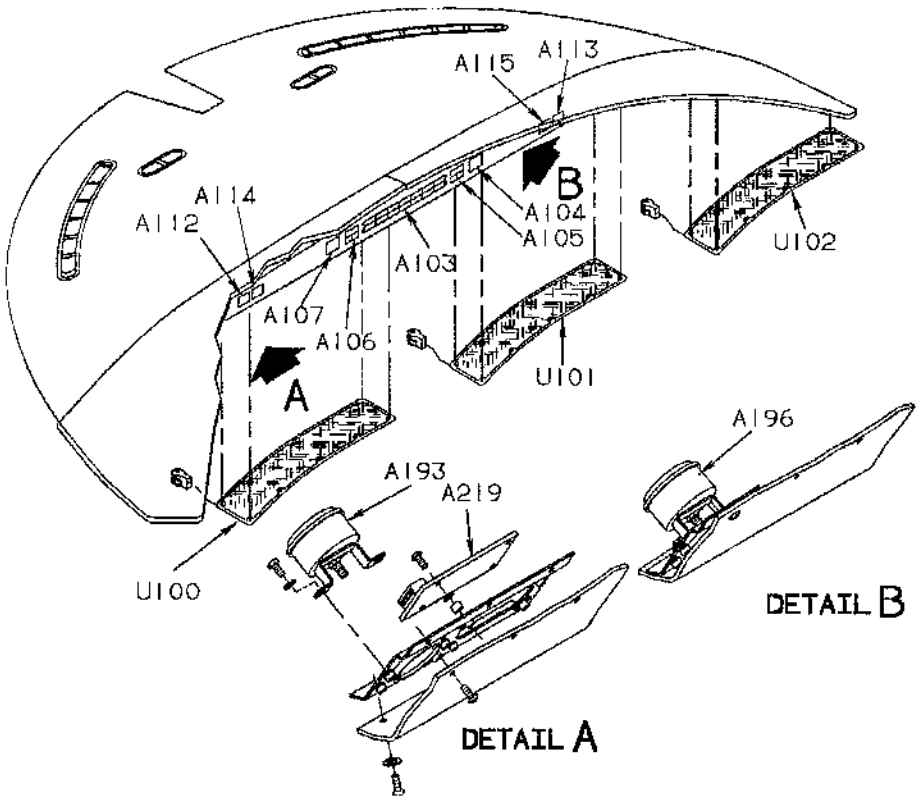
A222 - SURFACE DEICE SYSTEM FUSE PANEL

F144 - RH FORWARD WING DISTRIBUTOR VALVE
F145 - LH AFT WING DISTRIBUTOR VALVE
F146 - RH AFT WING DISTRIBUTOR VALVE
F147 - RH FORWARD WING DISTRIBUTOR VALVE
F148 - LH AFT WING DISTRIBUTOR VALVE
F149 - RH AFT WING DISTRIBUTOR VALVE

J944 - GROUND BLOCK CONNECTOR

2000-351-063

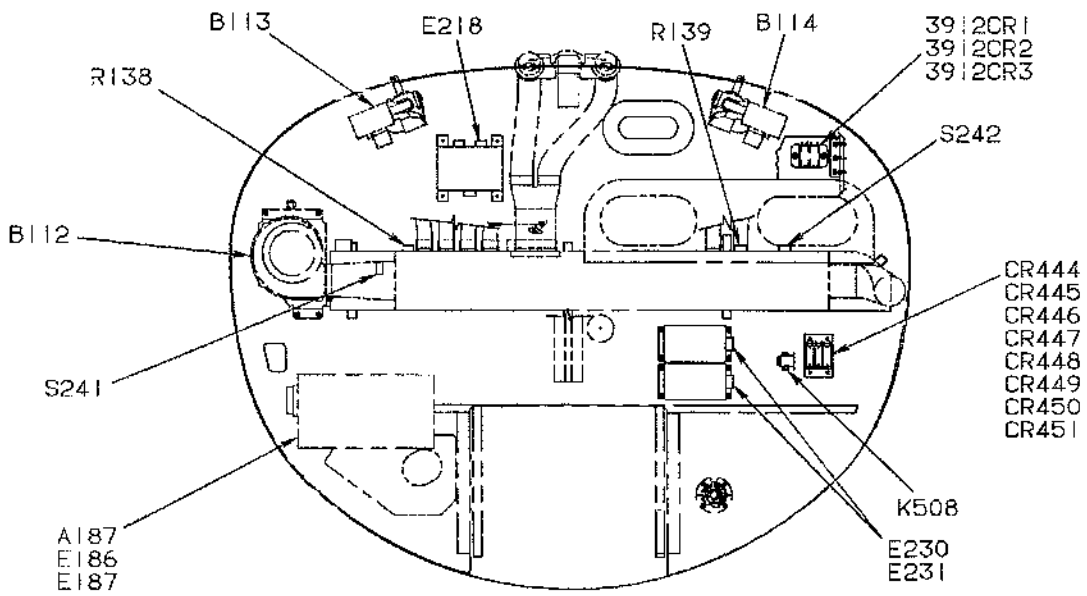
Surface Deice Electrical Equipment (Zone 211) (Effectivity: All)
Figure 6



A103 - WARNING ANNUNCIATOR PANEL
A104 - RH ENGINE FIRE AND FIREWALL FUEL SHUTOFF VALVE CLOSED SWITCH AND ANNUNCIATORS
A105 - RH ENGINE FIRE EXTINGUISHER LIGHT/SWITCH AND ANNUNCIATORS
A106 - LH ENGINE FIRE EXTINGUISHER LIGHT/SWITCH AND ANNUNCIATORS
A107 - LH ENGINE FIRE AND FIREWALL FUEL SHUTOFF VALVE CLOSED SWITCH AND ANNUNCIATORS
A112 - PILOT'S MASTER WARNING LIGHT/SWITCH
A113 - COPILOT'S MASTER WARNING LIGHT/SWITCH
A114 - PILOT'S MASTER CAUTION LIGHT/SWITCH
A115 - COPILOT'S MASTER CAUTION LIGHT/SWITCH
U100 - INSTRUMENT INDIRECT LIGHTING PANEL
U101 - INSTRUMENT INDIRECT LIGHTING PANEL
U102 - INSTRUMENT INDIRECT LIGHTING PANEL
A193 - LH STALL WARNING HORN
A196 - RH STALL WARNING HORN
A219 - COLUMN PUSHER TIME DELAY PCB

2000-351-017

Bareshield Electrical Equipment (Zone 257) (Effectivity: All)
Figure 7



A187 - CABIN PRESSURIZATION MODULE ASSEMBLY

E186 - CABIN PRESSURIZATION PRESET SOLENOID "B"

E187 - CABIN PRESSURIZATION DUMP SOLENOID "C"

S241 - LH FORWARD DUCT OVERTEMPERATURE SWITCH

B112 - AVIONICS ALTERNATE BLOWER

R138 - LH FORWARD DUCT TEMPERTURE SENSOR

B113 - LH WINDSHIELD WIPER MOTOR CONVERTER ASSEMBLY

E218 - WINDSHIELD WIPER MOTOR SPEED CONTROL ASSEMBLY

B114 - RH WINDSHIELD WIPER MOTOR CONVERTER ASSEMBLY

R139 - RH FORWARD DUCT OVERTEMPERATURE SENSOR

S242 - RH FORWARD DUCT OVERTEMPERATURE SENSOR

3912CR1 - RTU NO.1 TRANSZORB

3912CR2 - RTU NO.2 TRANSZORB

3912CR3 - SENSOR DISPLAY UNIT TRANSZORB

CR444 THRU CR451 - METAL OXIDE VARISTORS

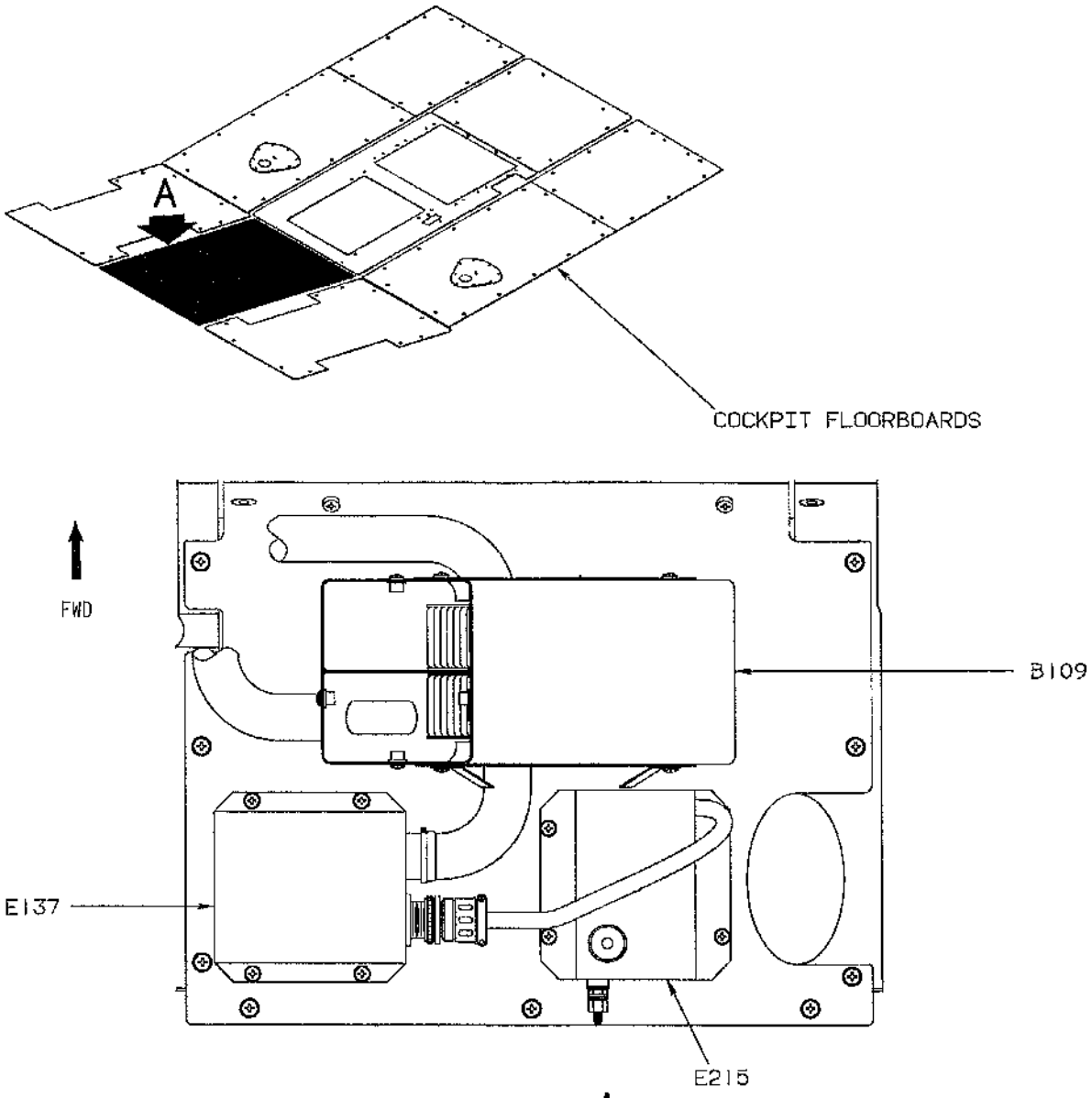
K508 - COLUMN PUSHER DISCONNECT RELAY

E230 - RH STALL WARNING COMPUTER

E231 - LH STALL WARNING COMPUTER

2000-351-068

Forward Pressure Bulkhead Electrical Equipment (Zones 221 and 222) (Effectivity: All)
Figure 8



DETAIL A

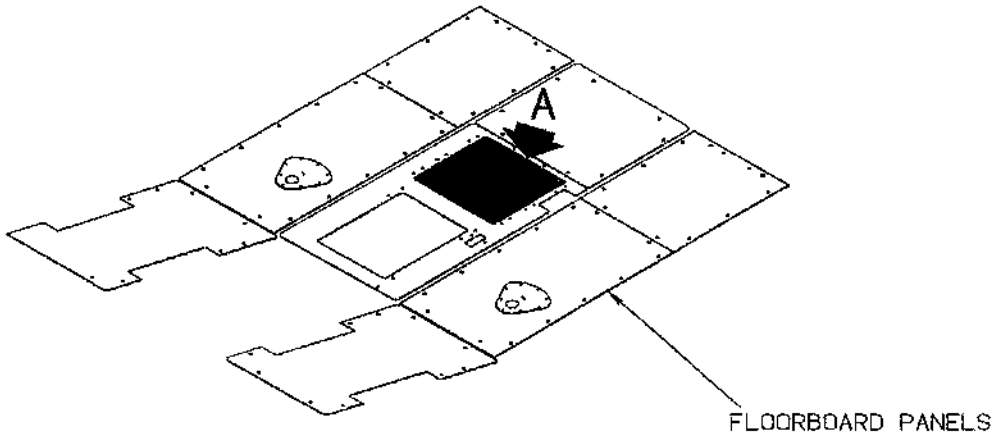
B109 - FORWARD VENT BLOWER ASSEMBLY

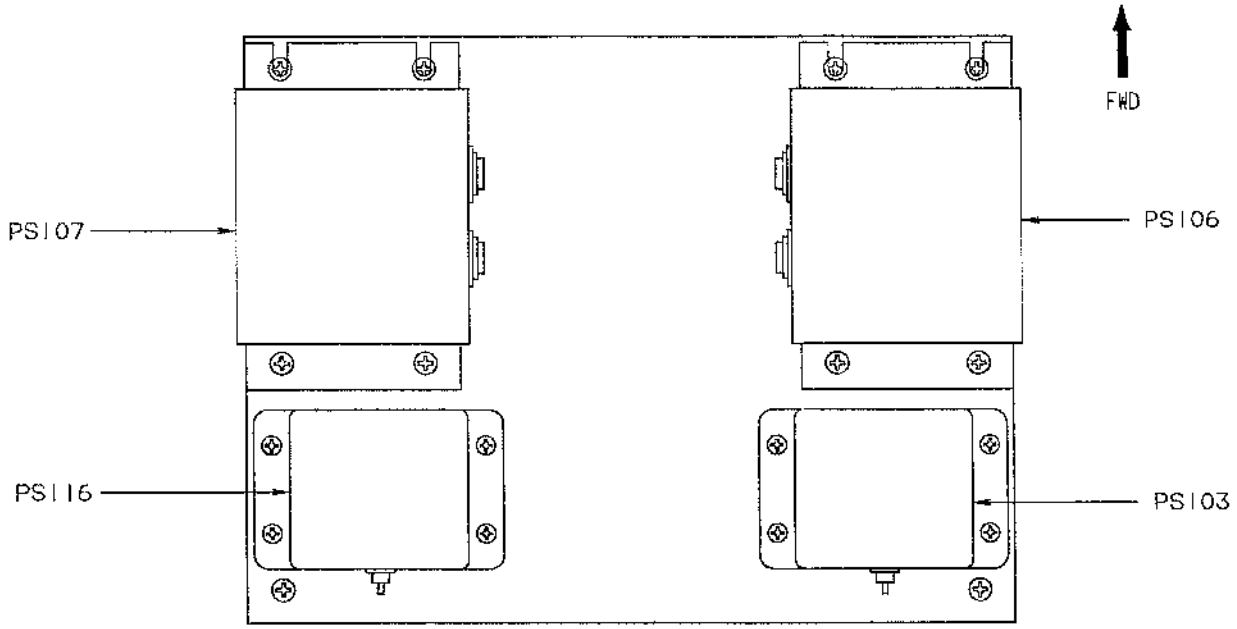
E137 - FORWARD VENT BLOWER SPEED CONTROL

E215 - FORWARD VENT BLOWER EMI FILTER

2000-361-065

Cockpit Center Floor Electrical Equipment (Zone 143) (Effectivity: All)
Figure 9





DETAIL A

PS103 - GLARESHIELD ELECTROLUMINESCENT LIGHT PANEL POWER SUPPLY

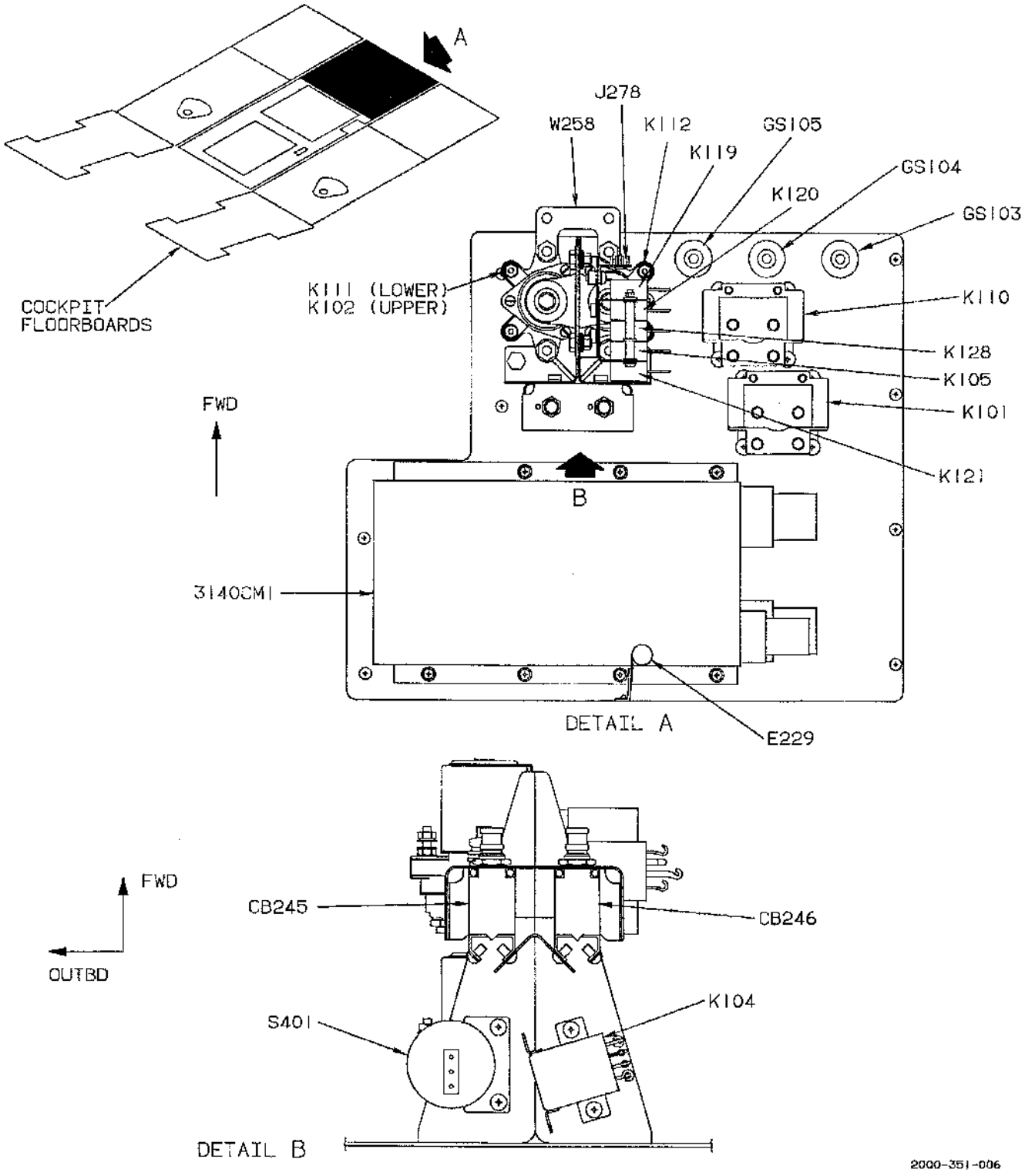
PS106 - SUBPANEL ELECTROLUMINESCENT PANEL POWER SUPPLY

PS107 - OVERHEAD AND CIRCUIT BREAKER ELECTROLUMINESCENT PANELS POWER SUPPLY

PS116 - INSTRUMENT PANEL ELECTROLUMINESCENT PANEL SUPPLY

2000-351-066

Cockpit Center Floor Electrical Equipment (Zone 143) (Effectivity: All)
Figure 10



Cockpit Center Floor Electrical Equipment (Zone 143) (Effectivity: All)
(Sheet 1 of 2)
Figure 11

K111 - COPILOT'S WINDSHIELD ANTI-ICE HIGH HEAT RELAY

K102 - PILOT'S WINDSHIELD ANTI-ICE HIGH HEAT RELAY

W258 - BUS BAR

J278 - COCKPIT GROUND BLOCK CONNECTOR

K112 - COPILOT'S WINDSHIELD ANTI-ICE LOW HEAT RELAY

K119 - RIGHT PITOT HEAT RELAY

K120 - LEFT PITOT HEAT RELAY

GS105 - GROUND STUD

GS104 - GROUND STUD

GS103 - GROUND STUD

K110 - PILOT'S WINDSHIELD ANTI-ICE REDUNDANT RELAY

K101 - PILOT'S WINDSHIELD ANTI-ICE LOW HEAT RELAY

K128 - NOSE GEAR LANDING LIGHT RELAY

K105 - RH WING LANDING LIGHT RELAY

K121 - LH WING LANDING LIGHT RELAY

E229 - G SWITCH (STALL WARNING SYSTEM)

3140CM1 - DATA ACQUISITION UNIT

CB245 - PILOT'S SIDE WINDOW HEAT CIRCUIT BREAKER

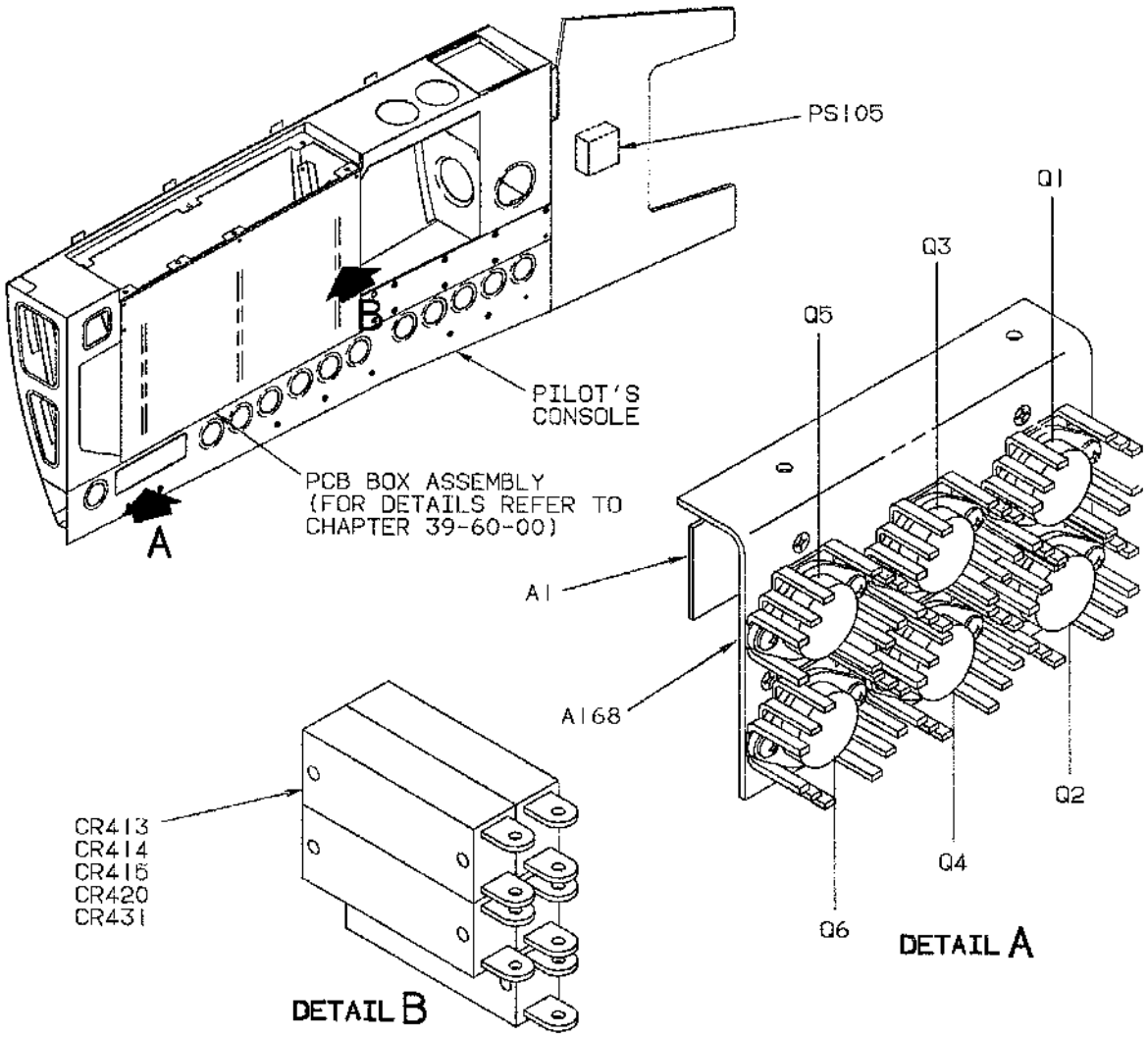
CB246 - COPILOT'S SIDE WINDOW HEAT CIRCUIT BREAKER

S401 - CABIN ALTITUDE WARNING SWITCH

K104 - TAXI LIGHT RELAY

2000-351-084

Cockpit Center Floor Electrical Equipment (Zone 143) (Effectivity: All)
(Sheet 2 of 2)
Figure 11



A168 - LIGHT DIMMING TRANSISTOR PANEL

A1 - LIGHT DIMMING PCB
CR1 THRU CR15 - METAL OXIDE VARISTORS

Q1 THRU Q6 - DIMMING TRANSISTORS

PS105 - COCKPIT LCD'S DIMMER CONTROL POWER SUPPLY

CR413 - POWER DISTRIBUTION CONTROL PCB TRANSZORB

CR414 - RH FLAP/FWD WING MONITOR & MONITOR TEST PCB TRANSZORB

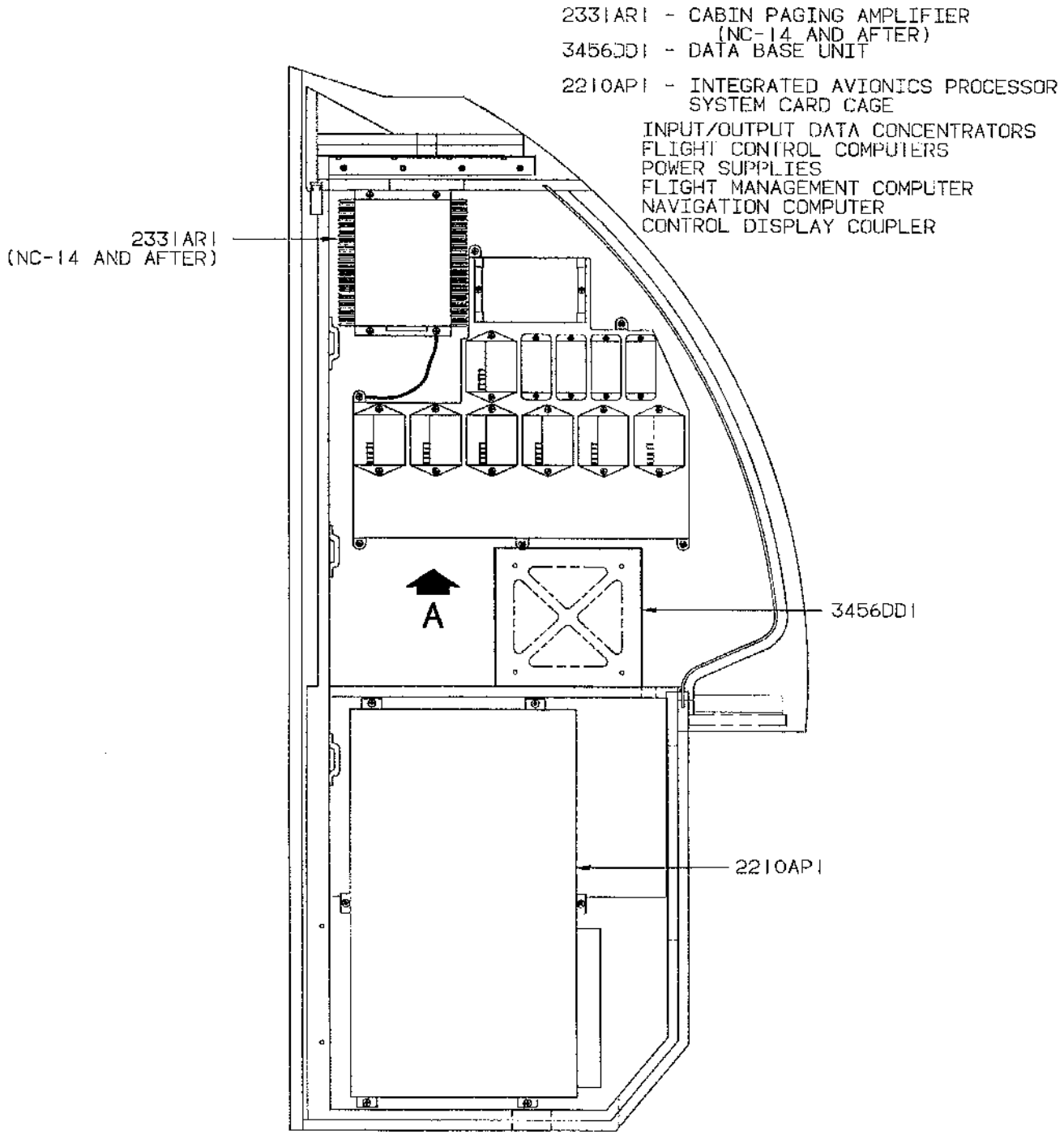
CR415 - LH FLAP/FWD WING MONITOR & MONITOR TEST PCB TRANSZORB

CR420 - POWER DISTRIBUTION CONTROL PCB TRANSZORB

CR431 - ANNUNCIATOR CONTROL PCB TRANSZORB

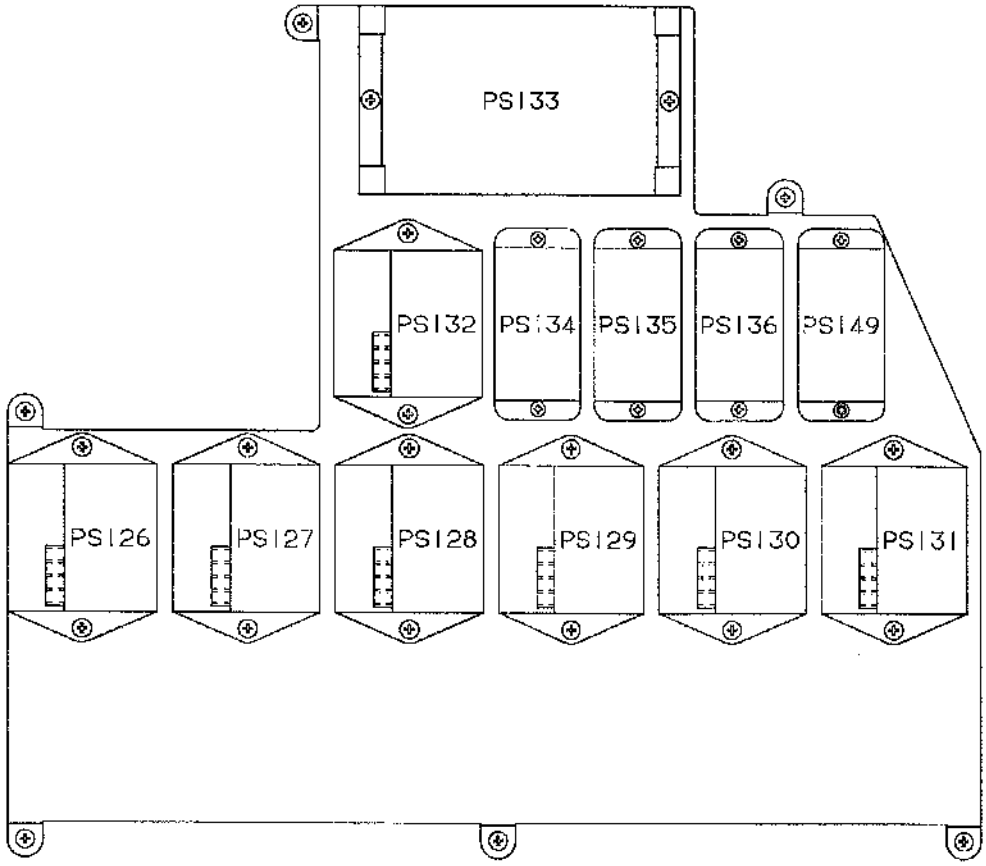
2000-351-026

Pilot's Console Electrical Equipment (Zone 235) (Effectivity: All)
Figure 12



2000-352-017

RH Forward Baggage Compartment Electrical Equipment (Effectivity: NC-4 thru NC-20 without BEECHCRAFT Kit No. 122-3010-1 installed)
(Sheet 1 of 2)
Figure 13

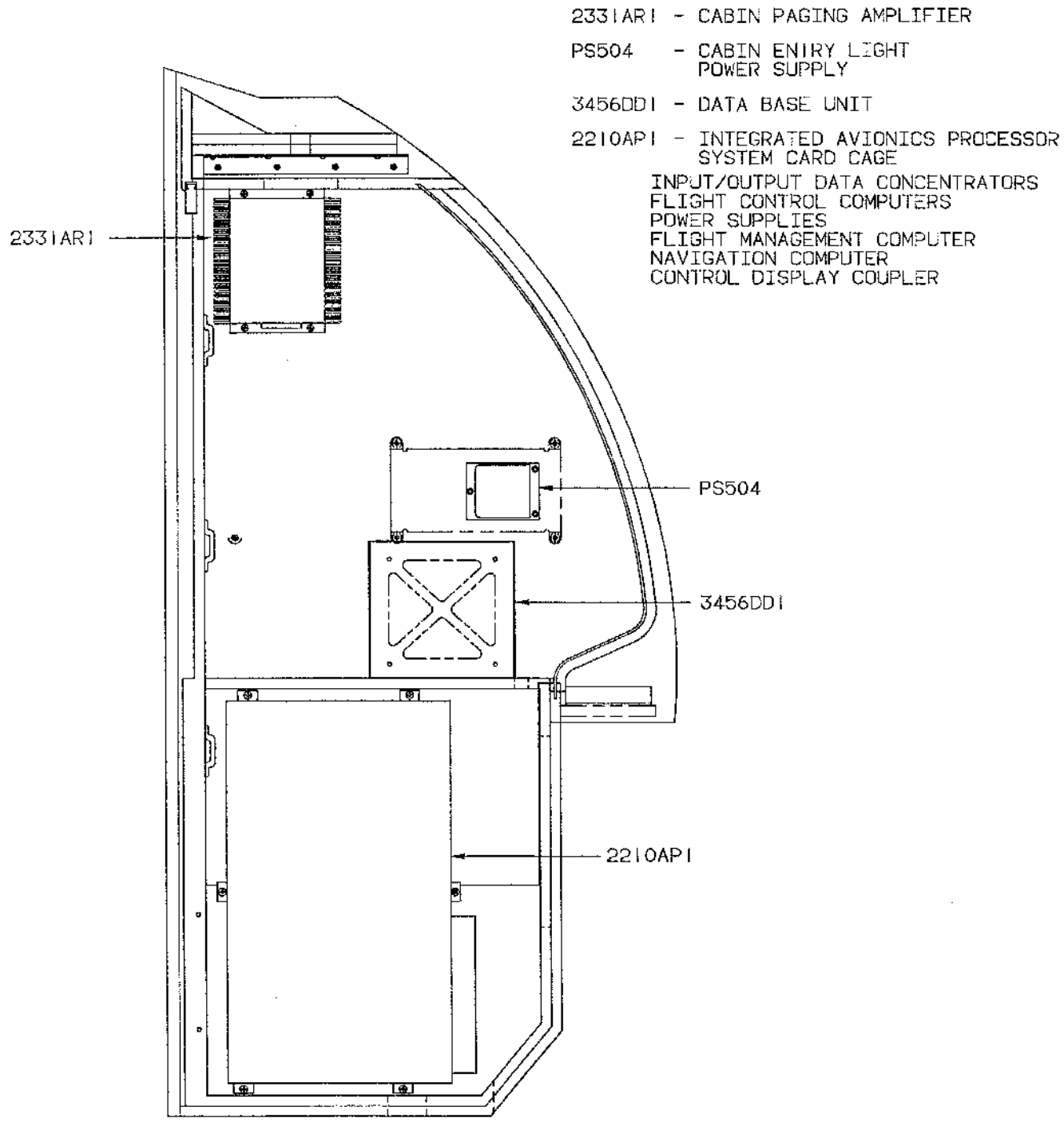


DETAIL A

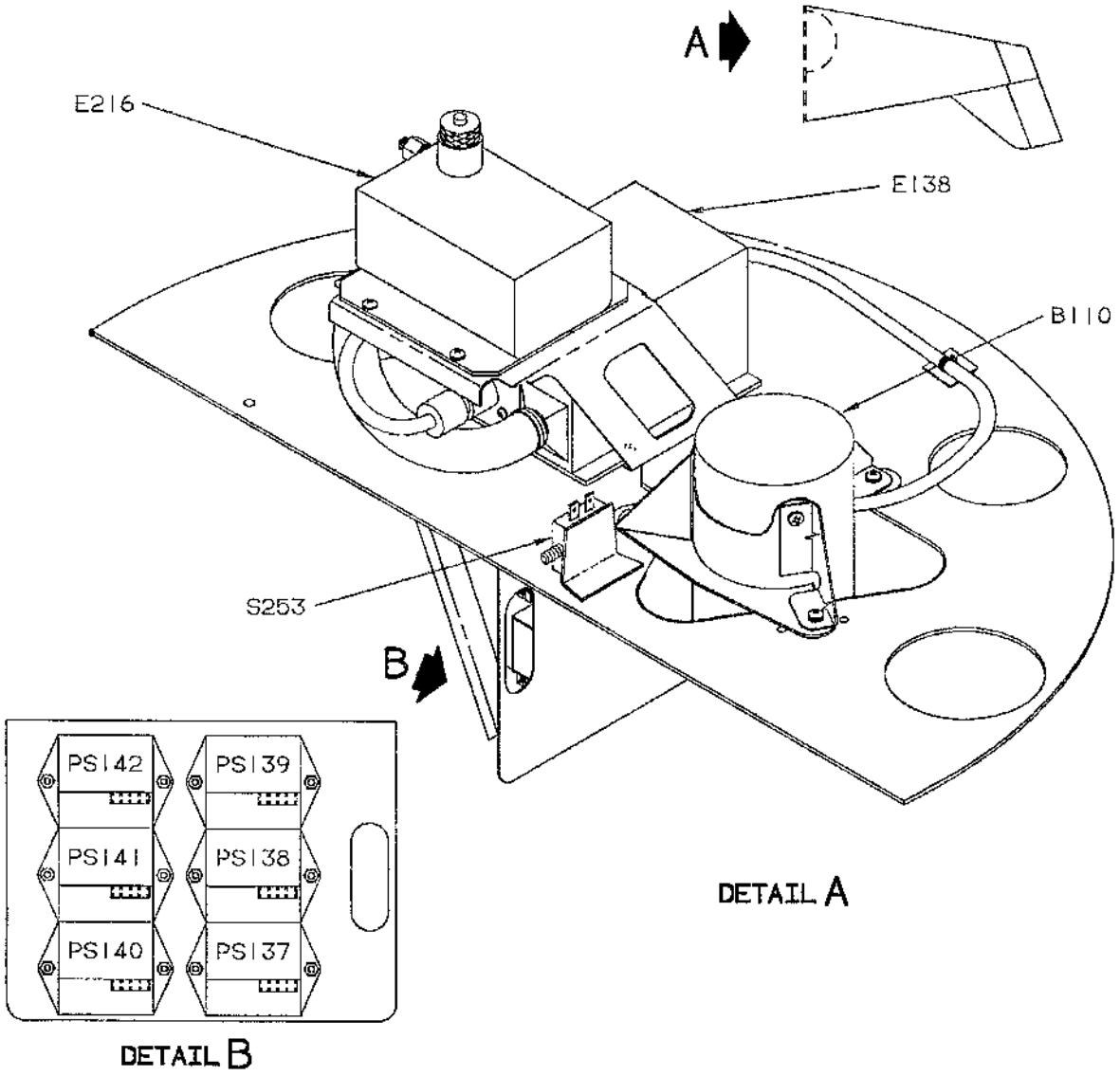
PS133 - CABIN OVERHEAD FLUORESCENT LIGHTS POWER SUPPLY
PS132 - CABIN ENTRY LIGHTS POWER SUPPLY
PS134 - CABIN OVERHEAD FLUORESCENT LIGHTS BALLAST
PS135 - CABIN OVERHEAD FLUORESCENT LIGHTS BALLAST
PS136 - CABIN OVERHEAD FLUORESCENT LIGHTS BALLAST
PS149 - CABIN OVERHEAD FLUORESCENT LIGHTS BALLAST
PS126 - CABIN READING LIGHT POWER SUPPLY
PS127 - CABIN TABLE LIGHT POWER SUPPLY
PS128 - CABIN READING LIGHT POWER SUPPLY
PS129 - CABIN READING LIGHT POWER SUPPLY
PS130 - CABIN TABLE LIGHT POWER SUPPLY
PS131 - CABIN READING LIGHT POWER SUPPLY

2000-352-018

RH Forward Baggage Compartment Electrical Equipment (Effectivity: NC-4 thru NC-20 without BEECHCRAFT Kit No. 122-3010-1 installed)
(Sheet 2 of 2)
Figure 13



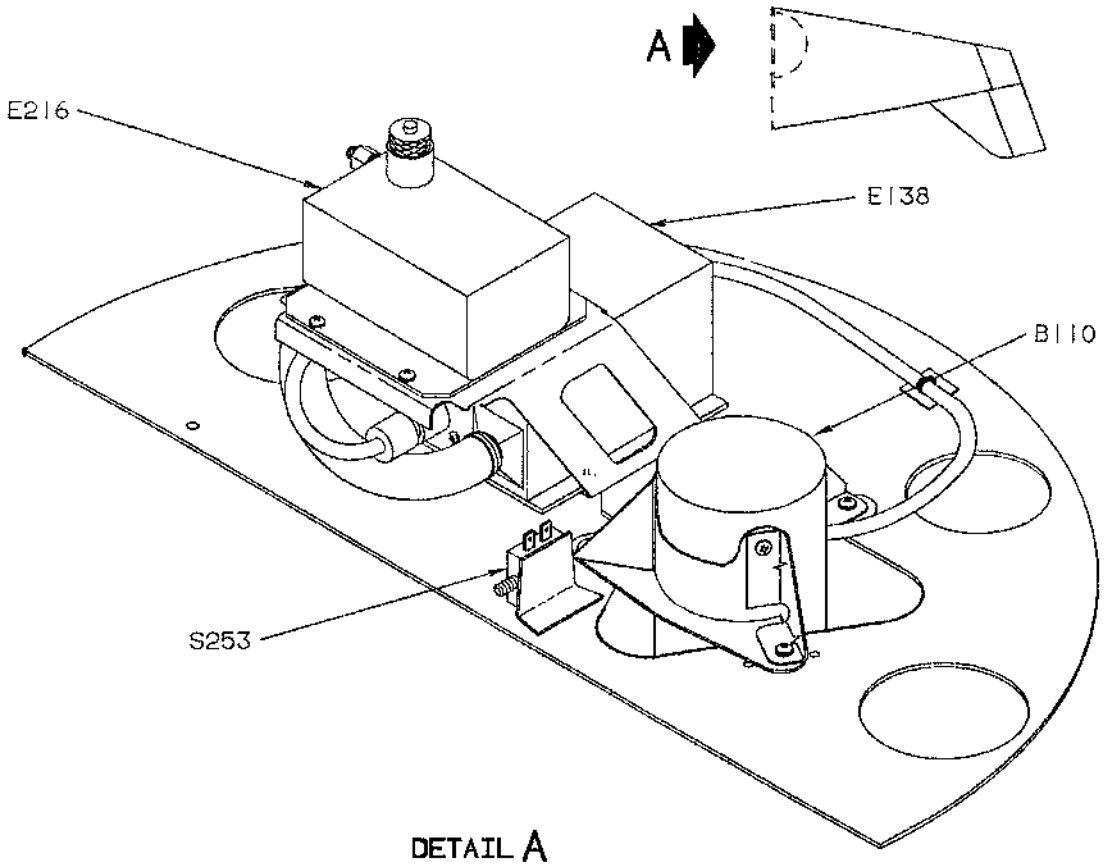
RH Forward Baggage Compartment Electrical Equipment (Effectivity: NC-21 and after and prior airplanes with BEECHCRAFT Kit No. 122-3010-1 installed)
Figure 14



B110 - AFT VENT BLOWER ASSEMBLY
E138 - AFT VENT BLOWER SPEED CONTROL
E216 - AFT VENT BLOWER EMI FILTER
S253 - AFT VENT BLOWER PRESSURE SWITCH
PS142 - CABIN READING LIGHT POWER SUPPLY
PS139 - CABIN READING LIGHT POWER SUPPLY
PS141 - CABIN TABLE LIGHT POWER SUPPLY
PS138 - CABIN TABLE LIGHT POWER SUPPLY
PS140 - CABIN READING LIGHT POWER SUPPLY
PS137 - CABIN READING LIGHT POWER SUPPLY

2000-352-016

Aft Pressure Dome Electrical Equipment (Zone 295) (Effectivity: NC-4 thru NC-20 without BEECHCRAFT Kit No. 122-3010-1 installed)
Figure 15

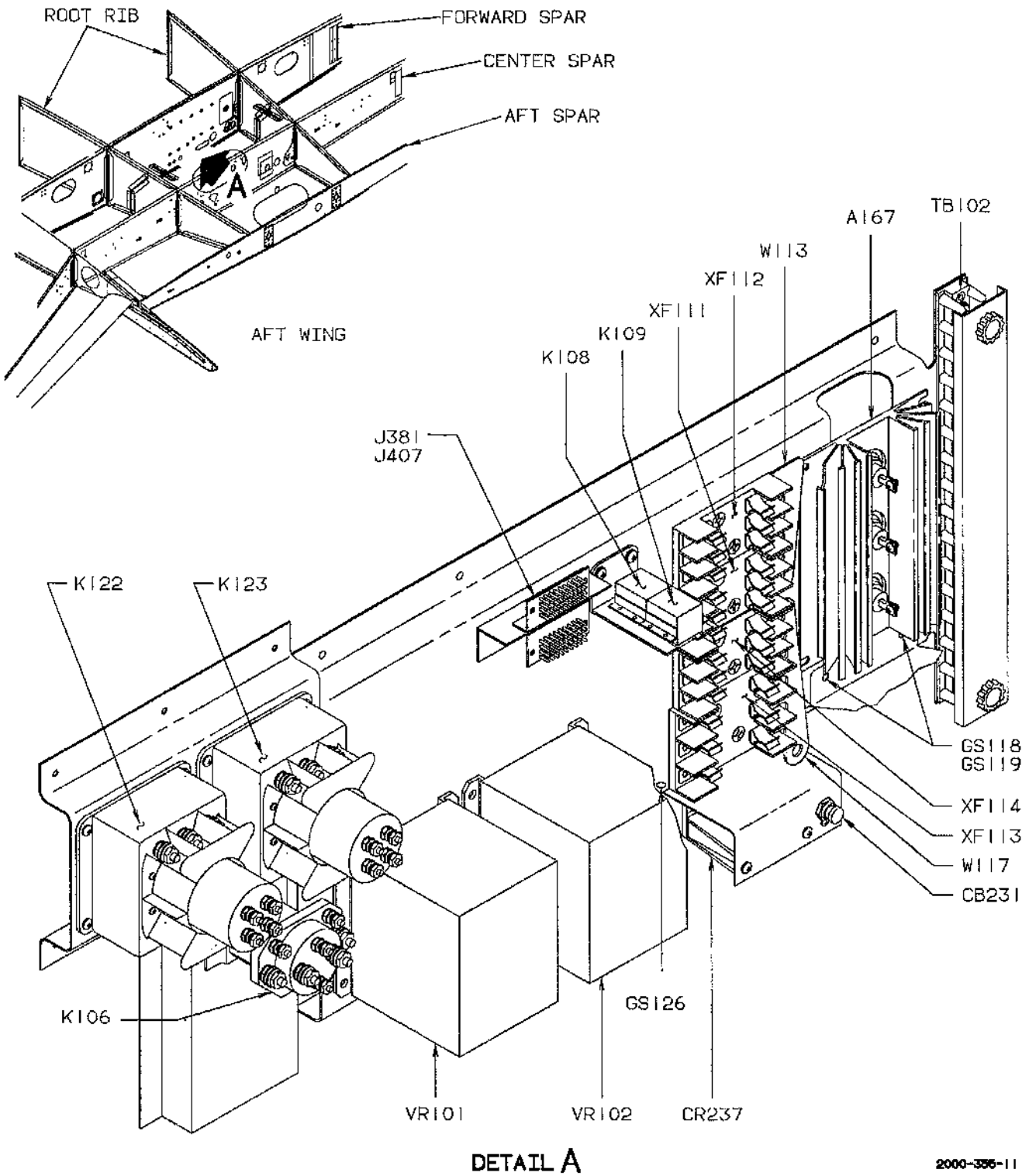


DETAIL A

B110 - AFT VENT BLOWER ASSEMBLY
E138 - AFT VENT BLOWER SPEED CONTROL
E216 - AFT VENT BLOWER EMI FILTER
S253 - AFT VENT BLOWER PRESSURE SWITCH

C94NC3980498

Aft Pressure Dome Electrical Equipment (Zone 295) (Effectivity: NC-21 and after and prior airplanes with BEECHCRAFT Kit No. 122-3010-1 installed)
Figure 16



DETAIL A

2000-356-11

Wing Center Section Electrical Equipment (Zone 547) (Effectivity: All)
(Sheet 1 of 2)
Figure 17

A167 - TRIPLE FED BUS DIODE ASSEMBLY
CR1 - DIODE (LH GENERATOR)
CR2 - DIODE (RH GENERATOR)
CR3 - DIODE (BATTERY)

K106 - FORWARD WING POWER RELAY

K108 - LH GENERATOR FIELD AND SENSE RELAY

K109 - RH GENERATOR FIELD AND SENSE RELAY

K122 - RH FLAP MONITOR RELAY

K123 - LH FLAP MONITOR RELAY

VR101 - GENERATOR CONTROL UNIT

VR102 - GENERATOR CONTROL UNIT

TB102 - TERMINAL BOARD

J381 - LEFT CENTER WING GROUND BLOCK CONNECTOR

J407 - RIGHT CENTER WING GROUND BLOCK CONNECTOR

XF112 - LIMITER BASE (TRIPLE FED BUS)
F132 - NO. 1 AVIONICS BUS LIMITER
F131 - NO. 1 SUBBUS FEEDER LIMITER
F130 - NO. 2 SUBBUS FEEDER LIMITER

XF111 - LIMITER BASE (TRIPLE FED BUS)
F129 - NO. 3 SUBBUS FEEDER LIMITER
F128 - NO. 4 SUBBUS FEEDER LIMITER
F127 - NO. 5 SUBBUS FEEDER LIMITER

XF114 - LIMITER BASE (CENTER BUS)
F125 - FORWARD WING MOTOR LIMITER
F124 - FORWARD SUBBUS FEEDER LIMITER
F138 - AFT SUBBUS FEEDER LIMITER

XF113 - LIMITER BASE (CENTER BUS)
F139 - FLAP LIMITER
F140 - EICAS POWER LIMITER

CB231 - CENTER BUS MOV CIRCUIT BREAKER

CR237 - METAL OXIDE VARISTOR (CENTER BUS)

W113 - BUS BAR (TRIPLE FED BUS)

W117 - BUS BAR (CENTER BUS)

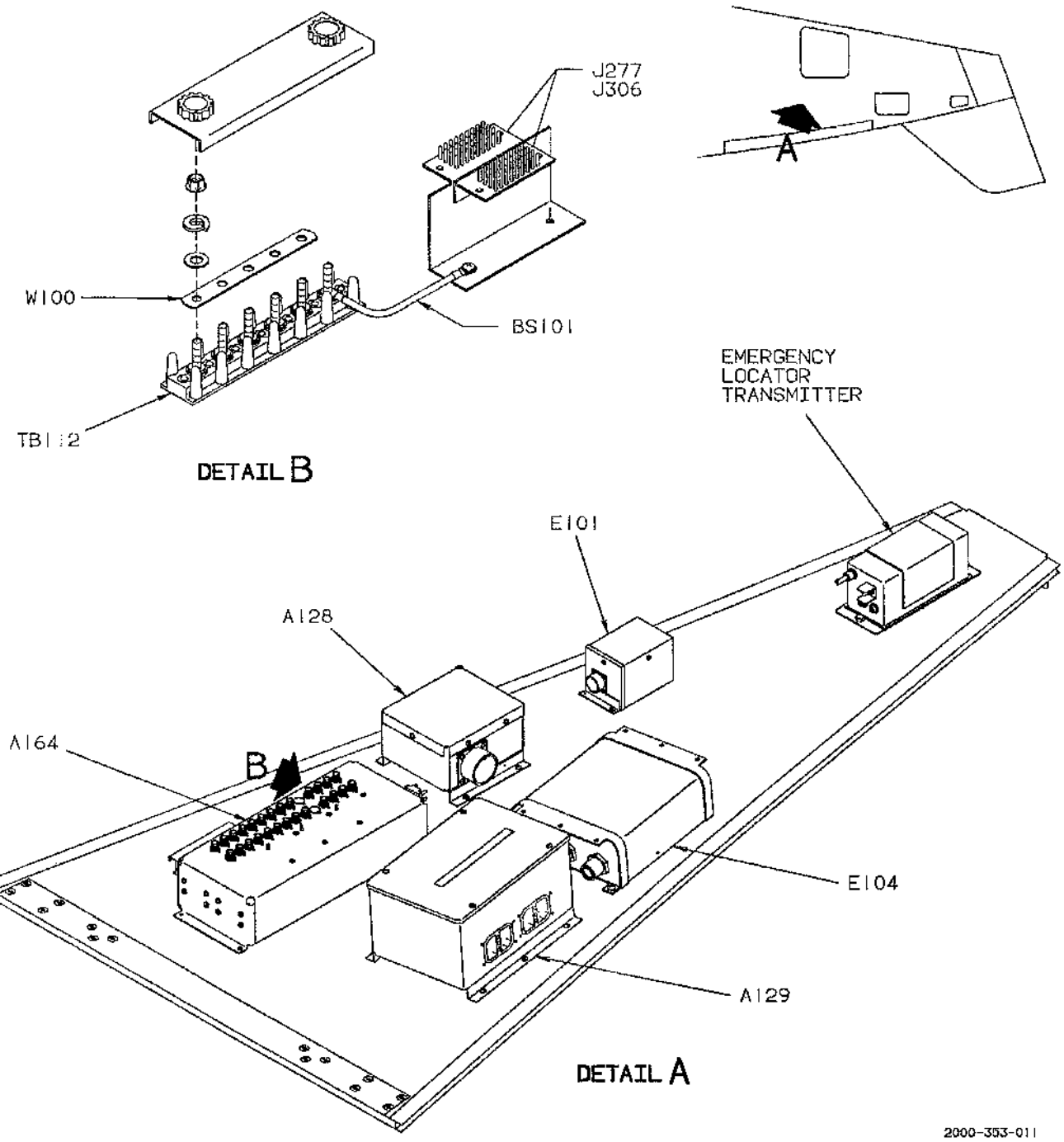
GS118 - GROUND STUD

GS119 - GROUND STUD

GS126 - GROUND STUD

2000-355-042

Wing Center Section Electrical Equipment (Zone 547) (Effectivity: All)
(Sheet 2 of 2)
Figure 17



Aft Fuselage Electrical Equipment (Zone 320) (Effectivity: All)
(Sheet 1 of 2)
Figure 18

A128 - AVIONICS POWER RELAY PANEL ASSEMBLY
- K1 - AVIONICS NO. 1 POWER RELAY
- K2 - AVIONICS NO. 2 POWER RELAY
- K3 - AVIONICS NO. 3 POWER RELAY
- K4 - AVIONICS NO. 4 POWER RELAY
- CR1 THRU CR4 - DIODES

A129 - AFT RELAY PANEL ASSEMBLY
- K1 THRU K23 - RELAYS
- K25 THRU 34 - RELAYS
- CR1 THRU CR4 - DIODES
- CR7 THRU CR18 - DIODES

A164 - AFT CIRCUIT BREAKER PANEL
- CB1 - STORM LIGHTS
- CB2 - READING LIGHTS
- CB3 - TOILET
- CB4 - CIGARETTE LIGHTER
- CB5 - CABIN FLUORESCENT LIGHT (NC-21 AND AFTER)
- CB6 - HEATED OVERBOARD DRAIN
- CB7 - DC TEST JACK
- CB8 - CONDENSER BLOWER CONTROL
- CB9 - CABIN LIGHTS
- CB10 - OVERHEAD FLOODLIGHTS
- CB11 - INSTRUMENT INDIRECT LIGHTS
- CB12 - LH EDC POWER (CHANNEL B)
- CB13 - RH EDC POWER (CHANNEL B)
- CB14 - LH SQUAT SWITCH
- CB15 - RH SQUAT SWITCH
- CB16 - FLIGHT INSTRUMENT LIGHTS
- CB17 - LCD DISPLAY LIGHTS
- CB18 - AVIONICS COPILOT BUS FEEDER
- CB19 - RH WINDSHIELD WIPER
- CB20 - OVERHEAD, SUBPANEL AND CONSOLE LIGHTS
- CB21 - A/C CLUTCH
- CB23 - WINDOW SHADE MOTOR (NC-4 THRU NC-20)
- CB24 - LH WINDSHIELD WIPER
- CB27 - ALT BLOWER POWER
- CB28 - ALT BLOWER CONTRCL
- GS1 - GROUND STUD
- GS2 - GROUND STUD
- GS3 - GROUND STUD
- K2 - ALT BLOWER RELAY
- CR3 - METAL OXIDE VARISTOR
- CR4 - METAL OXIDE VARISTOR
- CR5 - METAL OXIDE VARISTOR
- CR6 - DIODE
- W1 THRU W4 - BUS BARS

E104 - ENVIRONMENTAL SYSTEM CONTROL BOX

E101 - BLEED AIR LEAK DETECTION CONTROLLER

TB112 - TERMINAL BOARD

W100 - BUS BAR

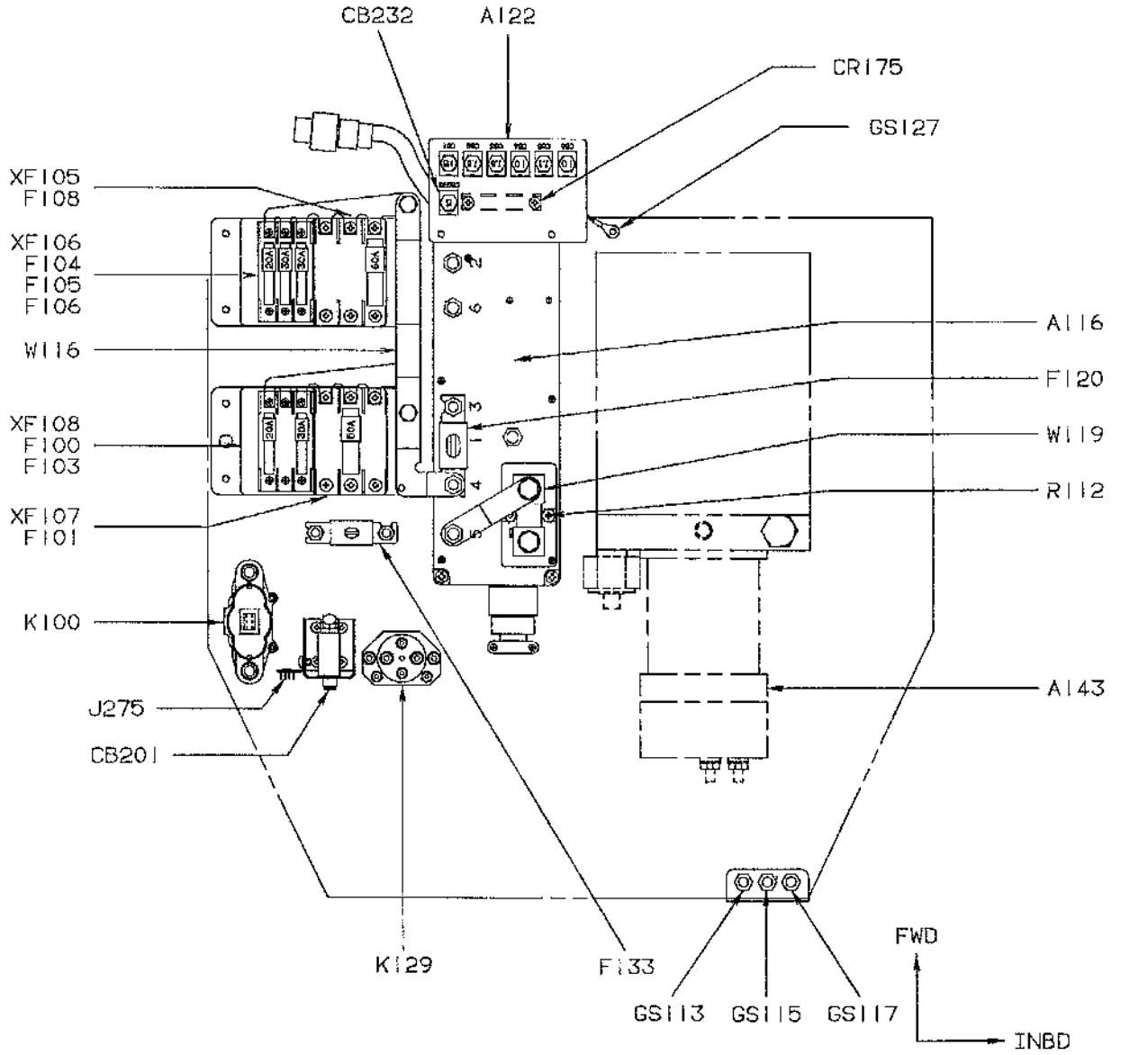
J277 - TAIL GROUND BLOCK NO. 1 CONNECTOR

J306 - TAIL GROUND BLOCK NO. 2 CONNECTOR

BS101 - BONDING STRAP

2000-353-012

Aft Fuselage Electrical Equipment (Zone 320) (Effectivity: All)
(Sheet 2 of 2)
Figure 18



LH Nacelle Electrical Equipment (Zone 583) (Effectivity: NC-4 thru 29)
(Sheet 1 of 2)
Figure 19

A116 - LH GENERATOR POWER RELAY PANEL
CURRENT SENSOR
LH BUS TIE RELAY
LINE CONTACTOR RELAY
LH ENGINE START RELAY

A122 - LH NACELLE CIRCUIT BREAKER PANEL
CB1 - LH GENERATOR AND FIELD SENSE
CB2 - LH GENERATOR LOAD METER
CB3 - LH LOAD METER
CB4 - LH GENERATOR CONTROL UNIT
CB5 - LH GENERATOR VOLTAGE
CB6 - LH BUS SENSE
W1 - BUS BAR

F120 - LH BUS TIE LIMITER

R112 - LH LOAD METER SHUNT

W116 - BUS BAR

W119 - BUS BAR

CR175 - LH GENERATOR BUS METAL OXIDE VARISTOR

CB232 - METAL OXIDE VARISTOR CIRCUIT BREAKER

GS127 - GROUND STUD

CB201 - LANDING GEAR POWER CIRCUIT BREAKER

A143 - HYDRAULIC LANDING GEAR POWER PACK

XF105 - LIMITER BASE
F108 - LH GENERATOR TRIPLE FED BUS FEEDER

XF106 - LIMITER BASE
F104 - NO.3 SUBBUS FEEDER
F105 - NO.2 SUBBUS FEEDER
F106 - NO.1 SUBBUS FEEDER

XF107 - LIMITER BASE
F101 - NO. 2 AVIONICS BUS

XF108 - LIMITER BASE
F100 - LH REFRESHMENT BAR
F103 - FORWARD EVAPORATOR BLOWER

F133 - PILOT'S WINDSHIELD ANTI-ICE LIMITER

K100 - LANDING GEAR POWER RELAY

K129 - FORWARD VENT BLOWER RELAY

GS113 - GROUND STUD

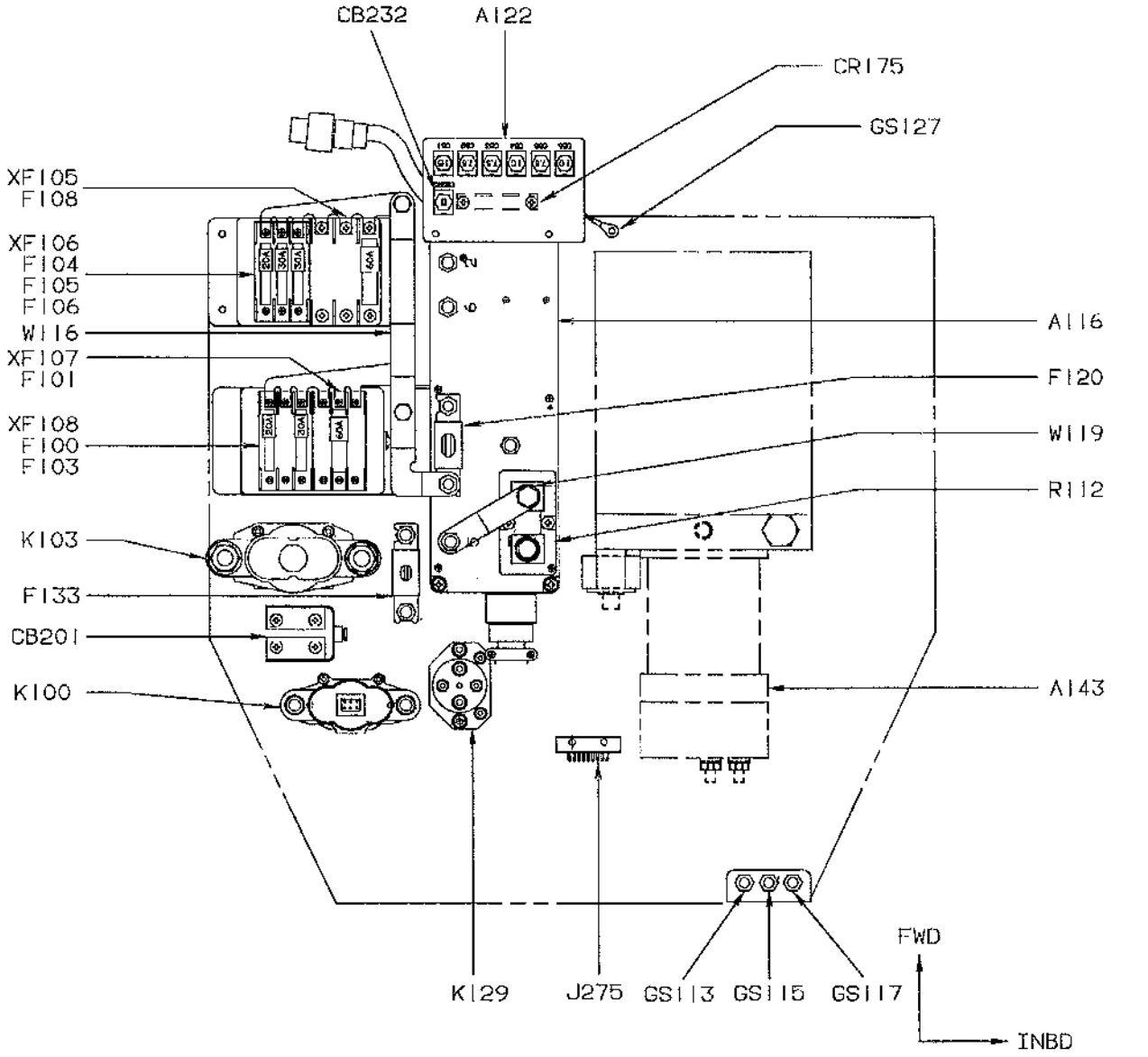
GS115 - GROUND STUD

GS117 - GROUND STUD

J275 - LH NACELLE GROUND BLOCK CONNECTOR

2000-355-043

LH Nacelle Electrical Equipment (Zone 583) (Effectivity: NC-4 thru 29)
(Sheet 2 of 2)
Figure 19



2000-355-046

LH Nacelle Electrical Equipment (Zone 583) (Effectivity: NC-30 and After)
(Sheet 1 of 2)
Figure 20

A116 - LH GENERATOR POWER RELAY PANEL
CURRENT SENSOR
LH BUS TIE RELAY
LINE CONTACTOR RELAY
LH ENGINE START REALY

A122 - LH NACELLE CIRCUIT BREAKER PANEL
CB1 - LH GENERATOR AND FIELD SENSE
CB2 - LH GENERATOR LOAD METER
CB3 - LOAD METER
CB4 - LH GENERATOR CONTROL UNIT
CB5 - LH GENERATOR VOLTAGE
CB6 - LH BUS SENSE
W1 - BUS BAR

F120 - LH BUS LIMITER
R112 - LH LOAD METER SHUNT
W116 - BUS BAR
W119 - BUS BAR

CR175 - LH GENERATOR BUS METAL OXIDE VARISTOR
CB232 - METAL OXIDE VARISTOR CIRCUIT BREAKER
GS127 - GROUND STUD
CB201 - LANDING GEAR POWER CIRCUIT BREAKER
A143 - HYDRAULIC LANDING GEAR POWER PACK
XF105 - LIMITER BASE
F108 - LH GENERATOR TRIPLE FED BUS FEEDER

XF106 - LIMITER BASE
F104 - NO. 3 SUBBUS FEEDER
F105 - NO. 2 SUBBUS FEEDER
F106 - NO. 1 SUBBUS FEEDER

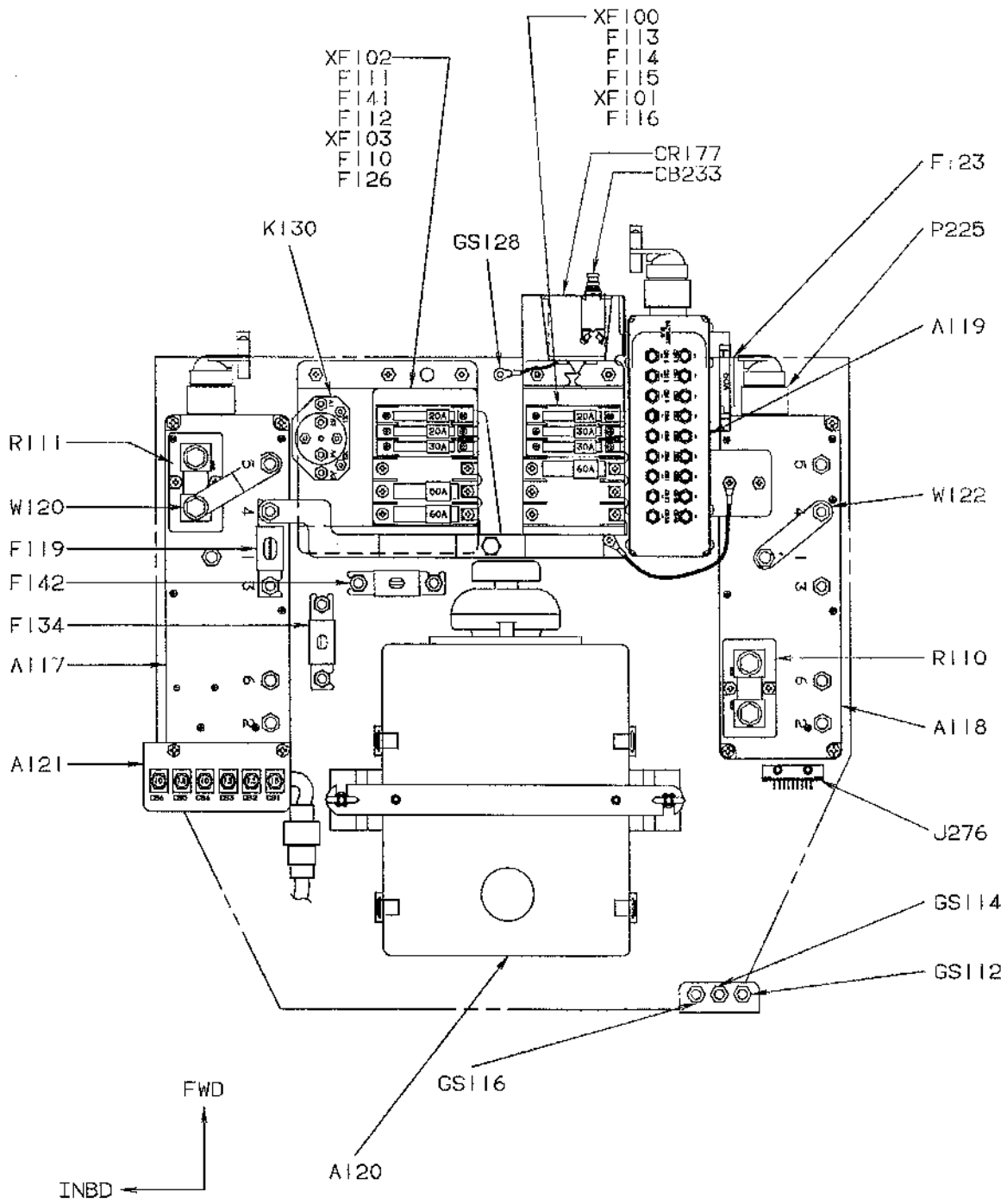
XF107 - LIMITER PANEL
F101 - NO. 2 AVIONICS BUS

XF108 - LIMITER PANEL
F100 - LH REFRESHMENT BAR
F103 - FORWARD EVAPORATOR BLOWER

F133 - PILOT'S WINDSHIELD ANTI-ICE LIMITER
K100 - LANDING GEAR POWER RELAY
K103 - EXTERNAL POWER RELAY
K129 - FORWARD VENT BLOWER RELAY
GS113 - GROUND STUD
GS115 - GROUND STUD
GS117 - GROUND STUD
J275 - LH NACELLE GROUND BLOCK CONNECTOR

2000-355-047

LH Nacelle Electrical Equipment (Zone 583) (Effectivity: NC-30 and After)
(Sheet 2 of 2)
Figure 20



RH Nacelle Electrical Equipment (Zone 683) (Effectivity: All)
(Sheet 1 of 2)
Figure 21

A117 - RH GENERATOR POWER RELAY PANEL ASSEMBLY
CURRENT SENSOR
RH BUS TIE RELAY
LINE CONTACTOR RELAY
RH ENGINE START RELAY
W120 - BUS BAR

A118 - BATTERY POWER RELAY PANEL
BATTERY RELAY
BATTERY TIE RELAY
CURRENT SENSOR
W122 - BUS BAR

A119 - BATTERY BUS CIRCUIT BREAKER PANEL
CB1 - LH ENGINE FIRE EXTINGUISHER
CB2 - RH ENGINE FIRE EXTINGUISHER
CB3 - ELECTRONIC OIL DIPSTICK
CB4 - AVIONICS MEMORY
CB5 - LEFT STANDBY PITCH TRIM
CB6 - CABIN LIGHTS
CB7 - GROUND RTU
CB8 - GROUND AUDIO
CB9 - GROUND COMM
CB10 - BAGGAGE COMPARTMENT LIGHTS
CB11 - FLOODLIGHTS
CB12 - LH FIREWALL FUEL SHUTOFF VALVE
CB13 - RH FIREWALL FUEL SHUTOFF VALVE
CB14 - BATTERY RELAY
CB15 - RIGHT STANDBY PITCH TRIM
CB16 - STEREO MEMORY
CB17 - STEERING DISCONNECT
CR1 - BATTERY BUS DIODE
CR2 - BATTERY BUS DIODE

A120 - BATTERY

A121 - RH NACELLE CIRCUIT BREAKER PANEL
CB1 - RH GENERATOR FIELD AND SENSE
CB2 - RH GENERATOR LOAD METER
CB3 - RH LOAD METER
CB4 - RH GENERATOR CONTROL PANEL
CB5 - RH GENERATOR VOLTAGE
CB6 - RH BUS SENSE
W1 - BUS BAR

XF102 - LIMITER BASE
F111 - RH REFRESHMENT BAR
F141 - ANTI-SKID PUMP
F112 - AFT EVAPORATOR BLOWER

XF103 - LIMITER BASE
F110 - NO. 3 AVIONICS BUS FEEDER
F126 - CONDENSER BLOWER

XF100 - LIMITER BASE
F113 - NO. 3 SUBBUS FEEDER
F114 - NO. 2 SUBBUS FEEDER
F115 - NO. 1 SUBBUS FEEDER

XF101 - LIMITER BASE
F116 - RH GENERATOR TRIPLE FED BUS FEEDER

K130 - AFT VENT BLOWER RELAY

R111 - RH LOAD METER SHUNT

F134 - CO-PILOT'S WINDSHIELD ANTI-ICE LIMITER

F142 - PILOT'S REDUNDANT WINDSHIELD ANTI-ICE LIMITER

F119 - RH GENERATOR BUS TIE LIMITER

F123 - BATTERY TRIPLE FED BUS FEEDER LIMITER

R110 - BATTERY AMMETER SHUNT

CR177 - RH GENERATOR BUS METAL OXIDE VARISTOR

CB233 - METAL OXIDE VARISTOR CIRCUIT BREAKER

GS128 - GROUND STUD

GS112 - GROUND STUD

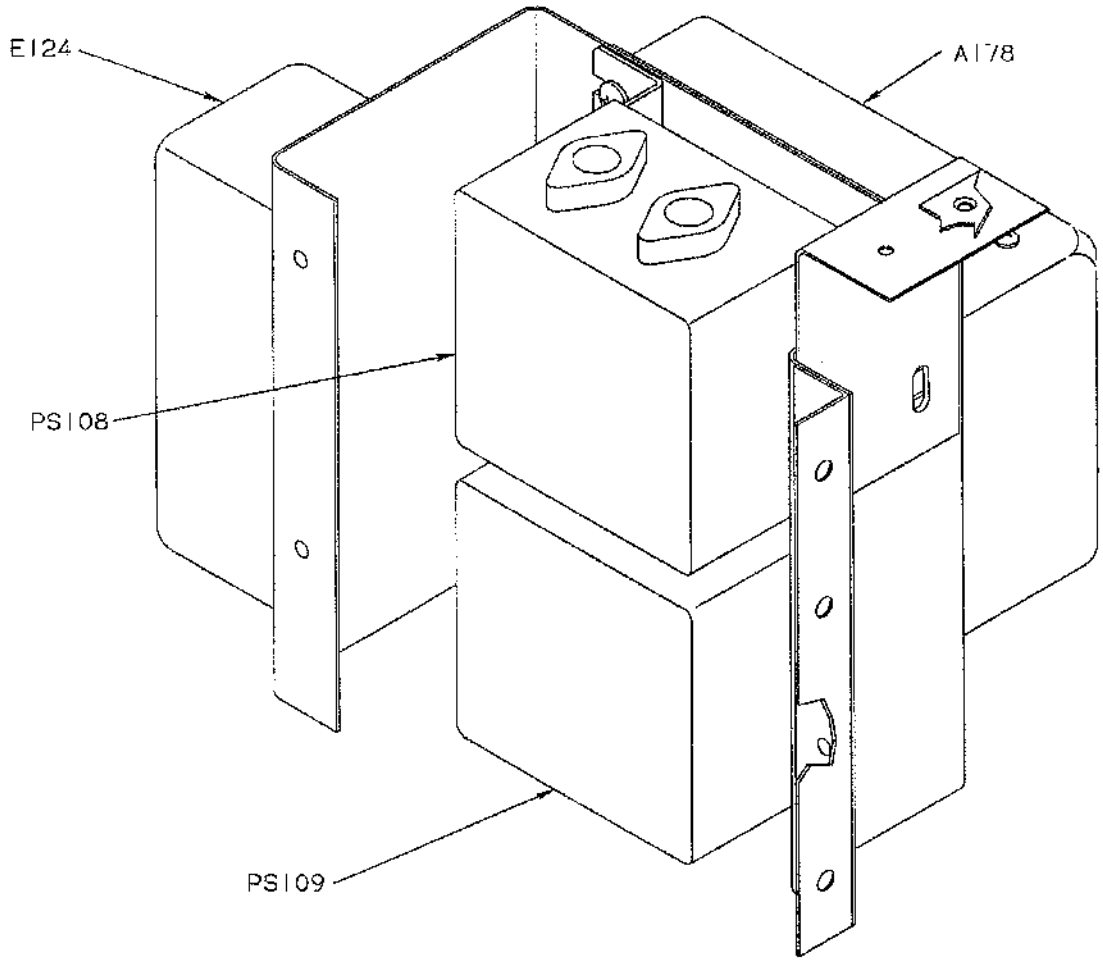
GS114 - GROUND STUD

GS116 - GROUND STUD

J276 - RH NACELLE GROUND BLOCK CONNECTOR

2000-355-044

RH Nacelle Electrical Equipment (Zone 683) (Effectivity: All)
(Sheet 2 of 2)
Figure 21



E124 - PROP SYNCHROPHASER CONTROL BOX

A178 - FLAP/FORWARD WING MONITOR BOX

PS108 - LEFT AISLE LIGHTS POWER SUPPLY

PS109 - RIGHT AISLE LIGHTS POWER SUPPLY

2000-354-006

Center Aisle Aft Step Electrical Equipment (Zone 175) (Effectivity: All)
Figure 22

PRINTED CIRCUIT BOARDS - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

Figure 1 in this chapter identifies each printed circuit board installed in the printed circuit board rack by name and designator. The printed circuit board rack is located in the pilot's console. Each printed circuit board reference designator corresponds with the designator used in the BEECHCRAFT Starship 1 Wiring Diagram Manual and the avionics wiring diagrams.

PRINTED CIRCUIT BOARDS - MAINTENANCE PRACTICES (Effectivity: All)

PRINTED CIRCUIT BOARD REMOVAL (Effectivity: All) (Figure 2)

CAUTION

Anytime maintenance is performed on or around printed circuit boards, it is essential that all Electrostatic Discharge (ESD) precautions are strictly observed. Refer to Chapter 20-00-02 for detailed information on the handling of ESD sensitive equipment.

a. Remove all electrical power from the airplane.

b. Refer to Chapter 25-10-00 and remove the pilot's seat and the aft upholstery panel from the pilot's console.

c. Remove the screws securing the upper wire tray cover (16) to the console structure, then remove the cover.

d. Disconnect the electrical connector from the desired printed circuit board (15).

e. Remove the screws (11) and washers (10) securing the plate assembly (13) to the printed circuit board box assembly (1) and slide the printed circuit board (15) out of the guides (2).

f. Remove the nut (8), washer (7) and screw (14) securing the clip (9) to the printed circuit board (15).

g. Remove the nuts (3), lock washers (4), washers (5) and screws (12) securing the connector (6) to the plate assembly (13), then remove the printed circuit board (15) from the airplane.

PRINTED CIRCUIT BOARD INSTALLATION (Effectivity: All) (Figure 2)

CAUTION

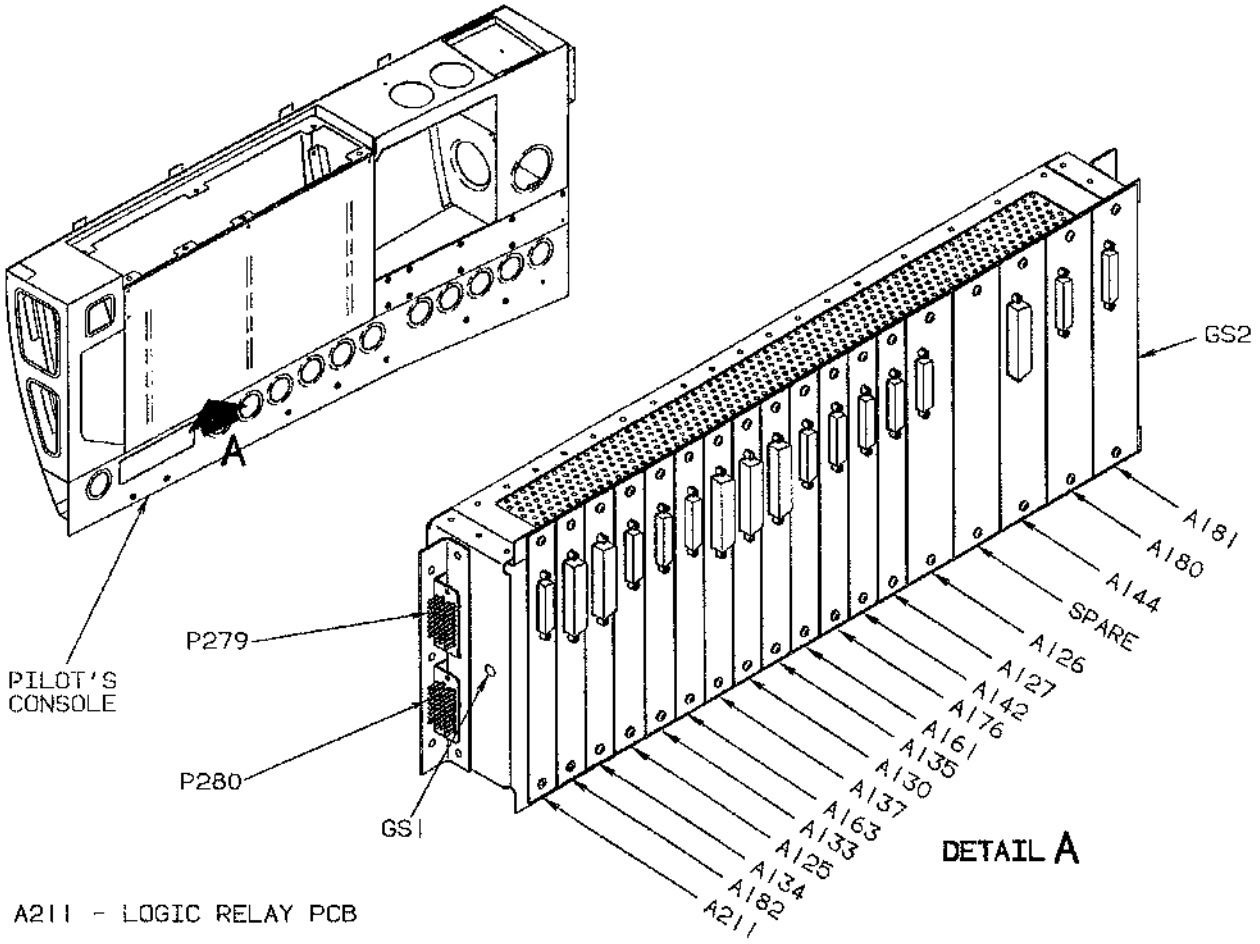
Anytime maintenance is performed on or around printed circuit boards, it is essential that all Electrostatic Discharge (ESD) precautions are strictly observed. Refer to Chapter 20-00-02 for detailed information on the handling of ESD sensitive equipment.

All printed circuit boards must meet electrical bonding requirements specified in Chapter 20-00-03. Proper bonding and installation of the clip (9) on the printed circuit board (15) should provide an adequate ground.

a. Align the printed circuit board (15) with the plate assembly (13) and secure the connector (6) to the plate assembly (13) with screws (12), washers (5), lock washers (4) and nuts (3).

b. Install screw (14), washer (7) and nut (8) to secure the clip (9) to the printed circuit board (15).

c. Slide the printed circuit board (15) into the guides (2), then install screws (11) and washers (10) to secure the printed circuit board (15) in the box assembly (1).



A211 - LOGIC RELAY PCB

A182 - LOGIC RELAY PCB

A134 - LOGIC RELAY PCB

A125 - BATTERY MONITOR PCB

A133 - N_1 SPEED CONTROL PCB

A163 - LIGHTING CONTROL PCB

A137 - ANNUNCIATOR FAULT DETECTION PCB

A130 - ANNUNCIATOR AND FIRE EXTINGUISHER TEST PCB

A135 - ANNUNCIATOR CONTROL PCB

A161 - LANDING GEAR INDICATION PCB

A176 - BRAKE DEICE TIMER AND TIME DELAY PCB

A142 - LANDING GEAR CONTROL PCB

A127 - WINDSHIELD ANTI-ICE AND EXTERNAL POWER MONITOR PCB

A126 - POWER DISTRIBUTION PCB

A144 - FLAP/FORWARD WING CONTROLLER AND ANNUNCIATION PCB

A180 - RH FLAP/FORWARD WING MONITOR AND MONITOR TEST PCB

A181 - LH FLAP/FORWARD WING MONITOR AND MONITOR TEST PCB

P279 - LEFT SUBPANEL GROUND BLOCK CONNECTOR

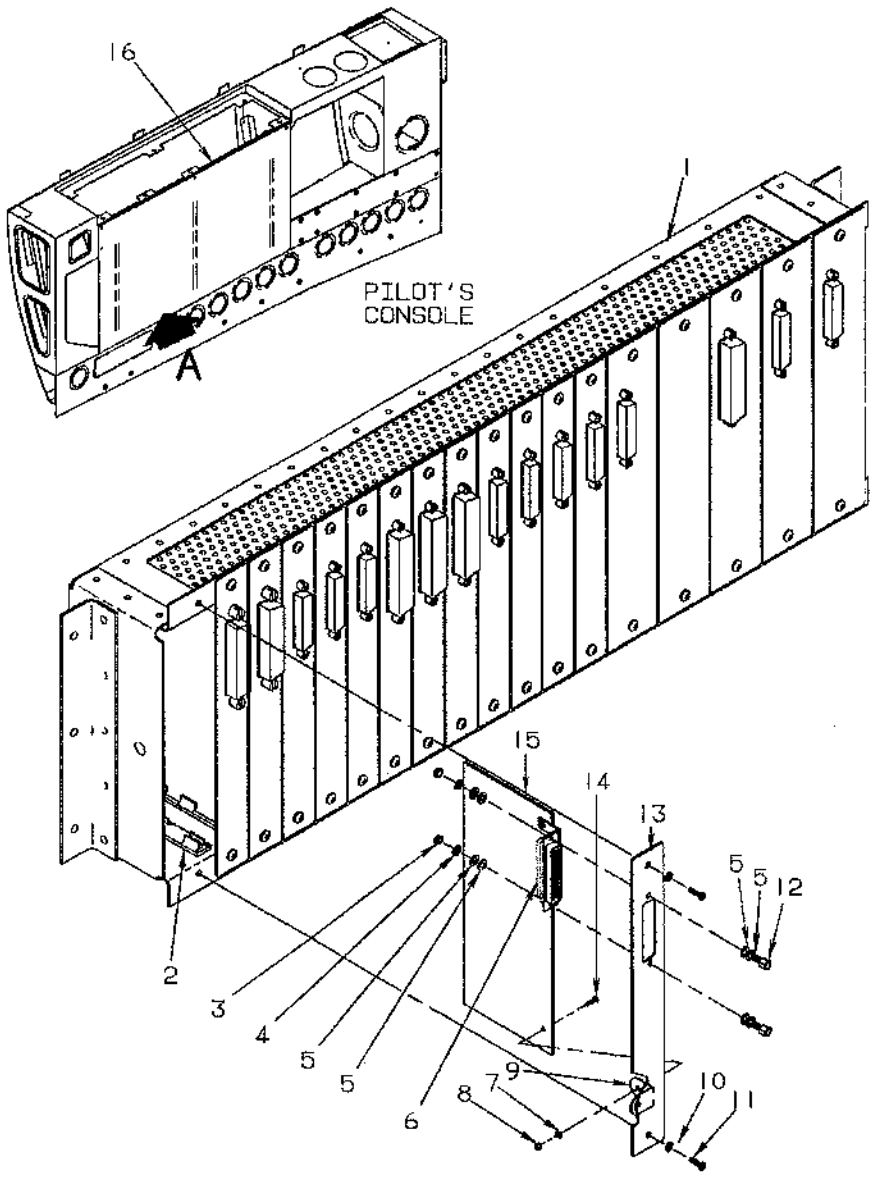
P280 - RIGHT SUBPANEL GROUND BLOCK CONNECTOR

GS1 - GROUND STUD

GS2 - GROUND STUD

2000-351-067

Printed Circuit Board Rack Assembly (Effectivity: All)
Figure 1



1. PRINTED CIRCUIT BOARD BOX ASSY
2. GUIDE
3. NUT
4. LOCK WASHER
5. WASHER
6. CONNECTOR
7. WASHER
8. NUT
9. CLIP
10. WASHER
11. SCREW
12. SCREWLOCK
13. PLATE ASSEMBLY
14. SCREW
15. PRINTED CIRCUIT BOARD
16. UPPER WIRE TRAY COVER

DETAIL A

2000-351-083

Printed Circuit Board Installation (Effectivity: All)
Figure 2

WARNING

The upper wire tray cover is load supporting and must be installed before flight of the airplane.

d. Align the wire tray cover (16) with the console structure and secure with screws.

e. Refer to Chapter 25-10-00 and install the pilot's seat and the aft upholstery panel on the pilot's console.

f. Restore electrical power to the airplane.

51 - STRUCTURES

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Beechcraft
STARSHIP 1
MAINTENANCE MANUAL

CHAPTER 51 - STRUCTURES

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The BEECHCRAFT Starship 1 airplane is constructed of composite materials. Composite construction technology has eliminated the use of metal skins, ribs and stringers.

The airplane structures are fabricated with layers of woven graphite fibers that are impregnated with epoxy or resin. The woven graphite fibers are laminated over a honeycomb core material, laid up in a mold and cured in an autoclave at 300 degrees Fahrenheit.

The fuselage is pressurized between the forward pressure bulkhead (FS 95.00) and the aft pressure bulkhead dome. The aft pressure bulkhead consists of an upper bulkhead and dome at FS 416.00, a lower bulkhead at FS 357.00 and the baggage shelf between the two bulkheads.

The composite constructed aft wing is a single unit with spars, ribs and skins. The main spars which run the length of the wing are identified as the front spar, center spar, inboard and outboard rear spars. The wing is centered on the fuselage with the spars passing through, and attached to the fuselage.

Information pertaining to the cabin door and emergency exit door is included in Chapter 52. Maintenance of the forward wing, ventral stabilizer and balance check of the control surfaces is included in Chapter 55. If repair to a structural member is necessary, refer to the BEECHCRAFT Starship 1 Structural Repair Manual P/N 122-590013-7.

SPECIAL TOOLS AND EQUIPMENT (Effectivity: All)

Each tool listed in Chart 1 is provided as an example of the equipment designed to perform a specific function. Generic or locally manufactured tools that are the equivalent with respect to accuracy, function and craftsmanship may be used in lieu of those listed.

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All)

TOOL NAME	PART NO.	VENDOR	USE
1. A. Scanner	MIA-2500	Staveley Instrument Inc., Kenwick, WA 99336	To set up a vibration mode in material and read the results
2. Probe	STO-3998	Staveley Instrument Inc., Kenwick, WA 99336	Used with A. Scanner
3. Ultrasonic Unit	NDT-132	Staveley Instrument Inc., Kenwick, WA 99336	To produce pulses for nondestructive testing inspections
4. Probe	RD-375 or RD-50-375	Utran State College PA 16801	Used witn Ultrasonic Unit
5. Fiber-Optic Borescope or Video Probe Borescope	IF-ND3-30 2000	Olympus Welch Allyn Box 2000 Skaneateles Falls NY 13153	To visually inspect most inaccessible areas

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All) *(Continued)*

TOOL NAME	PART NO.	VENDOR	USE
6. Tap Hammer (1-5 oz.)	TS1469	Beech Aircraft Corp. Wichita, KS 67201	To produce tapping sounds across a composite surface
7. Feeler Gage	Shim stock 0.005-inch thickness	Obtain locally	To probe for disbonds and/or delamination of a bond line

GENERAL - MAINTENANCE PRACTICES (Effectivity: All)

Maintenance of the airplane structures includes inspections to reveal possible damage, whether the damage requires cosmetic or supervised repair. Cosmetic type damage is allowable with no restrictions on flight operations and can be permitted to exist "as is" or be repaired by refinishing procedures. Repairable type damage (by standard repair) is damage that will weaken the structural strength of the airplane, such as damage to the skin, bond or core that cannot be allowed to exist without placing performance restrictions on the airplane. Supervised repair is of damage in excess of repairs allowed under standard repair and requires approval of BEECHCRAFT Service Engineering.

STRUCTURAL INSPECTION (Effectivity: All)

The following procedures specify requirements and instructions for in service inspection of the major structural components of the airplane.

Inspection of the airplane major structural components involves Routine Visual Inspection, Significant Structural Items Inspection, and Inspection of Suspect Areas. Inspection intervals are outlined in Chapter 4-00-00.

Methods of in-service inspections are visual/fiber-optic borescope, tap test, feeler gauge, mechanical impedance and ultrasonic inspection.

The usage of the special tools and equipment listed in Chart 1 is shown below:

a. Fiber-Optic Borescope or Video Probe Borescope (5, Chart 1): A device made from strands of fiber that allows the bending of light. This device is used to visually inspect areas that are inaccessible by normal means. The borescope may be rigid or flexible.

b. Tap Hammer (6, Chart 1): A metal hammer designed to produce repetitive tapping sounds across a composite surface. Recognition and comparison of sharp and dull tapping sounds aids in pin-pointing delamination, soft spots, and/or damage not visual on the surface.

c. Feeler Gauge (7, Chart 1): Shim stock of approximately 0.005 thickness with a pointed end. Used to probe for disbonds and/or delamination of a bond line.

d. Mechanical Impedance (1 and 2, Chart 1): Used herein to mean an electronic instrument capable of setting up a vibration mode of 1.0 to 8.0 kHz in a material and reading the resultant impedance plane ratio, phase shift or amplitude change.

e. Ultrasonic Inspection (3 and 4, Chart 1): The inspection of an item using an ultrasonic unit capable of producing pulses in a range of 0.5 to 10.0 mHz and a transducer(s).

Personnel performing visual or tap test inspections must be qualified for the applicable inspection by attending a BEECHCRAFT approved service school. Personnel performing ultrasonic inspections must meet the requirements for visual/tap test inspections and be qualified and certified in accordance with MIL-STD-410D. Personnel performing nondestructive inspections must be a Level II minimum.

Composite structural inspections are essential to reveal delamination/separation, exposed fibers, cracks, soft spots, fastener pull-through, punctures and/or impact damage. Impact damage may result in punctures, crushes or soft spots. Where damage has penetrated the honeycomb core, a check should be made for entry of moisture/fluid.

ROUTINE VISUAL INSPECTION (Effectivity: All)

NOTE

The routine visual inspection is performed to detect abnormalities such as obvious damage (holes, tears, cracks in paint, erosion or degradation of the paint, indentations and loose parts.

FUSELAGE EXTERIOR INSPECTION (Effectivity: All)

Perform a complete visual examination of the exterior of the fuselage. Inspect for abnormalities such as obvious damage, indentations, holes, tears, cracks, chips, erosion or degradation of the paint. Pay particular attention to the area around the windshield frame, window frames, nose section attach area.

FUSELAGE INTERIOR INSPECTION (Effectivity: All)

Perform a complete visual examination of the fuselage interior structure. Inspect for abnormalities such as obvious damage, indentations, holes, tears and cracks. Pay particular attention to the area around the cabin door frame, interior side and exposed edges of the door.

WING, FORWARD/AFT AND TIPSAILS EXTERIOR INSPECTION (Effectivity: All)

Perform a complete visual examination of the forward and aft wings and tipsails exterior. Inspect for abnormalities such as obvious damage, indentations, holes, tears, cracks, chips, erosion or degradation of the paint. Pay particular attention to the lower wing surfaces where they are susceptible to damage from runway debris. Inspect all control surface attachments and landing gear attachments.

WING, FORWARD/AFT AND TIPSAILS INTERIOR INSPECTION (Effectivity: All)

NOTE

Routine visual inspection is not required on the interior surfaces of the tipsails.

Visually inspect the inside area of the main wheel wells for obvious damage. In the main landing gear wheel wells, inspect for abnormalities such as indentations, holes, tears, cracks and chips. Pay particular attention to the exposed rear spar (landing gear spar), the spar-to-upper skin attach angles, and the landing gear attach fittings.

NACELLES EXTERIOR INSPECTION (Effectivity: All)

Perform a complete visual examination of the nacelle exterior surfaces. Inspect for signs of damage such as cracks, chips, erosion of the paint, buckling of the skins and broken fibers. Pay particular attention to the area around the fastener holes of the two upper access panels.

NACELLES INTERIOR INSPECTION (Effectivity: All)

NOTE

To inspect the interior of the nacelle, remove the two nacelle upper access panels.

Inspect the inner surface of the nacelles where accessible for broken fibers and buckling of the skin. Inspect the secondary bond between the "hat section" doubler and the interior side of the skin.

SIGNIFICANT STRUCTURAL ITEM INSPECTION (Effectivity: All)

NOTE

Prior to performing a significant structural item inspection, the Routine Visual Inspection must be performed.

FUSELAGE EXTERIOR INSPECTION (Effectivity: All)

NOTE

Remove all nose doors and panels, aft fuselage access doors and panels and the upper and lower wing-to-fuselage fairings. Disconnect and remove all electronic modules and removeable hardware from the avionics compartment. Do not remove the oxygen bottle unless it is necessary to access a suspected area. Remove the outflow valve exit ramp. Pull away wire bundles as far as possible when necessary.

a. Perform a thorough visual examination of all attach angles in the nose section. For inaccessible areas, the use of a mirror and flashlight or fiber-optic borescope (5, Chart 1) may be necessary. Inspect for any indication of bond line failure. These may be cracks in the exposed adhesive bond line edge, buckling of the attach member, or fiber splinters and tearing in the attach angle. Inspect for cracks in the adhesive bond line edge.

b. Perform a visual examination of the upper and lower nose shelves, center keels and forward side of the forward pressure bulkhead (FS 95.00). Inspect for any signs of damage such as buckling, fiber tearing, raised areas or any other irregularities. Pay particular attention to the area surrounding the forward wing

pivot point, both the upper and lower surfaces of the shelves, the inboard and outboard surfaces of the keels, the FS 95.00 bulkhead-to-upper shelf attach area, the FS 45.00 bulkhead-to-lower shelf attach area, and the attach angles for loose or missing fasteners.

c. Perform a visual inspection of the fuselage in the area of the wing cutout. A fiber-optic borescope (5, Chart 1) inspection is required in areas where visual inspection is not possible due to the close proximity of the wing. Inspect for cracks in the exposed adhesive bond line edge, buckling of the attach member, or fiber splinters and tearing in the attach angle. Inspect for any signs of damage such as buckling, fiber tearing, raised areas or any other irregularities.

d. Perform an inspection of the aft side of the FS 416.00 aft pressure bulkhead and the pressure dome through the lower aft fuselage access panel. This inspection is to be accomplished utilizing visual and fiber-optic borescope (5, Chart 1) inspection methods. Inspect for indications of bonding failure such as cracks in exposed adhesive bond line edge, buckling of the attach member, fiber splinters and tearing, or any other irregularities.

NOTE

In cases where further investigation is warranted, refer to INSPECTION OF SUSPECT AREAS in this chapter.

FUSELAGE INTERIOR INSPECTION
(Effectivity: All)

NOTE

Remove interior panels aft of the FS 95.00 forward pressure bulkhead, around the baggage shelf aft of the FS 357.00 pressure bulkhead, forward of the FS 416.00 pressure bulkhead, and the panels necessary to inspect around the cabin door frame and the emergency exit door frame. Remove the floorboards, the headliners and the baggage shelf protective honeycomb panel.

a. Using a flashlight and mirror or a fiber-optic borescope (5, Chart 1) visually inspect the secondary bond lines for any indication of bond line failure, such as cracks in the exposed adhesive bond line edge, buckling of the attach member or fiber splinter and tearing in the attach angle in the following areas:

1. Around the FS 95.00 forward pressure bulkhead attach angles to both the bulkhead and to the fuselage (where accessible without removing instruments).

2. Around the FS 357.00 aft pressure bulkhead attach angles both to the bulkhead and to the fuselage extending around the entire inner circumference of the fuselage.

3. Around the FS 416.00 aft pressure bulkhead to both the bulkhead and to the fuselage.

4. Around the pressure dome to the FS 416.00 pressure bulkhead.

5. Around the baggage shelf attach members both to the shelf and to the fuselage.

6. All bond lines on the cabin door frame, (the area on the mid or lower side of the forward bond line is not accessible).

7. Upper and lower BL 0.00 splice (also inspect for loose or missing fasteners).

8. Around the windshield frame attachment area.

b. Visually inspect the following areas for an indication of failure, such as buckling of skins, fiber tearing, raised areas or other irregularities.

1. FS 95.00 forward pressure bulkhead (where accessible without removing the instruments).

2. FS 357.00 aft pressure bulkhead.

3. FS 416.00 aft pressure bulkhead.

4. FS 416.00 pressure dome.

5. Baggage shelf (beneath the protective panel).

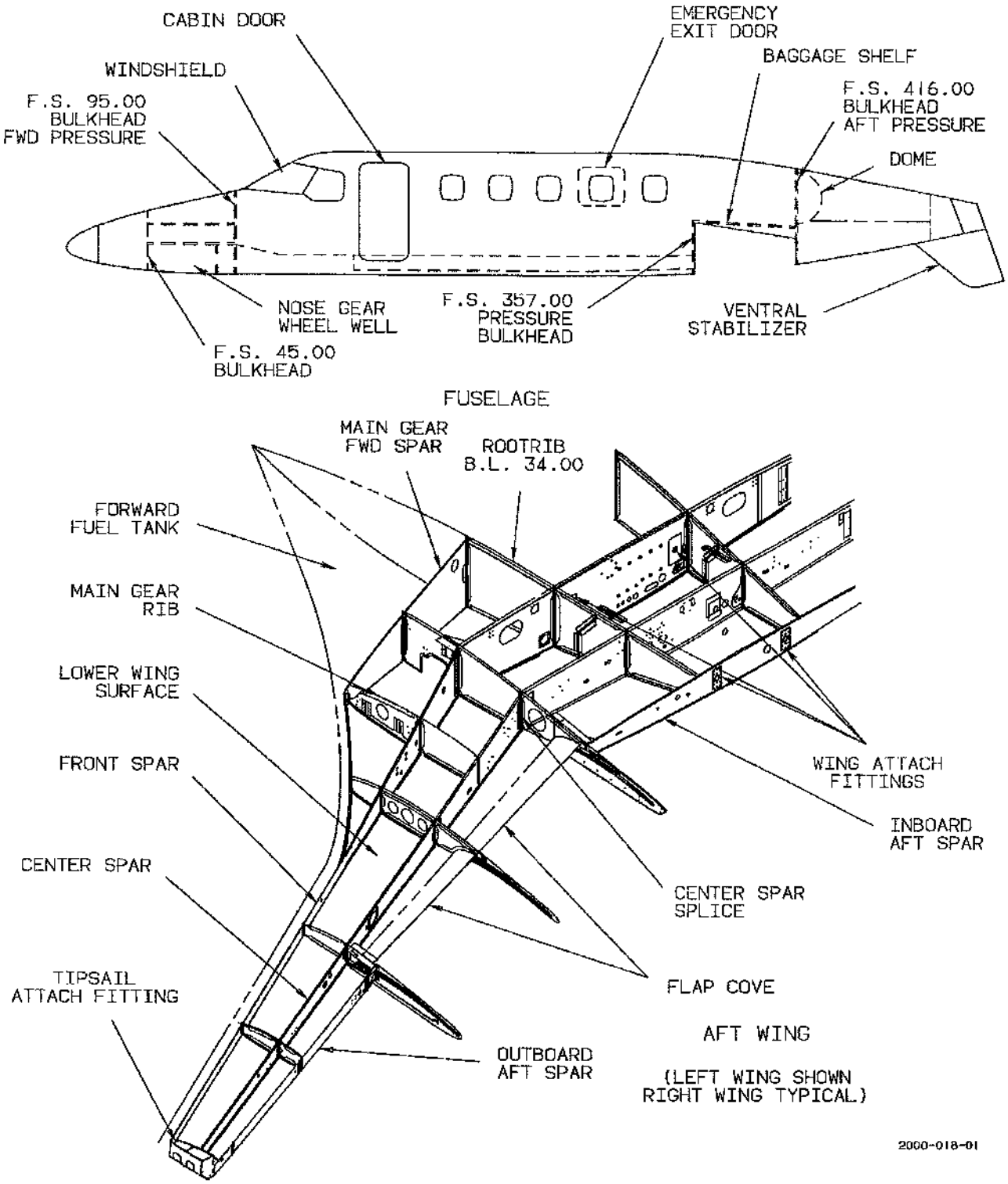
6. Emergency exit window frame close out.

NOTE

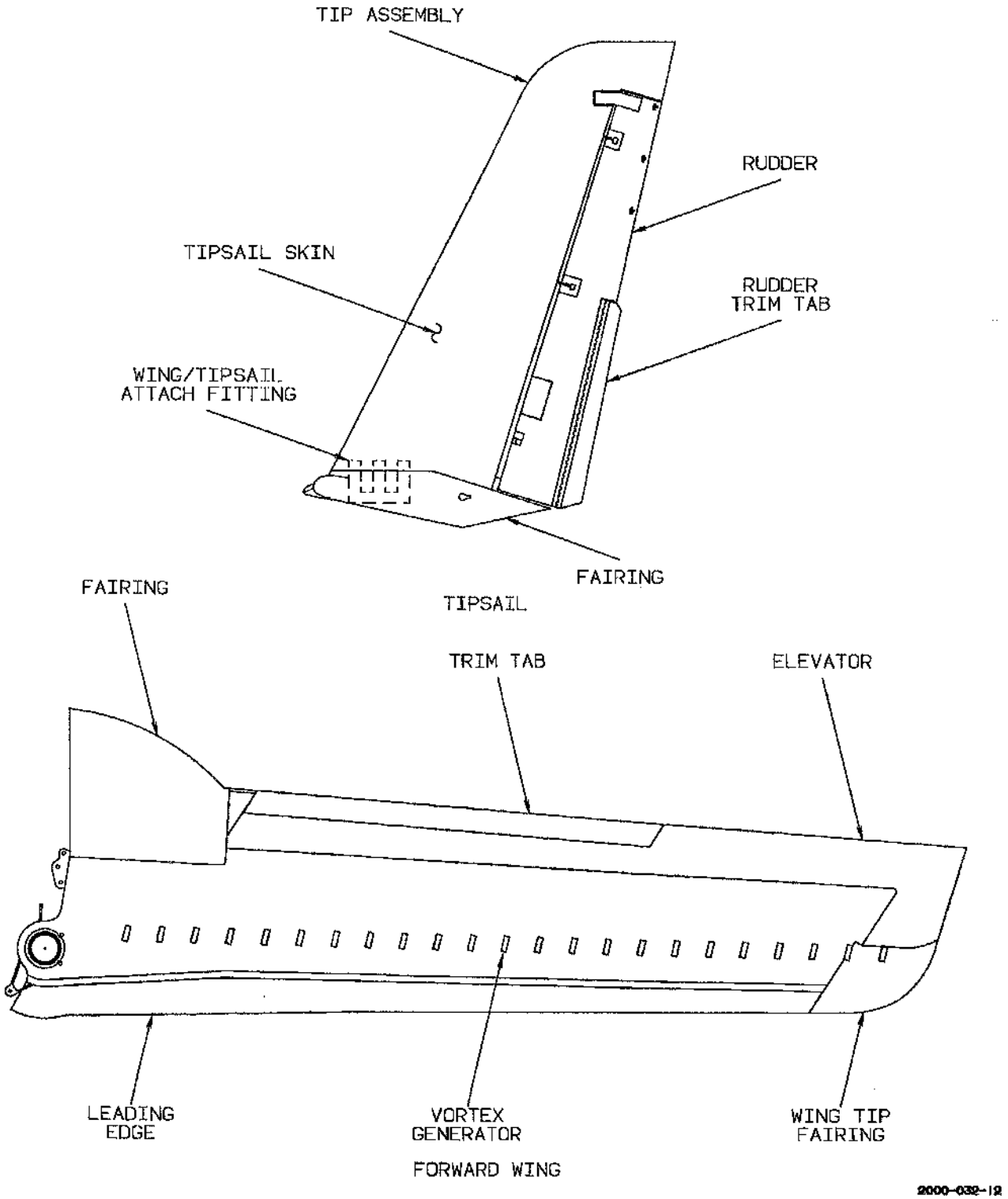
If any of the areas listed are suspect, or in cases where further investigation is necessary, refer to INSPECTION OF SUSPECT AREAS in this chapter.

FORWARD WINGS EXTERIOR INSPECTION
(Effectivity: All)

Perform a visual examination of the upper and lower surfaces of the forward wing. Inspect for indication of damage, such as skin buckling, fiber tears, raised areas or deformation of the leading edge. Pay particular attention to the area around the pivot pin and the actuator attach points.



Fuselage and Aft Wing Significant Structural Areas (Effectivity: All)
Figure 1



Tipsail and Forward Wing Significant Structural Areas (Effectivity: All)
Figure 2

FORWARD WINGS INTERIOR INSPECTION (Effectivity: All)

NOTE

To inspect the interior of the forward wings, remove the nose-to-wings fairings.

Using a fiber-optic borescope (5, Chart 1), inspect the forward and aft side of the main spar web (just aft of the leading edge) and the upper and lower interior wing surfaces. Access to the forward side is through the forward wheel well and through the outboard horn rear close out. Access to the aft side is through the outboard close out. Access to the aft side of the rear spar is by removing the trim tab actuator access cover. Due to the limited view of the first three feet, inspect through the stiffener where possible. Pay particular attention to the radius area between the upper and lower spar caps and the spar web both forward and aft sides.

NOTE

If further investigation is warranted, refer to INSPECTION OF SUSPECT AREAS in this chapter.

AFT WING EXTERIOR INSPECTION (Effectivity: All)

Using a flashlight, inspect all bond lines in the main wheel well openings. Inspect for irregularities such as cracking of adhesive on or near the bond line and buckling and tearing of the angle section. Inspect all control surfaces for obvious damage. Pay particular attention to the inboard and outboard rib secondary bonds on all control surfaces, the flap track roller assemblies, the flap tracks, all attaching hardware, exposed rear spar and rear spar bond lines, and reflective tape between the nacelle attach member and the rear spar.

AFT WING INTERIOR INSPECTION (Effectivity: All)

NOTE

To inspect the aft wing interior, remove all wing access doors (except the aft fuel cell door), the tipsail fairing assembly, and the aft wing shear panel.

a. Using a flashlight and mirror and/or fiber-optic borescope (5, Chart 1), visually inspect all wing bays. Inspect the secondary bond joints for such irregularities as cracked bonds on the edges of the attach members both to the wing and to the spars and ribs, buckling and fiber tearing of the attach angles.

b. Using a fiber-optic borescope (5, Chart 1), inspect the inner surfaces of the upper and lower skins. Check for obvious damage such as fiber tearing and raised areas indicating skin-to-core separation and/or other obvious irregularities. Pay particular attention to BL 34.00 root rib.

TIPSAILS EXTERIOR INSPECTION (Effectivity: All)

a. Inspect the C-channel to skin secondary bond lines located on the aft side of the tipsail forward of the rudder. Check for defects such as cracks in the exposed adhesive or buckling and tearing of the C-channel.

b. Inspect the rudder for any indication of damage caused by foreign objects or other defects. Pay particular attention to the exposed ribs on the lower side of the surface, checking for bond line cracks or other visible damage.

TIPSAILS INTERIOR INSPECTION (Effectivity: All)

NOTE

To inspect the interior of the tipsail, remove the tip assembly and the tipsail fairing assembly.

Using a fiber-optic borescope (5, Chart 1), inspect the forward and aft sides of the forward and aft C-channels. Inspect the inner surface of the tipsail. Check for any damage such as broken fibers, buckling, secondary bond and bond line cracking.

NACELLES EXTERIOR INSPECTION (Effectivity: All)

Perform a complete visual examination of the exterior surface of the nacelles. Check for signs of damage such as cracks, indentations, chips or erosion of the paint, buckling of the skins and broken fibers. Pay particular attention to the areas around the fastener holes on the two upper access panels.

NACELLES INTERIOR INSPECTION (Effectivity: All)

NOTE

To inspect the nacelle interior, remove the two upper access panels.

Visually inspect the interior surface of the nacelle where accessible. Inspect for any irregularities such as fiber splinters, or buckling of the skin. Inspect the "hat section" that is bonded to the left and right inner sides of the nacelle skin. Pay particular attention to the edges of the secondary bond on the "hat section", and check for any indications of cracking on the exposed adhesive. The use of a flashlight and mirror or fiber-optic borescope (5, Chart 1) may be necessary. Inspect the forward nacelle bulkhead section for indication of bond failure such as hairline cracks in the exposed adhesive edge.

NOTE

If any additional testing is required, refer to INSPECTION OF SUSPECT AREAS in this chapter.

INSPECTION OF SUSPECT AREAS (Effectivity: All)

NOTE

Initially, and in most cases, the use of a feeler gauge (7, Chart 1) and a tap hammer (6, Chart 1) will be sufficient to determine the extent or presence of defects in a suspect area. In cases where further investigation is warranted, refer to NONDESTRUCTIVE TESTING in this chapter.

PRIMARY BOND INSPECTION (Effectivity: All)

When impact damage is suspected, the tap hammer (6, Chart 1) method is used to define the area. When tapping in a repetitive light manner and moving about, a noticeable dull tapping sound should be heard identifying a damaged area. This is caused by the loss of support from the back of the area damaged; however, this should not be confused with a sharper sound such as when a core filled area is tapped. To find an outline of a damaged area, start tapping a few inches away from the center of the suspect area and tap toward it.

NOTE

If further testing is necessary, refer to NONDESTRUCTIVE TESTING in this chapter.

SECONDARY BOND INSPECTION (Effectivity: All)

Suspected disbonds may be detected with the use of a feeler gauge (7, Chart 1). An attempt should be made to insert the feeler gauge into the bond line at the area suspected of being cracked. In most cases this can easily be accomplished if the area is unbonded. If the feeler gauge cannot be inserted, use a tap hammer (6, Chart 1) as described in PRIMARY BONDS INSPECTION in this chapter.

NOTE

If further testing is necessary, refer to NONDESTRUCTIVE TESTING in this chapter.

NONDESTRUCTIVE TESTING (Effectivity: All)

NOTE

Personnel performing nondestructive testing must be certified in accordance with MIL-STD-410D in the ultrasonic discipline to a Level II minimum. It is required that the personnel be instructed by BEECHCRAFT authorized nondestructive testing personnel as to the considerations involved when selecting a test method, the expected results, and the availability of nondestructive testing standards for both primary and secondary bonds.

PRIMARY BOND NONDESTRUCTIVE TESTING (Effectivity: All)

a. Depending on the location of the suspected damage, a pulse-echo inspection with the ultrasonic unit and probe (3 and 4, Chart 1) with a 5.0 mHz transducer and a lucite stand-off may be performed. The thickness of the area should initially be determined by ultrasonic comparison method and standards. Select the proper standard and adjust the received signal to obtain a 50 percent screen height, then add 6 dB.

b. Scan the suspect area and note the areas that fall below 25 percent screen height, or the areas where the signal jumps forward in time on the base line, indicating interply-delamination.

NOTE

Care should be taken when utilizing ultrasonic inspection as the sole determining factor for making judgement. Many areas can be interpreted as defective if personnel are not familiar with the construction.

c. It is recommended that a second test method by use of the tap hammer (6, Chart 1) or mechanical impedance (1 and 2, Chart 1) be used to confirm the results.

SECONDARY BOND NONDESTRUCTIVE TESTING (Effectivity: All)

a. Mechanical impedance testing, by use of the A. scanner and probe (1 and 2, Chart 1), is recommended for secondary bonds of laminate to sandwich panel construction (keel attach angle to fuselage). First determine the thickness of the attach angle by the ultrasonic comparison method utilizing the ultrasonic unit and probe (3 and 4, Chart 1) then select the proper standard, set up the test and perform the inspection.

b. Ultrasonic inspection, by use of the ultrasonic unit and probe (3 and 4, Chart 1), is recommended for laminate-to-laminate secondary bonds (fuel tank skin-to-attach angles, tipsails aft close outs to C-channel). The finger damping method is recommended where access to the back side is available. Where access is not available, monitoring of the bond line interface for decreased signal height while watching the backwall signal for simultaneous decrease will provide the desired information.

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

This chapter is concerned with the description, operation and maintenance practices for the airplane's external doors and door warning system.

A door is provided for access to the cabin compartment, An emergency exit door is located on the right side of the cabin compartment. All doors are flush fitting. Except for the emergency exit door, all open outward. Further, a number of access doors are provided for servicing the airplane. For door servicing, refer to LUBRICATION procedures in Chapter 12-20-00.

RECOMMENDED MATERIALS (Effectivity: All)

The recommended materials listed in Chart 1 as meeting federal, military or vendor specifications are provided for reference only and are not specifically recommended by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in this chart have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor or by compliance with the applicable specification. Generic or locally manufactured products which conform to the requirements of the specification may be used even though not included herein. Only the basic number of each product specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
RECOMMENDED MATERIALS (Effectivity: All)

MATERIALS	SPECIFICATION	PRODUCT	VENDOR
1. Cleaning Solvent	TT-N-95	Naphtha	Locally available
2. Adhesive	MMM-A-121	EC1300L	Minnesota Mining & Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144
3. Adhesive	MMM-A-121	EC1357	Minnesota Mining & Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144
4. Adhesive		Dapcotac # 3300	D Aircraft Products., 1191 Hawk Circle, Anaheim, CA 92708
5. Lubricant	MIL-L-7870	Caltex Low Temp Oil	Caltex Oil Products Co. New York, NY
		Sinclair Aircraft Orbitlube	Sinclair Refining Co., 600 Fifth Ave. New York, NY 10020
		1692 Low Temp Oil	Texaco, Inc., 135 E. 42nd St. New York, NY 10017
6. Tape		Masking tape 2000	Minnesota Mining & Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144
7. Retaining Compound		Retaining Compound 609	Loctite Corporation, 705 N. Mountain Rd. Newington, CT 06111
8. Dry Film Lubricant	MIL-L-8937	Drilube # 2	Drilube Company 711 W. Broadway Glendale, CA 91204
		Molydag 254RFU	Acheson Colloids Co. P.O. Box 288 Port Huron, MI 48060
		Kaynar K-130	Kaynar, Microdot Co. P.O. Box 3001 800 S. State College Blvd. Fullerton, CA 92631
9. Sealant	MIL-S-8802, Type II	Pro Seal 890	Pro-Seal Corp., Div. of Essex Chemical Corp., Aerospace Products Compton, CA 90220
10. Cleaner	TT-M-261	MEK	Locally available

PASSENGER/CREW - DESCRIPTION AND OPERATION (Effectivity: All) (Figures 1 and 2)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

A swing-down cabin entrance door, located at the LH forward section of the fuselage, provides a convenient stairway for entry and exit. Plastic-encased cables support the door in the open position, provide a hand hold for passengers, and assist in closing the door from inside the cabin. A 115 vac power supply and three electroluminescent light panels are installed on the entrance door for illumination of the steps. The cabin entrance door exterior handle is provided with a key lock for security.

A spring-loaded, gas-pressurized damper assembly permits the entrance door to lower gradually as it opens and assists in the closing of the door. The damper, by metering hydraulic fluid, regulates the opening speed of the door but does not regulate the closing speed. The damper action, when malfunctioning, may be delayed, intermittent, or non-existent. If the damper malfunctions, the forces encountered when the opening door reaches the end of its travel may damage the door hinge or the attachment points for the damper or the assist cables. The person opening the door should be prepared to control the door opening forces should the damper function poorly. The damper should be rigged so that its travel limit (42 ± 0.5 inches eye-to-eye) is not exceeded when the door is in the fully open position. The door is supported in the fully open position by the assist cables, not the damper. The damper has a preload of 40 ± 5 pounds tending to shorten the assembly when retracted and a force of 140 ± 5 pounds tending to shorten the assembly when extended.

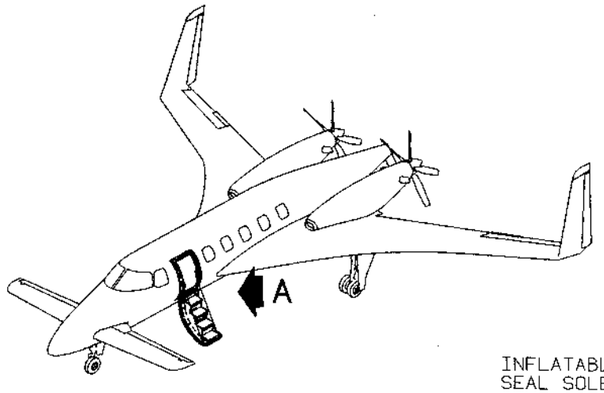
The cabin entrance door utilizes both compressive and inflatable seals to provide a positive seal of the pressure vessel during flight. The compressive door seal consists of a sponge rubber extrusion bonded to the mating edge of the entrance door. The inflatable door seal is installed in a seal retainer secured to the fuselage structure doorframe. A solenoid, located under the floorboard panel just inboard of the door hinge, receives triple-fed bus voltage through the PRESS SYS circuit breaker CB122 (pilot's CB panel) when the handle-position sense switch (S278) is actuated to the locked position by the interior handle. As the normally open door-seal solenoid energizes to the closed position, regulated bleed air pressure is routed to the inflatable seal. See Chapter 52-70-00 for an electrical schematic for the door-seal solenoid valve.

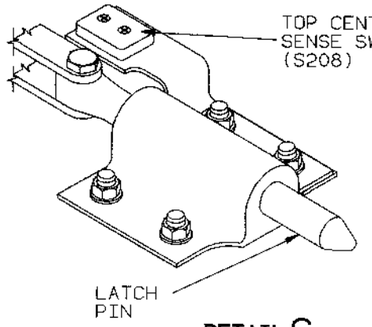
The locking mechanism for the cabin entrance door is operated by two mechanically interconnected handles (interior and exterior). When either handle is rotated to the closed and locked position, the seven latch pins enter their respective holes in the striker plates positioned around the perimeter of the cabin doorframe. The top latch pin enters the striker plate at the top center of the doorframe, and the remaining six enter the striker plates, three per side, on each side of the doorframe. There are six sight openings in the entrance door upholstery and a seventh in the cabin headliner which allow proper engagement to be visually verified for all seven latch pins. The door is considered locked when the green stripe on each latch pin is visible through its respective upholstery panel sight opening. Proper engagement of each latch pin is also verified by actuation of the individual latch-pin sense switches. The door sense switches are wired in parallel so that all seven sense switches must be actuated for the the red DOOR UNLOCKED warning annunciator to extinguish. Refer to Chapter 52-70-00 for detailed information on the door warning system.

WARNING

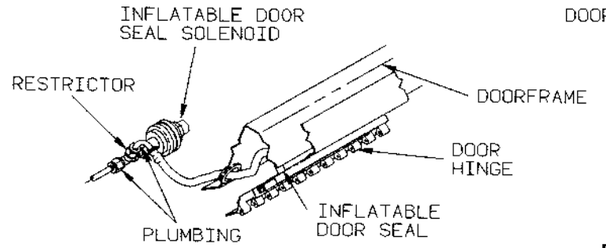
Never attempt to unlock, or even check the security of, the cabin entrance door during flight.

A pressure lock-release button, adjacent to both the interior and exterior door handles, must be pressed and held before either handle can be rotated to unlock the door. The interior door handle is locked in position when the lock-release button moves toward the operator. As the cabin is pressurized, cabin pressure moves the bellows in the air pressure lock assembly to prevent accidental opening of the cabin door while the airplane is still pressurized.





DETAIL C



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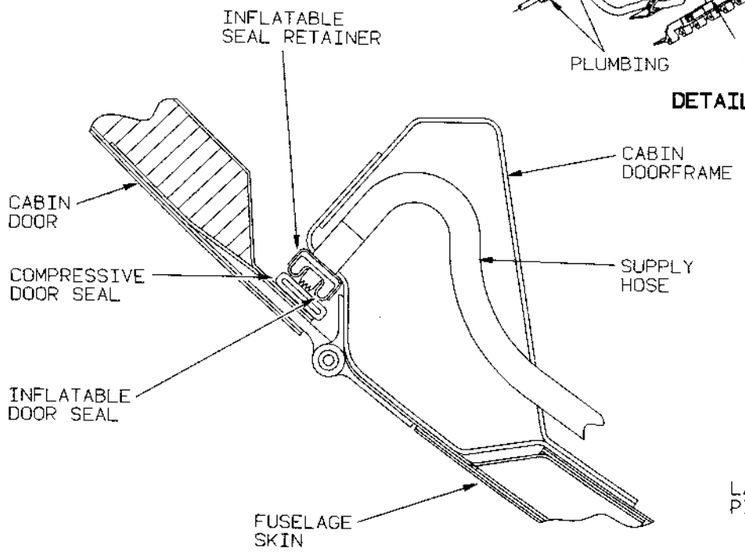
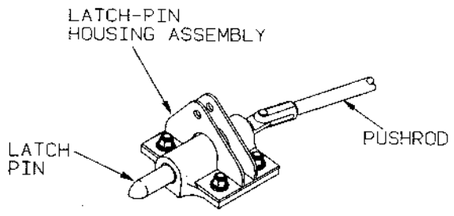


DETAIL B
(LOOKING FWD WITH DOOR CLOSED)



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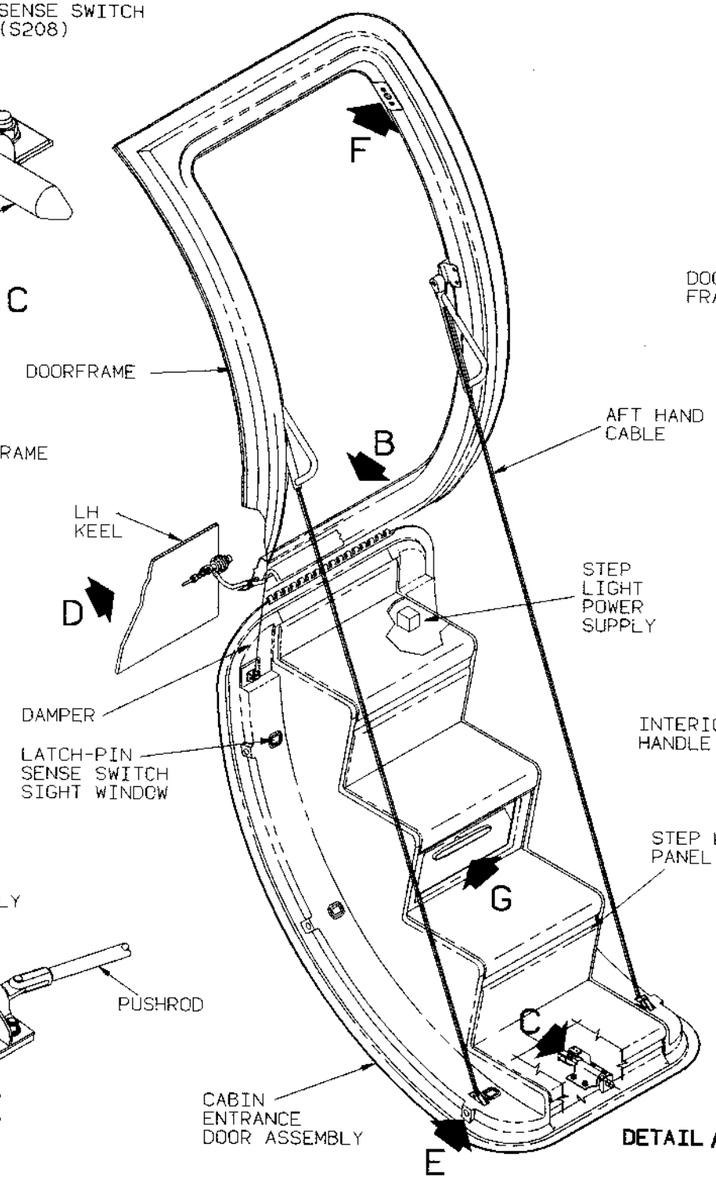
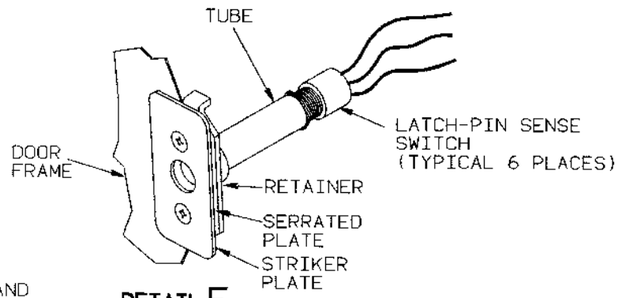


DETAIL A



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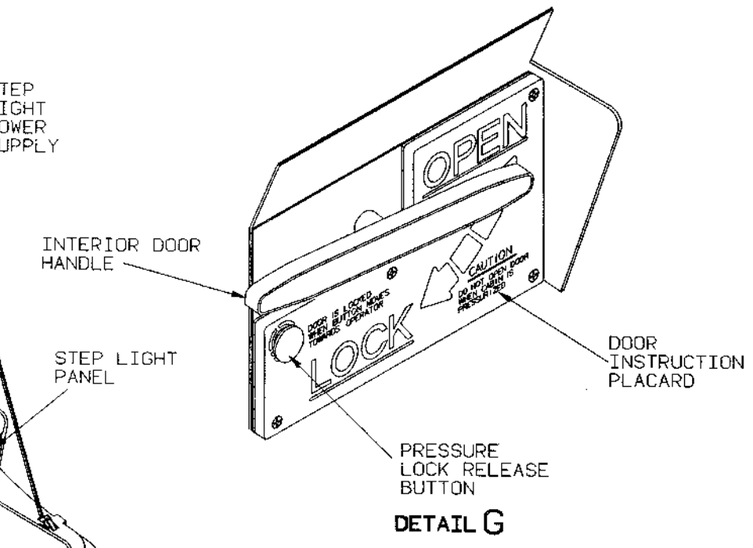

DETAIL G

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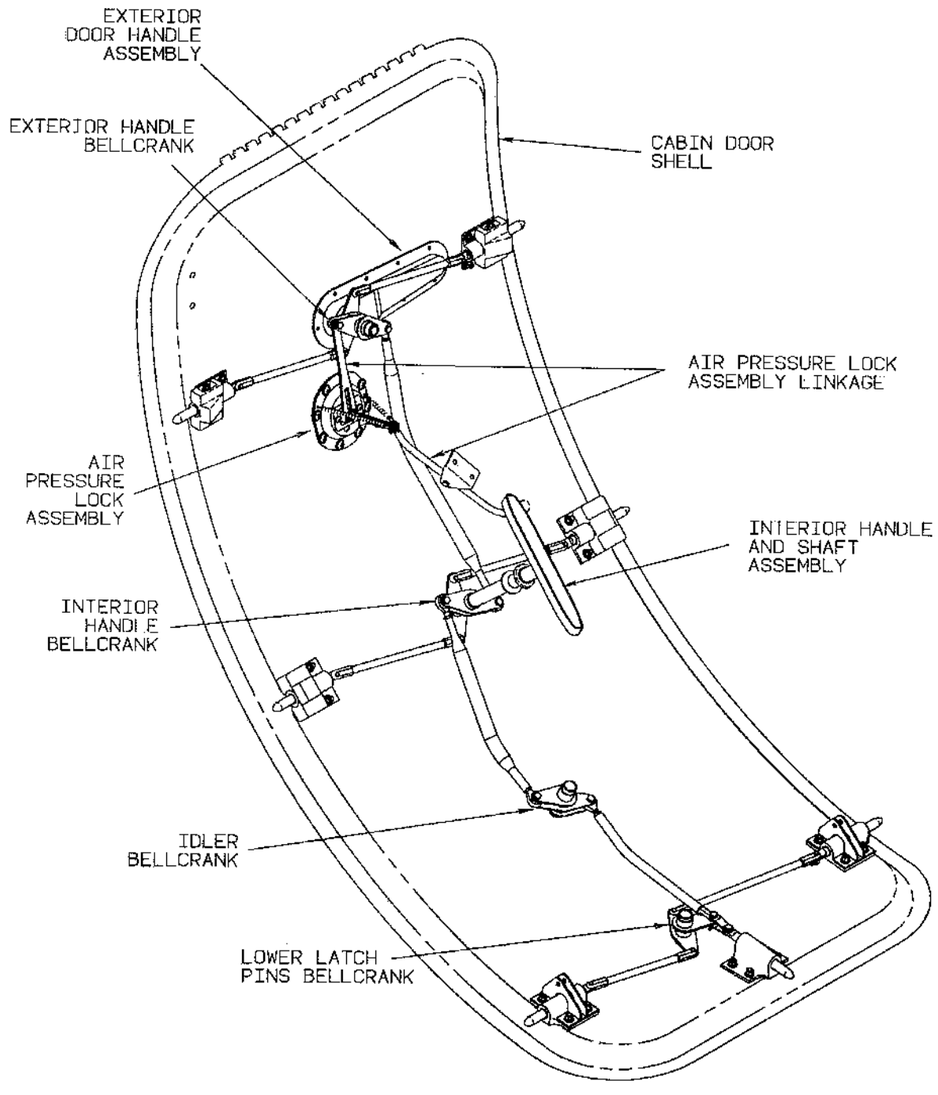
Cabin Entrance Door System
(Effectivity: All)
Figure 1



Cabin Entrance Door Mechanism (Effectivity: All)
Figure 2

PASSENGER/CREW - MAINTENANCE PRACTICES (Effectivity: All)

CABIN ENTRANCE DOOR COMPRESSIVE SEAL REMOVAL (Effectivity: All)

a. Use a white Pentel marking pencil to mark the entrance door around the inner edge of the compressive seal.

b. Use a flat nonmetallic scraper to carefully remove the old compressive seal and adhesive from the door shell edges. Never use a metal scraper as it may cause extensive damage to the composite surface.

NOTE

Use care not to remove the seal location marks made in step a to aid in locating the new seal on the door.

c. Use cleaning solvent (1, Chart 1, 52-00-00) to remove any remaining seal material or adhesive residue from the cabin door shell. Wipe the area dry with a clean, dry cloth.

CABIN ENTRANCE DOOR COMPRESSIVE SEAL INSTALLATION (Effectivity: All)

NOTE

It is acceptable to install the new seal in segments to ease installation or to facilitate repair of a small area. The minimum length of any individual segment is twelve inches.

a. Use 240-grit sandpaper to carefully scuff-sand a .75 inch wide area around the outside of the seal location mark. If the compressive seal is being installed on a new door, the .75 inch width should be centered on a line .40 inch from the outer edge of the door shell, except at the hinged edge. The portion of the seal along the hinged edge shall be bonded .82 inch from the centerline of the hinge pin.

b. Note the seal relief locations (every 12 inches) on the inside edge of the seal and ensure that they do not become closed off and prevent the seal from releasing air when compressed.

c. Carefully scuff-sand the contact side of the new seal (plugged tooling holes are located on the contact side).

d. Carefully apply adhesive (2, Chart 1, 52-00-00) to the seal bonding surface on the cabin door shell and to the bonding surface on the seal itself. Allow the adhesive to become tacky without adhering to the skin when lightly touched.

e. Align the compressive seal with the white reference line.

f. Apply light pressure to the outer edge of the compressive seal and allow the adhesive to set.

g. Seal all butt joints of the compressive seal with adhesive (3, Chart 1, 52-00-00). Apply the adhesive to both bonding surfaces and allow adequate time for the adhesive to set as recommended by the manufacturer's instructions.

CABIN DOOR INFLATABLE SEAL REMOVAL (Effectivity: All)

a. Refer to Chapter 53-20-00 and remove the cabin floorboard panel (151AT) just inboard of the cabin door threshold.

b. Remove and discard the plastic clamp from the supply hose.

c. Disconnect the supply hose at the door solenoid plumbing.

d. Use a flat, nonmetallic scraper to carefully remove the inflatable seal from the retainer.

e. Use cleaning solvent (1, Chart 1, 52-00-00) to remove any remaining adhesive or seal material from the seal retainer.

f. Wipe the retainer bonding surface with a clean, dry cloth.

CABIN DOOR INFLATABLE SEAL INSTALLATION (Effectivity: All)

a. Using 240-grit sandpaper, carefully scuff-sand the flat area on the retainer where the new seal is to be bonded.

b. Clean the prepared surface of the retainer with cleaning solvent (1, Chart 1, 52-00-00), then wipe dry with a clean, dry cloth to remove any residue.

c. Carefully scuff-sand the bonding surface on the inflatable seal with 240-grit sandpaper.

d. Clean the prepared surface on the inflatable seal with cleaning solvent (1, Chart 1, 52-00-00), then wipe dry with a clean, dry cloth to remove any residue.

e. Mix the adhesive (4, Chart 1, 52-00-00) according to the manufacturer's instructions.

f. Apply a thin layer of mixed adhesive to the bonding surfaces on the inflatable seal and seal retainer. Allow the adhesive to dry until tacky.

g. When the adhesive is still tacky, but can be touched lightly without lifting from the bonding surface, feed the supply hose into the fuselage and install the seal in the retainer.

h. Connect the supply hose on the door solenoid plumbing and secure with a new plastic clamp.

i. Allow for the adhesive to set, as recommended by the manufacturer's instructions.

j. Refer to Chapter 53-20-00 and install the cabin floorboard (151AT).

INFLATABLE DOOR-SEAL SOLENOID REMOVAL (Effectivity: All) (Figure 3)

a. Pull the circuit breaker placarded PRESS SYS on the pilot's circuit breaker panel.

b. Refer to Chapter 53-20-00 and remove the cabin floorboard panel (151AT) just inboard of the cabin door threshold.

c. Disconnect the two tube assemblies (1) from the tee fitting (9).

d. Disconnect the electrical connector from the solenoid (6).

e. Remove the nut (2) and washer (3) securing the mounting clamp (5) to the stud (12).

f. Remove the solenoid (6) and clamp (5) from the airplane.

g. If required, remove the clamp (5) from the solenoid (6).

h. If required, loosen the nut (8) and remove the restrictor (9) from the solenoid (6). Remove and discard the O-ring (7).

INFLATABLE DOOR-SEAL SOLENOID INSTALLATION (Effectivity: All) (Figure 3)

NOTE

Use care not to plug, cap or otherwise obstruct the solenoid exit port.

a. If required, install a new O-ring (7) on the solenoid (6).

b. If required, install the restrictor (9) in the solenoid (6) and tighten the nut (8).

c. If required, install the mounting clamp (5) on the solenoid (6).

d. Make sure the spacer (13) is installed on the stud (12). Position the solenoid (6) and the mounting clamp (5) on the stud (12) and install but do not tighten the washer (3) and the nut (2).

e. Position the solenoid (6) so that the tube assembly (1) for the supply hose (11) can be attached to the tee fitting (9).

f. Connect the two tube assemblies (1) to the tee fitting (9).

g. Secure the solenoid (6) in the clamp (5) by tightening the nut (2).

h. Connect the electrical connector to the solenoid (6).

i. Refer to Chapter 53-20-00 and install the cabin floorboard panel (151AT).

j. Close the circuit breaker placarded PRESS SYS on the pilot's circuit breaker panel.

CABIN ENTRANCE DOOR REMOVAL (Effectivity: All) (Figure 4)

a. Remove all electrical power from the airplane and pull the floodlight circuit breaker located on the battery bus circuit breaker panel.

b. Pull the circuit breaker placarded PRESS SYS on the pilot's circuit breaker panel.

c. Remove the upholstery surrounding the cabin doorframe in accordance with the ENTRY DOOR POST TRIM REMOVAL procedures in Chapter 25-20-00.

d. Identify, tag and disconnect the cabin door wiring and electrical connector.

CAUTION

Place a padded support under the cabin door anytime the forward or aft hand cable is disconnected. Never allow only one hand cable or the damper to support the entire weight of the cabin door and NEVER clamp anything directly onto the damper piston rod, as this may mar or gouge the surface and eventually cause the seals in the unit to fail and leak fluid.

e. Refer to CABIN ENTRANCE DOOR DAMPER REMOVAL and disconnect the damper assembly from the cabin door.

f. Refer to CABIN ENTRANCE DOOR FORWARD HAND CABLE REMOVAL and disconnect the forward hand cable from the cabin door.

g. Refer to CABIN ENTRANCE DOOR AFT HAND CABLE REMOVAL and disconnect the aft hand cable from the cabin door.

h. Remove the attaching screws (2) from the hinge pin retainers (1), then remove the hinge pin retainers.

i. Remove the cabin door hinge pin (4) to separate the hinge halves (3 and 5), then remove the cabin door from the airplane.

CABIN ENTRANCE DOOR INSTALLATION (Effectivity: All) (Figure 4)

NOTE

It is acceptable to apply a light coat of lubricant (5, Chart 1, 52-00-00) to the hinge pin (4) and hinge halves (3 and 5) to facilitate installation of the hinge pin.

a. While providing adequate support to the cabin door, align the hinge halves (3 and 5) and insert the hinge pin (4).

b. Install the hinge pin retainers (1) and secure with the hinge pin retainer attaching screws (2).

c. Refer to CABIN ENTRANCE DOOR AFT HAND CABLE INSTALLATION and reconnect the aft hand cable to the cabin door.

d. Refer to CABIN ENTRANCE DOOR DAMPER INSTALLATION and reconnect the damper assembly to the cabin door.

e. Refer to CABIN ENTRANCE DOOR FORWARD HAND CABLE INSTALLATION and reconnect the forward hand cable to the cabin door.

f. Connect the cabin door wiring and electrical connector.

g. Install the upholstery surrounding the cabin doorframe in accordance with the ENTRY DOOR POST TRIM INSTALLATION procedures in Chapter 25-20-00.

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

h. With the cabin door closed, turn the inside handle to the closed and locked position. Excessive force should not be required to engage the latch pins.

i. Repeat step h with the outside handle. If the door mechanism fails to operate smoothly, refer to the bellcrank installation procedures in this chapter for shimming instructions.

j. Close the circuit breaker placarded PRESS SYS on the pilot's circuit breaker panel.

k. Close the floodlight circuit breaker located on the battery bus circuit breaker panel.

l. Restore electrical power to the airplane.

CABIN ENTRANCE DOOR REPLACEMENT (Effectivity: All) (Figure 4)

Should the cabin entrance door require replacement, remove the old door from the airplane by performing the CABIN ENTRANCE DOOR REMOVAL procedures in this chapter.

NOTE

A hinge half is already installed on the new door assembly to facilitate ease of door installation. Perform the following procedure to properly install and match a new cabin door with the doorframe.

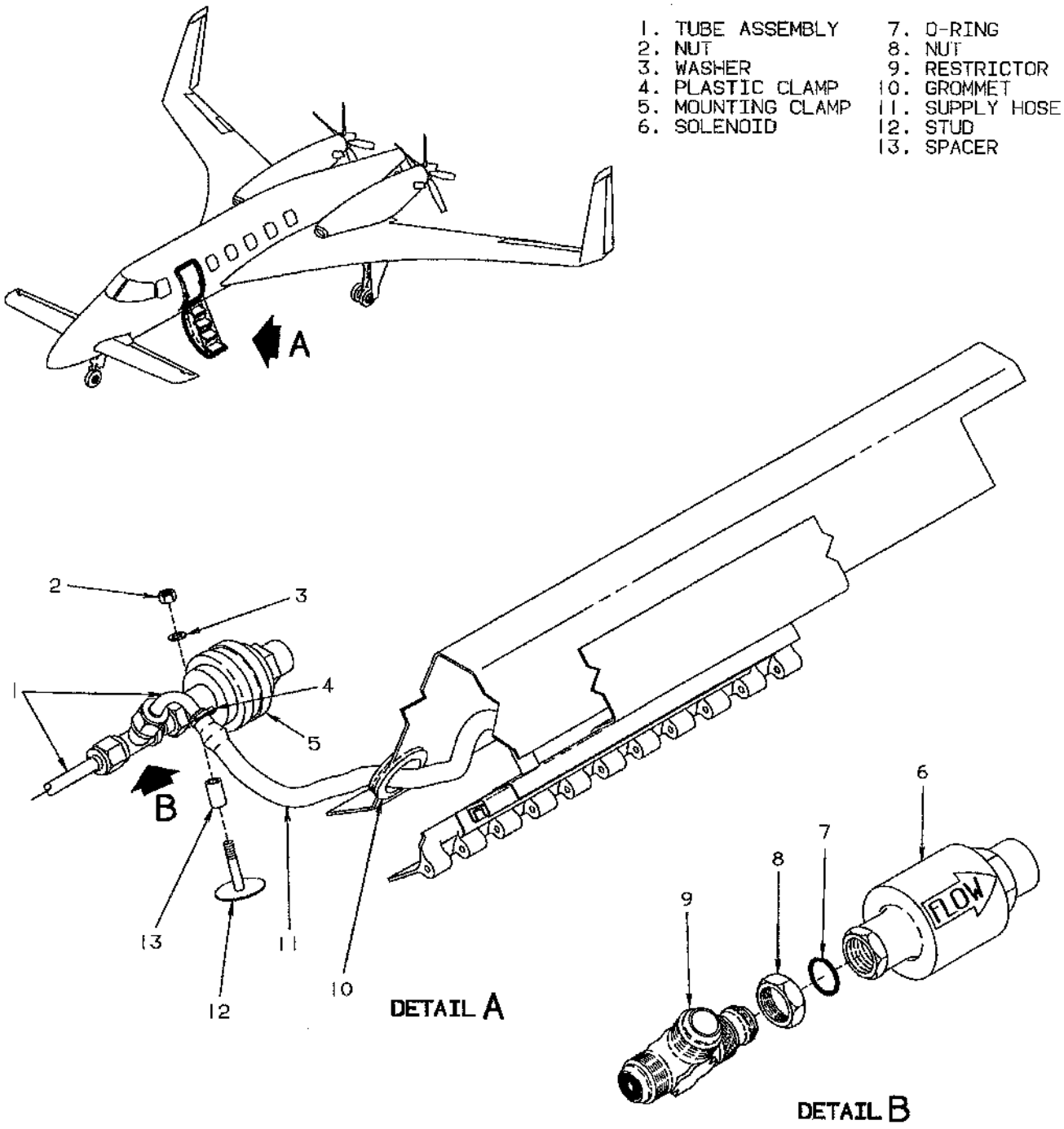
a. Position the old and new door in a convenient way to allow the transfer of the door components.

b. Note the position of each mechanical and electrical component on the old door as it is removed. Install the mechanical and electrical components on the new door.

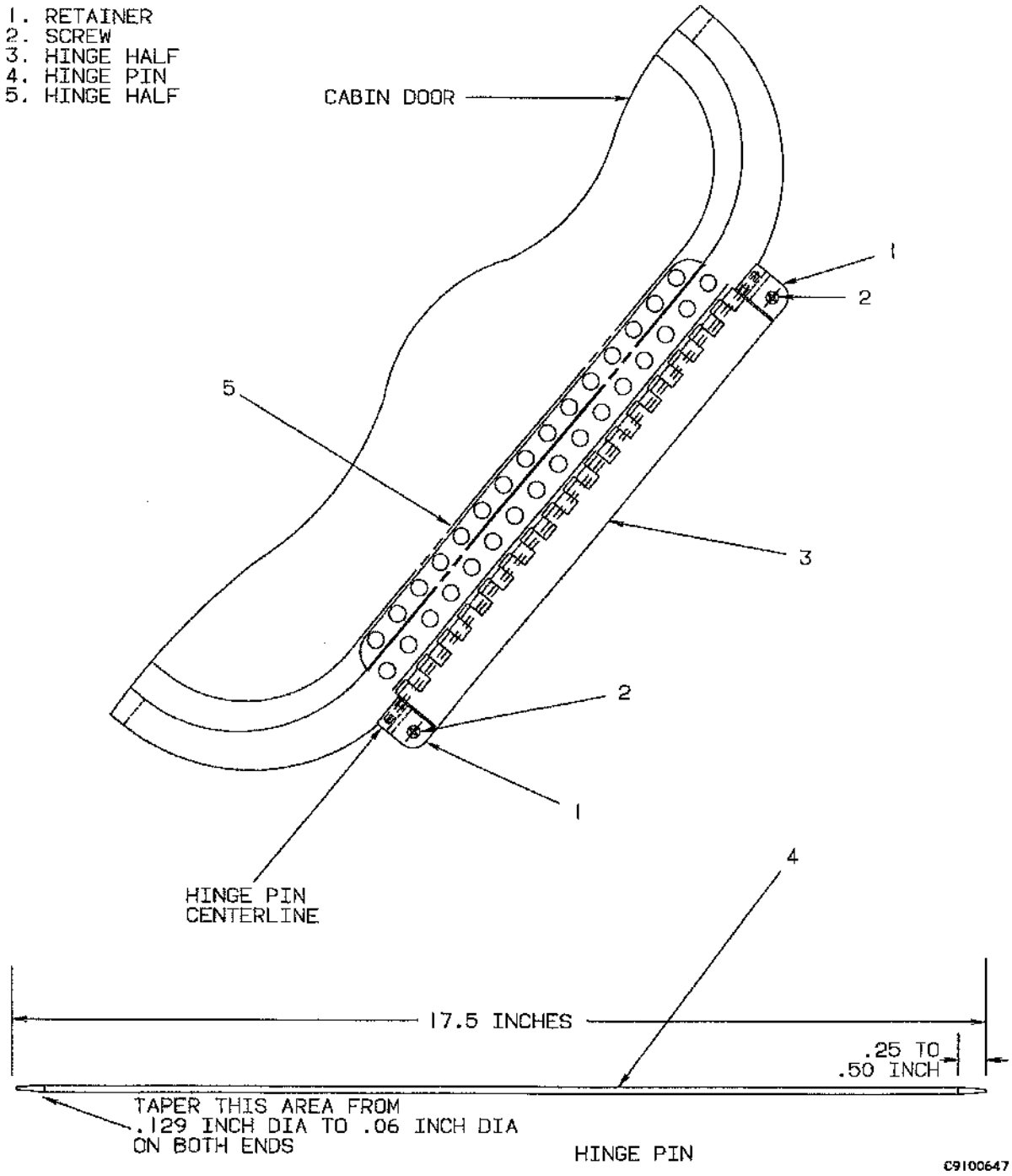
NOTE

It is acceptable to apply a light coat of lubricant (5, Chart 1, 52-00-00) to the hinge pin (4) to ease insertion.

c. While supporting the cabin door assembly, align the two hinge halves (3 and 5) and install the hinge pin (4).



Inflatable Door Seal Solenoid Installation (Effectivity: All)
Figure 3



Cabin Entrance Door Hinge Installation
(Effectivity: All)
Figure 4

d. Carefully place a piece of two-inch-wide masking tape (6, Chart 1, 52-00-00) around the perimeter of the doorframe where it meets the fuselage skin. The tape must closely match the outside contour of the doorframe to ensure a good fit.

e. Carefully close the cabin door as far as possible without binding the door skin in the frame.

f. Position a second layer of masking tape (6, Chart 1, 52-00-00) directly over the tape already on the fuselage skin. The outer edges of both tape strips must be approximately matched to form an accurate trim pattern on the door skin.

CAUTION

Use care not to cut into the fuselage and door skins while trimming the tape.

g. Use a sharp knife to carefully remove the excess tape extending past the edge of the door, then open the door.

h. Ensure that the cabin door is adequately supported from the bottom by a padded support.

CAUTION

Never sand the flat surface of the cabin door. Ensure that composite material is removed from the door edges only.

i. Install a new sanding disc (coarse grit) on a Datco air tool and carefully begin removing the excess material at the hinged side of the door assembly.

NOTE

Work upward from the contoured edges at the hinge line of the door. This will allow the technician to move the upper portion of the door closer to the doorframe since the .030- to .050-inch gap is to be attained at the lower end of the door.

j. After a .030- to .050-inch gap is established around the entire perimeter of the door, remove the remaining tape from the skins of the fuselage and door.

k. Refer to CABIN ENTRANCE DOOR COMPRESSIVE SEAL INSTALLATION and install the compressive seal on the door assembly.

l. Align the steps with the inserts in the door assembly structure and secure with screws as shown by Figure 12.

m. Refer to CABIN ENTRANCE DOOR AFT HAND CABLE INSTALLATION and reconnect the aft hand cable to the door.

n. Refer to CABIN ENTRANCE DOOR DAMPER INSTALLATION and reconnect the damper assembly to the door.

o. Refer to CABIN ENTRANCE DOOR FORWARD HAND CABLE INSTALLATION and reconnect the forward hand cable to the door.

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

p. With the cabin door closed, turn the inside handle to the closed and locked position. Excessive force should not be required to engage the latch pins. If excessive force is required to lock the door, refer to the bellcrank installation procedures in this chapter for shimming instructions.

q. Connect the cabin door wiring and electrical connector to the airplane harness.

r. Perform the HANDLE -POSITION SENSE SWITCH ADJUSTMENT procedures in Chapter 52-70-00.

s. Apply electrical power to the airplane.

t. Verify that the latch pins are properly engaged and that the DOOR UNLOCKED annunciator is no longer illuminated. Unlock and lock the door several times to verify proper operation of the door warning system.

u. Remove electrical power from the airplane.

v. Refer to Chapter 25-20-00 and install the cabin door upholstery.

w. Refer to Chapter 33-20-00 and install the cabin door step light panels.

x. Install the upholstery surrounding the cabin doorframe in accordance with the ENTRY DOOR POST TRIM INSTALLATION procedures in Chapter 25-20-00.

CABIN ENTRANCE DOOR DAMPER REMOVAL (Effectivity: All) (Figure 5)

WARNING

The damper is pressurized and may cause injury to personnel if the assembly is punctured or incinerated.

CAUTION

Place a padded support under the cabin door anytime the forward or aft hand cable is disconnected. Never allow only one hand cable or the damper to support the entire weight of the cabin door and NEVER clamp anything directly onto the damper piston rod, as this may mar or gouge the surface and eventually cause the seals in the unit to fail and leak fluid.

a. Close the cabin door to relieve strain from the damper attach point.

WARNING

The damper assembly is preloaded to the retracted position. Use care when disconnecting the damper assembly.

b. Remove the nut (16) and washer (15) from the bolt (13) attaching the damper piston rod end (17) to the support fitting (14).

c. Open the cabin door.

d. Remove the bolt (13) and washers (18 and 19), then disconnect the damper piston rod end (17) from the support fitting (14).

e. Remove the cotter pin (12), nut (1) and washer (2) from the bolt (9).

f. Slide the bolt (9) out far enough to remove the washers (3 and 5) and disconnect the damper (4) from the forward support fitting (6).

g. Remove the damper (4) from the airplane.

CABIN ENTRANCE DOOR DAMPER INSTALLATION (Effectivity: All) (Figure 5)

WARNING

The damper is pressurized and may cause injury to personnel if the assembly is punctured or incinerated.

a. With the bolt (9) still attaching the forward hand cable (10), align the damper (4) and washers (3 and 5) with the forward support fitting (6), then slide the bolt (9) completely through the forward support fitting (6).

b. Install the washer (2) and nut (1) on the bolt (9).

c. Install the cotter pin (12) and verify that the damper (4) and forward hand cable (10) both move freely.

WARNING

The damper assembly is preloaded to the retracted position. Use care when connecting the damper assembly.

d. Pull the door towards the closed position until the damper piston rod end (17) is aligned with the support fitting (14) on the cabin door.

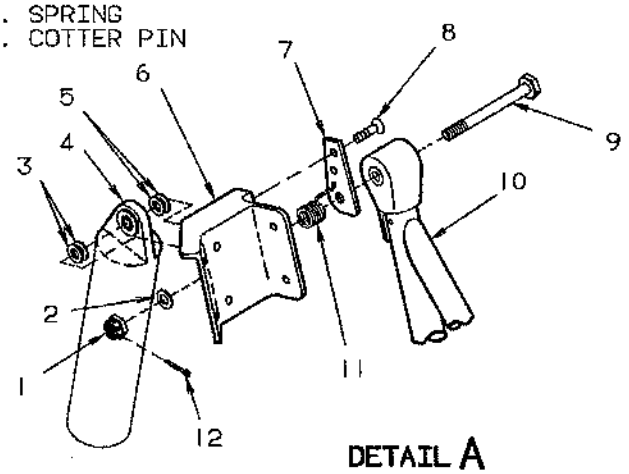
e. Align the washers (18 and 19) with the support fitting (14) and damper piston rod end (17), then slide the bolt (13) through the support fitting, rod end and washers. Secure the bolt (13) with washer (15) and nut (16).

CAUTION

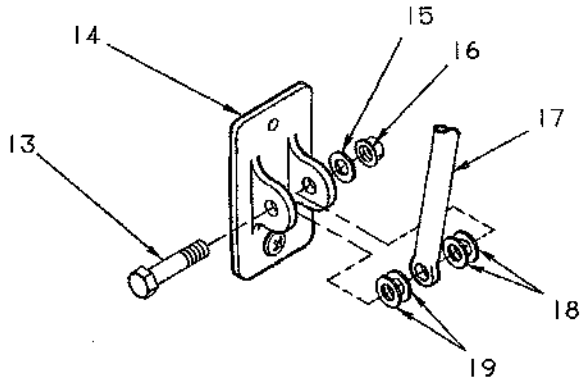
The maximum extension of the damper is 42 ± 0.5 inches from the center of the eyes on each end of the damper. Do not allow the damper to exceed this value when opening the door.

f. Allow the cabin door to slowly swing down to the fully open position. The damper assembly should move freely at both ends and permit the door to slowly extend to the fully open position. The hand cables should prevent the damper from exceeding its

1. NUT
2. WASHER
3. WASHERS
4. DAMPER
5. WASHERS
6. FORWARD SUPPORT FITTING
7. SUPPORT PLATE
8. SCREW
9. BOLT
10. FORWARD HAND CABLE
11. SPRING
12. COTTER PIN
13. BOLT
14. SUPPORT FITTING
15. WASHER
16. NUT
17. DAMPER PISTON ROD END
18. WASHERS
19. WASHERS



DETAIL A



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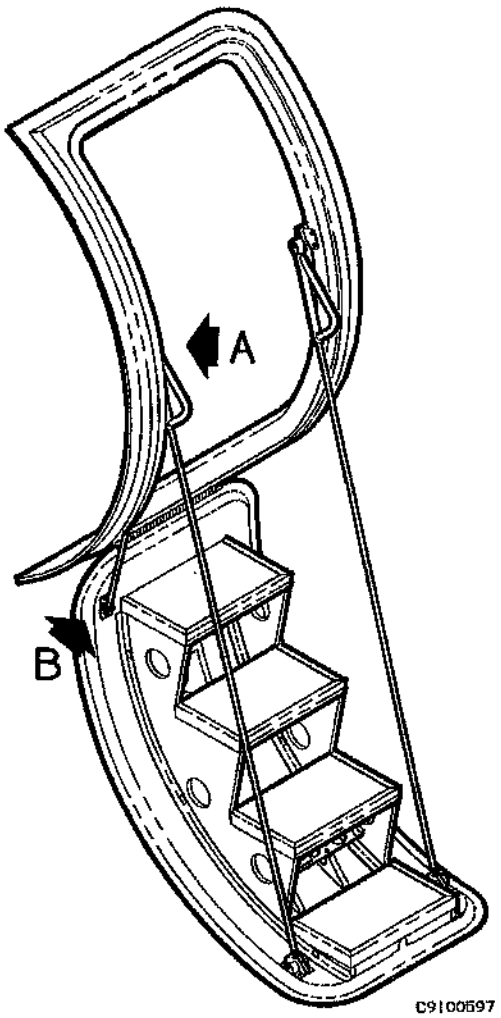


Cabin Entrance Door Damper Assembly Installation (Effectivity: All)
Figure 5

designed extension length. If the hand cables do not support the weight of the door and a person before the damper is fully extended, disconnect the applicable hand-cable clevis, loosen the jam nuts and turn the clevis as necessary to adjust cable length. After final adjustment, tighten the jam nuts and reconnect the clevis. The hand cables must supply support evenly when weight is applied to the door.

CABIN ENTRANCE DOOR FORWARD HAND CABLE REMOVAL (Effectivity: All) (Figure 6)

CAUTION

Place a padded support under the cabin door anytime the forward or aft hand cable is to be disconnected. Never allow only one hand cable or the damper assembly to support the entire weight of the cabin door, and NEVER clamp anything directly on to the damper piston rod as it may mar or gouge the surface and eventually cause the seals in the unit to fail and leak fluid.

NOTE

On airplanes NC-22 and after, the hand cable clevis (15) is secured with a bolt (17), washer (14) and nut (13) instead of a pin (22) and cotter pin (19). Airplanes NC-4 thru NC-21 may also have a bolt (17), washer (14) and nut (13) installed if the original hardware was replaced.

a. For airplanes NC-4 thru NC-21 with the original hardware installed, remove the cotter pin (19) and pull the pin (22) to disconnect the hand cable clevis (21).

b. For airplanes NC-4 thru NC-21 with new hardware installed, and NC-22 and after, remove the nut (13) and washer (14), then disconnect the hand cable clevis (15) from the attach bracket (18) by removing the bolt (17).

c. Remove the cotter pin (12), nut (1) and washer (2) from the bolt (9).

d. Slide the bolt (9) out far enough to disconnect the damper (4) and remove the washers (3 and 5).

e. Slide the bolt (9) the rest of the way out of the forward support fitting (6) and disconnect the forward hand cable (10) from the doorframe attach point.

NOTE

The spring (11) may be accessed by removing the screws (8) that secure the support plate (7) to the forward support fitting (6).

CABIN ENTRANCE DOOR FORWARD HAND CABLE INSTALLATION (Effectivity: All) (Figure 6)

NOTE

If the spring (11) was removed, reinstall it in the forward support fitting (6) and align the support plate (7) with the fitting. Apply retaining compound (7, Chart 1, 52-00-00) to the screw (8) threads and secure the support plate (7) with the screws (8).

a. Align the forward hand cable (10) with the forward support fitting (6) and insert the bolt (9).

b. Align the damper (4) and washers (3 and 5) with the forward support fitting (6) and slide the bolt (9) completely through the assembly.

c. Install the washer (2) and nut (1) on the bolt (9) and secure with cotter pin (12).

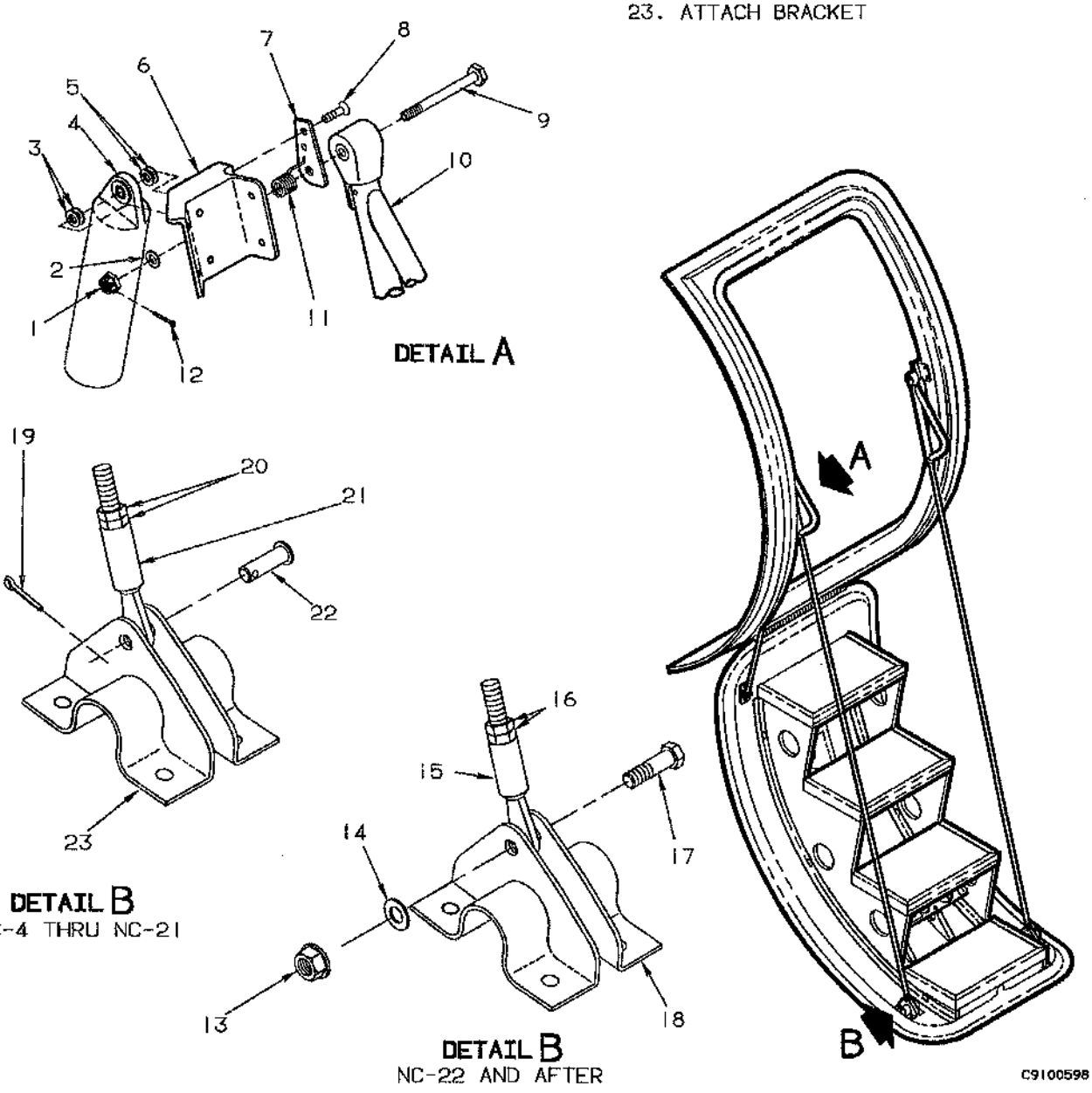
NOTE

On airplanes NC-22 and after, the hand cable clevis (15) is secured with a bolt (17), washer (14) and nut (13) instead of the pin (22) and cotter pin (19) installation. Airplanes NC-4 thru NC-21 may also have the bolt (17), washer (14) and nut (13) installed as a replacement.

d. For airplanes NC-4 thru NC-21 with the original hardware installed, align the hand cable clevis (21) with the attach bracket (23) and install the pin (22). Secure with cotter pin (19).

e. For airplanes NC-4 thru NC-21 with new hardware installed and on NC-22 and after, align the hand cable clevis (15) with the attach bracket (18) and install the bolt (17). Secure the bolt (17) with washer (14) and nut (13).

1. NUT
2. WASHER
3. WASHERS
4. DAMPER
5. WASHERS
6. FORWARD SUPPORT FITTING
7. SUPPORT PLATE
8. SCREW
9. BOLT
10. FORWARD HAND CABLE
11. SPRING
12. COTTER PIN
13. NUT
14. WASHER
15. HAND CABLE CLEVIS
16. JAM NUTS
17. BOLT
18. ATTACH BRACKET
19. COTTER PIN
20. JAM NUTS
21. HAND CABLE CLEVIS
22. PIN
23. ATTACH BRACKET



Cabin Entrance Door Forward Hand Cable Installation (Effectivity: All)
Figure 6

CAUTION

The maximum extension of the damper is 42 ± 0.5 inches from the center of the eyes on each end of the damper. Do not allow the damper to exceed this value when opening the door.

f. Allow the cabin door to slowly swing down to the fully open position. The damper assembly should move freely at both ends and permit the door to slowly extend to the fully open position. The hand cables should prevent the damper from exceeding its designed extension length. If the hand cables do not support the weight of the door and a person before the damper is fully extended, disconnect the applicable hand-cable clevis, loosen the jam nuts and turn the clevis as necessary to adjust cable length. After final adjustment, tighten the jam nuts and reconnect the clevis. The hand cables must supply support evenly when weight is applied to the door.

g. Close and open the cabin entrance door to verify that the hand cables are even, move freely and stow properly.

CABIN ENTRANCE DOOR AFT HAND CABLE REMOVAL (Effectivity: All) (Figure 7)

CAUTION

Place a padded support under the cabin door anytime the forward or aft hand cable is to be disconnected. Never allow only one hand cable or the damper assembly to support the entire weight of the cabin door, and NEVER clamp anything directly on to the damper piston rod as it may mar or gouge the surface and eventually cause the seals in the unit to fail and leak fluid.

a. Raise the cabin door enough to remove pressure from the hand cable door attach point.

NOTE

On airplanes NC-22 and after, the hand cable clevis (19) is secured with a bolt (20), washer (17) and nut (16) instead of a pin (14) and cotter pin (11). Airplanes NC-4 thru NC-21 may have the bolt (20), washer (17) and nut (16) installed as replacement hardware.

b. For airplanes NC-4 thru NC-21 with the original hardware installed, remove the cotter pin (11) and pull the pin (14) to disconnect the hand cable clevis (12).

c. For airplanes NC-4 thru NC-21 with new hardware installed and on NC-22 and after, remove the nut (16) and washer (17), then disconnect the hand cable clevis (19) from the attach bracket (21) by removing the bolt (20).

d. Remove the cotter pin (8), nut (7) and washer (6) from the bolt (10).

e. Slide the bolt (10) out, remove the spacer (4) and remove the aft hand cable (1) from the airplane.

NOTE

The spring (9) may be accessed by removing the two screws (2) that secure the support plate (3) to the aft support fitting (5).

CABIN ENTRANCE DOOR AFT HAND CABLE INSTALLATION (Effectivity: All) (Figure 7)

NOTE

If the spring (9) was removed, reinstall it in the aft support fitting (5) and align the support plate (3) with the fitting. Apply retaining compound (7, Chart 1, 52-00-00) to the screw (2) threads and secure the support plate (3) with screws (2).

a. Align the aft hand cable (1) and spacer (4) with the aft support fitting (5) and insert the bolt (10).

b. Install the washer (6) and nut (7) on the bolt (10) and secure with cotter pin (8).

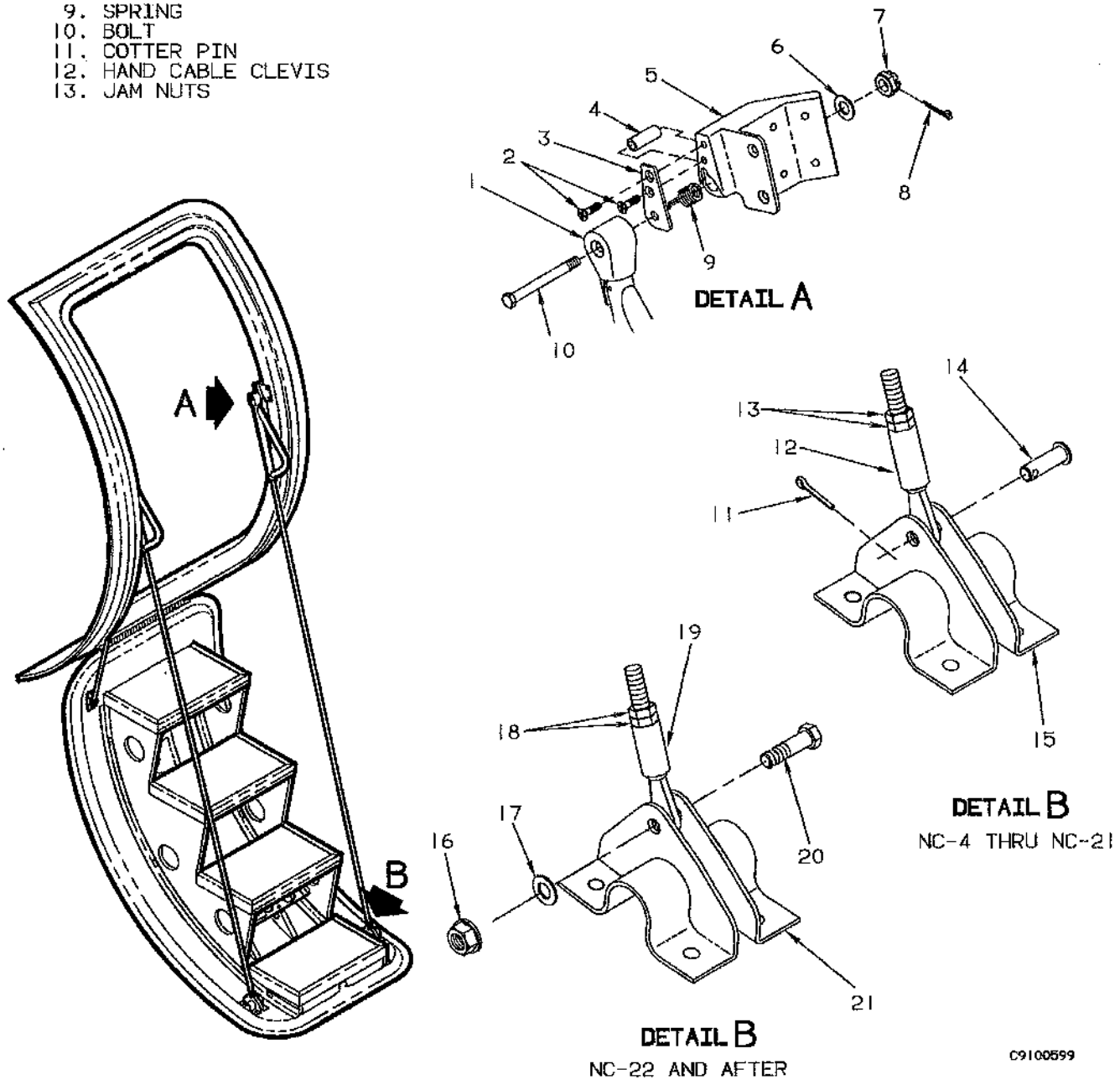
c. For airplanes NC-4 thru NC-21 with the original hardware installed, align the hand cable clevis (12) with the attach bracket (15) and install the pin (14). Secure with cotter pin (11).

d. For airplanes NC-4 thru NC-21 with new hardware installed and on NC-22 and after, align the hand cable clevis (19) with the attach bracket (21) and install the bolt (20). Secure the bolt (20) with washer (17) and nut (16).

CAUTION

The maximum extension of the damper is 42 ± 0.5 inches from the center of the eyes on each end of the damper. Do not allow the damper to exceed this value when opening the door.

1. AFT HAND CABLE
2. SCREWS
3. SUPPORT PLATE
4. SPACER
5. AFT SUPPORT FITTING
6. WASHER
7. NUT
8. COTTER PIN
9. SPRING
10. BOLT
11. COTTER PIN
12. HAND CABLE CLEVIS
13. JAM NUTS
14. PIN
15. ATTACH BRACKET
16. NUT
17. WASHER
18. JAM NUTS
19. HAND CABLE CLEVIS
20. BOLT
21. ATTACH BRACKET



Cabin Entrance Door Aft Hand Cable Installation (Effectivity: All)
Figure 7

e. Allow the cabin door to slowly swing down to the fully open position. The damper assembly should move freely at both ends and permit the door to slowly extend to the fully open position. The hand cables should prevent the damper from exceeding its designed extension length. If the hand cables do not support the weight of the door and a person before the damper is fully extended, disconnect the applicable hand-cable clevis, loosen the jam nuts and turn the clevis as necessary to adjust cable length. After final adjustment, tighten the jam nuts and reconnect the clevis. The hand cables must supply support evenly when weight is applied to the door.

f. Close and open the cabin entrance door and verify that the hand cables are even, move freely and stow properly.

TOP BELLCRANK REMOVAL
(Effectivity: All)
(Figure 8)

a. Refer to Chapter 33-20-00 and remove the step light panel from the lower step.

b. Remove upholstery from the cabin door as necessary to access the lower step attaching hardware.

c. Refer to Figure 12 and remove the lower step panel (2) from the step panel support (5) by removing the attaching screws (1) from the step angle (3).

NOTE

Ensure that all shims are saved for reinstallation. Refer to Chapter 20-00-05 for information on bushing replacement.

d. If the bushing (5) is to be replaced, mark the position of the upper bushing support (4), then remove the four screws (3) and washers (39) attaching it and any shims to the lower step panel (2, Figure 12).

e. Remove the cotter pin (22), nut (21), washer (20) and bolt (6) connecting the clip (14) and adjustable pushrod assembly (2) to the bellcrank (35).

f. Remove the cotter pin (24), washer (25) and pin (34) to disconnect the pushrod assembly (33) from the bellcrank (35).

g. Remove the cotter pin (36), washer (37) and pin (38) to disconnect the pushrod assembly (7) from the bellcrank (35).

h. Lift the bellcrank (35) from the lower bushing support (23).

NOTE

Ensure that all metal shims are saved for reinstallation of the bushing support (23). Refer to Chapter 20-00-05 for information on bushing replacement.

i. If the bushing (26) is to be replaced, mark the position of the lower bushing support (23), remove the four attaching screws (27) and washers (40), then remove the bushing support (23) and any metal shims from the door shell assembly.

TOP BELLCRANK INSTALLATION
(Effectivity: All)
(Figure 8)

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

a. If the lower bushing support (23) was removed, align it and any metal shims with the door shell as marked, then secure with screws (27) and washers (40).

b. Lubricate the end of the bellcrank (35) and insert it into the bushing (23).

c. Verify that the adjustable pushrod (2) measures 10.54 inches from eye-to-eye before installation. If not, disconnect it from the idler bellcrank (1) and adjust as necessary.

d. Align the adjustable pushrod assembly (2) with the bellcrank (35) and position the open end of the clip (14) over the adjustable pushrod assembly (2). Secure to the bellcrank (35) by installing the bolt (6), washer (20), nut (21) and cotter pin (22).

e. Align the pushrod assembly (7) with the bellcrank (35) and install the pin (38), washer (37) and cotter pin (36).

f. Align the pushrod assembly (33) with the bellcrank (35) and install the pin (34), washer (25) and cotter pin (24).

g. If the upper bushing support (4) was removed, position it and any existing shims on the lower step as marked, then secure with screws (3) and washers (39).

h. Capture the bellcrank (35) with the bushing (5) while positioning the lower step panel over the step panel support. The end of the bellcrank (35) must be lubricated before insertion into the bushing (5).

i. Align the lower step panel (2) and step angle (3) with the step panel support (5), then secure with screws (1 and 4) as shown by Figure 12.

j. Move the bellcrank (35) up and down to determine the amount of bellcrank end play. If end play is between .010 and .030 inch, proceed to step k, otherwise shim the lower bushing support as follows:

1. Remove the lower bushing support (23) and install a .060 inch MIL-S-22499 peelable shim over 100% of the bottom of the bushing support where it contacts the existing metal shim or door shell.

2. Peel the shim as necessary to attain .010 to .030 inch of bellcrank end play.

3. Reassemble the bushing support (4) with the bellcrank (35) and verify that bellcrank end play is now between .010 and .030 inch.

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

k. Actuate the cabin door handle to verify smooth operation of the entire locking mechanism. The excessive force should not be required to engage the latch pins and all seven latch pins must fully retract into the latch-pin housings with no evidence of binding. Adjust the mechanism as necessary.

l. Reinstall the upholstery on the cabin door.

m. Refer to Chapter 33-20-00 and install the step light panel.

TOP LATCH PINS REMOVAL
(Effectivity: All)
(Figure 8)

a. Refer to Chapter 33-20-00 and remove the step light panel from the lower step.

b. Remove upholstery from the cabin door as necessary to access the lower step attaching hardware.

c. Refer to Figure 12 and remove the lower step panel (2) from the step panel support (5) by removing the attaching screws (1) from the step angle (3).

NOTE

Perform step d for the top center latch pin (15), steps e thru g for the top forward latch pin (30) and steps h thru j for the top aft latch pin (9) removal.

d. Remove the cotter pin (19), nut (18), washer (17) and bolt (13) to disconnect the clip (14) from the latch pin (15), then remove the latch pin (15) from the housing assembly (16).

e. Remove the cotter pin (24), washer (25) and pin (34) to disconnect the pushrod assembly (33) from the bellcrank (35).

f. Pull the latch pin (30) inward until accesss to the attaching hardware can be obtained.

g. Remove the cotter pin (29), washer (28) and pin (32) to disconnect the pushrod assembly (33) from the latch pin (30), then remove the latch pin (30) from the support assembly (31).

h. Remove the cotter pin (36), washer (37) and pin (38) to disconnect the pushrod assembly (7) from the bellcrank (35).

i. Pull the latch pin (9) inward until accesss to the attaching hardware can be obtained.

j. Remove the cotter pin (12), washer (11) and pin (8) to disconnect the pushrod assembly (7) from the latch pin (9), then remove the latch pin (9) from the housing assembly (10).

TOP LATCH PINS INSTALLATION
(Effectivity: All)
(Figure 8)

NOTE

Perform step a and b for the top center latch pin (15), steps c thru e for the top forward latch pin (30) and steps f thru h for the top aft latch pin (9) installation, then perform steps i thru l.

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

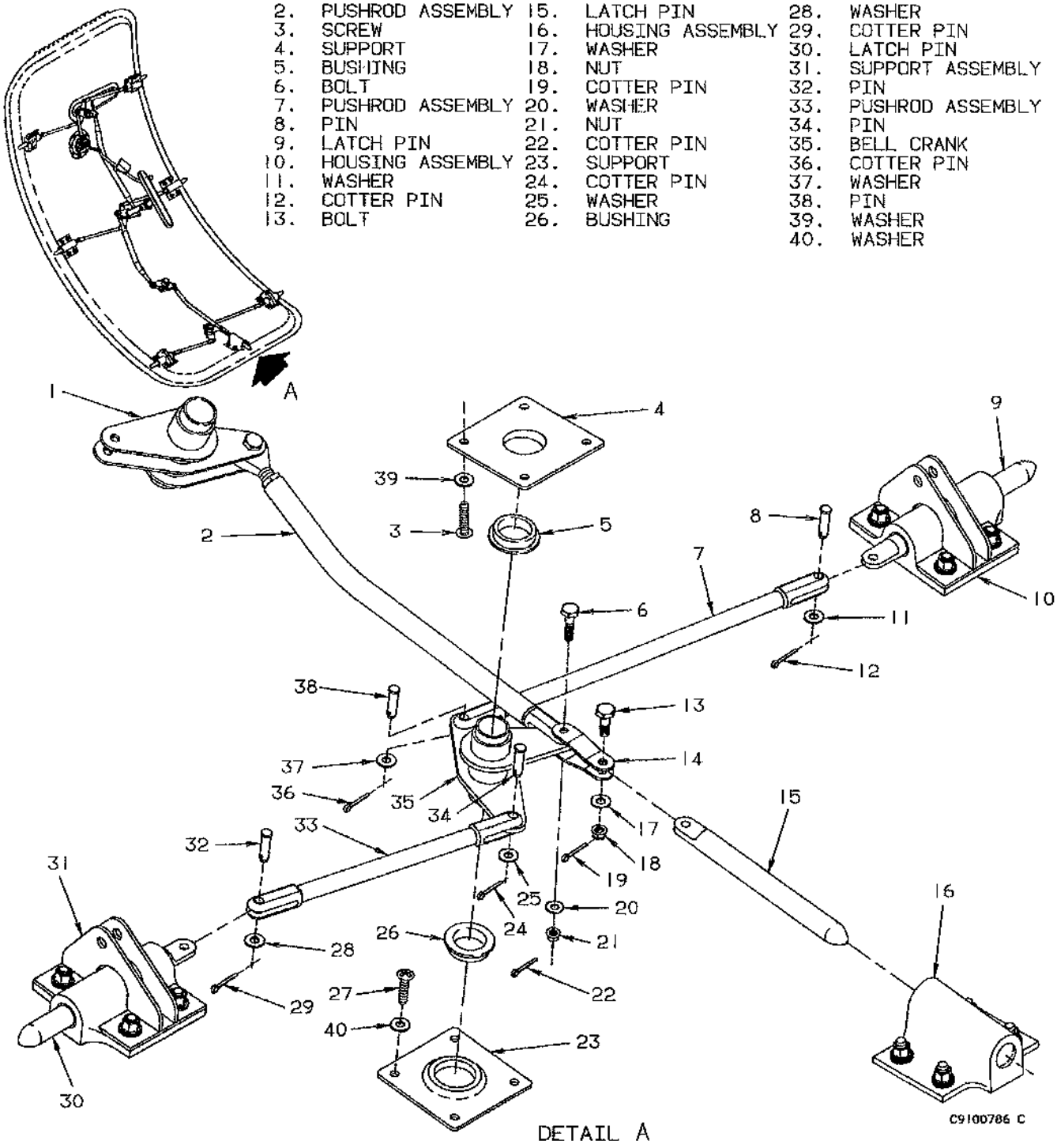
a. Lubricate the latch pin (15) sliding surface with dry lubricant (8, Chart 1, 52-00-00) and slide it into the housing assembly (16).

b. Secure the latch pin (15) to the clip (14) with bolt (13), washer (17), nut (18) and cotter pin (19).

c. Lubricate the latch pin (30) sliding surface with dry lubricant (8, Chart 1, 52-00-00) and slide it into the housing assembly (31).

d. Position the pushrod assembly (33) on the latch pin (30) and secure with the pin (32), washer (28) and cotter pin (29).

1.	IDLER BELL CRANK	14.	CLIP	27.	SCREW
2.	PUSHROD ASSEMBLY	15.	LATCH PIN	28.	WASHER
3.	SCREW	16.	HOUSING ASSEMBLY	29.	COTTER PIN
4.	SUPPORT	17.	WASHER	30.	LATCH PIN
5.	BUSHING	18.	NUT	31.	SUPPORT ASSEMBLY
6.	BOLT	19.	COTTER PIN	32.	PIN
7.	PUSHROD ASSEMBLY	20.	WASHER	33.	PUSHROD ASSEMBLY
8.	PIN	21.	NUT	34.	PIN
9.	LATCH PIN	22.	COTTER PIN	35.	BELL CRANK
10.	HOUSING ASSEMBLY	23.	SUPPORT	36.	COTTER PIN
11.	WASHER	24.	COTTER PIN	37.	WASHER
12.	COTTER PIN	25.	WASHER	38.	PIN
13.	BOLT	26.	BUSHING	39.	WASHER
				40.	WASHER



Top Bellcrank and Latch Pins Installation (Effectivity: All)
Figure 8

e. Align the pushrod assembly (33) with the bellcrank (35) and install the pin (34), washer (25) and cotter pin (24).

f. Lubricate the latch pin (9) sliding surface with dry lubricant (8, Chart 1, 52-00-00) and slide it into the housing assembly (10).

g. Position the pushrod assembly (7) on the latch pin (9) and secure with the pin (8), washer (11) and cotter pin (12).

h. Align the pushrod assembly (7) with the bellcrank (35) and install the pin (38), washer (37) and cotter pin (36).

i. Align the lower step panel (2) and step angle (3) with the step panel support (5), then secure with screws (1 and 4) as shown by Figure 12.

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

j. Actuate the cabin door handle to verify smooth operation of the entire locking mechanism. The excessive force should not be required to engage the latch pins and all seven latch pins must fully retract into the latch-pin housings with no evidence of binding. Adjust the mechanism as necessary.

k. Reinstall the upholstery on the cabin door.

l. Refer to Chapter 33-20-00 and install the step light panel.

IDLER BELLCRANK REMOVAL
(Effectivity: All)
(Figure 9)

a. Refer to Chapter 33-20-00 and remove the step light panel from the lower step.

b. Remove the upholstery from the cabin door as necessary to access the lower step attaching hardware.

c. Refer to Figure 12 and remove the step angle (3) and lower step panel (2) from the step panel support (5) by removing the attaching screws (1 and 4).

NOTE

MIL-S-22499 peelable shims are installed between the channel (55) and the support assembly (54). Record shim thickness or do not remove the shim material to provide proper mechanism alignment during reinstallation.

d. Remove the four screws and washers (57) attaching the angles (56) to the step panel supports (53). Remove the four screws and washers (58) attaching the channel (55) to the support brackets (54). Pull the channel upward to separate it from the idler bellcrank (24).

e. Disconnect the adjustable pushrod (15) from the idler bellcrank (24) by removing the cotter pin (21), nut (22), washer (23) and bolt (20).

f. Disconnect the opposite adjustable pushrod (29) from the idler bellcrank (24) by removing the cotter pin (28), nut (27), washer (26) and bolt (25).

g. Lift the idler bellcrank (24) out of the lower bushing support.

h. If the bushing is to be replaced, mark the position of the lower bushing support, remove the screws and washers, and remove the support from the cabin door shell. Note the number of shims used to ensure proper alignment of the bushing support for reinstallation.

NOTE

The upper bushing support is riveted to the channel (55). The upper bushing may be replaced without removal of the upper bushing support from the channel.

IDLER BELLCRANK INSTALLATION
(Effectivity: All)
(Figure 9)

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

a. If the lower bushing support was removed, position it with the metal shim as marked on the cabin door shell. Secure the bushing support and shim to the door shell with screws and washers.

b. Lubricate the end of the bellcrank (24) and insert it into the lower bushing.

c. Position the channel (55) while aligning the upper bushing with the bellcrank (24). When the end of the bellcrank (24) is properly aligned in both bushings, and the channel holes are aligned, install the four attaching screws and washers (58). Secure the angles (56) to the step panel support (53) with four screws and washers (57).

d. Move the idler bellcrank (24) up and down to determine the amount of bellcrank end play. If bellcrank end play is between .010 and .030 inch, proceed to step e, otherwise shim the support assembly as follows:

1. Remove the eight screws and washers (57 and 58) and remove the channel (55). Install a .030 or .060 inch MIL-S-22499 peelable shim over 100% of the area contacting the two support assemblies (54).

2. Peel the shims as necessary to attain .010 to .030 inch of bellcrank end play.

3. Reinstall the support assembly and verify that bellcrank end play is between .010 and .030 inch before proceding to step e.

e. Verify that the adjustable pushrod (15) measures 10.54 inches from eye-to-eye before installation.

f. Align the pushrod (15) with the idler bellcrank (24) and install the bolt (20) from the door shell side. Secure with washer (23), nut (22) and cotter pin (21).

g. Verify that the opposite adjustable pushrod (29) measures 12.06 inches from eye-to-eye before installation.

h. Align the adjustable pushrod (29) with the idler bellcrank (24) and secure with bolt (25), washer (26), nut (27) and cotter pin (28).

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

i. Actuate the cabin door handle to verify smooth operation of the entire locking mechanism. The excessive force should not be required to engage the latch pins and all seven latch pins must fully retract into the latch-pin housings without any evidence of binding. Adjust the mechanism as necessary.

j. Align the lower step panel (2) and step angle (3) with the step panel support (5) and secure with screws (1 and 4) as shown by Figure 12.

k. Reinstall the upholstery on the cabin door.

l. Refer to Chapter 33-20-00 and install the step light panel.

INTERIOR-HANDLE BELLCRANK REMOVAL (Effectivity: All) (Figure 9)

a. Refer to Chapter 33-20-00 and remove the step light panel from the step below the door handle assembly.

b. Remove the upholstery from the cabin door as necessary to access the step panel attaching hardware.

c. Refer to INTERIOR DOOR HANDLE REMOVAL procedures and remove the handle and bushing supports. Ensure that the bushing supports are marked prior to removal and note the locations of all metal shims to ensure proper mechanism alignment during reinstallation.

d. Remove the bolts (41 and 43), washers (42, 44 and 45) and nuts (46) then remove the shaft assembly (40) from the bellcrank (47).

NOTE

Refer to Chapter 20-00-05 for information on replacement of the bushings.

NOTE

The nonadjustable pushrods of the interior handle bellcrank are not shown to simplify the illustration in Figure 9. The nonadjustable pushrods shown by Figure 8 provide for a typical installation.

e. Disconnect the center latch-pin pushrods from the interior handle bellcrank (47) by removing the cotter pins (24 and 36, Figure 8), washers (25 and 37, Figure 8) and pins (34 and 38, Figure 8).

f. Disconnect the adjustable pushrod (29) from the bellcrank (47) by removing the cotter pin (31), nut (30), washer (32) and bolt (33).

g. Disconnect the opposite adjustable pushrod (10) from the interior handle bellcrank (47) by removing the cotter pin (14), nut (13), washer (12) and bolt (11).

h. Lift the bellcrank (47) out of the bushing support.

NOTE

If the bushing requires replacement, mark the position of the lower bushing support, then remove the four screws and washers to remove the support and metal shim from the cabin door shell. Refer to Chapter 20-00-05 for information on bushing replacement.

INTERIOR-HANDLE BELLCRANK INSTALLATION (Effectivity: All) (Figure 9)

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

a. If the lower bushing support was removed, align it and the metal shim in the proper position on the cabin door shell as marked, then secure with screws and washers.

b. Lubricate the end of the bellcrank (47) and insert it into the the bushing.

c. Insert the shaft assembly (40) into the bellcrank (47). Align the mounting holes and secure with bolts (41 and 43), washers (42, 44 and 45) and nuts (46).

d. Refer to INTERIOR DOOR HANDLE INSTALLATION and reinstall the interior handle bushing supports and handle assembly.

e. Move the interior handle bellcrank (47) up and down to determine the amount of bellcrank end play. If bellcrank end play is between .010 and .030 inch, proceed to step f, otherwise shim the lower bushing support as follows:

NOTE

Bellcrank end play may also be adjusted by shimming the bushing support at the top of the interior door handle as outlined under INTERIOR DOOR HANDLE INSTALLATION.

1. Remove the applicable bushing support and install a .030- or .060-inch MIL-S-22499 peelable shim over 100% of the bushing support where it contacts the door shell.

2. Peel the shim as necessary to attain .010 to .030 inch of bellcrank end play.

3. Reinstall the bushing support, bellcrank and handle assembly, then verify that bellcrank end play is between .010 and .030 inch before reconnecting the pushrods.

f. Verify that the adjustable pushrod (10) measures 16.57 inches from eye-to-eye before installation.

g. Align the adjustable pushrod (10) with the bellcrank (47) and secure with bolt (11), washer (12), nut (13) and cotter pin (14).

h. Verify that the opposite adjustable pushrod (29) measures 12.06 inches from eye-to-eye before installation.

i. Align the adjustable pushrod (29) with the bellcrank (47) and secure with bolt (33), washer (32), nut (30) and cotter pin (31).

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

j. Actuate the cabin door handle to verify smooth operation of the entire locking mechanism. Excessive force should not be required to engage the latch pins and all seven latch pins must fully retract into the latch-pin housings without any evidence of binding. Adjust the mechanism as necessary.

k. Reinstall the cabin door upholstery.

l. Refer to Chapter 33-20-00 and install and step light panels that were removed.

CENTER LATCH PIN REMOVAL (Effectivity: All) (Figure 9)

a. Refer to Chapter 33-20-00 and remove the step light panel from the step below the door handle assembly.

b. Remove the upholstery from the cabin door as necessary to access the step panel attaching hardware.

NOTE

The nonadjustable pushrods of the interior handle bellcrank and the latch pins and housing assemblies connected to these pushrods are not shown to simplify the illustration in Figure 9. The above items shown in Figure 8 provide for a typical installation.

c. Disconnect the center latch-pin pushrods from the interior handle bellcrank (47) by removing the cotter pins (24 and 36, Figure 8), washers (25 and 37, Figure 8) and pins (34 and 38, Figure 8).

d. Pull the latch pins (9 and 30, Figure 8) inward until access to the attaching hardware is obtained.

e. Disconnect the center latch-pin pushrods (7 and 33, Figure 8) from the latch pins (9 and 30, Figure 8)

by removing the cotter pins (12 or 29, Figure 8), washers (11 or 28, Figure 8) and pins (8 or 32, Figure 8) from the latch pins (9 or 30, Figure 8). Slide the latch pins (9 or 30, Figure 8) out of the respective latch-pin housing assemblies (10 or 31, Figure 8).

CENTER LATCH PIN INSTALLATION (Effectivity: All) (Figure 9)

NOTE

The nonadjustable pushrods of the interior handle bellcrank and the latch pins and housing assemblies connected to these pushrods are not shown to simplify the illustration in Figure 9. The above items shown in Figure 8 provide for a typical installation.

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

a. Lubricate the sliding surface of the latch pin (9 or 30, Figure 8) with dry lubricant (8, Chart 1, 52-00-00), insert them into the respective housing assemblies (10 or 31, Figure 8).

b. Align the pushrod (7, Figure 8) with the interior handle bellcrank (47) and install the pin (38, Figure 8), washer (37, Figure 8) and cotter pin (36, Figure 8).

c. Align the pushrod (33, Figure 8) with the interior handle bellcrank (47) and install the pin (34, Figure 8), washer (25, Figure 8) and cotter pin (24, Figure 8).

d. Connect the center latch pin pushrods connected to the interior handle bellcrank (47) by positioning the pushrods on the latch pin (9 or 30, Figure 8) and installing the pin (8 or 32, Figure 8), washer (11 or 28, Figure 8) and cotter pin (12 or 29, Figure 8).

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

e. Actuate the cabin door handle to verify smooth operation of the entire locking mechanism. Excessive force should not be required to engage the latch pins and all seven latch pins must fully retract into the latch-pin housings without any evidence of binding. Adjust the mechanism as necessary.

f. Reinstall the cabin door upholstery.

g. Refer to Chapter 33-20-00 and install and step light panels that were removed.

EXTERIOR-HANDLE BELLCRANK REMOVAL (Effectivity: All) (Figure 9)

a. Refer to Chapter 33-20-00 and remove the upper step light panel from the upper step to allow upholstery removal.

b. Remove the upholstery from the cabin door as necessary to access the step panel attaching hardware.

c. Refer to CABIN DOOR STEP PANEL REMOVAL and remove the upper step panel from the cabin door to access the support assembly (48 and 51) of the exterior handle bellcrank (3).

d. Remove the four screws and washers (49) attaching the angles (50) to the step panel support (53).

NOTE

The wire shielding terminal is installed with the wire harness stand-off clamp attaching hardware.

e. Remove the two screws and washers (52) attaching the wire harness stand-off clamps to the channel (51).

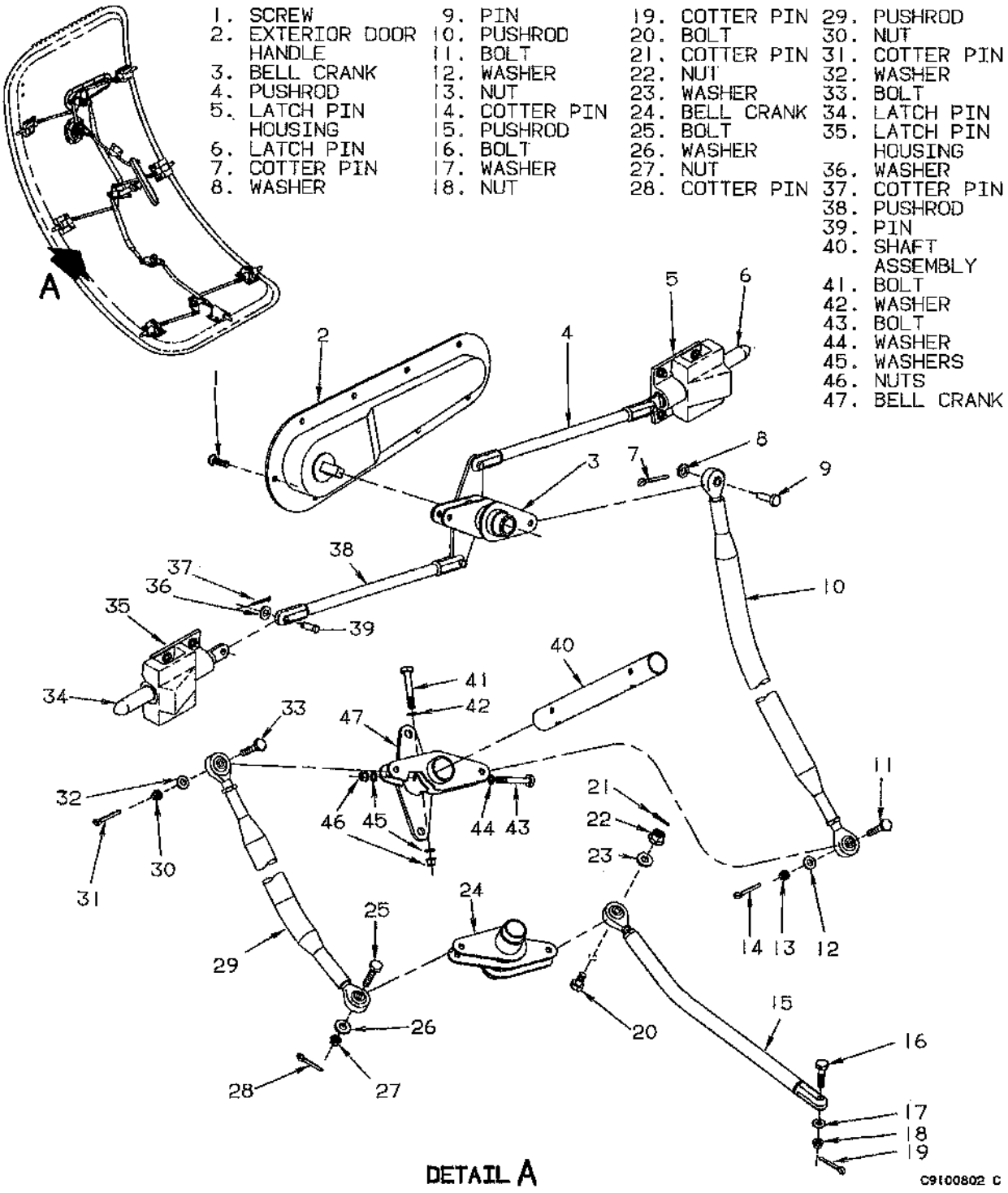
f. Remove the two remaining screws and washers (52) securing the channel (51) to the support assemblies (48) then pull the channel upward to separate the bushing from the exterior handle bellcrank (3).

NOTE

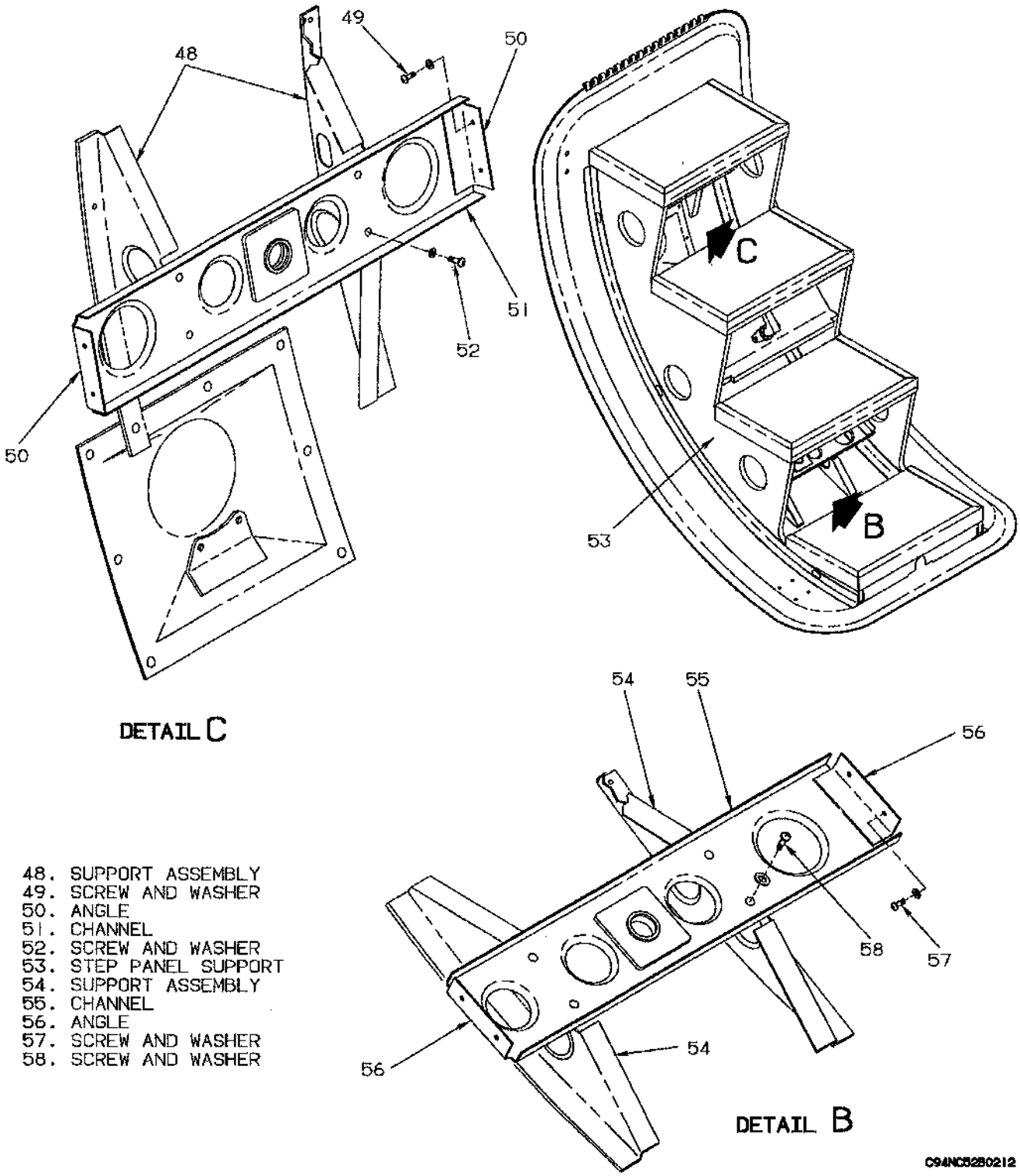
MIL-S-22499 peelable shims are installed between the channel (51) and the two support brackets (48). Record shim thickness or do not remove the shim material to provide proper mechanism alignment during reinstallation.

g. Refer to Figure 10 for an illustration of the air pressure lock assembly linkage and the index numbers included within this step. Remove the cotter pin (3), washer (2) and pin (5) to disconnect the link assembly (4) from the bellcrank (3, Figure 9). Push the lock assembly release button (15) and rotate the link assembly (4) out of the bellcrank (3, Figure 9).

h. Remove the cotter pin (7), washer (8) and pin (9) to disconnect the adjustable pushrod (10) from the bellcrank (3).



Interior and Exterior Handle Bellcranks and Latch Pin Installation (Effectivity: All) (Sheet 1 of 2)
Figure 9



48. SUPPORT ASSEMBLY
49. SCREW AND WASHER
50. ANGLE
51. CHANNEL
52. SCREW AND WASHER
53. STEP PANEL SUPPORT
54. SUPPORT ASSEMBLY
55. CHANNEL
56. ANGLE
57. SCREW AND WASHER
58. SCREW AND WASHER

Interior and Exterior Handle Bellcranks and Latch Pin Installation (Effectivity: All) (Sheet 2 of 2)
Figure 9

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i. Disconnect the nonadjustable pushrods (4 and 38) from the bellcrank (3) by removing the cotter pins, washers and pins at the bellcrank attach points (not shown). Remove the bellcrank (3) from the exterior door handle (2) drive shaft.

NOTE

The upper bushing support is riveted to the bellcrank support assembly. Bushing replacement may be accomplished without removing the upper bushing support from the support assembly. Refer to Chapter 20-00-05 for information on bushing replacement.

EXTERIOR-HANDLE BELLCRANK INSTALLATION (Effectivity: All) (Figure 9)

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

a. After lubricating the end of the bellcrank (3), install it in the proper position on the exterior door handle (2) drive shaft.

b. Reconnect the nonadjustable pushrods (4 and 38) to the bell crank (3) by installing the pins, washers and cotter pins (not shown).

c. Verify that the adjustable pushrod (10) measures 16.57 inches from eye-to-eye before reinstallation.

d. Align the adjustable pushrod (10) with the bellcrank (3) and install the pin (9), washer (8) and cotter pin (7).

e. Lubricate the end of the bellcrank (3) and position the channel (51) with the end of the bellcrank (3). When the end of the bellcrank (3) is properly inserted into the bushing and the channel holes are aligned, install the four screws and washers (52) into the support assemblies (50). Install the four screws and washers (49) through the angles (50) into the step panel support (53).

f. Move the bellcrank (3) up and down to determine the amount of bellcrank end play. If bellcrank end play is between .010 and .030 inch, proceed to step g, otherwise shim the entire support assembly as follows:

1. Remove the eight screws and washers (49 and 52) and the channel (51) from the support assemblies (48). Install a .030 or .060 inch MIL-S-22499 peelable shim over 100% of the area contacting the two support assemblies.

2. Peel the shims as necessary to attain .010 to .030 inch of bellcrank end play.

3. Reinstall the support assembly and verify that bellcrank end play is between .010 and .030 inch.

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

g. Actuate the cabin door handle to verify smooth operation of the entire locking mechanism. Excessive force should not be required to engage the latch pins and all seven latch pins must fully retract into the latch-pin housings without any evidence of binding. Adjust the mechanism as necessary.

NOTE

The wire harness shielding termination is secured with the stand-off clamp attaching hardware.

h. Align the wire harness over the support assembly and secure with the two stand-off clamps by installing two screws and washers (52). Verify that the wire shielding terminal is electrically bonded as specified by procedures in Chapter 20-00-03.

i. Refer to CABIN DOOR STEP PANEL INSTALLATION and reinstall the upper step panel.

j. Reinstall the upholstery on the cabin door.

k. Refer to Chapter 33-20-00 and install the upper step light panel.

BOTTOM LATCH PINS REMOVAL (Effectivity: All) (Figure 9)

a. Refer to Chapter 33-20-00 and remove the upper step light panel from the upper step to allow upholstery removal.

b. Remove the upholstery from the cabin door as necessary to access the step panel attaching hardware.

c. Disconnect the nonadjustable pushrods (4 and 38) from the bellcrank (3) by removing the cotter pins, washers and pins at the bellcrank attach points (not shown).

d. Pull the latch pins (6 and 34) inward until access to the attaching hardware is obtained.

e. Disconnect the pushrod (38) from the latch pin (34) by removing the cotter pin (37), washer (36) and pin (39), then slide the latch pin (34) out of the latch-pin housing (35). Disconnect the opposite nonadjustable pushrod (4) from the latch pin (6) in the same manner by removing the cotter pin, washer and pin, then slide the latch pin (6) out of the latch pin housing (5).

BOTTOM LATCH PINS INSTALLATION (Effectivity: All) (Figure 9)

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

a. Lubricate the latch pins (6 and 34) with dry lubricant (8, Chart 1, 52-00-00) and insert them into their respective latch-pin housings (5 and 35). Reconnect the pushrod (38) to the latch pin (34) by installing pin (39), washer (36) and cotter pin (37). Reconnect the opposite pushrod (4) to the latch pin (6) in the same manner by installing the pin, washer and cotter pin.

b. Connect the nonadjustable pushrods (4 and 38) to the bellcrank (3) by installing the cotter pins, washers and pins (not shown) at the bellcrank attach points.

NOTE

This check must be performed while the airplane is supported by the landing gear and should be performed anytime the door latch mechanism assembly has been adjusted.

c. Actuate the cabin door handle to verify smooth operation of the entire locking mechanism. Excessive force should not be required to engage the latch pins and all seven latch pins must fully retract into the latch-pin housings without any evidence of binding. Adjust the mechanism as necessary.

d. Refer to CABIN DOOR STEP PANEL INSTALLATION and reinstall the upper step panel.

e. Reinstall the upholstery on the cabin door.

f. Refer to Chapter 33-20-00 and install the upper step light panel.

AIR PRESSURE LOCK AND LINK ASSEMBLY REMOVAL (Effectivity: All) (Figure 10)

a. Refer to Chapter 33-20-00 and remove the upper and middle step light panels from the cabin door steps to allow upholstery removal.

b. Remove the upholstery from the cabin door as necessary to access the step panel attaching hardware.

c. Refer to CABIN DOOR STEP PANEL REMOVAL and remove the top and middle step panels from the cabin door to access the air pressure lock and linkage assembly.

d. Remove the cotter pin (18), washer (17) and pin (10) to disconnect the tube assembly (14) and springs (6 and 23) from the end of the link (25). Unhook the springs (6 and 23) from the holes in the air pressure lock assembly (1).

e. Remove the cotter pin (11), nut (12), washer (13) and bolt (16) to disconnect the tube assembly (14) from the bracket located on the step panel. Remove the interior lock-release button (15) from the tube assembly (14) as necessary.

f. Disconnect the bracket holding the handle-position sense switches to the air pressure lock.

g. Remove the screws, washers and nuts securing the handle-position sense switches to the switch bracket. Ensure that the sense switches are marked to allow reinstallation in their original position.

h. If it is necessary to remove the clip (24), mark the clip location on the link (25) to ensure proper sense switch actuation when the clip is reinstalled. Remove the nuts (19), washers (7 and 20) and screws (9) securing the clip (24) to the link (25).

i. Remove the cotter pin (21), washer (22) and pin (8) to disconnect the link (25) from the slotted bolt (30) on the air pressure lock assembly (1).

j. Remove the cotter pin (3), washer (2) and pin (5). Press the lock-release button (15) and rotate the link assembly (4) away from the exterior handle bellcrank.

NOTE

Follow the remaining steps to remove the air pressure lock assembly (1) from the pan assembly.

k. Unscrew the exterior lock-release button from the shafted nut (27).

NOTE

The air pressure lock assembly may be assembled with the screws (32) installed from the outboard side of the lock assembly. Disassembly may require that a 90° phillips head screwdriver to access the screw heads due to minimal clearance between the cover (39) and pan assembly.

l. Remove the eight nuts (26), washers (40) and screws (32) securing the two covers (34 and 39). Pull the covers apart without disturbing internal adjustment of pressure lock components.

m. Use an 1/8-inch drill bit to drill the rivets out securing the cover (39) to the pan assembly, then remove it from the door.

NOTE

Complete removal of the link assembly (4) requires removal of the slotted bolt (30) from the air pressure lock assembly (1).

AIR PRESSURE LOCK AND LINK ASSEMBLY INSTALLATION (Effectivity: All) (Figure 10)

If removed, reinstall the air pressure lock assembly as follows:

a. Clean any existing sealant from the door shell pan and pressure lock assembly cover (39) with solvent (1, Chart 1, 52-00-00).

b. Apply a .125-inch bead of sealant (9, Chart 1, 52-00-00) in a 2-inch-diameter circle on the base of the lock assembly cover (39).

c. Align the base of the lock assembly cover (39) with the rivet holes in the pan assembly and secure with four NAS1739B4-2, 1/8-inch rivets.

NOTE

Ensure that the air pressure lock assembly is not disturbed or altered from the original adjustment.

d. Align the cover (34) mounting holes with the opposite cover (39) and secure with screws (32), washers (40) and nuts (26) as originally installed.

e. Reinstall the exterior lock-release button on the shafted nut (27), using retaining compound (7, Chart 1, 52-00-00) to secure in place.

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

f. Depress the lock-release button (15), then rotate the link assembly (4) into proper alignment with the attach point on the exterior handle bellcrank (3 of Figure 9). Reconnect the exterior handle bellcrank to the link assembly (4) by installing the pin (5), washer (2) and cotter pin (3).

g. Align the link (25) with the slotted bolt (30) on the air pressure lock assembly (1), then install the pin (8) and washer (22). Trim the cotter pin (21) to .25 inch before installation to prevent interference with the linkage. Install the cotter pin (21) and push the lock-release button (15) to verify that there is no interference with the air pressure lock assembly and associated linkage.

NOTE

The link (25) and slotted bolt (30) must be angled at approximately 30° to ensure proper alignment with the tube assembly (14) during reinstallation.

h. If the clip (24) was removed from the link (25), align it in the original position as marked during removal, then secure with screws (9), washers (7 and 20) and nuts (19).

i. Install the handle-position sense switches in their original positions on the bracket assembly, then secure with screws, washers and nuts.

j. Position the middle step panel over the step panel supports and align the tube assembly (14) with the bracket. Secure the tube assembly (14) by installing the bolt (16), washer (13), nut (12) and cotter pin (11).

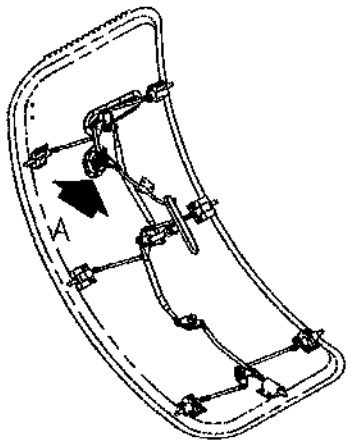
k. Hook the springs (6 and 23) into the holes on the air pressure lock assembly (1).

l. Reconnect the link (25) and springs (6 and 23) to the tube assembly (14) by installing the pin (10) through the spring (6), link (25) and spring (23), then install the washer (17) and cotter pin (18).

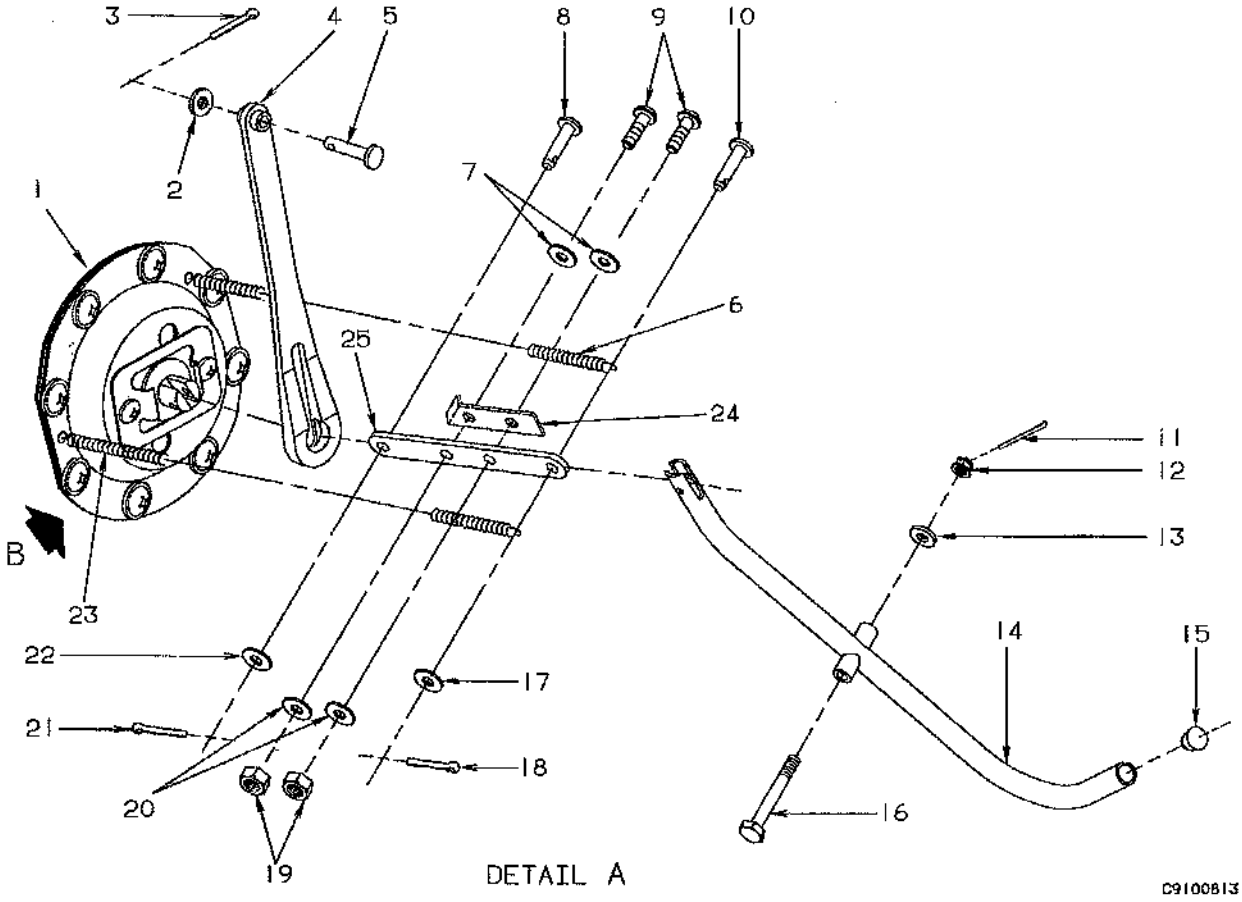
m. Reinstall the interior lock release button (15) on the tube assembly (14) and secure with retaining compound (7, Chart 1, 52-00-00).

n. Perform the HANDLE POSITION SENSE SWITCH ADJUSTEMNT procedures in Chapter 52-70-00.

o. Refer to CABIN DOOR STEP PANEL INSTALLATION and reinstall the step panels as required.

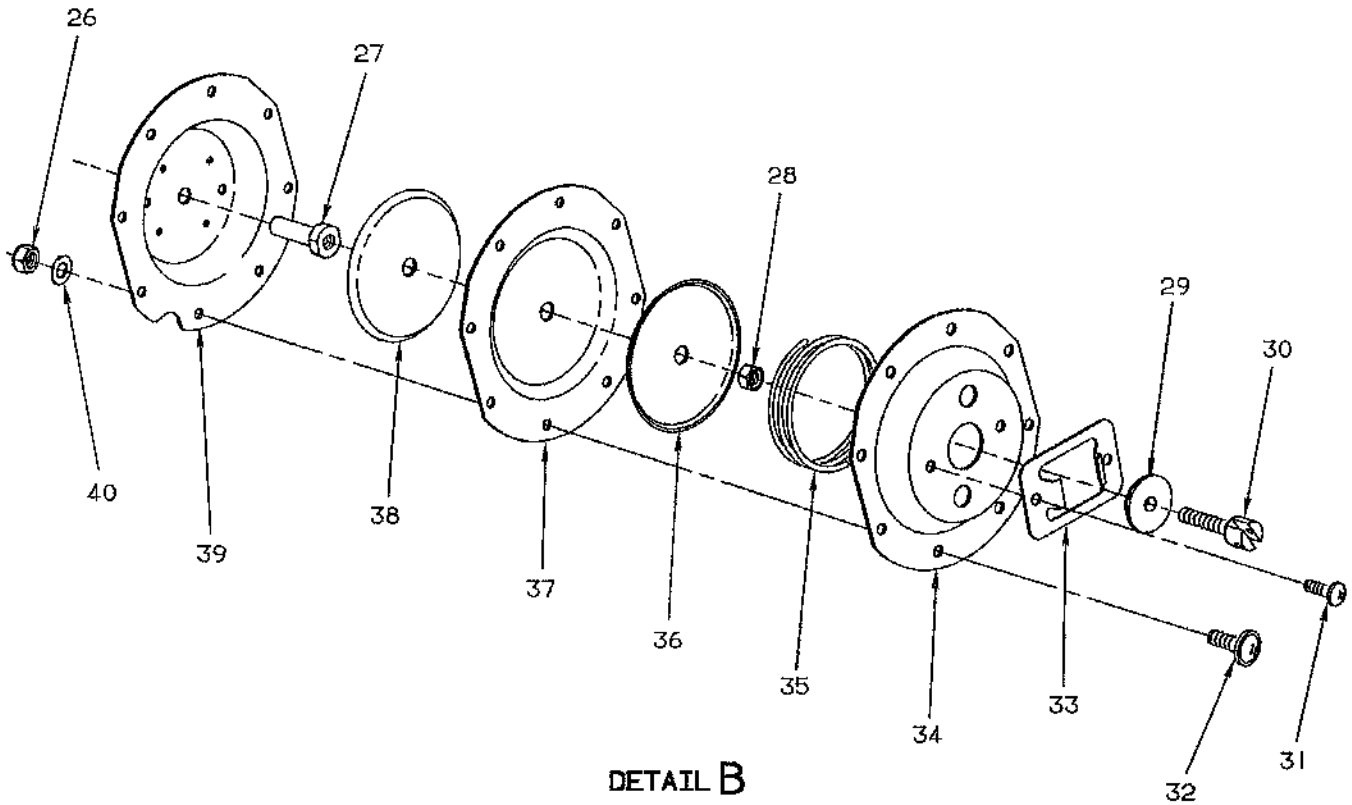


1. AIR PRESSURE LOCK ASSEMBLY
2. WASHER
3. COTTER PIN
4. LINK ASSEMBLY
5. PIN
6. SPRING
7. WASHERS
8. PIN
9. SCREWS
10. PIN
11. COTTER PIN
12. NUT
13. WASHER
14. TUBE ASSEMBLY
15. BUTTON
16. BOLT
17. WASHER
18. COTTER PIN
19. NUTS
20. WASHERS
21. COTTER PIN
22. WASHER
23. SPRING
24. CLIP
25. LINK



Air Pressure Lock and Link Assembly Installation (Effectivity: All)
(Sheet 1 of 2)
Figure 10

26.	NUT	34.	COVER
27.	SHAFTED NUT	35.	SPRING
28.	NUT	36.	SUPPORT
29.	BUSHING	37.	SEAL
30.	BOLT	38.	SUPPORT
31.	SCREW	39.	COVER
32.	SCREW	40.	WASHER
33.	GUIDE		



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Air Pressure Lock and Link Assembly
Installation (Effectivity: All)
(Sheet 2 of 2)
Figure 10

p. Reinstall all cabin door upholstery that was removed.

q. Refer to Chapter 33-20-00 and install the step light panels on the upper and middle cabin door steps.

r. Actuate the cabin door handle to verify smooth operation of the entire locking mechanism. Excessive force should not be required to engage the latch pins and all seven latch pins must fully retract into the latch-pin housings without any evidence of binding. Adjust the mechanism as necessary and verify that the pressure-lock-release assembly linkage moves freely and performs its locking operation properly.

INTERIOR DOOR HANDLE REMOVAL (Effectivity: All) (Figure 11)

a. Refer to Chapter 33-20-00 and remove the step light panel from the step panel below the door handle assembly.

b. Remove the upholstery from the cabin door as necessary to access the step panel attaching hardware.

c. Remove the nuts (1 and 14), washers (15 and 18), washers (2 and 16) and bolts (3 and 13) securing the interior door handle (8) to the shaft assembly (17). Slide the door handle out of the bushing supports (5 and 6).

d. For bushing replacement, remove the nuts (11 and 12) and screws (9 and 10) securing the bushing supports (5 and 6) to the step panel support.

e. Remove the bushing supports (5 and 6) and any existing metal shims from the step panel. Note the exact placement of shims to ensure proper bellcrank end play after reinstallation.

NOTE

Refer to Chapter 20-00-05 for information on bushing (4 and 7) replacement.

INTERIOR DOOR HANDLE INSTALLATION (Effectivity: All) (Figure 11)

NOTE

Lubricate all rotating parts with a light film of oil (5, Chart 1, 52-00-00) before assembly.

a. If the bushing supports (5 and 6) were removed, align them and any metal shims in their respective positions on the step panel, then secure with screws (9 and 10) and nuts (11 and 12).

b. Slide the interior door handle (8) into the handle shaft (17) and secure with bolts (3 and 13), washers (2 and 16), washers (15 and 18) and nuts (1 and 14).

c. Reinstall the cabin door upholstery.

d. Refer to Chapter 33-20-00 and install the middle step light panel.

CABIN ENTRANCE DOOR STEP PANEL REMOVAL (Effectivity: All) (Figure 12)

a. Remove the screws (1 and 4) from the step angles (3).

b. Remove the desired step panel (2) from the step panel supports (5).

CABIN ENTRANCE DOOR STEP PANEL INSTALLATION (Effectivity: All) (Figure 12)

a. Align the step panel (2) with the cabin door.

b. Install the screws (1 and 4) to secure the step angles (3) and step panel (2) to the cabin door step panel supports (5).

CABIN ENTRANCE DOOR INSTRUCTION PLACARD REMOVAL (Effectivity: All)

a. Remove the four placard attaching screws and remove the placard from the door.

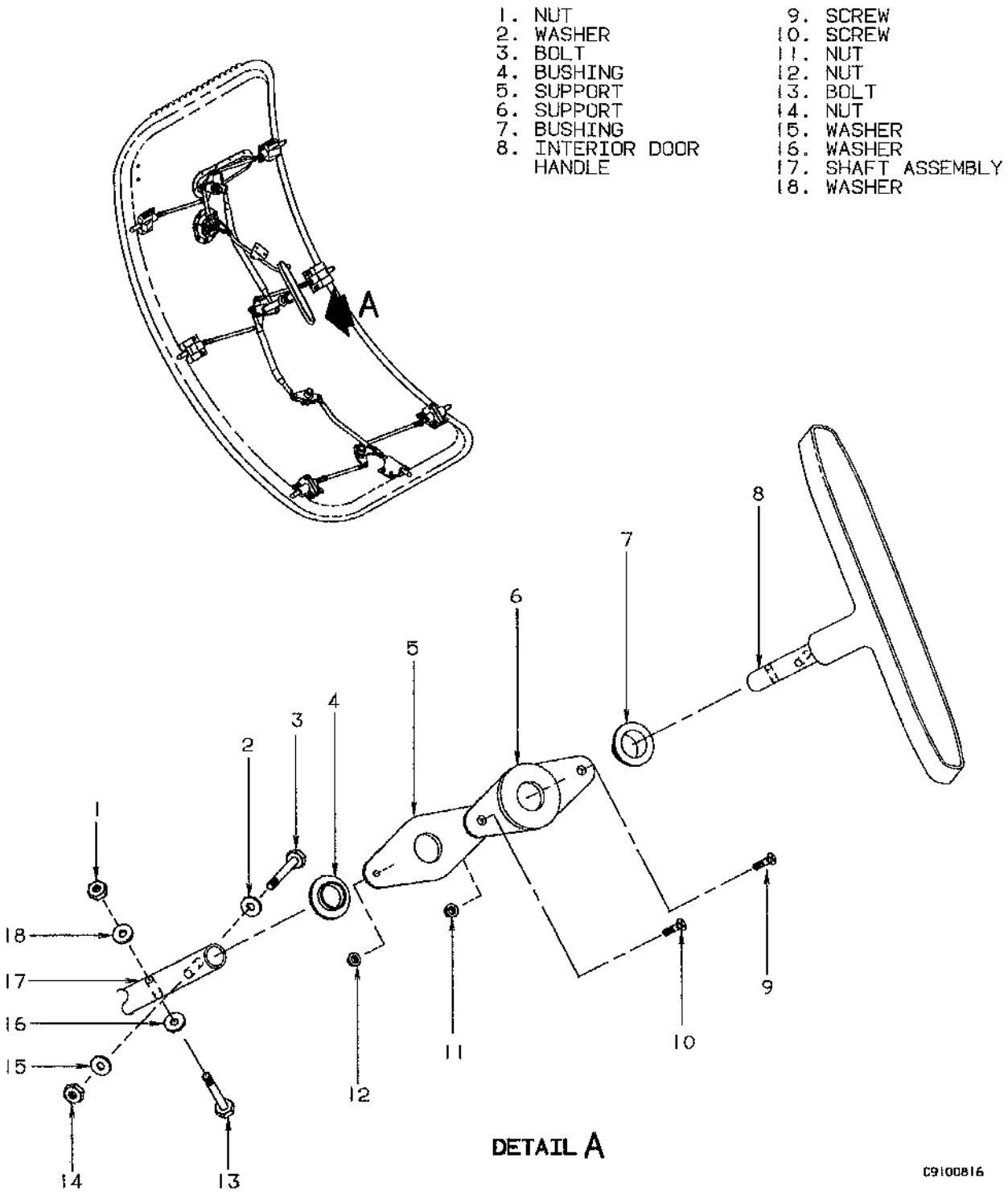
CABIN ENTRANCE DOOR INSTRUCTION PLACARD INSTALLATION (Effectivity: All)

a. Align the placard with the inserts in the cabin door structure and install the four attaching screws.

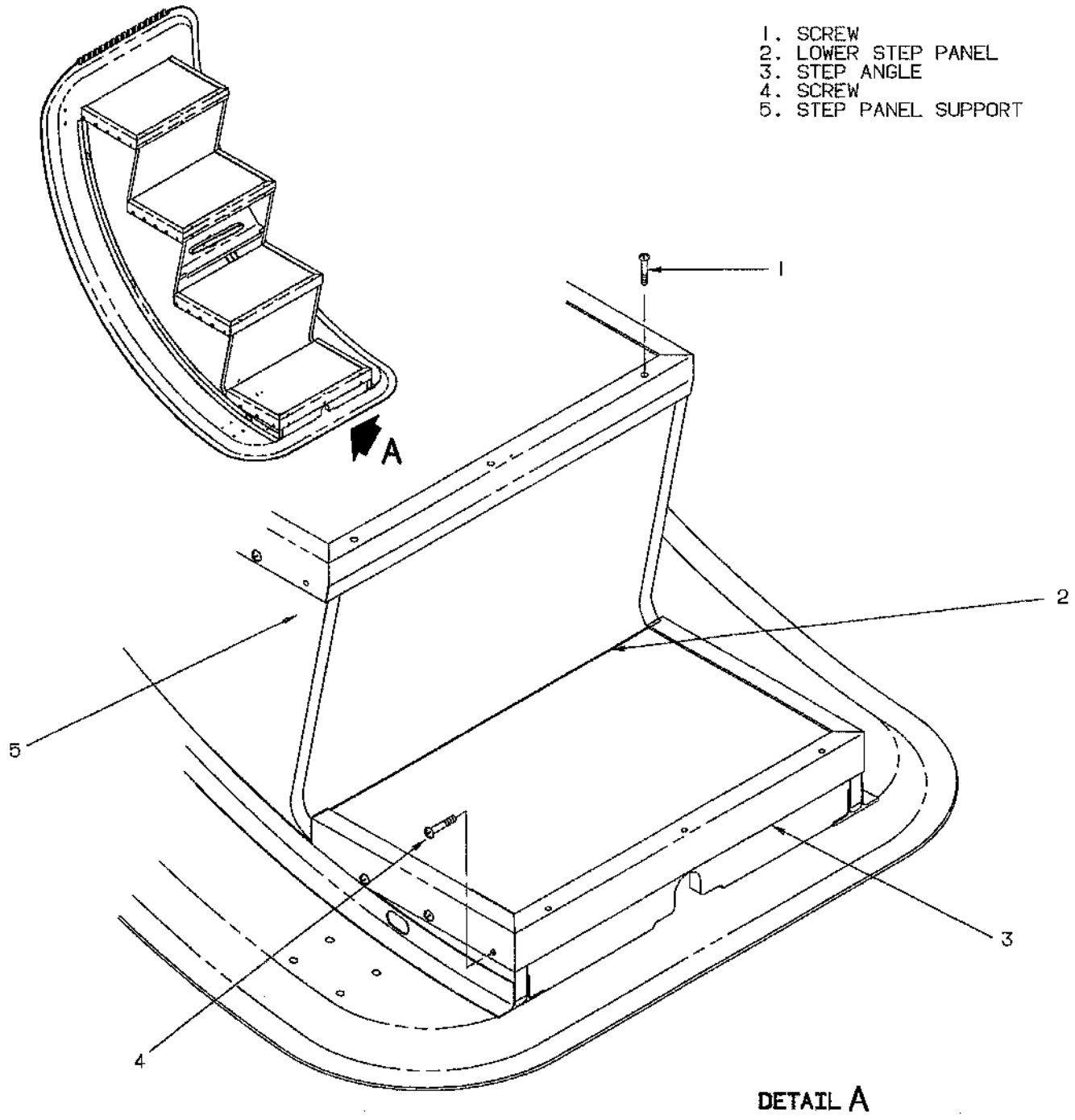
CABIN ENTRANCE DOOR STEP LIGHTS POWER SUPPLY REMOVAL (Effectivity: All) (Figure 13)

a. Remove all electrical power from the airplane and pull the floodlight circuit breaker located on the battery bus circuit breaker panel.

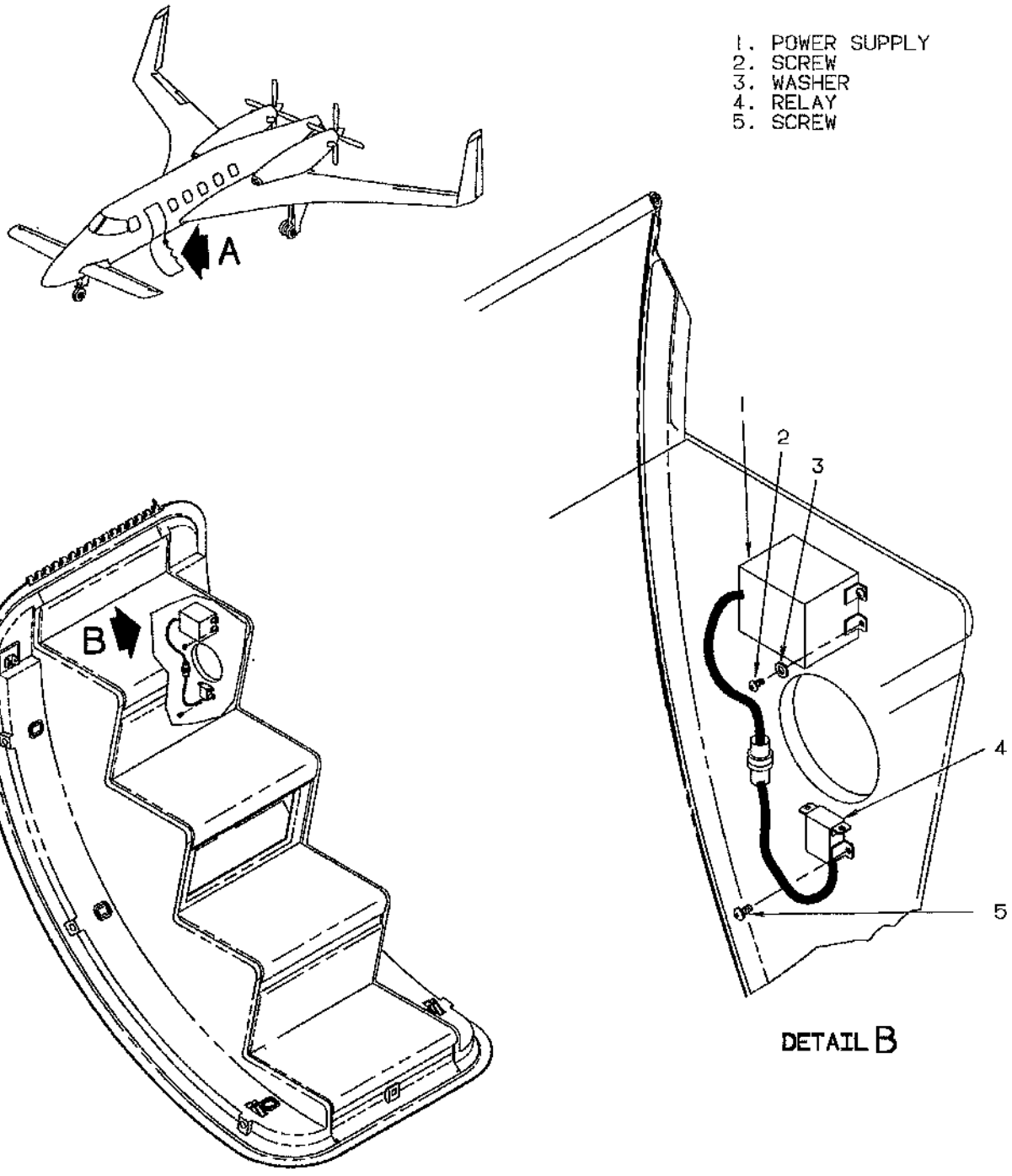
b. Refer to Chapter 33-20-00 and remove the step light panel from the top step.



Interior Door Handle Installation (Effectivity: All)
Figure 11



Cabin Entrance Door Step Panel Installation (Effectivity: All)
Figure 12



Step Lights Panel Power Supply and Relay Installation (Effectivity: All)
Figure 13

c. Remove the carpet and any other upholstery preventing removal of the top step.

d. Refer to CABIN ENTRANCE DOOR STEP PANEL REMOVAL and remove the desired step panel from the door assembly.

e. Tag and disconnect the electrical connector from the power supply (1).

f. Remove screws (2) and washers (3), then remove the power supply (1) from the airplane.

CABIN ENTRANCE DOOR STEP LIGHTS POWER SUPPLY INSTALLATION (Effectivity: All) (Figure 13)

a. Align the power supply (1) with the inserts in the cabin door structure and secure with screws (2) and washers (3).

b. Reconnect the electrical connector to the power supply according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

c. Refer to CABIN ENTRANCE DOOR STEP PANEL INSTALLATION and install the applicable step panel.

d. Reinstall the carpet and any other upholstery that was removed.

e. Refer to Chapter 33-20-00 and install the step light panel.

f. Close the floodlight circuit breaker and restore electrical power to the airplane.

CABIN ENTRANCE DOOR STEP LIGHTS RELAY REMOVAL (Effectivity: All) (Figure 13)

a. Remove all electrical power from the airplane and pull the floodlight circuit breaker on the battery bus circuit breaker panel.

b. Refer to Chapter 33-20-00 and remove the step light panel from the top step.

c. Remove the carpet and any other upholstery preventing removal of the top step.

d. Refer to CABIN ENTRANCE DOOR STEP PANEL REMOVAL and remove the desired step to gain access to the relay.

e. Tag and disconnect the electrical wiring from the relay (4).

f. Remove the relay attaching screws (5) and remove the relay (4) from the cabin door structure.

CABIN ENTRANCE DOOR STEP LIGHTS RELAY INSTALLATION (Effectivity: All) (Figure 13)

a. Align the relay (4) with the inserts in the cabin door structure and secure with the attaching screws (5).

b. Reconnect electrical wiring to the relay (4) according to tags or the BEECHCRAFT Starship 1 Wiring Diagram Manual.

c. Refer to CABIN ENTRANCE DOOR STEP PANEL INSTALLATION and install the attaching screws.

d. Reinstall the carpet and any other upholstery that may have been removed.

e. Refer to Chapter 33-20-00 and install the step light panel.

f. Close the floodlight circuit breaker and restore electrical power to the airplane.

EMERGENCY EXIT - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The inward opening, plug type, emergency exit door is located on the right side of the airplane near the aft end of the cabin. The door provides access to the forward portion of the right wing. The door is opened by pulling down on the handle which is marked EXIT - PULL (self-luminous). The door is secured at the bottom with a nonconnecting hinge and with two latch pins at the top of the door.

EMERGENCY EXIT - MAINTENANCE PRACTICES (Effectivity: All)

EMERGENCY EXIT DOOR REMOVAL (Effectivity: All)

CAUTION

Protect the emergency exit door windows from damage when working on or around the door.

a. Pull the EXIT-PULL handle.

b. Lower the door and then raise it from the nonconnecting hinge and pull inboard. Place clear of the work area.

EMERGENCY EXIT DOOR INSTALLATION (Effectivity: All)

CAUTION

Protect the emergency exit door windows from damage when working on or around the door.

a. Center the emergency exit door in its frame.

b. Lower the bottom of the door into the nonconnecting hinges.

c. Rotate the door upward.

d. Pull the EXIT-PULL handle to retract the latch pins, then rotate the door up and outboard.

e. When the door seats, push outboard on the EXIT-PULL handle until it snaps into the locked position.

f. If the door does not fit flush with the cabin exterior, perform the SHIM INSTALLATION procedure in this chapter.

EMERGENCY EXIT DOOR SEAL REMOVAL (Effectivity: All) (Figure 1)

a. Remove the emergency exit door in accordance with the EMERGENCY EXIT DOOR REMOVAL procedures in this chapter.

b. Use a plastic scraper to remove the door seal from the door.

c. Use cleaner (10, Chart 1, 52-00-00) to remove any old adhesive or seal residue from the door edge.

EMERGENCY EXIT DOOR SEAL INSTALLATION (Effectivity: All) (Figure 1)

a. Use cleaner (10, Chart 1, 52-00-00) to clean the seal and the door edge.

b. Mark a line 0.39 inches inboard from the outer edge of the door on the top and sides.

c. Mark a line 0.57 inches above the outer edge of the door on the bottom.

d. Position the seal on the door to check for alignment with marks.

e. Using the line as a centerline, apply adhesive (2, Chart 1, 52-00-00) in a band approximately 0.25 inches wide on either side of the line (the width of the seal).

f. Apply adhesive (2, Chart 1, 52-00-00) to the bottom of the door seal.

g. Allow the adhesive to dry until tacky.

NOTE

Start installing the seal at the bottom of the door with one free end at the bottom center of the door.

h. Install the seal on the door with the cut ends at the bottom center of the door. Join the cut ends with adhesive (2, Chart 1, 52-00-00).

i. Allow the seal to cure in accordance with the manufacturer's specifications.

j. Use cleaner (10, Chart 1, 52-00-00) to remove any excess adhesive from the door seal edges.

k. Install the emergency exit door in accordance with the EMERGENCY EXIT DOOR INSTALLATION procedures in this chapter.

EMERGENCY EXIT DOOR MARKER REMOVAL (Effectivity: All) (Figure 2)

a. Remove the emergency exit door in accordance with the EMERGENCY EXIT DOOR REMOVAL procedures in this chapter.

b. Pull the handle assembly (14) inboard to gain access to the back of the handle.

c. Remove the two screws (13) and washers (12) securing the marker (10) and grip (11) to the handle assembly (14).

EMERGENCY EXIT DOOR MARKER INSTALLATION (Effectivity: All) (Figure 2)

a. Pull the handle assembly (14) inboard to gain access to the back of the handle.

b. Position the marker (10) and grip (11) on the handle assembly (14) and install the two screws (13) and washers (12).

c. Install the emergency exit door in accordance with the EMERGENCY EXIT DOOR INSTALLATION procedures in this chapter.

HANDLE ASSEMBLY REMOVAL (Effectivity: All) (Figure 2)

a. Remove the emergency exit door in accordance with the EMERGENCY EXIT DOOR REMOVAL procedures in this chapter.

b. Remove the door upholstery in accordance with the EMERGENCY EXIT DOOR UPHOLSTERY REMOVAL procedures in Chapter 25-20-00.

c. Remove the cotter pin (20), washer (21) and pin (22) securing the link (23) to the handle (14).

d. Remove the two cotter pins (18), washers (17) and pins (8) securing the interlock links (7) to the arm assemblies (9).

e. Remove the four screws (15) securing the brackets (16) to the door.

HANDLE ASSEMBLY INSTALLATION (Effectivity: All) (Figure 2)

a. Position the handle assembly (14) on the door and install the four screws (15) through the brackets (16).

b. Secure the two interlock links (7) to the arm assemblies (9) with the pins (8), washers (17) and cotter pins (18).

c. Secure the link (23) to the handle (14) with the pin (22), washer (21) and cotter pin (20).

d. Install the door upholstery in accordance with the EMERGENCY EXIT DOOR UPHOLSTERY INSTALLATION procedures in Chapter 25-20-00.

e. Install the emergency exit door in accordance with the EMERGENCY EXIT DOOR INSTALLATION procedures in this chapter.

LATCH PIN REMOVAL (Effectivity: All) (Figure 2)

a. Remove the emergency exit door in accordance with the EMERGENCY EXIT DOOR REMOVAL procedures in this chapter.

b. Remove the door upholstery in accordance with the EMERGENCY EXIT DOOR UPHOLSTERY REMOVAL procedures in Chapter 25-20-00.

c. Remove the cotter pin (5), washer (4) and pin (19) securing the latch pin (2) to the interlock link (7).

d. Remove the latch pin (2) from the guide (3).

LATCH PIN INSTALLATION (Effectivity: All) (Figure 2)

a. Position the latch pin (2) in the guide (3).

b. Secure the interlock link (7) to the latch pin (2) with the pins (19), washers (4) and cotter pins (5).

c. Install the door upholstery in accordance with the EMERGENCY EXIT DOOR UPHOLSTERY INSTALLATION procedures in Chapter 25-20-00.

d. Install the emergency exit door in accordance with the EMERGENCY EXIT DOOR INSTALLATION procedures in this chapter.

SHIM INSTALLATION (Effectivity: All) (Figure 2)

NOTE

A maximum of two shims (1) may be installed under each guide (3) to make the emergency exit fit tightly against the inside surface of the cabin wall.

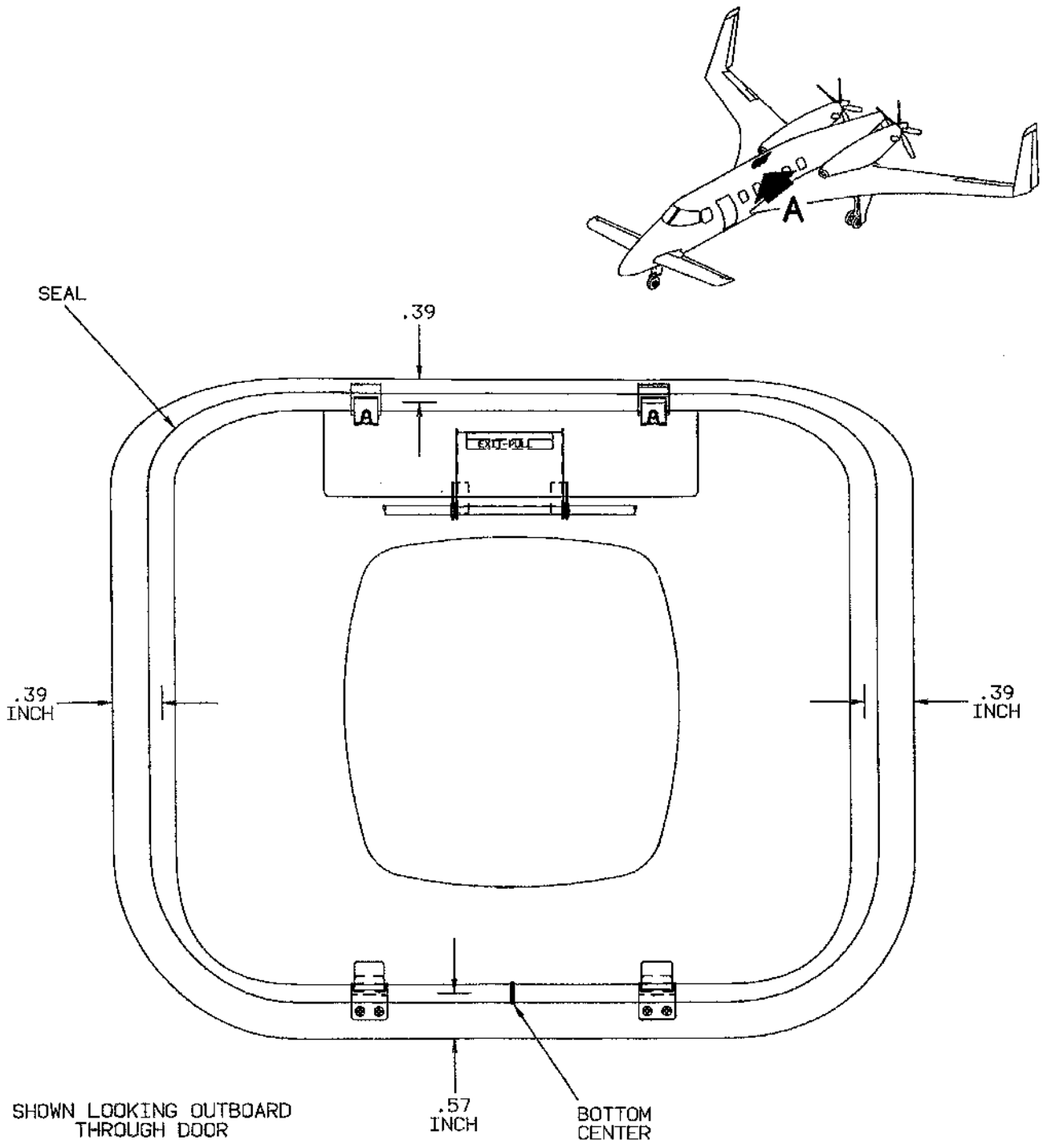
a. Remove the emergency exit door in accordance with the EMERGENCY EXIT DOOR REMOVAL procedures in this chapter.

b. Remove the door upholstery in accordance with the EMERGENCY EXIT DOOR UPHOLSTERY REMOVAL procedures in Chapter 25-20-00.

c. Remove the four screws (6) from the guide (3). Install the shims (1) as required behind the guide and install the screws.

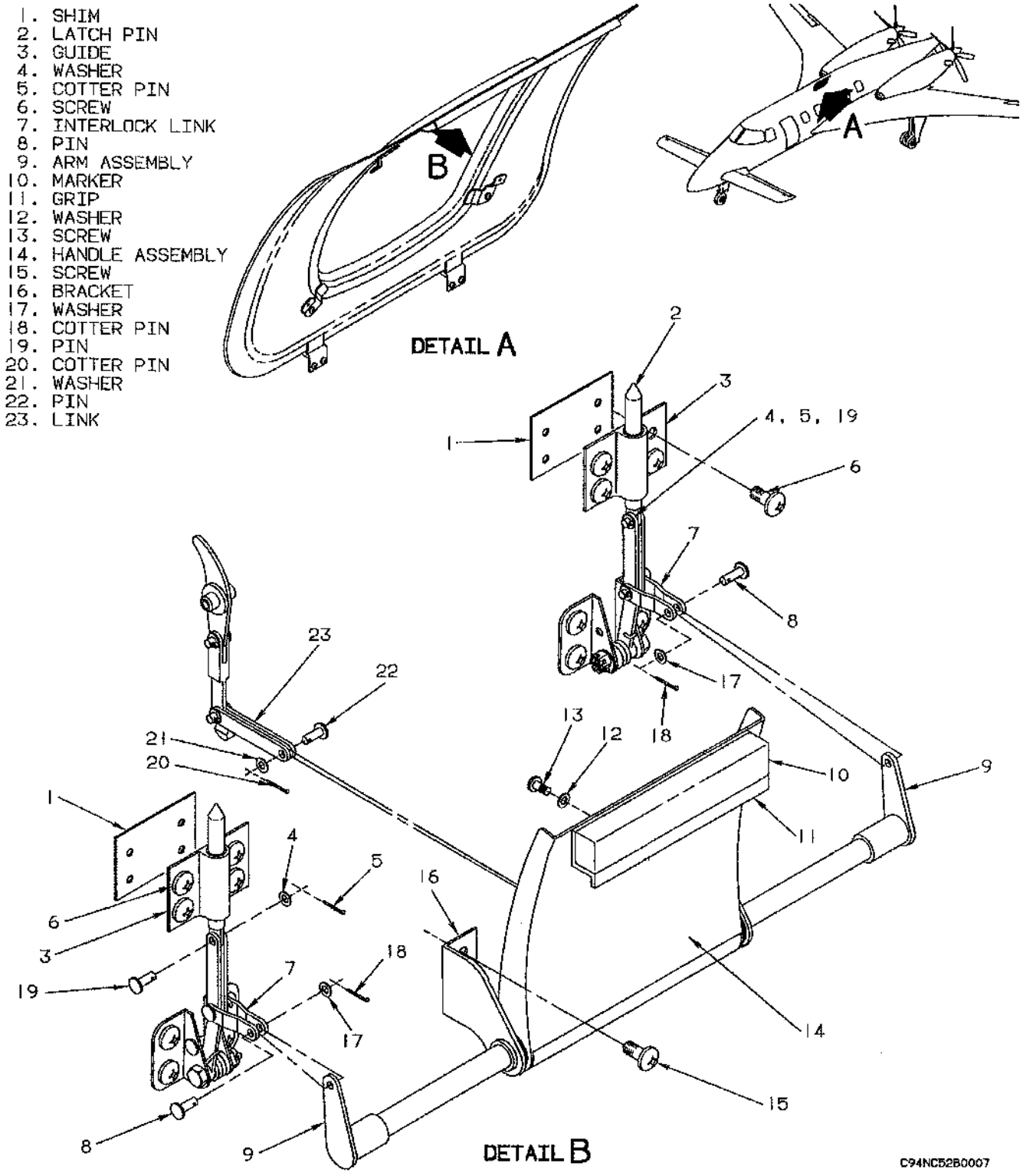
d. Install the door upholstery in accordance with the EMERGENCY EXIT DOOR UPHOLSTERY INSTALLATION procedures in Chapter 25-20-00.

e. Install the emergency exit door in accordance with the EMERGENCY EXIT DOOR INSTALLATION procedures in this chapter.



DETAIL A

Door Seal Installation (Effectivity: All)
Figure 1



Emergency Exit Door Component Installation (Effectivity: All)
Figure 2

SERVICE DOORS - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

Service doors allow access to components or systems in the airplane that require servicing. The service doors may be hinged doors with quick release latches or removable panels with screw type fasteners. The following service doors are covered in this chapter:

Air Conditioning System Service Compartment Access Panel - The air conditioning system service compartment access panel covers the air conditioning system service compartment located on the LH side of the aft fuselage.

Engine Oil Level Test Switch and OK Light Access Door - An access door for the oil level test switch and OK light is located in the lower fairing attached to each lower forward engine cowling. The doors are hinged and held in the closed position with quick release latches.

Engine Oil Filler Service Door - The oil filler service doors are located in the upper forward cowling of each engine. The oil filler service doors are hinged and held in the closed position with quick release latches.

Oxygen System Service Compartment Access Panel - The oxygen system service compartment access panel covers the oxygen system service compartment located in the RH lower nose access panel. The access panel is retained with captive fasteners.

NOTE

The brake fluid reservoir is located under the LH forward wing upper fairing, refer to Chapter 53-50-00 for information relating to this fairing.

SERVICE DOORS - MAINTENANCE PRACTICES (Effectivity: All)

AIR CONDITIONING SYSTEM SERVICE COMPARTMENT ACCESS PANEL REMOVAL (Effectivity: All) (Figure 1)

The air conditioning system service compartment access panel is removed as follows:

a. Loosen the attaching captive screws around the outside edge of the access panel.

b. Remove the access panel.

AIR CONDITIONING SYSTEM SERVICE COMPARTMENT ACCESS PANEL INSTALLATION (Effectivity: All) (Figure 1)

The air conditioning system service compartment access panel is installed as follows:

a. Place access panel in position.

b. Tighten the attaching captive screw fasteners.

ENGINE OIL LEVEL TEST SWITCH AND OK LIGHT ACCESS DOOR REMOVAL (Effectivity: All) (Figure 2)

The engine oil level test switch and OK light access door is removed as follows:

a. Refer to Chapter 71-10-00 and remove the lower fairing containing the oil level test switch and OK light access door from the lower forward engine cowling.

b. Using a small punch and a rawhide mallet, carefully drive out the hinge pin.

c. Remove the door.

ENGINE OIL LEVEL TEST SWITCH AND OK LIGHT ACCESS DOOR INSTALLATION (Effectivity: All) (Figure 2)

The engine oil level test switch and OK light access door is installed as follows:

a. Align the hinge pin holes in the hinge halves and install the hinge pin.

b. Crimp the end of the hinge halves to retain the hinge pin.

c. Refer to Chapter 71-10-00 and install the lower fairing containing the oil level test switch and OK light access door to the lower forward engine cowling.

ENGINE OIL FILLER SERVICE DOOR REMOVAL (Effectivity: All) (Figure 3)

The engine oil filler service door is removed as follows:

a. Refer to Chapter 71-10-00 and remove the upper forward engine cowling containing the oil filler service door.

b. Using a small punch and a rawhide mallet, carefully drive out the hinge pin securing the hinge halves together.

c. Remove the door from the upper forward cowling assembly.

ENGINE OIL FILLER SERVICE DOOR INSTALLATION (Effectivity: All) (Figure 3)

The engine oil filler service door is installed as follows:

a. Align the hinge pin holes in the hinge halves and install the hinge pin.

b. Crimp the end of the of the hinge halves together to retain the hinge pin.

c. Refer to Chapter 71-10-00 and install the upper forward engine cowling containing the oil filler access door.

OXYGEN SYSTEM SERVICE COMPARTMENT ACCESS PANEL REMOVAL (Effectivity: All) (Figure 4)

The oxygen system service compartment access panel is removed as follows:

a. Loosen the attaching captive fasteners installed around the edge of the access panel.

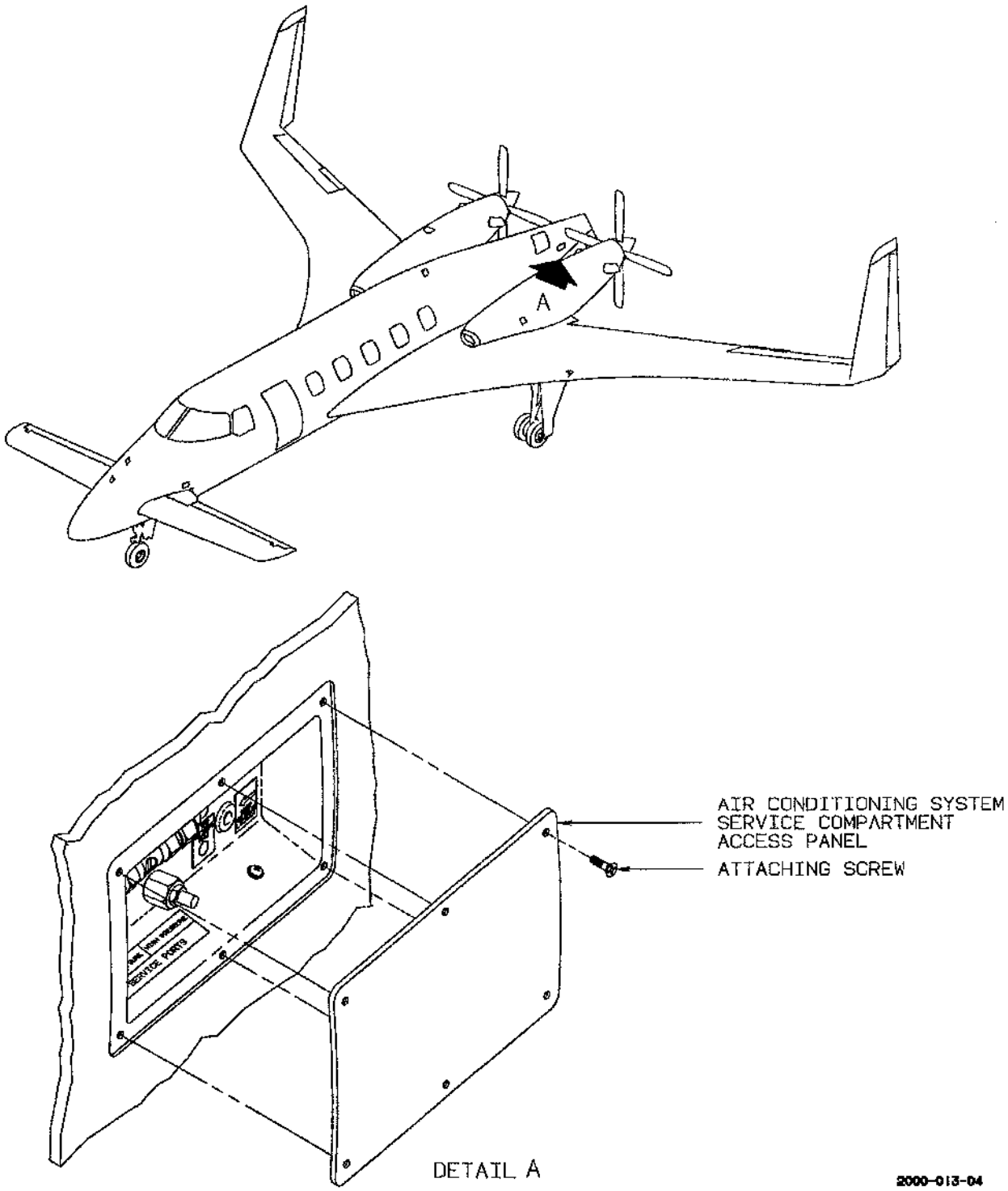
b. Remove the access panel from the airplane.

OXYGEN SYSTEM SERVICE COMPARTMENT ACCESS PANEL INSTALLATION (Effectivity: All) (Figure 4)

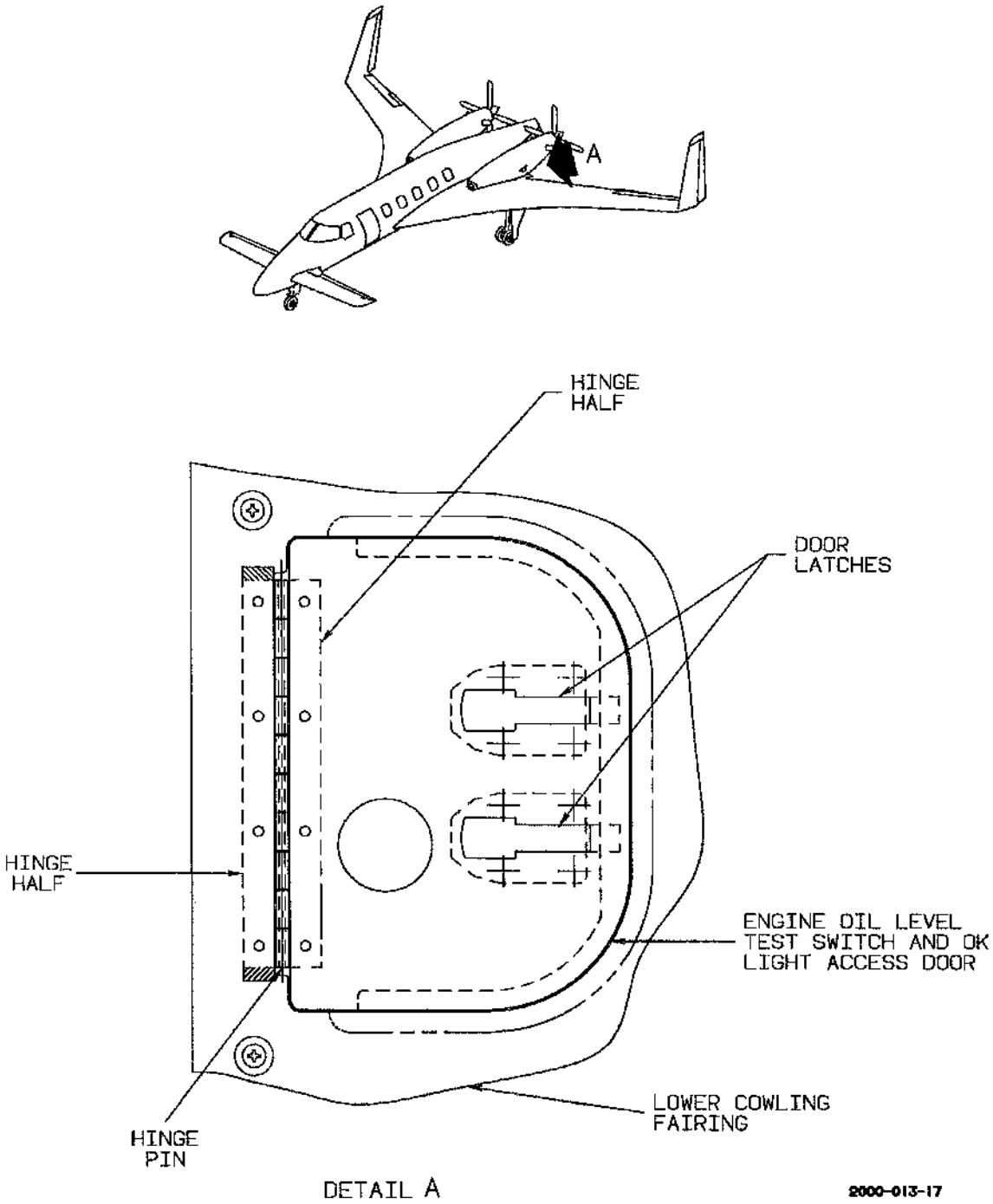
The oxygen system service compartment access panel is installed as follows:

a. Place the access panel in position on the RH side of the nose.

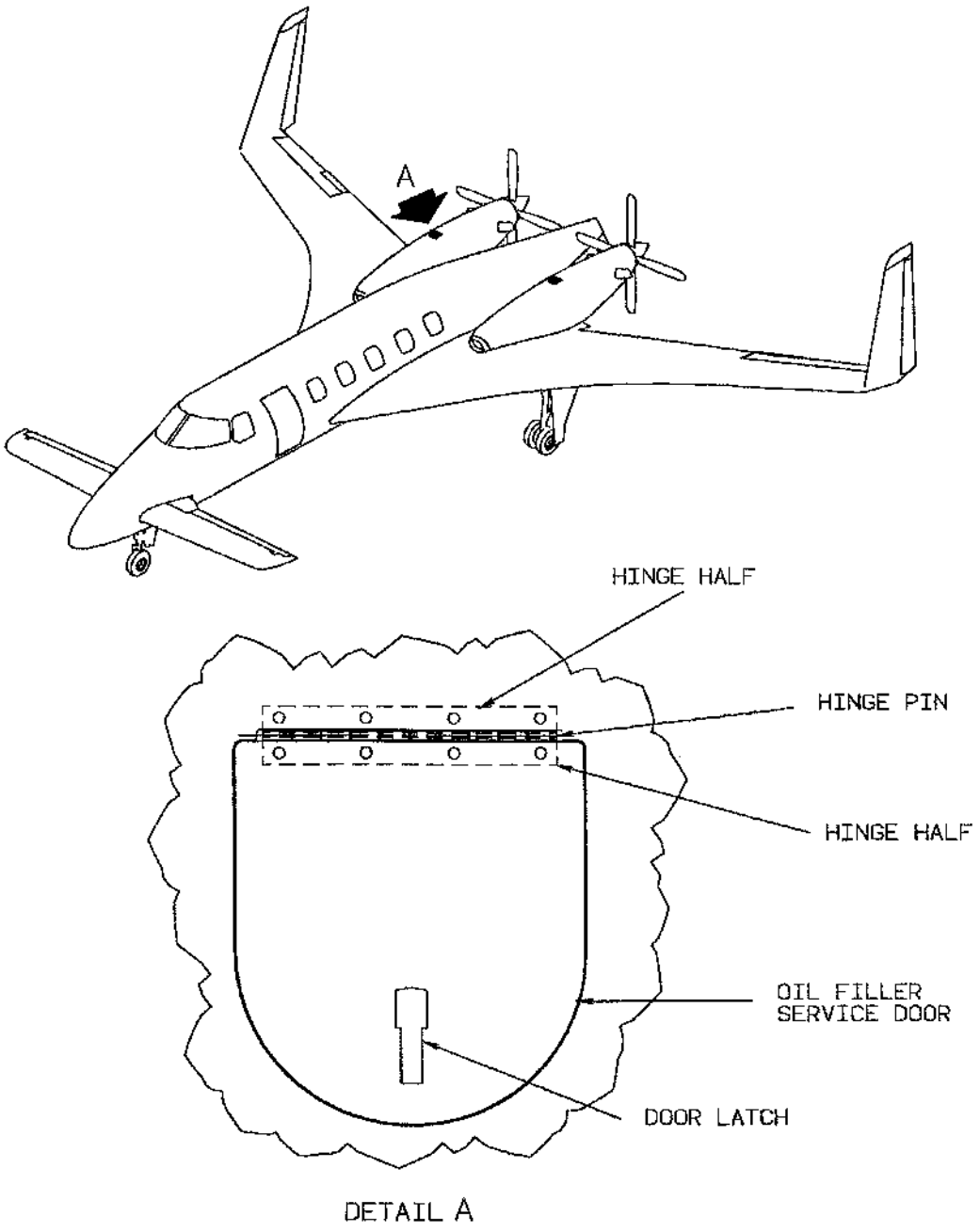
b. Tighten the attaching captive fasteners around the edge of the access panel.



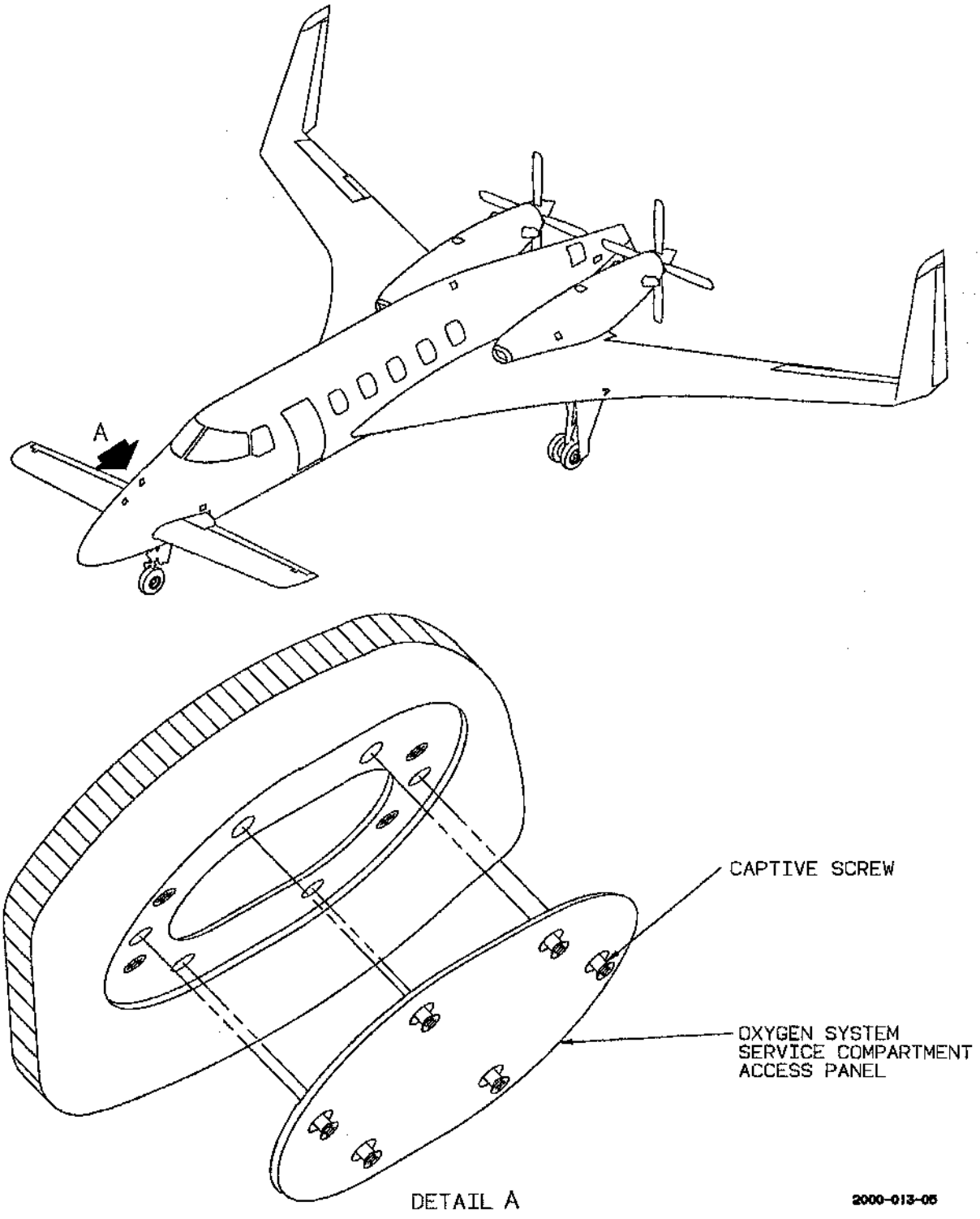
Air Conditioning System Service Compartment Access Panel Installation (Effectivity: All)
Figure 1



Engine Oil Level Test Switch and OK Light Access Door Installation (Effectivity: All)
Figure 2



Engine Oil Filler Service Door Installation
Figure 3



Oxygen System Service Compartment Access Panel Installation (Effectivity: All)
Figure 4

DOOR WARNING - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper latch pin installation procedures.

DOOR-UNLOCKED WARNING SYSTEM (Effectivity: All)

The Starship 1 uses seven latch-pin sense switches and a handle-position sense switch to provide signals to the warning annunciator circuitry that the cabin door is unlocked.

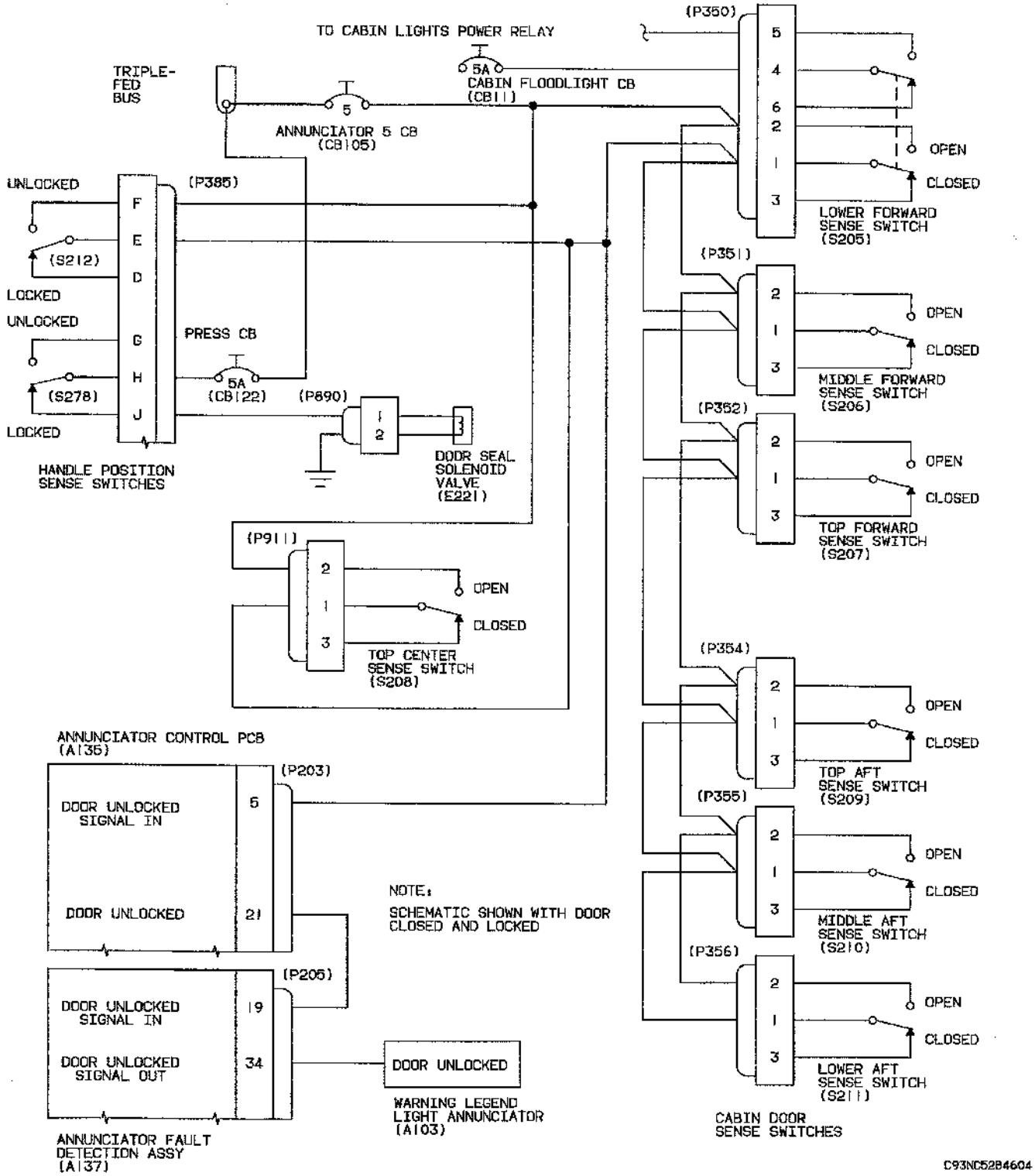
The door-handle-position sense switch is located on a bracket attached to the inboard side of the air pressure lock assembly. When the door is closed and locked, a clip attached to the link coming out of the air-pressure lock assembly will activate the sense switch. When activated, the sense switch will remove one source of power from the warning circuitry. The air-pressure lock assembly is behind the next to top step.

There are six latch-pin sense switches. Each switch is screwed into a threaded tube which attaches to the retainer assembly. The retainer assembly attaches to the inside of the doorframe opposite the frame doubler. A serrated cushion striker and striker are mounted on the side portion of the door frame with two screws which go through the doorframe and screw into two nut plates on the retainer assembly. The latch pins align with the holes in the striker plates and engage with the alignment portion of the threaded tube assembly until the pin reaches the fully extended position. At this point the latch-pin sense switch plunger should be depressed and the switch contacts should be in the "closed" position. Six of the plunger-type latch-pin sense switches are located on the forward and aft sides of the doorframe, three to a side. These six are adjusted by turning the switch the desired direction within the threaded tube.

The seventh latch-pin sense switch is located inside the cabin door on a bracket attached to the top latch-pin housing. The latch pin is beveled so that a roller on the switch actuator will ride up the bevel to actuate the switch when the latch pin is extended. This switch is located behind the bottom step. Refer to Chapter 52-10-00, Figure 1, Detail C for a view of the location of this switch.

DOOR-UNLOCKED ANNUNCIATION CIRCUIT (Effectivity: All) (Figure 1)

Refer to Figure 1 for a functional schematic of the system. All the switches are wired in parallel to provide an annunciation if one of the switches is not in the "locked" or "closed" position. Power to the circuit is provided by the number five annunciator circuit breaker (CB105) which receives power from the triple-fed bus. Twenty-eight vdc is routed to the "open" or "unlocked" contact of each switch. If the switch wiper is not in the "closed" or "locked" position, the power will be applied to pin 5 of the Annunciator Control PCB (A135). Since the annunciator is not dimmed, the power exits the PCB on pin 21 and goes to pin 19 of the Annunciator Fault Detection Assembly (A137). The assembly sends an output from pin 34 to pin 2 of the Warning Legend Light Assembly (A103) mounted on the glareshield where the red warning annunciation "DOOR UNLOCKED" will illuminate. For adjustment and maintenance of the door warning system components refer to maintenance practices in this section.



Door Warning System Schematic (Effectivity: All)
Figure 1

DOOR WARNING - TROUBLESHOOTING (Effectivity: All)

Refer to Charts 1 and 2 for door warning system troubleshooting.

CHART 1
TROUBLESHOOTING - DOOR WARNING (Effectivity: All)

"Door Unlocked" Message Annunciates When The Cabin Door Is Secured

STEP 1	Does the annunciator remain illuminated when a sense switch's electrical connector is disconnected? Refer to Figure 1.	**YES**	Repeat this step until all electrical connectors are disconnected. If the annunciator remains illuminated after all connectors are disconnected, check wiring for a short to power.
	NO		
STEP 2	Does the switch operate properly?	**YES**	Perform the appropriate switch adjustment procedure.
	NO		
	Replace defective switch.		

CHART 2
TROUBLESHOOTING - DOOR WARNING (Effectivity: All)

"DOOR UNLOCKED" Annunciator Does Not Light When Door Is Open

STEP 1	Is power available to switches? Refer to Figure 1.	**YES**	Proceed to Step 2.
	NO		
	Check power source, circuit breakers or repair power wiring.		
STEP 2	Check PCBs and wiring to annunciator.		

DOOR WARNING - MAINTENANCE PRACTICES (Effectivity: All)

SIDE LATCH-PIN SENSE SWITCH REMOVAL (Effectivity: All) (Figure 2)

NOTE

This procedure applies to all of the side mounted sense switches.

a. Remove power from the system by opening the ANNUNCIATOR 5 circuit breaker.

b. Remove the upholstery surrounding the cabin doorframe in accordance with the ENTRY DOOR POST TRIM REMOVAL procedures in Chapter 25-20-00.

c. Disconnect the switch electrical connector (2) from airplane harness.

d. Loosen the locking nut (3) on the threaded portion of the switch.

NOTE

Take care to prevent the tube assembly from being loosened when the switch is removed.

e. Unscrew the switch (1) from the tube assembly (4).

SIDE LATCH-PIN SENSE SWITCH INSTALLATION (Effectivity: All) (Figure 2)

NOTE

This procedure applies to all of the side mounted sense switches.

a. Screw the switch (1) a few turns into the tube assembly (4).

b. Perform the SIDE LATCH-PIN SENSE SWITCH ADJUSTMENT procedures in this chapter.

SIDE LATCH-PIN SENSE SWITCH ADJUSTMENT (Effectivity: All) (Figure 3)

NOTE

This procedure is for all side mounted sense switches.

a. Remove power from the system by opening the ANNUNCIATOR 5 circuit breaker.

b. Remove the upholstery surrounding the cabin doorframe in accordance with the ENTRY DOOR POST TRIM REMOVAL procedures in Chapter 25-20-00.

c. Disconnect the appropriate switch electrical connector (3) from airplane harness.

d. If the switch is not being installed or replaced, loosen the locking nut (1) on the threaded portion of the switch and unscrew the switch (2) until it will not contact the latch pin (5) when the door is closed and locked.

NOTE

Take care to prevent the tube assembly from being loosened when the switch is moved.

e. Close and lock the cabin door. Make sure the visual CLOSE indicators on the door pins line up.

f. Connect an ohmmeter to the switch electrical connector (3) contacts 1 and 3. The ohmmeter should read an open circuit.

g. Slowly screw the switch (2) into the tube (4) until the ohmmeter indicates continuity.

h. Turn the switch (2) four more complete revolutions.

i. Tighten the locking nut (1).

j. Unlock and lock the doors several times. The ohmmeter should show an open circuit when the door is unlocked and continuity when locked.

k. Connect the switch electrical connector to the airplane wiring.

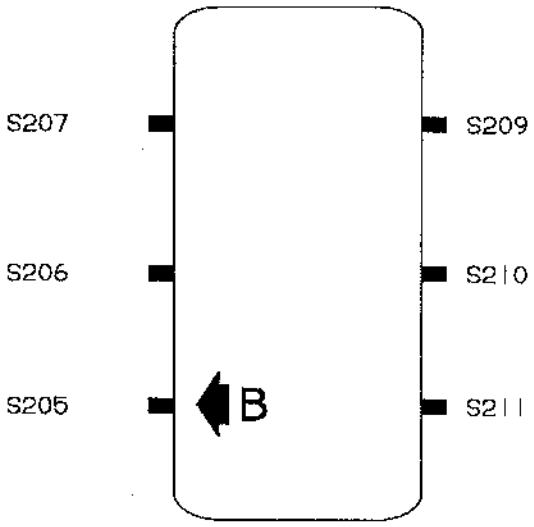
l. Close the ANNUNCIATOR 5 circuit breaker.

m. Apply electrical power to the airplane.

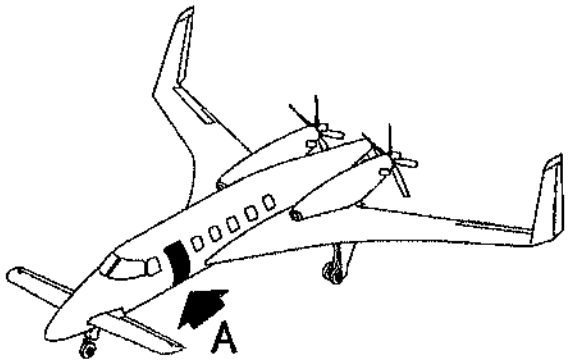
n. Verify that the DOOR UNLOCKED annunciator is extinguished when the door is locked.

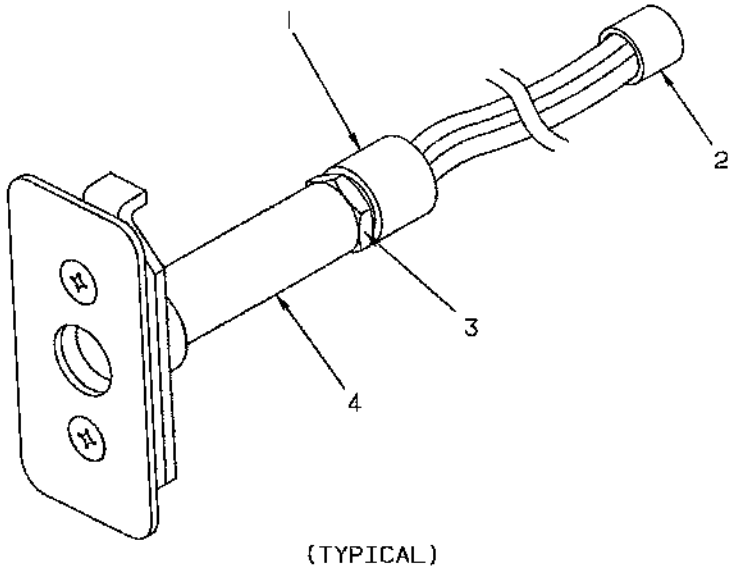
o. Remove electrical power from the airplane.

p. Install the upholstery surrounding the cabin doorframe in accordance with the ENTRY DOOR POST TRIM INSTALLATION procedures in Chapter 25-20-00.



DETAIL A





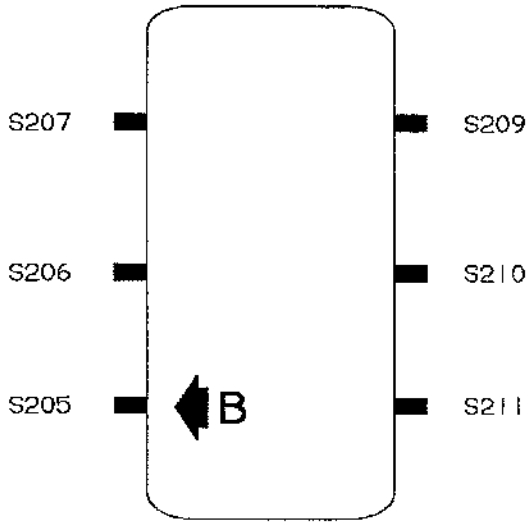
1. SWITCH
2. ELECTRICAL CONNECTOR
3. LOCKING NUT
4. TUBE ASSEMBLY

(TYPICAL)

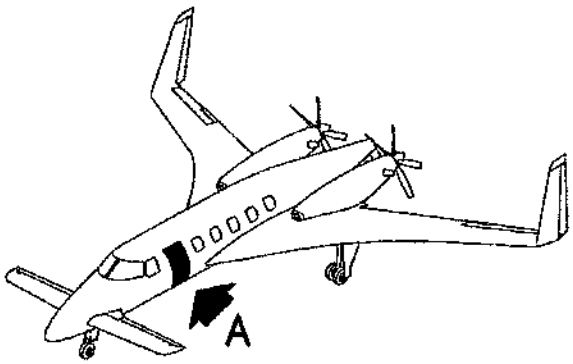
DETAIL B

C93NC52B4605

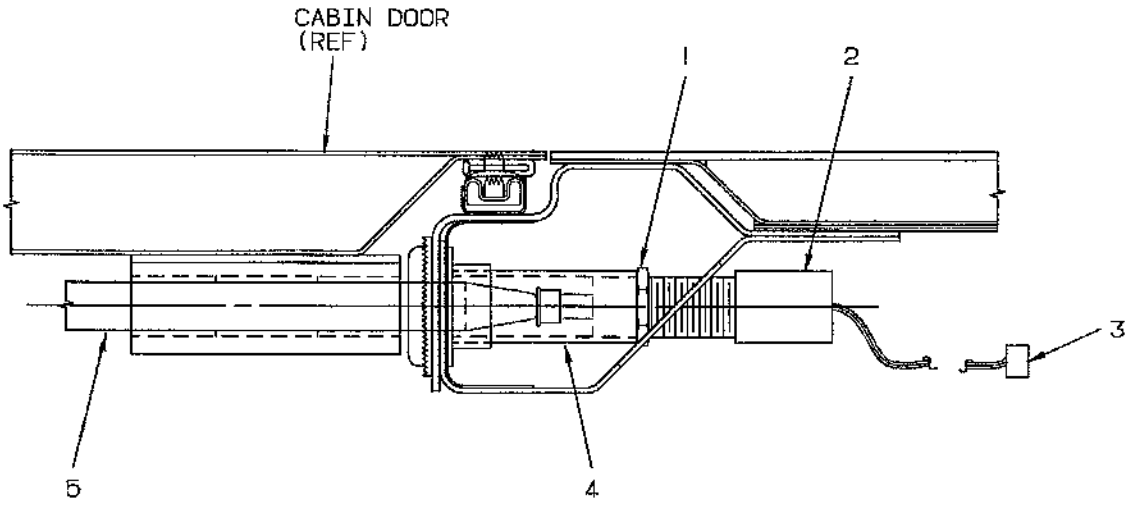
Side Latch-Pin Sense Switch Installation (Effectivity: All)
Figure 2



DETAIL A



1. LOCKING NUT
2. SWITCH
3. ELECTRICAL CONNECTOR
4. TUBE
5. LATCH PIN



DETAIL B

C93NC52B4606

Side Latch-Pin Sense Switch Adjustment (Effectivity: All)
Figure 3

TOP LATCH-PIN SENSE SWITCH REMOVAL (Effectivity: All) (Figure 4)

a. Remove power from the system by opening the ANNUNCIATOR 5 circuit breaker.

b. Remove the bottom step panel in accordance with the CABIN ENTRANCE DOOR STEP PANEL REMOVAL procedures in Chapter 52-10-00.

c. Disconnect the electrical connector (2) from the airplane harness.

d. Remove nuts and screws (1 and 3) attaching the switch (4) to the bracket (5).

TOP LATCH-PIN SENSE SWITCH INSTALLATION (Effectivity: All) (Figure 4)

a. Position the switch (4) in the mounting bracket (5) with the actuator roller facing the door latch pin.

b. Install, but do not tighten, the screws and nuts (1 and 3) through the switch (4) and bracket (5).

c. Perform the TOP LATCH-PIN SENSE SWITCH ADJUSTMENT procedures in this chapter.

TOP LATCH-PIN SENSE SWITCH ADJUSTMENT (Effectivity: All) (Figure 4)

a. Remove power from the system by opening the ANNUNCIATOR 5 circuit breaker.

b. Remove the bottom step panel in accordance with the CABIN ENTRANCE DOOR STEP PANEL REMOVAL procedures in Chapter 52-10-00.

c. Disconnect the electrical connector (2) from the airplane harness.

d. If the switch is not being installed or replaced, loosen the nuts and screws (1 and 3) attaching the switch (4) to the bracket (5) and adjust until the switch is not actuated.

e. With the cabin door in the open position, extend the latch pins to the locked position.

f. Connect an ohmmeter to the switch electrical connector (2) contacts 1 and 3. The ohmmeter should read an open circuit.

g. Slowly adjust the switch (4) toward the latch pin (6) until the ohmmeter indicates continuity.

h. Adjust the switch a further 0.02 inches past the operating point.

i. Tighten the screws and nuts (1 and 3).

j. Unlock and lock the doors several times. The ohmmeter should show an open circuit when the door is unlocked and continuity when locked.

k. Connect the switch electrical connector (2) to the airplane wiring.

l. Unlock the cabin door.

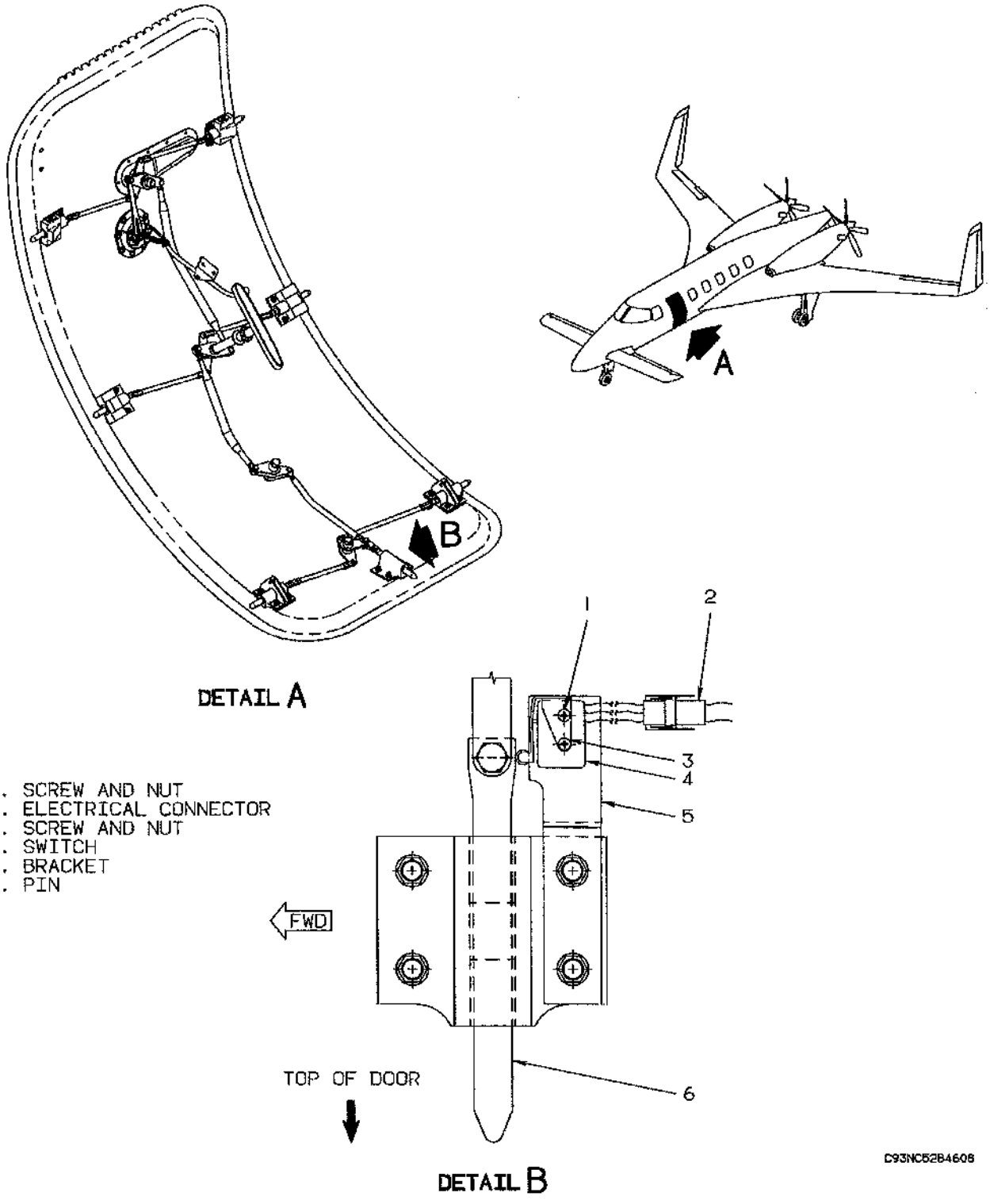
m. Close the ANNUNCIATOR 5 circuit breaker.

n. Apply electrical power to the airplane.

o. Verify that the DOOR UNLOCKED annunciator is extinguished when the door is closed and locked.

p. Remove electrical power from the airplane.

q. Install the bottom step panel in accordance with the CABIN ENTRANCE DOOR STEP PANEL INSTALLATION procedures in Chapter 52-10-00.



Top Latch-Pin Sense Switch Installation and Adjustment (Effectivity: All)
Figure 4

HANDLE-POSITION SENSE SWITCH REMOVAL (Effectivity: All) (Figure 5)

a. Remove power from the system by opening the ANNUNCIATOR 5 circuit breaker.

b. Remove the step panel and riser just below the top door step in accordance with the CABIN ENTRANCE DOOR STEP PANEL REMOVAL procedures in Chapter 52-10-00.

c. Remove the nuts (6), washers (7 and 11) and screws (10) securing the clip (12) to the link (8).

d. Remove the nuts and star washers (1) and screws (5) securing the switches (2 and 3) to the bracket (4). Remove the actuators (9) from the switches (2 and 3).

e. If required, identify, tag and disconnect the switch wiring.

HANDLE-POSITION SENSE SWITCH INSTALLATION (Effectivity: All) (Figure 5)

a. Align the actuators (9) with the switches (2 and 3) and insert into the bracket (4). Make sure the switches and actuators are level with each other. Secure the parts with the screws (5) and nuts and star washers (1).

b. Position the clip (12) on the link (8) and install the screws (10), washers (7 and 11) and nuts (6). Do not tighten the parts.

c. Perform the HANDLE-POSITION SENSE SWITCH ADJUSTMENT procedures in this chapter.

HANDLE-POSITION SENSE SWITCH ADJUSTMENT (Effectivity: All) (Figure 5)

a. Remove power from the system by opening the ANNUNCIATOR 5 circuit breaker.

b. Remove the step panel and riser just below the top door step in accordance with the CABIN ENTRANCE DOOR STEP PANEL REMOVAL procedures in Chapter 52-10-00.

c. If the switch is not being installed or replaced, loosen the nuts (6) and screws (10) securing the clip (12) to the link (8) and adjust the clip until the switches are not actuated.

d. With the cabin door in the open position, extend the latch pins to the locked position.

NOTE

Since the door-warning switch and the door-seal solenoid-valve switch are located side by side this procedure will adjust the actuation point for both switches. If both switches do not operate at approximately the same time, check the actuators for missmatches.

e. Connect an ohmmeter to the wiring connected to the C and NO contacts of S212. The ohmmeter should read an open circuit. This is for the adjustment of the door-warning switch.

f. Connect an ohmmeter to the wiring connected to the C and NO contacts of S278. The ohmmeter should read an open circuit. This is for the adjustment of the door-seal solenoid-valve switch.

g. Slowly adjust the clip (12) toward the switch actuators (9) until the ohmmeters indicate continuity. Both ohmmeters should indicate continuity at approximately the same time.

h. Adjust the clip (12) a further 0.02 inches past the operating point.

i. Tighten the screws (10) and nuts (6).

j. Unlock and lock the doors several times. The ohmmeters should show an open circuit when the door is unlocked and continuity when locked.

k. Unlock the cabin door.

l. If required, connect the switch wiring.

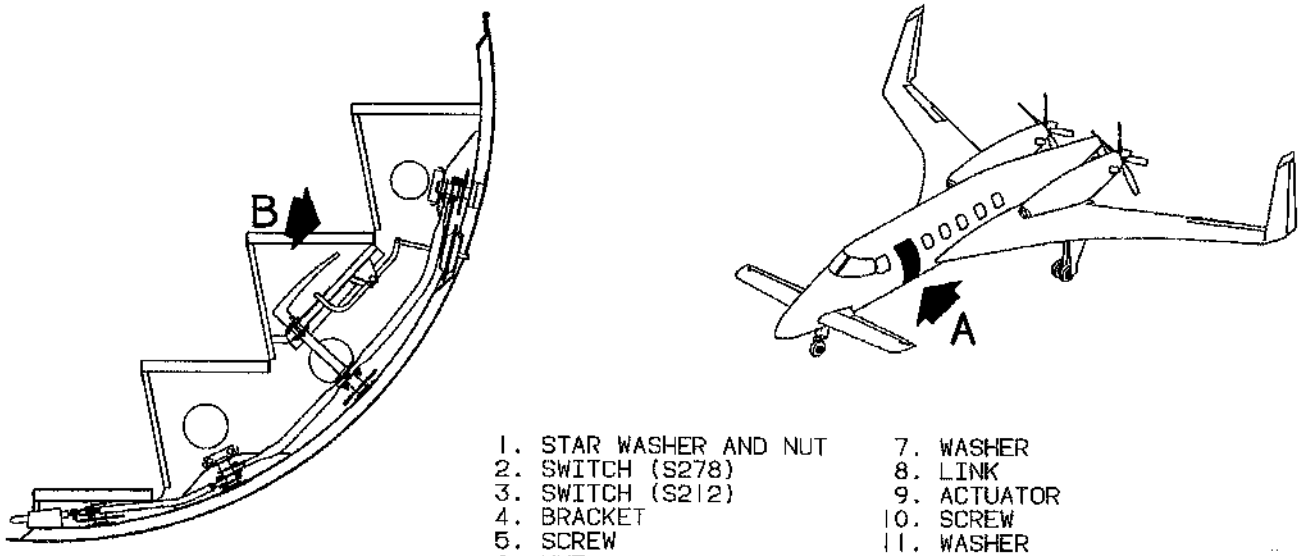
m. Close the ANNUNCIATOR 5 circuit breaker.

n. Apply electrical power to the airplane.

o. Verify that the DOOR UNLOCKED annunciator is extinguished when the door is closed and locked.

p. Remove electrical power from the airplane.

q. Install the step panel and riser in accordance with the CABIN ENTRANCE DOOR STEP PANEL INSTALLATION procedures in Chapter 52-10-00.



Handle-Position Sense Switch Installation and Adjustment (Effectivity: All)
Figure 5

LANDING GEAR DOORS - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

All of the landing gear doors are constructed of composite material.

MAIN LANDING GEAR DOORS (Effectivity: All)

The main landing gear doors are mounted on the main landing gear. The top of the doors are hinged to the wing structure.When the landing gear is retracted the landing gear doors cover the gear well opening.

NOSE LANDING GEAR DOORS (Effectivity: All)

FORWARD DOORS

The nose landing gear forward doors are hinged fore and aft to the airplane structure. The doors are open for retraction or extension, however as either cycle is completed the forward nose gear doors close. The position of the doors is controlled by the action of the nose landing gear door hydraulic actuator. The actuator is connected to the forward nose gear doors by means of a bellcrank and push rods.

AFT DOORS

The nose landing gear aft doors open and close with the action of the nose gear. The doors are open with the gear extended and closed with the gear retracted.

LANDING GEAR DOORS - MAINTENANCE PRACTICES

MAIN LANDING GEAR DOOR REMOVAL (Effectivity: All) (Figure 1)

a. Remove the wing access door (532AB or 632AB) to gain access to the main landing gear door hinges.

NOTE

Only the procedure for the left main landing gear door is shown. Procedures for the right main landing gear door are the same.

b. On the forward bracket (7), remove the nut (4), washer (5), and bolt (3). Remove the two washers (2) between the retract rod (1) and the bracket.

c. On the aft bracket (6), remove the nut (4), washer (5), and bolt (3). Remove the two washers (2) between the retract rod (1) and the bracket.

d. Support the gear door.

e. Reach through the access panel opening and on the forward bracket (18) remove the nut (9), washer (10), bolt (17), washer (16), spacers (11 and 15) and bushing (13).

f. Reach through the access panel opening and on the aft bracket (8) remove the nut (9), washer (10), bolt (17), washer (16), spacers (11 and 15) and bushing (13).

g. Remove the door from the airplane.

h. If required, remove the bushings (12 and 14) from the hinges (19).

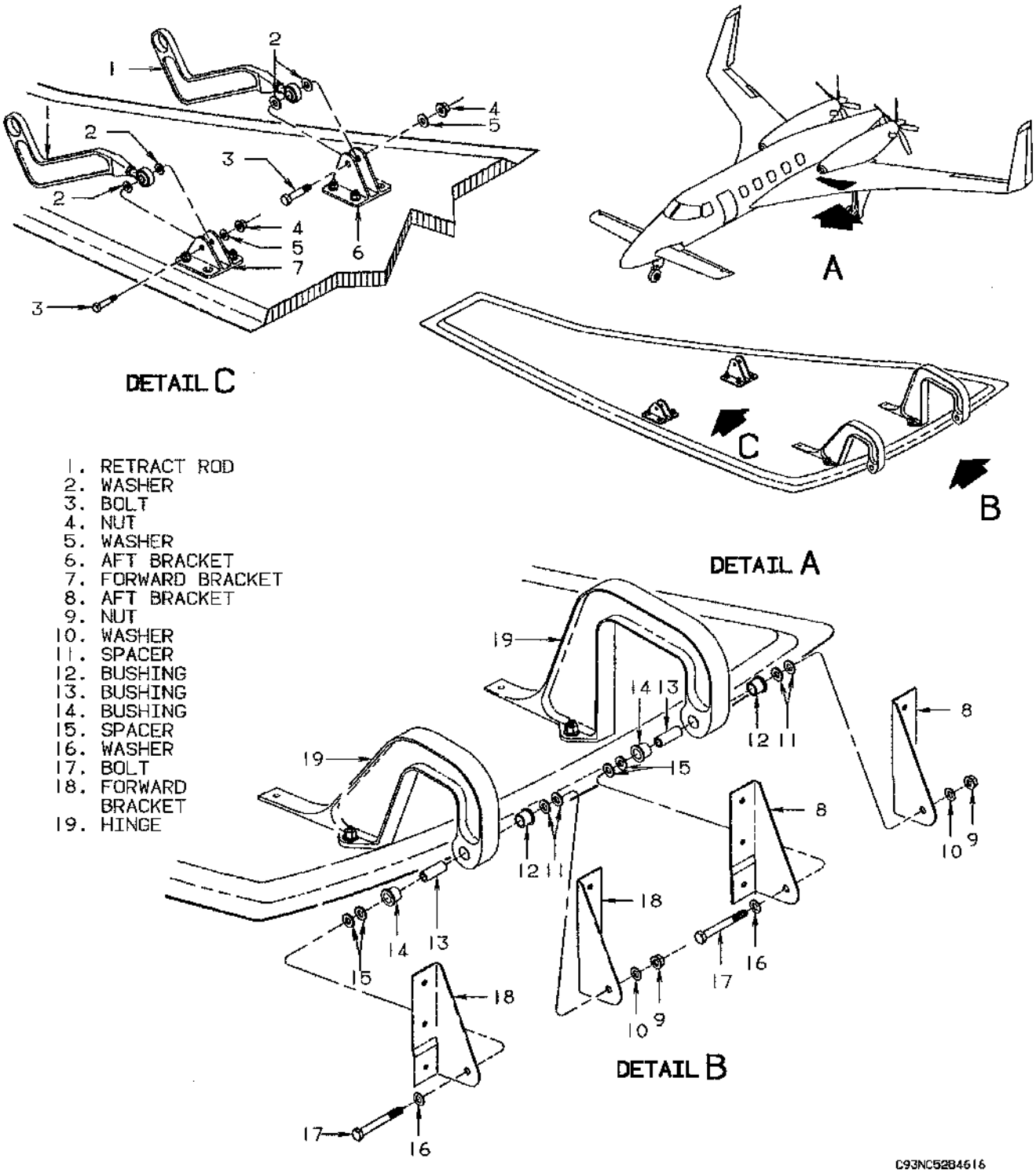
MAIN LANDING GEAR DOOR INSTALLATION (Effectivity: All) (Figure 1)

NOTE

Only the procedure for the left main landing gear door is shown. Procedures for the right main landing gear door are the same.

a. If required, install the bushings (12 and 14) in the hinges (19).

b. Position the gear door hinges (19) between the forward (18) and aft (8) brackets.



Main Landing Gear Doors Installation (Effectivity: All)
Figure 1

c. Install the bushings (13) in the hinges (19).

d. Place the spacers (11 and 15) between the hinges (19) and the brackets (8 and 18). Insert the bolts (17) and washers (16) through the brackets, spacers and hinges. Install the washers (10) and nuts (9) on the bolts.

e. Position the retract rod ends (1) between the brackets (6 and 7).

f. Position the washers (2) on either side of the rod ends (1).

g. Install the bolts (3) through the brackets (6 and 7) and washers (2) and secure with the washers (5) and nuts (4).

h. Install the wing access door (532AB or 632AB).

i. Rig the main landing gear doors as instructed in Chapter 32-30-00.

FORWARD NOSE LANDING GEAR DOORS REMOVAL (Effectivity: All) (Figure 2)

NOTE

This procedure can be used to remove either forward nose landing gear door.

a. Remove the appropriate nose access panel (131AL or 132AR) to gain access to the aft hinge of the forward nose landing gear door.

b. Pump open the nose gear doors by use of the alternate extension handle in the cockpit.

c. Perform the NOSE LANDING GEAR DOOR LOCKPIN INSTALLATION procedure in Chapter 32-20-00.

d. Remove the appropriate nose access panel (710AZ or 710BZ) to gain access to the forward hinge of the forward nose landing gear door.

e. Remove the screw (14), washer (13), lock washer (11) and nut (10) securing the ground strap (12) to the aft hinge (4).

f. Remove the cotter pin (16), nut (17), washer (18) and bolt (2) and disconnect the tube (1) from the bracket (3).

g. Support the nose gear door.

h. Reach through the access panel opening and remove the cotter pins (7), nut (8), spacers (9), and bolt (15) from the keel brackets (6) and remove the door.

i. If required, remove the bearing (5) from the hinge (4).

FORWARD NOSE LANDING GEAR DOORS INSTALLATION (Effectivity: All) (Figure 2)

NOTE

This procedure can be used to install either forward nose landing gear door.

a. If required, install the bearing (5) in the hinge (4).

b. Position the door hinges (4) in the keel brackets (6). Place the washers (9) on either side of the bearings (5) and install the bolts (15) through the brackets, washers and bearings. Install the nuts (8) and cotter pins (7) on the bolts.

c. Position the tube (1) in the bracket (3). Install the bolt (2), washer (18), nut (17) and cotter pin (16).

d. Install the ground strap (12) to the aft hinge (4) with the screw (14), washer (13), lock washer (11) and nut (10).

e. Install the appropriate nose access panel (710AZ or 710BZ).

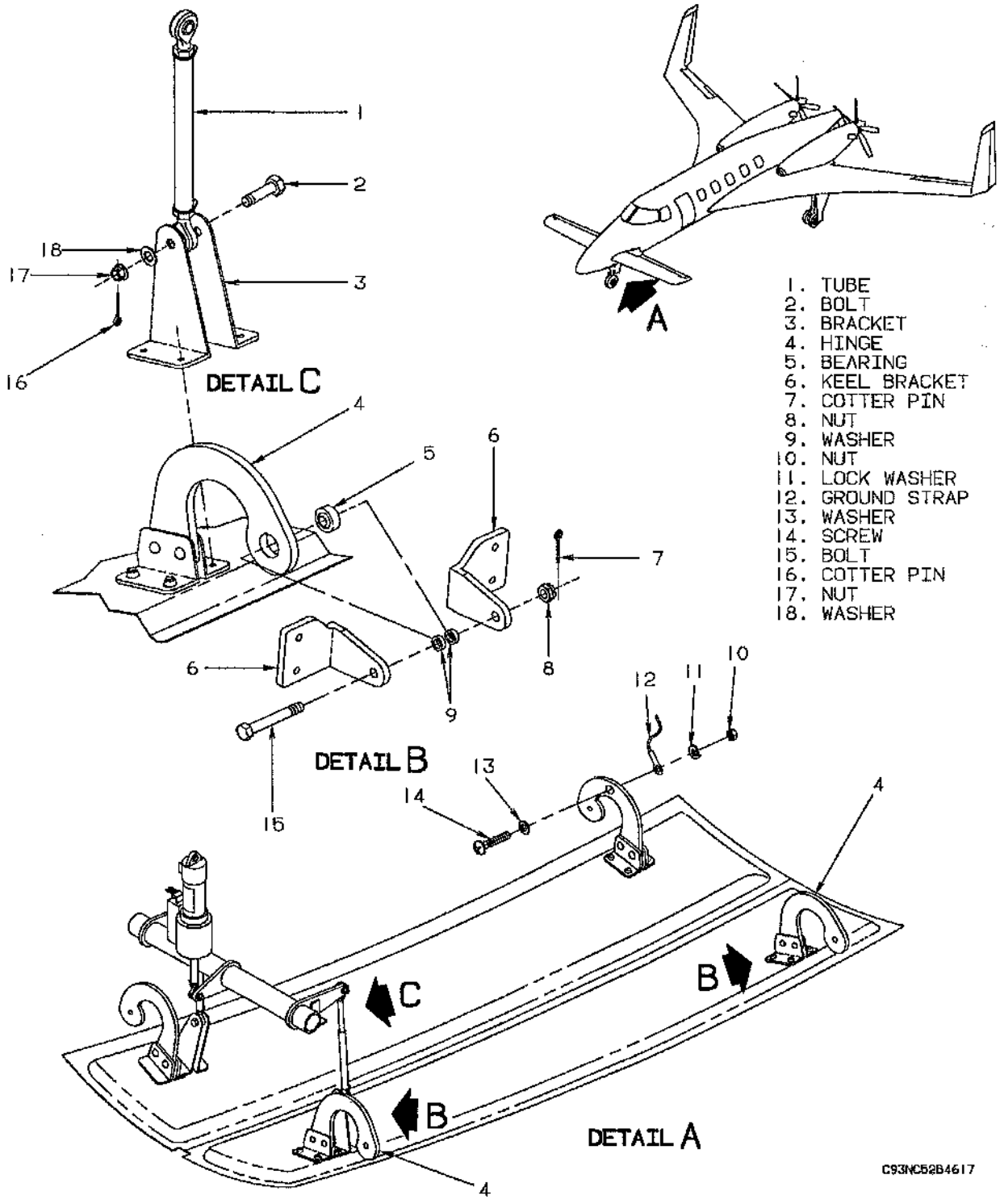
f. Remove the nose landing gear door lock pin.

g. Using hand pressure, carefully and firmly push up on the two nose landing gear doors until they lock in the closed position.

h. Gently pull down on the aft end of the doors to make certain the doors are locked.

i. Rig the nose landing gear forward doors in accordance with Chapter 32-30-00.

j. Install the appropriate nose access panel (131AL or 132AR).



Forward Nose Landing Gear Doors Installation (Effectivity: All)
Figure 2

FORWARD NOSE LANDING GEAR DOOR SEAL REMOVAL (Effectivity: All) (Figure 3)

CAUTION

To prevent damage to the airplane make sure that all the screws in the lower nose panel (131AL and 132AR) are fully installed before opening the nose landing gear doors.

a. Pump open the nose gear doors by use of the alternate extension handle in the cockpit.

b. Insert the nose landing gear door lockpin as instructed in Chapter 32-20-00.

NOTE

Do not use a metal scraper on composite material.

c. Use a nonmetallic scraper to carefully separate the door seal from the door.

d. Use cleaner (10, Chart 1, 52-00-00) to remove any old adhesive or seal residue from the door edge.

FORWARD NOSE LANDING GEAR DOOR SEAL INSTALLATION (Effectivity: All) (Figure 3)

NOTE

The door seal is installed on the left forward nose gear door.

a. The door seal is located 0.01 to 0.06 inches from each end of the door. Lay the seal out on the door and trim to the length that will cover the door except for the desired recess.

b. Use cleaner (10, Chart 1, 52-00-00) to clean the inner and outer edges of the door past the area that will be covered by the door seal.

c. Use a black Pentel marking pencil, draw a line on both sides of the gear door .28 inches from the edge.

d. Use cleaner (10, Chart 1, 52-00-00) to clean the attaching surfaces of the door seal.

NOTE

The adhesive must be allowed to dry until tack free before the seal can be installed. Keep the door seal attaching surfaces separated after applying the adhesive.

e. Apply a coat of adhesive (2, Chart 1, 52-00-00) to the line drawn on the door.

f. Apply a coat of adhesive (2, Chart 1, 52-00-00) to the cleaned area of the door seal.

g. When the adhesive is tack free, install the door seal on the door so that the outer edge of the attaching surface aligns with the line drawn on the door.

h. Use cleaner (10, Chart 1, 52-00-00) to remove any excess adhesive from the door seal edges.

i. Allow the adhesive to cure in accordance with the manufacturer's specifications.

j. Remove the nose gear door bellcrank lock pin.

k. Using hand pressure, carefully and firmly push up on the two nose landing gear doors until they lock in the closed position.

l. Gently pull down on the aft end of the doors to make certain the doors are locked.

AFT NOSE LANDING GEAR DOOR REMOVAL (Effectivity: All) (Figure 4)

a. Remove the cotter pin (10), nut (9), washers (4 and 8), spacers (5 and 7) and bolt (3) from the clevis (6). Move the tube clear of the door.

b. Support the door.

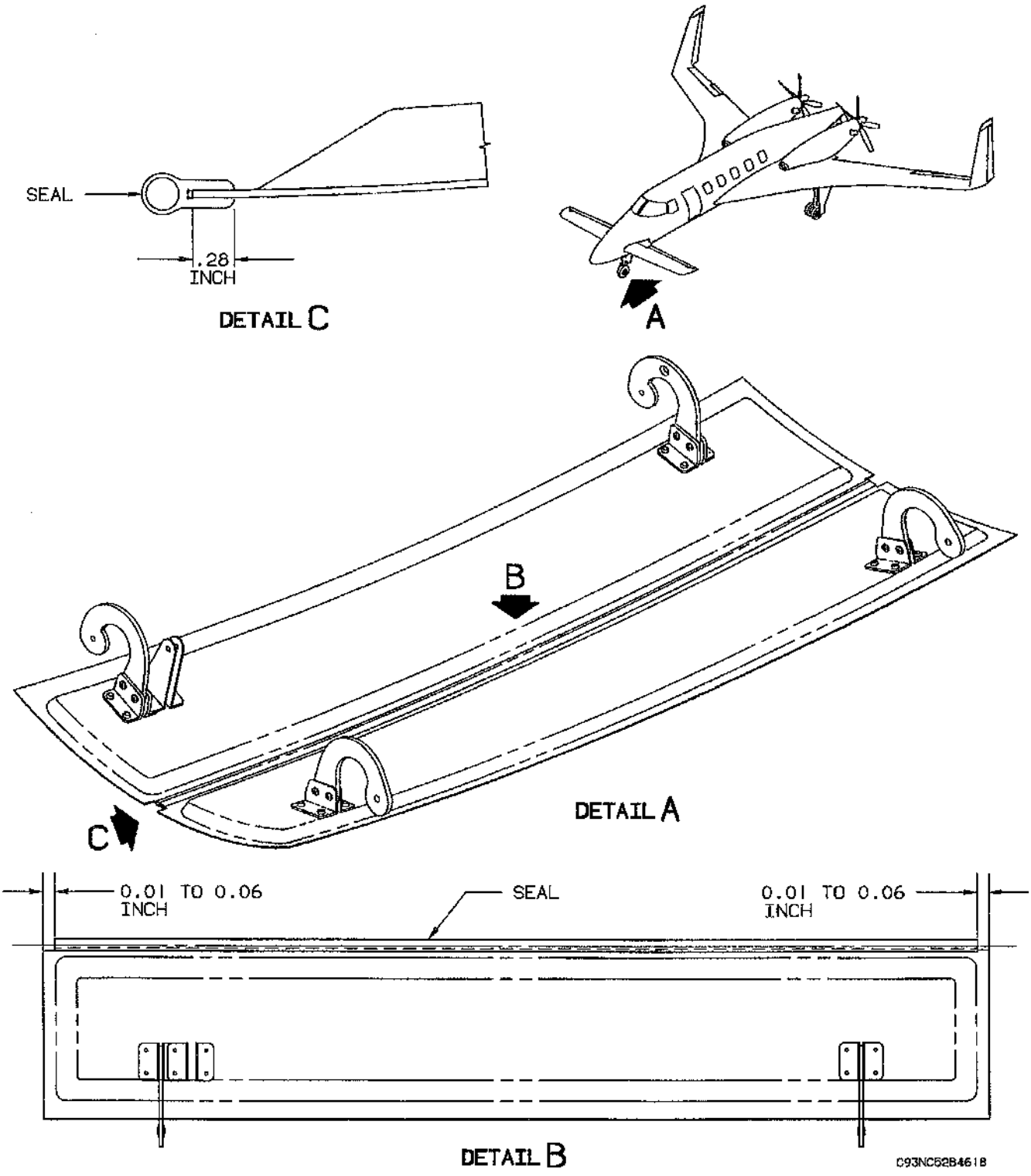
c. Remove the screws (1) and washers (2) attaching the door hinges to the airplane. Remove the ground strap (11) from the center hole of the forward hinge.

AFT NOSE LANDING GEAR DOOR INSTALLATION (Effectivity: All) (Figure 4)

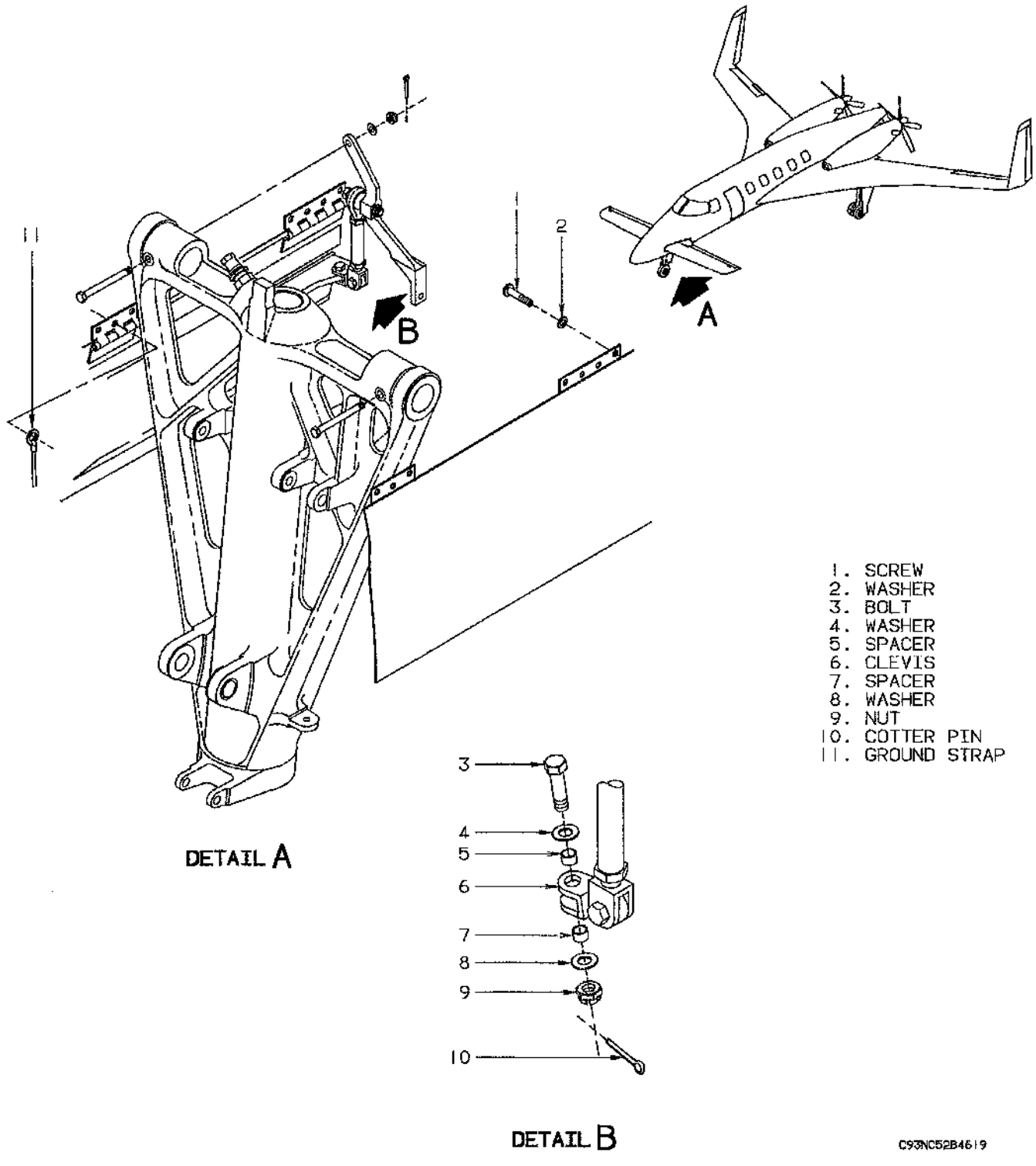
a. Position the door and attach the hinges to the airplane with the screws (1) and washers (2). The ground strap (11) attaches to the center hole of the forward hinge.

b. Position the clevis (6) on the door and insert the spacers (5 and 7). Install the bolt (3) and washer (4) through the bushings and clevis. Install the washer (8), nut (9) and cotter pin (10).

c. Rig the aft gear doors in accordance with Chapter 32-30-00.



Forward Nose Landing Gear Door Seal Installation (Effectivity: All)
Figure 3



Aft Nose Landing Gear Door Installation (Effectivity: All)
Figure 4

AFT NOSE LANDING GEAR DOOR SEAL REMOVAL (Effectivity: All) (Figure 5)

NOTE

Do not use a metal scraper on composite material.

a. Use a nonmetallic scraper to carefully separate the door seal (5) from the door.

b. Use cleaner (10, Chart 1, 52-00-00) to remove any old adhesive or seal residue from the door edge.

AFT NOSE LANDING GEAR DOOR SEAL INSTALLATION (Effectivity: All) (Figure 5)

NOTE

The door seal (5) is installed on the left aft nose gear door.

a. The door seal (5) is installed flush or back .05 inches from each end of the door. Lay the seal out on the door and trim to the length that will cover the door except for the desired recess.

b. Use cleaner (10, Chart 1, 52-00-00) to clean the inner and outer edges of the door past the area that will be covered by the door seal.

c. Use a black Pentel marking pencil, draw a line on both sides of the gear door .31 inches from the edge.

d. Use cleaner (10, Chart 1, 52-00-00) to clean the attaching surfaces of the door seal (5).

e. Apply a coat of adhesive (2, Chart 1, 52-00-00) to the line drawn on the door.

f. Apply a coat of adhesive (2, Chart 1, 52-00-00) to the cleaned area of the door seal (5).

NOTE

The adhesive must be allowed to dry until tack free before the seal can be installed. Keep the door seal attaching surfaces separated after applying the adhesive.

g. When the adhesive is tack free, install the door seal (5) on the door so that the outer edge of the attaching surface aligns with the line drawn on the door.

h. Use cleaner (10, Chart 1, 52-00-00) to remove any excess adhesive from the door seal edges.

i. Allow the adhesive to cure in accordance with the manufacturer's specifications.

AFT NOSE LANDING GEAR DOOR KEEL SEAL REMOVAL (Effectivity: All) (Figure 5)

NOTE

These procedures cover the installation of the three door seals (1, 3 and 4). Perform only the steps that apply to the seal being replaced.

Do not use a metal scraper on composite material.

a. Use a nonmetallic scraper to carefully separate the door seal (4) from the door.

b. Use cleaner (10, Chart 1, 52-00-00) to remove any old adhesive or seal residue from the keel attachment surfaces.

AFT NOSE LANDING GEAR DOOR KEEL SEAL INSTALLATION (Effectivity: All) (Figure 5)

NOTE

These procedures cover the installation of the three door seals (1, 3 and 4). Perform only the steps that apply to the seal being replaced.

a. The seal placement is as follows:

1. The forward and aft seal (1 and 4) ends, may be 0.0 to 0.05 inches from the hinge plates (2).

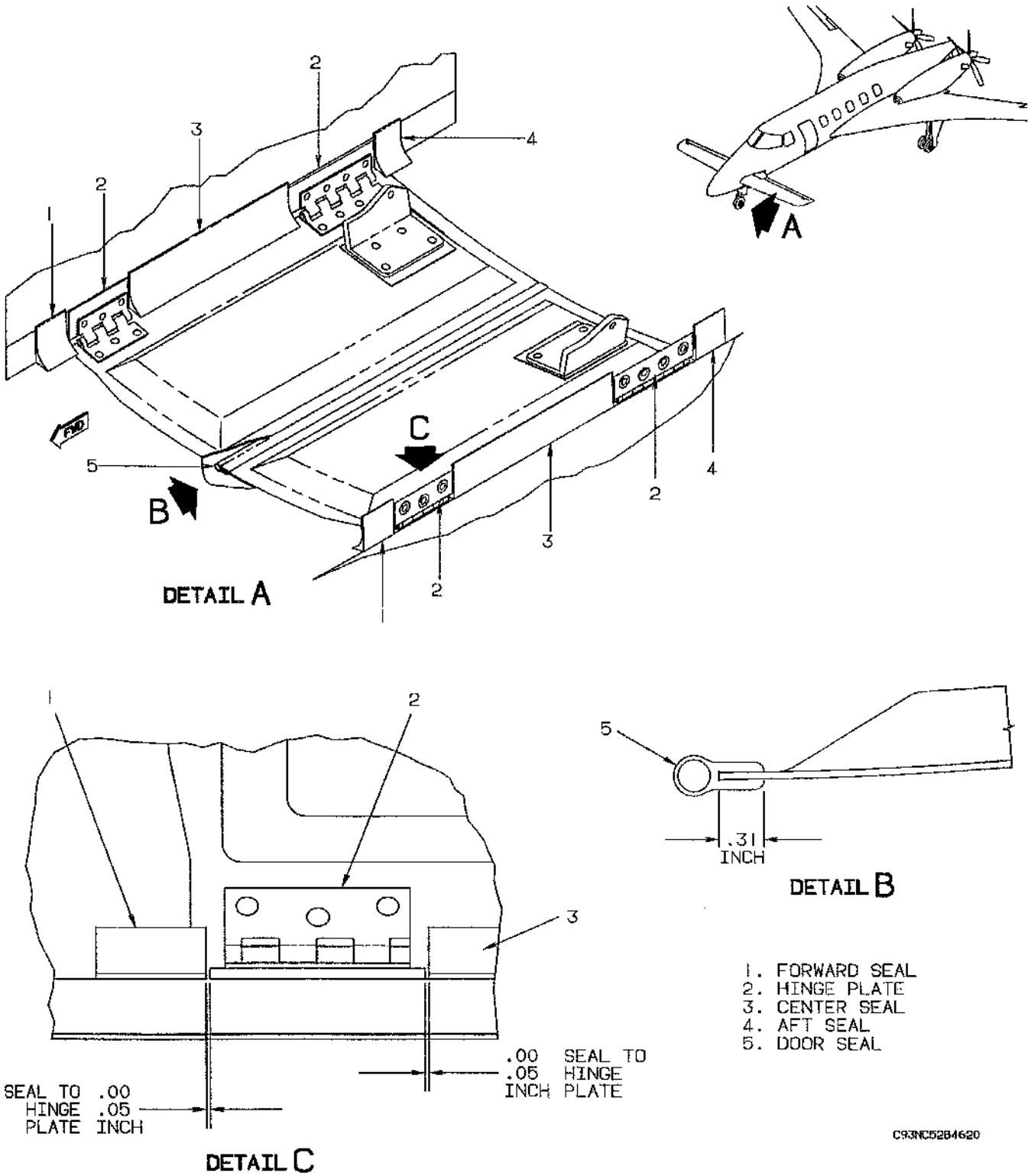
2. The center seal (3) ends, may be 0.0 to 0.05 inches from the hinge plates (2).

3. The seals (1, 3 and 4) should extend 0.01 to 0.05 inches below the lower edge of the keel.

b. Lay the seal out on the keel and trim to the length that will cover the door except for the desired recess.

c. Use cleaner (10, Chart 1, 52-00-00) to clean the 1.5 inch wide flange of the seal.

d. Use cleaner (10, Chart 1, 52-00-00) to clean the keel area.



Aft Nose Landing Gear Door Seal Installation (Effectivity: All)
Figure 5

NOTE

The adhesive must be allowed to dry until tack free before the seal can be installed.

e. Apply a coat of adhesive (2, Chart 1, 52-00-00) to the 1.5 inch wide flange of the seal (2).

f. Apply a coat of adhesive (2, Chart 1, 52-00-00) to the cleaned seal contact area.

g. When the adhesive is tack free, install the seal on the keel in the positions specified above.

h. Use cleaner (10, Chart 1, 52-00-00) to remove any excess adhesive from the door seal edges.

i. Allow the adhesive to cure in accordance with the manufacturer's specifications.

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Beechcraft
STARSHIP 1
MAINTENANCE MANUAL

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

This chapter describes the fuselage which is comprised of compartments for the crew, passengers, cargo and equipment. Covered in this Chapter are the fuselage main frame (shell), auxiliary structure, access openings, attach fittings and aerodynamic fairings.

The fuselage is of composite construction and fabricated of woven graphite fibers laminated over a honeycomb nomex core, laid up in a mold and cured in an autoclave. The fuselage consists of four basic sections. The nose section extends from the radome to the forward pressure bulkhead (FS 95.00). The radome is mounted on the forward end of the nose section. The cockpit section and the cabin section extend from the forward pressure bulkhead to the aft pressure bulkhead dome (FS 416.00). The cockpit and cabin sections are pressurized areas of the fuselage. The aft end of the cabin section is designed as the baggage compartment above the wing carry-through spars. The aft fuselage (tail section) is spliced at the aft pressure bulkhead. A ventral stabilizer is mounted under the aft end of the tail section. Internal keels and longerons run the length of the lower cockpit and cabin. The keels support the left and right inboard seat tracks and flooring. The longerons support the outboard seat tracks and flooring.

A single piece main entrance door is located on the left side of the fuselage at FS 180.00. An emergency exit door is located on the right side of the fuselage at FS 305.00. For information concerning the doors, refer to Chapter 52. Windows are incorporated on both sides of the fuselage. For information concerning the windows, refer to Chapter 56.

Moisture drain holes are incorporated along the lower surface of the fuselage. The drain holes, located under the cockpit and cabin, have seals to prevent the loss of pressurization. After depressurization, the drain holes open to allow moisture buildup to drain overboard. For maintenance of the moisture drain holes, refer to Chapter 53-10-00.

The cockpit and cabin flooring consists of panels constructed of fiberglass and honeycomb composite with ground plane expanded foil, which are reinforced with aluminum strips around the edges. The ground strips are bonded during co-cure manufacturing operations. For maintenance of the floorboards, refer to Chapter 53-20-00.

Access doors and panels are located on the nose section and aft fuselage (tail section). The left and right nose access panels are load-carrying panels and must be secured prior to towing, taxiing or moving the airplane. Seals and gaskets for the avionics compartment doors and nose access panels are bonded in place around the outer edge of the door or panel to prevent moisture from entering a compartment or area. For maintenance of the fuselage access doors and panels, refer to Chapter 53-30-00.

The attach fittings on the fuselage provide a means of attaching major components such as the nose landing gear, forward fuel tanks, aft wing and the ventral stabilizer. The nose landing gear is supported with trunnion fittings in the left and right nose keels. The forward fuel tanks are attached to the fuselage at FS 237.00 with left and right attach fittings. The aft wing is secured to the fuselage fittings located at FS 324.00, FS 357.00, FS 380.00 and FS 416.00. The ventral stabilizer is secured to the aft fuselage at a canted bulkhead and a "U" shaped fitting at FS 496.00. For description and operation of the attach fittings, refer to Chapter 53-40-00.

Aerodynamic fairings are installed to provide airflow smoothness between the aft wing/forward fuel tanks and the fuselage, and between the forward wing and the fuselage nose section. The radome forms the nose of the airplane, and while protecting the radar antenna, it permits radar transmission. The radome

incorporates five diverter strips for lightning strike purposes. For maintenance of aerodynamic fairings, refer to Chapter 53-50-00.

RECOMMENDED MATERIALS (Effectivity: All)

The recommended materials listed in Chart 1 as meeting federal, military or vendor specifications are provided for reference only and are not specifically recommended by Beech Aircraft Corporation. Any product conforming to the product listed may be used. The products included in this chart have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor or by compliance with the applicable specification. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included herein. Only the basic number of each product specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
RECOMMENDED MATERIALS (Effectivity: All)

MATERIALS	SPECIFICATION	PRODUCT	VENDOR
1. Cleaner	TT-M-261	MEK	
2. Cloth	CCC-C-440 Class II	Cheesecloth	
3. Adhesive	MIL-A-5092	EC847	Minnesota Mining and Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144
4. Adhesive		EC1368	Minnesota Mining and Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144
5. Double Tack Back Tape		445	Minnesota Mining and Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144
6. Cleaner	TT-N-95	Naphtha	
7. Adhesive		EC1403	Minnesota Mining and Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144
8. Adhesive	MMM-A-121	EC1300L	Minnesota Mining and Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144
9. Sealer	MIL-S-7502	EC801	Minnesota Mining and Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144

MAIN FRAME - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The fuselage main frame is constructed without frames and stringers. The fuselage (shell) is of composite construction and reinforced with bulkheads, keels, and longerons. The main bulkheads are the forward pressure bulkhead (FS 95.00) and a series of aft pressure bulkheads at FS 357.00, the baggage shelf and the dome (FS 416.00). The forward pressure bulkhead (FS 95.00) is sealed to hold cockpit and cabin pressurization, and is a support structure for loads imposed by the nose landing gear. The aft pressure bulkhead extends between the bulkhead at FS 357.00 and the bulkhead at FS 416.00. The bulkhead at FS 357.00 encloses the lower half of the fuselage and the bulkhead at FS 416.00 encloses the upper half of the fuselage. A baggage shelf is installed and sealed between the two bulkheads. The upper bulkhead at 416.00 is shaped as a dome.

The keels in the nose section form the left and right sides of the nose gear wheel well and support the nose landing gear left and right trunnion fittings. The left and right nose landing gear doors hinge on the keels. The cockpit/cabin keels are webs fabricated of composite with embedded expanded foil (ground plane). The keels extend along left and right BL 8.50 to FS 355.50. The keels are secured with angles along the interior fuselage shell, and form the cabin center dropped aisle. The keels support the inboard seat tracks (rails) and the floorboards.

Longerons are fabricated of composite and reinforced with bonded titanium straps. The longerons are utilized along the interior fuselage shell for support of the outboard seat tracks and floorboards. In the cabin, the longerons extend from FS 169.75 to FS 355.45. The windshield opening is reinforced with longerons and frames supporting the windshield acrylic panels. Refer to Chapter 56 for maintenance of the windshield.

Moisture drain holes are located in the lower surface of the fuselage where fluids may accumulate. In non-pressurized areas of the fuselage, the moisture drain holes are open at all times. In the pressurized areas of the fuselage, nine moisture drain holes have seals that close at 1-psi during cockpit/cabin pressurization and are open after depressurization. The moisture drain holes are located as follows:

a. Nose Section - FS 93.00, LBL 8.50, and RBL 8.50.

b. Cockpit Section - FS 157.00, LBL 8.50, BL 0.00 and RBL 8.50.

c. Cabin Section - FS 168.50, LBL 8.50, BL 0.00 and RBL 8.50.

d. Cabin Section - FS 280.00, LBL 8.50, BL 0.00 and RBL 8.50.

CAUTION

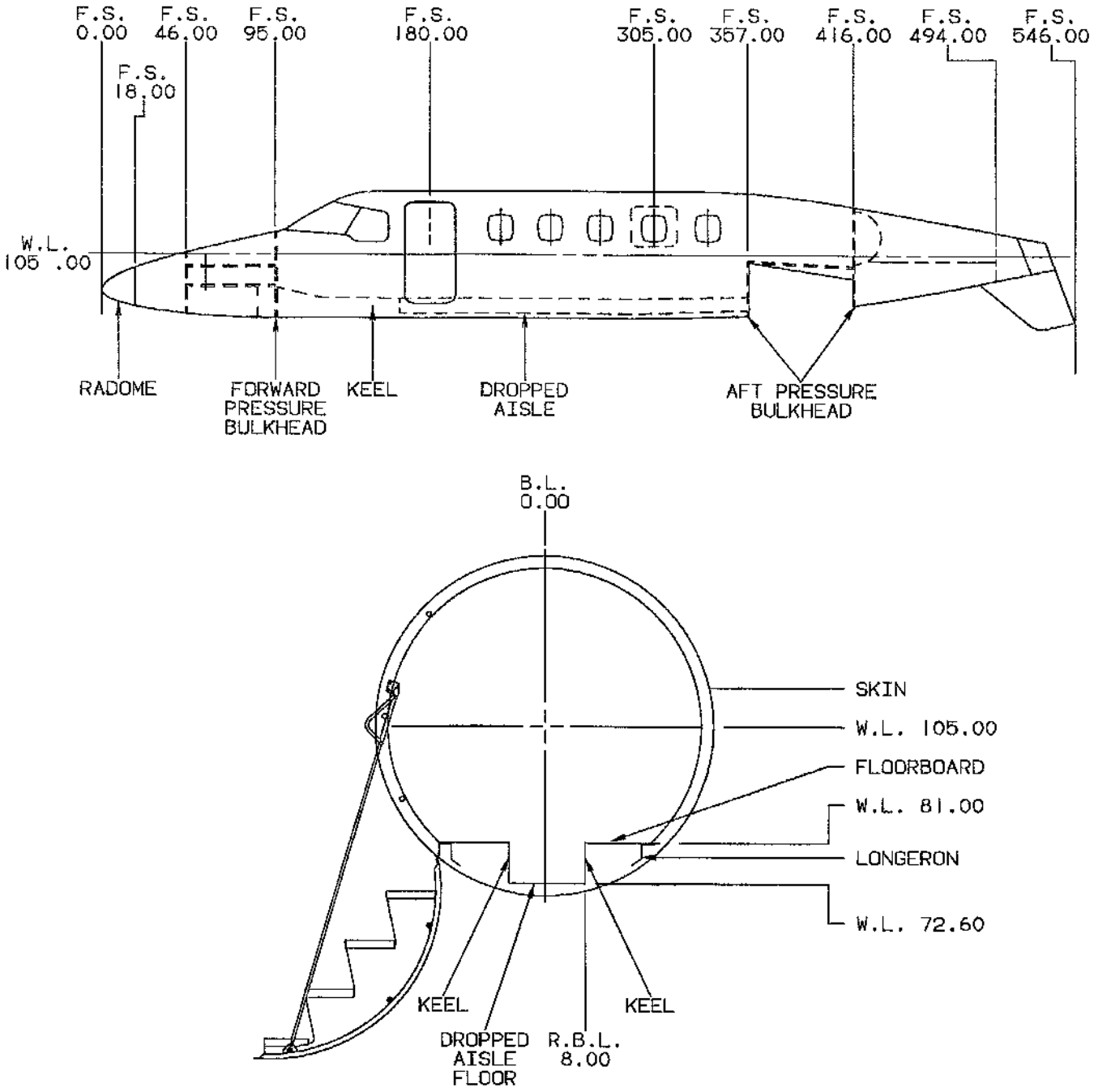
Do not allow solvents to contact the moisture drain seals. Solvents will cause deterioration of the drain seals and affect cabin pressurization.

MAIN FRAME - MAINTENANCE PRACTICES (Effectivity: All) (Figure 2)

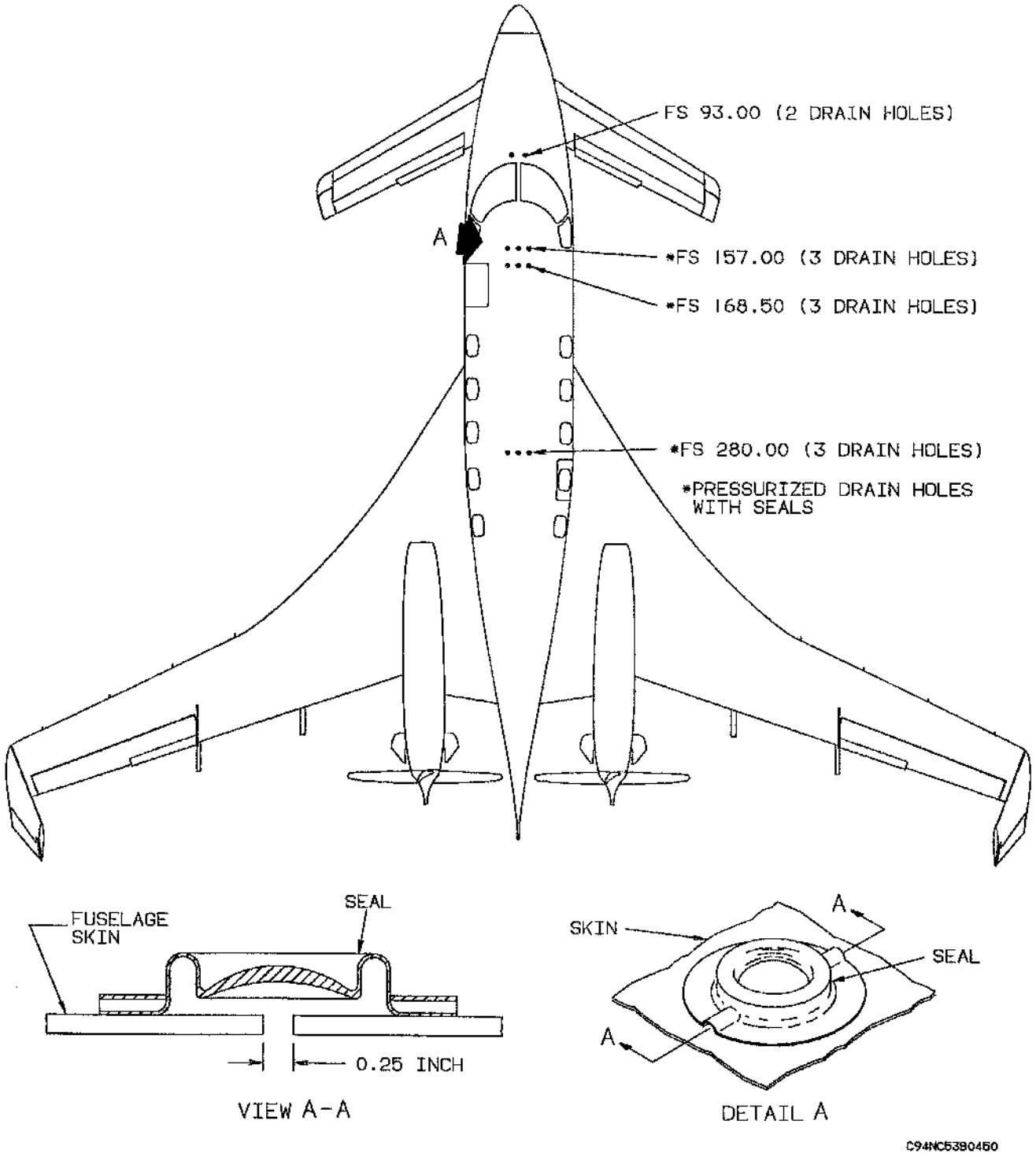
The moisture drain holes are 0.25-inch diameter through the lower composite skin. The drain seals are bonded with adhesive. A blocked or obstructed drain hole may be caused by dirt, debris or foreign objects. All moisture drain holes should be kept clear.

CAUTION

The insertion of sharp objects into the drain hole through the fuselage skin may puncture/damage the seal.



Fuselage Main Frame (Effectivity: All)
Figure 1



Fuselage Moisture Drains (Effectivity: All)
Figure 2

To clear an obstructed drain hole from outside of the fuselage, carefully insert a blunt wire into the drain hole. When a moisture drain hole/seal becomes clogged frequently, action should be taken to clear and clean the inside area around the drain hole.

Gain access to the moisture drain hole seals inside the cockpit area at FS 157.00 by removing the cockpit floorboard(s) 141CT, 143ATC and/or 142CT. Gain access to the moisture drain hole seals inside the cabin area at FS 168.50 by removing the cabin floorboard(s) 151AT, 163ATC and/or 152CT. For drain hole seals at FS 280.00 remove cabin floorboard(s) 161CT, 163ATC and/or 162CT.

MOISTURE DRAIN SEALS REPLACEMENT (Effectivity: All)

a. Gain access to the moisture drain seal to be replaced by removing the applicable cockpit/cabin floorboard. Refer to Chapter 53-20-00.

b. Carefully separate the rubber seal from the internal skin using a thin blade non-metallic tool to break the bond.

c. Clear the drain hole and clean the skin surface of old adhesive. Ensure that the surface around the drain hole is clean.

d. If necessary, scuff sand the surface with 180 grit sandpaper to remove any glaze on the surface. Do not cut or break the composite fibers.

e. Wipe the surface with Methyl Ethyl Ketone (1, Chart 1, 53-00-00) using clean cheesecloth (2, Chart 1, 53-00-00). Repeat as necessary until surface is free of dust, oil, finger prints, and other contamination. A clean dry wipe should follow before the solvent is allowed to dry.

f. Apply a coat of adhesive (3, Chart 1, 53-00-00) to the skin surface and the new drain seal. Permit the adhesive to become very tacky without transferring to the knuckle when touched.

g. Position and center the new drain seal over the drain hole and press in place. Allow to cure.

h. Reinstall the floorboards. Refer to Chapter 53-20-00.

AUXILIARY STRUCTURE - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

Cockpit flooring consists of panels fabricated of fiberglass honeycomb core composite with ground straps embedded along the outer edges of the panels. Four seat tracks (rails) extend from FS 134.00 to FS 163.00 along the L/RBL 8.50 and L/RBL 22.50. One center floorboard spans between the inboard seat tracks aft of the pedestal. A cover is installed below the aft portion of the pedestal extension. There are three floorboards on the left side and three on the right side of the cockpit. Supporting the floorboards are the keels under the inboard seat tracks and support assemblies along the interior shell of the cockpit.

The cabin flooring has a center dropped aisle formed by the two keels. The keels are covered on the inboard sides with drop panels and aisle light assemblies. The inboard seat tracks are secured on top of the keels. Six floorboards are located on each side of the center aisle with a step at the aft end of the aisle. The floorboards are fabricated of fiberglass and honeycomb core composite. The floorboards are secured to longitudinal and lateral supports.

The cabin center aisle floorboards are fabricated of fiberglass and honeycomb core composite. Stiffeners and pads under the center aisle area support the floorboards. The floorboards are secured through inserts. A step at the aft end of the center aisle is constructed with a drop (vertical) panel bonded on the forward end of the step (horizontal) panel with angles. Four screws in the corners and two screws in the center of the vertical panel secure the step assembly.

An aisle container is mounted at the aft end of the center aisle. The container cover slides aft to open and is latched in the closed position with a paddle type latch. The container is secured with eight screws through spacers to the left and right keels from inside the container. Carpet is bonded over the cover and the forward face of the container with adhesive (4, Chart 1, 53-00-00). The aisle container must be removed when removing the center aisle floor panel and/or step assembly.

The cockpit and cabin are separated by a partition (divider). The left divider is positioned at the main entrance door frame behind the pilot's seat. The right divider incorporates a map case behind the copilot's seat and a cabinet on the aft side of the divider. A sliding door between the left and right dividers closes off the cockpit. The dividers are secured with brackets and seat tracks. Depending on the configuration, the cabinet may be designed as a closet, toilet or both.

AUXILIARY STRUCTURE - MAINTENANCE PRACTICES (Effectivity: All) (Figures 1 and 2)

The cockpit floorboards are fabricated composite panels with expanded foil for static discharge and lightning strike protection. The cockpit floorboards are secured with stud and floating receptacle fasteners. The cabin floorboards are secured with screws into nutplates. The studs are held in the cockpit floorboard with retaining rings and align with floating receptacles which are riveted to the floor supports. The studs are rotated 1/4 turn to remove and/or install the floorboards.

The floorboards are covered with carpet which is trimmed around the pedestal and beveled at the seat tracks. The carpet has cutouts for rudder pedals, emergency landing gear handle, and the relief tube in the cockpit. Two-inch double tack tape (5, Chart 1, 53-00-00) is utilized to secure the carpet to the floorboards. Refer to Chapter 25.

Removal and installation of the cockpit and cabin floorboards is typical except for the fasteners securing the panels. The cockpit floorboards are secured with 1/4 turn of studs into receptacles, and the cabin floorboards are secured with screws into nutplates.

COCKPIT/CABIN FLOORBOARD REMOVAL (Effectivity: All)

a. When necessary, remove the seat that may interfere with the removal of the floorboard. Refer to Chapter 25.

b. Gain access to the stud/screw fasteners by separating the carpet and the double tack tape around the edges and cutouts of the floorboard.

c. For the cockpit floorboard, turn each stud fastener 1/4 turn counterclockwise.

d. For the cabin floorboard, remove the screws.

e. Lift and remove the floorboard.

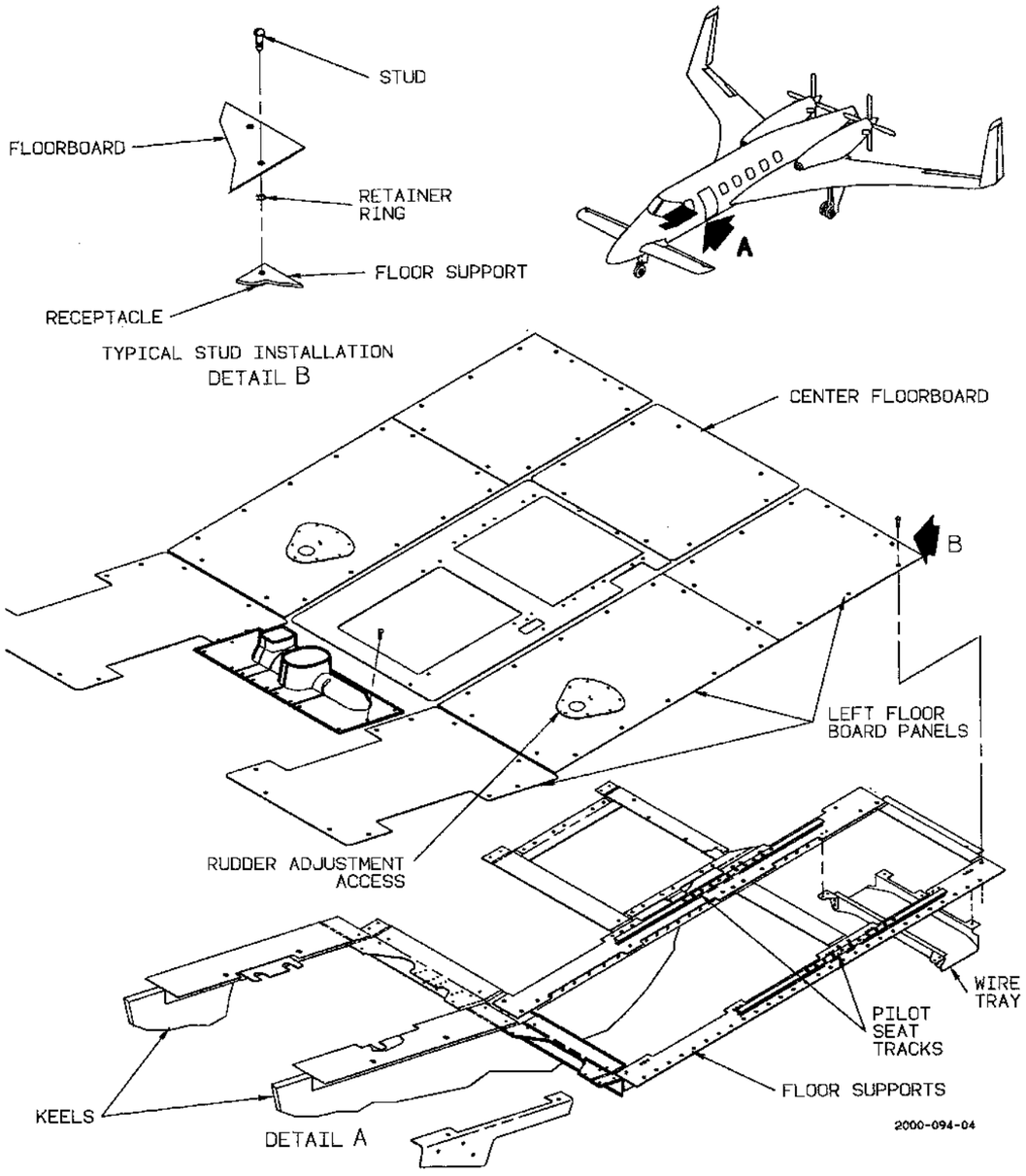
COCKPIT/CABIN FLOORBOARD INSTALLATION (Effectivity: All)

a. Place the cockpit floorboard in position and secure with stud fasteners.

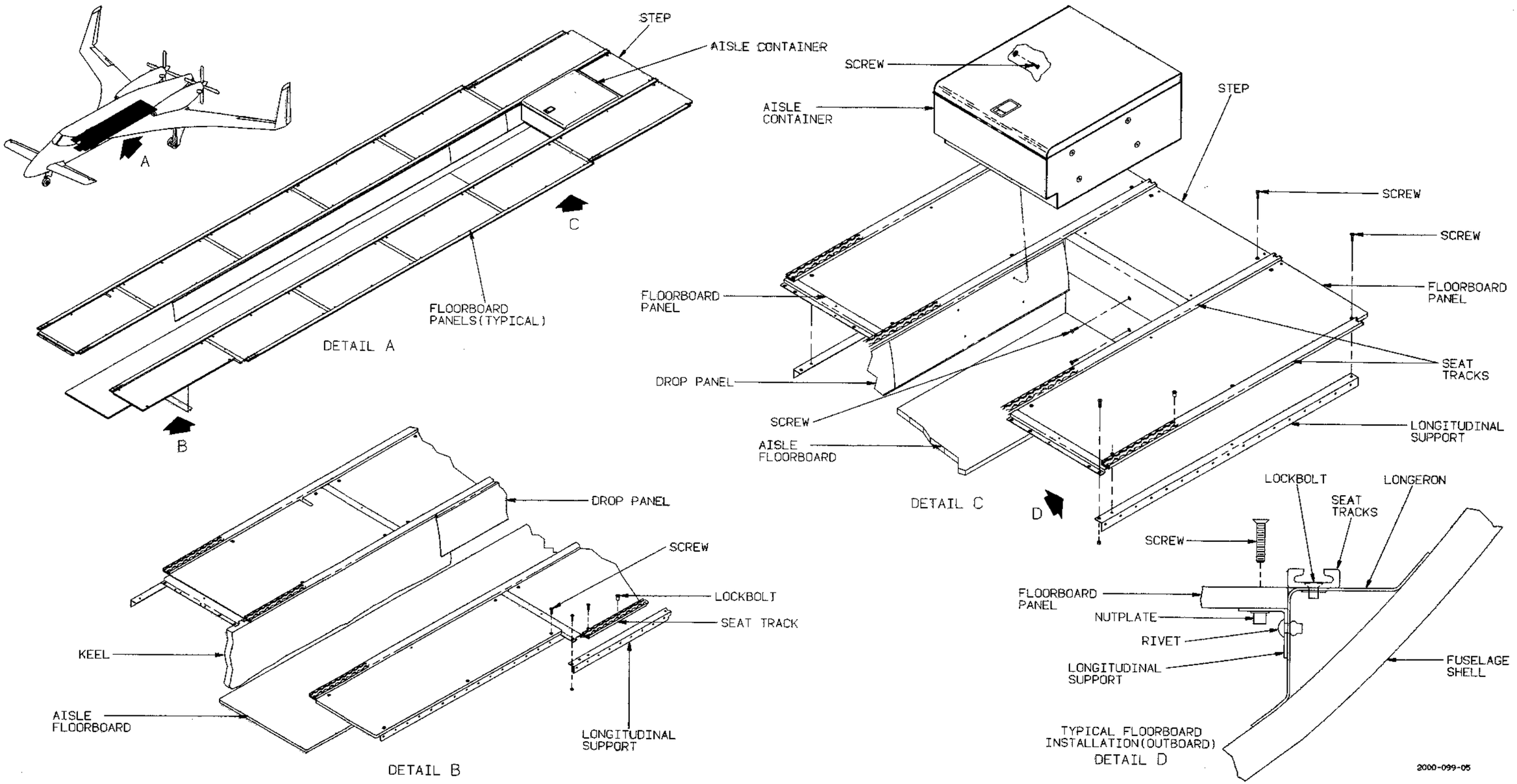
b. Place the cabin floorboard in position and secure with screws.

c. Check the double tack tape for its adhesion effectiveness and replace if necessary. Press the carpet against the double tack tape around the edges/cutouts of the floorboard.

d. When applicable, install the crew/passenger seat. Refer to Chapter 25.



Cockpit Floorboards and Supports (Effectivity: All)
Figure 1



Cabin Floorboard Installation (Effectivity: All)
Figure 2

ACCESS DOORS AND SKIN - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

This section describes the exterior fuselage structure (shell/skin) and the fuselage access doors/panels.

The exterior fuselage (shell/skin) is engineered utilizing advanced composite materials to form a rigid monocoque structure. The structure consist of a nomex honeycomb core bonded between layers of graphite fabric and cured in an autoclave. The fuselage access doors and panels are fabricated using the same advanced composite techniques as the fuselage structure, except kevlar fiber material is utilized in selected areas to increase the damage tolerance of the component.

CAUTION

The left and right nose skin panels are load-carrying panels. THE NOSE SKIN PANELS ARE REQUIRED TO BE INSTALLED AND SECURED PRIOR TO TOWING, TAXIING AND/OR MOVEMENT OF THE AIRPLANE. Refer to Chapter 9 for towing and taxiing the airplane.

The left and right avionics compartment doors allow access to the avionics and other components in the nose section. Door seals are bonded on the access doors to provide a weather-tight seal when compressed by the doors. The doors are secured with fasteners along the edge of each door.

Nose skin panels are located on the left and right nose section below the forward wing. The skin panels are secured with captive screws along the edge of the panels. The left nose skin panel is removed for access to the anti-skid brake system components. The oxygen system components are behind the right nose skin panel. Access to the oxygen service valve is through an access door on the right nose skin panel. Provision is made for installation of an ice detector on each skin panel at FS 88.00. Drain tubes are bonded through the left and right nose skin panels and scarfed aft. Connected to the drain tube is the anti-skid pump seal drain line (left side), and the oxygen overboard vent line (right side).

NOTE

The air conditioner service door is located on the left side of the aft fuselage (tail section). Refer to Chapter 52-40-00 for removal and installation of the service door.

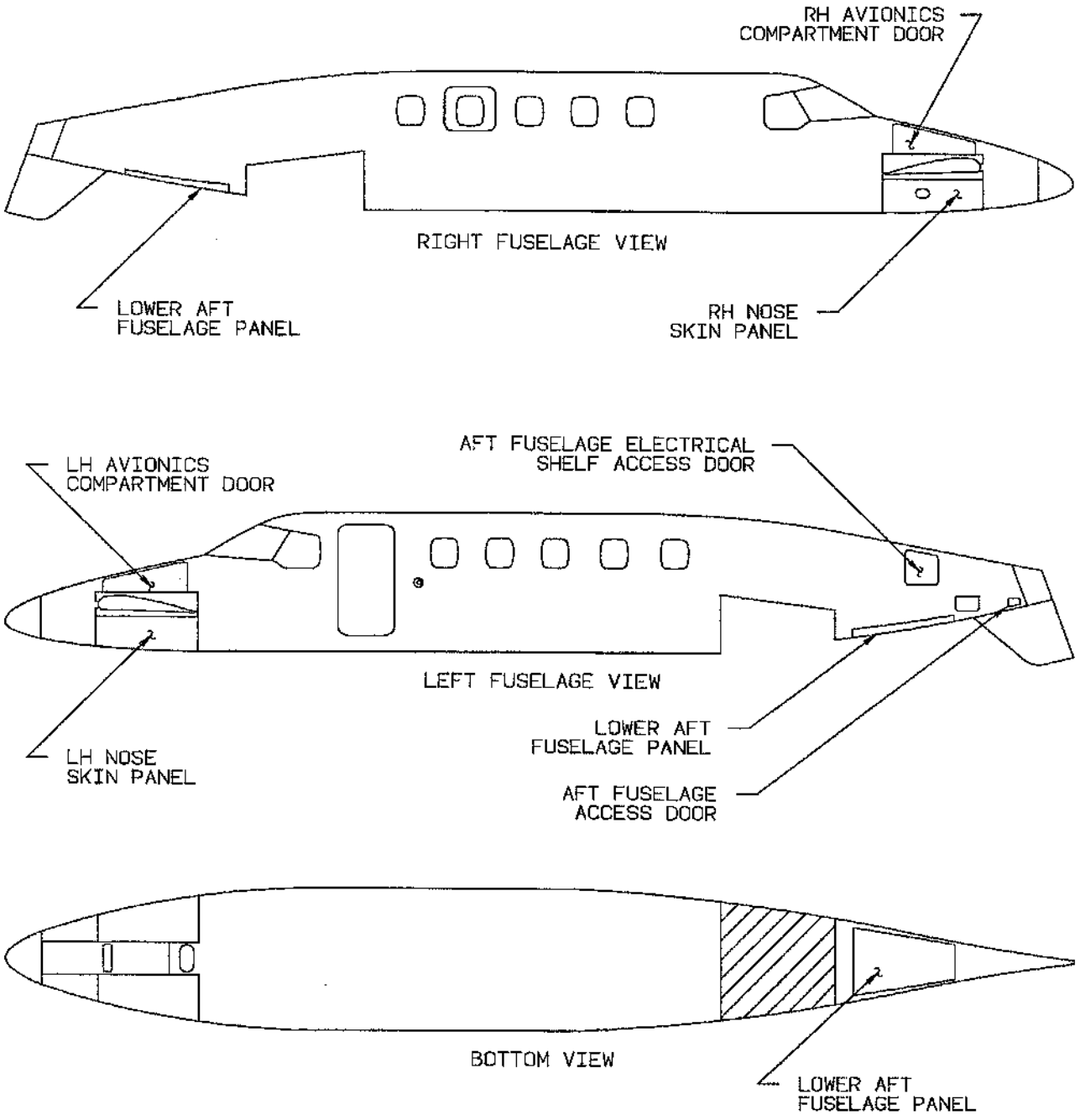
The aft fuselage access door is located on the left side of the aft fuselage near the tailcone. The access door is secured with captive screws along the edge of the door. A seal is bonded to the inside edge of the door.

The electrical shelf access door is located on the left side of the aft fuselage. The access door is secured with captive screws, and a seal is bonded to the inside edge of the door. Removal of the access door allows access to the electrical components located on the aft fuselage electrical shelf.

ACCESS DOORS AND SKIN - MAINTENANCE PRACTICES (Effectivity: All)

Maintenance of the fuselage skin is mainly preventive maintenance. Care should be taken to preserve the exterior finish of the fuselage and protect the skin from tool scratches, gouges and impact damage. Damage to the skin resulting in exposure of the honeycomb nomex core must be checked for moisture and repaired as soon as possible. For repair of the composite fuselage structure, refer to the BEECHCRAFT Starship 1 Structural Repair Manual.

Maintenance for most of the fuselage access doors/ panels is typical. Design in size, shape, type of seal and number of fasteners varies.



Access Doors and Skin Panels (Effectivity: All)
Figure 1

AVIONICS COMPARTMENT DOOR REMOVAL (Effectivity: All)
(Figure 2)

NOTE

Removal of the left and right avionics compartment doors is typical.

a. Unscrew the stud nuts securing the door.

b. Carefully remove the avionics compartment door.

c. Store the compartment door to protect it from damage.

AVIONICS COMPARTMENT DOOR INSTALLATION (Effectivity: All)
(Figure 2)

NOTE

Installation of the left and right avionics compartment doors is typical.

a. Check the door seal around the nose compartment opening for deterioration, cuts and security. Replace a defective seal. Refer to AVIONICS COMPARTMENT DOOR SEAL REPLACEMENT, this section.

b. Position the avionics compartment door to the compartment opening, aligning the stud nuts with their receptacles.

c. Screw the stud nuts into their receptacles.

AVIONICS COMPARTMENT DOOR SEAL REPLACEMENT (Effectivity: All)

NOTE

The left and right avionics compartment door seals are bonded to the edge of the compartment access door. Each access door has a two piece seal. One piece is bonded along the lower edge and the other along the sides and upper edge.

a. Use a non-metallic scraper to remove the door seal from around the edge of the door.

b. Remove any excess adhesive that might remain on the bonding surface, using a lint-free cheesecloth (2, Chart 1, 53-00-00) moistened with Methyl Ethyl Ketone or Naphtha (1, 6, Chart 1, 53-00-00).

c. With a brush apply a thin even coat of adhesive (7, Chart 1, 53-00-00) to each surface to be bonded. Allow to dry until the adhesive is tacky and will not transfer to the knuckle when touched.

d. Position the upper length of seal from the lower forward corner, around the upper edge, to the lower aft corner of the door. Press the seal in place.

e. Position the lower length of seal from the lower forward corner to the lower aft corner of the door. Press the seal in place.

f. Trim the seals in the lower forward and aft corners diagonally and join the seal ends with adhesive (8, Chart 1, 53-00-00).

NOSE SKIN PANEL REMOVAL
(Effectivity: All)
(Figure 3)

CAUTION

The left and right nose skin panels are load-carrying panels. THE NOSE SKIN PANELS ARE REQUIRED TO BE INSTALLED AND SECURED PRIOR TO TOWING, TAXIING AND/OR MOVEMENT OF THE AIRPLANE. Refer to Chapter 9 for towing and taxiing the airplane.

NOTE

Removal of the left and right nose skin panels is typical, except for the oxygen service access door on the right panel.

a. When removing the right nose skin panel, first remove the oxygen service access door, refer to Chapter 52-40-00.

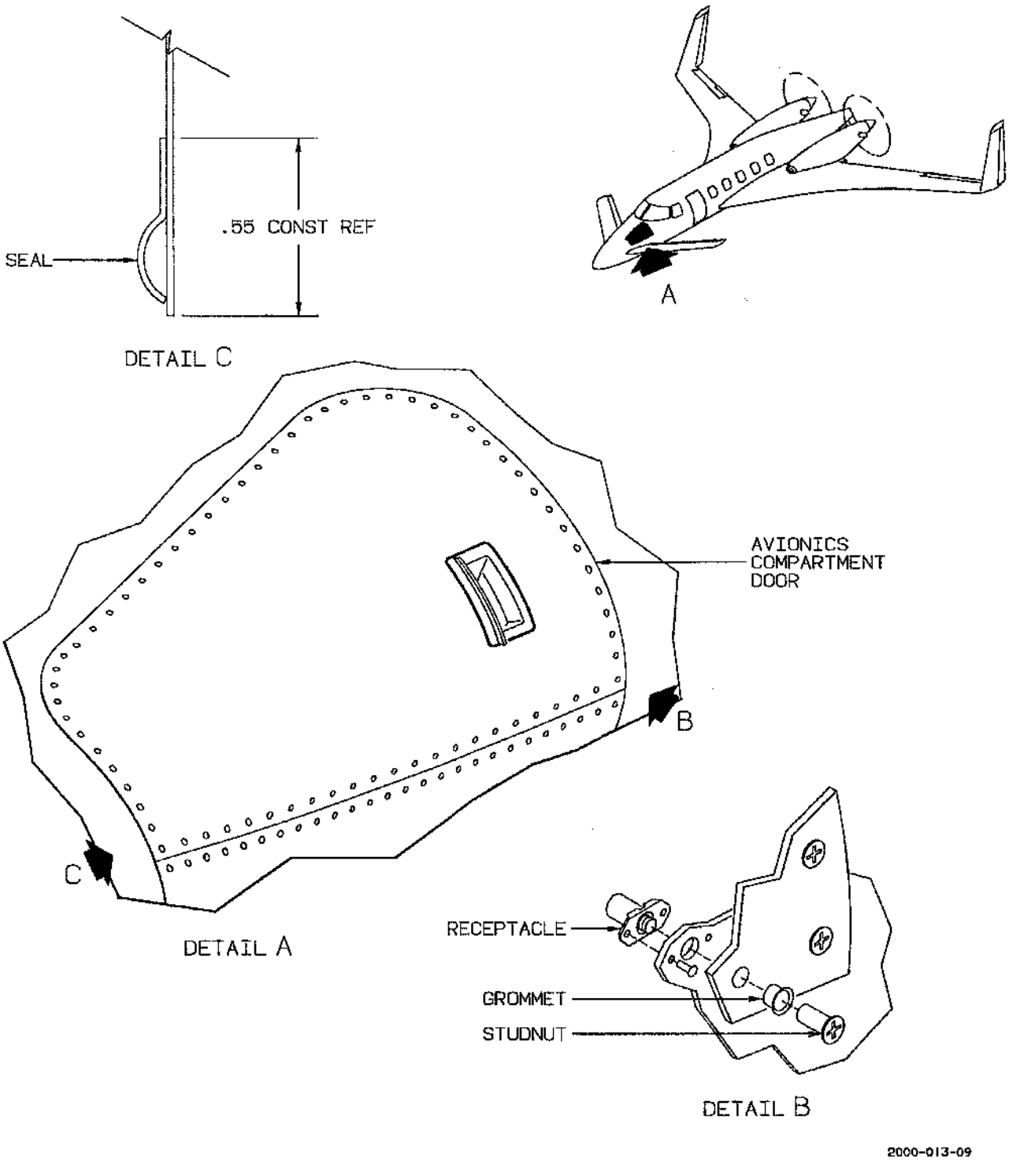
b. Unscrew the captive screws securing the nose skin panel. Lower the panel far enough to disconnect the ice detector electrical wiring and disconnect the drain from the drain tube bonded to the panel by removing the tube clamp.

c. Store the nose skin panel to prevent damage.

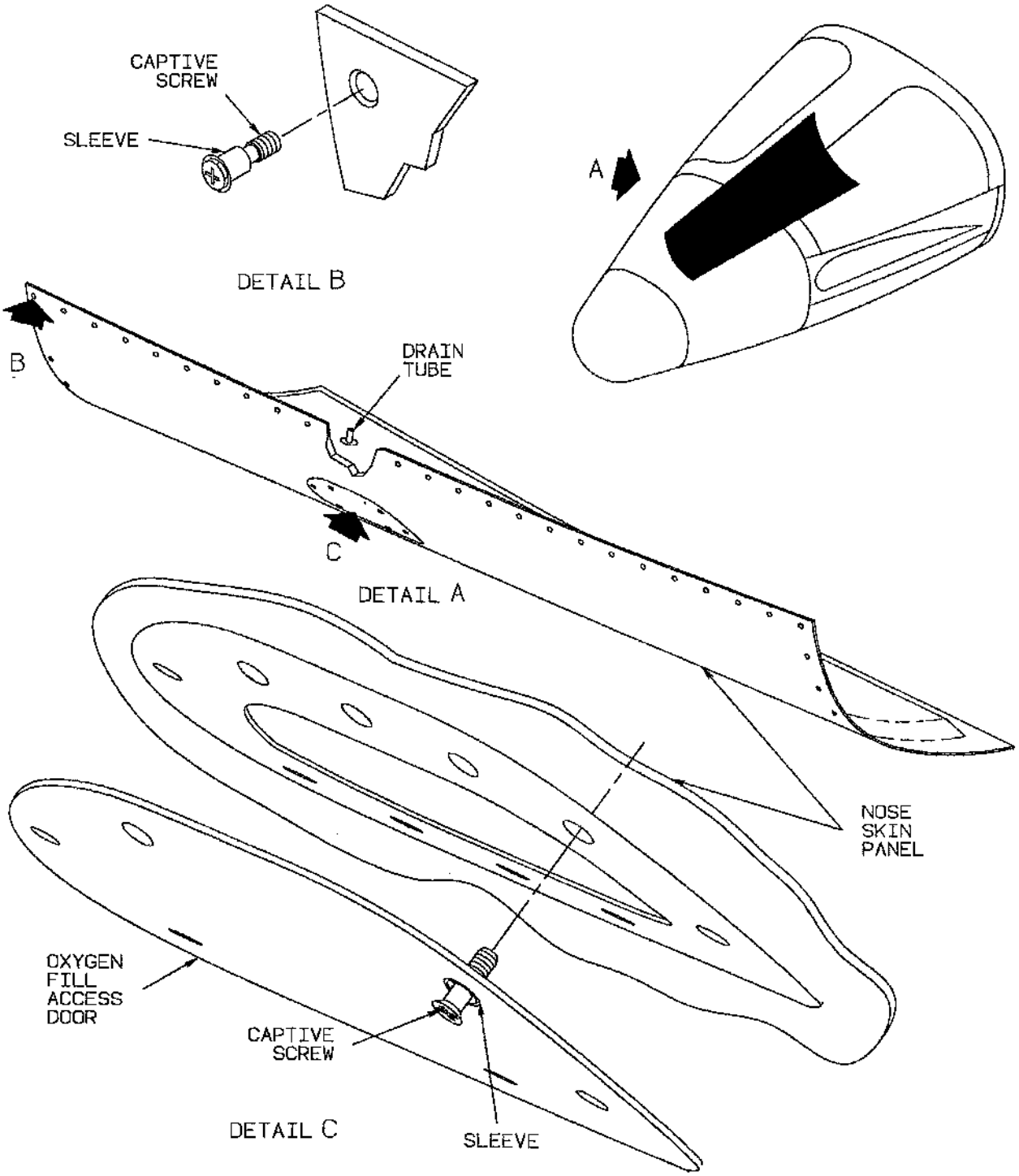
NOSE SKIN PANEL INSTALLATION
(Effectivity: All)
(Figure 3)

NOTE

Installation of the left and right nose skin panels is typical, except for the oxygen service access door on the right panel.



Avionics Compartment Door Installation (Effectivity: All)
Figure 2



CAUTION: THE LEFT AND RIGHT NOSE SKIN PANELS ARE LOAD CARRYING PANELS AND MUST BE INSTALLED AND SECURED PRIOR TO TOWING THE AIRPLANE. 2000-091-21

Nose Skin Panel Installation (Effectivity: All)
Figure 3

a. For the right nose skin panel: Check the seal on the oxygen service access door for deterioration, damage and security of bond. For replacement of the oxygen service door seal, refer to Chapter 35-00-00.

b. Place the skin panel near position. Connect the ice detector electrical wiring. Connect the drain to the drain tube bonded to the panel and secure with a clamp.

c. Align the skin panel to the nose section and secure the panel with the captive screws.

d. On the right nose skin panel, install the oxygen service access door. Refer to Chapter 52-40-00.

LOWER AFT FUSELAGE ACCESS PANEL REMOVAL (Effectivity: All) (Figure 4)

NOTE

The lower aft fuselage access panel is secured with captive screws into receptacles, and is sealed with gaskets bonded to the access panel.

a. Unscrew the captive screws securing the access panel to the fuselage.

b. Remove the access panel and store to prevent damage.

LOWER AFT FUSELAGE ACCESS PANEL INSTALLATION (Effectivity: All) (Figure 4)

a. Check access panel gaskets for deterioration, damage and/or security of bond. Replace defective gaskets as required. Refer to LOWER AFT FUSELAGE PANEL GASKET REPLACEMENT.

b. Place the access panel in position, aligning the fasteners with their receptacles.

c. Secure the access panel with the captive screws.

LOWER AFT FUSELAGE PANEL GASKET REPLACEMENT (Effectivity: All)

NOTE

There are four sections of 1/16 inch thick rubber gasket bonded around the inside edge of the access panel.

a. Remove the access panel, refer to LOWER AFT FUSELAGE ACCESS PANEL REMOVAL.

b. Use a non-metallic scraper to remove the gasket(s).

c. Remove any excess adhesive on the bonding surfaces using a lint-free cheesecloth (2, Chart 1, 53-00-00) moistened with Methyl Ethyl Ketone (1, Chart 1, 53-00-00) or Naphtha (6, Chart 1, 53-00-00).

d. Brush a thin even coat of adhesive (7, Chart 1, 53-00-00) on each surface to be bonded. Allow the adhesive to dry until it is tacky and will not transfer to knuckle when touched.

NOTE

When replacing all four gaskets, the forward and aft gaskets are to be installed first. The ends of the gaskets must butt together to ensure a weather tight seal.

e. Position the gasket(s) and press firmly in place with the ends of the gaskets butted together.

f. Install the access panel, refer to LOWER AFT FUSELAGE ACCESS PANEL INSTALLATION.

AFT FUSELAGE ACCESS DOORS REMOVAL (Effectivity: All) (Figure 5)

The electrical shelf access door and the aft fuselage access door, located on the left side of the aft fuselage (tail section), are secured with captive screws and receptacles. Removal of access doors is typical.

a. Unscrew the captive screws securing the access door.

b. Remove the access door and store to prevent damage.

AFT FUSELAGE ACCESS DOORS INSTALLATION (Effectivity: All) (Figure 5)

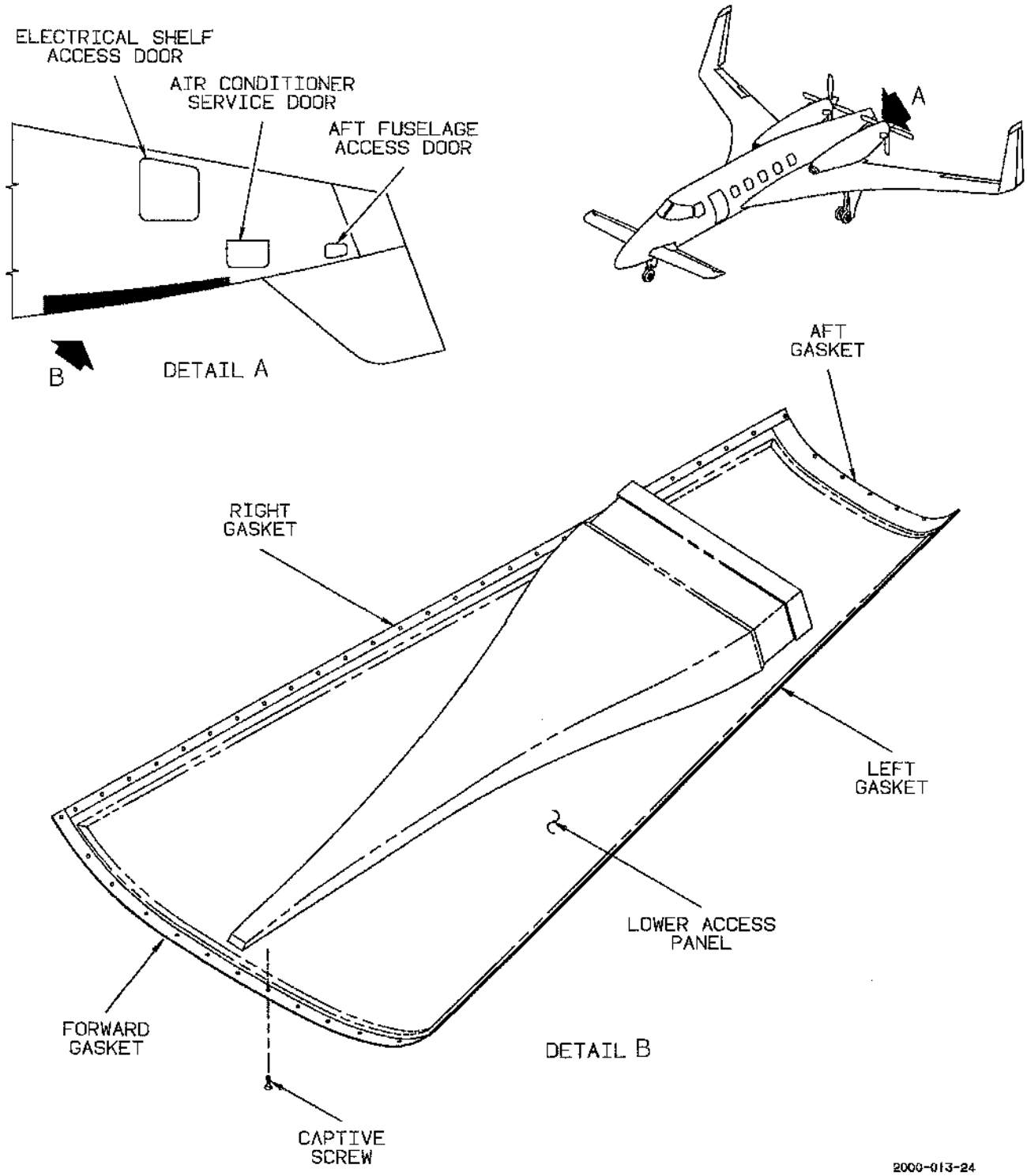
NOTE

Installation of the electrical shelf access door and the aft fuselage access door is typical.

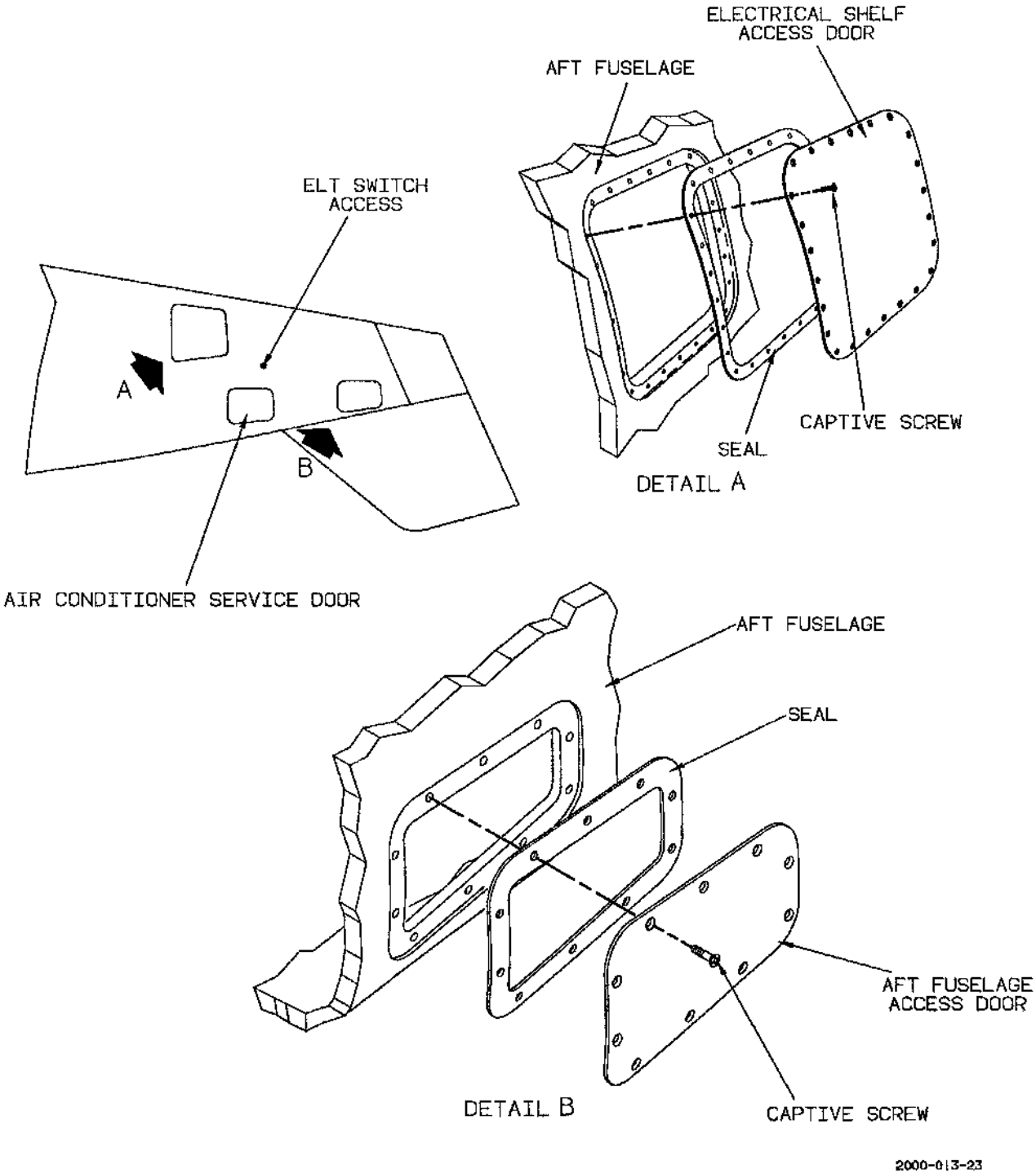
a. Check the access door seal for deterioration, cuts and damage. Replace seal if defective. Refer to AFT FUSELAGE ACCESS DOOR SEAL REPLACEMENT, this section.

b. Place the access door in position.

c. Secure the access door with the captive screws.



Lower Aft Fuselage Access Panel Installation (Effectivity: All)
Figure 4



Aft Fuselage Access Doors Installation (Effectivity: All)
Figure 5

AFT FUSELAGE ACCESS DOOR SEALS REPLACEMENT (Effectivity: All)

NOTE

Replacement of the electrical shelf access door seal and the aft fuselage access door seal is typical.

a. With the access door removed, carefully remove the seal using a non-metallic scraper to separate the seal from the access door.

b. Remove old adhesive from the bonding surface using a lint-free cheesecloth (2, Chart 1, 53-00-00) moistened with Methyl Ethyl Ketone or Naphtha (1, 6, Chart 1, 53-00-00).

c. With a brush, apply a thin coat of adhesive (7, Chart 1, 53-00-00) to each surface to be bonded. Allow adhesive to dry until it is tacky and will not transfer to knuckle when touched.

d. Position the seal on the access door and press firmly in place.

ATTACH FITTINGS - DESCRIPTION AND OPERATION (Effectivity: All)
(Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

This section describes the attach fittings designed to secure major structures/components to the fuselage, such as the nose landing gear trunnion fittings, the aft wing attach fittings, the forward fuel tank attach fittings, and the ventral stabilizer attach fitting.

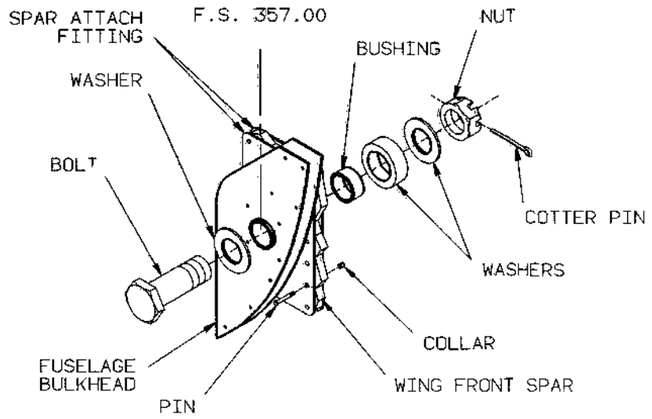
The forward wing attach fittings are located in the nose section. Left and right pivot bearings are secured in the upper and lower nose compartment shelves. The left and right pivots pins rotate in their respective upper and lower pivot bearings allowing the forward wings to rotate thirty degrees aft. Refer to Chapter 55 for maintenance of the forward wings.

The aft wing attach fittings consist of those fittings securing the wing to the fuselage. Included are the fittings securing the forward fuel tanks to the left and right sides of the fuselage at FS 237.00. The fuselage fittings at FS 324.00 provides attachment for the wing at the main landing gear forward spars. The wing front spar and rear spar are secured to the fuselage at the left and right wing bolt locations FS 358.50 and FS 416.00. Two drag shear fittings secure the upper wing surface to the fuselage at FS 380.00. The fuselage halves of the wing attach fittings are secured to the fuselage with adhesive using nylon knit scrim cloth and pin-collar fasteners.

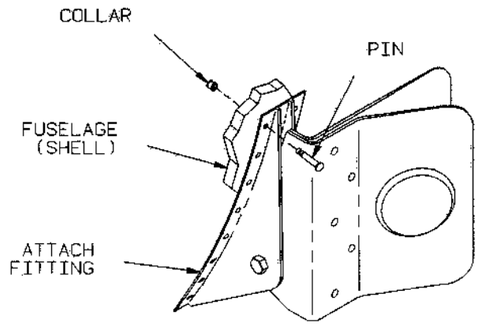
The nose landing gear trunnion pivots between the left and right trunnion fittings mounted in the left and right nose section keels (BL 8.5). Each trunnion fitting incorporates a bearing, shim and trunnion plate. A shim is placed under the fittings for protection of the keel. Fifteen bolts through the fitting, keel and trunnion plate secures the fitting. The center lower bolt of each fitting is a countersink head style to allow the nose landing gear trunnion fitting to slide in place. A bonding strap is secured to the center forward bolt. Refer to Chapter 32 for maintenance of the nose landing gear.

Seat tracks (rails) are located in the cockpit and cabin to support the pilot, copilot and passenger seats. The seat tracks are mounted on, and along the cockpit and cabin keels (inboard) and longerons (outboard). The cockpit seat tracks extend from FS 134.00 to FS 164.00. The cabin seat tracks extend the length of the cabin except at the entrance door. Incorporated in the seat tracks are holes which align with seat pins for forward and aft adjustment and to lock the seat in a selected position. Seat stops are installed to restrict to seat movement to an area of the seat track. The seat tracks are also utilized to secure cabin furnishings such as cabinets.

The ventral stabilizer has a forward "U" shaped fitting (FS 496.00) which secures the ventral stabilizer under the aft fuselage/tailcone. The aft spar of the ventral stabilizer is secured to the canted fuselage bulkhead. A seal is installed on the ventral stabilizer to mate with the fuselage/tailcone. A VHF navigation antenna is mounted on the left and the right sides of the ventral stabilizer. Refer to Chapter 34 for maintenance of the VHF navigation antenna.



DETAIL D



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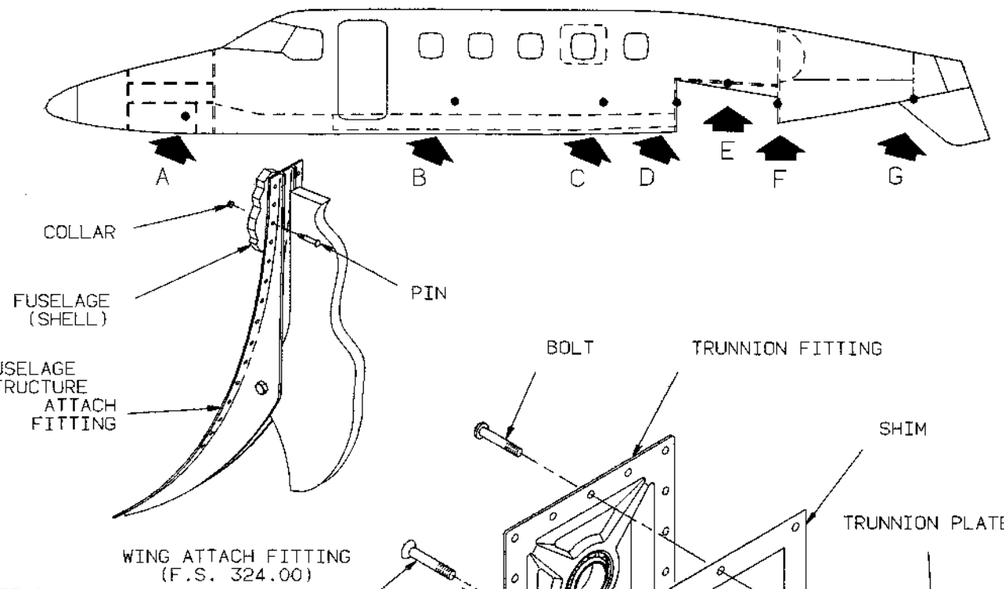

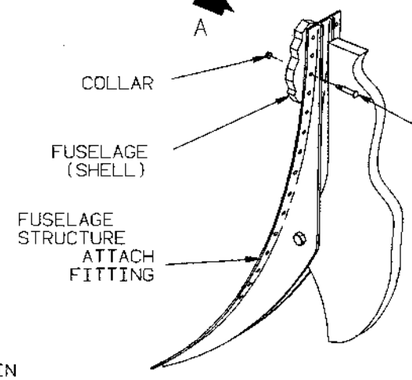

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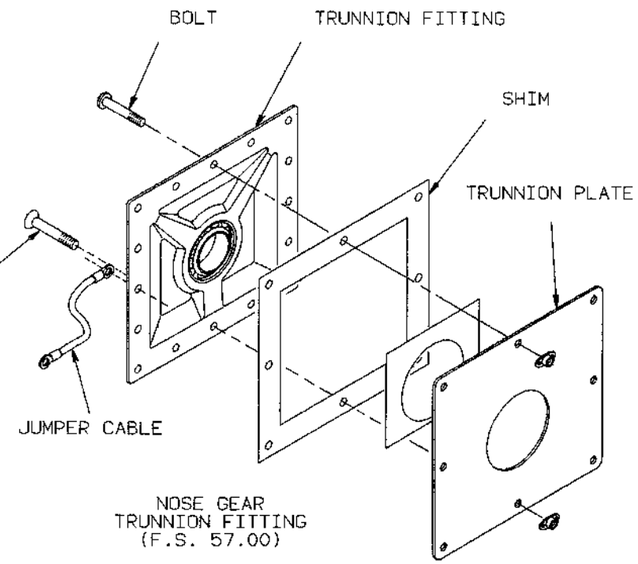
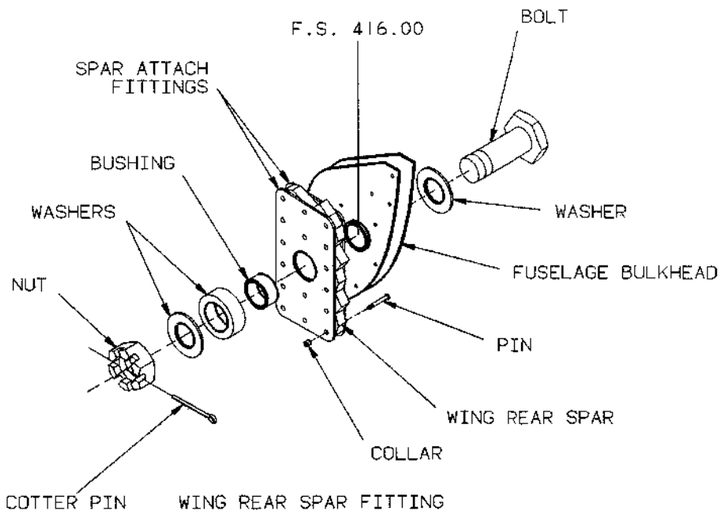
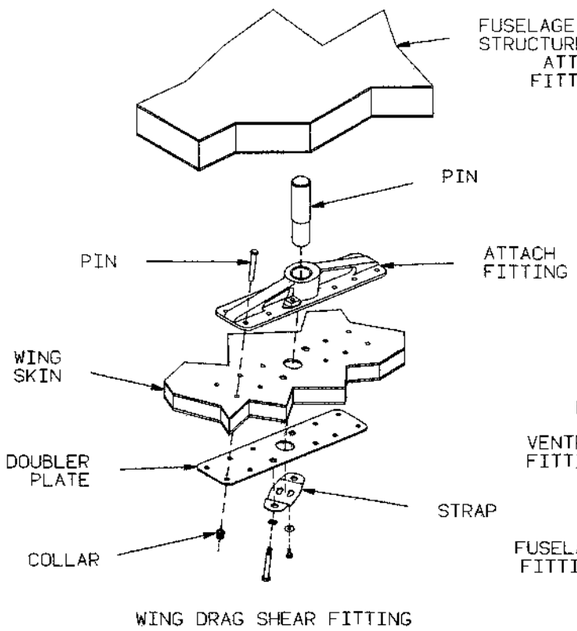


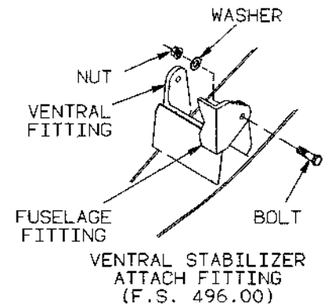
DETAIL A



DETAIL F



DETAIL E



DETAIL G

2000-090-09

Fuselage Attach Fittings (Effectivity: All)
Figure 1

AERODYNAMIC FAIRINGS - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

Aerodynamic fairing are installed between major structures of the airplane to provide smooth airflow and reduce in-flight drag. Aerodynamic fairings include the forward wing fairings, fuselage-to-forward fuel tanks/aft wing fairings, radome and tailcone. The fairings are constructed of composite materials and are secured with fasteners around the outer edge of each fairing.

The forward wing fairings areodynamically form around the upper and lower skin surfaces of the left and right forward wings. The fairings are secured to the fuselage nose section with captive screws.

The left and right fuselage-to-forward fuel tank/aft wing fairing panels extend along each wing root rib to the lower fuselage. In addition to providing an areodynamic surface, the fairings protect fuel lines and electrical wire bundles. A collar assembly is installed to form the forward fuel tank (leading edge) to the fuselage. The forward fuel tank/aft wing is faired to the fuselage with an upper forward seal, upper center seal, upper and lower aft seals. The seals are adhesive bonded to the fuselage/wing surfaces and edge dressed with sealant. An upper aft wing box fairing is located between the fuselage and wing aft cove on each side of the fuselage. The fairings are in two parts which are secured together with fasteners and nutplates.

The radome is fabricated of composite material (epoxy preimpregnated glass/fabric cloth with a core). The radome is mounted on the nose section of the fuselage to protect the radar antenna and aerodynamically streamline the nose. Five lightning diverter strips are incorporated in the radome, along with five captive screw type fasteners. A seal is installed to prevent the entry of moisture at the radome-to-nose section mating surfaces.

The tailcone is fabricated of composite material (woven kevlar) with provision for a navigation light. The tailcone is installed on the aft end of the aft fuselage and is secured with 16 screws and nutplates.

AERODYNAMIC FAIRINGS - MAINTENANCE PRACTICES (Effectivity: All)

Maintenance of the aerodynamic fairings is mainly removal and installation. Removal and installation of the left and right fairings is typical. After installation of the aerodynamic fairings, the fasteners should be touch-up painted for appearance as necessary.

FORWARD WING FAIRING REMOVAL (Effectivity: All) (Figure 1)

NOTE

The left/right upper and lower fairings may be removed individually. Removal of the left/right and upper/lower forward wing fairings is typical.

a. Unscrew the captive screws securing the fairing(s) to be removed.

b. Carefully remove the fairing(s) and store to prevent damage.

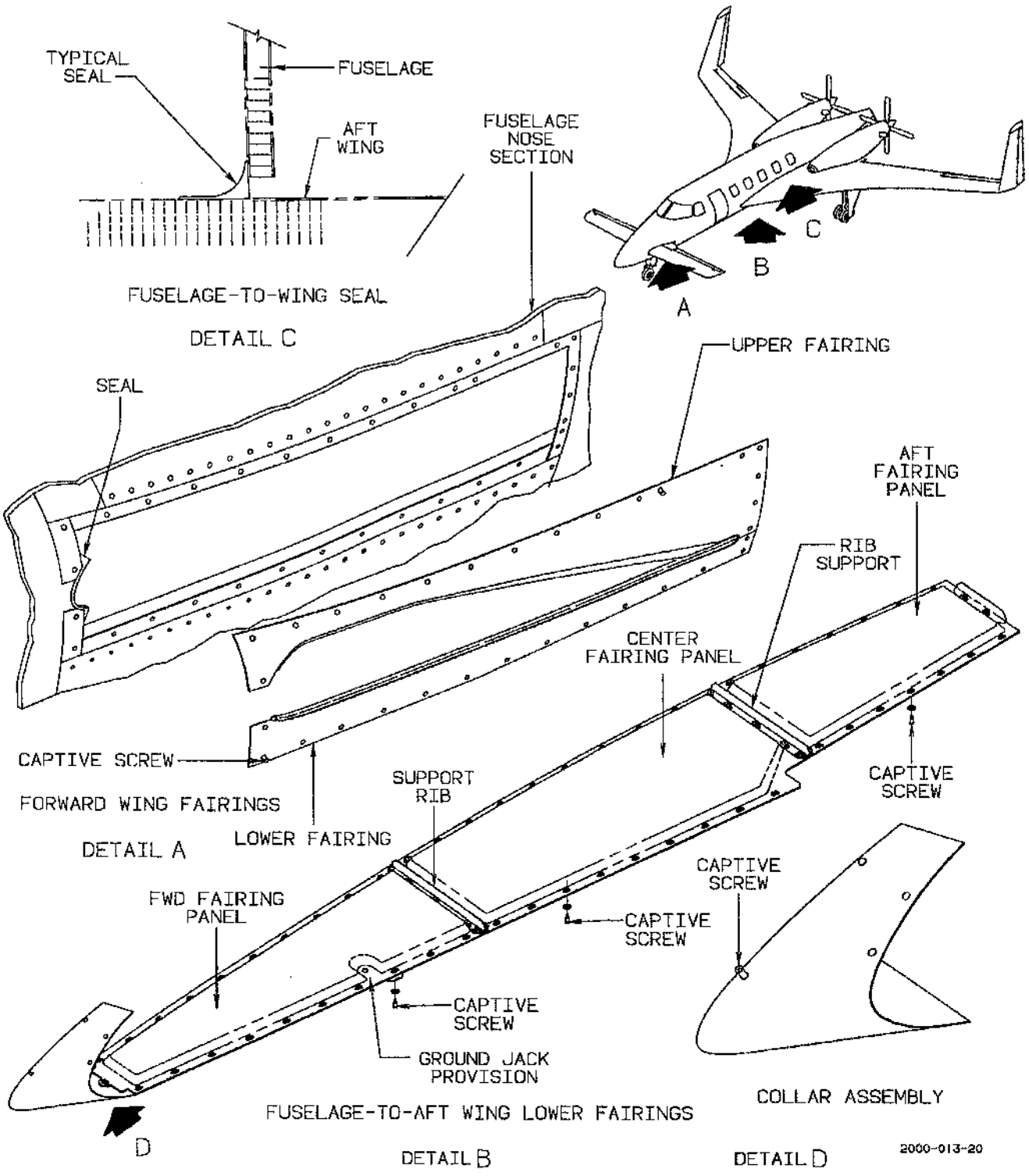
FORWARD WING FAIRING INSTALLATION (Effectivity: All) (Figure 1)

NOTE

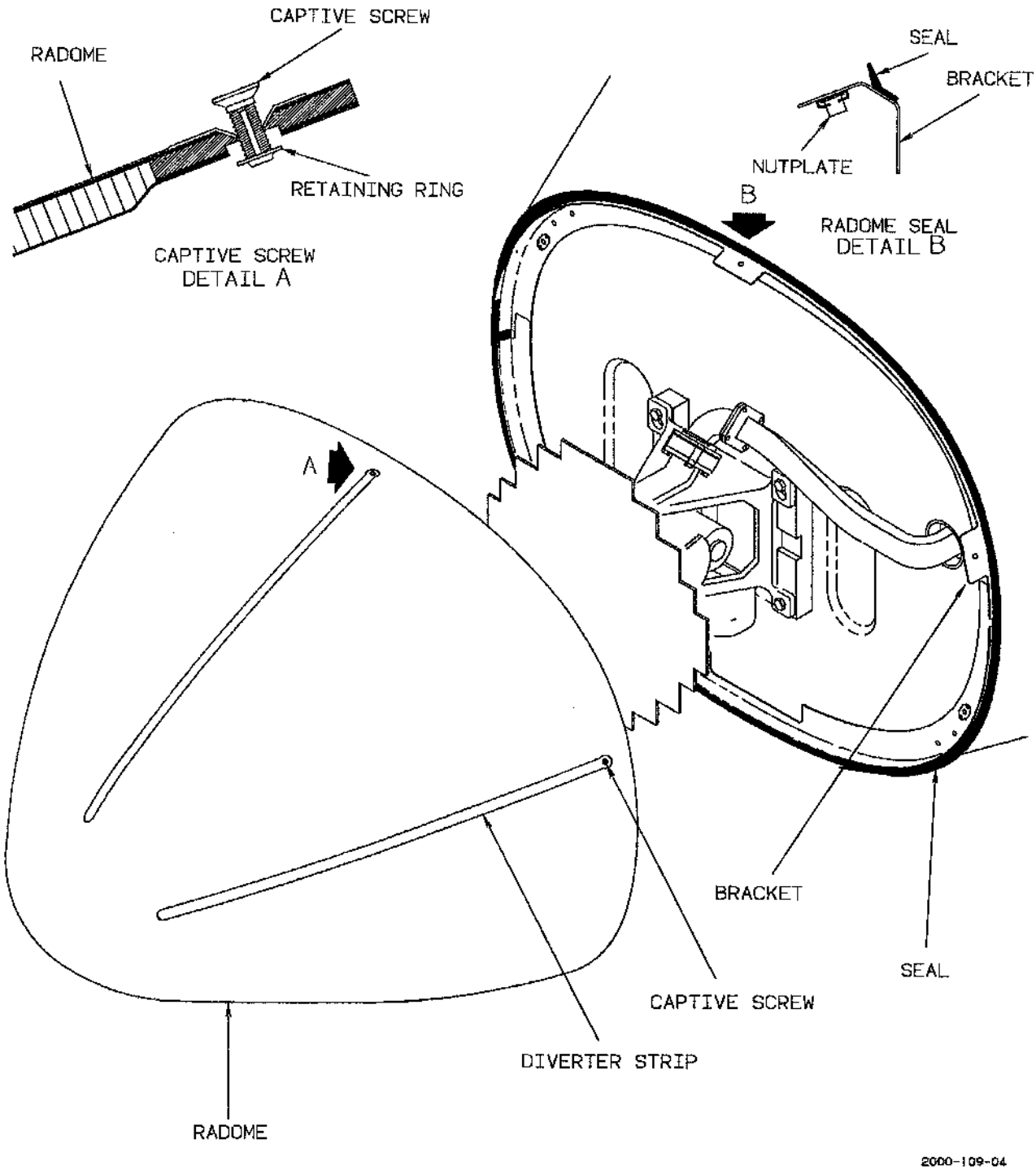
Installation of the left/right and upper/lower forward wing fairings is typical.

a. Identify and place the fairing(s) in position.

b. Secure the fairing(s) with their captive screws.



Aerodynamic Fairings Installation (Effectivity: All)
Figure 1



Radome Installation (Effectivity: All)
Figure 2

FUSELAGE-TO-FORWARD FUEL TANK/ AFT WING FAIRING REMOVAL (Effectivity: All)
(Figure 1)

a. Identify the fairing(s) to be removed and unscrew the captive screws securing the fairings(s).

b. Remove the fairing(s) and store to prevent damage.

FUSELAGE-TO-FORWARD FUEL TANK/ AFT WING FAIRING INSTALLATION (Effectivity: All)
(Figure 1)

a. Identify the fairing to be installed and place the fairing in position as it was originally installed.

b. Secure the fairing with the captive screws.

UPPER AFT WING BOX FAIRING REMOVAL (Effectivity: All)

NOTE

Removal of the left and right upper aft wing box fairing is typical.

a. Unscrew the captive screws securing the fairing.

b. Remove the fairing and store to prevent damage.

UPPER AFT WING BOX FAIRING INSTALLATION (Effectivity: All)

NOTE

Installation of the left and right upper aft wing box fairings is typical.

a. Place the fairing in position.

b. Secure the fairing with the captive screws.

FUSELAGE-TO-FORWARD FUEL TANK/ AFT WING SEALS REPLACEMENT (Effectivity: All)
(Figure 1)

NOTE

Replacement of the upper forward, upper center, upper and lower aft seals is typical.

a. With a thin nonmetallic blade, separate the seal from the fuselage and forward fuel tank/aft wing surfaces.

b. Clean old adhesive from the surfaces using lint-free cheesecloth (2, Chart 1, 53-00-00) moistened with Methyl Ethyl Ketone or Naphtha (1, 6, Chart 1, 53-00-00).

c. With a brush, apply a thin coat of adhesive (8, Chart 1, 53-00-00) to the seal and bonding surface. Allow the adhesive to dry until it is tacky and will not transfer to the knuckle when touched.

d. Position the seal and press firmly in place.

e. Edge dress along the seal with sealant (9, Chart 1, 53-00-00). Let sealant cure ten hours.

RADOME REMOVAL (Effectivity: All)
(Figure 2)

a. Unscrew the five fasteners securing the radome.

b. Remove the radome from the nose section and store to prevent damage.

RADOME INSTALLATION (Effectivity: All)
(Figure 2)

a. Check radome seal for deterioration, cuts, damage and security. For replacement of the radome seal, refer to RADOME SEAL REPLACEMENT, this section.

b. Place the radome in position and align the guide pins.

c. Secure the radome with the five captive screw fasteners.

RADOME SEAL REPLACEMENT (Effectivity: All)
(Figure 2)

a. With the radome removed, carefully remove the seal using a nonmetallic scraper to separate the seal from the upper and lower attach brackets.

b. Remove old adhesive from the upper and lower bracket surfaces using a lint-free cheesecloth (2, Chart 1, 53-00-00) moistened with Methyl Ethyl Ketone or Naphtha (1, 6, Chart 1, 53-00-00).

c. With a brush, apply a thin coat of adhesive (8, Chart 1, 53-00-00) to the surfaces of the seal and brackets. Allow the adhesive to dry until it is tacky and will not transfer to the knuckle when touched.

d. Position the seal around the upper and lower brackets with the seal ends butted at the bottom. Press the seal firmly in place.

FUSELAGE TAILCONE REMOVAL (Effectivity: All) (Figure 3)

a. Disengage the NAV circuit breaker located on the left circuit breaker panel in the cockpit.

b. Remove the screws securing the tailcone to the aft fuselage.

c. Remove the tailcone far enough to disconnect the electrical wiring to the navigation light.

d. Remove the tailcone and store to prevent damage.

FUSELAGE TAILCONE INSTALLATION (Effectivity: All) (Figure 3)

a. Place the tailcone near the aft fuselage and connect the electrical wiring to the navigation light.

b. Place the tailcone in position and secure with the correct screws.

c. Engage the NAV circuit breaker and check the navigation light for operation.

AFT WING BOX SHEAR PANEL REMOVAL (Effectivity: All) (Figure 4)

a. Remove electrical power from the airplane.

b. With an assistant supporting the weight of the panel (4), loosen the captive fasteners (1) along the LH and RH edge of the panel.

c. On airplanes NC-30 and after, remove the screws (3) and washers (2) along the aft edge of the panel (4).

d. Slide the panel (4) aft until the panel closure strip (9) is free of the front attach angle (11) mounted on the fuselage.

e. Lower the panel (4) to expose the antenna cables (10, 6 and 5) and the bonding strap (7).

f. Disconnect and tag the marker beacon and VLF antenna cables (10 and 6), and if installed, the FM antenna cable (5).

g. Disconnect the bonding strap (7) from the ground connector bracket (8).

h. Place the panel (4) in an area where it will not be damaged.

AFT WING BOX SHEAR PANEL INSTALLATION (Effectivity: All) (Figure 4)

a. With the help of an assistant, position the panel (4) under the panel opening.

b. Connect the bonding strap (7) to the ground connector bracket (8).

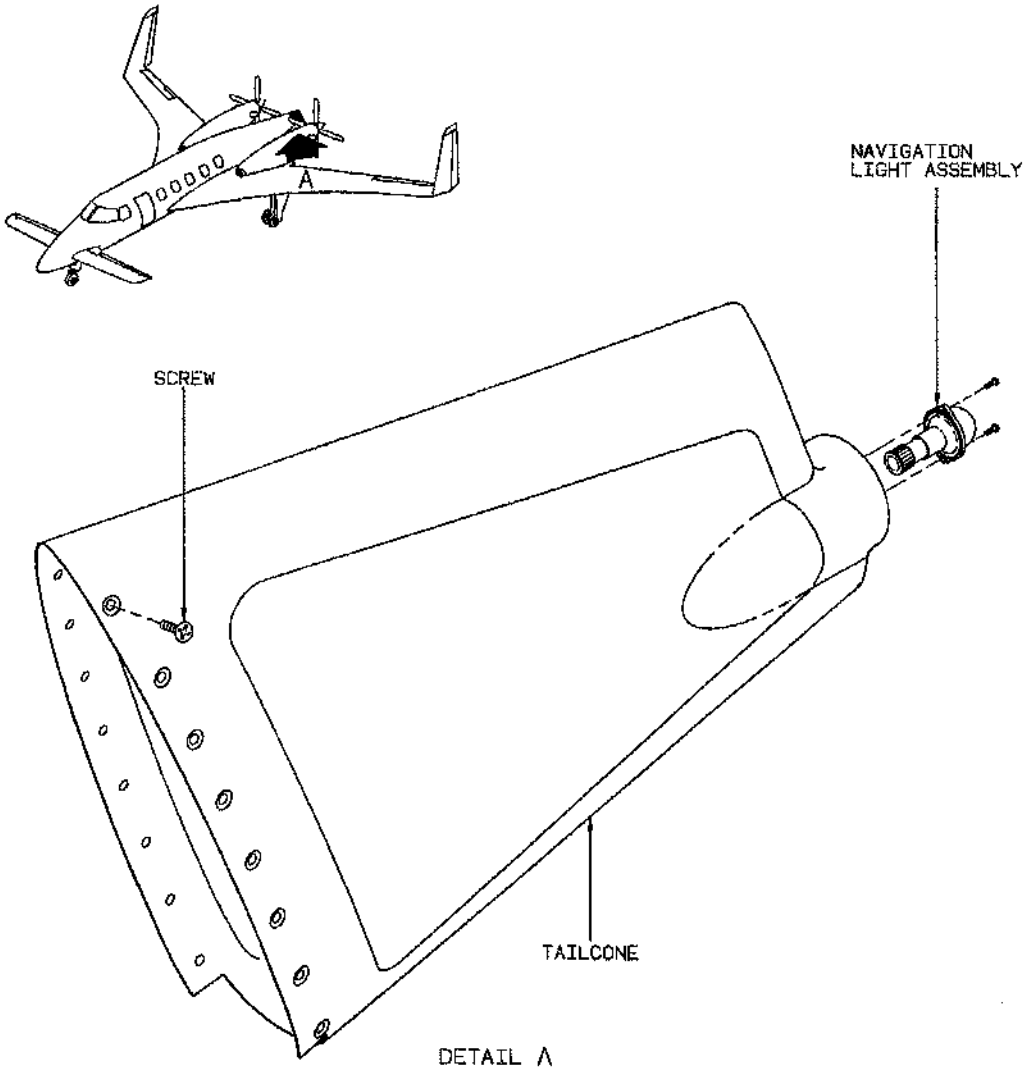
c. Connect the marker beacon and VLF antenna cables (10 and 6), and if installed, the FM antenna cable (5).

d. Slide the panel (4) forward until the panel closure strip (9) is over the front attach angle (11) mounted on the fuselage.

e. On airplanes NC-30 and after, install the screws (3) and washers (2) along the aft edge of the panel (4).

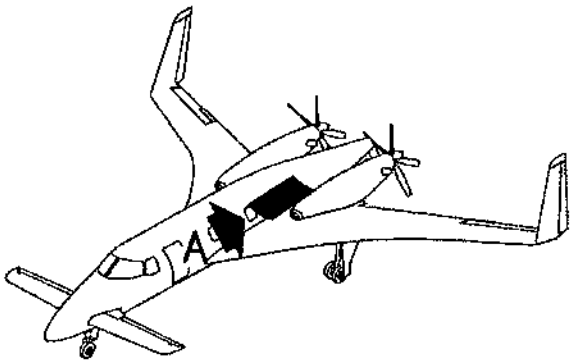
f. Tighten the captive fasteners (1) along the LH and RH edge of the panel (4).

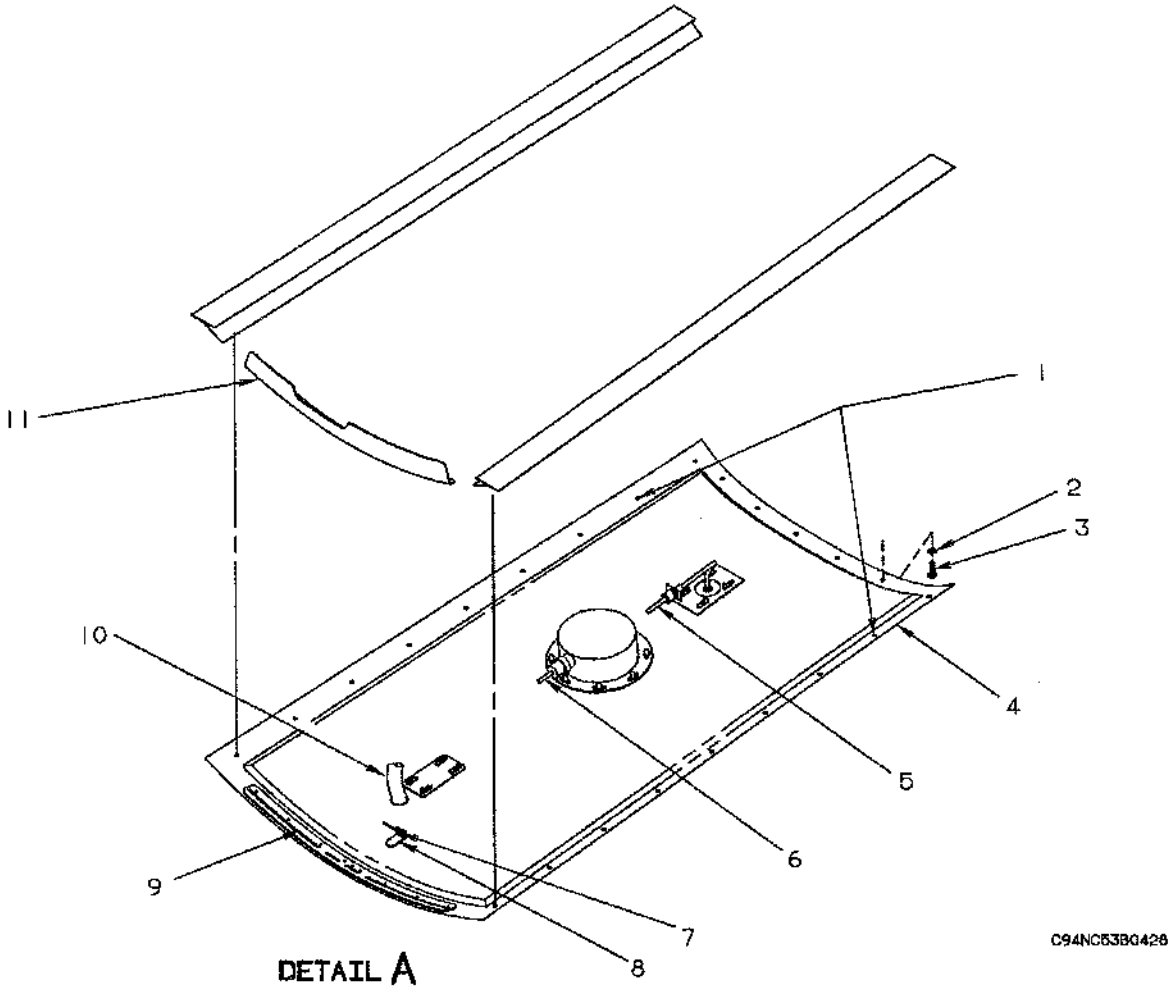
g. Restore electrical power to the airplane.



Tailcone Installation (Effectivity: All)
Figure 3

1. CAPTIVE FASTENERS
2. WASHER
3. SCREW
4. WING BOX SHEAR PANEL
5. FM ANTENNA CABLE
6. VLF ANTENNA CABLE
7. BONDING STRAP
8. GROUND CONNECTOR BRACKET
9. CLOSURE STRIP
10. MARKER BEACON ANTENNA CABLE
11. FRONT ATTACH ANGLE





Aft Wing Box Shear Panel Installation (Effectivity: All)
Figure 4

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The engine nacelles are mounted above the wing and forward of each engine. Each nacelle incorporates a firewall, shelf, skin and fairings, and houses such equipment as the engine fire extinguisher, electrical components, bleed air anti-ice components, and the engine air inlet ducting. Two access panels on top of each nacelle are removable for access to the equipment. There are two access doors located on the LH side of each nacelle. The aft LH access door has two vortilons riveted to the outside of the door. A louvered access door is located on the RH side of each nacelle. Both the access panels and the access doors are secured in place with captive stud nuts/stud fasteners (camloc).

Insulation blankets cover areas of the engine firewall, engine mounts (trusses) and the lower inboard and outboard ribs. Insulation blankets with exposed part numbers are shaped and sized for the area of the firewall they cover. The insulation blankets are secured in place with clips, fasteners, rivets and washers. No gaps are permitted where the insulation blankets are butted together. Insulation blankets covering the forward end of the engine mounts (trusses) are secured with straps, clamps, screws, washers, nuts and sealant.

A removable panel (angle), secured with four fasteners into nutplates, is located under the lower engine firewall for the purpose of inspecting the aluminum tape protecting the wing aft spar. The aluminum tape covers the entire wing aft spar in front of the engine lower firewall except under the engine-mount fittings.

RECOMMENDED MATERIALS (Effectivity: All)

The recommended materials listed in Chart 1 as meeting federal, military or vendor specifications are provided for reference only and are not specifically required by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor, or by compliance with the applicable specifications. GENERIC OR LOCALLY MANUFACTURED PRODUCTS WHICH CONFORM TO THE REQUIREMENTS OF THE SPECIFICATION MAY BE USED EVEN THOUGH NOT INCLUDED IN THE CHARTS. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
RECOMMENDED MATERIALS (Effectivity: All)

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
1. Adhesive/Sealer		RTV-106	General Electric Co. Waterford, NY 12188
2. Cleaner	TT-N-95	Naphta	Procure locally
3. Aluminum Tape		10100B-1/ 1A812	Mactac, Stowe, OH 44224

GENERAL - MAINTENANCE PRACTICES (Effectivity: All)

Maintenance of the nacelles consists mainly of removal and installation of the access panels, access doors, firewall insulation and the wing aft spar aluminum tape. Inspection of the insulation blankets and aluminum tape is accomplished at intervals outlined in Chapter 4-00-00.

NACELLE UPPER ACCESS PANELS REMOVAL (Effectivity: All) (Figure 1)

NOTE

Removal of the left and right nacelle upper access panels is typical. The aft access panels must be removed before the forward panels can be removed.

a. Unscrew the stud nuts around the edge of the aft access panel.

b. Lift the aft access panel from the nacelle and store to prevent damage.

c. Unscrew the stud nuts around the edge of the forward access panel.

d. Lift the forward access panel from the nacelle and store to prevent damage.

NACELLE UPPER ACCESS PANELS INSTALLATION (Effectivity: All) (Figure 1)

NOTE

Installation of the left and right nacelle upper access panels is typical. When installing the access panels, install the forward access panel first and then the aft access panel.

a. Place the forward access panel in position on the nacelle.

b. Secure the forward access panel with stud nuts.

c. Place the aft access panel in position on the nacelle and align the forward edge to the forward access panel.

d. Secure the aft access panel with the stud nuts.

NACELLE ACCESS DOORS REMOVAL (Effectivity: All) (Figure 1)

NOTE

Removal of all nacelle access doors is typical. Two vortilons are riveted on the LH access door and the RH aft access door is louvered.

a. Unscrew the stud fasteners securing the access door.

b. Remove the access door and store to prevent damage.

NACELLE ACCESS DOORS INSTALLATION (Effectivity: All) (Figure 1)

NOTE

Installation of all nacelle access doors is typical. The access doors having vortilons and louvers are installed with the vortilons/louvers at the aft end of the access doors.

a. Place the access door in position on the nacelle.

b. Secure the access door with the stud fasteners.

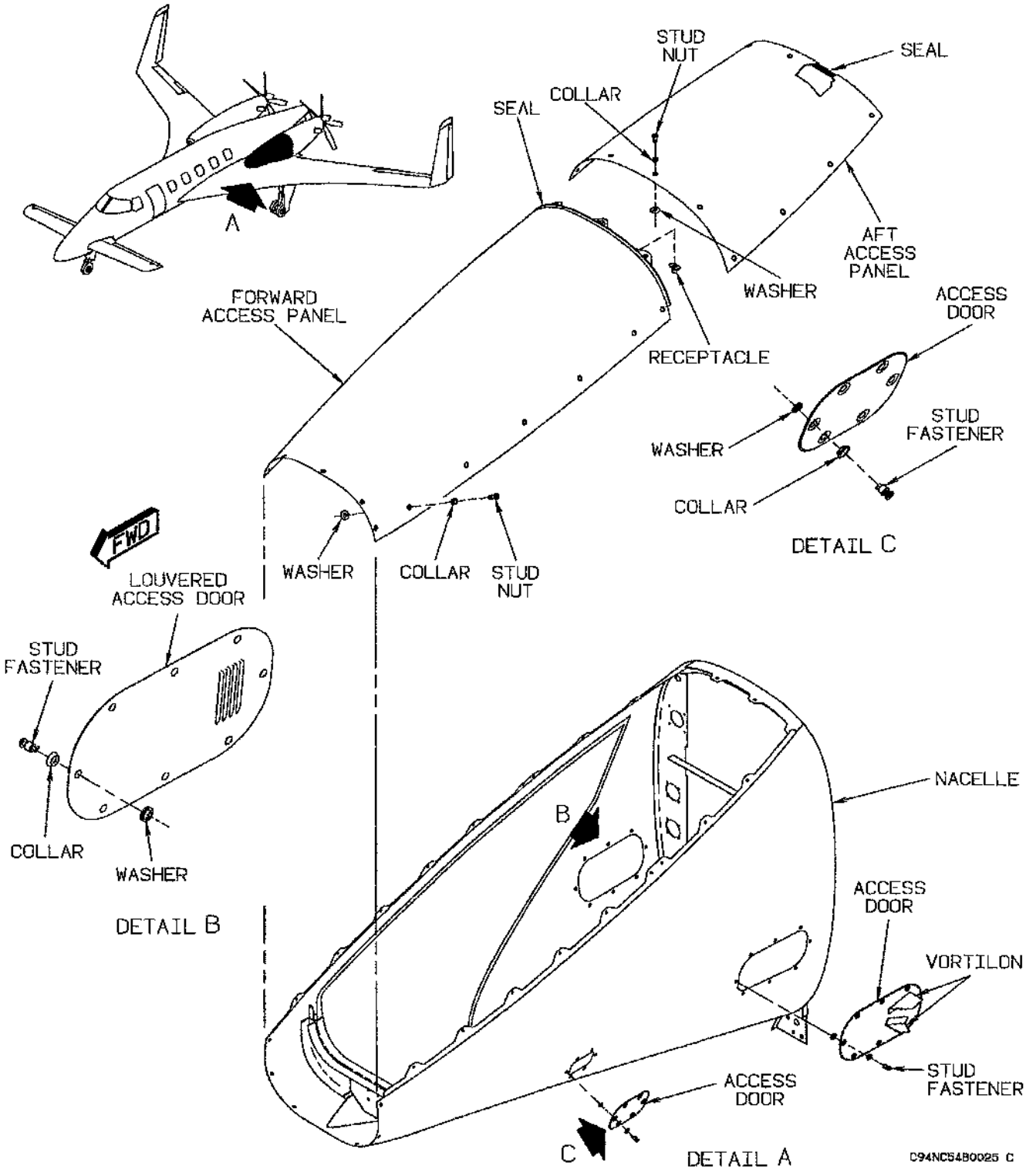
INSULATION BLANKET REPLACEMENT (Effectivity: All) (Figure 2)

NOTE

Replacement insulation blankets are designed and identified by part number for a particular area of the engine firewall, inboard and outboard ribs/coves and the engine mounts. The part numbers are stamped in ink on the face of the blanket except for the blanket on the inboard cove. Replace each insulation blanket with matching part numbers. Refer to BEECHCRAFT Starship 1 Illustrated Parts Catalog.

a. Gain access to the insulation blanket to be replaced by removing the necessary engine cowling as outlined in Chapter 71-10-00.

b. Remove the insulation blanket(s) to be replaced by drilling out, and/or removing the fasteners.



Nacelle Access Panels and Doors Installation (Effectivity: All)
Figure 1

c. For removal of the upper engine mount insulation blankets, remove the three nuts, washers, screws and two straps. Remove the clamps on the engine mounts.

d. For removal of the lower outboard engine mount/rib insulation blanket, remove the two screws, four washers and two nuts attaching the end tab to the side flap on the blanket. If the insulation blanket is sealed and/or retained with sealant, cut and unwrap the insulation blanket from the engine mount.

e. For removal of the lower inboard engine mount insulation blanket, remove the clamp securing the blanket around the engine mount. Cut and unwrap the blanket from the engine mount.

NOTE

The lower engine mount insulation blankets are sealed to the adjacent firewall insulation blankets. Removal of either the firewall or engine-mount insulation blankets may damage both blankets and require replacement.

f. Prepare and clean the firewall surface of oils, fluids and dirt for installation of the new insulation blanket(s).

NOTE

The insulation blankets are to be installed with the part number outward.

g. Align the edges of the new firewall insulation blankets to within ±0.10-inch of the corresponding features of the firewall and butt the new blankets up against the adjacent blankets with no gaps between them.

h. Secure the insulation blanket with rivets and washers, fasteners and/or clips as applicable.

i. For installation of the upper engine-mount insulation blanket, secure the straps together with three screws, washers and nuts. Secure clamps around the engine mounts.

j. For installation of the lower outboard engine mount/rib blanket, position the blanket to the outboard rib and wrap the end tab around the engine mount. Attach the end tab to the side flap of the blanket with two screws, a washer on each side of the blanket, and two nuts. Seal around the engine mount between the firewall blanket and the engine- mount blanket with RTV 106 sealant (1, Chart 1).

k. For installation of the inboard engine-mount insulation blanket, install part of the blanket in the short tube on the engine mount and wrap the blanket around each side of the mount. Apply RTV 106 sealant (1, Chart 1), between the mount blanket and the firewall blanket to seal and secure them together. Install clamp around the blanket and mount.

l. Install the cowling as outlined in Chapter 71-10-00.

WING AFT SPAR ALUMINUM TAPE REPLACEMENT (Effectivity: All)

Aluminum tape is applied to the wing aft spar behind the lower firewall of each engine. The purpose of the aluminum tape is to reflect heat and protect the wing spar.

a. Gain access to the aluminum tape by removing the four screws securing the angle below the engine lower firewall. Remove the angle.

b. Peel the strips of aluminum tape from the wing aft spar.

c. Clean the wing aft spar surface with Naphtha (2, Chart 1) in preparation for the new aluminum tape.

d. Cover the wing aft spar with strips of adhesive backed aluminum tape (3, Chart 1). The aluminum tape is not required under the engine mount fittings.

e. Position the angle and secure with four screws.

INSULATION BLANKETS INSPECTION (Effectivity: All)

The following procedures specify the requirements and instructions for in-service inspection of the firewall insulation blankets for the left and right engine. Inspection intervals are outlined in Chapter 4-00-00.

LEFT ENGINE FIREWALL INSULATION BLANKETS INSPECTION (Effectivity: All)

a. Visually inspect the insulation blankets for damage, security, missing sections and tears.

b. Insulation blankets that are damaged, loose, missing or torn must be replaced.

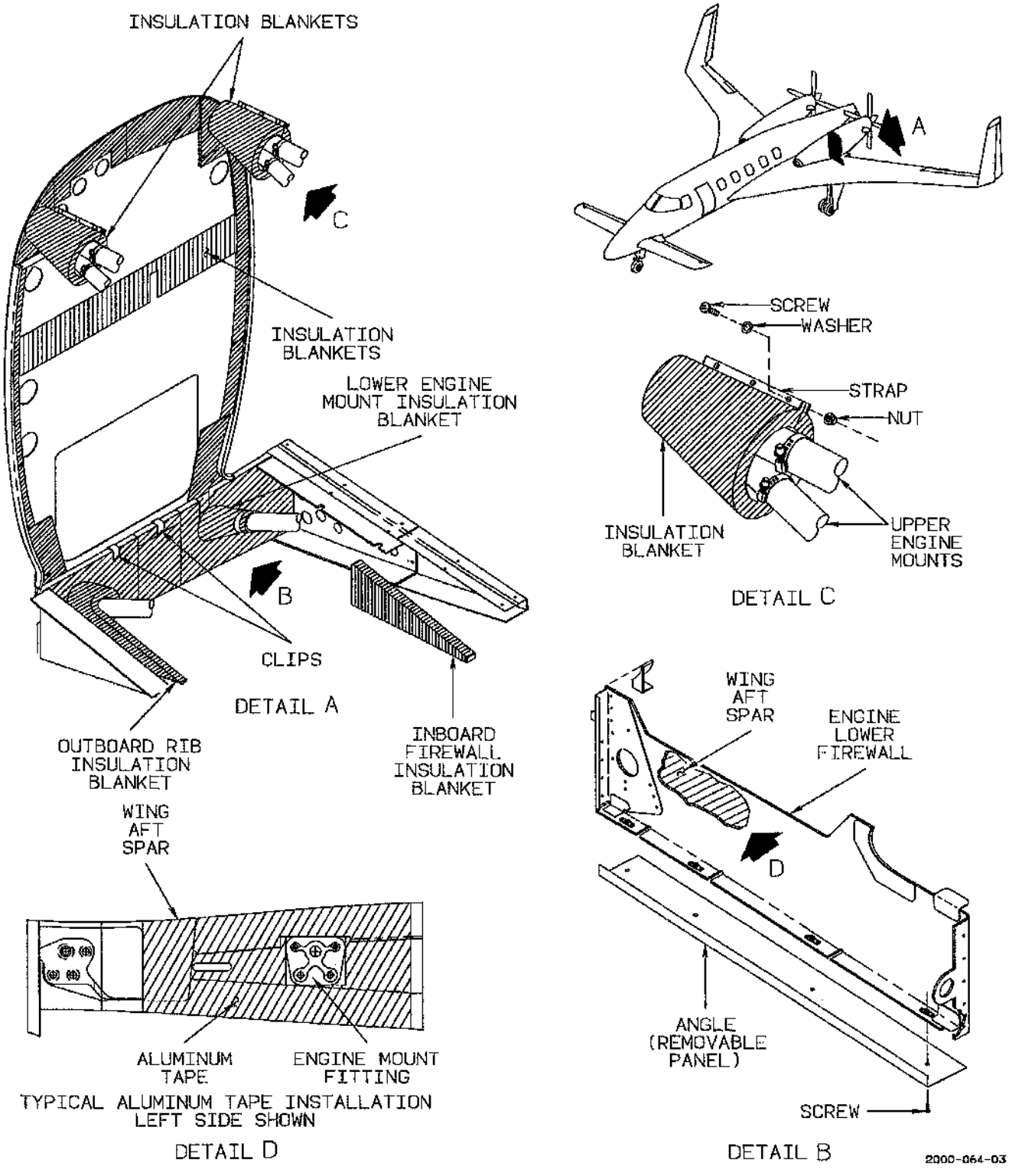
RIGHT ENGINE FIREWALL INSULATION BLANKETS INSPECTION (Effectivity: All)

a. Visually inspect the insulation blankets for damage, security, missing sections and tears.

b. Insulation blankets that are damaged, loose, missing or torn must be replaced.

LEFT ENGINE-MOUNT INSULATION BLANKETS INSPECTION (Effectivity: All)

a. Visually inspect the upper and lower engine-mount blankets for damage, tears and missing attached hardware.



Firewall Insulation Blankets and Aluminum Tape Installation (Effectivity: All)
Figure 2

b. Engine-mount blankets that are damaged or torn must be replaced. Replace missing attached hardware as applicable.

RIGHT ENGINE-MOUNT INSULATION BLANKETS INSPECTION (EFFECTIVITY: ALL)

a. Visually inspect the upper and lower engine-mount blankets for damage, tears and missing attached hardware.

b. Engine-mount blankets that are damaged or torn must be replaced. Replace missing attached hardware as applicable.

LEFT INBOARD FIREWALL INSULATION BLANKET INSPECTION (Effectivity: All)

a. Visually inspect the insulation blankets for damage, security, missing sections and tears.

b. Insulation blankets that are loose, missing or damaged must be replaced.

RIGHT INBOARD FIREWALL INSULATION BLANKETS INSPECTION (Effectivity: All)

a. Visually inspect the insulation blankets for damage, security, missing sections and tears.

b. Insulation blankets that are loose, missing or damaged must be replaced.

WING AFT SPAR ALUMINUM TAPE INSPECTION (Effectivity: All)

Aluminum tape is applied to the wing aft spar behind each engine lower firewall. Gain access to the aluminum tape by removing the angle below the engine lower firewall. Four screws secure the angle.

NOTE

For inspection of the aluminum tape protecting the left and right wing trailing edge inboard coves, refer to Chapter 57-00-00.

LEFT WING AFT SPAR ALUMINUM TAPE INSPECTION (Effectivity: All)

a. Wipe the aluminum tape clean and visually inspect the tape for disbonds, tears and discoloration (nonreflective).

b. If any of the above conditions exist, remove the aluminum tape in the affected area, clean the surface of the aft spar with Naphtha (2, Chart 1). Install new aluminum tape (3, Chart 1).

RIGHT WING AFT SPAR ALUMINUM TAPE INSPECTION (Effectivity: All)

a. Wipe the aluminum tape clean and visually inspect the tape for disbonds, tears and discoloration (nonreflective).

b. If any of the above condition exist, remove the aluminum tape in the affected area, clean the surface of the aft spar with Naphtha (2, Chart 1). Install new aluminum tape (3, Chart 1).

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

FORWARD WINGS, ELEVATORS AND ELEVATOR TRIM TABS (Effectivity: All)

The left and right forward wings pivot 30 degrees aft and -4 degrees forward on pivot pins and bearings which secure the wings to the upper and lower shelves in the nose assembly. Structurally, the forward wings are supported in the nose assembly by the nose keels, a bulkhead at FS 45.00, and the upper and lower avionic shelves. The left and right elevators are hinged on the forward wing trailing edges for pitch control of the airplane. Each elevator has a controllable trim tab.

For maintenance of the elevator and forward wing control systems refer to Chapters 27-30-00 and 27-50-00.

TIPSAILS (Effectivity: All)

The tipsails are mounted on the outboard ends of the aft wing. The tipsails are vertical airfoils constructed of spars, ribs and skins. A tip encloses and streamlines the top of each tipsail. Three hinge brackets on the trailing edge of the tipsails provide for the installation of the left and right rudders.

RUDDERS AND RUDDER TRIM TABS (Effectivity: All)

The left and right rudders are airfoils hinged on the trailing edge of the tipsails for yaw control of the airplane. Each rudder has a controllable trim tab.

For maintenance of the rudder and rudder trim tab control systems refer to Chapter 27-20-00.

VENTRAL STABILIZER (Effectivity: All)

The ventral stabilizer is a fixed airfoil mounted below the aft fuselage tailcone. The ventral stabilizer is constructed of spars, ribs and skin with a tip enclosing the lower rib.

SPECIAL TOOLS AND RECOMMENDED MATERIALS (Effectivity: All)

The special tools and recommended materials listed in Charts 1 and 2 as meeting federal, military or vendor specifications are provided for reference only and are not specifically prescribed by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All)

TOOL NAME	PART NO.	VENDOR	USE
1.Cradle	None	Obtain Locally	Support forward wing.
2.C-Wrench	TK 2031	Beech Aircraft Corp.,Wichita, Kansas	Torque forward wing pivot pin nuts.
3.Torque Wrench (36-inch, 1/2-inch or 3/4-inch drive)	None	Obtain Locally	Torque forward wing pivot pin nuts.
4.Spanner Wrench	TK 2032	Beech Aircraft Corp., Wichita, Kansas	Torque forward wing installation nuts.
5.Loading Fixture	None	Obtain Locally	Apply load to forward wing during free-play checks.
6.Dial Indicator	None	Obtain Locally	Measure deflection of forward wing during free-play check.
7.Pounds of Force Indicating Device	None	Obtain Locally	Measure force applied to forward wing during free-play check.
8.Stand with knife-edge supports.	None	Fabricate	Support elevator during balancing procedure.
9.Beam balance weighing device calibrated in units of .001 pounds or less.	None	Obtain Locally	Weight lead shot used in elevator balancing procedure.
10.Inclinometer	None	Obtain Locally	Determine position of elevator for balancing procedure.
11.Position Indicator Stand	None	Fabricate	Indicate elevator position during balancing procedure.

CHART 2
RECOMMENDED MATERIALS (Effectivity: All)

MATERIALS	SPECIFICATION	PRODUCT	VENDOR
1.Cleaner	TT-M-261	MEK	Obtain Locally
2.Cleaner	TT-N-95	Naphtha	Obtain Locally
3.Adhesive	None	EA9309NA	Hysol Div., The Dexter Corp., 211 Franklin St., Olean, NY 14760
4.Thickening Agent(Filler)	MIL-S-47129	CAB-O-SIL M5	Godfrey L. Cobot Corp, Tuscola, Il. 61953
5.Cloth	CCC-C-440 Class II	Cheesecloth	Obtain Locally
6.Adhesive	None	EC1403	Minnesota Mining and Manufacturing Co., 900 Bush Ave. St. Paul, MN 55144

FORWARD WING - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The forward wings are of composite construction with deice boots installed on the leading edge. The forward wings operate in conjunction with the flaps on the aft wing. With the flaps full down, the forward wing is at -4 degrees swept forward, and with the flaps full up, the forward wing is at approximately 30 degrees swept aft. For operation of the forward wing/flaps, refer to Chapter 27-50-00. The forward wings pivot on pivot pins and bearings mounted in the nose assembly. Elevators are attached to the forward wings at three hinge brackets along each forward wing trailing edge (cove). Each elevator has a trim tab positioned by an electrical actuator mounted in the elevator. Balancing the elevators is outlined in Chapter 55-20-00. Refer to Chapter 27-30-00 for maintenance and operation of the elevator/trim tab control systems.

FORWARD WING - MAINTENANCE PRACTICES (Effectivity: All)

FORWARD WING REMOVAL (Effectivity: All) (Figures 1 and 2)

a. Remove all electrical power from the airplane.

b. Place the airplane on jacks as outlined in Chapter 7-10-00.

c. Remove the upper and lower forward wing fairings (212AR and 212BR or 211AL and 211BL) for the left/right forward wing to be removed. Refer to Chapter 53-50-00 for removal of the forward wing fairings.

d. Remove the left or right lower nose panel (131AL or 132AR) for the left/right forward wing to be removed. Refer to Chapter 53-30-00 for removal of the lower nose panels.

e. Remove the left or right upper avionics compartment door (213AL or 214AR) for the left/right forward wing to be removed. Refer to Chapter 53-30-00 for removal of the avionics compartment doors.

f. Disconnect the forward wing actuator and elevator control linkage at the forward wing. Refer to Chapters 27-30-00 and 27-50-00.

g. Remove any safety wire from the pivot pin assembly nuts. Disconnect the ground strap between the lower shelf and the lower forward wing spar.

h. Disconnect the electrical wire bundle from the elevator trim tab actuator.

i. To counterbalance the weight of the forward wing and reduce any binding at the pivot pin (1), place a cradle (1,Chart 1, 55-00-00) under the forward wing. A 50-pound upward force at the location shown in Figure 4 will place the forward wing in the neutral up/down position.

NOTE

It is recommended that the wing support be on rollers with adjustable legs for height.

NOTE

All torquing and clockwise/counterclockwise rotating directions herein are viewed looking down at the forward wing pivot pin.

j. Loosen the lower nut (9) (just above the lower shelf) counterclockwise as required to provide a minimum .10-inch gap between the lower nut (9) and the ball of the lower pivot bearing (2).

k. Turn the middle flange nut (8) counterclockwise 1 to 1-1/2 full revolutions to withdraw the lower tapered collar (11) from the lower bonded spar bushing (6).

l. Remove the pivot pin nut (4) by turning it clockwise using a C-wrench (2, Chart 1, 55-00-00) or equivalent.

m. Remove the 3/8-inch bolts locking the pivot pin (1) and the forward wing spacer fitting (5).

n. From inside the pivot pin (1), loosen the upper and lower set screws (12) connecting the pivot pin to the upper and lower pivot bearings (2). Removal of the set screws (12) is not required.

o. Turn the upper flange nut (7) (just above the upper shelf) clockwise 1 to 1-1/2 full revolutions to withdraw the upper tapered collar (10) from the upper bonded spar bushing (6).

p. Tap the lower end of the pivot pin (1) with a mallet and wood block until the pivot pin is dislodged from the hardware.

q. Place a metal rod through the holes in the pivot pin (1) and pull/remove the pivot pin while supporting the forward wing.

r. Remove the forward wing from the nose assembly. Store the forward wing and protect the pivot pin hardware from damage.

FORWARD WING INSTALLATION (Effectivity: All) (Figures 1 and 2)

Prior to the installation of the forward wing, the pivot parts for the forward wing must be assembled as detailed under steps a through f. If the pivot parts are already assembled correctly on the forward wing, proceed to step g.

NOTE

The forward wing installation should be accomplished in sequential order and in the manner described to ensure that minimum free-play in the pivot joint is achieved.

a. With the forward wing on a bench, insert the upper tapered collar (10) into the upper bonded spar bushing (6) in the forward wing. Push the collar in place with light hand pressure only.

NOTE

Align the slot (machined in the tapered collar) with the stud protruding inside of the spar bushing.

b. Thread the upper flange nut (7) onto the upper tapered collar (10) until approximately .03-inch clearance exists between the nut flange and the upper bonded spar bushing (6).

NOTE

A slight clearance must be maintained to ensure that the upper flange nut (7) does not back the upper tapered collar (10) out of the upper bonded spar bushing (6).

c. Insert the lower tapered collar (11) into the lower bonded spar bushing (6) in the wing. Push the collar in place with light hand pressure only.

NOTE

Align the slot (machined in the tapered collar) with the stud protruding inside of the spar bushing.

d. Thread the middle flange nut (8) onto the lower tapered collar (11) until approximately .03-inch clearance exists between the nut flange and the lower bonded spar bushing (6).

NOTE

A slight clearance must be maintained to ensure that the middle flange nut (8) does not back the lower tapered collar (11) out of the lower bonded spar bushing (6).

e. Thread the lower nut (9) onto the lower tapered collar (11) until the nut bottoms out. Back off the nut just slightly to loosen.

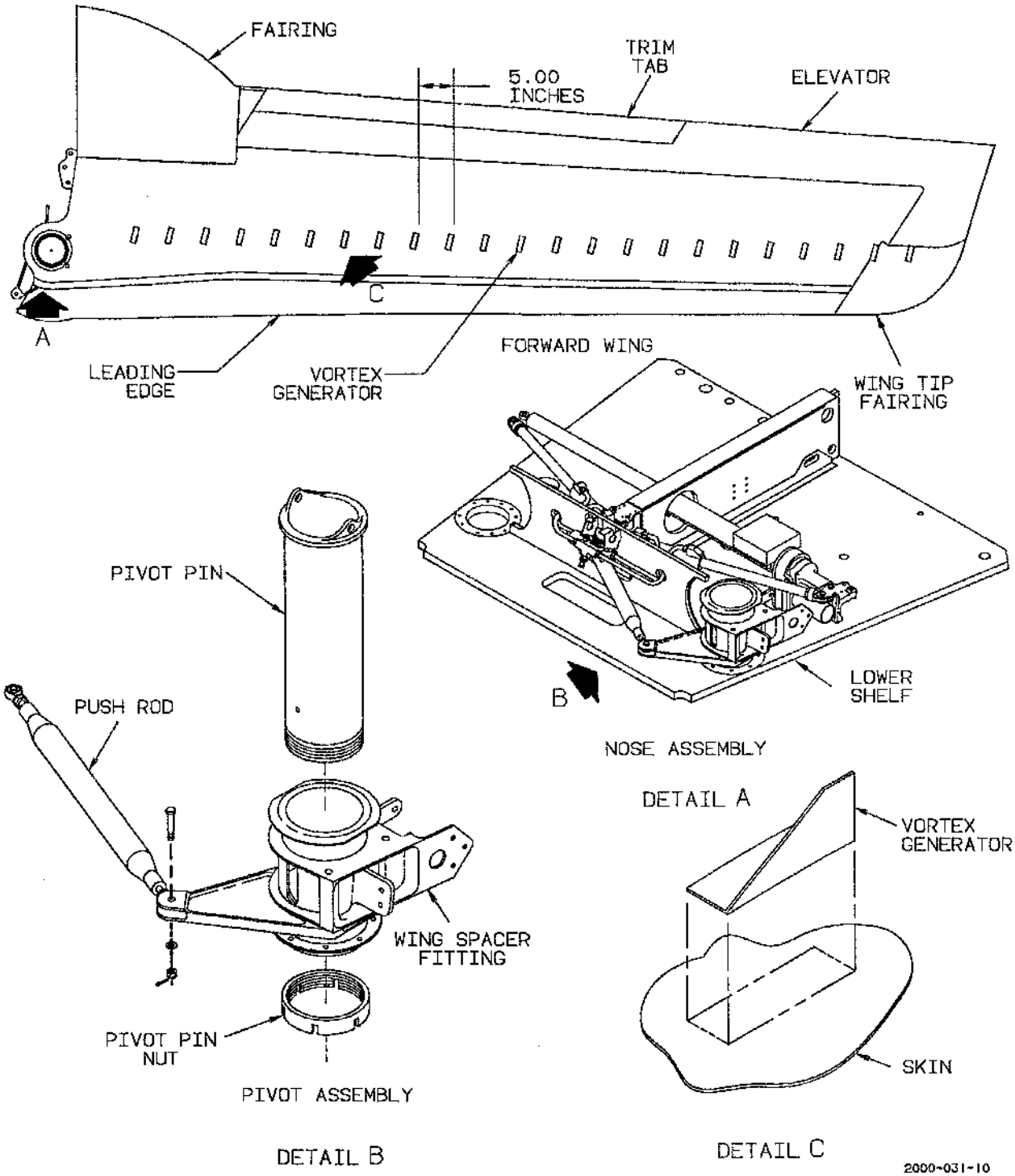
f. Check the total height of the resulting forward wing assembly at the location shown in Figure 2. To fit into the nose assembly properly, the forward wing assembly should not exceed 9.2 inches in height at the pivot area. If necessary for proper fit, repeat steps a through e.

g. Place the forward wing in position to the nose assembly. To counterbalance the weight of the forward wing and reduce any binding at the pivot pin (1), place a support under the forward wing. A 50-pound upward force at the location shown in Figure 4 will place the forward wing in the neutral up/down position.

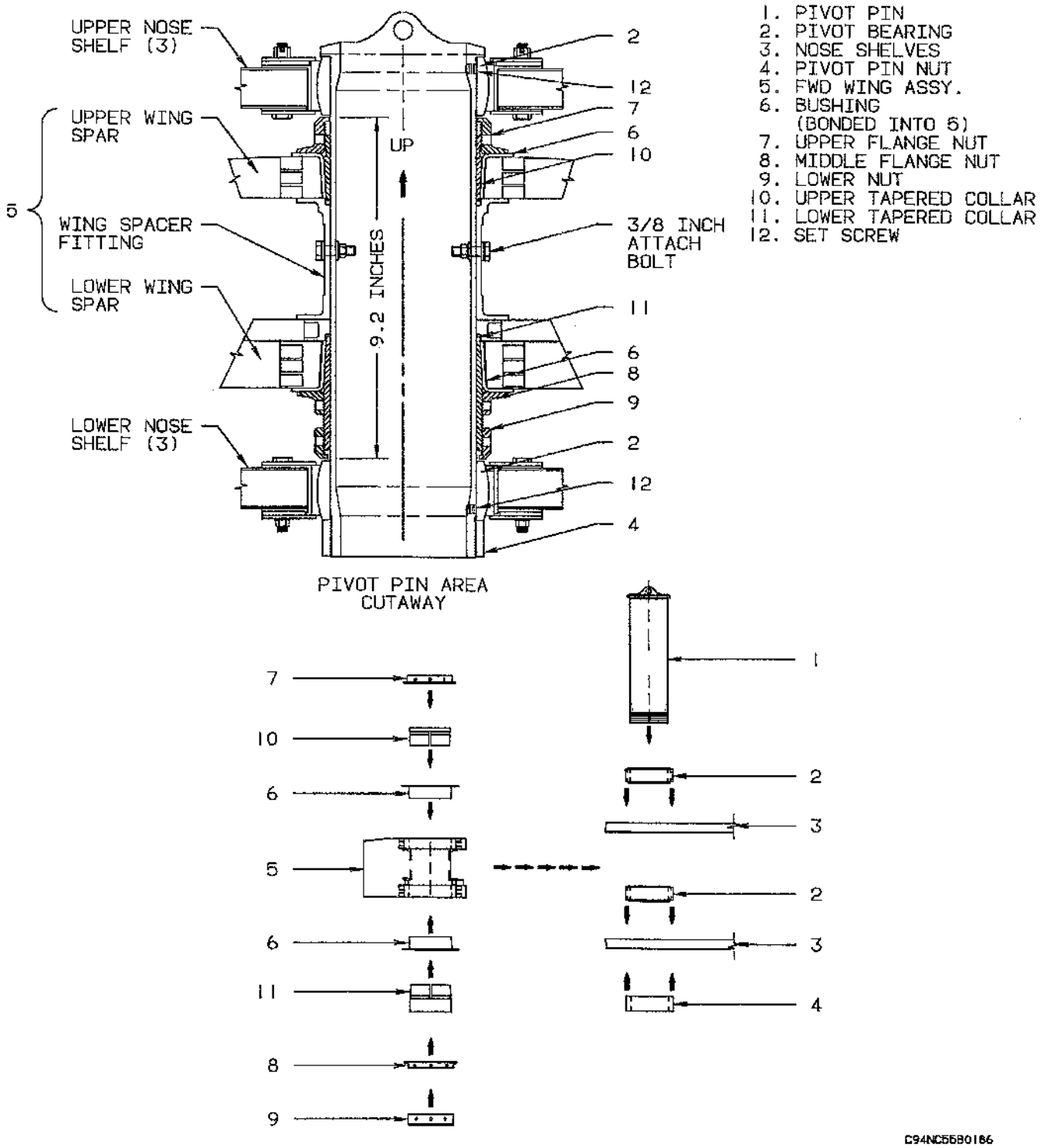
NOTE

It is recommended that the wing support be on rollers with adjustable legs for height.

h. Insert the pivot pin (1) through the upper pivot bearing (2), forward wing assembly and the lower pivot bearing (2) in the order noted.



Forward Wing and Pivot Assembly (Effectivity: All)
Figure 1



Forward Wing Pivot Pin Installation (Effectivity: All)
Figure 2

NOTE

A hand mallet and wood block may be used to assist the pivot pin through the assembly. AVOID USING EXCESSIVE FORCE WHICH MAY DAMAGE THE COMPOSITE SUPPORT STRUCTURE.

NOTE

Use care to align each 3/8-inch attaching bolt hole in the pivot pin (1) with the matching slotted holes in the forward wing spacer fitting (5). Align the slot machined in the upper and lower pivot bearings (2) with the set screws (12) in the pivot pin (1).

i. After the pivot pin is installed, install the 3/8-inch attaching bolts in the aligned holes. DO NOT TIGHTEN THE 3/8-INCH BOLTS AT THIS TIME.

j. Torque the forward wing pivot pin hardware as instructed under FORWARD WING PIVOT PIN TORQUING PROCEDURE in this chapter.

k. After the pivot pin hardware has been properly torqued, tighten the 3/8-inch attaching bolts that were previously left loose.

l. From inside the pivot pin (1), tighten the upper and lower set screws (12) into the slot machined in the upper and lower pivot bearings (2).

m. Connect the ground strap between the lower forward wing spar and the lower shelf.

NOTE

Refer to Chapter 20-00-03 for the proper bonding procedure.

n. Connect the electrical wire bundle for the elevator trim tab actuator.

o. Connect the forward wing actuator and elevator control linkage at the forward wing (refer to Chapters 27-30-00 and 27-50-00).

p. Perform a free-play check to verify the proper torque of the pivot pin assembly nuts. Refer to FORWARD WING FREE-PLAY CHECK in this chapter.

NOTE

Safety wiring/locking of the pivot pin assembly nuts may be delayed until the free-play check is performed; however, the safety wiring/locking must be accomplished immediately after the free-play check.

q. Secure the pivot pin assembly nuts using the following safety methods:

1. Safety wire the upper flange nut (7) to the head of the 3/8-inch attaching bolt in a manner resisting clockwise rotation of the upper flange nut (looking down at the pivot pin).

2. Safety wire the middle flange nut (8) to the shank of the 3/8-inch attaching bolt in a manner resisting clockwise rotation of the middle flange nut (looking down at the pivot pin).

3. Cross safety wire the middle flange nut (8) to the lower nut (9) in a manner resisting clockwise rotation of the middle flange nut and counterclockwise rotation of the lower nut (looking down at the pivot pin).

4. Safety lock the pivot pin nut (4) to the pivot pin (1) with a cotter pin. Align the cotter pin holes when tightening the pivot pin nut.

r. Remove the airplane from the jacks.

s. Install the upper and lower forward wing fairings (212AR and 212BR or 211AL and 211BL) for the left/right forward wing which was removed. Refer to Chapter 53-50-00 for installation of the forward wing fairings.

t. Install the left or right lower nose panel (131AL or 132AR) for the left/right forward wing which was removed. Refer to Chapter 53-30-00 for installation of the lower nose panels.

u. Install the upper left or right nose avionics compartment door (213AL or 214AR) for the left/right forward wing which was removed. Refer to Chapter 53-30-00 for installation of the avionics compartment doors.

v. Restore electrical power to the airplane.

FORWARD WING PIVOT PIN TORQUING PROCEDURE (Effectivity: All) (Figure 2)

Tighten the forward wing pivot installation by torquing the nuts in the sequence listed below. Using any other torquing sequence is cause for removing the forward wing and restarting the installation.

NOTE

All torque directions are given for a view looking down at the pivot pin.

CHART 1
FORWARD WING HARDWARE TORQUE SEQUENCE (Effectivity: All)

*ITEM	NAME	TORQUE (FT-LBS)	DIRECTION (LOOKING DOWN)
4	Pivot Pin Nut	30-60	CCW
7	Upper Flange Nut	Hand Tight	CW
8	Middle Flange Nut	Hand Tight	CCW
9	Lower Nut	60-80	CW
7	Upper Flange Nut	80-100	CCW
8	Middle Flange Nut	As Required	CW
9	Lower Nut	140-160	CW
8	Middle Flange Nut	140-160	CCW
7	Upper Flange Nut	140-160	CCW
	*Item numbers are those of figure 2.		

a. The following equipment is needed to torque the pivot pin hardware:

1. A cradle (1, Chart 1, 55-00-00) or other type of support capable of supporting a free-standing wing. It is recommended that the wing support be on rollers with adjustable legs for height.

2. A torque wrench (3, Chart 1, 55-00-00) at least 36 inches in length with a 1/2-inch or 3/4-inch square drive and capable of supplying 160 foot-pounds of torque is recommended.

3. A spanner wrench (4, Chart 1, 55-00-00) or equivalent is required to apply the torque to the forward wing installation nuts.

4. A C-wrench (2, Chart 1, 55-00-00) or equivalent is required to tighten the pivot pin nut.

NOTE

When using the spanner wrench, calculate the torque ratio caused by adding the spanner wrench to the torque wrench. Refer to Chapter 20.

b. To counterbalance the weight of the forward wing and reduce any binding at the pivot pin (1), place a support under the forward wing. A 50-pound upward force at the location shown in Figure 4 will place the forward wing in the neutral up/down position.

NOTE

It is recommended that the wing support be on rollers with adjustable legs for height.

c. Check top machined flange of the pivot pin (1) and make sure there is contact between the machined flange and the ball of the upper pivot bearing (2). Torque the pivot pin nut (4) counterclockwise to 30 to 60 foot-pounds. Use a special C-wrench (2, Chart 1, 55-00-00).

d. Hand torque the upper flange nut (7) clockwise to a hand- tight condition. Avoid excess tightening which may back the upper tapered collar (10) out of the wing spar bushing (6).

e. Hand torque the middle flange nut (8) counterclockwise to a hand-tight condition. Avoid excess tightening which may back the lower tapered collar (11) out of the wing spar bushing (6).

f. Using the spanner wrench (4, Chart 1, 55-00-00) and torque wrench (3, Chart 1, 55-00-00), torque the lower nut (9) clockwise to 60 to 80 foot-pounds. This should raise the forward wing and push the upper flange nut (7) against the ball of the upper pivot bearing (2). Check for proper contact of the upper flange nut (7) to the upper pivot bearing (2) using a .010-inch feeler gage.

NOTE

Proper "contact" is defined as contact over at least 50% of the possible contact area.

g. If proper contact is not made, the pivot area may be in a bind. Retorque the lower nut (9) to 60 to 100 foot-pounds while moving the wing tip slightly up and down. Recheck for proper contact of the upper flange nut (7) to the upper pivot bearing (2).

h. Torque the upper flange nut (7) counterclockwise to 80 to 100 foot-pounds.

i. Loosen the middle flange nut (8) clockwise as required until there is approximately .25-inch clearance between the middle flange nut (8) and the lower bonded spar bushing (6) in the wing

j. Torque the lower nut (9) clockwise to 140 - 160 foot-pounds.

k. Torque the middle flange nut (8) counterclockwise to 140 to 160 foot-pounds.

l. Torque the upper flange nut (7) counterclockwise to 140 to 160 foot-pounds.

FORWARD WING FREE-PLAY CHECK
(Effectivity: All)
(Figures 3 thru 6)

The purpose of the forward wing free-play check is to verify the tightness of the forward wing pivot pin hardware installation. A free-play check of each forward wing must be accomplished at the intervals specified in Chapter 4 and anytime the pivot pin hardware is removed, loosened, reinstalled and/or tightened. A free-play check of the forward wings consists of loading the wings in an up and down direction, measuring the amount of deflection, recording the amount of deflection on a copy of the Free-Play Determination Form, then working from the Free-Play Determination Form to determine the free play numerically. A sample and blank Free-Play Determination Form is provided in Figures 5 and 6.

NOTE

Free-play is defined as the deflection of the forward wing, measured at the wing tip, which is not due to the linear deflection of the structure under an applied load.

a. The following equipment is necessary to perform the forward wing free-play check:

1. A loading fixture (5, Chart 1, 55-00-00) capable of safely placing an upward load of 350 pounds of force and a downward load of 250 pounds of force to the forward wing structure in increments of 50 pounds. The loading fixture design shall include loading pads made to match the contour of the forward wing airfoil and should contact the wing surface at the main and rear spar locations or make continuous contact along the chord length. The pads must be fabricated from a material that will protect the forward wing surfaces from damage.

2. A dial indicator or deflection measurement device (6, Chart 1, 55-00-00) to measure the vertical deflection of the forward wing. The measuring device must be capable of measuring a minimum of 2 inches in the up direction and 2 inches in the down direction from a "zeroed" position.

3. A pounds-of-force indicating device (7, Chart 1, 55-00-00) to indicate the amount of force being applied to the forward wing by the loading fixture. The device must be capable of indicating up to 350 pounds of force.

NOTE

To ensure the validity of the free-play check, do not allow any movement of the airplane during the check.

b. Jack the airplane until all weight is off of the landing gear. Refer to Chapter 7-10-00 for jacking procedures.

c. Visually check that the pivot pin assembly installation is complete.

d. All forward wing and elevator actuating components must be connected, refer to Chapters 27-30-00 and 27-50-00.

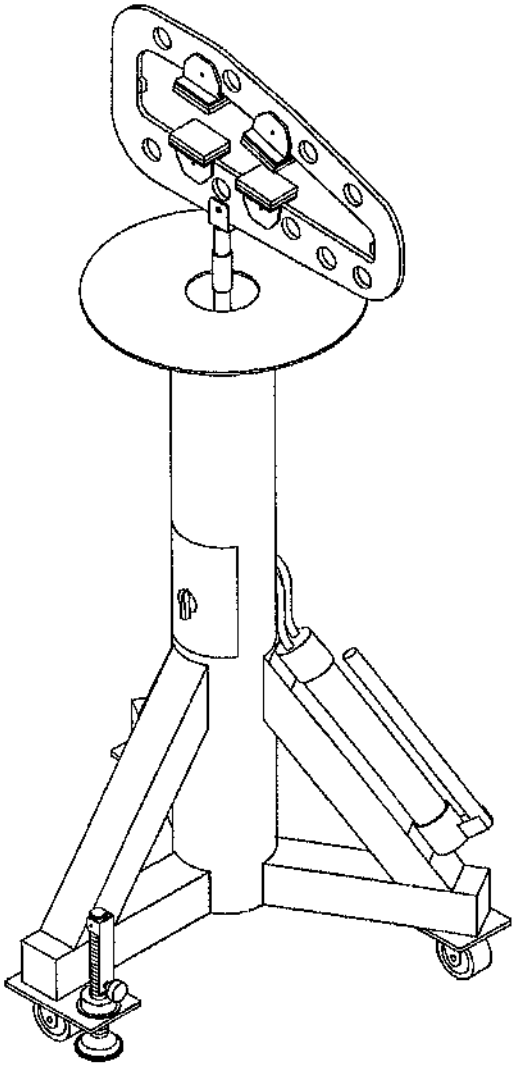
e. Verify that the forward wings are in the full forward position ±5°.

f. Set up the loading fixture to the forward wing to be checked. Refer to Figure 4 for positioning the loading fixture.

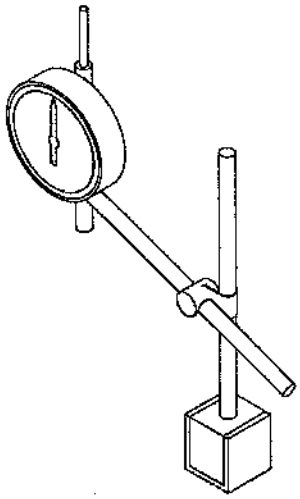
g. Set up the dial indicator under the aft fastener on the underside of the wing tip fairing (refer to Figure 4). Place double-sided tape on the wing contacting surface, or otherwise ensure that the dial indicator arm movement will not slide inboard or outboard.

h. Using the loading fixture and the pounds-of-force indicating device, apply a 250-pound load in the down direction. Slowly and steadily reverse the load direction until a load of 50 pounds in the up direction is obtained. This condition is identified as the "Zero Position" since this initial 50-pound load is required to balance the weight of the forward wing.

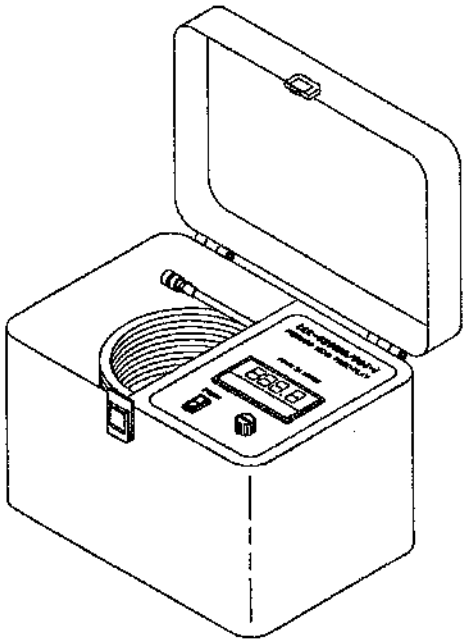
i. Position the dial indicator vertically so that a 2-inch deflection up and down may be measured. Zero the dial indicator. Record the dial indicator reading on the Free-Play Determination Form (Step #2).



LOADING FIXTURE



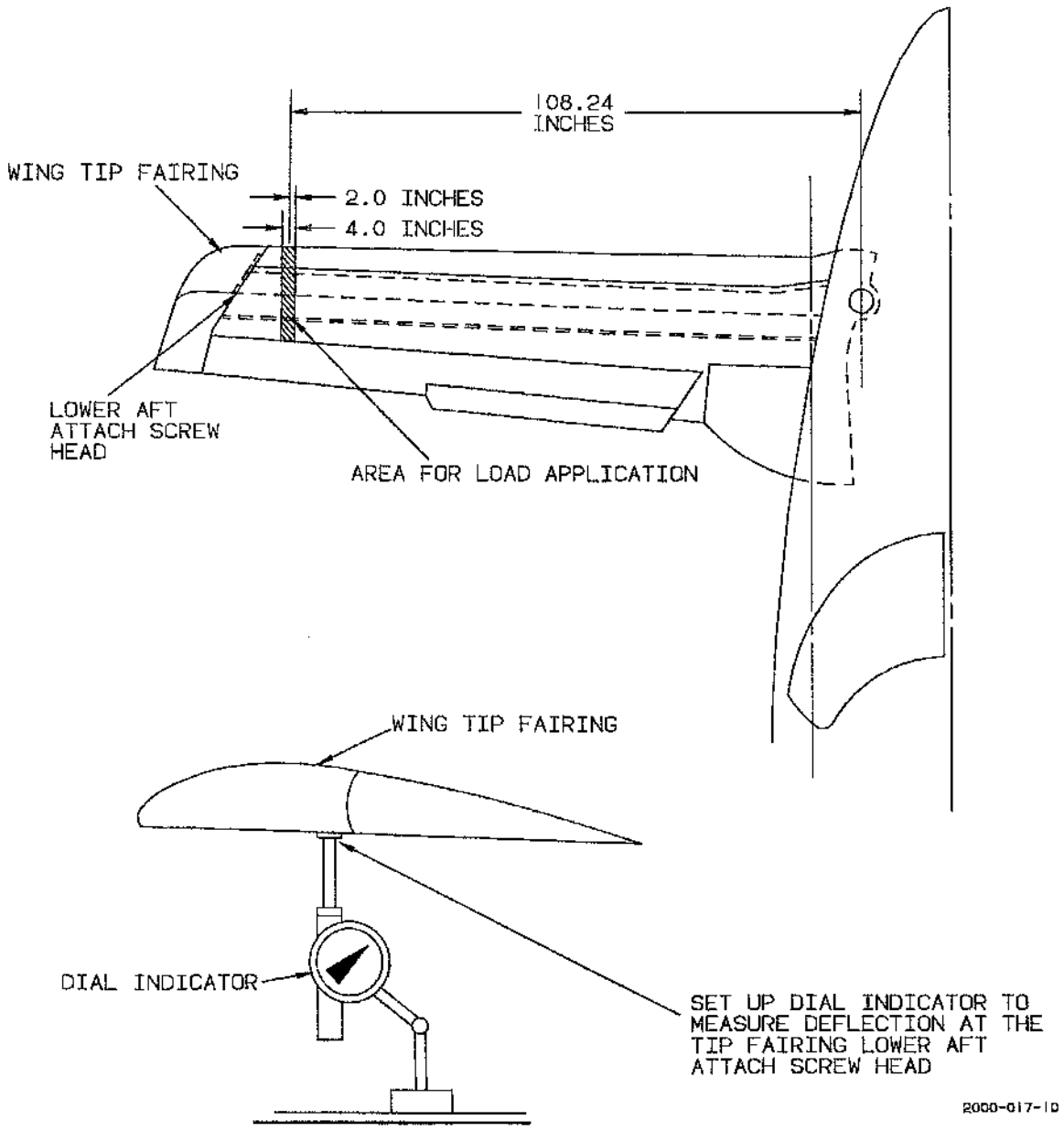
DIAL INDICATOR



POUNDS OF FORCE INDICATOR

2000-017-11

Free-Play Check Equipment (Effectivity: All)
Figure 3



Forward Wing Free-Play Equipment Setup (Effectivity: All)
Figure 4

CAUTION

Do not subject the forward wing to a rapid impulse load when applying the pounds of force loads. Incorrect readings may result.

NOTE

Do not take the readings out of sequence. Should any variation in the procedure occur while checking the deflection in the up direction, restart the procedure at step h.

j. Slowly and carefully apply an additional 200-pound load in the up direction, making a total of 250 pounds up and a net load of 200 pounds up from the "zero position". Record the dial indicator reading on the Free-Play Determination Form (Step #3).

k. Slowly and carefully add an additional 100-pound load in the up direction, up to the final 350-pound load (300 pounds of net load). Record the dial indicator reading on the Free-Play Determination Form (Step #4).

l. Prepare for the downward defection loading by slowly and gradually unloading the wing to return it to the "Zero Position" (50-pound upward load). The reading on the dial indicator will not match the original 50-pound up-load reading due to the hysteresis effect. Read the dial indicator and record the dial indicator reading on the Free-Play Determination Form (Step #5).

m. Slowly and carefully remove the upward load of 50 pounds.

NOTE

Do not take the readings out of sequence. Should any variation in the procedure occur while checking the deflection in the down direction, restart the procedure by loading the wing in the up direction until a load of 350 pounds is obtained. Take no dial indicator reading at this time. Slowly and carefully reduce the load until a load of 50 pounds up is obtained. Reset the dial indicator; this is to be the "Zero Position" for downward deflection measurements; record the dial indicator reading on the Free-Play Determination Form (Step #5) and proceed with step n.

n. Slowly and carefully add a 150-pound load in the down direction (net load of 200 pounds) and record the dial indicator reading on the Free-PLay Determination Form (Step #6).

o. Slowly and carefully add an additional 100-pound load in the down direction, up to the final 250-pound load (300 pounds at net load). Record the dial indicator reading on the Free-Play Determination Form (Step #7).

p. Slowly and gradually unload the forward wing.

q. Determine the free-play value for each forward wing numerically by using the deflection values recorded on the Free-Play Determination Form. This data is obtained in a manner which will allow the free-play to be calculated totally from data taken in the up direction only. This free-play value shall be subsequently verified by calculating the total free-play with the down direction data and comparing the two values.

1. Determine the up direction free-play value using the following formula: X = 3(A) - 2(B), where A and B are from Steps #3 and #4 of the Free-Play Determination Form.

2. Determine the down direction free-play value using the following formula: Y = 3(C) - 2(D), where C and D are from Steps #6 and #7 of the Free-Play Determination Form.

r. Treating each forward wing separately, the free-play is acceptable if the following conditions are met:

1. If the value of X is less than or equal to .350.

2. If the value of Y is less than or equal to .350.

3. If the comparison of X to Y shows that the values agree to within a tolerance of .125.

s. In the event that the forward wing does not meet free-play limits accomplish the following;

1. To counterbalance the weight of the forward wing and reduce any binding at the pivot pin (1, Figure 2), place a support under the forward wing. A 50-pound upward force at the location shown in Figure 4 will place the forward wing in the neutral up/down position.

NOTE

It is recommended that the wing support be on rollers with adjustable legs for height.

AIRCRAFT IDENTIFICATION:

SAMPLE ONLY

SERIAL # = NC-4
TEST DATE = 6-2-89
FLIGHT HRS = 0
TEMPERATURE = 57° F

LEFT HAND FWD WING

STEP	APPLIED LOAD (LBS)	NET LOAD-REF ONLY(LBS)	DIRECTION	DIAL INDICATOR READING (IN)	DEFLECTION OF WING (IN)
#1	250	300	DOWN	NO READING	
#2	50	0	UP	2.000	ZERO POSITION
#3	250	200	UP	1.051	A= .949
#4	350	300	UP	.689	B= 1.311
#5	50	0	UP	1.850	ZERO POSITION
#6	150	200	DOWN	2.662	C= .812
#7	250	300	DOWN	3.034	D= 1.184

RIGHT HAND FWD WING

STEP	APPLIED LOAD (LBS)	NET LOAD-REF ONLY(LBS)	DIRECTION	DIAL INDICATOR READING (IN)	DEFLECTION OF WING (IN)
#1	250	300	DOWN	NO READING	
#2	50	0	UP	2.000	ZERO POSITION
#3	250	200	UP	1.206	A= .794
#4	350	300	UP	.865	B= 1.135
#5	50	0	UP	1.900	ZERO POSITION
#6	150	200	DOWN	2.701	C= .801
#7	250	300	DOWN	3.057	D= 1.157

FREE-PLAY CALCULATION:

LH FWD WING- X = 3(A)-2(B) = 3(.949)-2(1.311) = .225 Y = 3(C)-2(D) = 3(.812)-2(1.184) = .068 X - Y = .157

RH FWD WING- X = 3(A)-2(B) = 3(.794)-2(1.135) = .112 Y = 3(C)-2(D) = 3(.801)-2(1.157) = .089 X - Y = .023

NOTE: LH FWD WING NUMBERS ILLUSTRATES BAD/FAIL READINGS.

RH FWD WING NUMBERS ILLUSTRATES GOOD/PASS READINGS.

C94NC55B0010

Sample Free-Play Determination Form (Effectivity: All)
Figure 5

AIRCRAFT IDENTIFICATION:

SERIAL # = _____
TEST DATE = _____
FLIGHT HRS = _____
TEMPERATURE = _____

LEFT HAND FWD WING					
STEP	APPLIED LOAD (LBS)	NET LOAD-REF ONLY(LBS)	DIRECTION	DIAL INDICATOR READING (IN)	DEFLECTION OF WING (IN)
#1	250	300	DOWN	NO READING	
#2	50	0	UP		ZERO POSITION
#3	250	200	UP		A=
#4	350	300	UP		B=
#5	50	0	UP		ZERO POSITION
#6	150	200	DOWN		C=
#7	250	300	DOWN		D=

RIGHT HAND FWD WING					
STEP	APPLIED LOAD (LBS)	NET LOAD-REF ONLY(LBS)	DIRECTION	DIAL INDICATOR READING (IN)	DEFLECTION OF WING (IN)
#1	250	300	DOWN	NO READING	
#2	50	0	UP		ZERO POSITION
#3	250	200	UP		A=
#4	350	300	UP		B=
#5	50	0	UP		ZERO POSITION
#6	150	200	DOWN		C=
#7	250	300	DOWN		D=

FREE-PLAY CALCULATION:

LH FWD WING- X = 3(A)-2(B) Y = 3(C)-2(D) X - Y =

RH FWD WING- X = 3(A)-2(B) Y = 3(C)-2(D) X - Y =

C94NC55B0009

Free-Play Determination Form (Effectivity: All)
Figure 6

2. Remove the safety wire and loosen the lower nut (9, Figure 2) counterclockwise as required to provide a minimum .10-inch gap between the lower nut and the ball of the lower pivot bearing (2, Figure 2).

3. Remove the safety wire and turn the middle flange nut (8, Figure 2) counterclockwise 1 to 1-1/2 full revolutions to withdraw the lower tapered collar (11, Figure 2) from the lower bonded spar bushing (6, Figure 2).

4. Remove the safety wire and loosen the pivot pin nut (4, Figure 2) by turning it clockwise using a C-wrench (2, Chart 1, 55-00-00) or equivalent.

5. Remove the safety wire and loosen the 3/8-inch bolts locking the pivot pin (1, Figure 2) and the forward wing spacer fitting (5, Figure 2).

6. From inside the pivot pin (1, Figure 2), loosen the upper and lower set screws (12, Figure 2) connecting the pivot pin to the upper and lower pivot bearings (2, Figure 2).

7. Refer to FORWARD WING PIVOT PIN TORQUING PROCEDURE in this chapter and retorque the pivot pin hardware.

8. After the torquing procedure has been completed, perform this free play check again for the affected wing.

VORTEX GENERATOR REPLACEMENT (Effectivity: All) (Figure 1)

Vortex generators are bonded along the upper surface of the forward wings to increase control effectiveness of the elevators in flight. The 23 aluminum vortex generators are spaced 5.00 inches apart.

NOTE

Replacement of a vortex generator(s) is required when an installed vortex generator is damaged, broken or missing. Replacement of the vortex generators is typical.

a. With a straightedge along the horizontal and vertical edges of the vortex generator that is to be replaced, mark straight lines extending well beyond the vortex generator to identify the position for the new vortex generator, or refer to Figure 1 to properly locate the vortex generators.

WARNING

When using a heat gun, hold the gun approximately twelve inches away from the vortex generator and do not allow the surface temperature to exceed 190° F.

b. Using a heat gun, apply heat (160° F nominal) to the vortex generator to soften the adhesive. Continually monitor the surface temperature and do not allow it to exceed 190° F.

c. When the adhesive softens, use a nonmetallic scraper to remove the vortex generator from the forward wing. Using the scraper remove all remaining adhesive while it is still soft.

d. Scuff sand, with a maximum of 180-grit aluminum-oxide sandpaper, an area slightly larger than the bonding surface where the new vortex generator is to be installed.

NOTE

Do not sand the facesheet deep enough to cut the composite fibers.

e. Scuff sand the bonding surface on the new vortex generator.

f. Clean the bonding surfaces using a clean, white, cloth dampened with solvent (1 or 2, Chart 2, 55-00-00) and follow immediately with a dry clean white cloth. Do not allow the solvent to evaporate on the surface.

NOTE

The surface to receive the adhesive must be clean for proper adhesion. Any type of contamination on the surface will not allow the adhesive to adhere to the surface.

g. With a straightedge held along the locator lines previously drawn, extend the lines to intersect the position for the new vortex generator, or refer to Figure 1 for locating the new vortex generator.

h. Mix Part A and Part B adhesive (3, Chart 2, 55-00-00) per instructions on the container. If desired, a thickening agent (4, Chart 2, 55-00-00) may be added to the adhesive. No more than 5 percent by weight of the thickening agent should be added to the adhesive.

i. Apply the adhesive to the bonding surfaces.

j. Cut a piece of scrim cloth the same size as the vortex generator bonding surface and align it on the vortex generator.

k. Align the vortex generator with the locator marks and use tape across the vortex generator which will apply a light pressure to the bond.

l. Cure the adhesive under heat lamps for 2 hours at a constant temperature of 130° to 150° F. After 2 hours, shut off the heat lamps and allow the surface to cool to ambient temperature before removing the tape.

WARNING

While warming the surface with heat lamps, care must be taken to ensure that the surface temperature does not exceed 190° F. The surface temperature must be monitored at all times with a temperature sensing device.

ELEVATOR - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

ELEVATOR (Effectivity: NC-4 Thru NC-23) (Figure 1)

An elevator and trim tab are mounted on each forward wing. Each elevator is constructed of woven cloth graphite and honeycomb core. The honeycomb core skins cover inboard and outboard ribs and five riblets located in the leading edge. Three hinge assemblies mate with the forward wing hinge brackets. A moisture drain hole is located in the lower outboard aft corner of each elevator.

Two static discharger bases are mounted on each elevator outboard trailing edge. Static discharge wicks thread into the discharger bases. A ground strap is installed between the elevator outboard hinge assembly and the hinge bracket on the forward wing.

A permanent counterbalance weight is mounted on the outboard side of the elevator tip fairing closeout rib. Trim weights mounted adjacent to the counterbalance weight are added and removed to balance the elevator. Both the trim weights and the counterbalance weight are accessible when the elevator tip fairing is removed.

The elevator trim tabs are hinged on the elevators with piano type hinges and hinge pins. The trim tabs are fabricated of woven graphite and honeycomb core. Movement of the trim tab is accomplished by an electrical motor-driven actuator mounted in the elevator leading edge and connected to the trim tab with push-pull rods.

ELEVATOR (Effectivity: NC-24 and After) (Figure 2)

An elevator and trim tab are mounted on each forward wing. Each elevator is constructed of woven cloth graphite and honeycomb core. The honeycomb core skins cover inboard and outboard ribs and five riblets located in the leading edge. Three hinge assemblies mate with the forward wing hinge brackets. A moisture drain hole is located in the lower outboard aft corner of each elevator.

Two static discharger bases are mounted on each elevator outboard trailing edge. Static discharge wicks thread into the discharger bases. A ground strap is installed between the elevator outboard hinge assembly and the hinge bracket on the forward wing.

Two counterbalance weights are mounted on the outboard side of the elevator tip fairing closeout rib. Trim weights mounted adjacent to the counterbalance weights are added and removed to balance the elevator. If necessary, one of the counterbalance weights can be removed to facilitate balancing elevator. Both the trim weights and the counterbalance weights are accessible when the elevator tip fairing is removed.

The elevator trim tabs are hinged on the elevators with piano type hinges and hinge pins. The trim tabs are fabricated of woven graphite and honeycomb core. Movement of the trim tab is accomplished by an electrical motor-driven actuator mounted in the elevator leading edge and connected to the trim tab with push-pull rods.

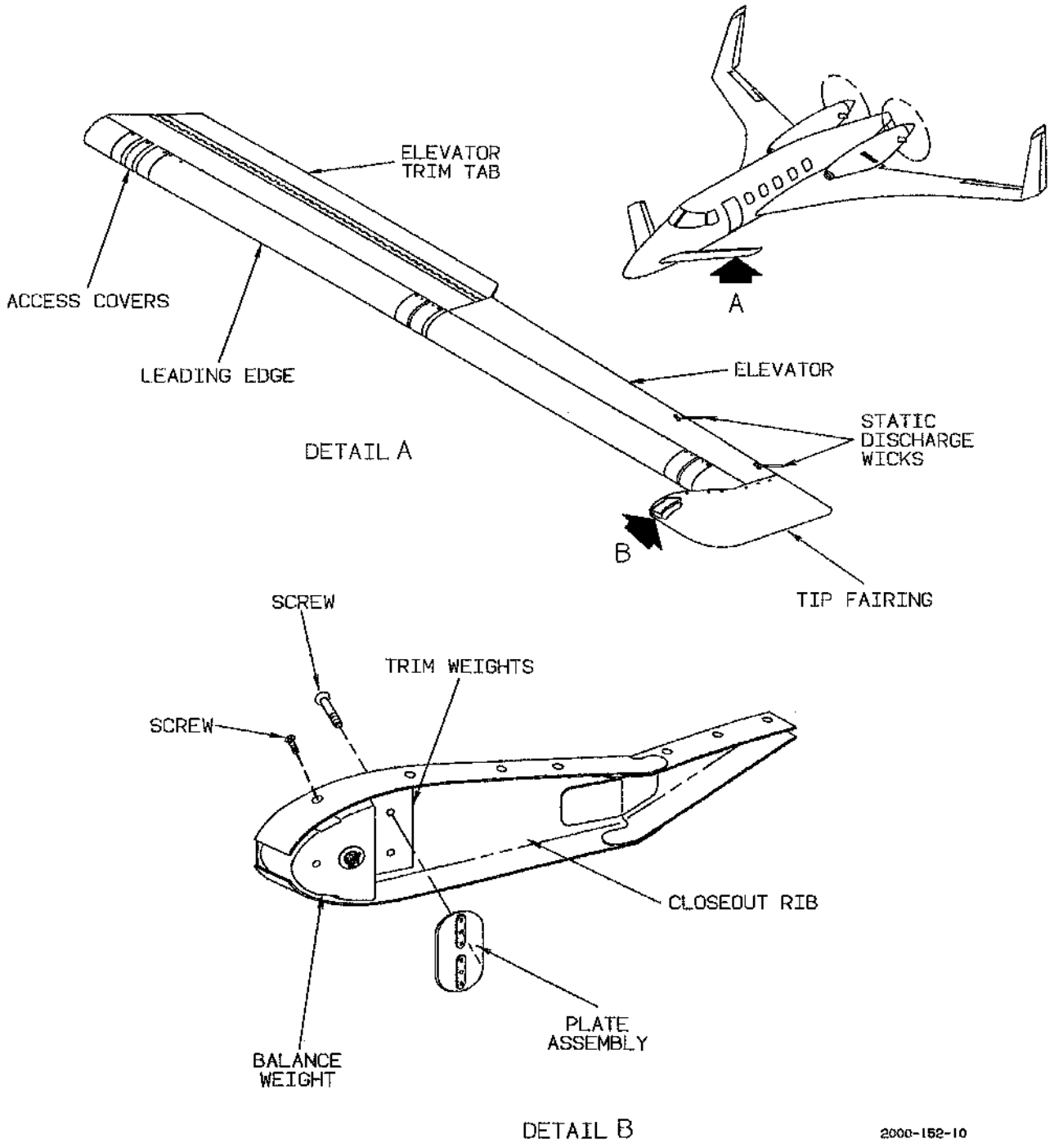
ELEVATOR - MAINTENANCE PRACTICES (Effectivity: All)

NOTE

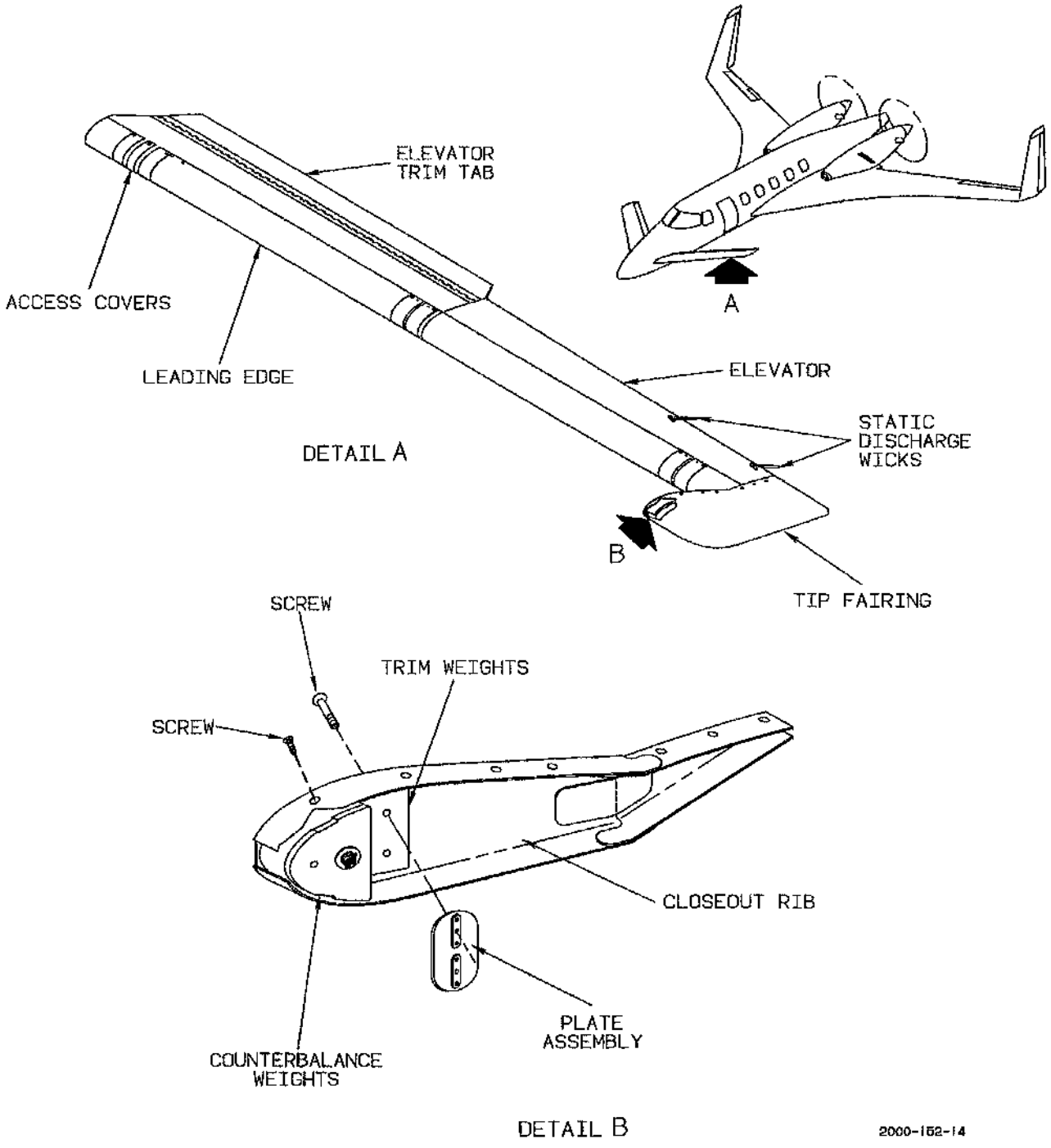
For removal/installation and maintenance of the elevators and their trim tab control systems, refer to Chapter 27-30-00.

ELEVATOR BALANCING (Effectivity: All)

This procedure provides the information necessary to measure the amount of underbalance (tail heavy) or overbalance (nose heavy) of the elevators and to



Elevator and Balance Weights (Effectivity: NC-4 Thru NC-23)
Figure 1



Elevator and Balance Weights (Effectivity: NC-24 and After)
Figure 2

properly balance the elevators. Although many different methods of check balancing exist, counterbalancing is the simplest. Counterbalancing is the application of a known force or weight at a measured distance from the hinge line to counter the unbalance moment of the control surface. The static moment of the elevator is determined by multiplying the unbalanced weight of the elevator assembly by the perpendicular distance from the hinge centerline to the center of gravity when the elevator trailing edge is in a 14 degrees down position. The weight is measured in pounds and the distance is measured in inches. The elevator must be in the following configuration prior to balancing:

- The hinge attachments must be complete with the same bolts, nuts and washers that are required for attachment on the airplane.

- The trim tab, trim tab actuator and push-pull rods must be installed with the trim tab positioned in the neutral position.

- All access covers must be installed and secured with all fasteners.

- All static discharge wicks must be installed (not temporarily taped or laid upon the surface).

- The elevator must be finish painted, including stripes and touch-up.

NOTE

Whenever the elevator and/or trim tab is replaced, modified, reworked or painted in any way, or the trim tab actuator is replaced, the elevator must be balance checked to ensure that its static moment about the hinge line is within limits. When a new or replacement elevator is to be installed, a balance check must be made just prior to installation.

EQUIPMENT REQUIRED TO PERFORM CHECK BALANCING BY COUNTERBALANCING METHOD (Effectivity: All)

The following equipment is required to check balance an elevator using the counterbalancing method:

a. A stand with knife edge supports (8, Chart 1, 55-00-00). The knife edges must be in the same horizontal plane.

b. A paper cup or similar light weight container.

c. Approximately 7 pounds of lead shot.

d. A certified beam balance weighing device calibrated in units of .001 pound or less (9, Chart 1, 55-00-00).

e. A ruler and a control surface inclinometer (10, Chart 1, 55-00-00).

f. A position indicator stand (11, Chart 1, 55-00-00).

ELEVATOR BALANCE CHECK (COUNTERBALANCE METHOD) (Effectivity: NC-4 Thru NC-23) (Figure 3)

The static moment of a 100 percent balanced elevator assembly is 0.0 pound-inches. Tail heaviness indicates static underbalance while nose heaviness indicates static overbalance.

NOTE

The static moment of all elevator assemblies must fall within the range of 9.20 to 12.20 pound-inches tail heavy at the measured moment about the hinge line.

a. Verify that the trim tab, trim tab actuator and push-pull rods are installed with the trim tab positioned in neutral.

b. Verify that all access covers are installed and secured with all fasteners.

c. Verify that all static discharge wicks are installed (not temporarily taped or laid upon the surface).

d. The elevator must be finish painted, including stripes and touch-up.

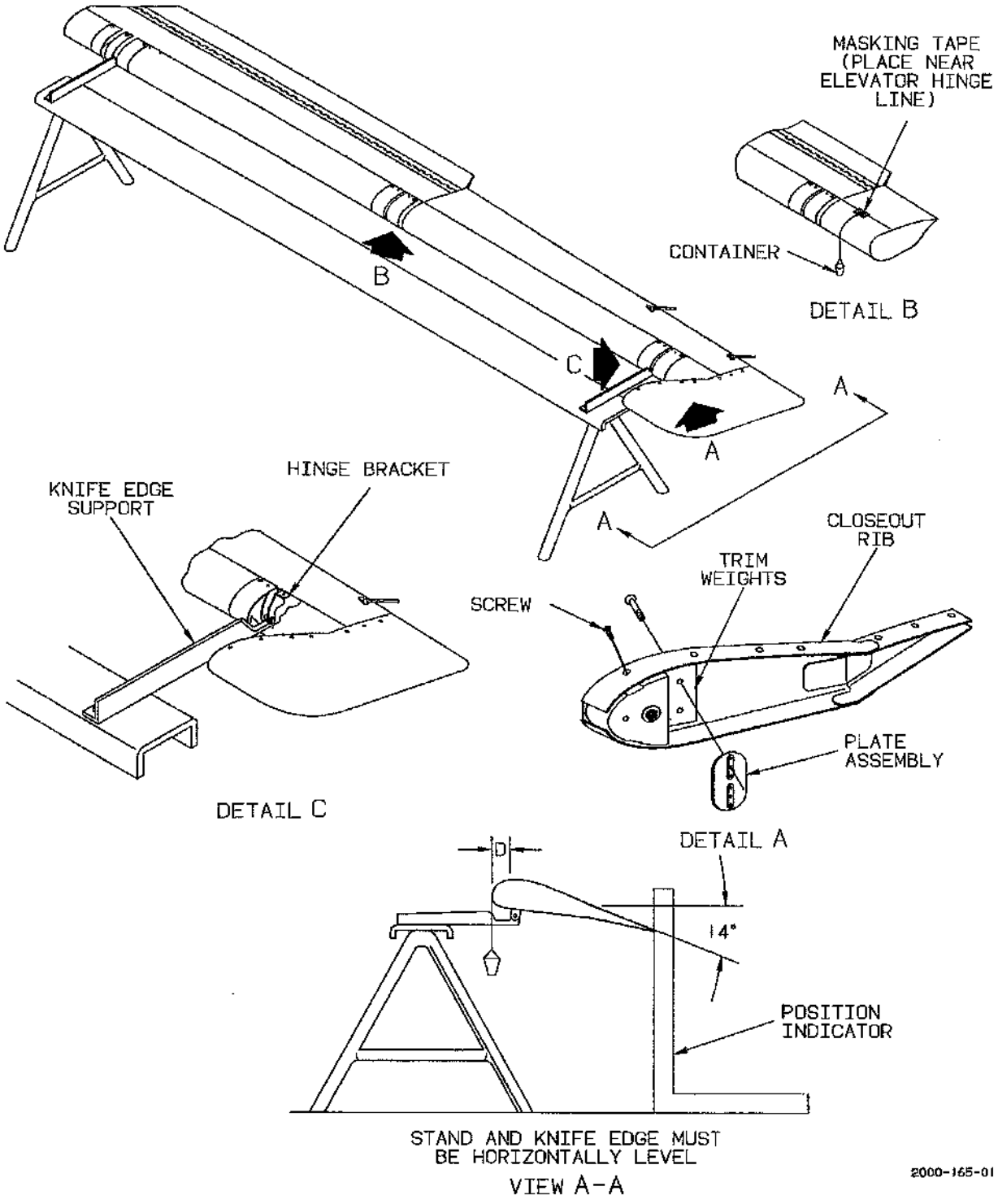
e. Verify that the hinge attach bolts, washers and nuts are assembled in the hinges, and mount the elevator on the knife edge supports. Ascertain that the elevator is free to rotate about the hinge centerline.

f. With the trailing edge in the 14 degrees down position as measured with the inclinometer on the upper skin, mark the trailing edge position on the indicator stand with a marking pencil.

g. To determine if trim weights should be added or removed, suspend a paper cup from the elevator leading edge. Use a short length of small diameter string secured to the surface with a small piece of masking tape as shown in Figure 3. The cup must be free to hang vertically.

NOTE

Static balance must be checked in a draft free area.



Elevator Static Balancing (Effectivity: NC-4 Thru NC-23)
Figure 3

h. Add small quantities of lead shot to the cup until the elevator balances in the 14 degrees trailing edge down position marked on the indicator stand.

i. Carefully measure the perpendicular distance "D" within 0.1 inch from the hinge centerline to the point of suspension of the cup.

j. Remove the cup, contents and string, then weigh them to within .005-pound.

NOTE

Since any weighing error is magnified by the distance "D", weighing the cup, contents and string is most important and must be done carefully on scales that are certified for accuracy.

k. Calculate the elevator static balance from the formula M = W x D as follows:

1. The weight of the cup and contents is designated by "W".

2. The distance between the centerline of the hinge and the suspension point of the cup is designated by "D".

3. The overbalance or underbalance moment is designated by "M".

l. The following is a typical example of a balancing calculation:

1. Assume the elevator is slightly underbalanced (tail heavy) and the paper cup was suspended from the leading edge.

2. If the elevator balances at (W = 6.69 pounds) and (D = 1.6 inches), then. . .M = 6.69 x 1.6.

3. M = 10.7 pound-inches (the product of "W x D" must be accurate to within 0.05 pound-inches). In this instance, "M" is within the required static balance range and is considered acceptable.

m. If the elevator static balance does not fall within the range of 9.20 to 12.20 pound-inches tail heavy (underbalance), trim weights must be added or removed and the balance rechecked.

n. If the trim weight requires adjustment, remove the screws securing the elevator tip fairing to the elevator and remove the fairing. Cut the safety wire between the two screws and back the screws out of the nut plates installed in the plate assembly. Remove the plate assembly. Add or remove trim weights as required. A maximum of twelve trim weights (P/N 122-610057-27) may be used on each elevator. Secure the trim weights to the closeout rib with the screws and plate assembly. Safety wire the screws together and install the tip fairing.

ELEVATOR BALANCE CHECK (COUNTERBALANCE METHOD) (Effectivity: NC-24 and After) (Figure 4)

The static moment of a 100 percent balanced elevator assembly is 0.0 pound-inches. Tail heaviness indicates static underbalance while nose heaviness indicates static overbalance.

NOTE

The static moment of all elevator assemblies must fall within the range of 9.20 to 12.20 pound-inches tail heavy at the measured moment about the hinge line.

a. Verify that the trim tab, trim tab actuator and push-pull rods are installed with the trim tab positioned in neutral.

b. Verify that all access covers are installed and secured with all fasteners.

c. Verify that all static discharge wicks are installed (not temporarily taped or laid upon the surface).

d. The elevator must be finish painted, including stripes and touch-up.

e. Verify that the hinge attach bolts, washers and nuts are assembled in the hinges, and mount the elevator on the knife edge supports. Ascertain that the elevator is free to rotate about the hinge centerline.

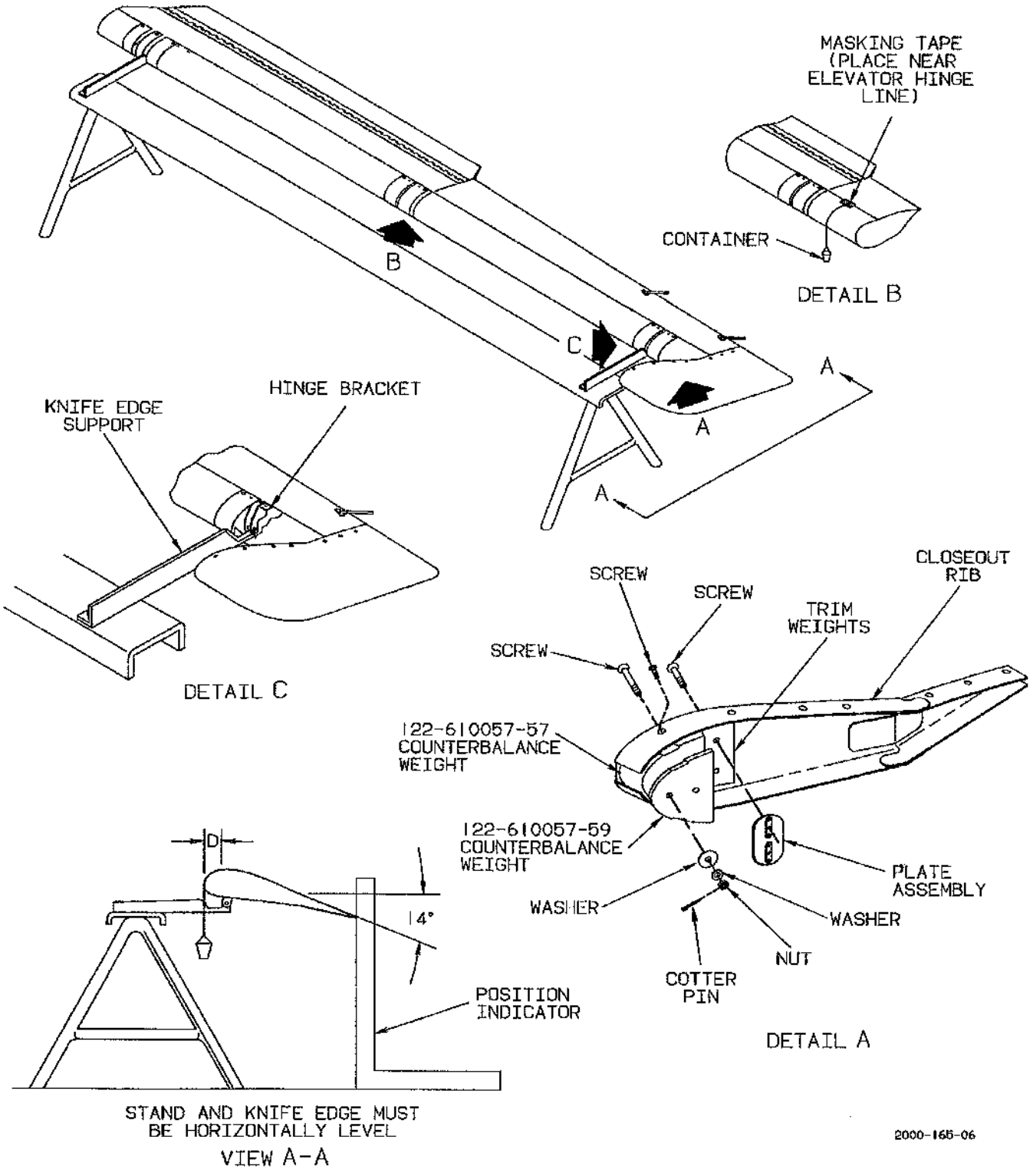
f. With the trailing edge in the 14 degrees down position as measured with the inclinometer on the upper skin, mark the trailing edge position on the indicator stand with a marking pencil.

g. To determine if trim weights should be added or removed, suspend a paper cup from the elevator leading edge. Use a short length of small diameter string secured to the surface with a small piece of masking tape as shown in Figure 4. The cup must be free to hang vertically.

NOTE

Static balance must be checked in a draft free area.

h. Add small quantities of lead shot to the cup until the elevator balances in the 14 degrees trailing edge down position marked on the indicator stand.



Elevator Static Balancing (Effectivity: NC-24 and After)
Figure 4

i. Carefully measure the perpendicular distance "D" within 0.1 inch from the hinge centerline to the point of suspension of the cup.

j. Remove the cup, contents and string, then weigh them to within .005 pound.

NOTE

Since any weighing error is magnified by the distance "D", weighing the cup, contents and string is most important and must be done carefully on scales that are certified for accuracy.

k. Calculate the elevator static balance from the formula M = W x D as follows:

1. The weight of the cup and contents is designated by "W".

2. The distance between the centerline of the hinge and the suspension point of the cup is designated by "D".

3. The overbalance or underbalance moment is designated by "M".

l. The following is a typical example of a balancing calculation:

1. Assume the elevator is slightly underbalanced (tail heavy) and the paper cup was suspended from the leading edge.

2. If the elevator balances at (W = 6.69 pounds) and (D = 1.6 inches), then. . .M = 6.69 x 1.6.

3. M = 10.7 pound-inches (the product of "W x D" must be accurate to within 0.05 pound-inches). In this instance, "M" is within the required static balance range and is considered acceptable.

m. If the elevator static balance does not fall within the range of 9.20 to 12.20 pound-inches tail heavy (underbalance), trim weights must be added or removed and the balance rechecked.

n. If the trim weight requires adjustment, remove the screws securing the elevator tip fairing to the elevator and remove the fairing. Cut the safety wire between the two screws and back the screws out of the nut plates installed in the plate assembly. Remove the plate assembly. Add or remove trim weights as required. A maximum of twelve trim weights (P/N 122-610057-27) may be used on each elevator.

NOTE

If the elevator cannot be properly balanced adjusting the trim weights, it will be necessary to adjust the counterbalance weight. Two counterbalance weights are available to accomplish this, a 122-610057-59 (thin) and a 122-610057-57 (thick) weight. When adjusting the counterbalance weight, use either one -57 weight, one -59 weight, or a combination of one -57 and one -59 weight. At least one counterbalance weight must be installed on the elevator. Remove the cotter pins, nuts, washers and screws securing the counterbalance weight(s) to the closeout rib and remove the weight(s). Adjust the counterbalance as necessary and secure the weight(s) to the closeout rib with the screws, washers, nuts and cotter pins. Use the following screw and washer combinations when installing the counterbalance weights: one -57 weight, use 122-610057-29 screw with one 105090PD050-3F, one AN960-10L and two AN960-10 washers under the nut; one -59 weight, use 122-610057-61 screw with one AN960-10L and one 105090PD050-3F washer under the nut; one -57 and one -59 weight, use 122-610057-29 screw with one 105090PD050-3F and one AN960-10L washer under nut.

o. Recheck the balance of the elevator if the counterbalance weight was adjusted and then adjust the trim weights as necessary. Secure the trim weights to the closeout rib with the screws and plate assembly. Safety wire the screws together and install the tip fairing.

TIPSAILS - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The tipsails are airfoils fabricated of composite materials (woven cloth graphite). The structure of the tipsails include spars, ribs, honeycomb core skins and attach fittings. Three hinge brackets in the rudder cove provide for installation of the rudder assemblies. The tipsails are canted inboard at the top and attach to the aft wing center and aft spars with two fittings and a bracket. Pin-collar type fasteners secure the tipsail spars to wing spars at the attach fittings. Fifteen equally spaced vortex generators are bonded on the inboard side of each tipsail.

Fairings enclose the left and right tipsail attach fittings and are fabricated of Kevlar woven cloth. A landing light lens is installed forward of each fairing. A moisture drain hole is located at the lowest point of each fairing. The fairings incorporate a shelf for mounting a strobe light power supply with the strobe light located on the outboard side of each fairing. For maintenance of the strobe, navigation, recognition and landing light systems, refer to Chapter 33.

The tip assemblies enclose the top of the tipsails. The tip assemblies are fabricated using woven graphite prepreg fabric with aluminum mesh along the bottom and aft edges. Two vortex generators are bonded on the inboard side of each tip assembly to provide improved handling qualities in flight. Screws secure the tip assemblies to the tipsails.

NOTE

For maintenance of the rudder and rudder trim tab, refer to Chapter 27-20-00. For balancing the rudder, refer to Chapter 55-40-00.

TIPSAILS - MAINTENANCE PRACTICES (Effectivity: All)

TIPSAIL FAIRING REMOVAL (Effectivity: All) (Figure 1)

a. Remove all electrical power from the airplane.

b. Remove the screws attaching the tipsail lens to the the tipsail fairing, and remove the lens.

WARNING

High voltage may be present in the wiring between the strobe light power supply and strobe flash tube. Always allow at least ten minutes after both STROBE switches are set to OFF before attempting any maintenance. Disconnecting or touching the wiring before ten minutes have elapsed may result in electrical shock and/or injury to personnel.

c. Remove the screws attaching the tipsail fairing to the tipsail, then lower the tipsail fairing and disconnect the electrical connector at the tipsail fairing wire harness break.

NOTE

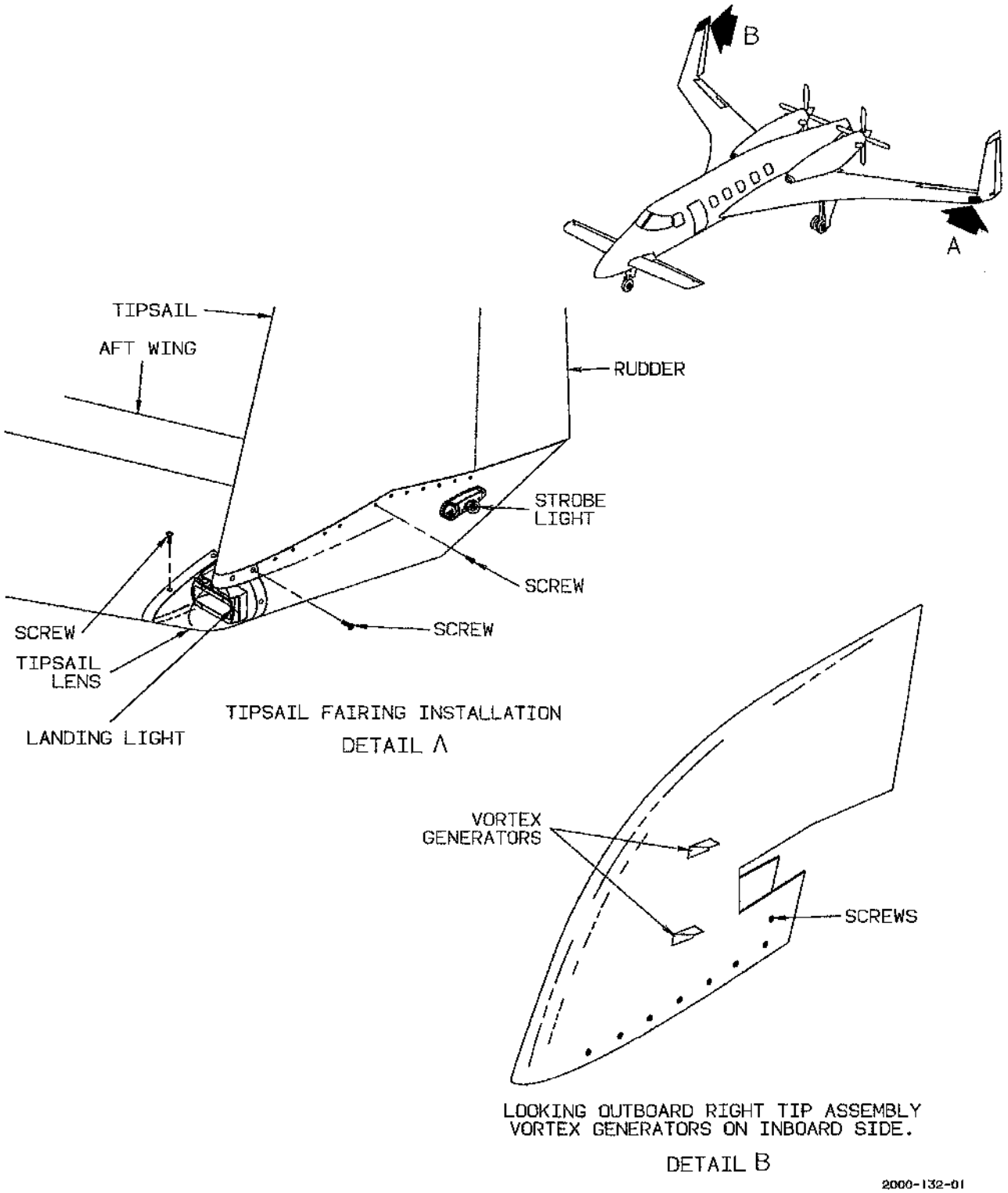
Do not allow wire harness to support the weight of the tipsail fairing.

d. Disconnect both strobe light power supply connectors at the power supply.

e. Store the tipsail fairing to protect it from damage.

TIPSAIL FAIRING INSTALLATION (Effectivity: All) (Figure 1)

a. Connect the electrical connector at the tipsail fairing wire harness break.



Tipsail Fairing and Tip Installation (Effectivity: All)
Figure 1

NOTE

Do not allow wire harness to support the weight of the tipsail fairing.

b. Connect both strobe light power supply connectors at the power supply.

c. Position and align the tipsail fairing to the tipsail.

d. Secure the tipsail fairing to the tipsail and wing tip with screws around the fairing.

e. Refer to Chapter 33-40-00 and install the tipsail lens as detailed under TIPSAIL LENS INSTALLATION.

TIPSAIL TIP ASSEMBLY REMOVAL (Effectivity: All) (Figure 1)

a. Remove the screws on the inboard and outboard sides of the tip assembly.

b. Remove the tip assembly and store to prevent damage.

TIPSAIL TIP ASSEMBLY INSTALLATION (Effectivity: All) (Figure 1)

a. Position and align the tip assembly on the tipsail.

b. Secure the tip assembly to the tipsail with screws on the inboard and outboard sides of the tip assembly.

TIPSAIL VORTEX GENERATOR REPLACEMENT (Effectivity: All) (Figure 2)

NOTE

Replacement of a vortex generator(s) is required when an installed vortex generator is damaged, broken or missing. Replacement of the vortex generators is typical.

a. With a straight edge along the horizontal and vertical edges of the vortex generator to be replaced, mark straight lines extending well beyond the vortex generator to identify the position for the new vortex generator, or refer to Figure 2 to properly locate the vortex generators.

WARNING

When using a heat gun, hold the gun approximately twelve inches away from the vortex generator and do not allow the surface temperature to exceed 190° F.

b. Using a heat gun, apply heat (160° F nominal) to the vortex generator to soften the adhesive. Continually monitor the surface temperature and do not allow it to exceed 190° F.

c. When the adhesive softens, use a non-metallic scraper to remove the vortex generator from the tipsail. Using the scraper remove all remaining adhesive while it is still soft.

d. Scuff sand, with a maximum of 180 grit aluminum oxide sandpaper, an area slightly larger than the bonding surface where the new vortex generator is to be installed.

NOTE

Do not sand the facesheet deep enough to cut the composite fibers.

e. Scuff sand the bonding surface on the new vortex generator.

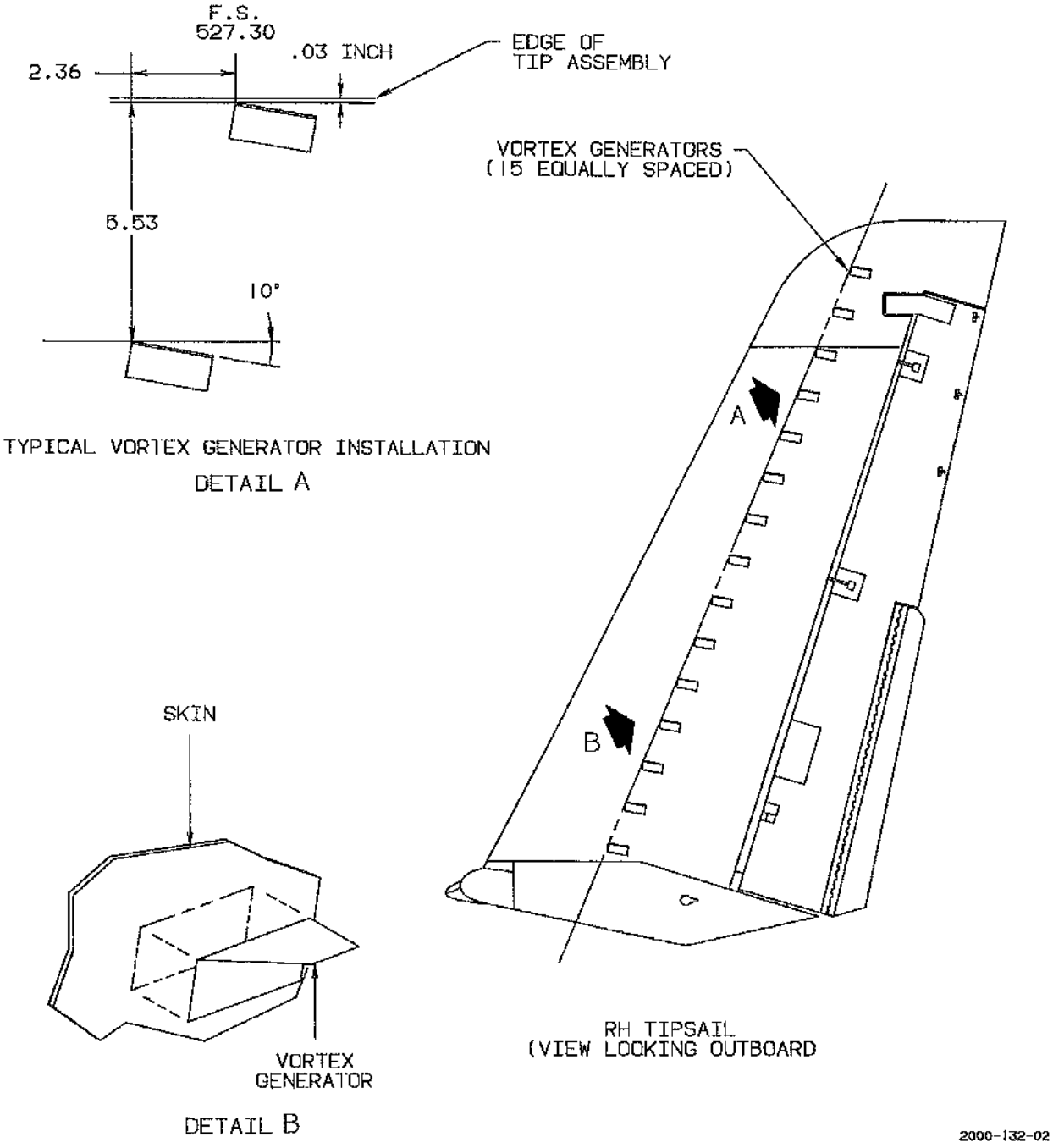
f. Clean the bonding surfaces using a clean white cloth dampened with solvent (1 or 2, Chart 2, 55-00-00) and follow immediately with a dry clean white cloth. Do not allow the solvent to evaporate on the surface.

NOTE

The surface to receive the adhesive must be clean for proper adhesive adhesion. Any type of contamination on the surface will not allow the adhesive to adhere to the surface.

g. With a straight edge held along the locator lines previously drawn, extend the lines to intersect the position for the new vortex generator, or refer to Figure 2 for locating the new vortex generator.

h. Mix Part A and Part B adhesive (3, Chart 2, 55-00-00) per instructions on the container. If desired, a thickening agent (4, Chart 2, 55-00-00) may be added to the adhesive. No more than 5 percent by weight of the thickening agent should be added to the adhesisve.



TYPICAL VORTEX GENERATOR INSTALLATION

DETAIL A

DETAIL B

RH TIPSAIL
(VIEW LOOKING OUTBOARD

Tipsail Vortex Generator Replacement (Effectivity: All)
Figure 2

i. Apply the adhesive to the bonding surfaces.

j. Cut a piece of scrim cloth the same size as the vortex generator bonding surface and align it with the vortex generator.

k. Align the vortex generator with the locator marks and use tape to apply a light pressure to the bond.

l. Cure the adhesive under heat lamps for 2 hours at a constant temperature of 130° to 150° F. After 2 hours, shut off the heat lamps and allow the surface to cool to ambient temperature before removing the tape.

WARNING

While warming the surface with heat lamps, care must be taken to ensure that the surface temperature does not exceed 190° F. The surface temperature must be monitored at all times with a temperature sensing device.

RUDDER - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

RUDDER (Effectivity: NC-4 Thru NC-25) (Figure 1)

The rudders are airfoils hinged on the trailing edge of the tipsails to provide yaw control of the airplane. Each rudder has a controllable trim tab.

Each rudder is fabricated of woven cloth graphite and honeycomb core. The rudder skins cover spars and ribs. The three rudder hinges and the trim tab actuator have access covers. Moisture drain holes are located at the upper aft corner of the rudder and in the lower closeout rib.

Three static discharge bases are bonded to the outboard side of the rudder trailing edge. Static wicks thread into the discharger bases. A grounding strap is installed between the rudder center hinge assembly and the hinge bracket of the tipsail.

The rudder trim tabs are hinged on the rudders with a piano type hinge and hinge pin. The trim tabs are fabricated of woven cloth graphite and honeycomb core. Movement of the trim tab is accomplished by an electrical motor-driven actuator mounted in the rudder leading edge and connected to the trim tab with push-pull rods. The leading edge cover between the the center and lower rudder hinges is removable to gain access to the trim tab actuator. A fairing is installed where the trim tab push-pull rods exit the rudder skin (inboard side).

Permanent balance weights are contoured to the inside of the leading edge and are secured with adhesive. A permanent balance weight is also bolted in the upper rudder fairing extending forward of the rudder. Trim weights are bolted to the lower closeout rib of the rudder and may be added or removed during balance checks.

RUDDER (Effectivity: NC-26 and After) (Figure 2)

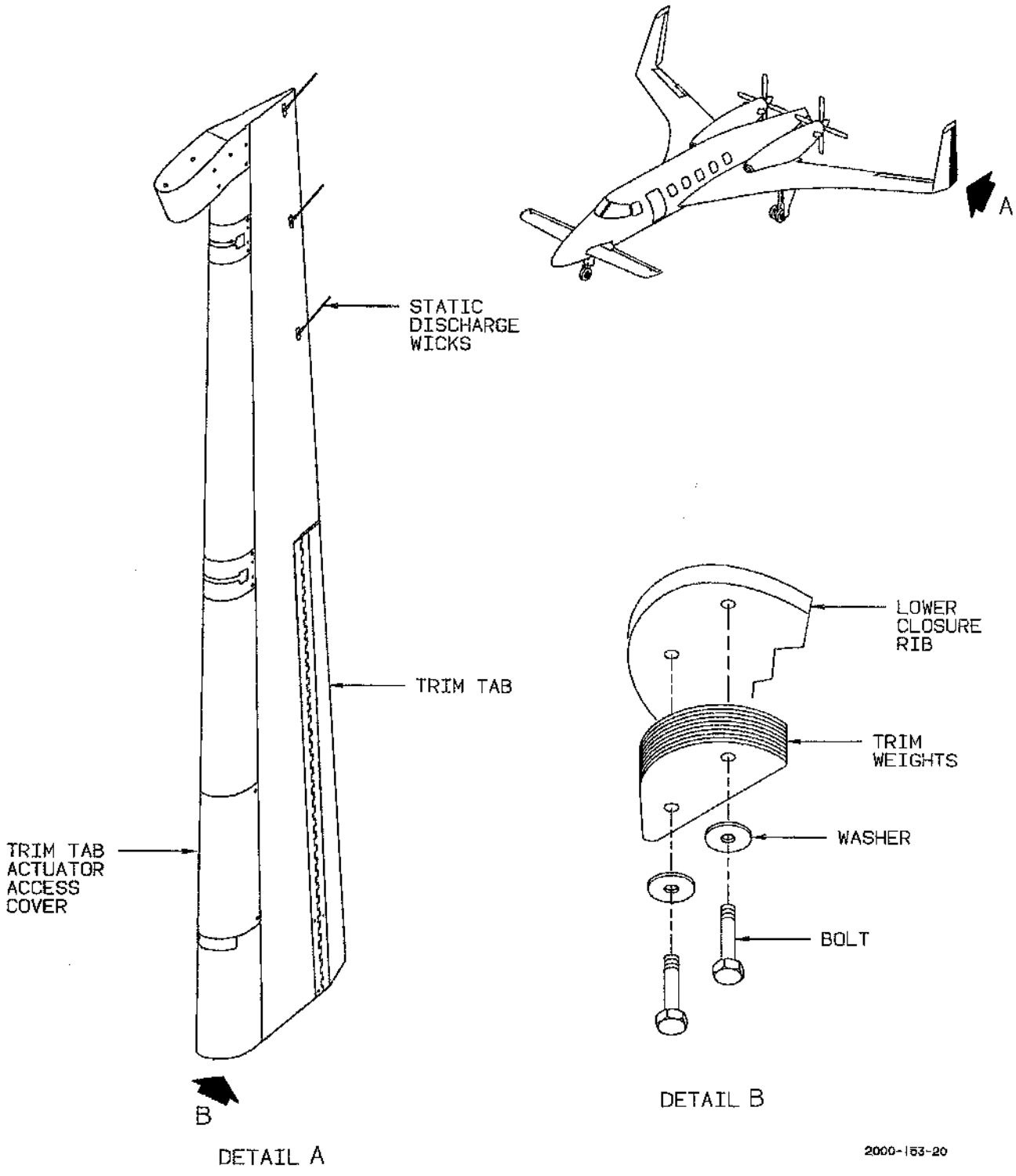
The rudders are airfoils hinged on the trailing edge of the tipsails to provide yaw control of the airplane. Each rudder has a controllable trim tab.

Each rudder is fabricated of woven cloth graphite and honeycomb core. The rudder skins cover spars and ribs. The three rudder hinges and the trim tab actuator have access covers. Moisture drain holes are located at the upper aft corner of the rudder and in the lower closeout rib.

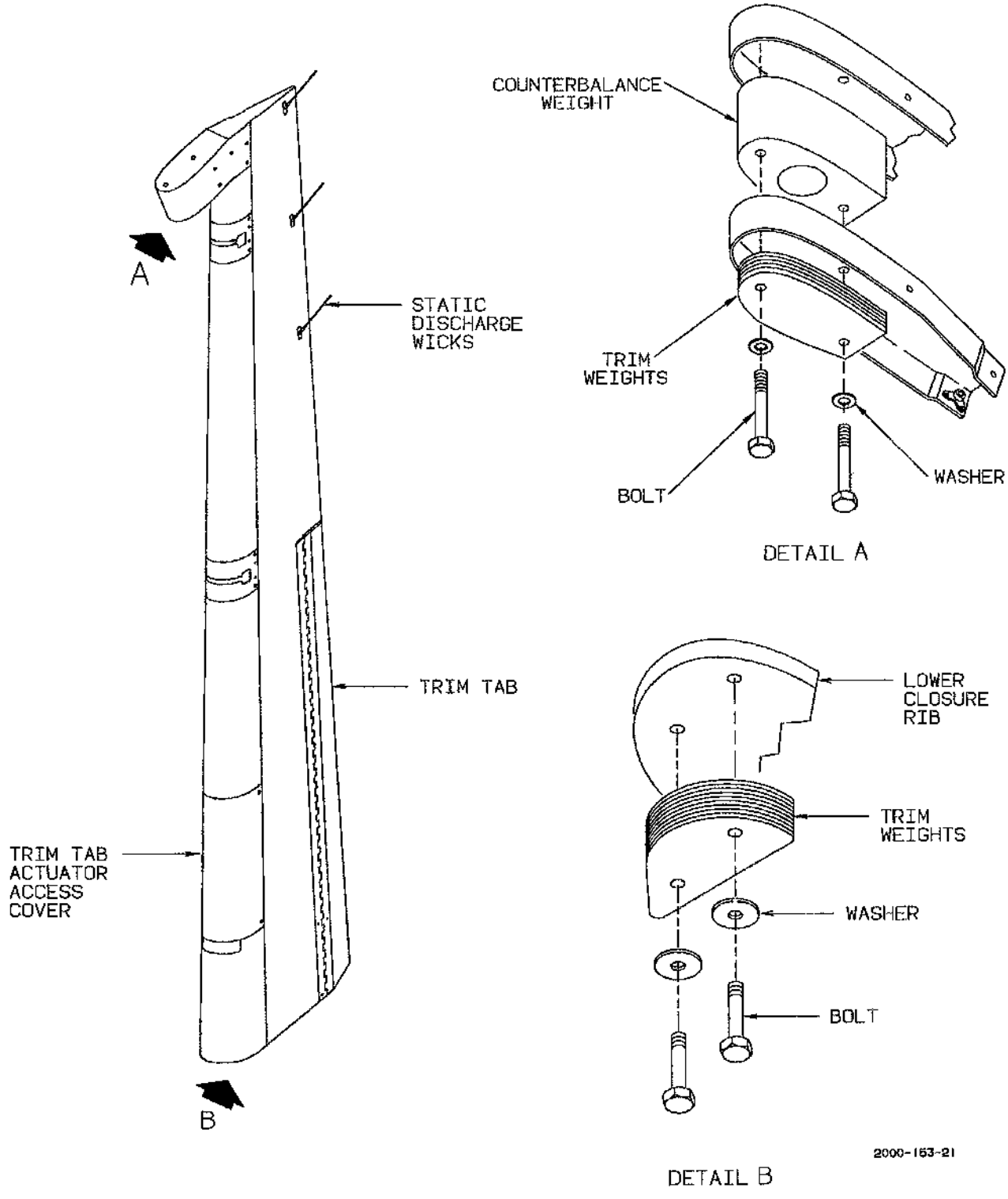
Three static discharge bases are bonded to the outboard side of the rudder trailing edge. Static wicks thread into the discharger bases. A grounding strap is installed between the rudder center hinge assembly and the hinge bracket of the tipsail.

The rudder trim tabs are hinged on the rudders with a piano type hinge and hinge pin. The trim tabs are fabricated of woven cloth graphite and honeycomb core. Movement of the trim tab is accomplished by an electrical motor-driven actuator mounted in the rudder leading edge and connected to the trim tab with push-pull rods. The leading edge cover between the the center and lower rudder hinges is removable to gain access to the trim tab actuator. A fairing is installed where the trim tab push-pull rods exit the rudder skin (inboard side).

Counterbalance weights are installed in the leading edge and in the upper fairing extending forward of the rudder. Trim weights that may be added or removed to balance the rudder are located on the lower closeout rib and and adjacent to the counterbalance weight at the top of the rudder.



Rudder and Balance Weights (Effectivity: NC-4 Thru NC-25)
Figure 1



Rudder and Balance Weights (Effectivity: NC-26 and After)
Figure 2

RUDDER - MAINTENANCE PRACTICES (Effectivity: All)

NOTE

For removal/installation and maintenance of the rudder and rudder trim tab systems, refer to Chapter 27-20-00.

RUDDER BALANCING (Effectivity: All)

This procedure provides the information necessary to measure the amount of underbalance (tail heavy) or overbalance (nose heavy) of the rudders and to properly balance the rudders. Although many different methods of check balancing exist, counterbalancing is the simplest. Counterbalancing is the application of a known force or weight at a measured distance from the hinge line to counter the unbalance moment of the control surface. The static moment of the rudder is determined by multiplying the unbalanced weight of the rudder assembly by the perpendicular distance from the hinge centerline to the center of gravity when the rudder trailing edge is in a 3.75 degrees down position. The weight is measured in pounds and the distance is measured in inches. The rudder must be in the following configuration prior to balancing:

- The hinge attachments must be complete with the same bolts, nuts and washers that are required for attachment on the airplane.
- The trim tab, trim tab actuator and push-pull rods must be installed with the trim tab positioned in the neutral position.
- All access covers must be installed and secured with all fasteners.
- All static discharge wicks must be installed (not temporarily taped or laid upon the surface).
- The rudder must be finish painted, including stripes and touch-up.

NOTE

Whenever the rudder and/or trim tab is replaced, modified, reworked or painted in any way, or the trim tab actuator is replaced, the rudder must be balance checked to ensure that its static moment about the hinge line is within limits. When a new or replacement rudder is to be installed, a balance check must be made just prior to installation.

EQUIPMENT REQUIRED TO PERFORM CHECK BALANCING BY COUNTERBALANCING METHOD (Effectivity: All)

The following equipment is required to check balance a rudder using the counterbalancing method:

a. A stand with knife edge supports. The knife edges must be in the same horizontal plane.

b. A paper cup or similar light weight container.

c. Approximately 5 pounds of lead shot.

d. A certified beam balance weighing device calibrated in units of .001 pound or less.

e. A ruler and a control surface inclinometer.

f. A position indicator stand.

RUDDER BALANCE CHECK (COUNTERBALANCE METHOD) (Effectivity: NC-4 Thru NC-25) (Figure 3)

The static moment of a 100 percent balanced rudder assembly is 0.0 pound-inches. Tail heaviness indicates static underbalance while nose heaviness indicates static overbalance.

NOTE

The static moment of all rudder assemblies must fall within the range of 7.0 to 10.5 pound-inches tail heavy (underbalance) at the measured moment about the hinge line.

a. Verify that the trim tab, trim tab actuator and push-pull rods are installed with the trim tab positioned in neutral.

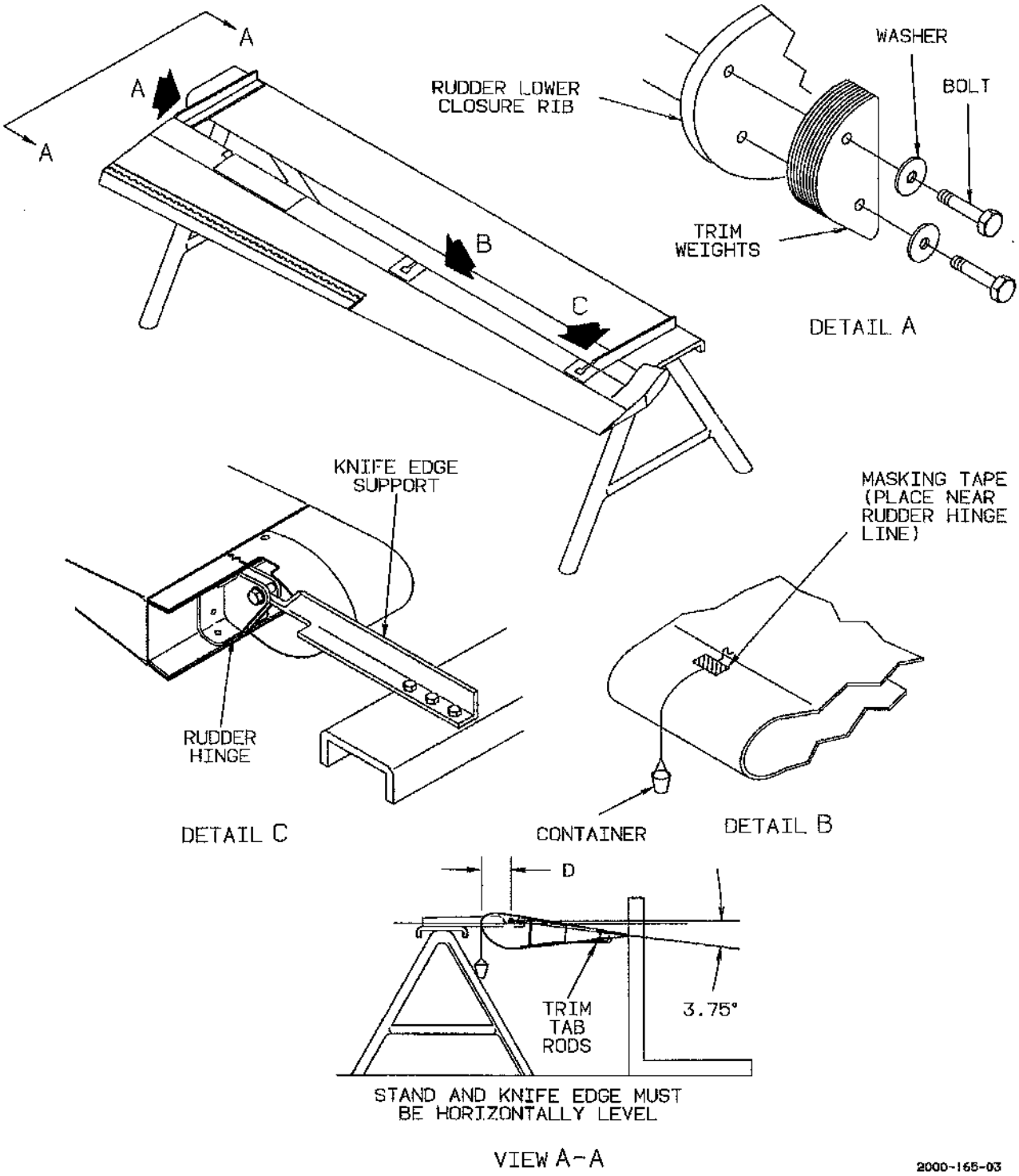
b. Verify that all access covers are installed and secured with all fasteners.

c. Verify that all static discharge wicks are installed (not temporarily taped or laid upon the surface).

d. The rudder must be finish painted, including stripes and touch-up.

e. Verify that the hinge attach bolts, washers and nuts are assembled in the hinges. Mount the rudder on the knife edge supports with the outboard surface of the rudder up. Ascertain that the rudder is free to rotate about the hinge centerline.

f. With the trailing edge in the 3.75 degrees down position as measured with the inclinometer on the outboard skin near the trailing edge, mark the trailing edge position on the indicator stand with a marking pencil.



Rudder Balancing (Effectivity: NC-4 Thru NC-25)
Figure 3

g. To determine if trim weights should be added or removed, suspend a paper cup from the rudder leading edge. Use a short length of small diameter string secured to the surface with a small piece of masking tape as shown in Figure 3. The cup must be free to hang vertically.

NOTE

Static balance must be checked in a draft free area.

h. Add small quantities of lead shot to the cup until the rudder balances in the 3.75 degrees trailing edge down position marked on the indicator stand.

i. Carefully measure the perpendicular distance "D" within 0.1 inch from the hinge centerline to the point of suspension of the cup.

j. Remove the cup, contents and string, then weigh them to within .005 pound.

NOTE

Since any weighing error is magnified by the distance "D", weighing the cup, contents and string is most important and must be done carefully on scales that are certified for accuracy.

k. Calculate the rudder static balance from the formula M = W x D as follows:

1. The weight of the cup and contents is designated by "W".

2. The distance between the centerline of the hinge and the suspension point of the cup is designated by "D".

3. The overbalance or underbalance moment is designated by "M".

l. The following is a typical example of a balancing calculation:

1. Assume the rudder is slightly underbalanced (tail heavy) and the paper cup was suspended from the leading edge.

2. If the rudder balances at W = 4.79 pounds and D = 1.8 inches, then. . .M = 4.79 x 1.8.

3. M = 8.6 pound-inches (the product of "W x D" must be accurate to within 0.05 pound-inches). In this instance, "M" is within the required static balance range and is considered acceptable.

m. If the rudder static balance does not fall within the range of 7.0 to 10.5 pound-inches tail heavy (underbalance), trim weights must be added or removed and the balance rechecked.

n. A maximum of nine trim weights (P/N 122-630042-123) may be used on each rudder. The trim weights are secured to the forward end of the lower closure rib with two AN3H10A bolts with a AN960C10 washer under each bolt head.

o. Safety wire the two bolts together after the proper number of trim weights have been added to the rudder.

RUDDER BALANCE CHECK (COUNTERBALANCE METHOD) (Effectivity: NC-26 and After) (Figure 4)

The static moment of a 100 percent balanced rudder assembly is 0.0 pound-inches. Tail heaviness indicates static underbalance while nose heaviness indicates static overbalance.

NOTE

The static moment of all rudder assemblies must fall within the range of 7.0 to 10.5 pound-inches tail heavy (underbalance) at the measured moment about the hinge line.

a. Verify that the trim tab, trim tab actuator and push-pull rods are installed with the trim tab positioned in neutral.

b. Verify that all access covers are installed and secured with all fasteners.

c. Verify that all static discharge wicks are installed (not temporarily taped or laid upon the surface).

d. The rudder must be finish painted, including stripes and touch-up.

e. Verify that the hinge attach bolts, washers and nuts are assembled in the hinges. Mount the rudder on the knife edge supports with the outboard surface of the rudder up. Ascertain that the rudder is free to rotate about the hinge centerline.

f. With the trailing edge in the 3.75 degrees down position as measured with the inclinometer on the outboard skin near the trailing edge, mark the trailing edge position on the indicator stand with a marking pencil.

g. To determine if trim weights should be added or removed, suspend a paper cup from the rudder leading edge. Use a short length of small diameter string secured to the surface with a small piece of masking tape as shown in Figure 4. The cup must be free to hang vertically.

NOTE

Static balance must be checked in a draft free area.

h. Add small quantities of lead shot to the cup until the rudder balances in the 3.75 degrees trailing edge down position marked on the indicator stand.

i. Carefully measure the perpendicular distance "D" within 0.1 inch from the hinge centerline to the point of suspension of the cup.

j. Remove the cup, contents and string, then weigh them to within .005 pound.

NOTE

Since any weighing error is magnified by the distance "D", weighing the cup, contents and string is most important and must be done carefully on scales that are certified for accuracy.

k. Calculate the rudder static balance from the formula M = W x D as follows:

1. The weight of the cup and contents is designated by "W".

2. The distance between the centerline of the hinge and the suspension point of the cup is designated by "D".

3. The overbalance or underbalance moment is designated by "M".

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1. Assume the rudder is slightly underbalanced (tail heavy) and the paper cup was suspended from the leading edge.

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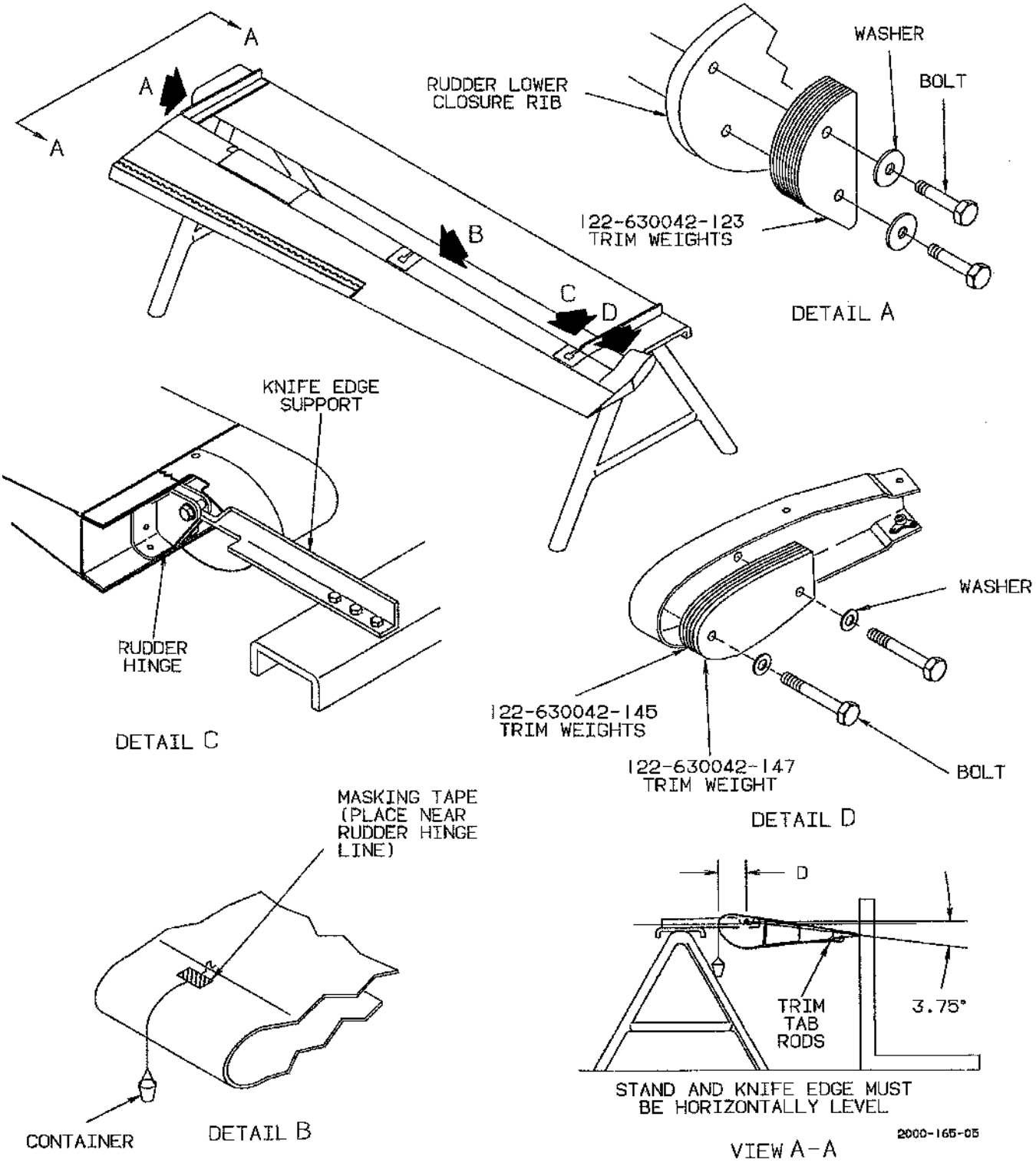
3. M = 8.6 pound-inches (the product of "W x D" must be accurate to within 0.05 pound-inches). In this instance, "M" is within the required static balance range and is considered acceptable.

m. If the rudder static balance does not fall within the range of 7.0 to 10.5 pound-inches tail heavy (underbalance), trim weights must be added or removed and the balance rechecked.

n. The trim weights are located in two places; in the forward end of the lower closure rib and in the fairing at the top of the rudder. When adjusting the trim weights use a maiximum of nine 122-630042-123 weights, a maximum of four 122-630042-145 weights and a maximum of one 122-630042-147 weight. Secure the 122-630042-123 trim weights to the forward end of the lower closure rib with two AN3H10A bolts with a AN960C10 washer under each bolt head. Secure the 122-630042-145 and -147 trim weights with two AN4CH27A bolts, use AN960C4 washers under the bolt heads as required. Safety wire the bolts after the proper number of trim weights have been installed in the rudder.



Rudder Balancing (Effectivity: NC-26 and After)
Figure 4

VENTRAL STABILIZER - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The ventral stabilizer is mounted under the aft fuselage tailcone and is secured to the aft fuselage canted bulkhead (FS 509.73) and a forward attach fitting (U-shaped bracket) at FS 496.00.

The ventral stabilizer is a fixed airfoil constructed with a forward spar, rear spar, upper rib, lower rib, skin and a tip assembly. An aluminum access door with a seal is located an the left side for access to antenna connections. The antenna wiring is routed in a conduit bonded to the forward spar. The ventral stabilizer is fabricated of woven cloth graphite and honeycomb core. The tip assembly is woven cloth kevlar with a moisture drain hole located at the lowest point of the tip assembly. The tip assembly is secured with screws into nutplates.

VENTRAL STABILIZER - MAINTENANCE PRACTICES (Effectivity: All)

VENTRAL STABILIZER REMOVAL (Effectivity: All) (Figure 1)

a. Remove all electrical power from the airplane.

b. Gain access to the ventral stabilizer rear spar attachment to the fuselage canted bulkhead by removing the fuselage tailcone, refer to Chapter 53-50-00.

c. Gain access to the ventral stabilizer forward spar attach fitting by removing the access door above the ventral stabilizer (left side).

d. Remove the two nuts, washers and bolts securing the ventral stabilizer forward spar attach fitting and ground strap.

e. At the ventral stabilizer forward spar, disconnect the avionics antenna wire bundle.

f. Support the ventral stabilizer and remove the six bolts and washers securing the rear spar to the canted bulkhead.

g. Remove the ventral stabilizer and store properly to prevent damage.

VENTRAL STABILIZER INSTALLATION (Effectivity: All) (Figure 1)

a. Check the seal around the top of the ventral stabilizer for cuts, deterioration and security of bond. To replace the seal, refer to VENTRAL STABILIZER SEAL REPLACEMENT in this chapter.

b. Place the ventral stabilizer in position, mating the forward spar fitting and the rear spar to the fuselage canted bulkhead.

c. Secure the rear spar to the fuselage canted bulkhead with six bolts and washers.

d. Secure the forward spar attach fitting and ground strap with two bolts and washers.

NOTE

Refer to Chapter 20 for the electrical bonding requirements.

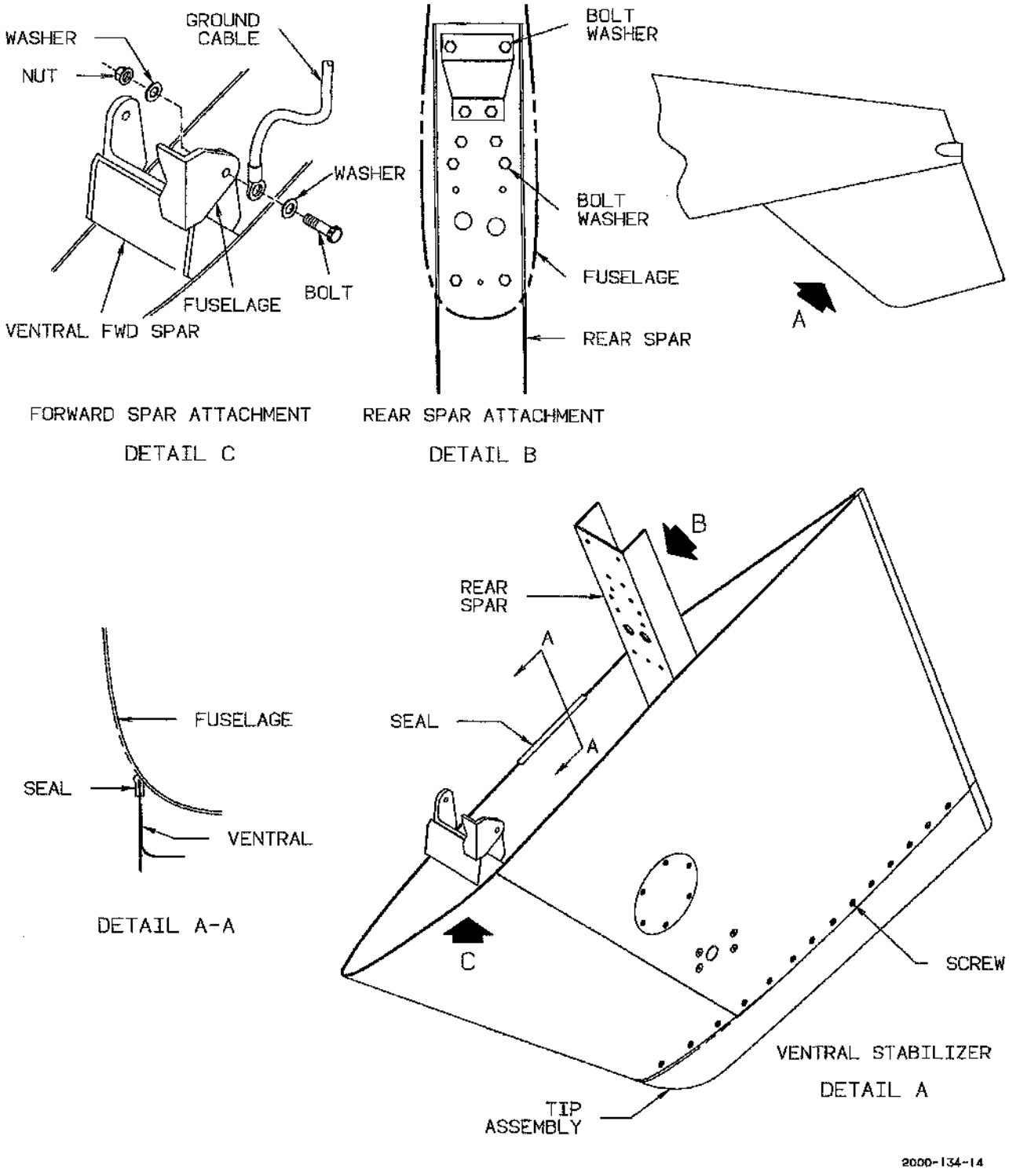
e. Connect the antenna wire bundle at the forward spar fitting area.

f. Install the access door above the ventral stabilizer.

g. Install the fuselage tailcone, refer to Chapter 53-50-00.

VENTRAL STABILIZER SEAL REPLACEMENT (Effectivity: All)

a. With the ventral stabilizer removed from the airplane, carefully remove the seal and clean old adhesive from the surface with cheesecloth (5, Chart 2, 55-00-00) moistened with Naphtha (2, Chart 2, 55-00-00).



Ventral Stabilizer Installation (Effectivity: All)
Figure 1

b. Temporarily place a length of seal on the ventral stabilizer. Cut the seal to length as required and trim the inboard leg of the seal in the area of the spar.

c. Apply a thin even coat of adhesive (6, Chart 2, 55-00-00) with a brush to the seal and ventral stabilizer surface to be bonded. Allow to dry until adhesive is quite tacky, but will not transfer to the knuckle when touched.

d. Align and press the seal in place.

e. Install the ventral stabilizer.

CHAPTER 56 - WINDOWS

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CHAPTER 56 - WINDOWS

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WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The cockpit windshields and side windows are constructed of two layers of laminated glass, with an inner layer of urethane. An antistatic coating is applied on the outer surface. An integral heating element is installed between the outer layer of glass and the layer of urethane.

The structurally mounted cabin windows are constructed of laminated stretched acrylic. The inter-surface has a urethane antifog coating.

The polarized windows consist of two panes, each of which are composed of a film of polarizing material laminated between two sheets of acrylic plastic.

The tinted windows for the aft toilet installed on NC-21 and after consist of a single sheet of acrylic plastic.

SPECIAL TOOLS AND RECOMMENDED MATERIALS (EFFECTIVITY: ALL)

The special tools and recommended materials listed in Charts 1 and 2 as meeting federal, military or supplier specifications are provided for reference only and are not specifically prescribed by Raytheon Aircraft Company. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Raytheon Aircraft Company, by the supplier, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the supplier of the product to be used.

CHART 1
SPECIAL TOOLS AND EQUIPMENT (EFFECTIVITY: ALL)

TOOL NAME	PART NO.	SUPPLIER	USE
1. Barfield Fuel Calibration Instrumentation, Ohmmeter or Volt-Ohmmeter capable of registering over 100 million ohms (100 megohms)	2548H	Barfield	Test Resistance of windshield anti-static coating
2. Orbital sander	None	Obtain locally	Refurbish acrylic cabin windows

CHART 2
RECOMMENDED MATERIALS (EFFECTIVITY: ALL)

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
1. Cleaner	TT-M-261	MEK	
2. Plastic Window Scratch Remover		HP-100C Micro-Mesh	Micro Surface Finishing Products, Box 818 Wilton, IA 52778
3. Cleaner (Solvent)	TT-N-95	Naphtha	Obtain Locally
4. Adhesive		Dapcotac #3300	D Aircraft Products Co., 1191 Hawk Circle Anaheim, CA 92708
5. Primer		S2260	Dow Corning Corp., 3901 S. Saginaw Rd P.O. Box 997 Midland, MI 48640
6. Sealant	MIL-S-8802, Type II	PR 1440 B-4	Courtaulds Aerospace Corp., 5454 San Fernando Rd P.O. Box 1800 Glendale, CA 91209
7. Sealant		PR 1425 B-2	Courtaulds Aerospace Corp., 5454 San Fernando Rd P.O. Box 1800 Glendale, CA 91209
8. Copper Wire Mesh Pad			Obtain Locally

GENERAL - MAINTENANCE PRACTICES (EFFECTIVITY: ALL)

CLEANING WINDSHIELDS AND CABIN WINDOWS (EFFECTIVITY: ALL)

Refer to Chapter 12.

PRECAUTIONS REQUIRED WHEN STRIPPING, PAINTING OR DEGREASING (EFFECTIVITY: ALL)

Paint, paint strippers, degreasing compounds, and most solvents will seriously damage acrylic windows. If any of these activities are to be performed in the vicinity of the windows, the windows must be completely sealed against the chemical compounds and fumes. Pressure sensitive polyester film (mylar) and pressure sensitive painting/stripping masks of polyester, aluminum and other composite materials are commercially available for this purpose. When installed, the mask must extend beyond the window cutouts in the skin. This will prevent the chemical compounds from attacking the window edges. If paint is inadvertently applied to the windshield glass during airplane repainting, remove the paint as instructed in the procedure that follows.

REMOVING PAINT FROM WINDSHIELDS AND SIDE WINDOWS (EFFECTIVITY: ALL)

a. Mask the paint on the airplane skin to prevent damage from the solvent.

CAUTION

To prevent damage, use care not to contact the airplane paint with the methyl ethyl ketone or other solvent. To avoid damage to the antistatic coating, never use any rubbing compound or polish, or any substance containing acid, on the windshield or side window.

b. If the paint on the windshield or side window is dry, contact the Customer Support Department of Raytheon Aircraft Company for consultation.

c. If the paint on the windshield or side window is still wet, use a clean, soft cloth dampened with a nonacid solvent or the preferred methyl ethyl ketone (1, Chart 2) to remove the paint. Wipe the glass surface in a straight rubbing motion. Never use any abrasive materials or acids to remove the paint.

d. Rinse the glass thoroughly and dry, but do not apply wax.

COCKPIT - MAINTENANCE PRACTICES

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

WINDSHIELD AND SIDE WINDOW ANTISTATIC COATING INSPECTION

Inspect the electrically heated windshields and side windows every 600 hours or 18 months which ever occurs first.

a. Remove all electrical power from the airplane.

b. Clean the exterior of the windshield or side window glass as instructed in Chapter 12.

c. Remove the paint from the head of one of the screws attaching the windshield or side window to the frame.

NOTE

A Barfield Fuel Calibration Instrumentation P/N 2548H or equivalent or a volt-ohmmeter capable of reading in excess of 100 million ohms (100 megohms) will be required for the following inspection (1, Chart 1, 56-00-00).

d. Set the ohmmeter range dial on 100 megohms and adjust the needle reading to zero.

CAUTION

Do not scratch the glass surface with the wire mesh pad. Press the wire mesh pad straight down on the glass surface with as little side-to-side motion as possible.

e. Contact the windshield or side window glass with the wire mesh pad. Check the ohmmeter reading. If the ohmmeter reading is 100 megohms or less in at least 10 different locations, no further action is required. Repaint the screw head where the paint was removed to match the surrounding finish.

NOTE

Do not remove the windshield or side window from the airplane as a result of this inspection.

f. If some of the readings were more than 100 megohms, draw a diagram noting the locations of the excessive readings, and contact the Customer Support Department of Beech Aircraft Corporation for consultation.

WINDSHIELD AND SIDE WINDOW REPLACEMENT CRITERIA

The electrically heated laminated glass windshields and cockpit side windows, are subject to a gradual process of delamination due to the effects of chemical action and differentials of temperature and pressure incurred during

pressurized flights at varying altitudes and weather conditions. This delamination is not detrimental to the structural integrity of the windshield or side window, but may significantly decrease visibility or the deicing capability of the windshield. Beyond certain limits, either of these effects will require replacement of the windshield or side window. The following guidelines are provided as an aid to determine when a windshield or side window should be replaced:

a. VISIBILITY IMPAIRMENT: Replace the windshield or side window when areas of delamination enlarge to the point of impairing vision, whether or not these areas extend to the edge of the glass. Smaller delaminations appearing as bubbles are no cause for concern. The rate of growth of delaminated areas should be used as a guide for scheduling windshield or side window replacement.

b. HEATING IMPAIRMENT LIMITS: Replace the windshield or side window when the lost heating area exceeds 1/4 to 1/3 of the total heated area.

NOTE

The windshield has one spare temperature controller element which can be used if required.

NOTE

For flights with cracked windshield glass refer to AIRWORTHINESS LIMITATIONS, Chapter 4-00-00.

WINDSHIELD REMOVAL

NOTE

The LH and RH windshields are symmetrical to one another. The following procedures are applicable to either windshield.

a. Remove electrical power from the airplane.

b. Remove the glareshield, compass cover, sunvisor, windshield cover, and upper sidewall upholstery panel. Refer to Chapter 25-10-00.

c. Identify, tag and disconnect the windshield heating wiring (4). Refer to Figure 1.

d. Note the position of the windshield wiper arm and remove it.

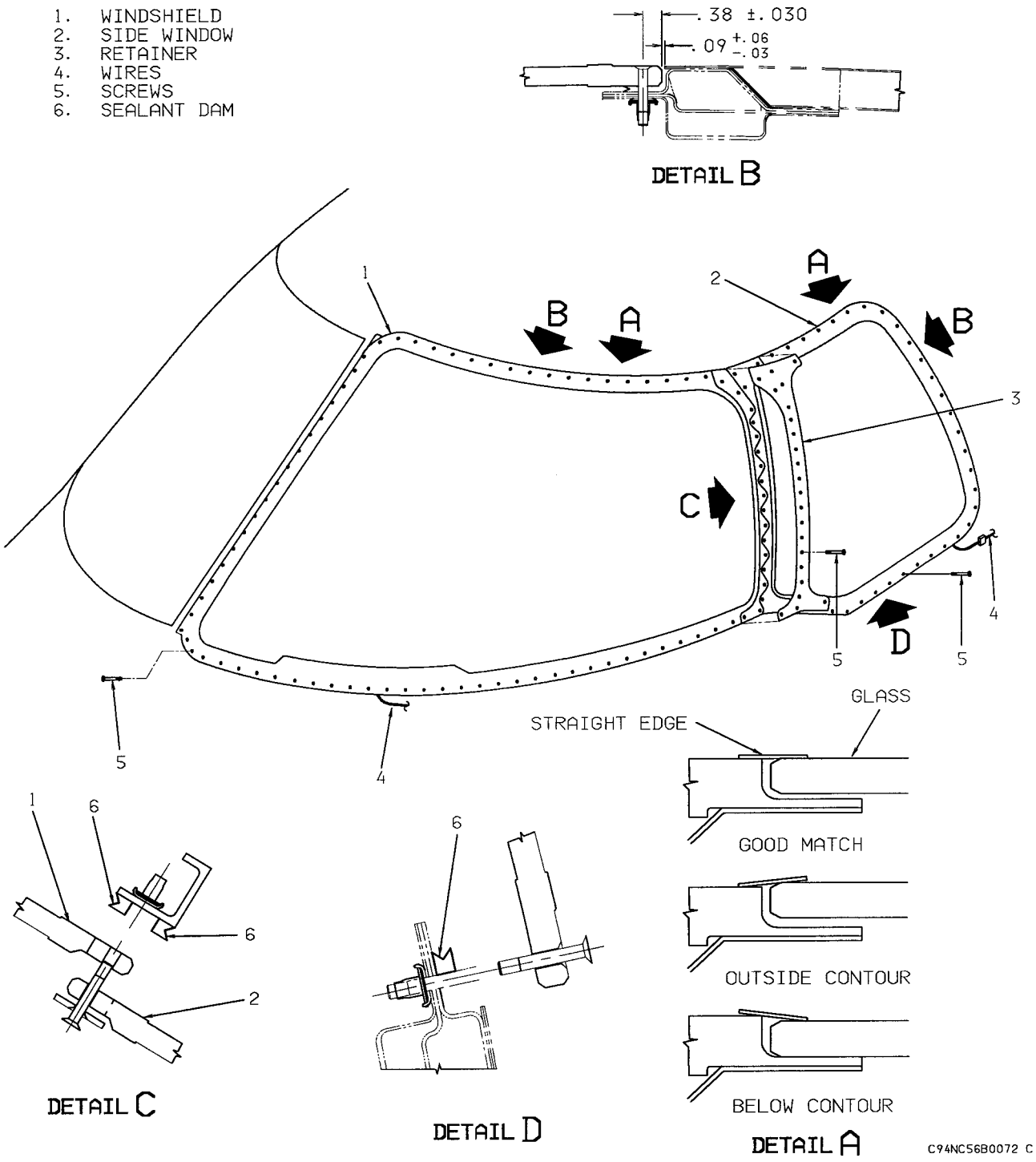
e. Mask the fuselage around the outside of the windshield (1). If the windshield is cracked, apply masking tape to contain the splintering glass. Place a cover over the instrument panel to protect it from debris.

NOTE

Some of the screws (5) securing the windshield (1) may be of odd lengths. Therefore it is advisable to make a map of the screw locations to ensure that the proper length screws are installed in each hole.

NOTE

After the screws (5) have been loosened from the nutplates, they may be pried loose from the windshield sealant with a small prying tool. Take care not to scratch the fuselage surface.



Windshield and Side Window Installation
Figure 1

NOTE

Airplanes NC-4 thru NC-23 have floating nutplates installed for windshield (1) and side window (2) attachment. Damaged nutplates may be replaced as described under ALTERNATE METHOD OF WINDSHIELD AND SIDE WINDOW NUTPLATE REPLACEMENT in this chapter.

f. Remove the screws (5) from the retainer (3) mounted on the windshield side post. Using a thin flat bladed tool, cut the sealant from the retainer and carefully remove the retainer. Clean the retainer with cleaner (1, Chart 2, 56-00-00) and keep it for use on the new windshield (1).

g. Remove the remaining screws (5) from the windshield (1).

h. Cut the sealant around the windshield (1) on the outside of the airplane using a knife.

i. Cut the sealant around the windshield (1) on the inside of the airplane using a knife. Begin cutting at the aft edge of the windshield and work forward, pushing outward on the windshield. When the sealant has been completely broken loose, remove the windshield.

j. Remove the remaining sealant and sealant dam (6) from the frame using a nonmetallic scraper.

WINDSHIELD INSTALLATION

NOTE

The following procedures are based on a room temperature of 75° to 77° F. Refer to the manufacturers instructions for curing times of bonding and sealing materials if the temperature varies from this range.

a. Before fitting the new windshield (1) to the windshield frame (Refer to Figure 1), bond the new sealant dam (6) to the frame as follows:

1. Wipe the windshield frame clean with cleaner (3, Chart 2, 56-00-00).

2. Apply two coats of adhesive (4, Chart 2, 56-00-00) to the bonding area. Allow the first coat to dry for approximately 30 minutes before the second coat is applied.

3. Apply primer (5, Chart 2, 56-00-00) to the area where the sealant dam (6) is to be bonded. Allow the primer to dry for approximately 15 minutes.

4. When the second coat of adhesive can be touched with a knuckle without transferring (approximately 15 to 30 minutes), install the sealant dam (6) in the windshield frame. Allow bond to set for 2 to 4 hours.

NOTE

Airplanes NC-4 thru NC-23 have floating nutplates installed for windshield (1) and side window (2) attachment. Damaged nutplates may be replaced as described under ALTERNATE METHOD OF WINDSHIELD AND SIDE WINDOW NUTPLATE REPLACEMENT in this chapter.

b. Lay the new windshield (1) in the frame and install screws (5) along the center post, tightening them only enough to start the screws into the nutplates on the frame.

c. Install screws (5) only enough to engage the nutplates alternately along the top and bottom of the windshield (1), working from the center post outward. When holes in the windshield and the frame do not align, tighten the screws in the center post slightly, as well as the screws already started. This will align more windshield and frame

holes along the top and bottom. Continue in this way until all of the screws in the windshield have been installed.

d. The windshield (1) must be as close as possible to flush with the fuselage once it has been installed. Check for match of contour between the windshield and fuselage by placing a straightedge across the gap between the two. Tighten the screws (5) until the best average contour match has been made. See Detail A of Figure 1.

e. The windshield (1) shall fit into the frame and meet the following criteria, or must be trimmed to do so:

1. The distance from the centerline of a hole to the edge shall be .380 ±.030 inch. See Detail B of Figure 1.

2. The gap around the periphery of the windshield shall be .09 +.06/-.03 inch. See Detail B of Figure 1.

f. Remove the windshield (1) from the frame.

g. To prevent sealant from dripping through the nutplates, loosely place masking tape over the lower side of the nutplates.

h. Fill the frame outside of the sealant dam (6) with sealant (6, Chart 2, 56-00-00). The sealant should fill the frame to approximately level with the top of the sealant dam.

NOTE

When tightening the screws (5) in the windshield (1) before the sealant has cured, tighten the screws only until the windshield has reached the best average contour. See Detail A of Figure 1.

i. Place the windshield (1) in the frame, and work the windshield into place as described in steps c and d, except leave the aft (rippled) edge unattached.

j. Apply a coating of sealant to the retainer (3) and install it over the aft (rippled) edge of the windshield (1) with screws (5).

k. Smooth the sealant around the retainer (3) and remove the excess as required.

l. Remove the masking tape from the fuselage around the windshield (1) and remove the cover from the instrument panel.

m. Allow the sealant to cure for approximately 30 hours. After the sealant has cured, loosen and retorque the screws (5) to a maximum of 20 inch-pounds.

n. Install the windshield wipers on the airplane in the same position they were in before removal.

o. Connect the windshield heating wiring (4) to the windshield (1).

p. Install the upper sidewall upholstery panel, windshield cover, sunvisor, compass cover and glareshield. Refer to Chapter 25-10-00.

q. Restore electrical power to the airplane.

SIDE WINDOW REMOVAL

NOTE

The LH and RH side windows (2) are symmetrical to one another. The following procedures are applicable to either window.

a. Remove electrical power from the airplane.

b. Remove the sunvisor, windshield cover and upper sidewall upholstery panel. Refer to Chapter 25-10-00.

c. Identify, tag and disconnect the side window heating wiring (4). Refer to Figure 1.

d. Mask the fuselage around the outside of the window (2). If the window is cracked, apply masking tape to contain the splintering glass.

NOTE

Some of the screws (5) securing the window (2) may be of odd lengths; therefore, it is advisable to make a map of the screw locations to ensure that the proper length screws are installed in each hole.

After the screws (5) have been loosened from the nutplates, they may be pried loose from the window sealant with a prying tool. Take care not to scratch the fuselage surface.

Airplanes NC-4 thru NC-23 have floating nutplates installed for windshield (1) and side window (2) attachment. Damaged nutplates may be replaced as described under ALTERNATE METHOD OF WINDSHIELD AND SIDE WINDOW NUTPLATE REPLACEMENT in this chapter.

e. Remove the screws (5) from the retainer (3) mounted on the windshield side post. Using a thin flat bladed tool, cut the sealant from the retainer and carefully remove the retainer. Clean the retainer and keep it for use on the new side window (2).

f. Remove the remaining screws (5) from the side window (2).

g. Cut the sealant around the side window (2) on the outside of the airplane using a knife.

h. Cut the sealant around the side window (2) on the inside of the airplane using a knife. Begin cutting at the forward edge of the side window and work aft, pushing outward on the window. When the sealant has been completely broken loose, remove the side window.

i. Remove the remaining sealant and dam (6) from the frame using a nonmetallic scraper.

SIDE WINDOW INSTALLATION

NOTE

The following procedures are based on a room temperature of 75° to 77° F. Refer to the manufacturers instructions for curing times of bonding and sealing materials if the temperature varies from this range.

a. In order to insure proper fitting of the new side window (2), bond the new sealant dam (6) in the side window frame before fitting the window. Refer to Figure 1.

NOTE

Airplanes NC-4 thru NC-23 have floating nutplates installed for windshield (1) and side window (2) attachment. Damaged nutplates may be replaced as described under ALTERNATE METHOD OF WINDSHIELD AND SIDE WINDOW NUTPLATE REPLACEMENT in this chapter.

a. Lay the new side window (2) in the frame and install screws (5) along its aft edge, tightening them only enough

to start the screws into the nutplates on the frame.

b. Install screws (5) only enough to engage the nutplates alternately along the top and bottom of the side window (2), working from the aft edge forward. When holes in the side window and the frame do not align, tighten the screws on the aft edge slightly, as well as the screws already started. This will align more side window and frame holes along the top and bottom. Continue in this way until all of the screws in the side window have been installed.

c. The side window (2) must be as close as possible to flush with the fuselage once it has been installed. Check for match of contour between the side window and fuselage by placing a straightedge across the gap between the two. Tighten the screws until the best average contour match has been made. See Detail A of Figure 1.

d. The side window (2) shall fit into the frame, and meet the following criteria, or be trimmed to do so:

1. The distance from the centerline of a hole to the edge shall be .380 ± .030 inch. See Detail B of Figure 1.

2. The gap around the periphery of the side window shall be .09 +.06/-.03 inch. See Detail B of Figure 1.

e. Remove the side window (2) from the frame.

f. To prevent sealant from dripping through the nutplates, loosely place masking tape over the lower side of the nutplates.

g. Fill the frame outside of the sealant dam (6) with sealant (6, Chart 2, 56-00-00). The sealant should fill the frame to approximately level with the top of the sealant dam.

NOTE

When tightening the screws (5) in the side window (2) before the sealant has cured, tighten the screws only until the side window has reached the best average contour. See Detail A of Figure 1.

h. Place the side window (2) in the frame, and work the side window into place as described in steps c and d, except leave the forward (rippled) edge unattached.

i. Apply a coating of sealant to the retainer (3) and install it over the forward (rippled) edge of the side window (2) with screws (5).

j. Smooth the sealant around the retainer (3) and remove the excess as required.

k. Remove the masking tape from the fuselage around the side window (2).

l. Allow the sealant to cure for approximately 30 hours. After the sealant has cured, loosen and retorque the screws (5) to a maximum of 20 inch-pounds.

m. Connect the side window heating wiring (4) to the side window (2).

n. Install the upper sidewall upholstery panel, windshield cover and sunvisor. Refer to Chapter 25-10-00.

o. Restore electrical power to the airplane.

ALTERNATE METHOD OF WINDSHIELD AND SIDE WINDOW NUTPLATE REPLACEMENT (NC-4 THRU NC-23)

a. Remove the damaged nutplate.

b. Mount an appropriate NAS1791, NAS1792 or NAS1793 nutplate on a piece of .040-inch aluminum 1-inch

square.

c. Fit the nutplate assy into place, trimming the aluminum base as required.

d. Bond the nutplate to the windshield or side window frame with sealant (6, Chart 2, 56-00-00).

CABIN - MAINTENANCE PRACTICES (EFFECTIVITY: ALL)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

CABIN WINDOWS (EFFECTIVITY: ALL)

The cabin windows in the Beech Starship 1 airplanes are made of a fail-safe, multi-ply stretched acrylic material. Inspect these windows at intervals as specified in Chapter 5. Inspect the windows for deep scratches, chips, excessive crazing, and all other evidence of damage. See Chart 1 for damage types and recommended action.

ACRYLIC WINDOW REFURBISHING PROCEDURE (EFFECTIVITY: ALL)

Chart 1 outlines the types and limits of window damage which may be repaired utilizing this procedure. If any unusual conditions are found that are not covered in Chart 1, contact the Customer Support Department of Raytheon Aircraft Company for consultation. Use of this procedure is prohibited except as specifically instructed by Chart 1 or by the Customer Support Department of Raytheon Aircraft Company.

a. Mask off any major portions of the window which are not involved in the refurbishing.

b. Use wet abrasives only.

c. Flush the window surface with a liberal amount of water after using each abrasive grade.

d. Use a straight-line sanding motion and alternate the line of sanding at right angles to cross the sanding pattern.

e. Gradually lighten the working pressure on the abrasive as each finer grade is utilized. Do not exert excessive force on the window pane.

f. Work a marginally larger area as each finer abrasive is used. Completely remove each previous sanding pattern before moving to the next finer abrasive.

g. Do not overheat the acrylic window.

h. Start the refurbishing procedure with 240-grit wet/dry silicone carbide cloth or equivalent and an orbital sander (2, Chart 1, 56-00-00). Use this combination until all traces of the damage have been removed.

i. Use a 320-grit wet/dry silicone carbide cloth with the orbital sander and work the window until the 240-grit pattern has been removed.

j. Repeat step i using 400-grit, 600-grit, 1800-grit, 2400-grit, 3200-grit, 4000-grit, and 6000-grit silicone carbide cloth in succession.

k. Finish the refurbishing procedure using an 8000-grit cushioned abrasive cloth or cerium oxide.

l. Inspect the finished window following the inspection procedure shown in Figure 1 to ensure that all damage and grit patterns have been removed.

m. When satisfied that the window has been properly refurbished, apply a small amount of antistatic cleaner or wax to the window and buff to a high luster. For removal of minor crazing or scratches, an HP-100C Micro-Mesh kit (2, Chart 2, 56-00-00) is recommended.

CHART 1 - WINDOW DAMAGE (EFFECTIVITY: ALL)

Damage Type	Location	Probable Cause	Recommended Action
Scratches, abrasions, gouges and chips with a maximum depth of .015 inch	Inner ply	Improper maintenance or cleaning procedures	Airplane may be operated unpressurized only until the window is replaced.
	Outer ply	Improper maintenance cleaning procedures or object impact	Damage may be worked out. Refer to ACRYLIC WINDOW REFURBISHING PROCEDURES
Scratches, abrasions, gouges and chips with a depth in excess of .015 inch	Inner ply	Improper maintenance or cleaning procedures	Immediate replacement of the window is mandatory before further flight
	Outer ply	Improper maintenance, cleaning procedures or object impact	Airplane may be operated unpressurized only until the window is replaced
Crazing with a maximum depth of .015 inch	Inner ply	Contact with unapproved cleaners, solvents or chemical compounds; stress fatigue	Airplane may be operated unpressurized only until the window is replaced
	Outer ply	Contact with unapproved cleaners, solvents or chemical compounds; stress fatigue	Damage may be worked out. Refer to ACRYLIC WINDOW REFURBISHING PROCEDURES
Crazing with a maximum depth in excess of .015 inch	Inner ply	Contact with unapproved cleaners, solvents or chemical compounds; stress fatigue	Immediate replacement of the window is mandatory before further flight
	Outer ply	Contact with unapproved cleaners, solvents or chemical compounds; stress fatigue	Airplane may be operated unpressurized only until the window is replaced
Haziness and cloudiness	Inner ply	Contact with unapproved cleaners, solvents or chemical compounds; stress fatigue	Airplane may be operated unpressurized only until the window is replaced
	Outer ply	Contact with unapproved cleaners, solvents or chemical compounds; stress fatigue	Airplane may be operated unpressurized only until the window is replaced
Pane delamination		Stress; extreme heat at high humidities	Airplane may be operated unpressurized only until the window is replaced

CHART 1 - WINDOW DAMAGE (EFFECTIVITY: ALL) (CONTINUED)

Damage Type	Location	Probable Cause	Recommended Action
Out of contour		High humidity; water absorption, excessive heat or pressurization; chemical or solvent absorption	Immediate replacement of the window is mandatory before further flight
Distortion		Improper restoration when removing damage from outer ply; chemical or solvent damage	If distortion is caused by poor restoration techniques, the window may be replaced at will with no flight limitations. However, if the distortion is the result of chemical or solvent damage, the airplane may be operated unpressurized only until the window is replaced.
Circumferential cracks - at least 2 inches from the window frame	Inner ply	Improper maintenance; stress fatigue	Airplane may be operated unpressurized only until the window is replaced
	Outer ply	Improper maintenance; stress fatigue; object impact	Airplane may be operated for a maximum of 20 hours at a reduced pressure of 4.6 psid with a maximum altitude of 25,000 feet
Circumferential cracks - within 2 inches of the window frame	Inner ply	Improper maintenance; stress fatigue	Airplane may be operated unpressurized only until the window is replaced
	Outer ply	Improper maintenance: stress fatigue; object impact	Airplane may be operated unpressurized only until the window is replaced

CABIN WINDOW REMOVAL (EFFECTIVITY: ALL) (FIGURE 2)

a. Remove the cabin window upholstery and polarized or tinted window.

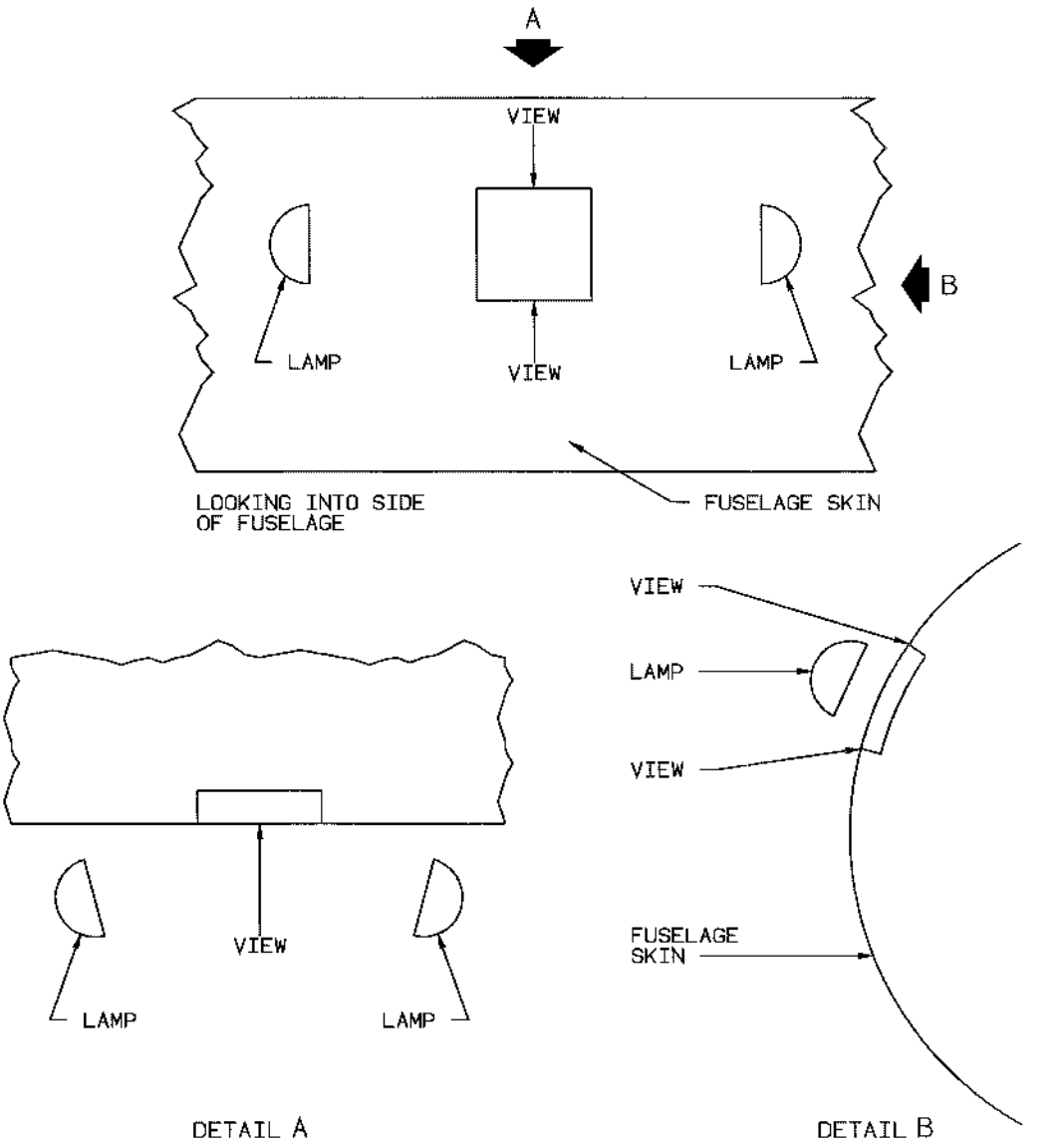
b. Remove the window clips (1) and retain them.

c. If Kit 122-5031-1, -3 or -5 has been installed, disconnect the window pigtail tubing.

d. Cut the sealant on the inside of the airplane between the window (2) and the window opening flange (3) with a knife.

e. Cut the sealant on the outside of the airplane between the window (2) and the window opening flange (3) with a knife. When the sealant has been loosened, push the window out of the window opening.

f. Remove any sealant remaining on the window opening flange (3) with a nonmetallic scraper.



THE WINDOW INSPECTION PROCEDURE SHOWN IN THIS FIGURE SHOULD BE PERFORMED IN A DARKENED AREA, OR AT NIGHT. USE A GOOD LIGHT SOURCE AND MOVE IT AROUND THE WINDOW AT VARIOUS ANGLES WHILE VIEWING THROUGH THE WINDOW FROM DIFFERENT DIRECTIONS. INSPECT ALL OF THE FUSELAGE SIDE WINDOWS IN THIS MANNER. SEE CHART 1 FOR DAMAGE DESCRIPTION AND RECOMMENDED ACTION.

2000-110-07

Window Inspection Procedure (Effectivity: All)
Figure 1

CABIN WINDOW INSTALLATION (EFFECTIVITY: ALL)
(FIGURE 2)

a. Ensure that the surface of the window opening flange (3) is smooth and clean.

b. Apply masking tape (5 and 7) around the periphery of the window opening on the exterior of the fuselage and around the periphery of the machined edge of the window .

c. Place a bead of sealant along the center of the window opening flange (3) all around the window opening.

d. Place the window (2) into the opening and center it. The window shall be oriented so that the part number is at the bottom of the window. There shall be a gap of .15 +.06/-.03 inch between the periphery of the window opening on the exterior of the fuselage and the periphery of the machined edge of the window. The flange on the outer edge of the window shall overlap the window opening flange a minimum of .62 inch at all points around the edge of the window. See Detail C of Figure 2.

e. Install the window clips (1).

WARNING

Do not use a razor blade or sharp instrument on the window (2) to trim the sealant. Do not leave masking tape (7) on the window for an extended amount of time. This could damage the window.

f. Fill the exposed gap between the masked areas with sealant (6) (7, Chart 2, 56-00-00). Smooth the sealant and remove the masking tape (5 and 7).

g. Fillet seal between the window and the window opening flange with sealant (4) (7, Chart 2, 56-00-00).

h. Allow the sealant to cure for the time/temperature specified by the manufacturer. Remove the protective paper on the inboard side of the window (2) and clean the window with a cloth dampened with solvent (3, Chart 2, 56-00-00), followed immediately with a clean dry cloth.

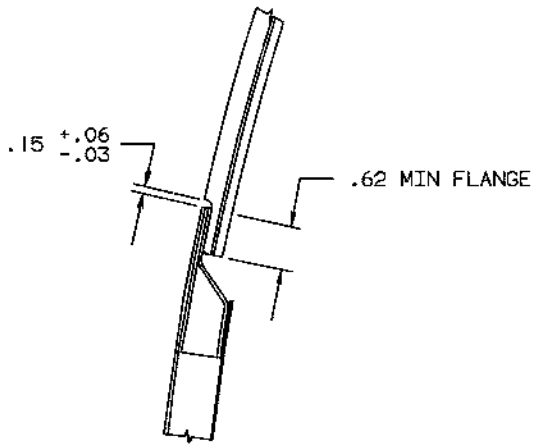
CAUTION

If Kit 122-5031-1, -3 or -5 is not installed, ensure that the window pigtail tubing is not obstructed or kinked. Also ensure that the pigtail is routed away from the bleed air vent.

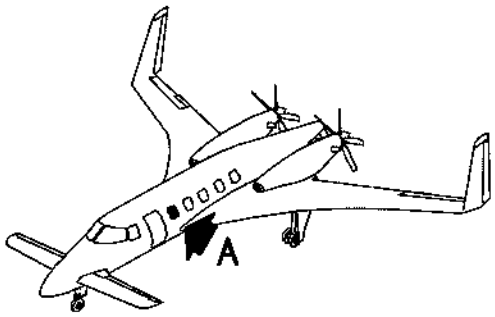
i. If Kit 122-5031-1,-3 or -5 is installed, perform the following steps:

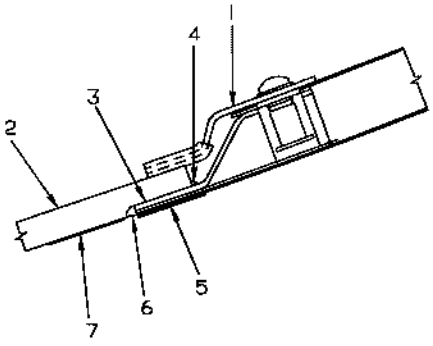
1. Locate the existing desiccant bottle, and remove. See Figure 3.
2. Remove and discard the cap from one end of the new desiccant bottle.
3. Connect the open end of the bottle to the window tubing and attach the bottle to the frame with ty-strap.
4. Remove and discard the cap from the other end of the desiccant bottle.
5. Connect the window pig tail tubing to the existing window tubing system.

j. Install the polarized or tinted window and the cabin window upholstery.

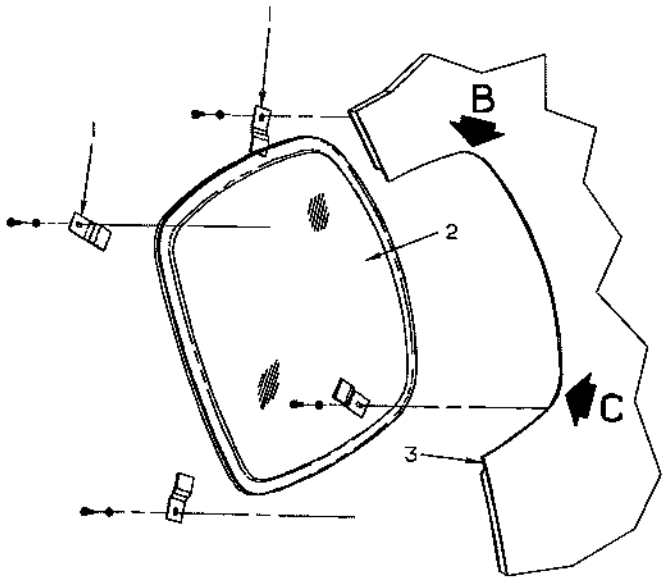


TYPICAL FOR ENTIRE WINDOW
DETAIL C





TYPICAL FOR ENTIRE WINDOW
DETAIL B



DETAIL A

1. WINDOW CLIPS
2. CABIN WINDOW
3. WINDOW OPENING FLANGE
4. SEALANT
5. MASKING TAPE
6. SEALANT
7. MASKING TAPE

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Cabin Window Installation (Effectivity: All)
Figure 2

DESICCANT BOTTLE REPLACEMENT
(EFFECTIVITY: AIRPLANES WITH KIT 122-5031-1, -3 OR -5 INSTALLED)
(FIGURE 3)

a. Remove the cabin window upholstery. Refer to Chapter 25.

b. Locate the existing desiccant bottles and replace each one as follows:

1. Remove and discard the existing desiccant bottle.

2. Remove and discard the caps from each end of the new desiccant bottle.

3. Connect one end of the new bottle to the window tubing and attach the bottle to the frame with ty-strap.

c. Install the cabin window upholstery. Refer to Chapter 25.

ACTUATOR MOTOR REMOVAL (EFFECTIVITY: NC-4 THRU NC-20)
(FIGURE 4)

a. Remove electrical power from the airplane.

b. Remove the upholstery around the window. Refer to Chapter 25.

c. Remove the screws (11) and washers (10) attaching the block (5) to the bracket (2).

d. Remove the motor (3) from the bracket (2), and disconnect and tag the wires (4).

e. Loosen the set screw (8) and remove the drive roller (7) from the motor shaft.

f. If the O-ring (9) needs to be replaced, remove the O-ring from the drive roller (7).

g. Remove the screws (6) from the block (5) and remove the block from the motor (3).

ACTUATOR MOTOR INSTALLATION (EFFECTIVITY: NC-4 THRU NC-20)
(FIGURE 4)

a. Install the block (5) on the motor (3) with screws (6).

b. Install the drive roller (7) on the motor (3) and tighten the set screw (8).

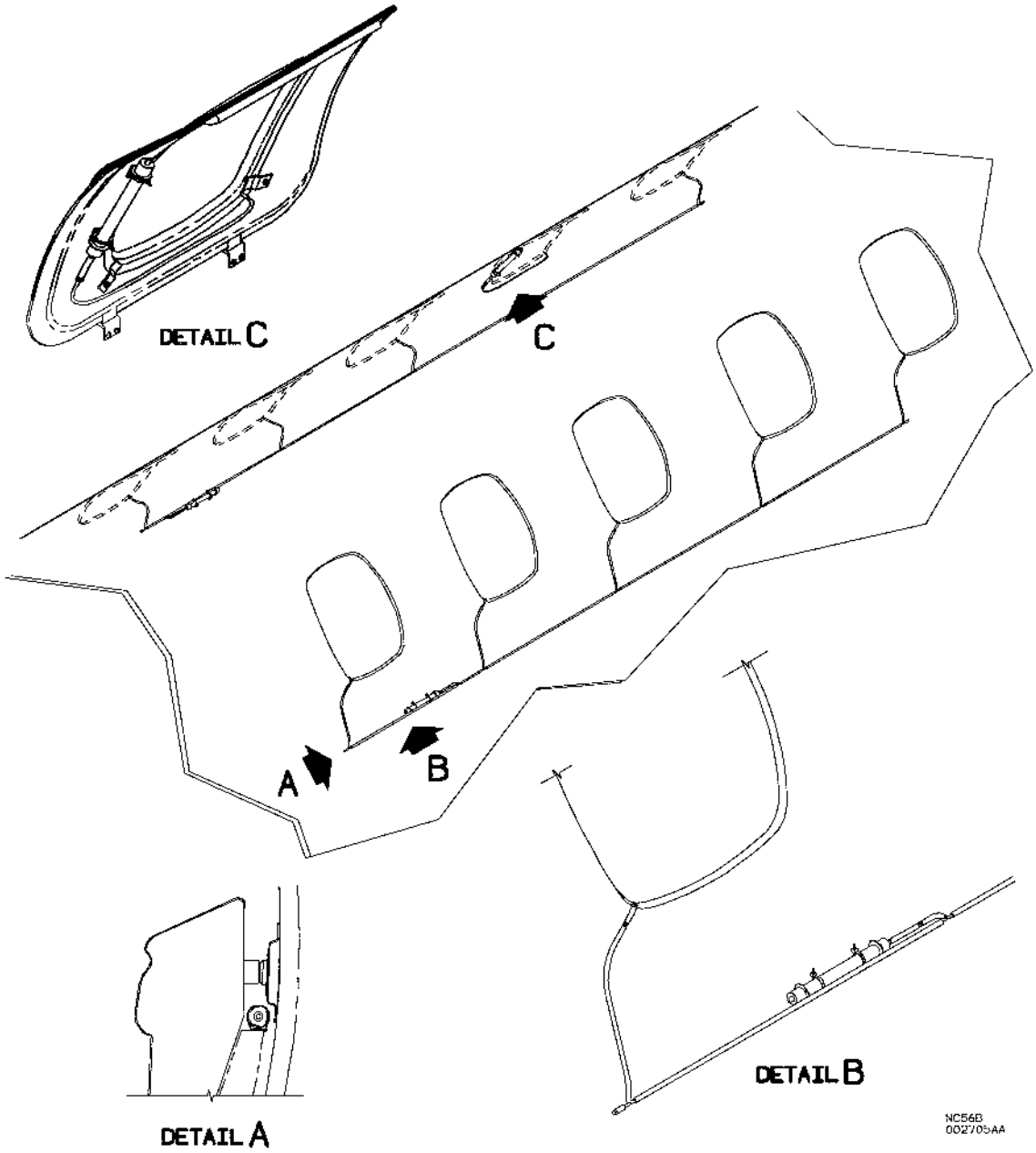
c. Install the new O-ring (9) (if required) on the drive roller (7).

d. Connect the wires (4) to the motor as tagged during removal.

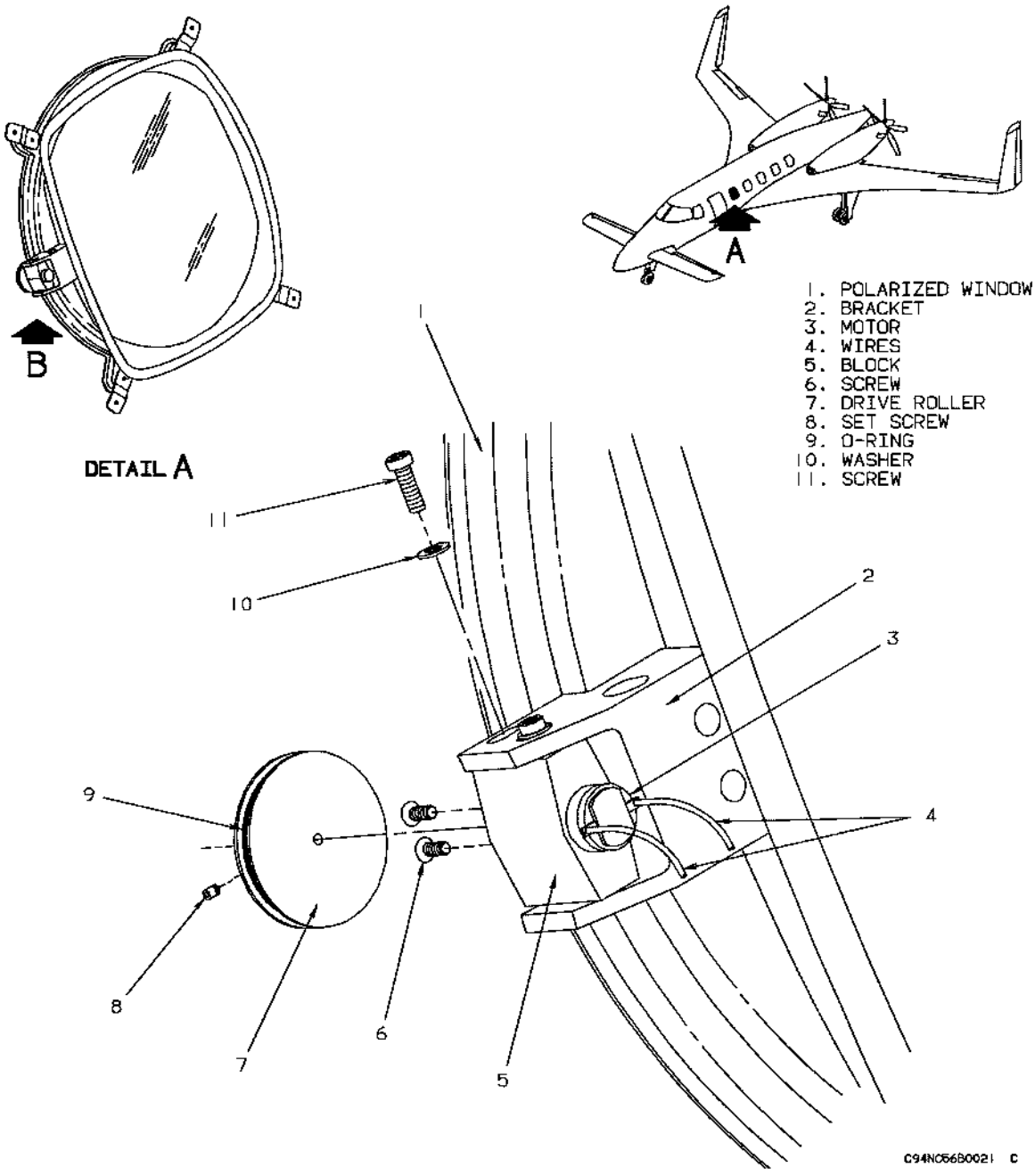
e. Mount the motor (3) in the bracket (2), positioning the motor so that the O-ring (9) on the drive roller (7) will create adequate friction with the polarized window (1) to turn it smoothly. Adjust the position of the motor on the bracket as necessary.

f. Restore electrical power to the airplane and test the motor for proper operation.

g. Install the upholstery around the window. Refer to Chapter 25-20-00.



Desiccant System (Effectivity: Airplanes with Kit No. 122-5031-1, -3 or -5)
Figure 3



Actuator Motor Installation (Effectivity: NC-4 thru NC-20)
Figure 4

ACTUATOR MOTOR SWITCH REMOVAL (EFFECTIVITY: NC-4 THRU NC-20) (FIGURE 5)

a. Remove electrical power from the airplane.

b. Remove the sidewall console assembly (1). Refer to Chapter 25-20-00.

c. Remove the wires (2) from the switch (3) and tag them.

d. Press the retaining springs (4) and remove the switch (3) from the console (1).

ACTUATOR MOTOR SWITCH INSTALLATION (EFFECTIVITY: NC-4 THRU NC-20) (FIGURE 5)

a. Press the switch (3) into the sidewall console assembly (1).

b. Attach the wires (2) to the switch (3) as tagged.

c. Install the sidewall console assembly (1). Refer to Chapter 25-20-00.

d. Restore electrical power to the airplane.

POLARIZED WINDOW REMOVAL (EFFECTIVITY: NC-4 THRU NC-20) (FIGURE 6)

a. Remove electrical power from the airplane.

b. Remove the upholstery from around the window. Refer to Chapter 25-20-00.

c. Remove the wires (9) from the actuator motor (14) and tag them.

d. Remove the screws (5) attaching the window shroud (7) to the fuselage. Remove the shroud and window clips (6).

e. Loosen the screws (12) attaching the actuator motor block (10) and position the block so that the drive roller (11) is clear of the inboard window (2).

f. Tighten the screws (3) in the roller block assemblies so the rollers (4) are free of the inboard window (2).

g. Carefully remove the inboard window (2) and retainer (1) from the shroud (7).

h. Remove the outboard window (8) from the shroud (7).

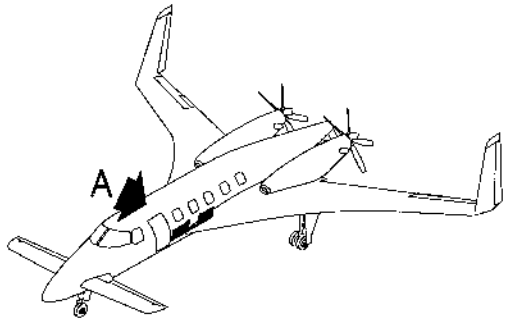
i. Carefully remove the retainer (1) from the inboard window (2).

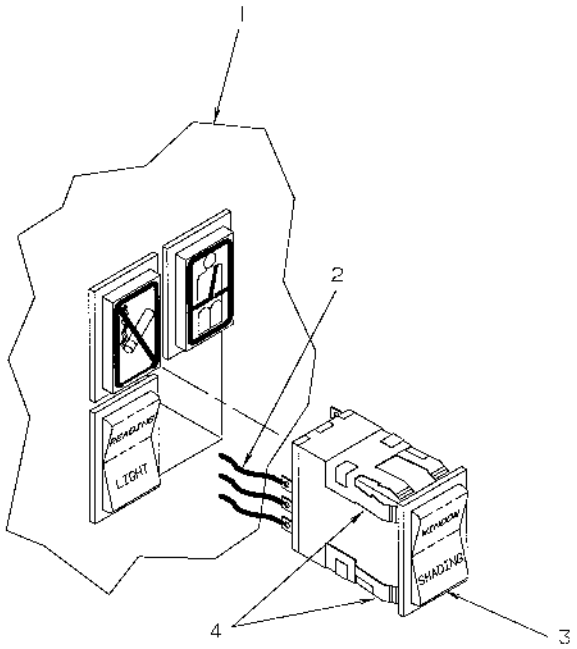
POLARIZED WINDOW INSTALLATION (EFFECTIVITY: NC-4 THRU NC-20) (FIGURE 6)

a. Carefully install the retainer (1) on the inboard window (2).

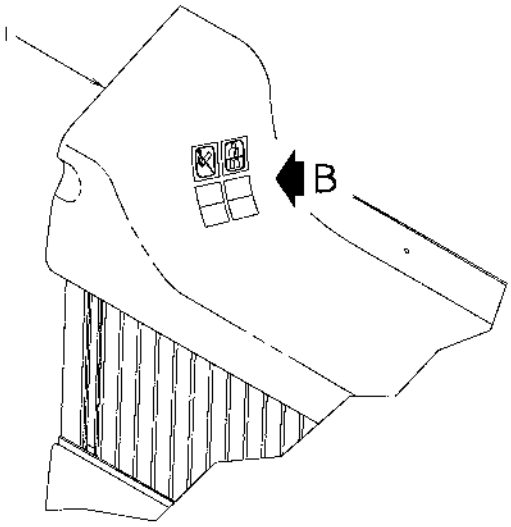
b. Place the outboard window (8) in the shroud (7) so that the flat edge of the window is at the bottom and the trademark will be facing inboard.

1. SIDEWALL CONSOLE ASSEMBLY
2. WIRES
3. SWITCH
4. RETAINING SPRING





DETAIL B



DETAIL A

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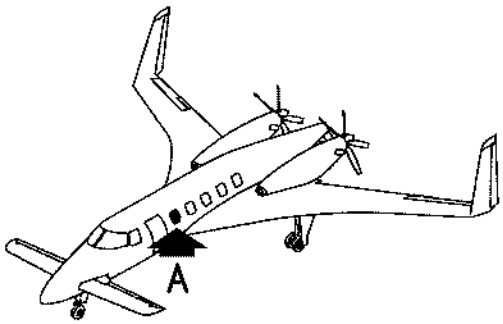
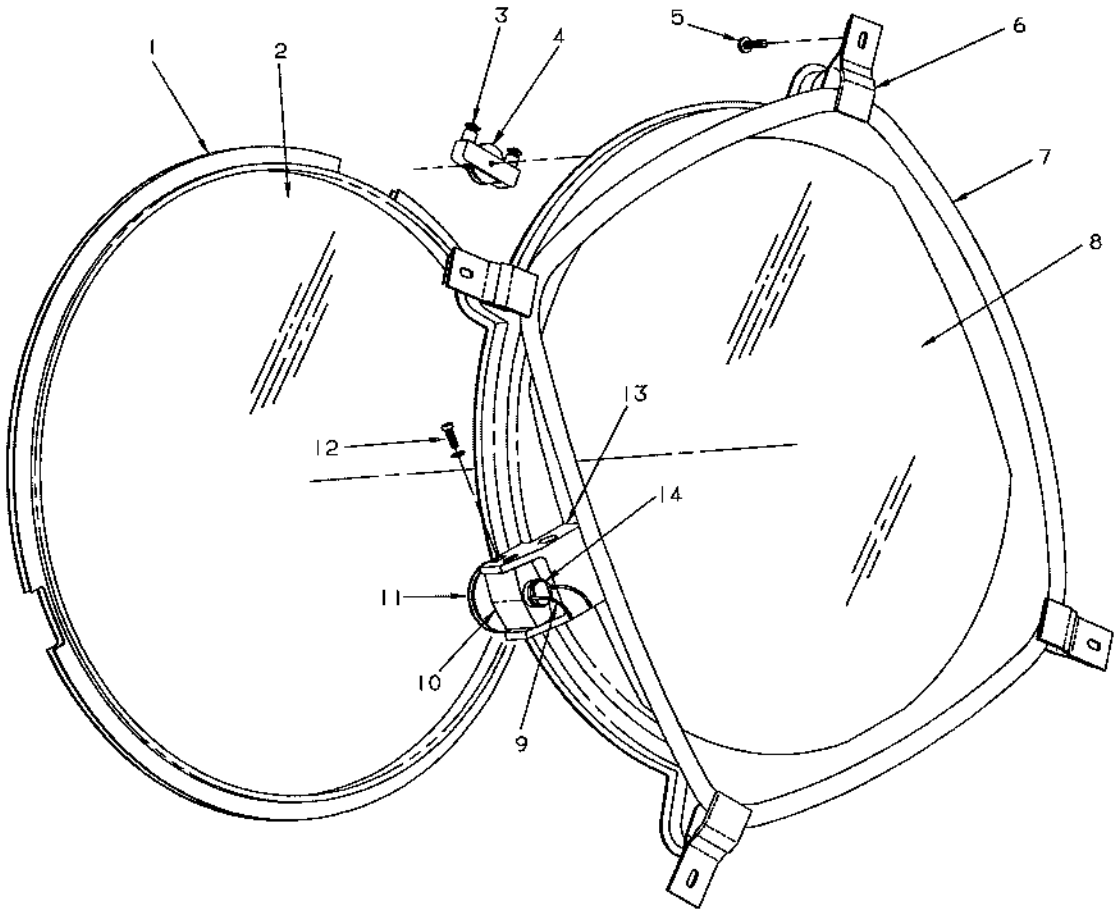
Actuator Motor Switch Installation (Effectivity: NC-4 thru NC-20)
Figure 5

1. RETAINER
2. INBOARD WINDOW
3. SCREW
4. ROLLER
5. SCREW
6. WINDOW CLIP
7. SHROUD
8. OUTBOARD WINDOW
9. WIRE
10. MOTOR BLOCK
11. DRIVE ROLLER
12. SCREW
13. BRACKET
14. ACTUATOR MOTOR





DETAIL A

C94NC56001AB

Polarized Window Installation (Effectivity: NC-4 thru NC-20)
Figure 6

c. Carefully install the inboard window (2) and retainer (1) in the shroud (7) so that the trademark will be facing inboard and the openings for the rollers (4 and 11) are properly aligned.

d. Loosen the roller block assembly screws (3) so the rollers (4) have contact with the inboard window (2).

e. Position the actuator motor block (10) so the drive roller (11) has contact with the inboard window (2) and tighten the screws (12).

f. Before attaching the polarized window assembly to the airplane, check for smooth operation of the assembly by applying 18 volts to the actuator motor (14). If necessary, adjust the roller block assembly screws (3) or the position of the motor block (10) on the bracket (13).

g. Position the window clips (6) and the shroud (7) over the cabin window, and attach them with screws (5).

h. Attach the wires (9) to the actuator motor (14) as tagged during removal.

i. Restore electrical power to the airplane.

j. Press the actuator motor switch to check for proper operation of the polarized window assembly.

k. Install the upholstery around the cabin window. Refer to Chapter 25-20-00.

POLARIZED WINDOW REMOVAL (EFFECTIVITY: NC-21 AND AFTER) (FIGURE 7)

a. Remove the upholstery from around the window. Refer to Chapter 25-20-00.

b. Remove the screws (5) attaching the window shroud (7) to the fuselage. Remove the shroud and window clips (6).

c. Tighten the screws (3) in the roller block assemblies until the rollers (4) are free of the inboard window (2).

d. Carefully remove the inboard window (2) and retainer (1) from the shroud (7).

e. Remove the outboard window (8) from the shroud (7).

f. Carefully remove the retainer (1) from the inboard window (2).

POLARIZED WINDOW INSTALLATION (EFFECTIVITY: NC-21 AND AFTER) (FIGURE 7)

a. Carefully install the retainer (1) on the inboard window (2).

b. Place the outboard window (8) in the shroud (7) so that the flat edge of the window is at the bottom and the trademark will be facing inboard.

c. Carefully install the inboard window (2) and retainer (1) in the shroud (7) so that the knob is at the bottom and the openings for the rollers (4) are properly aligned.

d. Loosen the roller block assembly screws (3) so the rollers (4) have contact with the inboard window (2).

e. Position the window clips (6) and the shroud (7) over the cabin window, and attach them with screws (5).

f. Check for smooth operation of the polarized window assembly by turning the knob clockwise, then counterclockwise. Adjust the roller block assembly screws (3) as required.

g. Install the upholstery around the window. Refer to Chapter 25-20-00.

AFT TOILET TINTED WINDOW REMOVAL (EFFECTIVITY: NC-21 AND AFTER) (FIGURE 8)

a. Remove the upholstery from around the window. Refer to Chapter 25-20-00.

b. Remove the screws (3) attaching the window shroud (5) to the fuselage. Remove the shroud and the window clips (4).

c. Remove the screws (9), washers (7), nuts (6) and brackets (8) attaching the tinted window (2) to the shroud (5).

d. Remove the tinted window (2) from the shroud (5).

e. Remove the molding (1) from the tinted window (2).

AFT TOILET TINTED WINDOW INSTALLATION (EFFECTIVITY: NC-21 AND AFTER) (FIGURE 8)

a. Install the molding (1) on the tinted window (2).

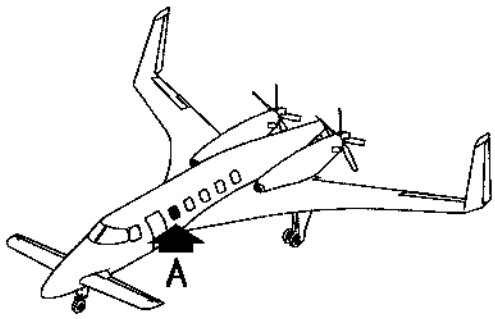
b. Place the tinted window (2) in the shroud (5).

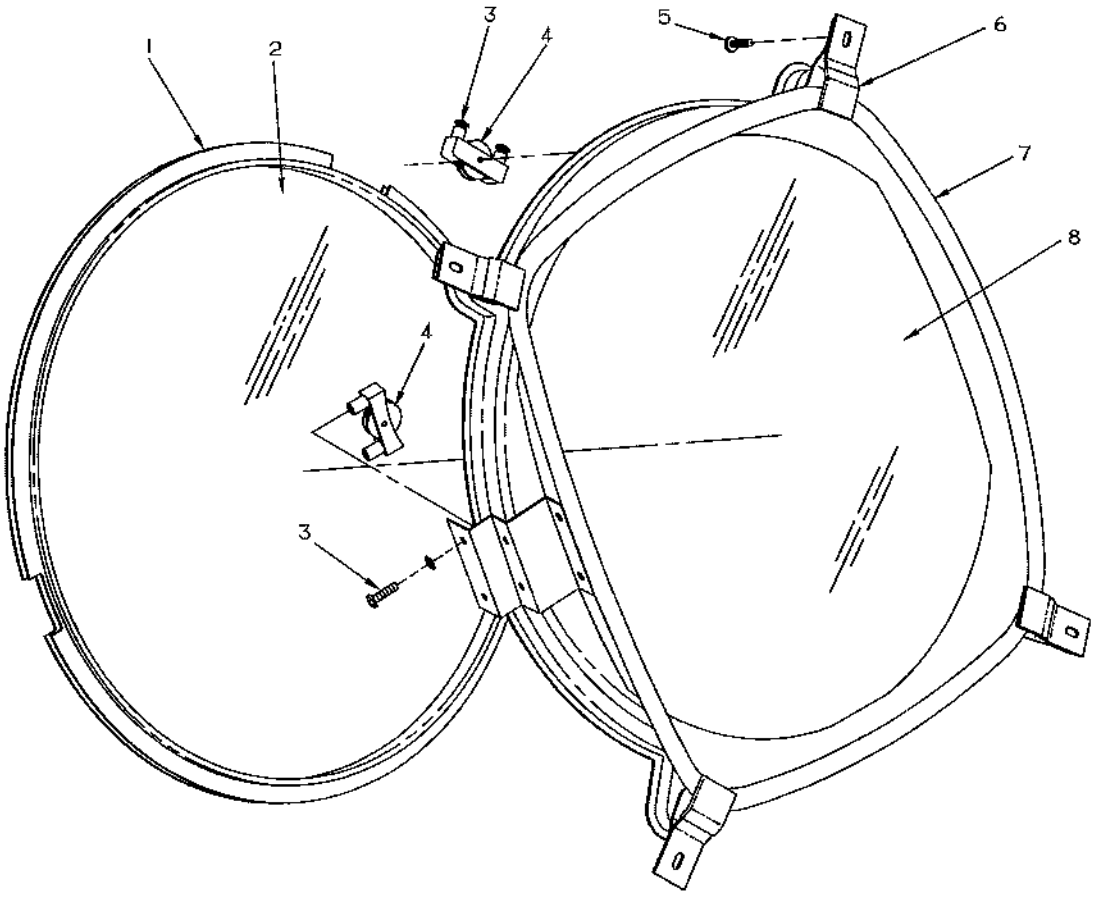
c. Attach the tinted window (2) to the shroud (5) with screws (9), washers (7), nuts (6) and brackets (8).

d. Position the window clips (4) and the shroud (5) over the cabin window, and attach them with screws (3).

e. Install the upholstery around the window. Refer to Chapter 25-20-00.

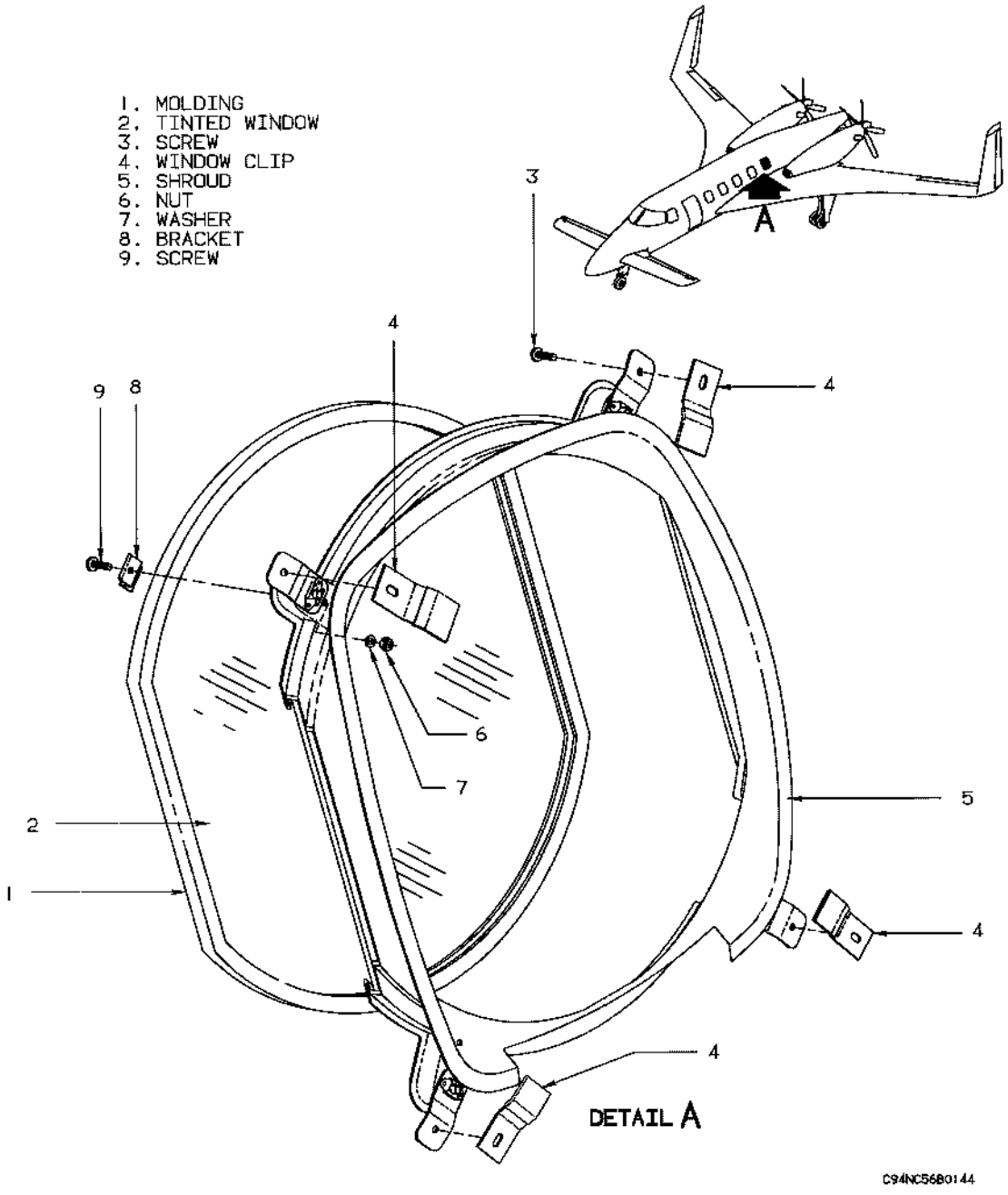
1. RETAINER
2. INBOARD WINDOW
3. SCREW
4. ROLLER
5. SCREW
6. WINDOW CLIP
7. SHROUD
8. OUTBOARD WINDOW





DETAIL A

Polarized Window Installation (Effectivity: NC-21 and after)
Figure 7



Aft Toilet Tinted Window Installation (Effectivity: NC-21 and after)
Figure 8

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CHAPTER 57 - WINGS

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CHAPTER 57 - WINGS

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The aft wing is constructed as a single unit and fabricated of composite materials. Three main spars extend the full length of the wing and are identified as the front spar, center spar and rear spar. Ribs installed between the spars begin inboard with the left and right root ribs. Ribs extend aft at L/RBL 72.34, 139.00 and 206.22 to support the wing flaps. Upper and lower skins cover the spars and ribs. The lower wing surface has access doors for gaining access to components and performing structural inspections.

MAIN LANDING GEAR SUPPORT CLOSURE (Effectivity: All)

The main landing gear is supported between fittings on the wing front spar and the landing gear spar. The main landing gear wheel wells are designed for enclosing the main landing gear when retracted in the wing. Main landing gear support closures outline the retracted landing gear and provide for an aerodynamic lower wing surface. The support closure is constructed of composite materials and consists of bonded support assemblies, angles and shear clips. The support closures are secured with screws. With the landing gear retracted, the attached door of the landing gear is mated with the support closure to complete the enclosure of the main landing gear. For maintenance of the main landing gear doors, refer to Chapter 52-80-00.

AFT FUEL CELL WING CAVITIES (Effectivity: All)

Aft of the main landing gear between the wing front spar and the center spar, cavities in the wing provide for installation of the left and right aft fuel tanks (bladder cells). Access to the aft fuel cell and fuel cell cavity is through an access door on the wing lower surface. For maintenance of the aft fuel cells and the cavities, refer to Chapter 28-10-00.

WING FLAPS (Effectivity: All)

Composite-fabricated wing flaps are located on the wing trailing edge, two wing flaps on each side. Construction of the wing flaps consists of a front spar, rear spar, and inboard and outboard closeout ribs covered with skin. Riblets support the wing flap leading edge. Wire mesh is bonded along the wing flap trailing edge for lightning strike protection. Fittings installed on the inboard and outboard ends of each wing flap have antifriction rollers that travel in flap tracks. For maintenance of the wing flap control system refer to Chapter 27-50-00.

ELEVONS (Effectivity: All)

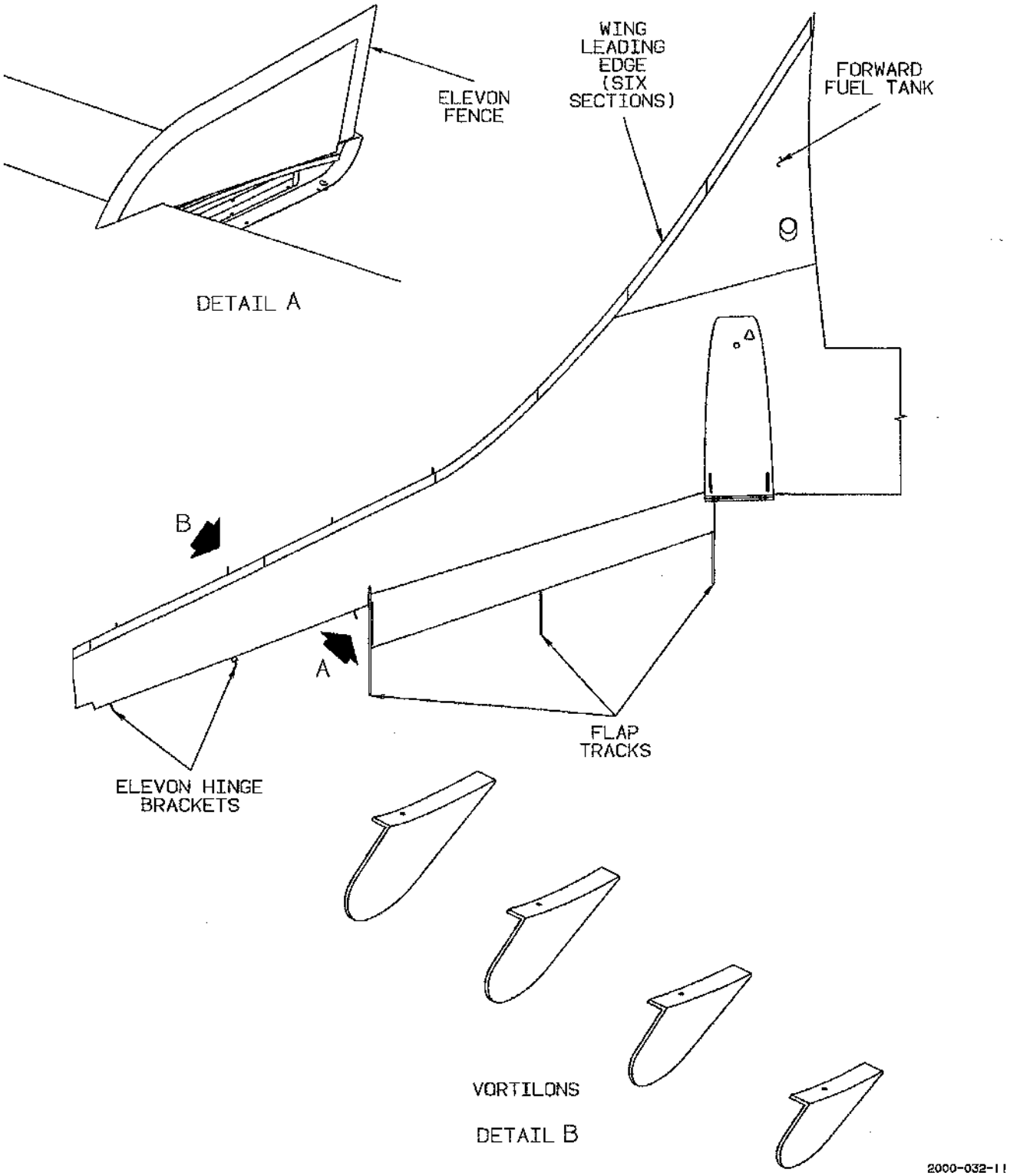
The left and right elevons are secured to the outboard wing trailing edge with hinge brackets. The elevons are constructed of composite material spars, ribs and skins. Riblets are shaped to, and support, the elevon leading edge. A trim tab is hinged on each elevon and is positioned by an electric actuator mounted in the elevon leading edge.

ELEVON FENCES (Effectivity: All)

Left and right elevon fences are mounted to the wing ribs at the outboard wing flap tracks. The elevon fences are fabricated of woven-cloth kevlar and fiberglass with a nomex core. The elevon fences are bonded in place with paste adhesive and secured with rivets.

WING LEADING EDGES (Effectivity: All)

The wing leading edges extend from the fuselage along the forward fuel tank and wing to the wing tip fairing. Each wing leading edge is made up of six sections secured to the forward fuel tank and wing with flush head blind rivets. Deice boots cover the wing leading edges to prevent ice build-up. There are four vortilons mounted under the wing leading edges forward of each elevon.



Aft Wing Assembly (Effectivity: All)
Figure 1

WING ACCESS DOORS (Effectivity: All)

Access doors are located on the lower surface of the wing. The location of the access doors on the right side are typical of the access doors on the left side. The access doors are fabricated of woven-cloth graphite or aluminum. The five inboard wing access doors are secured with captive screws into inserts.

Removal and installation of the access doors on the forward fuel tank, aft fuel tank and the vent scoop access are outlined in Chapter 28. For identification of the access doors, refer to Chapter 6.

WING TRAILING EDGE INBOARD COVES (Effectivity: All)

Left and right wing trailing edge inboard cove assemblies streamline the wing trailing edges between the fuselage and the engine nacelles. The cove assemblies are fabricated of nomex honeycomb core and woven graphite. A spar is bonded in the cove with a doubler riveted to the inboard end. A stud (pin) is secured through the doubler which mates with a bracket during installation of the cove assembly. The cove assembly is secured with stud-nuts along the outboard edge, captive screws along the forward edge and the stud (pin). Reflective aluminum tape is applied inside the outboard upper and lower surfaces to protect the cove assembly from heat.

SPECIAL TOOLS AND RECOMMENDED MATERIALS (Effectivity: All)

The special tools and recommended materials listed in Charts 1 and 2 as meeting federal, military or vendor specifications are provided for reference only and are not specifically prescribed by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All)

TOOL NAME	PART NO.	VENDOR	USE
1.Stand with knife edge supports.	None	Fabricate	Support elevon during balancing procedure.
2.Beam balance weighing device calibrated in units of .001 pounds or less.	None	Obtain Locally	Weigh lead shot used in elevon balancing procedure.
3.Inclinometer	None	Obtain Locally	Determine position of elevon for balancing procedure.
4.Position Indicator Stand	None	Fabricate	Indicate elevon position during balancing procedure.

CHART 2
RECOMMENDED MATERIALS (Effectivity: All)

MATERIALS	SPECIFICATION	PRODUCT	VENDOR
1.Solvent	TT-T-548C	Toluene	Obtain Locally
2.Sealant	MIL-S-8802, Type B-1/2	Pro-Seal 890	Pro-Seal Corp.,Div of Essex Chemical Corp., Aerospace Products Compton, CA 90220
3.Cleaner	TT-N-95	Naphtha	Obtain Locally
4.Aluminum Tape		10100B-1/ 1A 812	Mactac, Stowe, OH 44224

GENERAL - MAINTENANCE PRACTICES (Effectivity: All)

WING LEADING EDGE VORTILON REMOVAL (Effectivity: All) (Figure 1)

Removal of the vortilon from the wing leading edge is typical for each location.

a. Carefully cut around vortilon to loosen the sealant around the edges of the deicer boot cutout.

b. Drill and punch out the rivet securing the vortilon to the wing leading edge structure.

c. Pry vortilon from the wing leading edge using care not to damage the structure under the vortilon.

d. Remove any old sealant from the wing leading edge structure and vortilon.

WING LEADING EDGE VORTILON INSTALLATION (Effectivity: All) (Figure 1)

Installation of the vortilon on the wing leading edge structure is typical for each location. If the vortilons have been removed due to deicer boot replacement, installation of the vortilons will not be accomplished until after the deicer boot has been installed as outlined in Chapter 30-10-00.

a. Ensure that the bonding areas of the vortilon and wing structure are clean and free of old sealant.

b. Brush the sealant (2, Chart 2), on the bonding areas of the vortilon and wing structure as evenly as possible.

c. Align the vortilon hole with the hole in the wing leading edge structure and insert a wet rivet (P/N NAS1921M06S06) using sealant (2, Chart 2), in the vortilon and wing leading edge holes. Leave the rivet loose at this time.

d. Align the vortilon surface within ±2° to the BL plane and secure the vortilon to the wing leading edge with the rivet inserted in the step above.

e. Seal all edges as instructed in the EDGE SEALING procedure outlined in Chapter 30-10-00.

f. After the sealant has cured, clean the vortilon and surrounding area with a clean white rag dampened with solvent (1, Chart 2).

MAIN LANDING GEAR SUPPORT CLOSURE REMOVAL (Effectivity: All) (Figure 2)

Removal of the left and right main landing gear support closure is typical.

a. Remove the screws along the main landing gear forward spar.

b. Remove the screws along the wing front spar.

c. Remove the screws securing the support closure to the angles.

d. Remove the support closure and store to prevent damage.

MAIN LANDING GEAR SUPPORT CLOSURE INSTALLATION (Effectivity: All) (Figure 2)

Installation of the left and right main landing gear support closure is typical.

a. Place the support closure in position to the main gear wheel well.

b. To assist in screw installation, align the screw holes and start the screws in the angles, along the root rib, main landing gear forward spar and the wing front spar.

c. Tighten all screws securing the support closure.

WING ACCESS DOORS REMOVAL (Effectivity: All) (Figure 3)

Removal of the access doors on the wing lower surface is typical except for the type and number of screws/captive screws. The access doors should be identified for location and position so they can be reinstalled as they were removed.

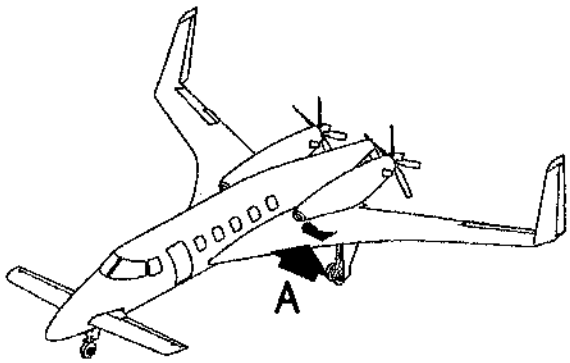
a. Identify the access door(s) to be removed and remove/unscrew the fasteners.

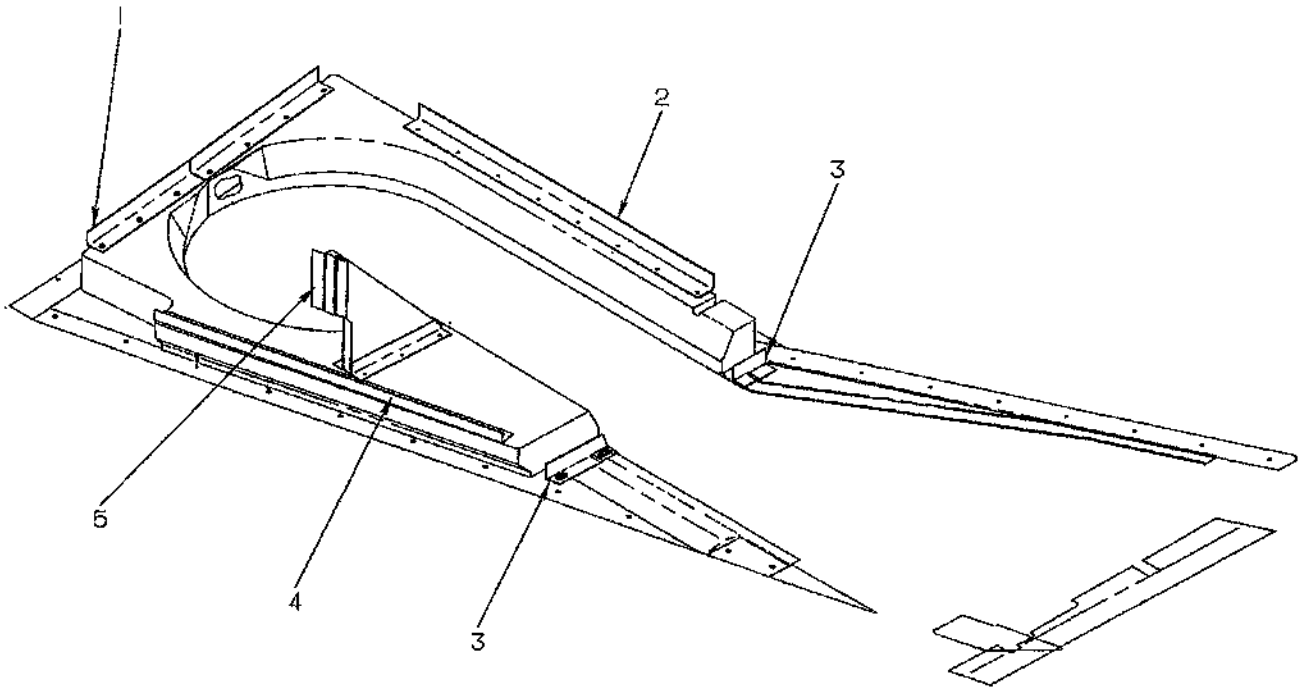
b. Remove the access door(s) and store to prevent damage.

WING ACCESS DOORS INSTALLATION (Effectivity: All) (Figure 3)

Installation of the access doors on the wing lower surface is typical except for the type and number of screws/captive screws.

1. LH ANGLE
2. ANGLE
3. SHEAR CLIP
4. ANGLE
5. LH INBD ANGLE





DETAIL A

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Main Landing Gear Support Closure (Effectivity: All)
Figure 2

a. Identify and position the access door(s) to the access opening.

b. Secure the access door(s) with the fasteners.

WING TRAILING EDGE INBOARD COVE ASSEMBLY REMOVAL (Effectivity: All) (Figure 4)

Removal of the left and right cove assembly is typical.

a. Unscrew the four upper and four lower stud-nut fasteners along the outboard flanges of the cove assembly.

b. Unscrew the six upper and seven lower captive screws along the forward edge of the cove assembly.

c. Remove the cove assembly by pulling aft to separate the stud (pin) from its bracket on the inboard side of the cove assembly.

d. Store the cove assembly to prevent damage.

WING TRAILING EDGE INBOARD COVE ASSEMBLY INSTALLATION (Effectivity: All) (Figure 4)

Installation of the left and right cove assembly is typical.

a. Position and slide the cove assembly forward to engage the stud (pin) in its bracket on the inboard side of the cove assembly. Align the fasteners with their receptacles.

b. Secure the upper and lower forward edge of the cove assembly with captive screws.

c. Secure the upper and lower outboard flanges of the cove assembly with stud-nut fasteners.

TRAILING EDGE INBOARD COVE ALUMINUM TAPE REPLACEMENT (Effectivity: All) (Figure 4)

Aluminum tape is applied to the upper and lower outboard flange area of the cove assemblies. The purpose of the aluminum tape is to reflect heat and protect the cove assembly.

a. Gain access to the aluminum tape by removing the cove assembly. Refer to WING TRAILING EDGE INBOARD COVE ASSEMBLY REMOVAL in this chapter.

b. Peel the strips of aluminum tape from the cove assembly.

c. Clean the cove assembly flanges and inside surface with Naphtha (3, Chart 2), in preparation for the new aluminum tape.

d. Cover the cove assembly flanges and inside surface with aluminum tape (4, Chart 2), in the zone illustrated in Figure 4.

NOTE

Do not apply aluminum tape in a bond joint. Ensure that there is no exposed graphite in the aluminum tape zone.

e. Install the cove assembly. Refer to WING TRAILING EDGE INBOARD COVE ASSEMBLY INSTALLATION in this chapter.

TRAILING EDGE INBOARD COVE ALUMINUM TAPE INSPECTION (Effectivity: All)

The trailing edge inboard cove aluminum tape is to be inspected at intervals outlined in Chapter 4-00-00. For inspection of the aluminum tape on the wing aft spar behind the engine lower firewall and the engine firewall insulation blankets, refer to Chapter 54-00-00.

LEFT INBOARD COVE ALUMINUM TAPE INSPECTION (Effectivity: All)

a. Remove the left inboard cove assembly, refer to WING TRAILING EDGE INBOARD COVE ASSEMBLY REMOVAL in this chapter.

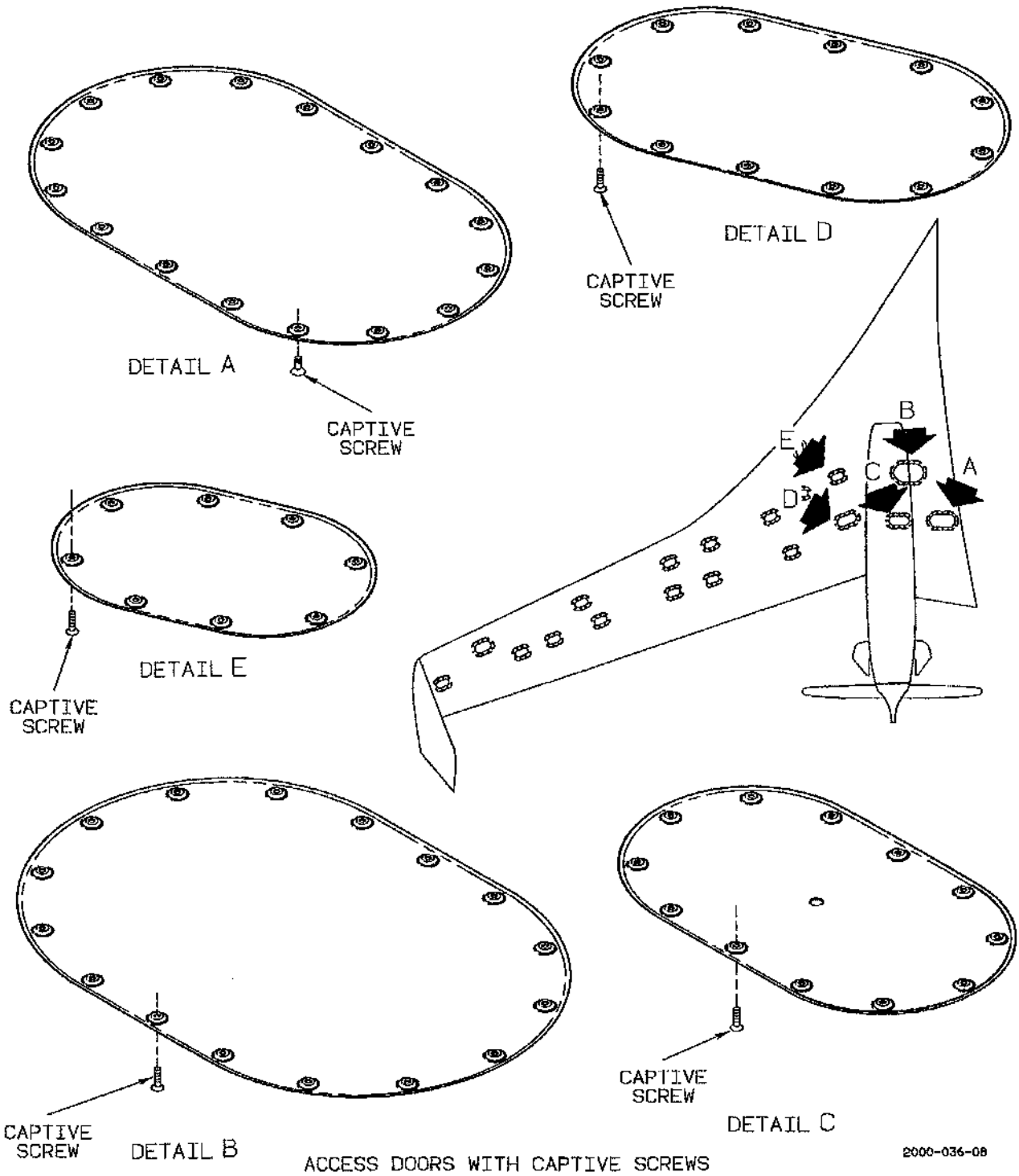
b. Wipe the aluminum tape clean and visually inspect the aluminum tape for disbonds, tears and discoloration (surface not reflective).

c. If any of the above conditions exist, replace the aluminum tape in the affected area, refer to TRAILING EDGE INBOARD COVE ALUMINUM TAPE REPLACEMENT in this chapter.

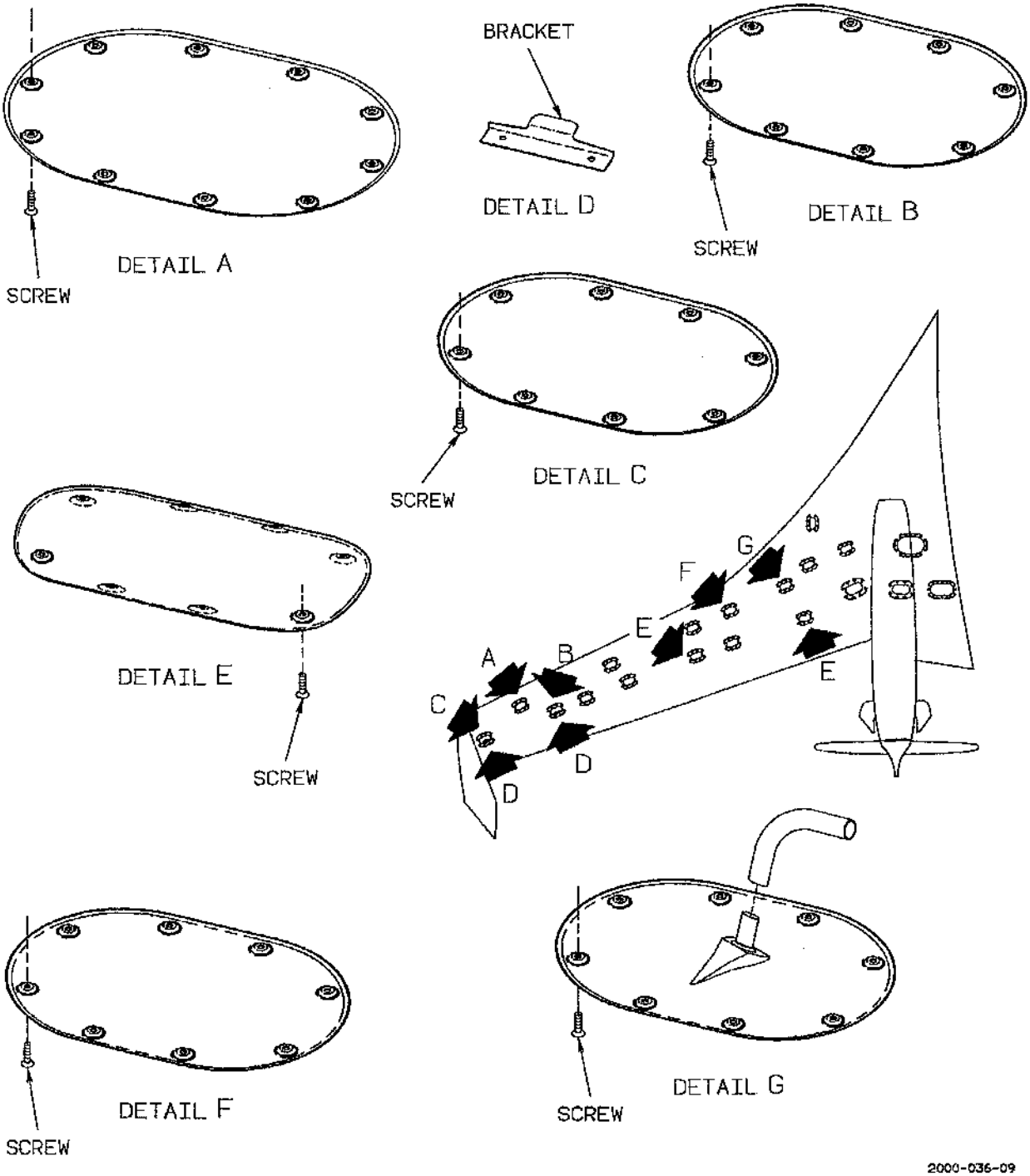
d. Install the left inboard cove, refer to WING TRAILING EDGE INBOARD COVE ASSEMBLY INSTALLATION in this chapter.

RIGHT INBOARD COVE ALUMINUM TAPE INSPECTION (Effectivity: All)

a. Remove the right inboard cove assembly, refer to WING TRAILING EDGE INBOARD COVE ASSEMBLY REMOVAL in this chapter.



Wing Access Doors Installation (Effectivity: All)
(Sheet 1 of 2)
Figure 3

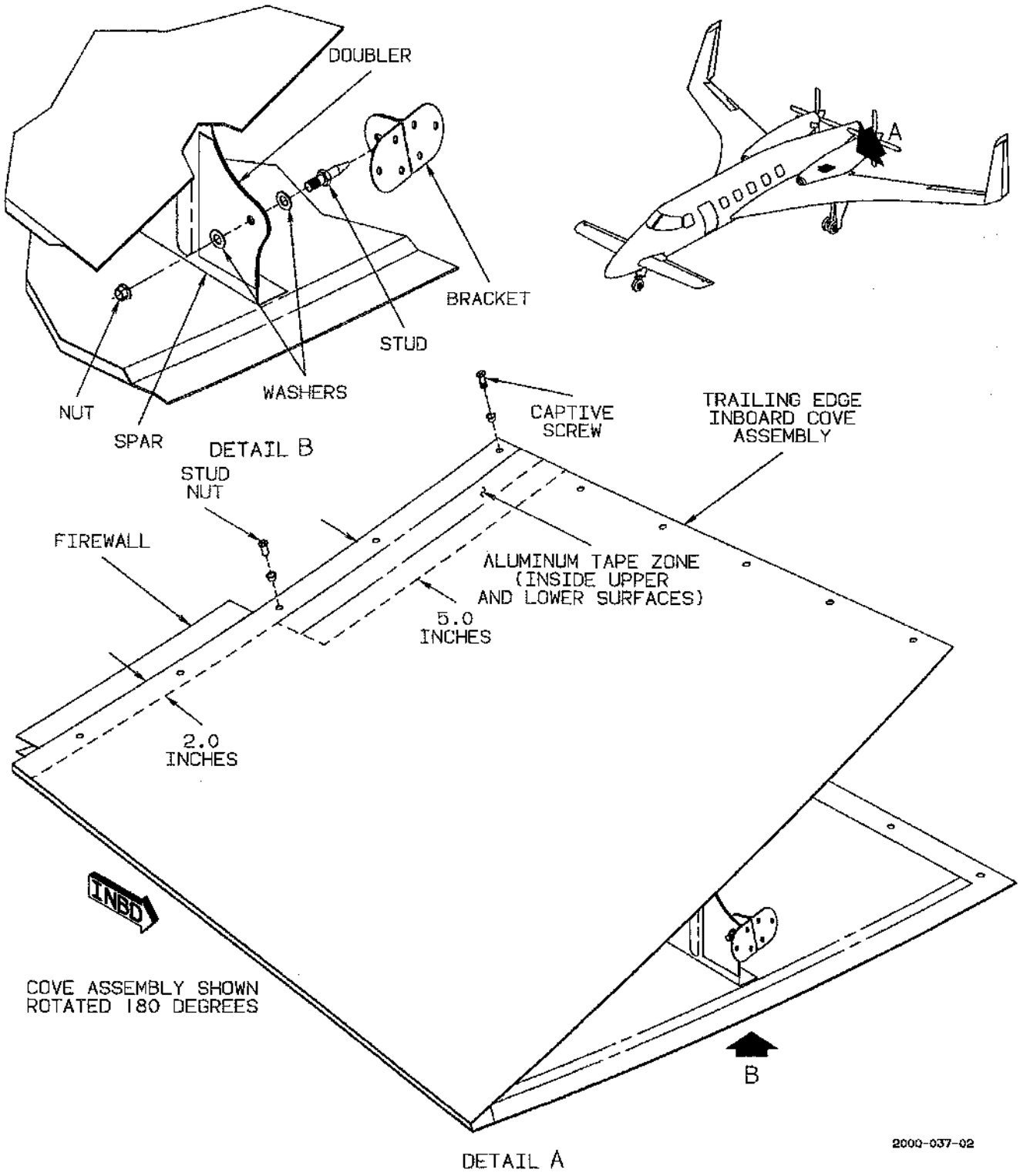


Wing Access Doors Installation (Effectivity: All)
(Sheet 2 of 2)
Figure 3

b. Wipe the aluminum tape clean and visually inspect the aluminum tape for disbonds, tears and discoloration (Surface not reflective).

c. If any of the above conditions exist, replace the aluminum tape in the affected area, refer to WING TRAILING EDGE INBOARD COVE ALUMINUM TAPE REPLACEMENT in this chapter.

d. Install the right inboard cove, refer to WING TRAILING EDGE INBOARD COVE ASSEMBLY INSTALLATION in this chapter.



Wing Trailing Edge Inboard Cove Assembly Installation (Effectivity: All)
Figure 4

ELEVON - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

An elevon and trim tab are mounted on the outboard ends of the aft wing. The elevons are constructed of woven cloth graphite and honeycomb core. The elevon upper and lower skins cover a spar along the cove for the trim tab, inboard and outboard ribs and riblets in the leading edge. Three hinge assemblies mate with the hinge brackets on the aft wing. Each hinge assembly and the trim tab actuator has an access cover. Moisture drain holes are located in the elevon lower skin just outboard of the trim tab and near the elevon inboard rib, aft of the inboard hinge assembly. Wire mesh is bonded along the elevon trailing edge.

Four static discharger bases are mounted (riveted) on each elevon outboard trailing edge. Static discharge wicks thread into the discharger bases. A ground strap is installed between the elevon inboard hinge assembly and the hinge bracket of the aft wing.

The elevon trim tabs are hinged on the elevons with piano type hinges and hinge pins. The trim tabs are fabricated of woven graphite and honeycomb core. Movement of the trim tab is accomplished by an electrical motor-driven actuator mounted in the elevon leading edge and connected to the trim tab with push-pull rods. The leading edge cover between the inboard and center hinges is removable to gain access to the trim tab actuator.

Balance weights are contoured to the elevon leading edge and are secured with adhesive and mechanical fasteners. Trim weights are secured to the riblets on the inboard and outboard sides of the center hinge assembly. The trim weights may be removed or added during balance checks.

NOTE

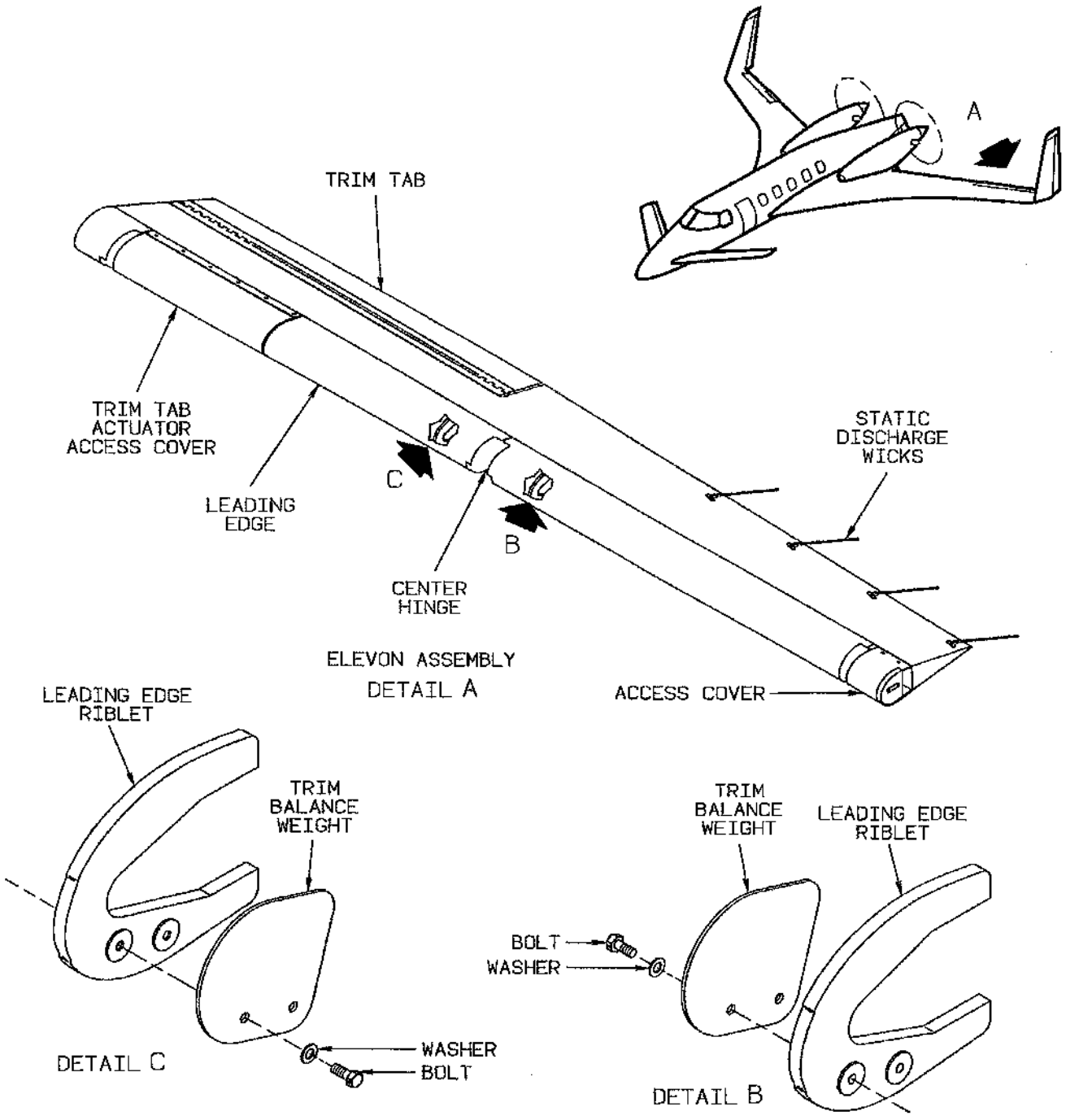
For removal/installation and maintenance of the elevators, elevons and their trim tab control systems, refer to Chapter 27-30-00.

ELEVON - MAINTENANCE PRACTICES (Effectivity: All)

ELEVON BALANCING (Effectivity: All) (Figure 2)

This procedure provides the information necessary to measure the amount of overbalance (nose heavy) of the elevons and to properly balance the elevons. Although many different methods of check balancing exist, counterbalancing is the simplest. Counterbalancing is the application of a known force or weight at a measured distance from the hinge line to counter the unbalance moment of the control surface. The static moment of the elevon is determined by multiplying the unbalanced weight of the elevon assembly by the perpendicular distance from the hinge centerline to the center of gravity when the elevon trailing edge is in a 12 degrees down position. The weight is measured in pounds and the distance is measured in inches. The elevon must be in the following configuration prior to balancing:

- The hinge attachments must be complete with the same bolts that are required for attachment on the airplane.
- The trim tab, trim tab actuator and push-pull rods must be installed with the trim tab positioned in neutral while balancing.
- All access covers must be installed and secured with all fasteners.
- All static discharge wicks must be installed (not temporarily taped or laid upon the surface).
- The elevon must be finish painted, including stripes and touch-up.



Elevon and Trim Balance Weights (Effectivity: All)
Figure 1

NOTE

Whenever the elevon and/or trim tab is replaced, modified, reworked or painted in any way, or the trim tab actuator is replaced, the elevon must be balance checked to ensure that its static moment about the hinge line is within limits. When a new or replacement elevon is to be installed, a balance check must be made just prior to installation.

EQUIPMENT REQUIRED TO PERFORM CHECK BALANCING BY COUNTERBALANCING METHOD (Effectivity: All)

The following equipment is required to check balance an elevon using the counterbalancing method:

a. A stand with knife edge supports (1, Chart 1, 57-00-00), as illustrated in Figure 2. The knife edges must be in the same horizontal plane.

b. A paper cup or similar light weight container.

c. Approximately 1 pound of lead shot.

d. A certified beam balance weighing device calibrated in units of .001 pound or less (2, Chart 1, 57-00-00).

e. A ruler and an inclinometer (3, Chart 1, 57-00-00).

f. A position indicator stand (4, Chart 1, 57-00-00).

ELEVON BALANCE CHECK (COUNTERBALANCE METHOD) (Effectivity: All) (Figure 2)

The static moment of a 100 percent balanced elevon assembly is 0.0 pound-inches. Tail heaviness indicates static underbalance while nose heaviness indicates static overbalance.

NOTE

The static balance for the elevon assemblies must fall between 0.0 and 6.50 pounds-inches (nose heavy) at the measured moment about the hinge line.

a. Verify that the trim tab, trim tab actuator and push-pull rods are installed with the trim tab positioned in neutral.

b. Verify that all access covers are installed and secured with all fasteners.

c. Verify that all static discharge wicks are installed (not temporarily taped or laid upon the surface).

d. The elevon must be finish painted, including stripes and touch-up.

e. Verify that the hinge attach bolts are assembled in the hinge brackets and mount the elevon on the stand with knife edge supports (1, Chart 1, 57-00-00). Ascertain that the elevon is free to rotate about the the hinge centerline.

f. With the elevon in the upright position and the trailing edge 12 degrees down as measured with the inclinometer (3, Chart 1, 57-00-00) on the upper skin, mark the trailing edge position on the indicator stand (4, Chart 1, 57-00-00) with a marking pencil.

g. To determine if trim balance weights should be added or removed, suspend a paper cup from a point near the center of the elevon trailing edge. Use a short length of small diameter string secured to the surface with a small piece of masking tape as illustrated in Figure 2. The cup must be free to hang vertically.

NOTE

Static balance must be checked in a draft free area.

h. Add small quantities of lead shot to the cup until the elevon balances in the 12 degrees trailing edge down position marked on the indicator stand.

i. Carefully measure the perpendicular distance "D" within 0.1 inch from the hinge line to the point of suspension of the cup.

j. Remove the cup, contents and string, then weigh them to within .005 pound.

NOTE

Since any weighing error is magnified by the distance "D", weighing the cup, contents and string is most important and must be done carefully on scales that are certified for accuracy.

k. Calculate the elevon static balance from the formula M = W x D as follows:

1. The weight of the cup and contents is designated by "W".

2. The distance between the centerline of the hinge and the suspension point of the cup is designated by "D".

3. The overbalance or underbalance moment is designated by "M".

l. The following is typical example of a balancing calculation:

1. Assume the elevon is overbalanced (nose heavy) and the paper cup was suspended from the trailing edge.

2. If the elevon balances at "W = .140 pound" and "D = 12.8 inches", then M :eq. .140 X 12.8.

3. M = 1.79 pound-inches (the product of "W x D" must be accurate to within 0.05 pound-inches). In this example, "M" is within the required static balance range and is therefore acceptable.

m. If the static balance does not fall within the range of 0.0 to 6.5 pound-inches nose heavy, trim balance weights must be added or removed and the elevon balance rechecked.

NOTE

A maximum of six trim balance weights (P/N 122-130110-7) or twelve trim balance weights (P/N 122-130110-9) are allowed per elevon, refer to Chart 1. Adjust the elevon balance by adding or removing weights (P/N 122-130110-7 and/or -9) to both of the riblets (inboard and outboard) at the elevon center hinge. When installing the trim weights, evenly divide the weight between the inboard and outboard riblets.

n. Safety wire (MS20995-C20-6) must be used if any trim weights are installed.

CHART 1
ELEVON TRIM WEIGHTS AND BOLTS (EFFECTIVITY: ALL)

Combination Wts Required		P/N Of Bolt For Each Installation	Number Bolts Required (One AN960-C10 Washer Per Bolt)
-7 Outbd Side of Hinge	-9 Inbd Side of Hinge		
1	0	AN3H4A	2
0	1	AN3H3A	2
1	1	AN3H4A	2
2	1	AN3H5A	2
3	0	AN3H6A	2
2	0	AN3H5A	2

Large Weights (122-130110-7) = .28 lb each, .125 inch thick, maximum 6
Small Weights (122-130110-9) = .14 lb each, .0625 inch thick, maximum 12

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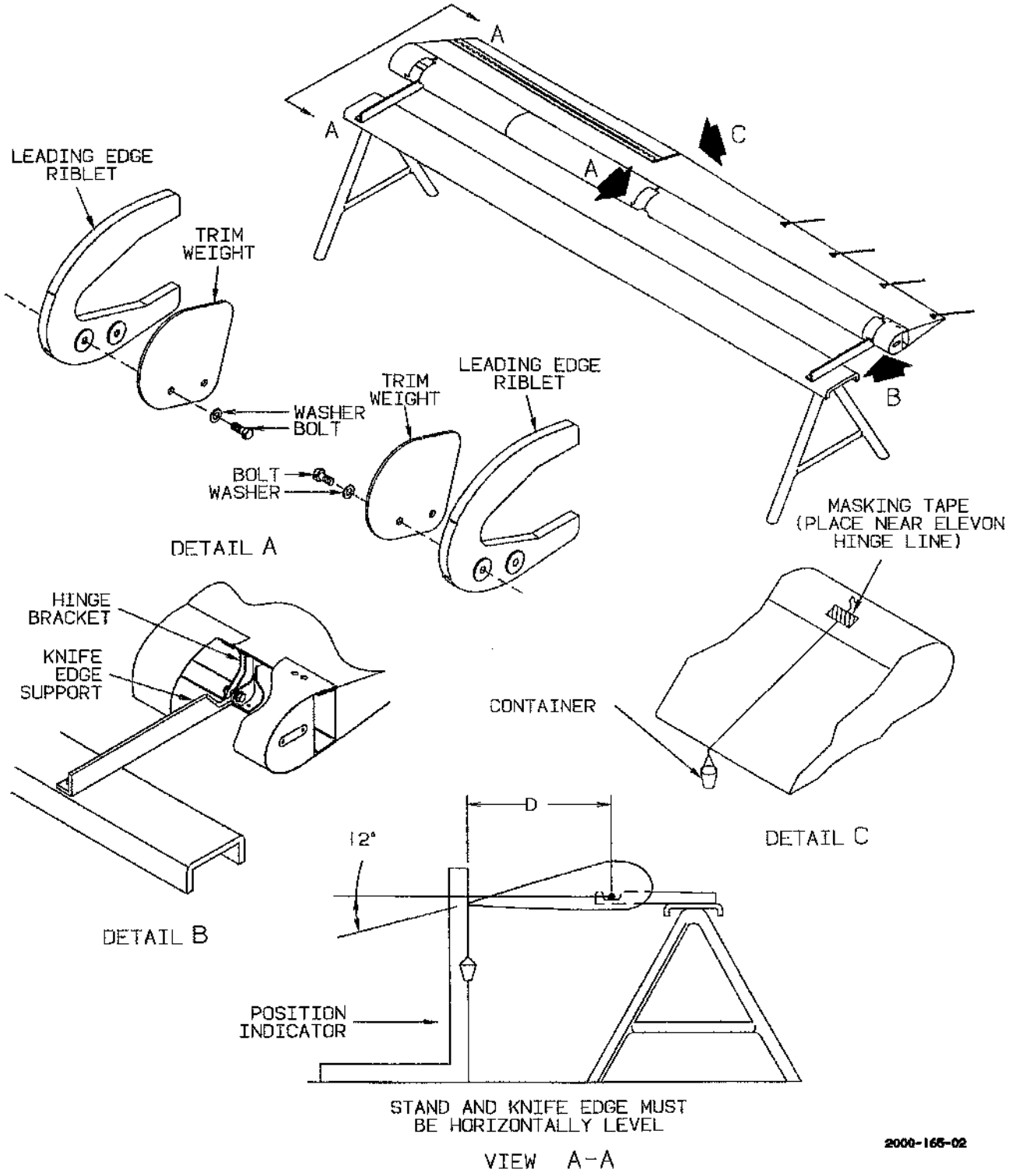

Elevon Balancing (Effectivity: All)
Figure 2

CHAPTER 61 - PROPELLERS

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The Starship 1 utilizes two McCauley five-bladed propellers fitted with standard hubs mounted on the aft end of their respective engine in a pusher configuration. The installation is unique due to a slight difference in feather angle between the LH and RH propellers. The LH propeller is a McCauley P/N 5JFR36C1003-F/D-L104DSZ-0 (P10036062-03) and the RH propeller is a McCauley P/N 5JFR36C1003-F/D-L104DSZ-0 (P10036062-04). The propellers are of the constant speed, reversible, full-feathering type and are hydraulically controlled by oil flow diverted from the engine.

SPECIAL TOOLS AND RECOMMENDED MATERIALS (Effectivity: All)

The special tools and recommended materials listed in Charts 1 and 2 as meeting federal, military or vendor specifications are provided for reference only and are not specifically prescribed by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All)

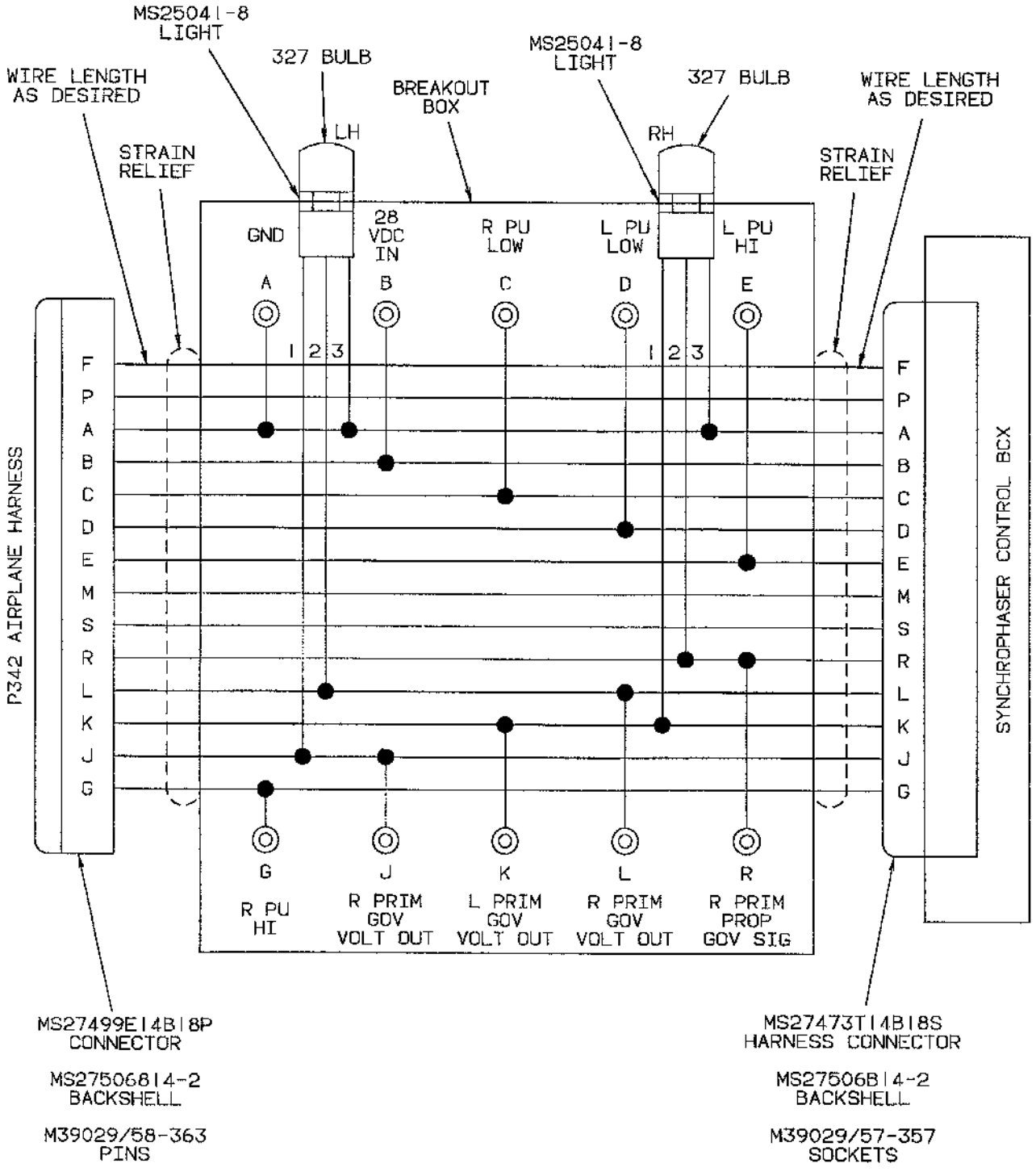
TOOL NAME	PART NO.	VENDOR	USE
1. Beta Retractor Tool	E-5945	McCauleyAccessory Division	Retract beta feedback collar.
2. Dynamic Balancer	Model 8500	Chadwick-Helmuth Aviation Products Co., Inc. 4601 N. Avden Dr. El Monte, CA 91731	Propeller Dynamic Balancing
3. Adapter Cable	10390	Chadwick-Helmuth Aviation Products Co., Inc. 4601 N. Avden Dr. El Monte, CA 91731	Propeller Dynamic Balancing
4. Junction Box	9110	Chadwick-Helmuth Aviation Products Co., Inc. 4601 N. Avden Dr. El Monte, CA 91731	Propeller Dynamic Balancing

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All) *(Continued)*

TOOL NAME	PART NO.	VENDOR	USE
5. Breakout Box	None	Fabricate in accordance with Figure 1.	Check of synchrophasing system.

CHART 2
RECOMMENDED MATERIALS (Effectivity: All)

MATERIALS	SPECIFICATION	PRODUCT	VENDOR
1. Anti-Seize Compound	MIL-T-5544	Petrolated Graphite	
2. Threadlock Adhesive		Locktite 222	Locktite Corporation 705 N. Mountain Rd Newington, CT 06111
3. Grease Lubricating		Plastilube No. 3	Parr Inc. Cleveland, OH 44104



Synchrophaser Breakout Box Schematic (Effectivity: All)
Figure 1

PROPELLER ASSEMBLY - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

PROPELLER ASSEMBLY (Effectivity: All) (Figures 1, 2, 3, and 4)

Propeller speed control is accomplished by a single acting engine-driven primary governor. An overspeed governor and a (NF) power turbine governor are used as a backup in case of a primary governor failure. A servo piston mounted on the aft side of the propeller spider hub moves the propeller blades through links connected to the trailing edges. Centrifugal counterweights on each blade, in conjunction with a feathering spring on the servo piston, increase pitch (which decreases rpm) toward the feathered position as oil pressure from the governor is relieved. The feathering spring completes the feathering operation when centrifugal twisting movement is lost as the propeller stops rotating. The system also provides a means of immediately dumping oil from the primary governor, thereby enabling the feathering springs to feather the propellers immediately after oil pressure at the engine torquemeter drops below 4.7 psi at power settings above 86 to 89% N_1 speed.

The servo piston is connected by five spring-loaded sliding rods (beta rods) to the beta feedback collar which is located forward of the propeller. The movement of the beta feedback collar is transmitted by the feedback bearing assembly through the reversing lever and connecting linkage to a beta valve. The beta valve is positioned to maintain propeller blade angle while propeller rpm is lower than that of the governor pilot valve, as selected by the control. The reversing lever is also connected to the power turbine governor to limit propeller rpm in the reverse position. A push-pull cable extends from the reversing lever forward, to a control beta cam box connected to the power lever control and fuel control unit. Movement of the control is transmitted through the beta cam box and interconnecting linkage to the fuel control unit and governor to provide propeller speed and pitch regulation.

FIVE-BLADED PROPELLER - SETTINGS (Effectivity: All)

The McCauley five-bladed propellers have different blade angle settings. The LH propeller has a full-feathered angle of 90.9°. The RH propeller has a full-feathered angle of 91.7°. Reverse blade angle is hydraulically controlled and is set to a specific torque reading. Maximum propeller rpm is 1700 at take-off. Refer to Chapter 76-00-00 for detailed information on propeller rigging and propeller adjustment procedures.

NOTE

The synchrophaser must be turned OFF while making these adjustments.

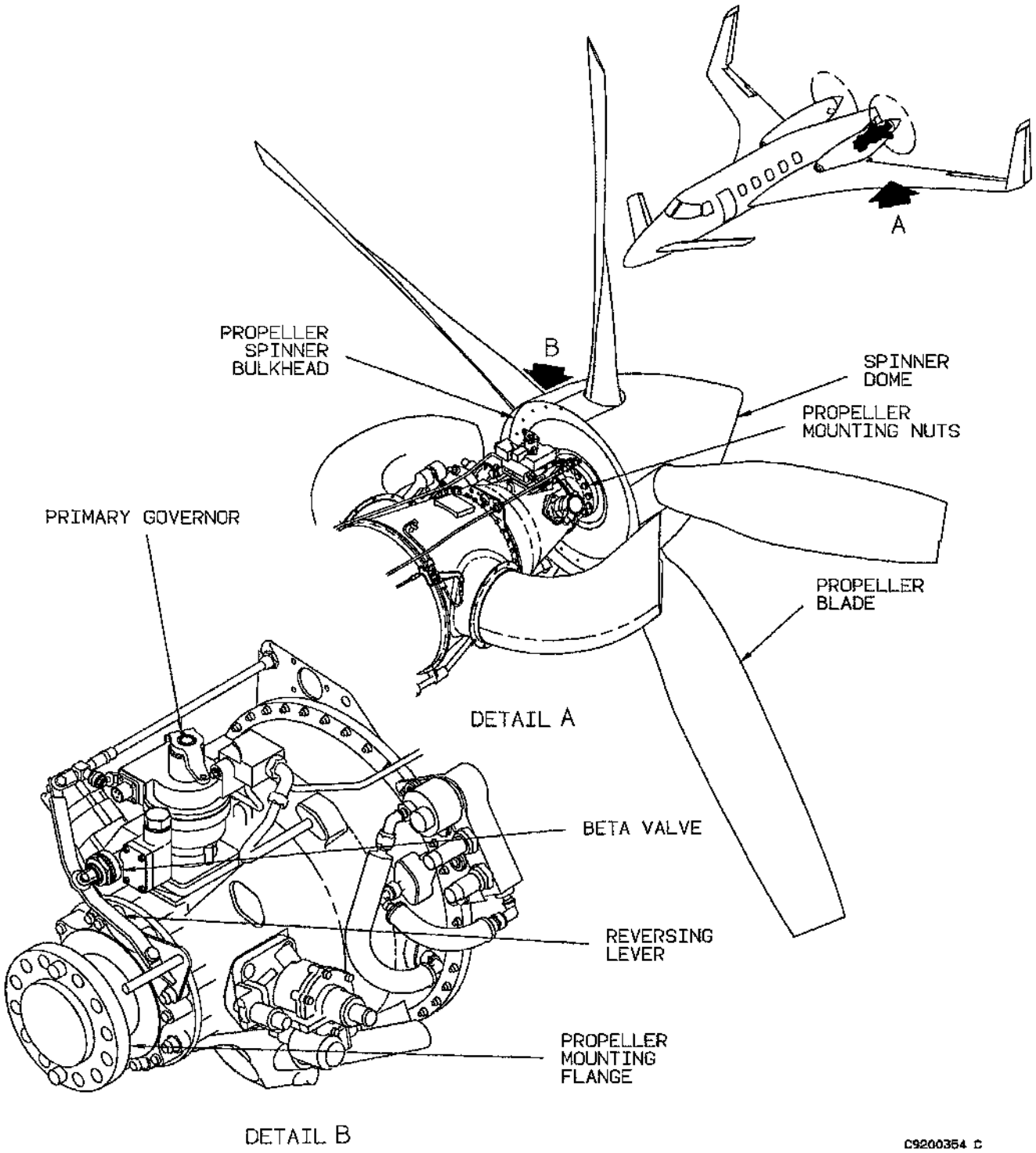
PROPELLER ASSEMBLY - MAINTENANCE PRACTICES (Effectivity: All)

WARNING

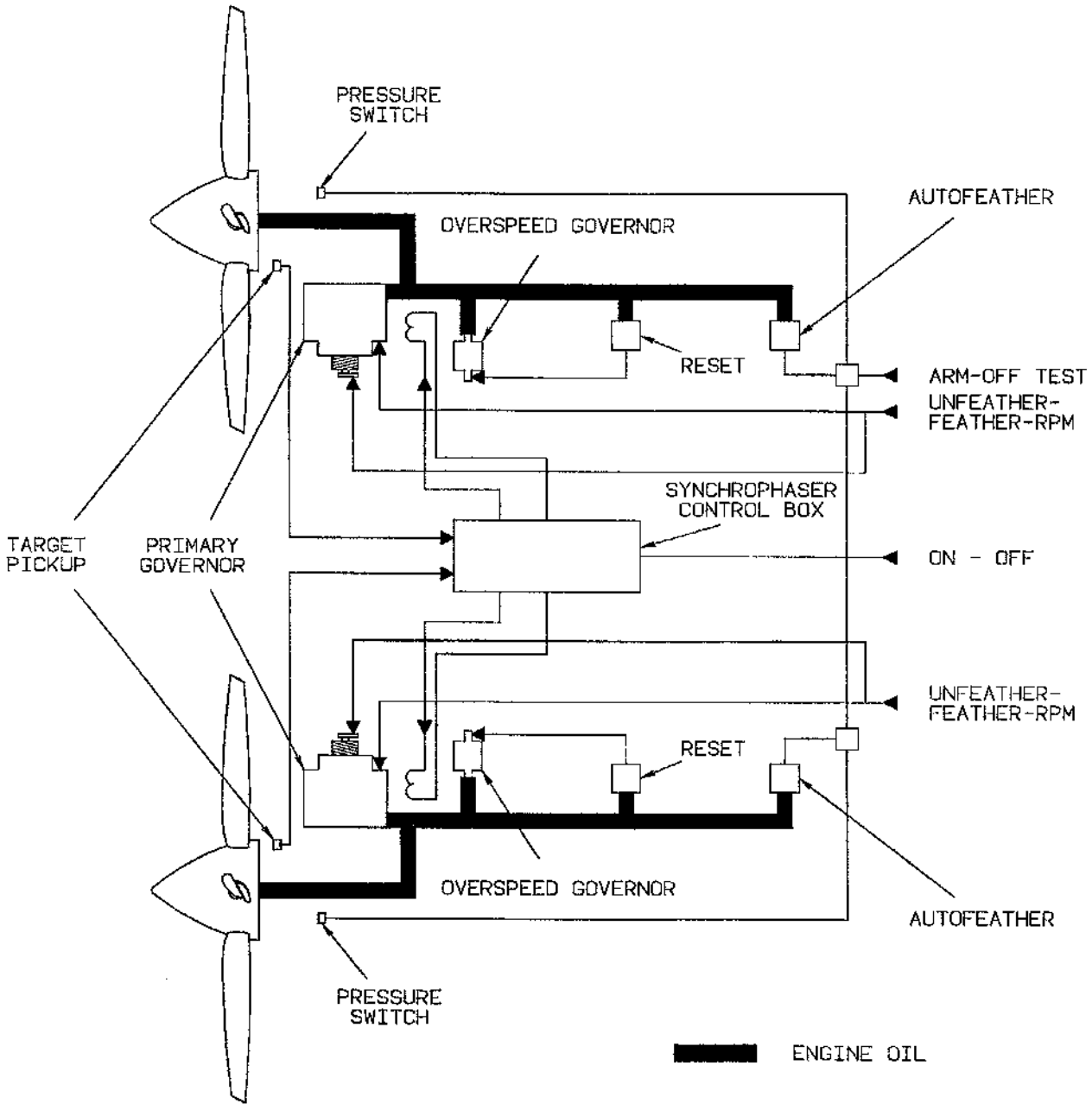
Prior to performing engine maintenance, pull and tag the START CONTROL and IGNTR POWER circuit breakers for the engine being worked on.

PROPELLER REVERSING HARDWARE CLEANING (Effectivity: All) (Figure 3)

The propeller reversing hardware (nonaluminum parts) should be rubbed clean with steel wool and coated with an clear lacquer spray.



Propeller Assembly (Effectivity: All)
Figure 1



Propeller Control (Effectivity: All)
Figure 2

PROPELLER REMOVAL (Effectivity: All) (Figures 3 and 4)

a. Remove all electrical power from the airplane and disconnect the battery.

b. Remove the aft engine cowling panels as outlined in Chapter 71-10-00.

c. Remove the propeller spinner dome by removing the attaching screws from the circumference of the spinner dome and pulling the spinner dome away from the support bulkhead.

d. Disconnect the propeller reversing lever from its top end at the propeller control linkage.

CAUTION

Ensure that the retractor tool is centered and straight to avoid damaging the propeller. Caution should be exercised to avoid bending or damaging the spring-loaded rods and the low pitch stop collar (brass ring).

e. Install the beta retractor tool (1, Chart 1, 61-00-00) on the propeller assembly and move the beta feedback collar to allow access to the attachment area.

CAUTION

Provide adequate support to the propeller assembly (engine hoist, etc.) before removing the propeller mounting nuts (the propeller weighs 203 pounds).

f. Remove the twelve propeller mounting nuts from the propeller mounting studs.

g. Remove the propeller from the propeller shaft.

PROPELLER INSTALLATION (Effectivity: All) (Figures 3 and 4)

a. Install a new O-ring seal on the engine shaft.

b. Prior to installing the propeller, inspect the threads of both the studs and nuts for nicks, burns or other damage. Verify that the threads are clean and free from foreign material.

c. Liberally apply an antiseize compound (graphite) (1, Chart 2, Chapter 61-00-00) to the threads of the propeller mounting studs and to the face of the propeller mounting nuts per MIL-T-5544.

d. Install the propeller on the engine by inserting the propeller mounting studs on the propeller into the mounting holes in the propeller shaft.

e. Secure the propeller to the propeller shaft by screwing the propeller mounting nuts onto the propeller mounting studs. Install the propeller mounting nuts in a star pattern and torque to 68 to 72 foot pounds.

NOTE

Refer to Chapter 20-00-01 for torque wrench usage and extension formulas. All torques in this procedure are "wet torques".

f. Connect the propeller reversing lever to the propeller control linkage.

g. Remove the beta retractor tool.

h. Install the feedback bearing assembly in the beta feedback collar ring.

NOTE

Side clearance between the bearing and the groove in the beta feedback collar should be .001 to .002 inch through one complete 360° turn of beta ring travel.

A slight amount of carbon may be removed from the sides of the feedback bearing assembly to obtain the required clearance.

i. Check the propeller reversing linkage for proper rigging and adjustment as outlined in Chapter 76-00-00.

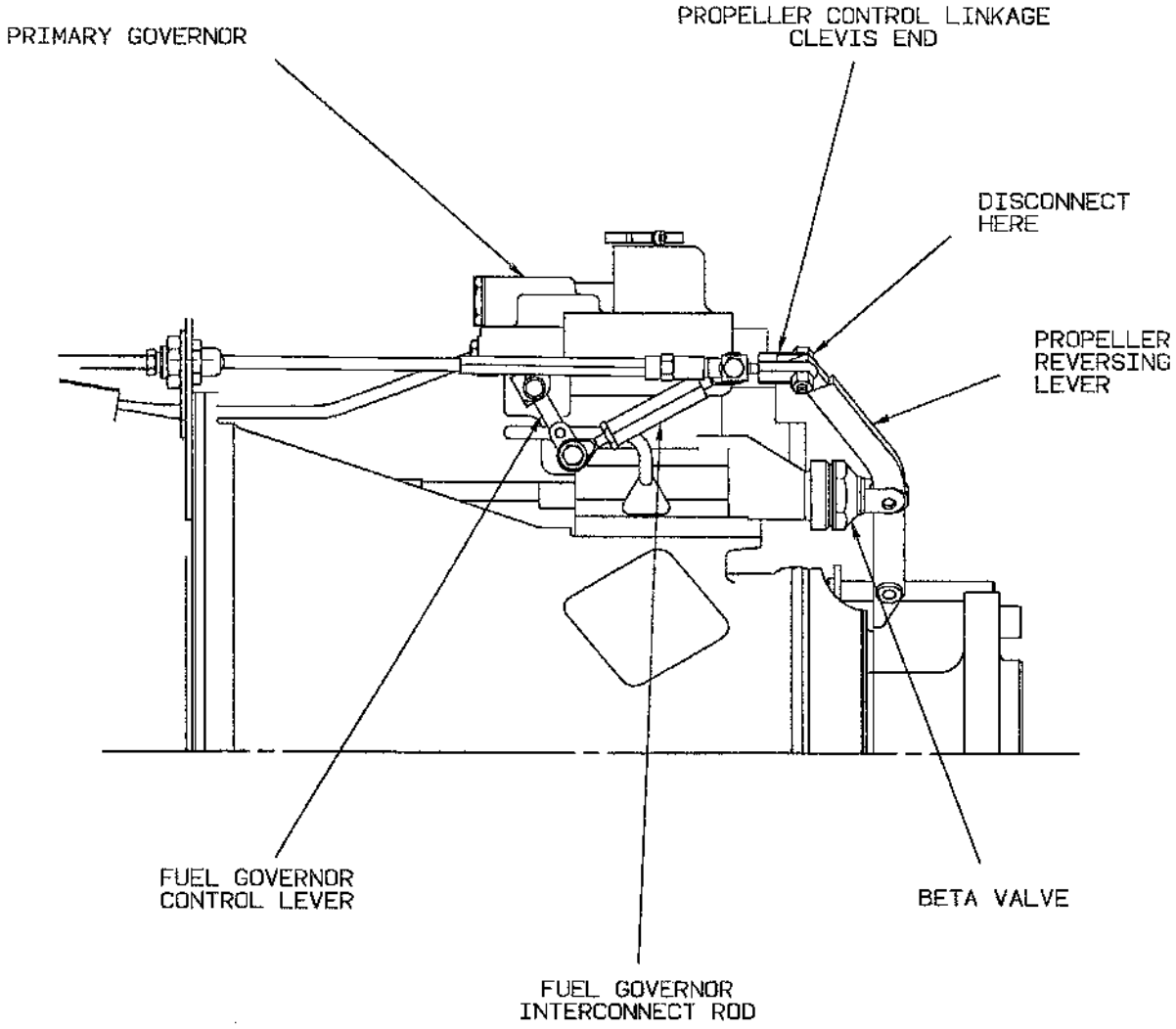
j. Install the propeller spinner dome and secure with the screws.

k. Check the synchrophaser magnetic pickup clearance as outlined in MAGNETIC PICKUP INSTALLATION, Chapter 61-22-00.

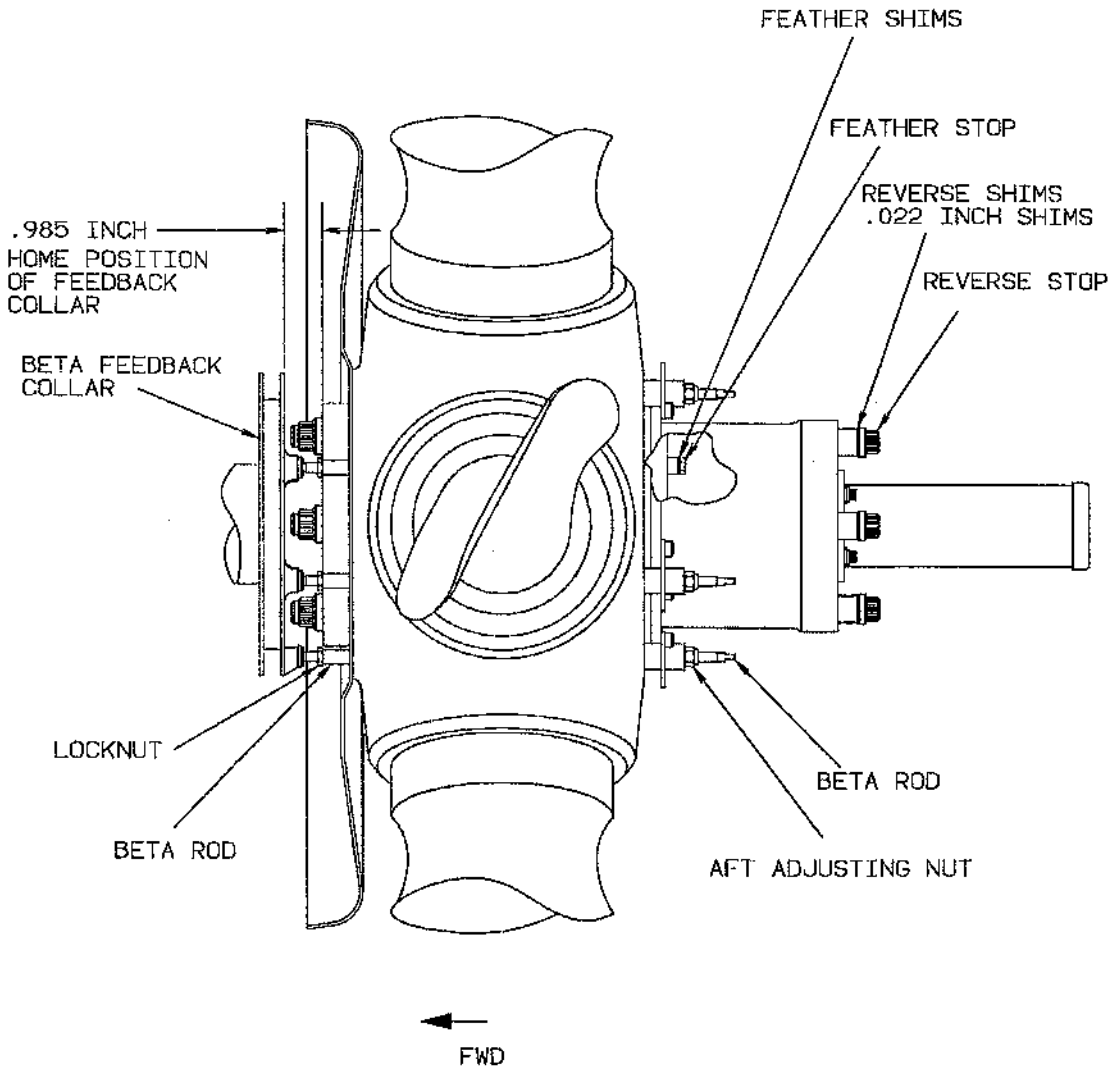
l. Install the engine cowling as outlined in Chapter 71-10-00.

CAUTION

Do not run engines with the propeller spinners off or damage to the oil cooler exit duct may occur when the propeller blade closeouts bend as a result of centrifugal force.



Propeller Control Linkage (Effectivity: All)
Figure 3



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Propeller LH View (Effectivity: All)
Figure 4

m. Refer to Chapter 76-00-00 and perform the PROPELLER CONTROLS RIGGING.

PROPELLER SHAFT OIL SEAL REPLACEMENT (Effectivity: All) (Figures 5 and 6)

A propeller shaft oil seal of nonsplit configuration is installed on all new or overhauled engines. Should it become necessary to replace the propeller shaft oil seal between overhauls, a like nonsplit type may be installed as follows. Refer to Chapter 72-16-01 in the latest revision of the Pratt & Whitney PT6A-67A Engine Maintenance Manual for proper handling of the garter spring and propeller flange gasket.

a. Remove the upper and lower aft engine cowlings as outlined in Chapter 71-10-00.

b. Remove the propeller as outlined in this section.

c. Remove the eight bolts that secure the seal retaining plate to the thrust bearing cover. Slide the seal retaining plate aft against the propeller shaft flange.

d. Remove and discard the gasket.

e. Remove the seal from the cavity in the thrust bearing cover while exercising care not to damage the bore of the thrust bearing cover assembly or the surface of the oil seal race.

f. Remove the garter spring from the back of the seal. Cut and discard the seal.

g. Coat the new seal with clean engine oil. See the latest revision of Pratt & Whitney Service Bulletin No. 14001 for the correct type of engine oil.

h. Gently work the seal over the propeller mounting flange and work the seal retaining ring on to the propeller shaft. The garter spring will not be visible when the seal is properly installed.

CAUTION

Avoid applying loads locally to the lip of the oil seal. The seal element is soft and subject to damage unless handled with care at all times.

i. Reinstall the garter spring as follows:

1. Grasp the narrow end of the spring with the right-hand, and grasp the straight end of the spring with the left hand.

2. While holding the straight end (LH) firmly, rotate the narrow end (RH) 2 turns counterclockwise (the spring helix is right-handed).

3. Insert the narrow end (RH) into the straight end (LH) and rotate the narrow end 2 turns clockwise to secure the ends together.

4. Position the spring in the slot on the backside of the seal.

j. Gently press the seal into its cavity.

NOTE

Verify that the seal is evenly pushed into the proper position.

k. Align the gasket profile with the bolt pattern and install it over the propeller flange and seal retaining ring.

l. Place the seal retaining ring in position and secure it to the thrust bearing cover with the eight bolts. Torque the bolts to 36-40 inch-pounds and secure them in pairs with safety wire.

m. Install the propeller as outlined in this section.

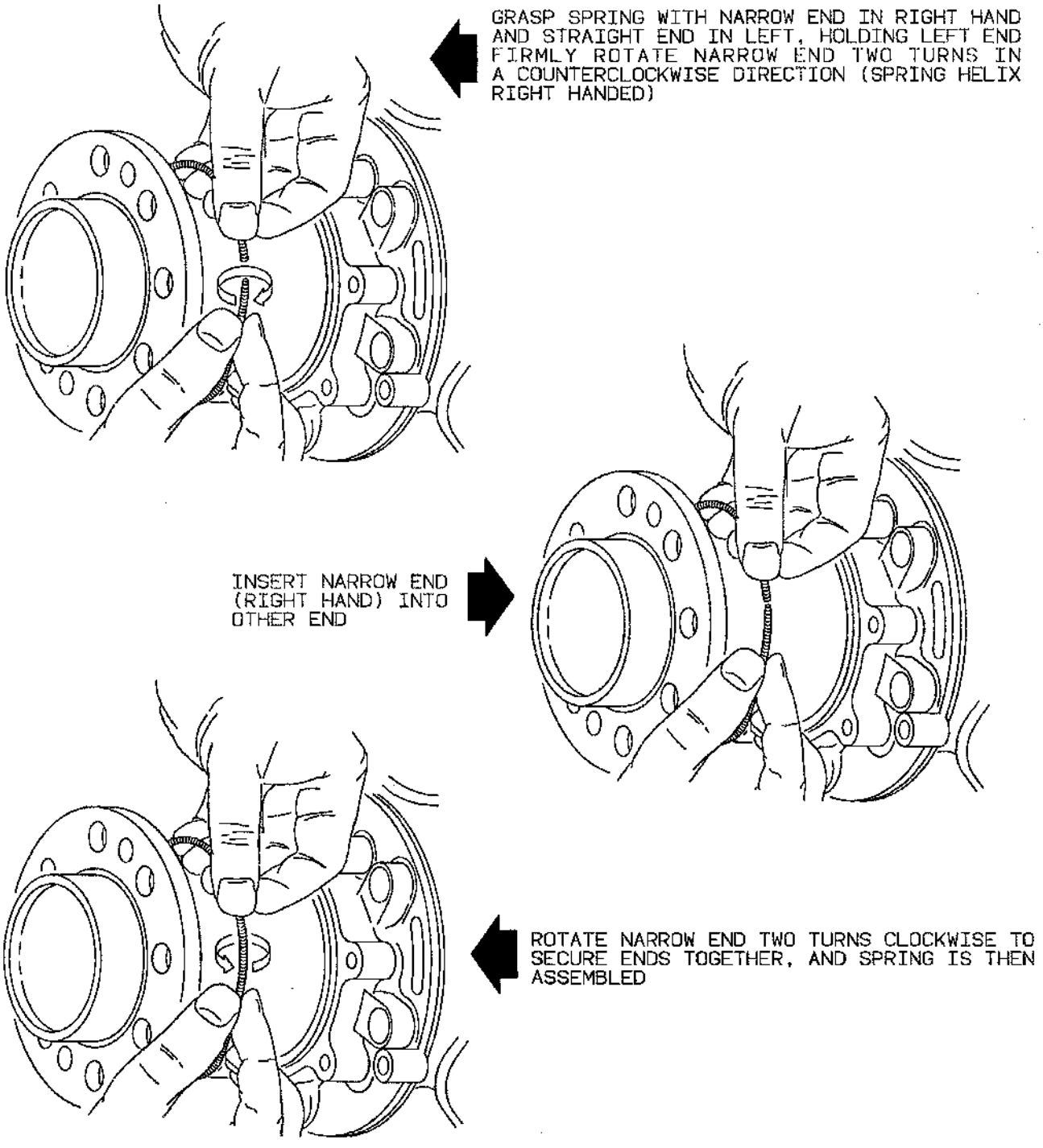
n. Install the upper and lower aft cowling as outlined in Chapter 71-10-00.

PROPELLER INSPECTION (Effectivity: All) (Figures 3 and 4)

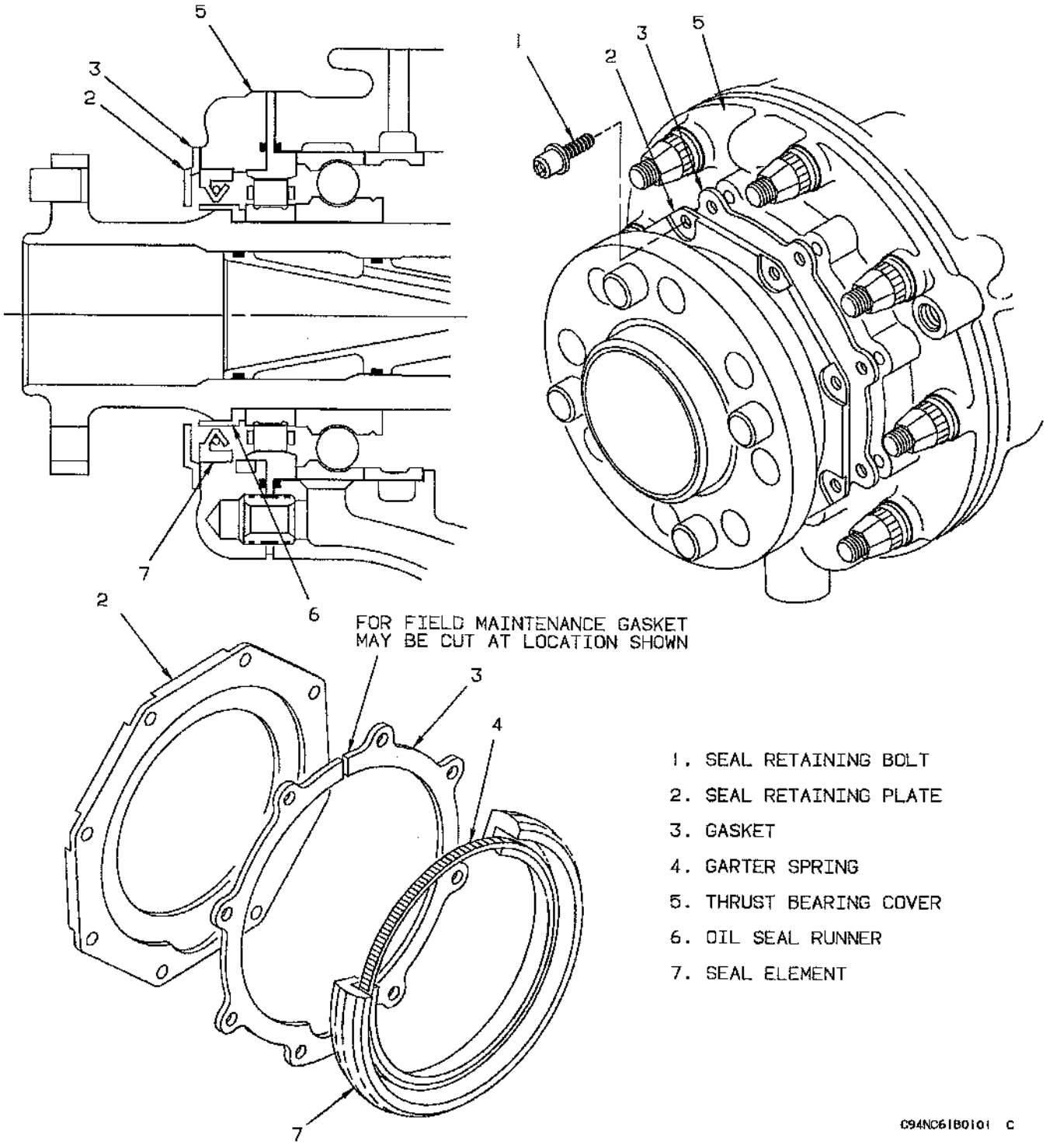
Check the propellers for condition and attachment and for condition of the beta feedback collar, beta rods and springs. Check the propeller reversing lever and linkage for evidence of binding and security of attachment. Check the beta valve and plumbing for security of attachment and leakage as specified in Chapter 5-20-00.

PROPELLER DYNAMIC BALANCING (Effectivity: All) (Figure 7)

The turbine engine is lighter in weight and is generally more free of the vibration levels commonly associated with the reciprocating engine, consequently, propeller imbalance has a tendancy to be more readily detected or noticed on a turboprop engine. Small differences in mass distribution in blades, blade twist or their settings in the hub can result in aerodynamic and mass imbalance that will not be detected or corrected by static balancing. To help ensure trouble-free and vibration-free operation from the propellers, all propellers installed at the factory have been dynamically balanced prior to delivery of the airplane.



Garter Spring Installation (Effectivity: All)
Figure 5



Propeller Shaft Oil Seal Replacement (Effectivity: All)
Figure 6

CAUTION

Dynamic balancing of the propellers consists of balancing each propeller with its respective engine. A dynamically balanced propeller must be rebalanced if it is to be reinstalled on any engine other than the one to which it was initally balanced. Dynamic balancing of a propeller is required anytime maintenance is performed that could affect balance of the propeller and its respective engine.

Dynamic balancing may be accomplished using any one of the many commercially available balance analyzers. When balancing the propellers, strict adherence to the operating instructions furnished with the equipment is required. Dynamic balancing of the propellers must be accomplished ONLY by personel who are fully qualified to operate the service equipment. The propellers are dynamically balanced when installed and should be rechecked at every 600 hours of operation. Dynamically balance the props to 0.1 ips (inches per second) or less (0.2 ips. maximum).

NOTE

When balancing the propeller, do not move any of the static balance weights to adjust the dynamic balance.

Airplanes NC-35 and after are hard wired at the factory for field connection of the Chadwick-Helmuth Model 8500 Balancer (2, Chart 1, Chapter 61-00-00). Airplanes NC-4 thru NC-34 are not prewired at the factory and will require additional set-up to perform the balancing procedure.

TEST PREPARATION (Effectivity: All)

NOTE

The following procedure was written for use with the Dynamic Balancer (2, Chart 1, 60-00-00).

a. Remove the upper engine cowling nearest the propeller.

b. If not previously installed, install the vibration sensor brackets on the lower stud of the tach generator nearest the propeller. Index mark the spinner and the spinner bulkhead.

c. Install the vibration sensors on the brackets. Ensure that the vibration sensor is vertical with respect to the plane of the propeller.

NOTE

Proceed to step e for airplanes NC-35 and after.

d. On airplanes NC-4 thru NC-34 connect the vibration sensor wires to the sensor and route it along the fuselage through the emergency exit door. Secure the vibration sensor wires to the nacelle, wing and fuselage with tape to hold them stationary during the balancing operation.

e. Gain access to the synchrophaser control box by removing the aft step panel as shown in Chapter 53-30-00.

f. Disconnect the electrical connector from the synchrophaser control box and plug it into the adapter cable (McCauley P/N 10390).

g. Plug the adapter cable leads and the vibration sensor leads into the junction box (McCauley P/N 9110) as follows: LH engine wiring to channel 1 and the RH engine wiring to channel 2. Select position 1 on the junction box and turn the synchrophaser power OFF.

CAUTION

Do not run engines with the propeller spinners off or damage to the oil cooler exit duct may occur when the propeller blade closeouts bend as a result of centrifugal force.

h. Refer to the BEECHCRAFT Starship 1 Airplane Flight Manual and start the engines. Taxi the airplane to the run-up pad and position the airplane so it is headed into the wind. Shut down the RH engine and begin the TEST PROCEDURE.

TEST PROCEDURE (Effectivity: All)
(Figure 7)

GROUND CHECK (Effectivity: All)

a. Turn ON the synchrophaser power switch and verify that the dynamic balancer performs its SELF TEST function. Increase speed of the LH propeller to cruise rpm at minimum torque. Press the BALANCE button.

b. After propeller speed has stabilized at cruise rpm, press the START button. When the measurement is complete, make a note of the vibration level and phase angle. Press the RE DO button and, when the measurement is complete, note the vibration level and

phase angle. Repeat this sequence as necessary until the technician is confident of reliable results. Press PRINT and the dynamic balancer will print a hard copy of the results from the last reading.

c. Increase engine power to approximately 80% torque at the same rpm as before. Repeat step b and compare the readings taken at both power settings. If they differ more than 0.2 ips, the propeller should be tracked or sent to the propeller shop for a blade angle check and/or static balance. Retard the power levers to IDLE.

d. Start the RH engine as outlined by the BEECHCRAFT Starship 1 Airplane Flight Manual. Shut down the LH engine.

e. Select position 2 on the junction box. Increase speed of the RH propeller to cruise rpm, minimum torque. Perform the tests in steps b and c on the RH propeller. When the tests are complete, shut down the RH engine. If the vibration levels are 0.1 ips or less, proceed to step a under FLIGHT TEST. If the vibration levels are above 0.1 ips, proceed to the next step.

f. Compute the weight amount and location using the balance charts. Attach the weights as computed per the propeller manufacturer's service instructions.

g. Perform a normal start of the LH engine as outlined by the BEECHCRAFT Starship 1 Airplane Flight Manual. Repeat steps a through f until propeller balance is within acceptable limits.

FLIGHT TEST (Effectivity: All)

a. Establish the following flight conditions:

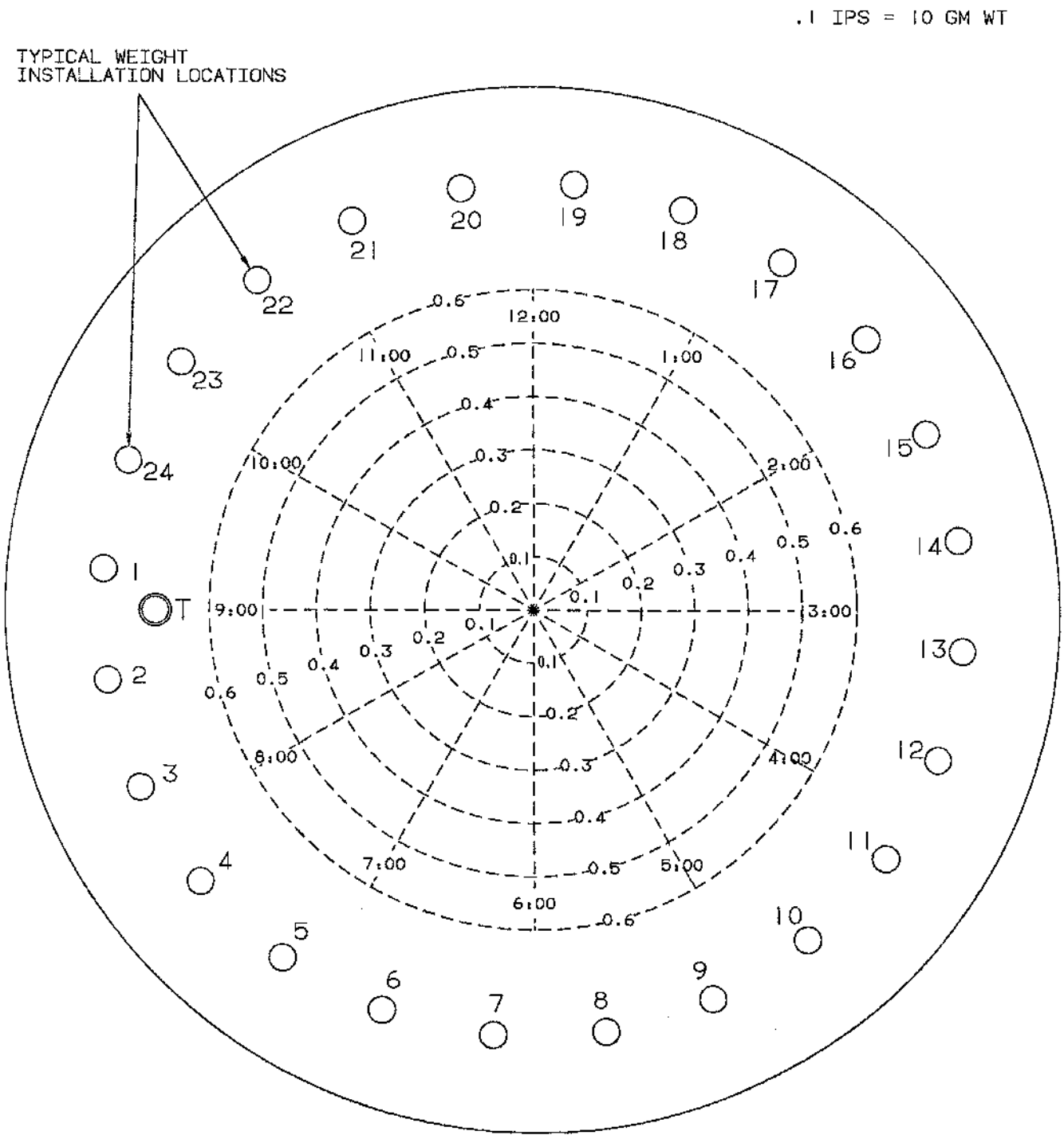
1. ALTITUDE - 16,000 ft.; CRUISE POWER; CRUISE AIRSPEED.

2. Perform a normal shutdown of the RH engine as outlined by the BEECHCRAFT Starship 1 Airplane Flight Manual.

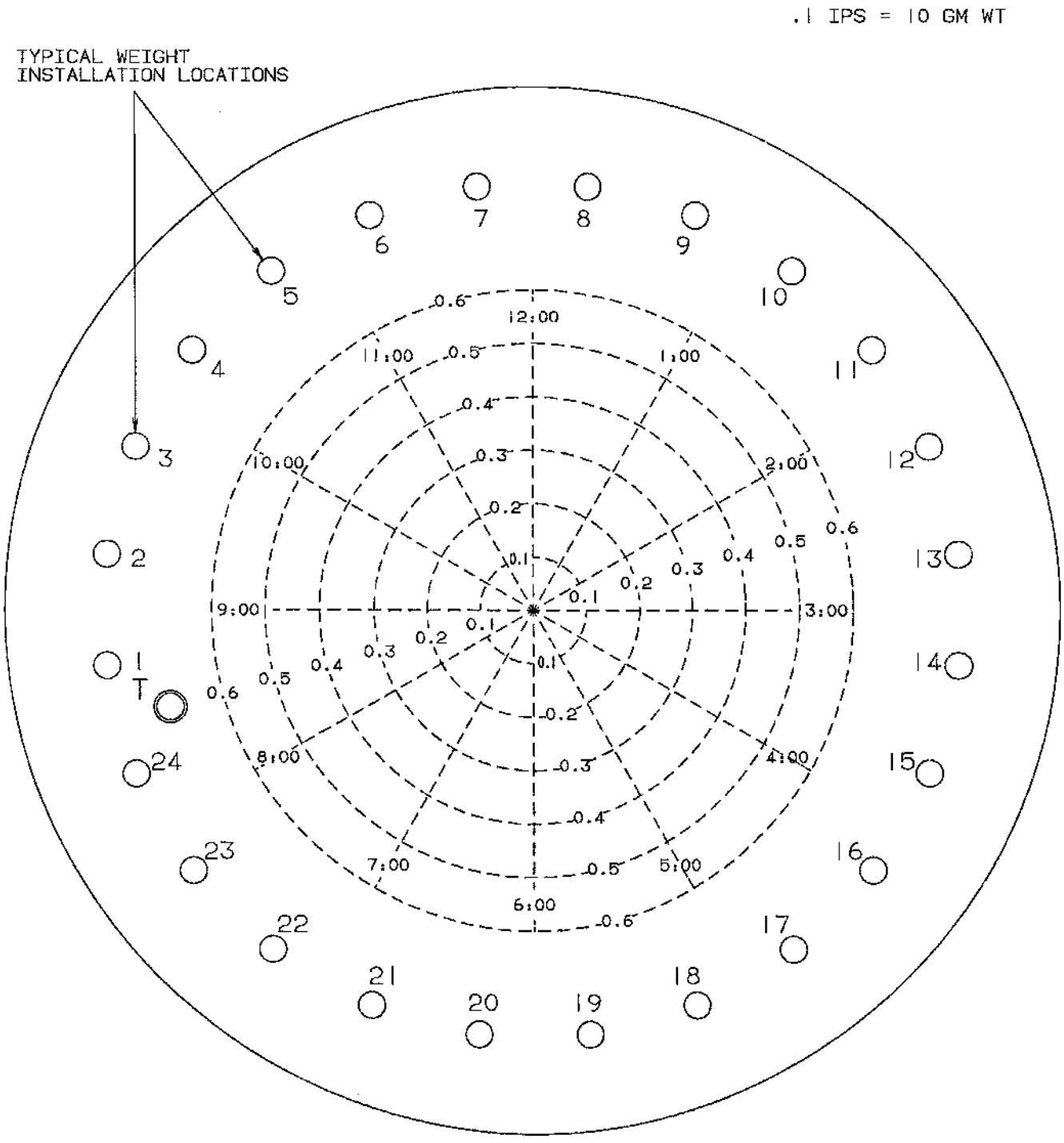
b. When the propeller speed has stabilized at cruise rpm, press the start button. When the measurement is complete, make a note of the vibration level and phase angle. Press the RE DO button and when the measurement is complete, note the vibration level and phase angle. Repeat the above sequence until confident of the results. Press PRINT and the dynamic balancer will print the results of the last reading.

c. Restart the RH engine and when cruise power has been established, perform a shutdown of the LH engine. Repeat step b on the RH engine.

d. Make the necessary balance corrections on the ground per step f of the GROUND TEST. If the vibration levels are 0.1 ips or less, no corrections are needed and the test is complete.



Balance Chart (Effectivity: All)
(Sheet 1 of 2)
Figure 7



Balance Chart (Effectivity: All)
(Sheet 2 of 2)
Figure 7

CONTROLLING - DESCRIPTION AND OPERATION (EFFECTIVITY: ALL)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

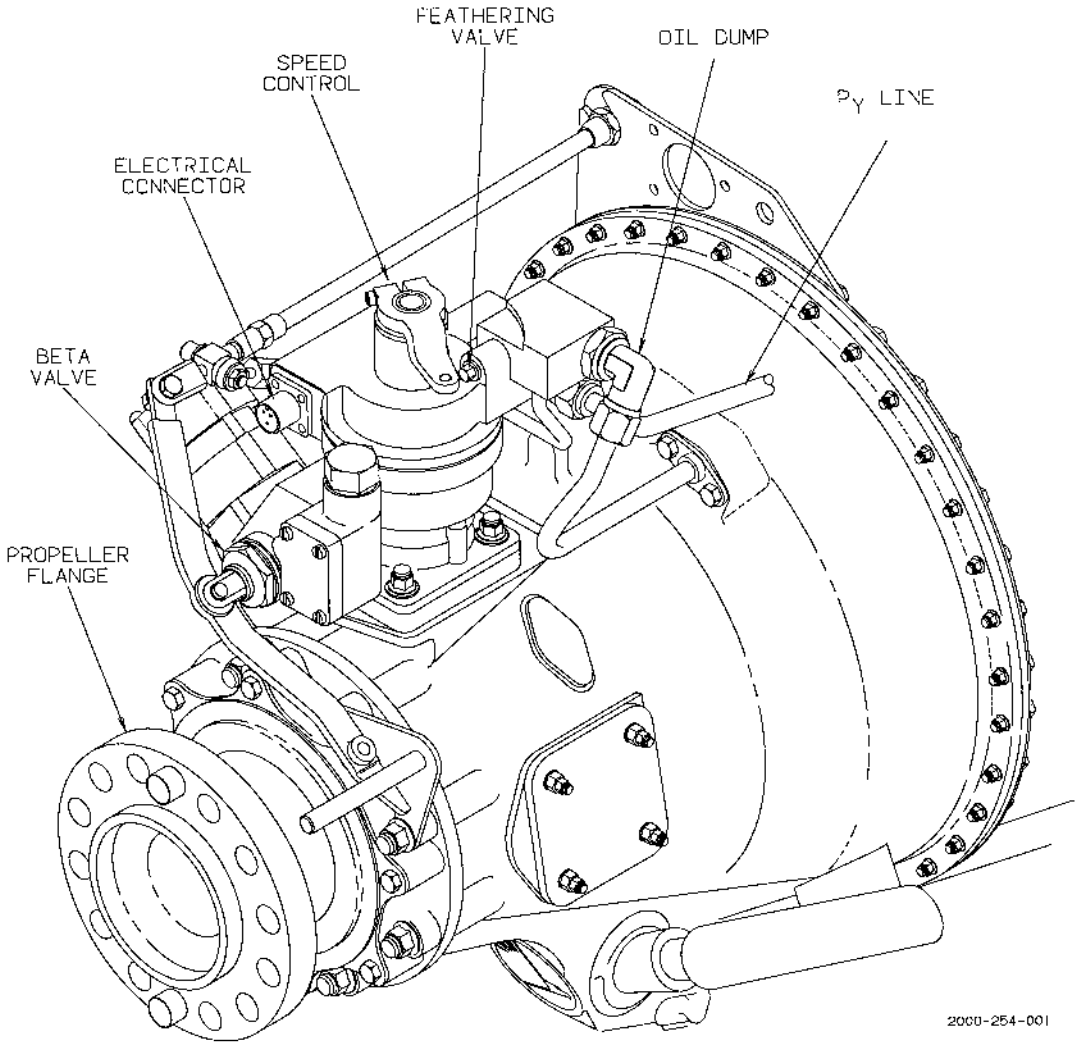
PRIMARY GOVERNOR (EFFECTIVITY: ALL) (FIGURES 1 AND 2)

The single acting governor passes engine oil to the propeller, through the oil transfer housing and the hollow center of the propeller shaft, to move the propeller blades towards the low pitch (high rpm) hydraulic stop and reverse positions. The primary governor, mounted on top of the gear reduction housing, controls the power turbine (N_2) speed by varying the propeller blade pitch. The governor consists of a gear-type oil pump with a pressure relief valve, a pair of of pivoting flyweights mounted on a rotating head and a spring-loaded pilot valve that regulates the flow of oil to and from the propeller servo piston.

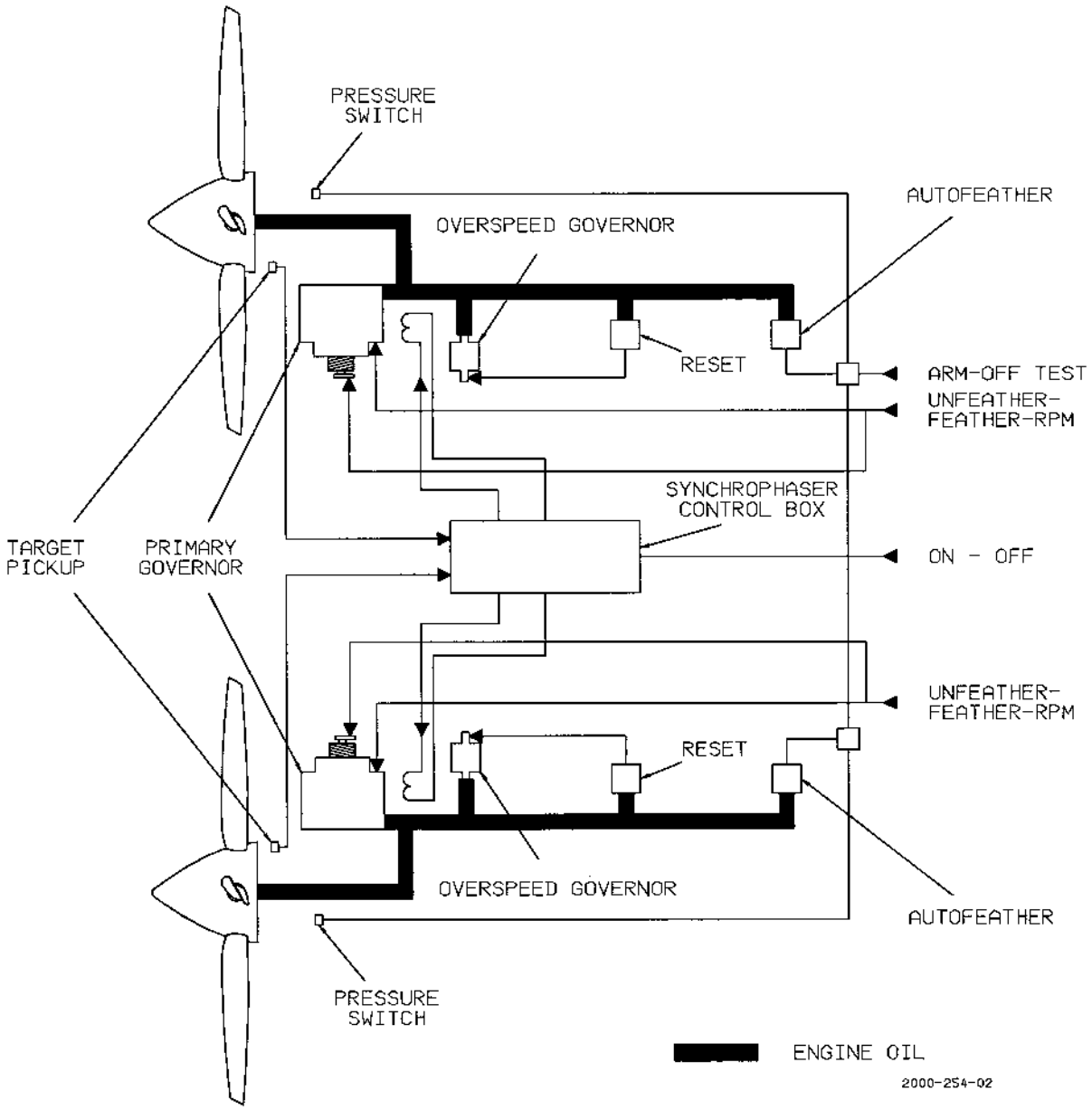
The position of the pilot valve is controlled by the combination of rotating flyweights (engine speed), the spring load imposed by the externally mounted speed control lever through the central cable from the propeller control lever in the crew compartment, and the synchrophaser magnetic force (when the synchrophaser is on). When the engine rpm drops to an underspeed condition below the control setting, speeder-spring force overcomes flyweight force to lower the pilot valve plunger and open the port in the governor drive gear shaft, through which oil flows to the propeller servo piston and decreases blade angle. The decrease in pitch reduces the load on the engine. The resultant increase in engine rpm also increases the centrifugal force of the rotating flyweights, which then lift the pilot valve plunger to cover the port in the governor drive gear shaft and shut off the flow of oil to the propeller.

The forces exerted on the pilot valve plunger by the flyweights and speeder spring then balance, causing a state of equilibrium to exist between the spring and flyweights (on-speed condition). An overspeed condition occurs with a decrease in propeller load or with movement of the propeller control to decrease rpm. Flyweight force then overcomes speeder-spring force and raises the pilot valve plunger to open the port through which oil drains from the propeller through the governor to the sump. The load on the engine increases and rpm drops as the counterweights and feathering spring increase propeller pitch. The pilot valve then centers in the governor drive gear shaft to block the flow of oil to and from the propeller as the governor flyweight force and speeder-spring force reach a state of equilibrium.

The primary governor contains a fuel section designed to protect the engine against overspeeding in the event of a malfunction, such as sticking propeller blades. During reverse thrust operation, the fuel section is set below the speed selected by the governor to permit control of propeller speed by the FCU (fuel control unit) servo system. As speed is increased, a pneumatic orifice is opened by a yoked bellcrank, operated off the flyweight head, to reduce the fuel metered to the engine. The speed of governor fuel section operation is determined by the speed selected by the governor, and the position of the reset lever, which is normally set so that the fuel section fully regulates power turbine (N_2) speed at approximately 6 percent higher than the setting of the governor, and at 4 percent lower than the governor setting in the reset (reverse) position. The reset lever causes the yoke to contact the pilot valve. The resultant spring load, in addition to the speeder-spring force, must be overcome by the flyweights in order for them to control propeller speed. This effect increases governed speed by about one percent over that normally selected.



Governor Nomenclature (Effectivity: All)
Figure 1



Propeller Governor System (Effectivity: All)
Figure 2

FLIGHT IDLE LOW PITCH STOP (EFFECTIVITY: ALL)

The flight idle low pitch stop operates in the non-energized position through the solenoid clevis. The propeller servo piston is connected by five spring-loaded sliding rods to the low pitch stop collar mounted forward of the propeller. A carbon brush block riding on the slip ring transfers the movement of the slip ring through the propeller reversing lever to the beta valve of the governor. The initial aft motion of the beta valve blocks off the flow of oil to the propeller. Further motion aft dumps the oil from the propeller into the reduction gearbox sump. A mechanical stop limits the rearward movement of the beta valve. Forward movement of the beta valve does not effect normal propeller control. When the propeller is rotating at a speed lower than the speed selected on the governor, the governor pump provides oil pressure to the servo piston and decreases the pitch of the propeller blades until the feedback of motion from the low pitch stop collar pulls the beta valve into a position blocking the supply of oil to the propeller, thus preventing further pitch changes.

GROUND IDLE STOP SYSTEM (EFFECTIVITY: ALL)

The ground idle stop system uses an electrical solenoid which is mounted at the rear of the propeller reversing push/pull cable. It is designed to operate on the ground only by resetting the propeller blades to maintain an angle of approximately 4° below the flight idle condition. The solenoid is connected to the propeller reversing lever by means of a slotted clevis which allows the reversing lever to be pulled forward, resetting the beta valve.

The electrical solenoid is wired through the LH landing gear squat switch. As the airplane lands, the squat switch activates the solenoid, which in turn pulls the reverse lever forward to reset the beta valve. This enables the propeller blades to reset to a lower blade angle. The travel on the reverse lever is limited by the slot in the clevis. While the airplane is on the ground, the system remains activated. When the squat switch deactivates the solenoid, the spring force from the beta valve pushes the reverse lever forward to its original position.

A propeller low pitch stop switch is provided in the pedestal to ensure operation of the ground idle stop system under low gross weight operation of the airplane. The switch is activated by a linkage to the throttle levers. Lifting either throttle lever to the beta range will cause the throttle lever detent pin to contact the linkage to actuate the low pitch stop switch. Returning both throttle levers to the idle position will deactivate the switch.

For rigging and adjustment of the ground idle stop system, refer to Chapter 76-00-00.

PROPELLER REVERSING SYSTEM (EFFECTIVITY: ALL)

Propeller reversing is accomplished by moving the propeller controls into the reverse range. This enables the propeller servo piston to reverse the pitch of the propeller blades with engine oil pressure from the governor sump. The power lever controls are connected to the engine control beta cam box, which is also linked to the fuel control unit. A push-pull cable extends from the beta cam box aft to the reversing lever which is attached to the brush that rides in the propeller brass or low pitch stop collar. Movement of the power control lever controls is transmitted through the beta cam box and interconnecting linkage to the fuel control unit. Prop control levers that are connected to the governors regulate propeller speed and pitch.

OVERSPEED GOVERNOR (EFFECTIVITY: ALL)

A propeller overspeed governor, mounted on the RH side of the reduction gear housing, acts as a safeguard against propeller overspeed should the primary governor fail. The overspeed governor regulates the flow of oil to the propeller pitch-change mechanism by means of a flyweight and speeder spring arrangement similar to that of the primary governor. The overspeed unit governs at 104% N_2 speed. Since it has no mechanical controls, the overspeed governor is equipped with a testing solenoid that resets the normal overspeed setting to approximately 1565 propeller rpm for ground testing.

CONTROLLING - MAINTENANCE PRACTICES (EFFECTIVITY: ALL)

PRIMARY GOVERNOR REMOVAL (EFFECTIVITY: ALL)
(FIGURE 3)

NOTE

The power turbine governor is an integral part of the primary governor.

a. Remove the upper aft engine cowling as outlined in Chapter 71-10-00.

b. Remove the cotter pin, castellated nut, washers and bolt securing the primary governor control lever to the propeller control cable rod end.

c. Remove the cotter pin, castellated nut, washer and bolt securing the fuel governor interconnecting rod to the fuel governor control lever.

d. Remove the cotter pin, castellated nut, washer, bolt and spacer that secure the clevis end to the propeller reversing lever.

e. Remove the cotter pin, washer and clevis pin securing the propeller reversing lever to the beta valve.

f. Remove the safety wire and disconnect the propeller synchrophaser electrical plug from the solenoid on the governor.

g. Remove the safety wire and disconnect the Py line and oil dump line from the governor. Cap or plug any open lines and ports.

h. Remove the four nuts and washers that anchor the governor to the mounting pad on the reduction gearbox case.

i. Remove the governor and governor mounting pad gasket.

PRIMARY GOVERNOR INSTALLATION (EFFECTIVITY: ALL)
(FIGURE 3)

NOTE

Refer to Chapter 70-06-00, Standard Practices, of the Pratt & Whitney PT6A-67A Engine Maintenance Manual for procedures on linkage assembly and adjustments to the linkage assembly.

a. Install a new gasket over the four studs on the governor mounting pad.

b. Lightly coat the splined shaft of the governor with lubricating grease (3, Chart 2, 61-00-00).

c. Position the governor on the mounting pad and secure it in place with the four attaching washers and nuts. Torque the nuts to 170-190 inch-pounds.

d. Secure the propeller reversing lever to the beta valve with the attaching bushing, clevis pin, washer and cotter pin.

e. Secure the clevis end to the propeller reversing lever with the attaching spacer, bolt, washers, castellated nut and cotter pin.

f. Secure the fuel governor interconnecting rod to the fuel governor control lever with the attaching bolt, washer, castellated nut and cotter pin.

g. Secure the primary governor control lever to the propeller control cable rod end with the attaching bolt, washers, castellated nut and cotter pin.

h. Connect the Py and oil dump lines to the governor and safety wire.

i. Connect the propeller synchrophaser electrical plug to the solenoid and safety wire.

j. Perform the following checks before returning the airplane to service:

1. Overspeed Governor Check (found in this sub-chapter).

2. Feathering Operation Check.

a) Set the power lever to 65% N_1.

b) Feather Propellers.

c) N_1 should not change more than .3%.

3. Underspeed Fuel Governing Check.

a) Disconnect and hold the propeller governor reset lever against the MINIMUM stop.

b) Set the propeller control lever to HIGH RPM.

c) Accelerate engine slowly until Np (propeller speed) is 1615±17 rpm.

d) Connect the propeller governor reset lever and set the lost motion link free play at .100 inch.

4. Maximum Propeller Speed Check.

a) Set the propeller control to obtain 100% Np (a propeller speed of 1700 rpm).

b) Adjust the maximum speed stop screw as necessary (reference the Pratt & Whitney PT6A-67A Engine Maintenance Manual).

5. Interference Check between the Normal Governing and Overspeed Fuel Governing Functions of the Propeller Governor.

a) Ensure propeller governor reset lever is contacting the MAXIMUM stop and that the lost motion link has .100 inch of free play.

b) Set the propeller control lever to HIGH RPM.

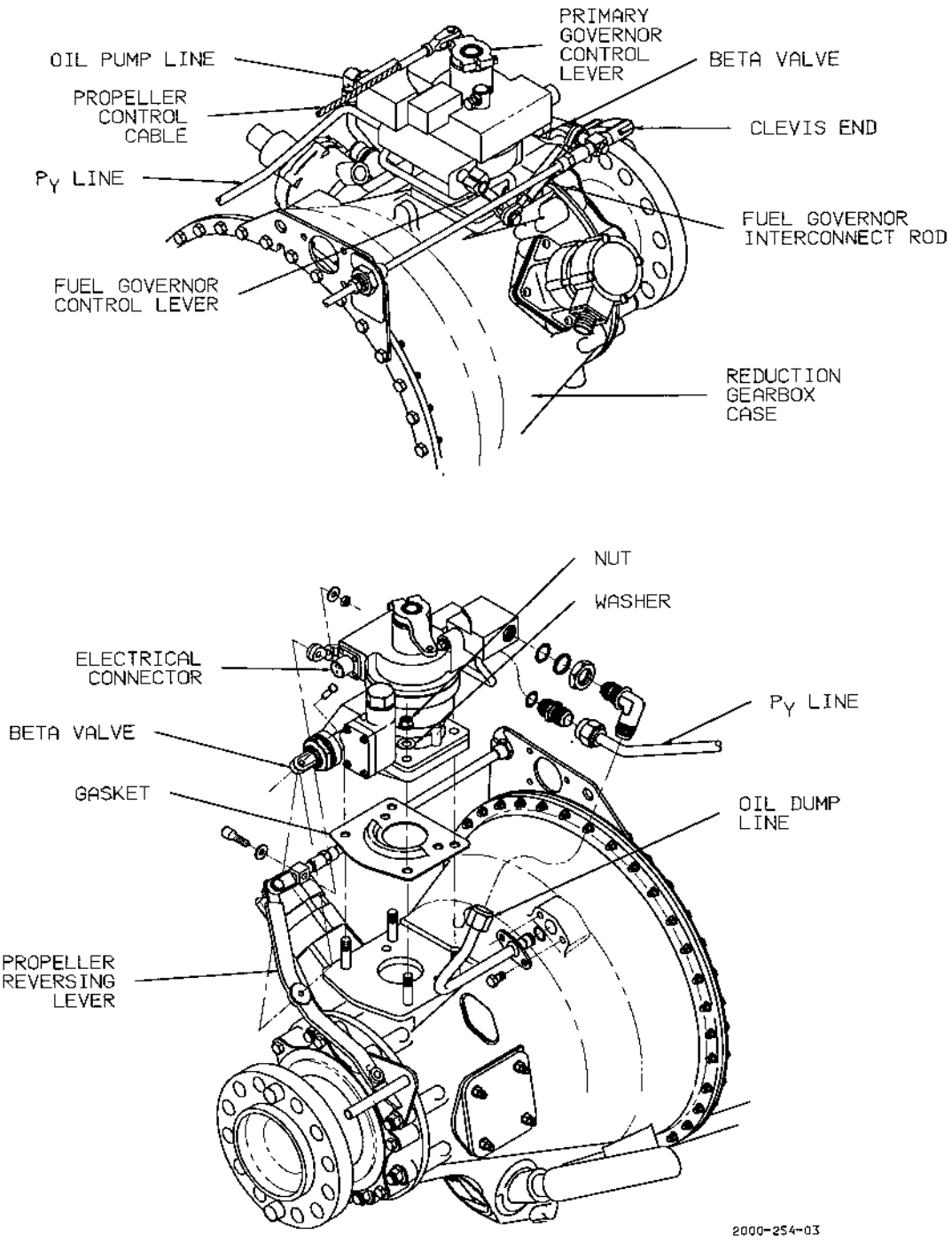
c) Set the power control to obtain 74% torque and observe N_1.

d) Activate the overspeed governor test solenoid to obtain 95% Np (1615 propeller rpm) by holding the propeller test switch in the OVERSPEED GOV position.

e) Check that the propeller governor reset lever is still contacting the MAXIMUM stop.

f) An increase in N_1 speed of more than .4% is unacceptable. A decrease in N_1 speed is acceptable.

k. Reinstall the upper aft engine cowling as outlined in Chapter 71-10-00.



Propeller Governor Installation (Effectivity: All)
Figure 3

FEATHERING ADJUSTMENT (EFFECTIVITY: ALL) (FIGURE 4)

Position the propeller lever in the full HIGH RPM position and advance the power lever to obtain a propeller speed of 1700 rpm. Pull the propeller lever back against the detent, then adjust the propeller governor control arm until there is a .20-inch gap between the feathering valve adjustment bolt and the governor stop plate (reference Chapter 76-00-00). Pull the propeller lever past the detent (feather position) and observe that the propeller begins to feather.

PROPELLER OVERSPEED GOVERNOR REMOVAL (EFFECTIVITY: ALL) (FIGURE 5)

a. Remove the upper aft engine cowling as outlined in Chapter 71-10-00.

b. Remove the safety wire and disconnect the electrical connectors (5 and 6) from the governor solenoids.

c. Remove the four self-locking nuts (4) and plain washers (3) securing the governor (2) and remove the governor from the RH side of the reduction gear housing.

PROPELLER OVERSPEED GOVERNOR INSTALLATION (EFFECTIVITY: ALL) (FIGURE 5)

a. Install a new gasket (1) on the mounting pad.

b. Apply lubricating grease (3, Chart 2, 61-00-00) sparingly to the governor splined drive.

c. Position the governor (2) on the mounting pad and install the four plain washers (3) and self-locking nuts (4). Apply a torque of 170 to 190 inch-pounds to the mounting nuts.

d. Connect the electrical connectors (5 and 6) to the governor (2). The solenoid with the receptacle on top is the speed reset solenoid used to test governor operation and the remaining solenoid is used for autofeathering. Safety wire the plugs.

e. Reinstall the upper aft engine cowling as outlined in Chapter 71-10-00.

OVERSPEED GOVERNOR CHECK (EFFECTIVITY: ALL)

The overspeed governor unit, mounted on the left side of the reduction gear housing, is preset at the factory to govern at 104% N_2 speed (approximately 1768 propeller rpm) and should not normally require readjusting. This setting can be reduced to 1565 ±45 rpm for testing purposes by means of a testing solenoid valve. A second solenoid valve attached to the governor provides a means of immediately dumping oil from the unit for autofeathering. Perform the overspeed governor check as follows:

a. Move the propeller levers full forward.

b. Advance the power levers until the propeller speed is increased to 1500 rpm.

CAUTION

Be extremely careful. DO NOT exceed the engine torque limits during this test.

c. Hold the propeller test switch (located on the left subpanel) in the OVERSPEED GOV position.

d. Advance the power levers until the rpm stabilizes. A correctly operating overspeed governor will hold a stable

propeller speed of 1520 to 1610 rpm.

NOTE

Overspeed units that fail to govern at the prescribed settings should be replaced and returned to a qualified service station for repair.

e. Set the power lever in the idle position, then release the propeller test switch.

GROUND IDLE STOP CHECK (EFFECTIVITY: ALL)
(FIGURE 6)

a. Place the condition levers to high idle and the propeller control levers to the full forward position. Note the stabilized ground idle RPM of each propeller.

b. Advance the power levers forward untill each propeller RPM reaches 1500 RPM and stabilizes.

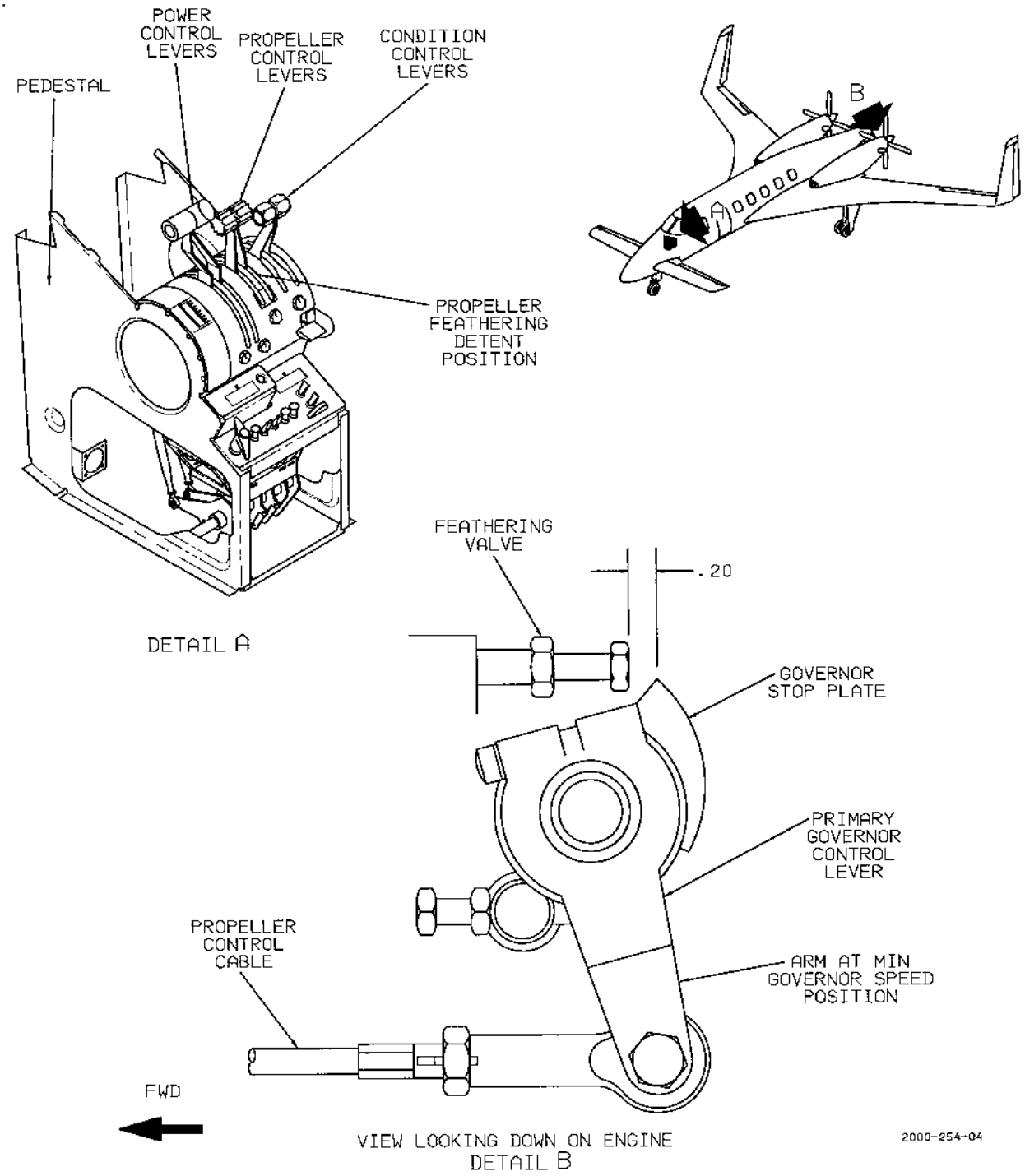
c. Hold the GRD IDLE STOP switch in the test position.

d. Verify that each propeller RPM decreases to 1100 to 1200 RPM and stabilizes.

e. Release the GRD IDLE STOP switch and confirm that each propeller returns to 1500 RPM and stabilizes.

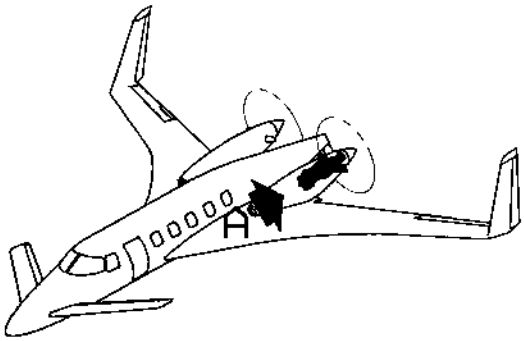
f. If the decrease in RPM is not within limits:

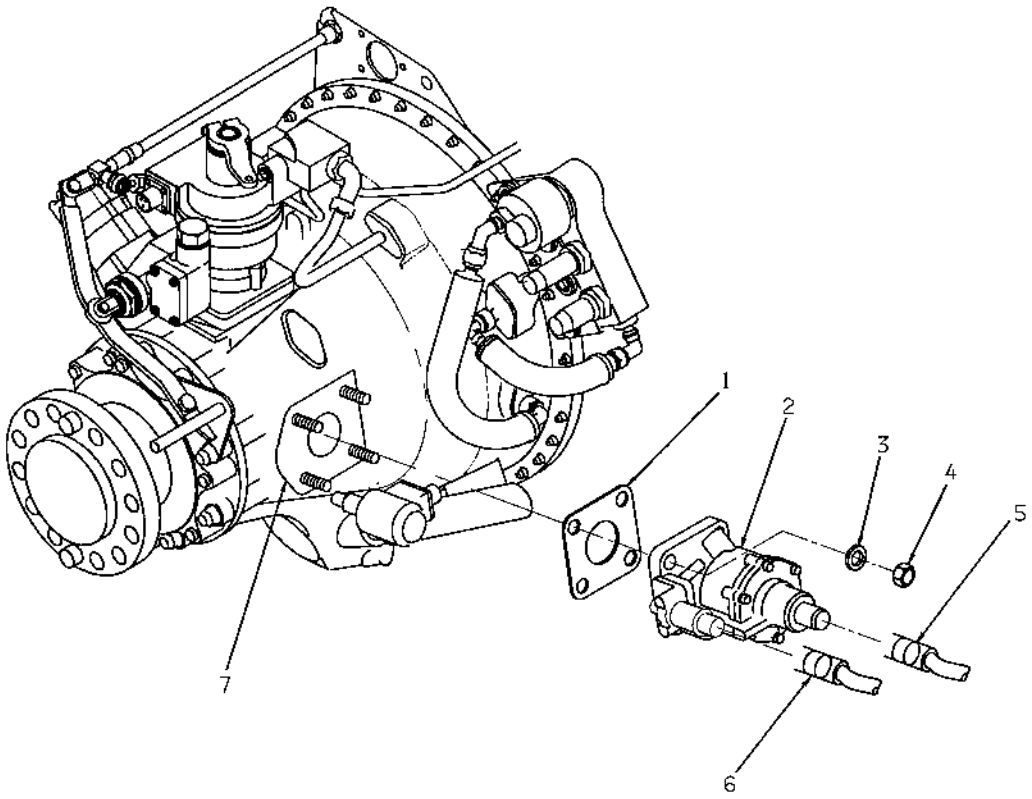
1. Loosen the lower screws on the solenoid bracket. Refer to Figure 6.
2. Move the solenoid aft to increase RPM drop. Move the solenoid forward to decrease RPM drop.
3. Tighten the lower screws on the solenoid bracket.
4. Repeat the solenoid check procedure to verify correct operation.



Feathering Detents (Effectivity: All)
Figure 4

1. GASKET
2. OVERSPEED GOVERNOR
3. WASHER
4. NUT
5. ELECTRICAL CONNECTOR
6. ELECTRICAL CONNECTOR
7. STUDS





DETAIL A

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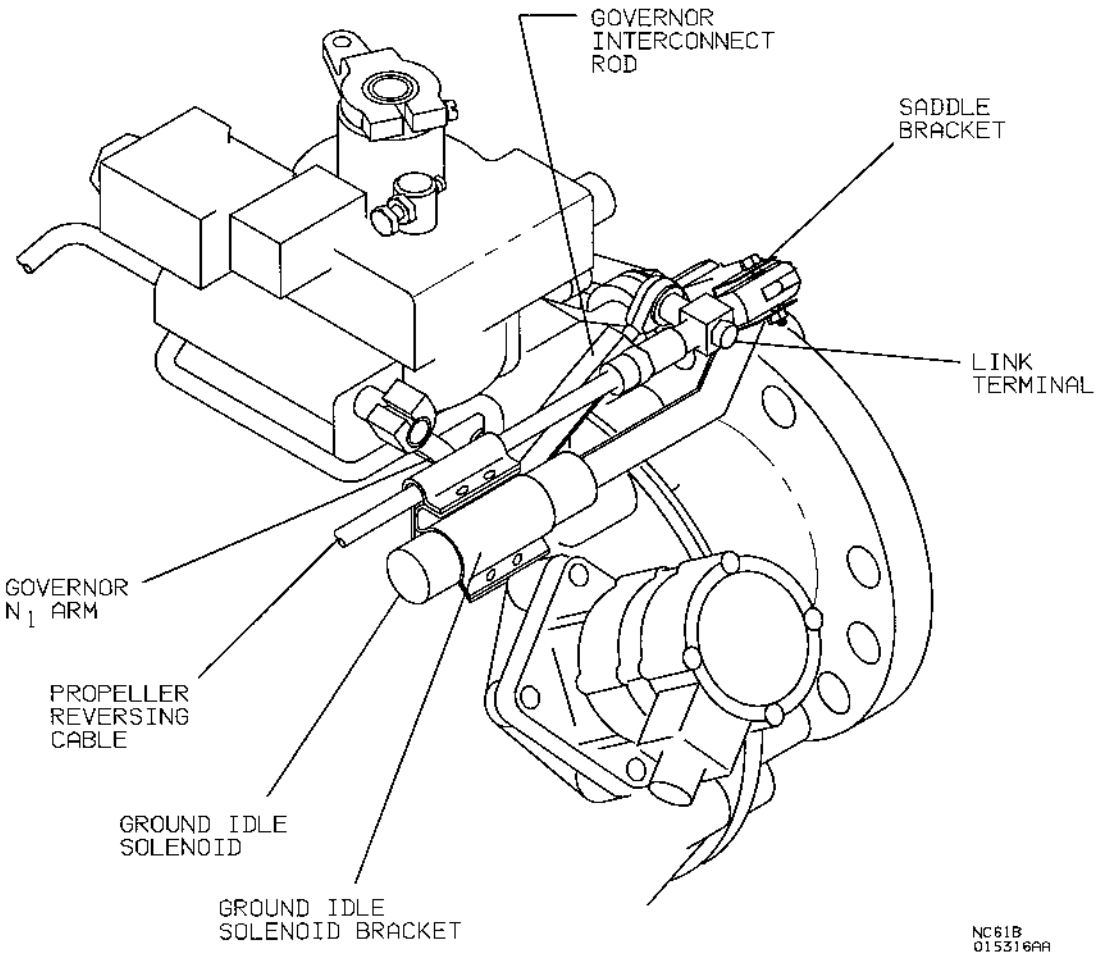
Propeller Overspeed Governor Installation (Effectivity: All)
Figure 5



Ground Idle Solenoid (Effectivity: All)
Figure 6

AUTOFEATHERING - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

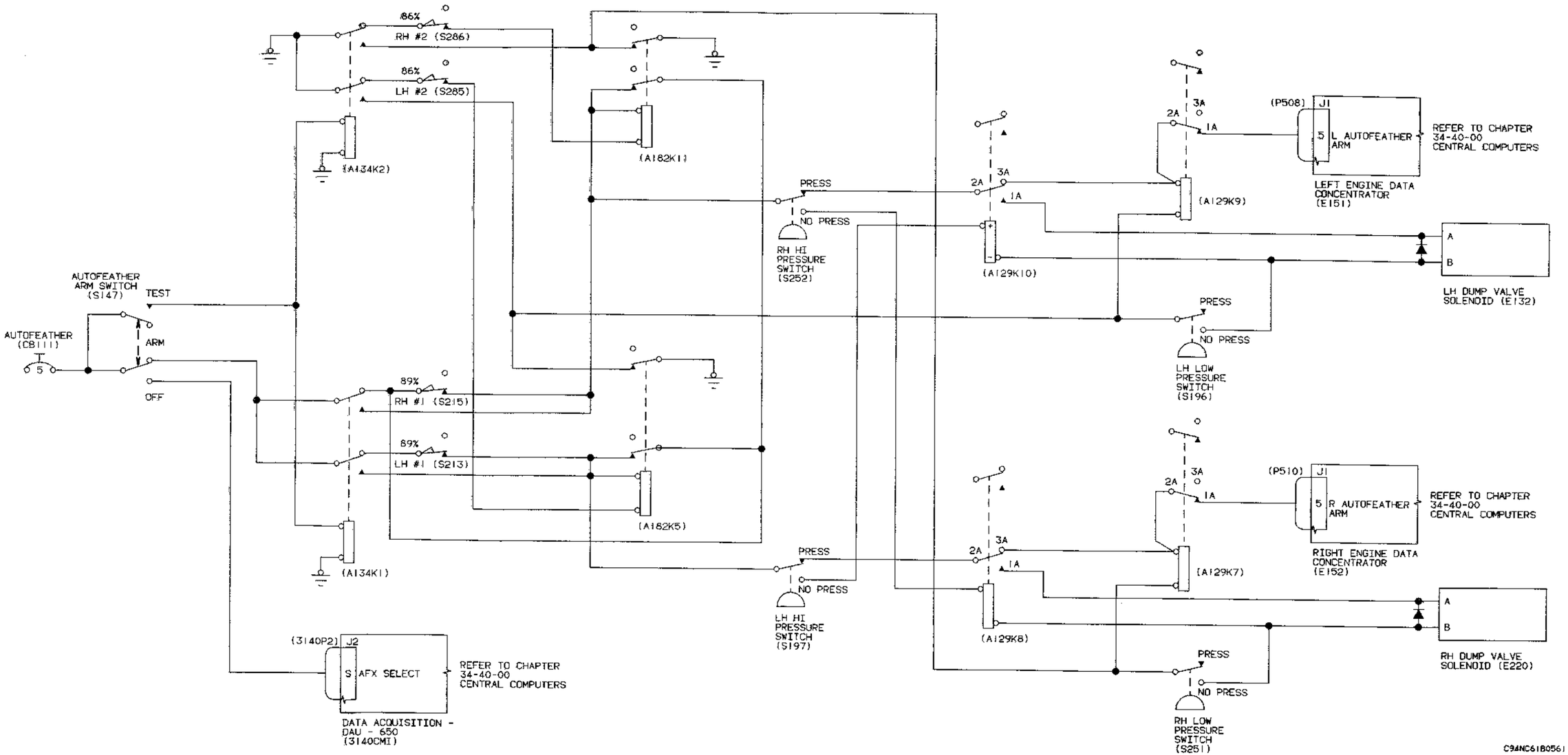
The automatic feathering system is designed to dump engine oil from the primary governor when engine torquemeter oil pressure drops below approximately 13% torque (at power settings greater than 89% to 91% N_1). When engine oil is dumped from the primary governor, the feathering springs force the propeller blades to feather. The system is intended for use during takeoff, climb, approach and landing. It should be kept in the ARM position until the airplane has gained enough altitude so that the loss of one engine and the resultant propeller drag will not present an immediate problem to the pilot.

The autofeathering system is controlled by a lever lock type of arming switch that has functional modes of ARM (the center position), OFF (the down position), and TEST (the up position). The system is activated by placing the arming switch, located in the center subpanel, in the ARM position. This closes the circuit, and power is drawn from the circuit breaker (CB111) labeled AUTO FEATHER (located in the left circuit breaker panel) to the A134 relay PCB which is located in the forward console. The autofeathering system will arm when the power levers are advanced through the 89% to 91% N_1 position. The system will not be disabled until the power levers are pulled back through the 86% to 88% N_1 position. The autofeather system will remain disabled as long as the power levers are retarded below the 85% to 87% N_1 position.

When the power levers are advanced to the 89% to 91% N_1 position the left and right 89% power lever switches are closed. This completes the circuit from CB111 (the autofeather circuit breaker) through S147 (the autofeather arm switch) through the contacts of A134K1 to energize the relays A182K1 and A182K5. With the A182 relays energized power bypasses the 89% switches to latch the relays. The A182 relays will remain energized until the power levers are retarded enough to open the 86% switches and remove the ground. With the A182 relays energized power is applied to the left and right high pressure switches (S197 and S252 respectively). Engine torquemeter oil pressure closes S197 and S252 which applies power to energize relays A129K7 and A129K9. With the A129 relays energized the autofeather system is armed and power is provided to the left and right engine data concentrators (EDCs). The EDCs provide annunciation signals to the Engine Instrument Crew Alerting System (EICAS), refer to chapter 31-40-00 for information on this system. A decrease in engine torquemeter oil pressure opens S197 and S252 which will deenergize relays A129K7 and A129K9 and apply power to relays A129K8 and A129K10. A further decrease in engine torquemeter oil pressure closes the autofeather low pressure switches S196 (LH) and S251 (RH) and applies power to energize A129K8 (RH) or A129K9 (LH) which opens the respective autofeather dump valve solenoid.

When the AUTOFEATHER switch on the pilot's center subpanel is set to the ARM position (while the power levers are advanced), an AFX message is displayed above each propeller rpm message inside the propeller speed gauges on the EICAS display. A green AFX ENABLED message will also be displayed on the EICAS display list when the autofeather system is armed. The engine must be above approximately 89% N_1 (the autofeather power lever switches must be engaged) for the autofeather system to arm.

The autofeathering system will automatically disarm propeller autofeathering on the normally functioning engine when the power (torque) on the malfunctioning engine drops to a point where propeller autofeathering on the malfunctioning engine is imminent. If the RH engine torque falls far enough to allow the RH autofeather high pressure switch to open (in the case of a low or lost power situation), the A129K9 relay will



Autofeather System Schematic (Effectivity: All)
Figure 1

de-energize and the left engine autofeather arm will be disabled. When the LH engine autofeather arm is disabled, the AFX message in the LH propeller speed gauge will extinguish. If the airplane is on the ground while the LH autofeather arm is disabled, the EICAS message list will read "AFX DISABLED" in yellow. If the airplane is airborne while the LH autofeather arm is disabled, the EICAS message will read "AFX DISABLED" in white. If the RH engine torque falls below the autofeather threshold (approximately 6% to 12% engine torque), the autofeather low pressure switch will de-energize. Current then flows through the A129K7 relay which energizes the solenoid dump valve on the propeller governor. The solenoid dump valve immediately dumps oil from the servo, reducing governor oil pressure to zero which allows the propeller feathering springs to quickly move the propeller blades toward feather. As the propeller blades feather, the AFX message in the RH propeller speed gauge will blink on and off. When the RH propeller blades have feathered, the AFX annunciator located inside the RH prop display will extinguish.

For testing purposes, the AUTOFEATHER switch is equipped with a TEST mode by which the power lever switches in the pedestal can be bypassed to check the system when the power levers are retarded below the 89% to 91% N_1 position. When the AUTOFEATHER switch is placed in the TEST position, power from CB111 is passed through S197 to energize A134K1 and A134K2. With the A134 relays energized power bypasses the power lever switches and is applied directly to the A182 relays.

AUTOFEATHER HIGH PRESSURE SWITCH (Effectivity: All)

The autofeather high pressure switch is a double-pole switch that moves to the normally closed set of contacts when the torquemeter oil pressure drops below 4.7 ±.4 psi and moves to the normally open set of contacts when the pressure rises again to 6.1 ±.5 psi. When actuated to its normally open set of contacts (torquemeter pressure above 6.1 ±.5 psi), the high pressure switch provides current from the A134 relay PCB to the engine data concentrator for the opposite side EICAS AFX message display and also to the opposite side autofeather dump solenoid valve and to A129K7 or A129K9. Should a fault cause the torquemeter pressure to drop below 4.7 ±.4 psi on either autofeather system, the high pressure switch on the faulted side will return to its normally closed position. This will remove current from the opposite side to disable that system and provide current to its own dump valve solenoid control relay A129K8 or A129K10. When the opposite side dump valve solenoid control relay is disabled, the automatic feathering of both propellers at the same time is prevented.

AUTOFEATHER LOW PRESSURE SWITCH (Effectivity: All)

The autofeather low pressure switch provides for the switching of ground to the A129 relay panel and autofeather dump valve solenoid. The low pressure switch is a single-pole pressure-actuated switch permanently set to actuate on rising pressure to its normally open contact at 4.0 ±.8 psi. On falling pressure, it will return to its normally closed contact at 3.1 ±.6 psi. In its normally closed position (torquemeter pressure below 3.1 ± .6 psi), the low pressure switch completes the ground circuit to the autofeather dump valve solenoid and to its respective A129 relay.

AUTOFEATHERING - MAINTENANCE PRACTICES (Effectivity: All)

AUTOFEATHER PRESSURE SWITCH REMOVAL (Effectivity: All) (Figure 2)

NOTE

Removal and installation of the high pressure and low pressure switches are identical. The low pressure switch is mounted adjacent to the torque pressure transmitter just aft and above the right hand exhaust outlet on each engine. The high pressure switch is mounted just below and to the right of the low pressure switch. Also reference Chapter 74-00-00 IGNITION, for autofeather high pressure switch removal and installation.

a. Remove all electrical power and disconnect the battery.

b. Remove the upper aft cowling as outlined in Chapter 71-10-00.

c. Unsafety and remove the electrical connector from the autofeather switch.

d. Unsafety and remove the mounting nut at the base of the switch and remove the switch.

e. Install a protective plug in the switch mounting hole to prevent contamination of the torquemeter chamber oil supply.

AUTOFEATHER PRESSURE SWITCH INSTALLATION (Effectivity: All) (Figure 2)

a. Remove the protective plug from the switch mounting hole and secure the switch in place with the mounting nut. Torque the mounting nut to 30 inch-pounds maximum), then lock the nut in place with safety wire.

b. Connect the electrical connector to the switch and lock it in place with safety wire.

c. Install the upper forward cowling as outlined in Chapter 71-10-00.

d. Connect the battery and perform an AUTOFEATHER OPERATIONAL CHECK.

AUTOFEATHER POWER LEVER SWITCH REMOVAL (Effectivity: All) (Figure 3)

NOTE

Removal of any of the four power lever switches is the same.

a. Remove all electrical power from the airplane and disconnect the battery.

b. Remove the cover on the LH side of the control pedestal.

c. Disconnect the electrical wiring (1) from the switch (2).

d. Unscrew the mounting screws (3) and washers (4) and remove the switch (2).

AUTOFEATHER POWER LEVER SWITCH INSTALLATION (Effectivity: All) (Figure 3)

NOTE

Installation of any of the four power lever switches is the same.

a. Place the switch (2) in position and install the mounting screws (3) through the slotted mounts.

b. Connect the electrical wiring (1) to the switch (2).

c. Adjust the switch as outlined in AUTOFEATHER POWER LEVER SWITCH ADJUSTMENT, in this section.

d. Install the cover on LH side of the control pedestal.

e. Connect the battery and perform an AUTOFEATHER OPERATIONAL CHECK.

AUTOFEATHER POWER LEVER SWITCH ADJUSTMENT (Effectivity: All) (Figure 3)

Access to the autofeather power switches is through the cover on the left side of the pedestal. The switches may be adjusted as follows:

CAUTION

Do not run engines with the propeller spinners off or damage to the oil cooler exit duct may occur when the propeller blade closeouts bend as a result of centrifugal force.

a. Start the engines as outlined in the Starship 1 AIRPLANE FLIGHT MANUAL.

b. Place the AUTOFEATHER switch in the ARM position.

c. Advance the power levers to 86% N_1.

d. Adjust the No. 2 RH and No. 2 LH autofeather power lever switches in their mounting slots so that they open when the power levers are moving down through 86 +2 -0% N_1 Tighten the switches in their mounting slots until secure.

e. Advance the power levers to 89% N_1.

f. Adjust the No. 1 RH and No. 1 LH autofeather power lever switches in their mounting slots so that they close when the power levers are moving up through 89 +2 -0% N_1. Tighten the switches in their mounting slots until secure.

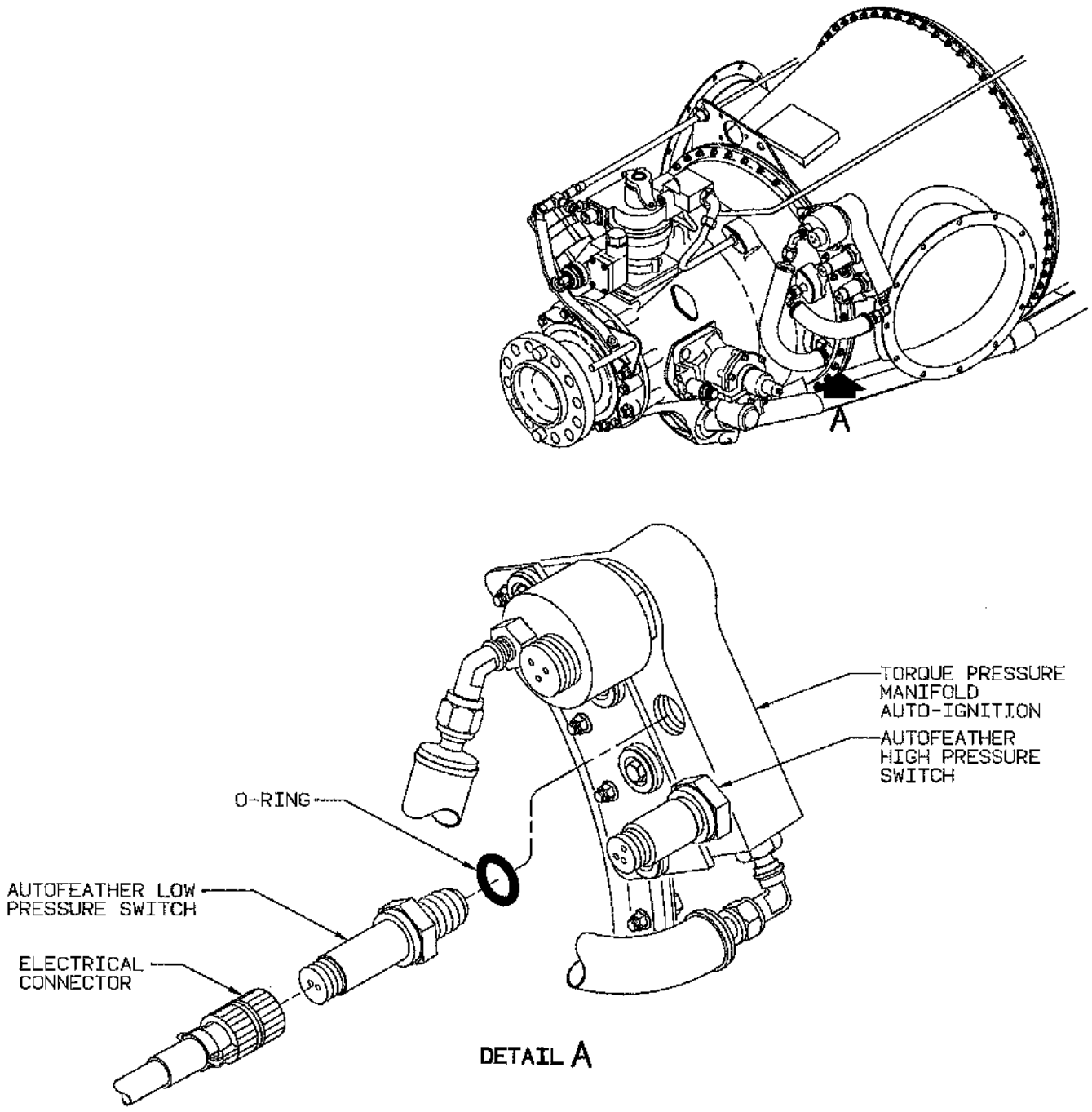
g. Shut down the engines as outlined in the Starship 1 AIRPLANE FLIGHT MANUAL.

h. Perform an AUTOFEATHER OPERATIONAL CHECK.

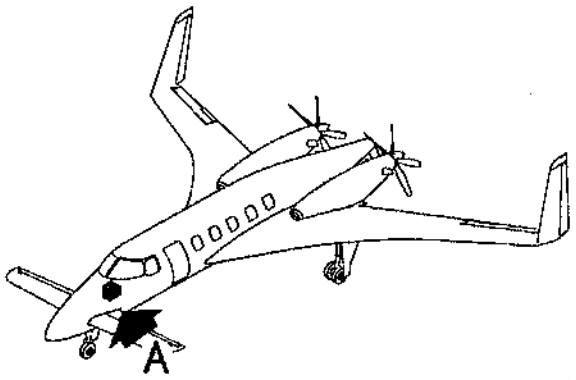
AUTOFEATHER OPERATIONAL CHECK (Effectivity: All)

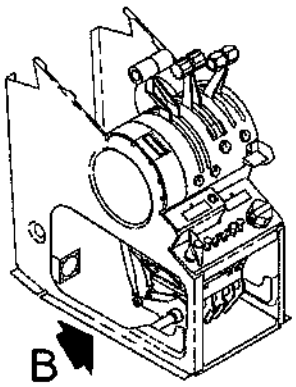
CAUTION

Do not run engines with the propeller spinners off or damage to the oil cooler exit duct may occur when the propeller blade closeouts bend as a result of centrifugal force.

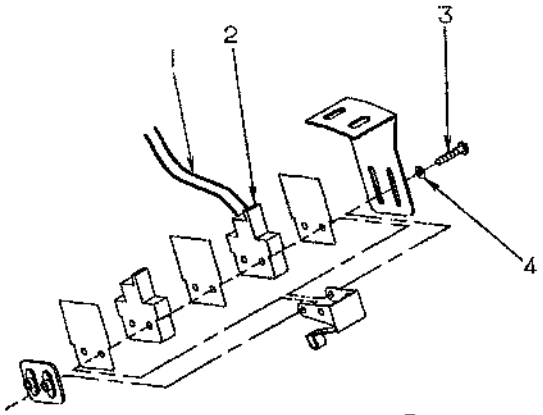


Autofeather Pressure Switches (Effectivity: All)
Figure 2





DETAIL A



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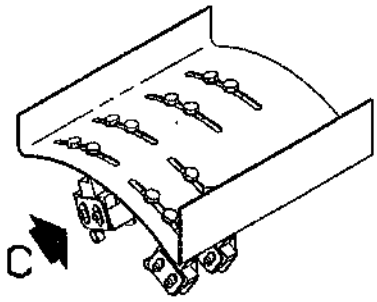

DETAIL B

1. ELECTRICAL WIRES
2. SWITCH
3. SCREWS
4. WASHERS

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Autofeather Power Lever Switches Installation (Effectivity: All)
Figure 3

a. While both engines are running, set the power levers to idle, the autofeather switch to OFF and verify that the yellow AFX OFF message is illuminated.

b. Select and hold the propeller AUTOFEATHER switch in the TEST position. The yellow AFX OFF message will extinguish. This will indicate that the high pressure switches are in the unactuated (open) position. Observe that the propellers remain unfeathered and that the green AFX ENABLED message is not illuminated. Continue to hold the propeller AUTOFEATHER switch in the TEST position during the autofeather operational check.

c. Advance both power levers to approximately 22% engine torque pressure. Verify that the green AFX ENABLED message is illuminated in the EICAS list and that the green AFX messages are illuminated in the center of each propeller speed gauge. Illumination of these messages indicate that the high pressure switches have been actuated to the closed position.

d. Slowly retard the LH power lever. The AFX message in the right propeller speed gauge and the green AFX ENABLED message in the EICAS should extinguish when LH engine torque indicates between 13% and 19%. At between 6% and 12% torque on the LH engine, the LH propeller will begin to feather. As the LH propeller feathers, the AFX message in the left propeller speed gauge will blink on and off. This indicates that the LH low pressure switch is closed and that the LH dump valve solenoid is working properly. It also ascertains that the circuit between the LH and RH systems has functioned properly and that the LH high pressure switch has deactuated (opened). Advance the LH power lever to 22% torque on the LH engine. Both AFX messages in the propeller speed gauges and the green AFX ENABLE in the EICAS should illuminate.

e. Slowly retard the RH power lever. The AFX message in the LH propeller speed gauge and the green AFX ENABLE in the EICAS will extinguish when the RH engine torque indicates between 13% and 19%. At between 6% to 12% RH engine torque the RH propeller will begin to feather. As the propeller feathers, the AFX message in the RH propeller speed gauge will blink on and off. This confirms that the circuit between the RH and LH systems is functioning properly and that the RH high pressure has deactuated (opened). It also indicates the RH low pressure switch is closed and that the RH dump valve solenoid is working properly. Advance the RH power lever to 22% RH engine torque. Both AFX messages in the propeller speed gauges and the green AFX ENABLE message in the EICAS should illuminate.

f. Retard both power levers back to the IDLE position together. The AFX messages in both propeller speed gauges and the AFX ENABLE message on the EICAS list should extinguish. The propellers should not feather.

g. Release the propeller AUTOFEATHER switch from the TEST position and verify that the propeller AUTOFEATHER switch is placed in the ARM position.

h. Advance each power lever individually above 92% N_1 and verify that neither of the AFX messages (in the propeller tachometer gauges) illuminate.

i. Set the LH power lever to 92% N_1 and slowly advance the RH power lever. The autofeather system will arm and the AFX messages should illuminate at 89% to 91% N_1. This indicates that the RH power lever switches are closing properly.

j. Retard the RH power lever. The autofeather system will disarm and the AFX messages should extinguish at 85% to 87% N_1. This indicates that the RH power lever switches are opening properly.

k. Set the RH power lever to 92% N_1 and slowly advance the LH power lever. The autofeather system will arm and the AFX messages should illuminate at 89% to 91% N_1. This indicates that LH power lever switches are closing properly.

l. Retard the LH power lever. The autofeather system will disarm and the AFX messages should extinguish at 85% to 87% N_1. This indicates that the LH power lever switches are opening properly.

SYNCHROPHASING - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The Type II propeller synchrophaser system automatically matches the rpm of both propellers by maintaining a specific phase relationship between the blades of the left and right propellers. The control box senses pulses which are generated by pickups mounted at identical locations in relation to the engine centerline on both engines. Ferrous metal targets, mounted on the propeller spinner bulkheads, provide the pulse reference for the pickups.

The magnetic pickups operate on the magnetoelectric induction principle: when the ferrous metal target passes in close proximity to the magnetic field of the pickup, an alternating current is induced in the pickup. The waveform of the current is essentially square-wave with a very narrow bandwidth at the peaks of the wave. The amplitude of the positive signal may not necessarily be equal to the amplitude of the negative signal; therefore, it will be necessary to measure peak-to-peak voltages (the magnitude of difference between the voltage of the positive peak and the voltage of the negative peak) in order to properly assess the operating outputs of the pickups. The frequency and voltage of the alternating current is dependent upon propeller rpm. The entire waveform cycle occurs during a time span of less than 1/5000 second. As a result of the peculiar characteristics of the AC waveform produced by the pickups, an oscilloscope must be used to measure peak-to-peak voltages. The magnitude of the peak-to-peak voltage should be at least 2.4 volts at minimum cruise speeds.

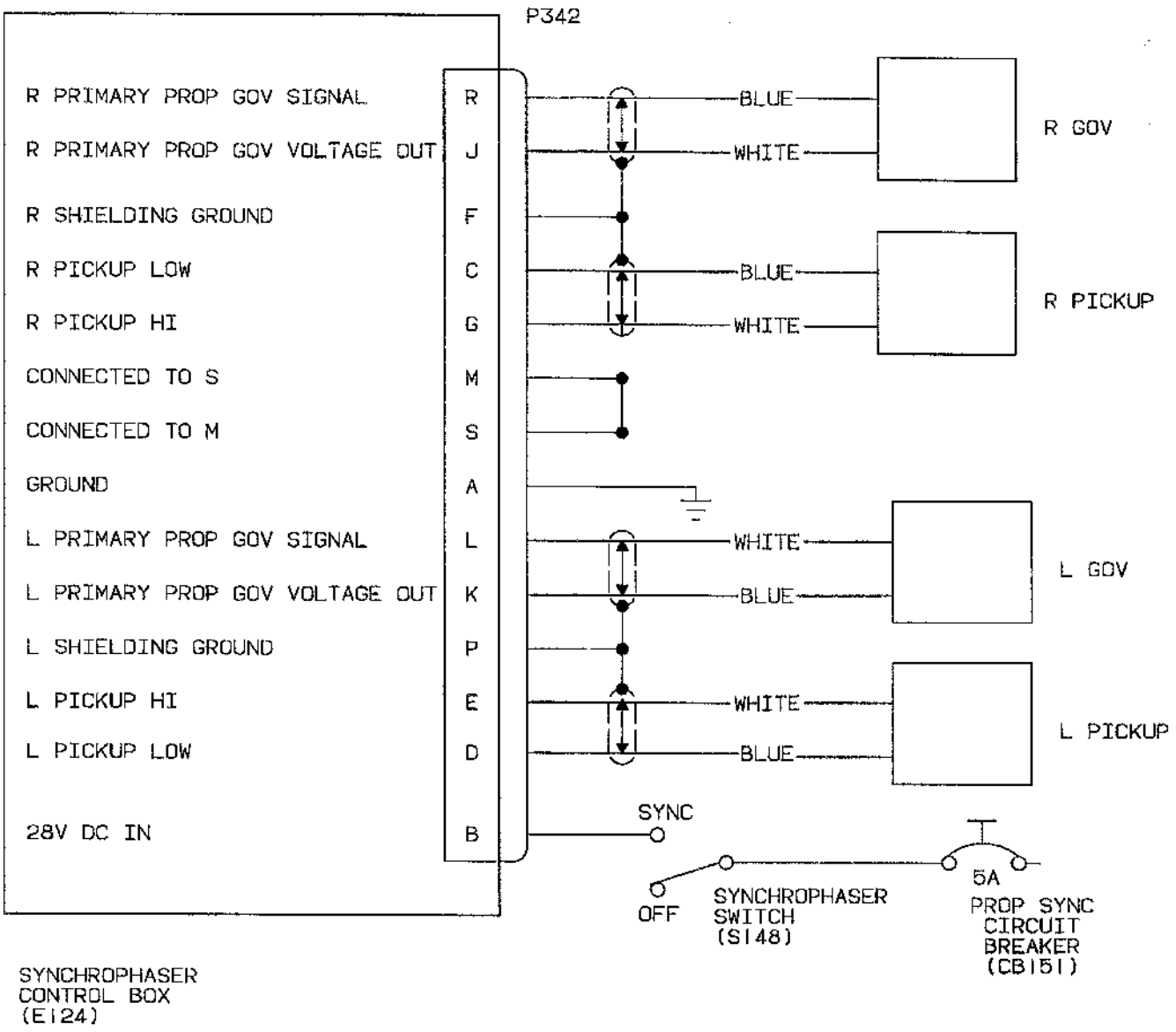
The control box does not respond to amplitude of the input signals, but to phasing of the waveforms. A specific amount of voltage change, approximately 0.85 volts, is the minimum necessary to trigger the control box. Maintaining a time-phase relationship is the control box's only priority. The control box senses the pulses from each pickup and attempts to superimpose the waveforms by trimming the speed of each propeller.

Speed trim of the propellers is accomplished by the control box which sends correction commands to each propeller governor. The character of these correction commands is in the form of pulse-width-modulated 28-volt direct current: the duration of the pulse of current is regulated to produce the proper amount of speed trim. The amount of speed trim per pulse-width-modulation by the control box is a function of the governor and will always be within a very narrow range (holding range, approximately 20 rpm).

The governor servo can increase slightly, but never decrease, the speed set by the propeller control lever. The rpm of one propeller will follow the changes in rpm of the other propeller over the predetermined holding range of the governor. This limited holding range prevents either propeller from losing more than a limited rpm if the rpm of the other propeller is manually reduced, such as in power changes or propeller feathering, while the synchrophaser is on.

The synchrophaser system is controlled through a toggle switch placarded PROPELLER, SYNC - OFF. To operate the system, synchronize the propellers in the normal manner and place the synchrophaser switch in the SYNC position. To change rpm, adjust both propellers at the same time. This will keep the setting within the holding range of the system. If the synchrophaser is on, but will not synchronize propellers, the propeller speeds are not within the limits required for the system to assume control (outside capture range). Turn the synchrophaser off, synchronize the propellers manually, then turn the snychrophaser on.

The propeller synchronizer display consists of a series of square white boxes which are located on the EICAS screen below the propeller display. If the white boxes are motionless, the propellers are synchronized. When the white boxes move to the right, the RH propeller rpm is higher. If the white boxes are moving to



Control Box Wiring Schematic (Effectivity: All)
Figure 1

the left, the LH propeller rpm is higher. The white boxes will disappear from the EICAS display if either propeller signal fails.

SYNCHROPHASING - TROUBLESHOOTING (Effectivity: All)

The first phase of troubleshooting the synchrophaser system is the functional check. Performing a functional check of the synchrophaser system before continuing through the troubleshooting sequence should yield information essential to identifying faults within the system. The functional check which follows should be performed as the first phase of troubleshooting. The second phase of troubleshooting will include performing continuity checks on the synchrophaser system wiring and peripheral components. The breakout box (5, Chart 1, Chapter 61-00-00) can be fabricated in the shop and will provide convenient access to the various circuits of the synchrophaser system. The breakout box can also be used to check the operation of the propeller governors. If continuity checks fail to reveal the nature of the fault, it will be necessary to begin the third phase of troubleshooting which is to evaluate the operation of the propeller governors. The fourth phase of troubleshooting, if no faults became apparent during previous troubleshooting attempts, is to evaluate the character of the outputs from the magnetic pickups.

If the previous checks or procedures do not reveal the exact nature of the fault, the fault may be assumed to be in the control box, and a control box, known to be operational, should be installed.

Troubleshooting Chart 1 proposes a logical sequence of checks and activities to be performed during the troubleshooting process and should aid the technician in eliminating confusion and maintaining logical order. The most probable causes of faults in the system are detailed in this chart.

CHART 1
TROUBLESHOOTING - PROPELLER SYNCHROPHASER SYSTEM (Effectivity: All)

Propeller Synchrophaser

STEP 1	Do the engines synchronize within a few seconds to one minute? (Note 1)	**YES**	System operating normally.
	NO		
STEP 2	Is the circuit breaker open?	**YES**	Close the circuit breaker and repeat step 1.
	NO		
STEP 3	Are the propellers within the 10 rpm capture range?	**YES**	Perform circuit integrity test (Note 2). Proceed to step 4.
	NO		
	Turn system off, manually synchronize propellers within 10 rpm and turn system on.		
STEP 4	Are circuit continuity checks correct?	**YES**	Proceed to step 5.
	NO		
	Correct wiring fault or replace faulty pickup or governor.		
STEP 5	Can both governors be manually operated? (Note 3)	**YES**	Check pickup outputs (Note 4) and proceed to step 6.
	NO		
	Replace the faulty governor.		
STEP 6	Are peak to peak voltages within tolerance?	**YES**	Proceed to step 9.
	NO		

CHART 1
TROUBLESHOOTING - PROPELLER SYNCHROPHASER SYSTEM (Effectivity: All) *(Continued)*

Propeller Synchrophaser

STEP 7	Can voltages be adjusted by varying the pickup gap? (Note 5)	**YES**	If system is still inoperative replace the control box.
	NO		
STEP 8	Is the face of the target parallel to the face of the pickup?	**YES**	Replace the pickup.
	NO		
	Correct the target alignment.		
STEP 9	Are AC waveforms identical in appearance?	**YES**	Replace the control box.
	NO		
	Adjust or replace targets.		

1. Perform synchrophaser FUNCTIONAL CHECK procedure.
2. Perform synchrophaser CONTINUITY CHECK procedure.
3. Perform propeller GOVERNOR OPERATIONAL CHECK procedure.
4. Perform synchrophaser PICKUP OUTPUT CHECK procedure.
5. Refer to synchrophaser PICKUP INSTALLATION AND ADJUSTMENT procedure.

SYNCHROPHASING FUNCTIONAL CHECK (Effectivity: All)

The proper operation of the synchrophaser system may be ground checked as follows:

CAUTION

Do not run engines with the propeller spinners off or damage to the oil cooler exit duct may occur when the propeller blade closeouts bend as a result of centrifugal force.

a. Start the engines and increase power until the speed on both propellers is at 1700 rpm with the propeller levers in high rpm.

b. Using the propeller levers, reduce propeller rpm on both engines to 1600 rpm as nearly as possible. The propellers must be matched within 10 rpm or less to engage the synchrophasers.

CAUTION

Do not exceed torque redline.

c. Turn the synchrophaser switch to the SYNC (on) position. Allow 30 seconds for the propellers to drift within phasing range (30 rotational degrees) before the system corrects the phase relationship electronically. The rpm of both engines will be raised slightly when the synchrophaser engages.

d. Using the propeller lever, increase the rpm of the LH propeller slightly and check that the synchrophaser increased the rpm on the RH propeller to match.

1. If the previous check yields negative results, turn the synchrophaser off and repeat the above procedure.

e. If the previous check indicated that the synchrophaser is operating, continue increasing the propeller rpm with the propeller lever until synchronization is lost and record this rpm.

f. Using the propeller lever, slowly decrease the propeller rpm on the left engine below the rpm of the right engine until synchronization is again lost and record this rpm.

g. The difference between the two propeller speeds at which synchronization was lost is the holding range and should be at least 20 rpm. This check evaluates the ability of the right propeller governor to respond to correction commands from the control box, as well as the control box's ability to issue commands properly.

1. Failure of this check will necessitate performing the synchrophaser continuity check.

h. Reset the system by turning the synchrophaser switch to the OFF position; match propeller rpm to within 10 rpm of each other and turn the synchronizer to the SYNC (on) position.

i. Repeat the functional check procedure by varying the propeller rpm of the right engine while letting the left propeller governor respond.

j. During this functional check, determine that the communications and navigation equipment is not adversely affected by the synchrophaser operation and that synchrophaser operation is not adversely affected by the operation of other electrical equipment.

SYNCHROPHASING CONTINUITY CHECK (Effectivity: All) (Figures 1 and 2)

This check will help locate the source of the fault should the synchrophaser system fail the functional check. A sensitive volt/ohmmeter and the breakout box (5, Chart 1, Chapter 61-00-00) will be required to perform these continuity checks. The control box is located forward of the aft pressure bulkhead, under the aft step electrical equipment panel.

a. Check that the battery master switch is OFF, and PULL (open) the synchrophaser circuit breaker.

b. Remove the connector from the control box.

c. Check the synchrophaser system wiring for conformity to the schematic in Figure 1.

d. Perform continuity checks for the control box as indicated in Chart 2.

NOTE

Resistance readings noted in Chart 2 may be as much as 20% higher during heat soak following engine shutdown.

e. After performing the continuity checks, close the synchrophaser circuit breaker, turn the battery master switch on and check for battery voltage between pins A and B.

f. Correct any wiring faults that may have been revealed during these continuity checks. Replace any component which failed to meet the proper resistance specified for that component. Restore the system to operational status and perform the functional check again.

CAUTION

DO NOT connect the plug to the control box before turning the battery master switch OFF and pulling (open) the synchrophaser circuit breaker.

g. If no faults were found by the continuity checks, perform the propeller GOVERNOR OPERATIONAL CHECK.

CHART 2 SYNCHROPHASER CONTINUITY CHECKS

Test Between Receptacle Pins	Ohmeter Indication	Component
A and ground	0 ohms	
A and B	Open circuit	
C and G	52 - 68 ohms	Right Pickup
D and E	52 - 68 ohms	Left Pickup
J and R	115 - 206 ohms	Right Governor
K and L	115 - 206 ohms	Left Governor

Resistances may be as much as 20% higher during heat soak after engine shutdown.

GOVERNOR OPERATIONAL CHECK (Effectivity: All) (Figures 1 and 2)

The following checks assess the operation of the governors and should be performed before evaluating the outputs of the magnetic pickups.

NOTE

The pulsed lights on the breakout box (5, Chart 1, 61-00-00) inform the operator that the breakout box is providing a full or partial ground to the governors.

a. Connect the breakout box (5, Chart 1, 61-00-00) between the airplane harness and the control box.

CAUTION

Do not run engines with the propeller spinners off or damage to the oil cooler exit duct may occur when the propeller blade closeouts bend as a result of centrifugal force.

CAUTION

Do not exceed torque redline.

b. Start the engines and increase the propeller rpm to 1700 rpm with the propeller levers at high rpm.

c. Using the propeller levers, reduce the rpm to 1600 rpm to ensure that the propellers are being controlled by the governors.

CAUTION

Do not insert jumpers larger than .040 inch in diameter into the connector socket or the connector will be damaged.

d. Press to test the lights on the breakout box.

e. Check for 28 vdc between pin J and ground and pin K and ground on the breakout box.

f. Install a jumper between pins A and R of the breakout box. The speed of the right propeller should noticeably increase when the synchrophaser switch is turned on (propeller rpm should increase approximately 1%). Turn the synchrophaser switch OFF.

g. Install the jumper between pins A and L on the breakout box. The speed of the left propeller should noticeably increase when the synchrophaser is turned on. (propeller rpm should increase approximately 1%)

h. A failure of either propeller governor to increase propeller speed indicates that there is a fault in that governor.

i. If no faults were revealed by this operational check, evaluate the character of the outputs from the pickups before condemning the control box.

PICKUP OUTPUT CHECK (Effectivity: All) (Figures 1, 2 and 3)

This procedure makes it possible to evaluate the character of the pickup outputs and to make adjustments as required. An oscilloscope is required to display the output waveform and measure the peak-to-peak voltages.

a. Be sure the battery master switch is turned OFF and the synchrophaser circuit breaker is pulled (open).

b. Disconnect the plug from the control box.

c. Connect the breakout box (5, Chart 1, 61-00-00) to the airplane connector.

d. Connect a jumper between pins F and ground and P and ground on the breakout box.

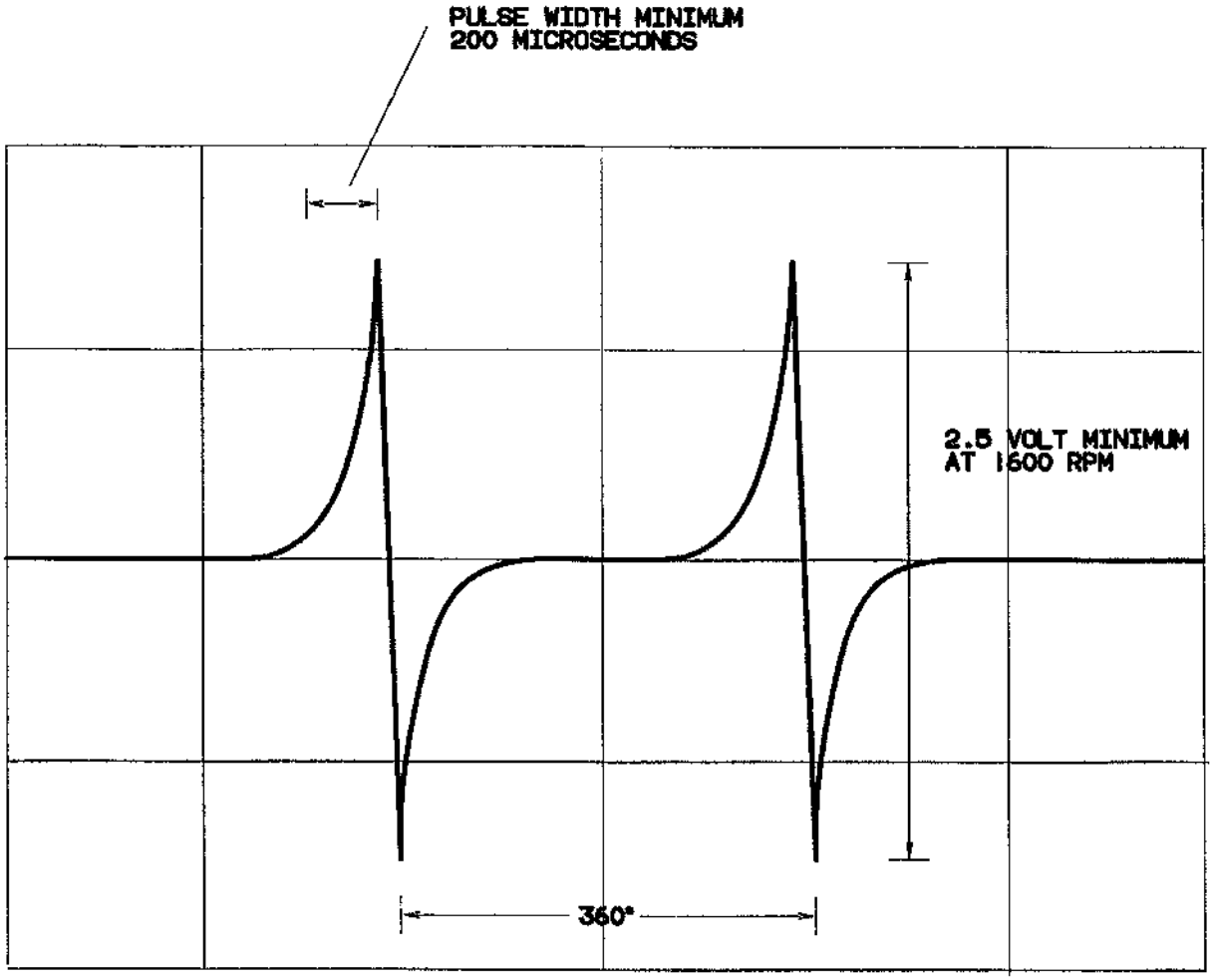
CAUTION

Ensure that the breakout box plug is disconnected from the control box. If pins K or J are connected to ground while the control box is operational, the control box will be damaged.

e. Connect an oscilloscope between pins G and C for the right pickup and between pins E and D for the left pickup.

CAUTION

Do not run engines with the propeller spinners off or damage to the oil cooler exit duct may occur when the propeller blade closeouts bend as a result of centrifugal force.



Typical Pickup Waveform (Effectivity: All)
Figure 2

CAUTION

Do not exceed engine torque redline.

f. Start the engines and adjust the propeller speeds to 1700 rpm as nearly as possible.

g. The waveform generated by the pickups should depart smoothly from the baseline, rise smoothly to the positive peak, drop off to the negative peak and return to the baseline (see Figure 2). There should be no secondary peaks at either the positive or negative peaks of the waveform. The ascending and descending sides of the peaks should be symmetrical. Electromagnetic interference should not exceed +0.5 volts on the input signal baseline.

h. Retard the propeller levers until propeller speeds of 1600 rpm are obtained. Check the output amplitude as outlined in step e.

1. A ragged waveform may indicate an intermittent fault in the pickup.

2. An asymmetrical waveform may indicate that the target is not parallel to the tip of the pickup.

3. A secondary peak in the waveform may indicate that the target has been damaged.

i. Record the peak-to-peak voltage of the waveforms from each pickup. The peak-to-peak voltage should be at least 2.4 volts at 1600 rpm. Waveform pulse width should be a minimum of 200 microseconds. There should be only 1 waveform per propeller shaft rotation.

1. If peak-to-peak voltage of the pickups is not within tolerance, perform the MAGNETIC PICKUP INSTALLATION AND ADJUSTMENT procedure.

SYNCHROPHASING - MAINTENANCE PRACTICES (Effectivity: All)

The synchrophaser magnetic pickups are located in a bracket which is bolted to the aft side of the reduction gearbox. The magnetic pickup should be removed, installed and adjusted as follows:

MAGNETIC PICKUP REMOVAL (Effectivity: All) (Figure 3)

a. Remove the lower aft engine cowling to gain access to the magnetic pickup (4).

b. Disconnect the electrical leads from the magnetic pickup (4).

c. Unscrew the locknut (3) and remove the magnetic pickup (4) from the mounting bracket (1).

MAGNETIC PICKUP INSTALLATION AND ADJUSTMENT (Effectivity: All) (Figure 3)

CAUTION

Do not adjust the synchrophaser target gap unless the propeller spinner dome is installed. Installing the propeller spinner dome will cause the propeller bulkhead to be forced aft, which will widen the synchrophaser target gap.

WARNING

Be sure that the propeller shaft is forced forward while the synchrophaser target gap is adjusted. Propeller shaft movement will cause the target gap to decrease when the engine is started, damaging the magnetic pickup and propeller spinner bulkhead.

a. Screw the magnetic pickup (4) into the mounting bracket (1).

b. Adjust the magnetic pickup (4) to provide a clearance of .035 to .040 inch between the magnetic pickup (4) and the target (2).

NOTE

This adjustment is a preliminary setting. A final adjustment may be necessary to meet the peak-to-peak voltage requirements while performing the pickup output check.

c. Torque the locknut (3) on the magnetic pickup (4) to a maximum of 25 inch-pounds and secure the locknut (3) with threadlock (2, Chart 2, Chapter 61-00-00).

d. Connect the electrical leads to the magnetic pickup (4).

e. Measure the output of the magnetic pickup (4) according to the synchrophaser PICKUP OUTPUT CHECK procedure.

f. If the connector plug was previously disconnected, reconnect the plug to the control box.

g. Perform the synchrophaser FUNCTIONAL CHECK procedure.

h. Reinstall the lower aft engine cowling.

CONTROL BOX REMOVAL (Effectivity: All) (Figure 4)

a. Remove electrical power from the airplane.

b. Refer to Chapter 53-20-00 and remove the aft step.

c. Disconnect electrical connector (3) from the control box (4).

d. Remove the screws (1), washers (5) and spacers (2) from the control box (4).

e. Remove the control box (4) from the airplane.

CONTROL BOX INSTALLATION (Effectivity: All) (Figure 4)

a. Position the control box (4) in the airplane.

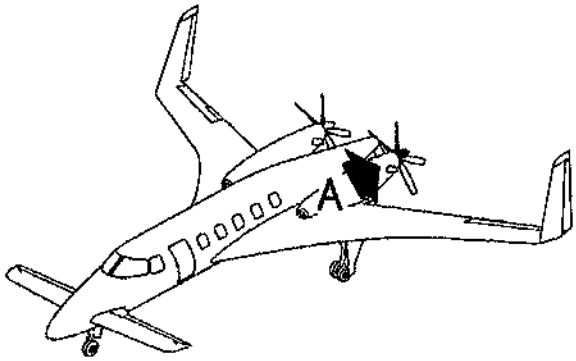
b. Install the spacers (2), washers (5) and screws (1) to secure the control box (4).

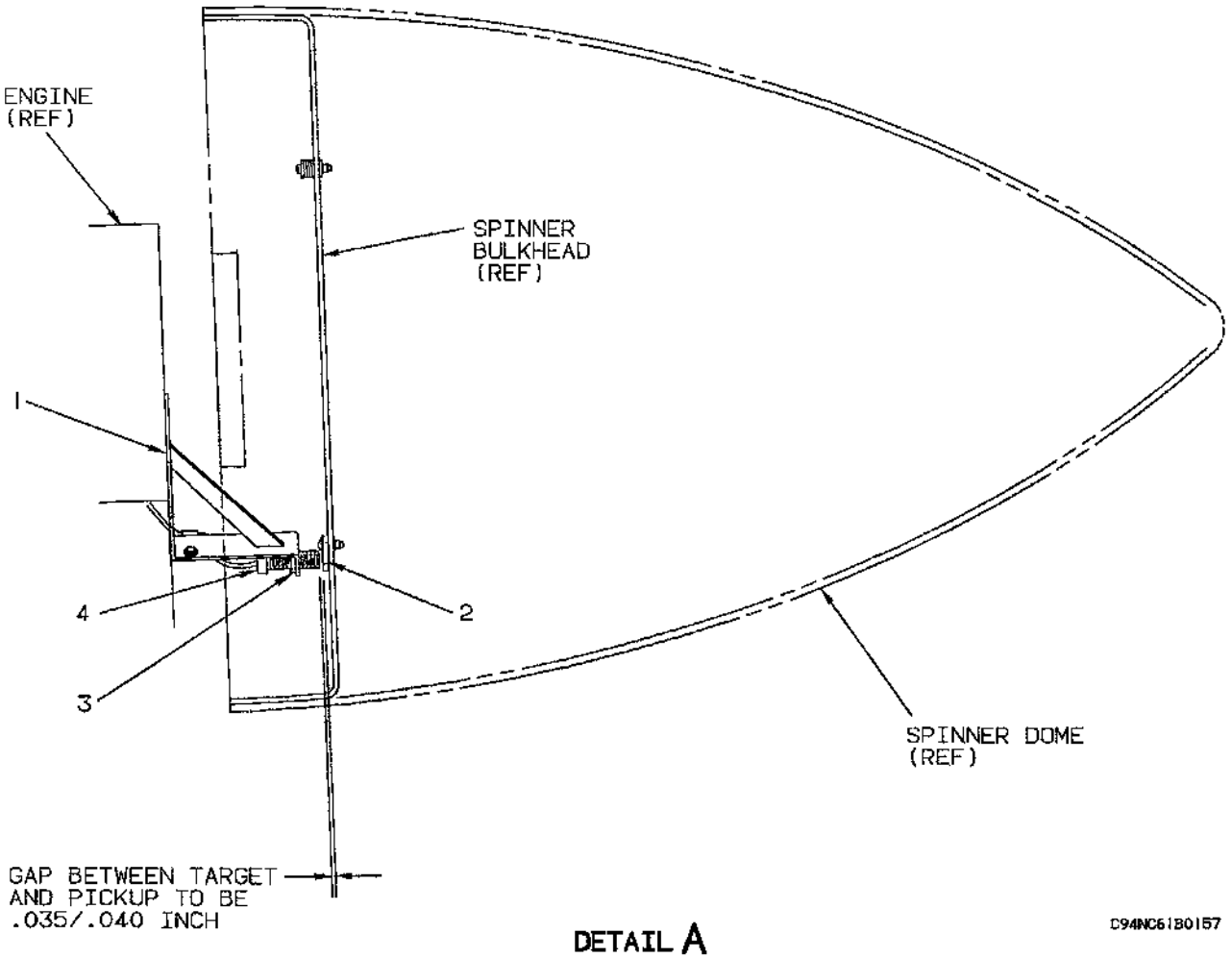
c. Connect the electrical connector (3) to the control box (4).

d. Refer to Chapter 53-20-00 and install the aft step.

e. Perform the SYNCHROPHASING FUNCTIONAL CHECK in this section.

1. BRACKET
2. MAGNETIC TARGET
3. LOCK NUT
4. MAGNETIC PICKUP

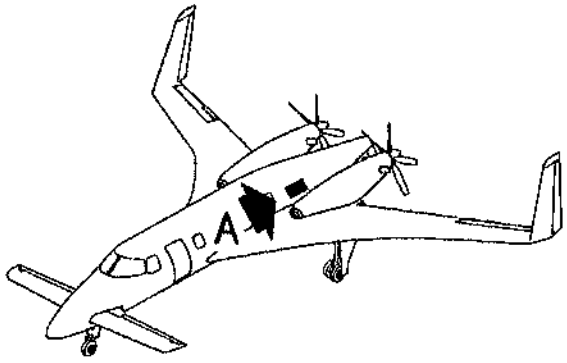


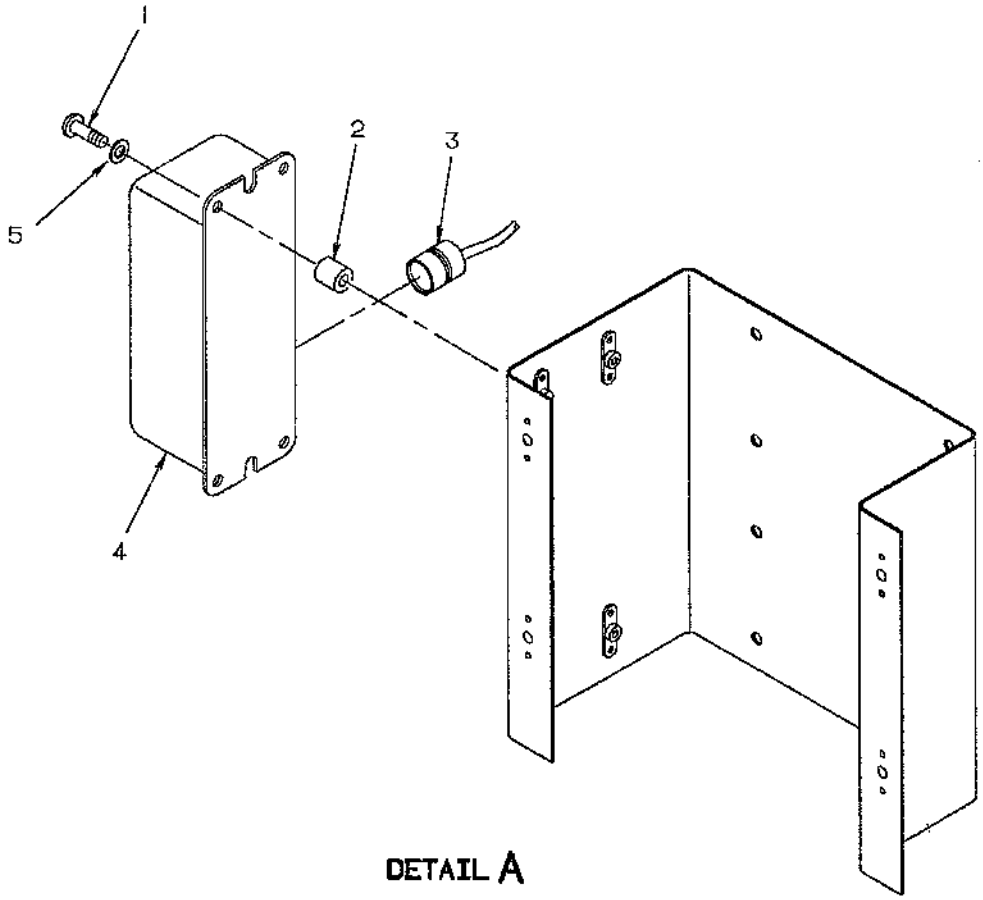


DETAIL A

Magnetic Pickup Installation and Adjustment (Effectivity: All)
Figure 3

1. SCREW
2. SPACER
3. ELECTRICAL CONNECTOR
4. CONTROL BOX
5. WASHER





DETAIL A

C94NC61B0156

Control Box Installation (Effectivity: All)
Figure 4

INDICATING - DESCRIPTION AND OPERATION (Effectivity: All)

FX SWITCHES (Effectivity: All)

The FX switches are mounted on a bracket in the lower pedestal and are configured to actuate when the propellers are feathered. When the propeller levers are pulled back into the feathered position, the propeller control bellcranks are rotated to a position that actuates the FX switches. When the switches actuate, signals are sent to the Engine Data Concentrator which removes the AFX messages from the Engine Instrument, Crew Alerting System (EICAS) display.

LOW PITCH STOP THROTTLE SWITCH (Effectivity: All)

The low-pitch-stop throttle switch is designed to actuate the ground-idle-stop solenoid when the power levers are placed in the GND FINE position. The low-pitch-stop throttle switch is a momentary contact type lever switch and will only function while the airplane is on the ground. Power to the ground-idle low-pitch-stop system is provided through the landing gear safety switch (squat switch). When the GND FINE position is selected, the low-pitch-stop throttle switch sends electrical power from the A129 aft relay panel to the ground-idle-stop solenoid. The ground-idle-stop solenoid reduces the propeller blade pitch angle.

When the power levers are pulled back and lifted into the GND FINE position, the power levers lift the low-pitch-stop lever. A link connected to the low-pitch-stop lever raises a lever arm that contacts the low-pitch-stop throttle switch. When the power levers are raised, the low-pitch-stop throttle switch is deactuated and the propeller blade pitch is reduced.

PROPELLERS INDICATING - MAINTENANCE PRACTICES (Effectivity: All)

PROPELLER FX SWITCH REMOVAL (Effectivity: All) (Figure 1)

a. Remove all electrical power from the airplane and disconnect the battery.

b. Remove the LH cover from the side of the pedestal assembly.

c. Disconnect the electrical wires from the FX switches.

d. Remove the two screws and washers which retain each switch to the bracket. Save the clips (located below the pedestal bracket) for installation.

e. Remove the switch hangers (with the FX switches still attached) from the pedestal.

PROPELLER FX SWITCH INSTALLATION AND ADJUSTMENT (Figure 1)

a. Make sure all electrical power is removed from the airplane and that the battery is disconnected.

b. Put the switch hanger on top of the clip and place in position below the pedestal bracket.

c. Align the holes in the clip and switch hanger with the slots in the pedestal and attach the switch hanger and clip to the pedestal bracket with the screws and washers.

d. Adjust the switch actuation point by sliding the switch hanger in the pedestal bracket slot. Set the switch hangers so that the switches are actuated when the prop levers are in the feathered position. Tighten the screws until the switch hangers are secure.

e. Connect the electrical wires to the FX switches.

f. Install the cover on the LH side of the pedestal assembly.

g. Connect the battery and restore electrical power to the airplane.

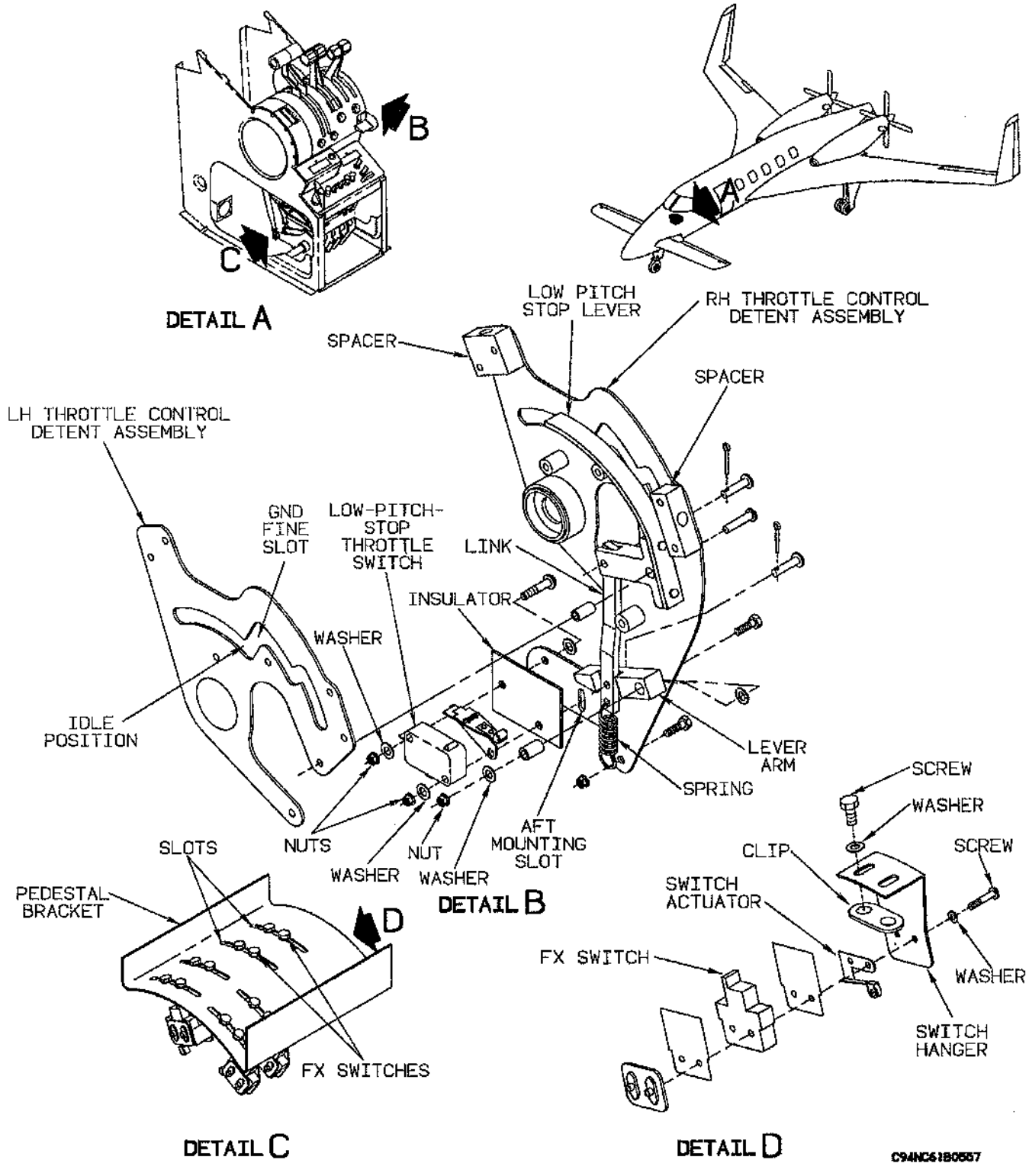
THROTTLE SWITCH ADJUSTMENT (Effectivity: All) (Figure 1)

The low-pitch-stop throttle switch is attached to the LH side of the RH throttle control detent assembly which is located below the power levers. Access to the low-pitch-stop throttle switch is gained through the upper LH side of the pedestal. Adjust the low-pitch-stop throttle switch as follows:

a. Remove the cover from the LH side of the pedestal.

b. Move the power lever switches into the IDLE position.

c. Loosen the two nuts that retain the insulator and low-pitch-stop throttle switch to the LH side of the RH throttle control detent assembly.



Propeller Low Pitch Stop and FX Switches (Effectivity: All)
Figure 1

d. Adjust the low-pitch-stop throttle switch in the aft mounting slot so that the switch deactivates when the low-pitch-stop lever is raised midway up through the GND FINE slot. Check the switch to see that it activates before the low-pitch-stop lever is lowered to the idle slot in the control detent assembly.

e. Tighten the nuts that retain the low-pitch-stop throttle switch to the LH side of the RH throttle control detent.

f. Verify that the low-pitch-stop throttle switch deactuates as the power levers are pulled up into the GND FINE position. Make sure that the switch fully actuates as the power levers are lowered into the IDLE position.

g. Install the cover on the upper LH side of the pedestal.

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CHAPTER 71 - POWER PLANT

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GENERAL - DESCRIPTION AND OPERATION

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

POWERPLANT

The Starship 1 is equipped with two PT6A-67A power plants. The PT6A-67A power plant is a light weight free turbine engine and is mounted to the airframe in pusher configuration. The engine utilizes two independent turbine sections: the first section drives the compressor in the gas generator and the second section (two-stage power turbine) drives the propeller shaft through a reduction gearbox. This engine is self-sufficient because its gas generator driven oil system provides lubrication for all areas of the engine, pressure for the torque meter and power for the propeller pitch control.

Inlet air enters the engine though an annular plenum chamber, formed by the compressor inlet case, where the air is directed rearward to the compressor. The compressor consists of four axial stages that are combined with a centrifugal stage. The compressor is assembled as an integral unit. The engine is equipped with a water wash ring at the compressor air inlet case.

A row of stator vanes, located between each stage of compression, diffuses the air, raises its static pressure and directs it to the next stage of compression. The compressed air passes through diffuser tubes which turn the air through ninety degrees in direction, converting air velocity into static air pressure. The diffused air then passes through straightening vanes in route to the annulus surrounding the combustion chamber liner assembly.

The combustion chamber liner consists of two annular weldments or wrappers bolted together at the front dome-shaped end. The outer wrapper incorporates a large integral exit duct. The liner assembly has perforations of various sizes that allow entry of compressor delivery air. The flow of air changes direction 180 degrees as it enters and mixes with fuel. The fuel/air mixture is ignited and the resultant expanding gasses are directed to the turbines. The location of the liner eliminates the need for a long shaft between the compressor and the compressor turbine which reduces the overall length and weight of the engine.

Fuel is injected into the combustion chamber liner through 14 simplex nozzles arranged in two sets of seven and supplied by a dual manifold consisting of primary and secondary transfer tubes and adapters. The fuel/air mixture is ignited by two spark igniters which protrude into the liner. The resultant gasses expand from the liner, reverse direction in the exit duct zone and pass through the compressor turbine inlet guide vanes to the single-stage compressor turbine. The guide vanes ensure that the expanding gasses impinge on the turbine blades at the correct angle, with minimum loss of energy. The gasses (still expanding) are then directed aft to drive the power turbine section.

The two-stage power turbine consists of the first-stage inlet guide vane and turbine, and a second-stage inlet guide vane and turbine. The power turbine drives the propeller shaft via a reduction gearbox.

The compressor and power turbines are located near the center of the engine with their respective shafts extending in opposite directions. Exhaust gas from the power turbine is collected and ducted in the bifurcated (twin chambered) exhaust duct assembly and directed to the atmosphere via twin opposed exhaust stacks.

The interturbine temperature (T5) is monitored by an integral bus-bar, probe and harness assembly. The probe and harness assembly is installed between the compressor turbine and the power turbine, with probes projecting into the gas path. A terminal block mounted on the gas generator case provides a connection point for cockpit instrumentation.

All engine-driven accessories, with the exception of the propeller governor, overspeed governor and N_1 tachometer-generator, are mounted on the accessory gearbox at the front of the engine. These components are compressor driven by means of a coupling shaft extending the drive through a tube at the center of the oil tank.

The engine oil supply is contained in an integral oil tank which forms the rear section of the compressor inlet case. The tank has a total capacity of 2.5 U.S. gallons and is provided with a filler, an oil level sight gage, and remote electrical oil level indicator.

Fuel supplied to the engine from an external source is further pressurized by an engine-driven fuel pump and its flow to the fuel manifold is controlled by the fuel control unit (FCU).

The power turbine drives the propeller through a two-stage planetary reduction gearbox located at the rear of the engine. The gearbox embodies an integral torquemeter device which is instrumented to provide an accurate indication of engine power. A chip detector is installed at the bottom of the gearbox. Refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual for a complete description of the engine, engine accessories and additional maintenance information.

RECOMMENDED MATERIALS

The recommended materials listed in Chart 1 as meeting federal, military or vendor specifications are provided for reference only and are not specifically recommended by Raytheon Aircraft Corporation. Any product conforming to the product listed may be used. The products included in this chart have been tested and approved for aviation usage by Raytheon Aircraft Corporation, by the vendor or by compliance with the applicable specification. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included herein. Only the basic number of each product specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
RECOMMENDED MATERIALS

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
1. Pipe Sealant with Teflon		LOCTITE 592	Loctite Corporation, 705 N. Mountain Rd. Newington, CT 06111
2. Paint stripper		Turco 5351	Turco Products Inc., 24600 S. Main Los Angeles, CA 90746
3. Epoxy Primer	MIL-P-23377	Primer	Sterling Lacquer Mfg. Co., 3150 Brannon Ave St. Louis, MO 63139-1422
4. Aluminum Epoxy Paint	MIL-C-16173	Paint	
5. Lubricant, Petroleum	VV-P-236		

GENERAL - MAINTENANCE PRACTICES

ENGINE REMOVAL

Although it is possible to remove the engine from the engine truss mount before removing the engine mount from the firewall, it is recommended that the engine and the engine truss mount be removed from the firewall together. This will facilitate maintenance on the engine. The engines are removed and installed as described in the following procedures. Refer to Figures 1, 2, 3, and 4.

NOTE

Airplanes idle for more than 7 days due to maintenance or other reasons should either have the engines operated or preserved. Refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual or Chapter 10-10-00 in this maintenance manual for the proper procedures.

a. Turn the Master-Battery switch ON.

b. Press the LH "ENG FIRE F/W VALVE PUSH CLOSED" and RH "ENG FIRE F/W VALVE PUSH CLOSED" switches. These switches are located on the glareshield and will close the firewall fuel shut off valves to prevent fuel from flowing to the engine. Wait at least 30 seconds.

CAUTION

When the LH and RH "ENG FIRE F/W VALVE PUSH CLOSED" switches are pressed, the LH and RH "EXTINGUISHER PUSH" switches will illuminate. DO NOT press the "EXTINGUISHER PUSH" switches. If either "EXTINGUISHER PUSH" switch is pressed, the contents of the fire extinguishing system on that side will be released.

c. Turn the master-battery switch OFF.

d. Pull all engine circuit breakers.

e. Remove all electrical power from the airplane and disconnect the battery.

f. Remove the engine cowling as outlined in ENGINE COWLING REMOVAL, Chapter 71-10-00.

g. Place a suitable drip pan underneath the engine to catch fuel and oil.

h. Drain the engine oil as outlined in Chapter 12-10-00.

NOTE

Tag and identify all hoses, plumbing and electrical connections to ensure correct installation. Cap or cover all open hoses, plumbing, engine ports and electrical connections to prevent contamination. Note the position of all securing clamps and standoffs (before removing) for correct location when installing.

i. Disconnect the No. 1 and No. 2 electrical wire harness plugs from the firewall connectors as outlined in Chapter 71-50-00.

j. Disconnect the oil level OK test switch assembly wire and oil level OK light sending wire from the connector near the firewall as outlined in Chapter 71-50-00.

k. Remove the propeller as outlined in Chapter 61-10-00.

l. Remove the engine air plenum and air inlet duct as outlined in Chapter 71-60-00.

m. Place a suitable cover over the engine air inlet screen and secure it to prevent foreign matter from entering the engine air inlet.

n. Remove all engine drain tubes and hoses as outlined in Chapter 71-70-00.

o. Remove the engine oil cooler and engine oil cooler supports as outlined in Chapter 79-00-00.

p. Disconnect the P3 bleed air fitting from the engine case.

NOTE

Drain all excess fuel from the engine fuel hoses into a suitable clean container as the engine fuel hoses are disconnected. See Chapter 73-10-00 for details on engine fuel hose removal.

q. Disconnect the motive fuel flow hose from the TEE (located between the engine driven boost pump and the oil-to-fuel heater) by unscrewing the fitting on the TEE and removing the motive fuel flow hose from the TEE.

r. Disconnect the engine fuel supply hose from the lower pump inlet elbow fitting on the engine driven fuel boost pump by unscrewing the fitting and removing the engine supply hose from the pump inlet elbow fitting.

s. Disconnect the purge hose assembly from the engine fuel pump by unscrewing the fitting and removing the purge hose assembly from the top of the engine fuel pump.

t. Disconnect the power control cable rod (3) from the lever extension (5) on the LH side of the accessory gearbox mounting pad by removing the cotter pin (7), nut (6), washer (8) and screw (4). Remove the clamp that retains the power control cable (1) to the engine truss mount. Free the power control cable (1) from the engine by removing the nuts (9) that retain the power control mounting bracket (2) to the engine. (Refer to Figure 4)

u. Disconnect the propeller control cable (10) from the primary governor control lever (24) by removing the bolt (26) and washer (25) that retains the propeller control cable rod end (23) to the primary governor control lever (24). Remove the clip (17) and clamps (11, 12, 16 and 18) which retain the propeller control cable (10) to the engine truss mount (two places), air plenum (14) (two places), engine flange and engine bracket (22). (Refer to Figure 4)

v. Disconnect the condition control cable (27) at the rod end (37) by removing the cotter pin (35), nut (34), washer (33) and bolt (38) from the lever (36). Remove the clamps that retain the condition control cable (27) to the engine truss mount and engine air plenum. Free the condition control cable (27) from the engine by removing the nuts (32) that retain the condition control mounting bracket (31) to the engine. (Refer to Figure 4)

w. Remove the insulation blankets from the engine truss mount as outlined in Chapter 54-00-00.

x. Attach an engine sling to the engine hoisting lugs. The forward engine hoisting lug is located on the upper forward side of the compressor inlet case. The aft engine hoisting lug is located on the upper side of the reduction gearbox to exhaust duct flange.

y. Place a suitable engine hoist directly above the eye of the engine sling. Position the hoist so it can be backed away from the airplane.

z. Attach the engine hoist to the engine sling and raise the engine hoist until the weight of the engine and engine truss mount are supported by the engine sling.

NOTE

Identify and retain all engine truss mounting bolts, washers and nuts to ensure correct installation.

aa. While supporting the engine and engine truss mount, remove the engine and engine truss mount from the firewall as follows. (Refer to Figure 1)

1. Remove the lower inboard mounting bolt (13) from the bar nut (27).

2. Remove the lower outboard mounting bolt (18) from the locking nut (25) and engine truss mount (4).

3. Remove the upper inboard and upper outboard mounting bolts (3) from the barrel nuts (30) and retainers (31).

CAUTION

Before pulling the engine clear of the airplane, check to be sure that all wiring, cables, tube and hose assemblies are disconnected and free from snagging.

4. After the engine truss mounting bolts have been removed, slowly move the engine and engine truss mount from the firewall by rolling the engine hoist away from the airplane.

ab. When the engine and engine truss mount are clear of the airplane, lower the engine hoist and install the engine in a suitable floor stand as follows: (Refer to Figure 1)

1. Lower the engine hoist until the engine and engine truss mount are at a comfortable working level.

NOTE

Retain the spacer, castellated nut (8), washers (7) and isolator bolt (10) for installation. See Chapter 71-20-00 for details on engine mount isolator removal.

2. Cut the safety wire and remove the four engine bolts (6) that attach each of the five engine mount isolators (5) to the engine. Discard the gaskets.

3. Remove the cotter pin (9), castellated nut (8), spacer, washers (7) and isolator bolt (10) from the engine mount isolator (5).

4. Remove all five engine mount isolators (5) from the engine truss mount.

5. Attach a pair of engine stand adapter brackets to the engine at the engine mount isolator holes.

6. Lower the engine into the engine stand and secure the engine stand adapter brackets to the engine stand.

7. Disconnect the engine sling from the engine hoisting lugs.

ac. Remove the lower aft cross tube from the engine truss mount.

ad. Remove the engine truss mount from the engine.

ENGINE INSTALLATION

Before the engine is installed on the airplane, the engine truss mount and engine mount isolators should be installed,

and the engine should be built up. See Chapter 71-00-00 ENGINE BUILD UP, for engine build up and engine truss mount installation procedures.

a. Attach an engine sling to the engine hoisting lugs. The forward engine hoisting lug is located on the upper forward side of the compressor inlet case. The aft engine hoisting lug is located on the upper side of the reduction gearbox to exhaust duct flange.

b. Position a suitable engine hoist directly over the eye of the engine sling and connect the engine hoist to the engine sling.

CAUTION

Take care to prevent damage to the engine components, plumbing and electrical wiring when the engine is lowered into position.

c. Raise the engine hoist until the weight of the engine and engine truss mount are supported by the engine hoist. Remove the engine stand adapter brackets and engine stand from the engine.

d. Hoist the engine and engine truss mount from the floor stand and position it over the nacelle, aft of the firewall.

e. Slowly lower the engine and engine truss mount into position and align the bolt holes in the engine truss mount with the bolt holes in the firewall.

CAUTION

Attach the upper engine truss mount bolts to the firewall first. When this has been completed, relax the hoist and attach the lower engine truss mount bolts to the firewall. Torque each bolt to 450 to 500 inch-pounds.

f. While supporting the weight of the engine and engine truss mount with the engine hoist, attach the engine truss mount to the firewall in the following sequence:

1. Attach the engine truss mount (4) to the firewall (29) at the upper outboard and upper inboard bolt holes (1 and 12). Install the bolts (3) from the aft side of the engine truss mount (4), with a countersunk washer (2) installed under the head of each bolt (3). Secure each bolt (3) with a self locking barrel nut (30) and retainer (31). Torque to 450 to 500 inch-pounds. (Refer to Figure 1)

NOTE

To compensate for varying material thickness, a maximum of three flat washers (20) may be used to prevent bottoming out of the threads on the lower outboard bolt (18). If a flat washer(s) is used, a countersunk washer (19) must also be used (with the countersink towards the bolt head).

2. Attach the engine truss mount (4) to the firewall (29) at the lower outboard bolt hole (23). Relax tension on the engine hoist. Install the bolt (18), countersunk washer (19) [required only if flat washers (20) are used], flat washers (20) (if required, a maximum of three), convex washers (21), and concave washer (22) from the aft side of the engine truss mount (4). Install the flat washer (24) and locking nut (25) on the bolt (18). Torque to 450 to 500 inch-pounds. (Refer to Figure 1)

NOTE

To compensate for varying material thickness, a maximum of three flat washers (15) may be used to prevent bottoming out of the threads on the lower inboard bolt (13). If a flat washer(s) is used, a countersunk washer (14) must also be used (with the countersink towards the bolt head).

3. Attach the engine truss mount (4) to the firewall (29) at the lower inboard bolt hole (28). Install the bolt (13), countersunk washer (14) [required only if flat washers (15) are used], flat washers (15) (if required, a maximum of three), convex washers (16), and concave washer (17) from the aft side of the engine truss mount (4). Thread the bolt (13) into the bar nut (27). Torque to 450 to 500 inch-pounds. Insure by inspection that the threads on the bolt (13) are visible through the inspection hole (26) in the bar nut (27). (Refer to Figure 1)

g. Remove the engine hoist and sling from the engine.

h. Install the insulation blankets on the engine truss mount as outlined Chapter 54-00-00.

NOTE

Apply a light coating of thread sealant (1, Chart 1) to the male threads on all fuel hoses and connectors. See Chapter 73-10-00 for details on engine fuel hose installation.

i. Connect the purge hose assembly to the engine fuel pump by screwing the purge hose assembly onto the fitting at the top of the engine fuel pump.

j. Connect the engine fuel supply hose to the lower elbow fitting on the engine driven fuel boost pump.

k. Connect the motive fuel flow hose to the TEE (located between the engine driven boost pump and the oil-to-fuel heater) by screwing the fitting on the motive fuel flow hose onto the TEE.

l. Install all engine drain hoses and drain tubing as outlined in Chapter 71-70-00.

m. Connect the No. 1 and No. 2 firewall connector plugs to the connectors on the firewall as outlined in Chapter 71-50-00. Secure the No. 1 and No. 2 connector plugs with safety wire as outlined in Chapter 71-50-00.

n. Connect the oil level OK test switch assembly and oil level OK light sending wire to the connector located near the firewall as outlined in Chapter 71-50-00.

o. Install the propeller as outlined Chapter 61-00-00.

p. Connect the P3 bleed air tubing to the fitting on the RH side of the engine case.

q. Install the bleed air tubing by connecting the bleed air tubing to the bleed air connection on the firewall.

r. Remove the cover which was placed over the air inlet screen during ENGINE REMOVAL.

s. Install the engine air plenum and air inlet duct as outlined in Chapter 71-60-00.

t. Install the engine oil cooler and oil cooler supports as outlined in Chapter 79-00-00.

u. Install the clamps (11, 12, 16 and 18) and clip (17) that retain the control cable (10) to the engine bracket (22), engine flange, air plenum (14) (2 places) and engine truss mount (2 places). (Refer to Figure 4)

v. Connect the propeller control cable rod end (23) to the lever on the primary governor control (24) with the bolt (26) and washer (25). (Refer to Figure 4)

w. Attach the power control cable mounting bracket (2) to the engine with the nuts (9) on the LH side of the acces-

sory gearbox mounting pad. (Refer to Figure 4)

x. Connect the power control cable rod (3) to the lever extension (5) at the LH side of the accessory gearbox mounting pad. Secure the rod (3) to the lever extension (5) with the screw (4), washer (8), nut (6) and cotter pin (7). Attach the power control cable (1) to the engine truss mount with the clamps. (Refer to Figure 4)

y. Install the clamps that retain the condition control cable (27) to the engine truss mount. Attach the condition control cable bracket (31) to the engine with the two nuts (32). Connect the condition control rod end (37) to the lever (36) with the bolt (38), washer (33), nut (34) and cotter pin (35). (Refer Figure 4)

z. Rig the propeller reversing mechanism as outlined in Chapter 61-20-00.

aa. Rig the power control and condition control mechanisms as outlined in Chapter 76-00-00.

ab. Add oil to the engine as outlined in Chapter 12-10-00.

ac. Remove the drip pan from underneath the engine.

ad. Install the engine cowling as outlined in Chapter 71-10-00.

ae. Connect electrical power to the airplane as follows:

1. Connect the battery.

2. Reset the engine circuit breakers.

3. Turn the Master-Battery switch ON.

4. Press the LH "ENG FIRE F/W VALVE PUSH CLOSED" and RH "ENG FIRE F/W VALVE PUSH CLOSED" switches. These switches are located on the glareshield and will open the firewall shut off valves. Wait at least 30 seconds.

5. Turn the Master-Battery switch OFF.

ENGINE TEARDOWN

Engine teardown consists of the removal of all Raytheon Aircraft supplied components from the engine. These steps should be accomplished after the engine has been removed from the airplane and is installed on an engine stand. See ENGINE REMOVAL Chapter 71-00-00. Refer to the Engine Maintenance Manual for additional information and procedures.

NOTE

Tag and identify all hoses, plumbing and electrical connections to ensure correct installation. Cap or cover all open hoses, plumbing, engine ports and electrical connections to prevent contamination. Note all securing clamps and standoff locations for correct installation on the new engine.

a. Remove the wire harness from the engine as outlined in Chapter 71-50-00.

b. Remove the clamps that attach the ignition exciter to the engine truss mount and remove the ignition exciter from the engine truss mount as outlined in Chapter 74-00-00.

c. Remove the starter/generator and mounting adapter from the accessory gearbox mounting pad as outlined in Chapter 24-30-00.

d. Remove the turbine tach generator by unscrewing the nuts that retain it to the engine as outlined in Chapter 77-40-00.

e. Remove the autofeather pressure switches and the engine torque transducer from the torque pressure manifold as outlined in Chapter 61-21-00 and Chapter 77-40-00.

f. Disconnect the torque pressure manifold hoses (two) from the engine and remove the torque pressure manifold from the engine flange.

g. Remove the propeller overspeed governor from the engine as outlined in Chapter 61-20-00.

h. Remove the ground idle stop solenoid and mounting bracket from the engine as outlined in Chapter 76-00-00.

i. Remove the engine propeller tachometer generator from the reduction gearbox case as outlined in Chapter 77-40-00.

j. Remove the overboard oil breather tubing from the fitting on the forward side of the engine and from the fitting on the RH exhaust stack as outlined in Chapter 79-00-00.

k. Cut the safety wire and remove the nuts, bolts and washers that retain the exhaust stacks to the exhaust stack flanges.

l. Remove the fuel tubing and TEE that is located between the engine driven fuel boost pump and the oil-to-fuel heater as outlined in Chapter 73-10-00.

m. Remove the engine driven fuel boost pump as outlined in Chapter 73-10-00.

n. Disconnect and remove the propeller reversing interconnect linkage from the engine as outlined in Chapter 61-10-00.

o. Remove the oil drain line and fitting from the engine as outlined in Chapter 79-00-00.

p. Remove the air conditioning compressor and air conditioning compressor bracket from the engine as outlined in Chapter 21-50-00. (RH engine only).

q. Remove the engine from the engine stand as follows:

1. Attach a suitable engine sling to the forward and aft engine hoisting lugs. The forward engine hoisting lug is located on the upper forward side of the compressor inlet case. The aft engine hoisting lug is located on the upper side of the reduction gearbox to exhaust duct flange.

2. Attach an engine hoist to the eye of the engine sling.

3. Raise the engine hoist until the weight of the engine is supported by the engine hoist.

4. Remove the engine stand adapter brackets and engine stand from the engine.

r. Prepare the engine for return as outlined in ENGINE RETURN Chapter 71-00-00 and lower into the shipping container as outlined in the Pratt & Whitney PT6A-67A Engine Maintenance Manual.

ENGINE BUILDUP

Engine build-up consists of installing accessories and equipment on the new engine which were removed from the old engine. When a new engine is received it should be removed from the shipping container and installed in an engine stand as outlined below. Refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual for additional

information and procedures. Complete the engine build-up as follows:

a. After the engine has been removed from the shipping container, attach a suitable engine sling to the forward and aft engine hoisting lugs.

b. Connect an engine hoist to the eye of the engine sling.

c. Raise the engine hoist until the engine is at a level that will allow it to fit onto the engine stand.

d. Attach engine stand adapter brackets to the engine at the engine mount isolator holes.

e. Lower the engine into the engine stand and secure the engine stand adapter brackets to the engine stand.

f. Lower the engine hoist and remove the engine sling from the engine hoisting lugs.

g. Install the turbine tach generator as outlined in Chapter 77-40-00.

h. Install the air conditioning compressor and air conditioning compressor bracket as outlined in Chapter 21-50-00. (RH engine only).

i. Install the oil drain line and fitting on the engine as outlined in Chapter 79-00-00.

j. Connect the propeller reversing interconnect linkage to the engine as outlined in Chapter 61-10-00.

NOTE

Apply a light coating of thread sealant (1, Chart 1) to the male threads on all fuel hoses and connectors. See Chapter 73-10-00 for details on engine fuel hose installation.

k. Install the engine-driven fuel boost pump as outlined in Chapter 73-10-00.

l. Connect the fuel tubing and TEE that runs between the engine-driven fuel boost pump and the oil-to-fuel heater.

m. Attach the exhaust stacks to the exhaust stack flanges on the gas generator case with the nuts, washers and bolts. Safety wire the bolts that retain the exhaust stacks to exhaust stack flanges on the gas generator case.

n. Install the engine propeller tachometer generator as outlined in Chapter 77-40-00.

o. Install the overboard oil breather tubing as outlined in Chapter 79-00-00.

p. Install the ground idle stop solenoid and mounting bracket as outlined in Chapter 76-00-00.

q. Install the propeller overspeed governor as outlined in Chapter 61-20-00.

r. Install the torque pressure manifold onto the engine flange (aft of the RH exhaust stack) and connect the hoses (two) on the torque pressure manifold to the engine.

s. Install the autofeather pressure switches and the engine torque transducer on the torque pressure manifold as outlined in Chapter 61-21-00 and Chapter 77-40-00.

t. Install the starter/generator as outlined in Chapter 24-30-00.

u. Install the low oil pressure switch as outlined in Chapter 79-30-00.

v. Install the fuel flowmeter transmitter, oil temperature bulb and oil pressure transducer as outlined in Chapter 77-40-00.

w. Attach a suitable engine sling to the engine hoisting lugs. The forward engine hoisting lug is located on the upper forward side of the compressor inlet case. The aft engine hoisting lug is located on the upper side of the reduction gearbox to exhaust duct flange.

x. Place a suitable engine hoist directly above the eye of the engine sling.

y. Attach the engine hoist to the engine sling and raise the engine hoist until the weight of the engine is supported by the engine sling.

z. Remove the engine stand adapter brackets from the engine stand.

aa. Remove the engine stand adapter brackets from the engine.

ab. Install the five engine mount isolators on the engine with the four engine bolts. Use new gaskets (one per isolator) and torque the engine bolts to 225 to 300 inch-pounds. Safety wire the engine bolts.

ac. Adjust the position of the engine truss mount and engine hoist so that the holes in the engine truss mount line up with the holes in the engine mount isolator.

ad. Assemble the isolators and install the isolator bolts in the isolators and secure as outlined in ENGINE MOUNT ISOLATOR INSTALLATION, Chapter 71-20-00. Torque the nut on the isolator bolt to 450 to 500 inch-pounds. Secure the nut with a new cotter pin.

ae. Install the lower aft cross tube on the engine truss mount.

af. Attach the ignition exciter to the to the truss mount with the clamps as outlined in Chapter 74-00-00.

ag. Attach the engine electrical wire harness to the engine components with the connectors and clamps as outlined in Chapter 71-50-00.

ENGINE RETURN

When an engine is returned, the following steps must be accomplished, and the following items should be enclosed with the engine before it is shipped.

NOTE

Refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual for specific instructions on preparing the engine for shipping.

a. All openings, bosses and lines must be plugged, capped or covered.

b. All electrical connections must be covered.

c. The propeller hub must covered.

d. The exhaust ports and engine air inlet screen must be covered.

e. All linkage and loose parts must be secured to the engine to prevent movement.

f. The ignition exciter must be returned with the engine.

g. The engine log book must be returned with the engine.

h. Use the new engine shipping container to return the old engine. Be sure the old engine is secured to the shipping

bracket.

FUEL CONTROL UNIT PURGE AND SOAK PROCEDURE

CAUTION

Throughout the following procedure, observe the starter operating limits of 30 seconds ON, 5 minutes OFF, 30 seconds ON, 5 minutes OFF, 30 seconds ON, then 30 minutes OFF.

a. To purge the fuel control unit of preservation oil, install the unit on the engine and connect it to the airplane fuel supply as outlined in Chapter 73-30-00. Install a suitable line between the fuel control outlet and a clean container.

b. With the IGNITION/STARTER switch in the STARTER ONLY position and the fuel CONDITION control lever in the START position, motor the engine until clean fuel flows from the fuel outlet. During the motoring run, move the POWER lever from IDLE to full power and back to IDLE. The fuel CONDITION control lever should be moved from the START position to the FUEL CUTOFF position to ascertain fuel shutoff.

c. Remove the line used to purge the fuel control unit and connect the engine plumbing to the fuel control outlet as outlined in Chapter 73-30-00. Check the installation for leakage.

CAUTION

Before attempting a start, motor the engine with the fuel shut off to ensure that all fuel has been purged from the engine.

HANDLING FUEL NOZZLES

Extreme care should be exercised when handling fuel nozzles, even finger prints on the orifice may produce a poor spray pattern. If fuel nozzles are not to be installed on the engine immediately, they should be placed in a covered container to prevent exposure to dirt. If fuel nozzles are to be shipped or stored, they must be separately and securely wrapped. If the fuel nozzles are packed together in a container that allows relative movement, the fuel orifice faces will be damaged.

ENGINE WASHING PROCEDURES

CAUTION

Before any external washing, cover or protect all electrical components located on the engine or in the engine compartment. Do not allow water to enter the engine air inlet or exhaust. Precautions should be taken against contaminating the clutch portion of the air conditioning compressor with cleaning solvent during the engine wash procedure. Such contamination may dilute the grease in the compressor pulley bearing and result in premature failure. High pressure water or solvent should not be directed on electrical components or mechanical parts having air vent holes. After washing the engine, blow dry the components with dry shop air.

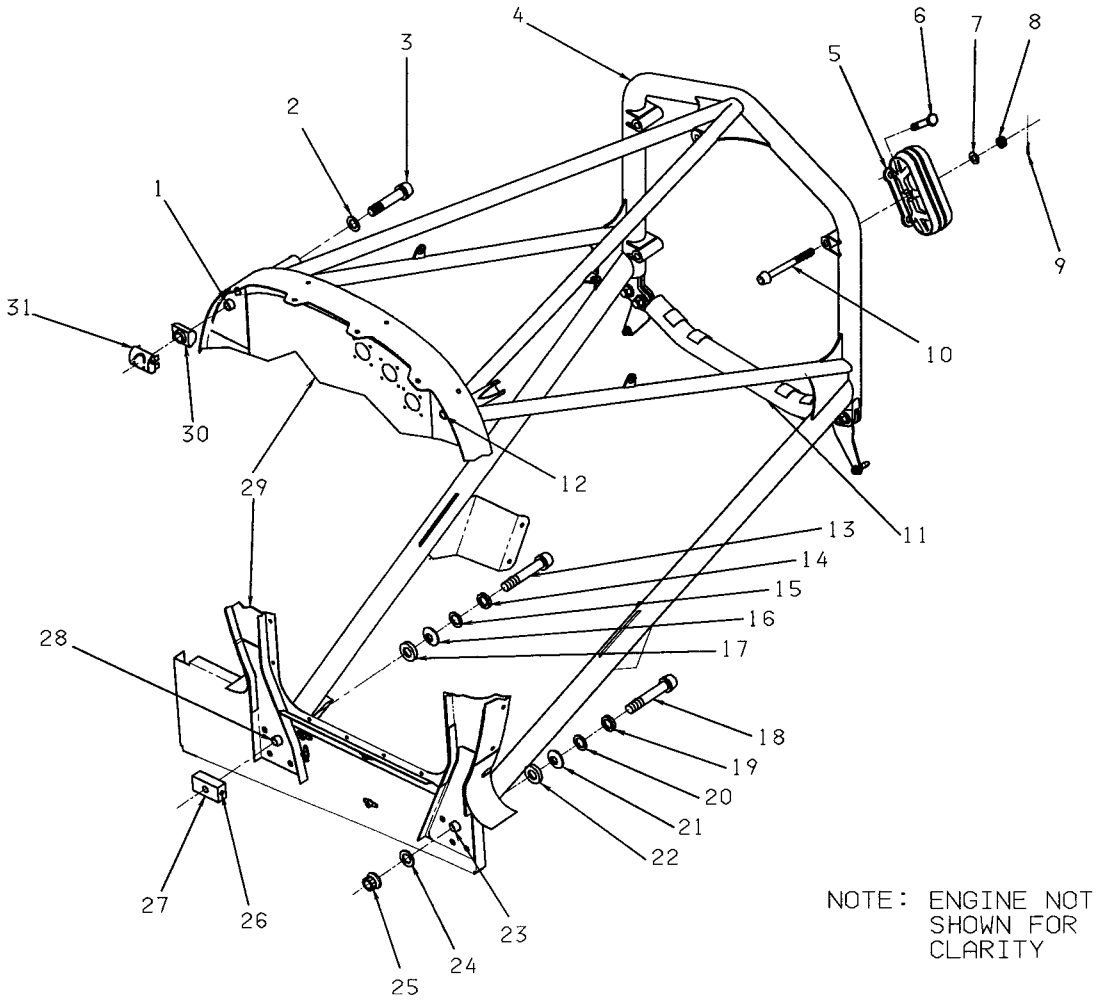
CAUTION

Never wash an engine while it is hot or running. After the engine has been shut down, allow it to cool with the cowling removed for at least one hour prior to washing.

For detailed engine washing procedures recommended by the engine manufacturer, refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual and the Pratt & Whitney Aircraft Gas Turbine Operation Letter No. 7, Rev. 2, dated January 23, 1976, or subsequent by Pratt & Whitney of Canada Limited.

ENGINE SURFACE SALT CONTAMINATION

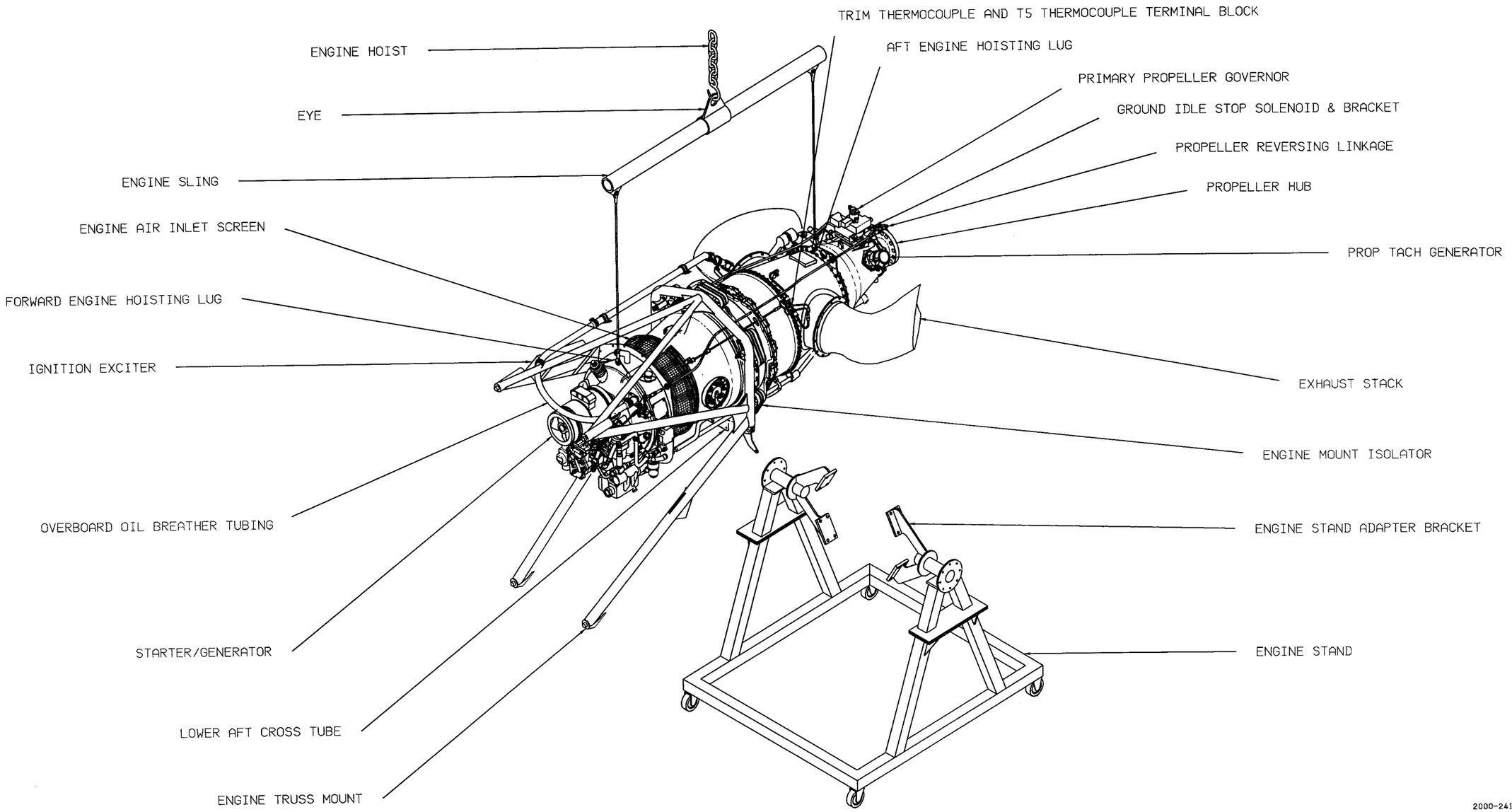
When the exterior surface of the engine is contaminated with salt, it should be washed clean with water prior to flight. Demineralized water is not required for this purpose. At no time should an engine be left in a contaminated (salted) condition for any extended period of time (such as overnight).



1. UPPER INBOARD BOLT HOLE
2. COUNTERSUNK WASHER
3. UPPER INBOARD MOUNTING BOLT
4. ENGINE TRUSS MOUNT
5. ENGINE MOUNT ISOLATOR
6. ENGINE BOLT
7. WASHER
8. CASTELLATED NUT
9. COTTER PIN
10. ISOLATOR BOLT
11. LOWER AFT CROSS TUBE
12. UPPER OUTBOARD BOLT HOLE
13. LOWER INBOARD MOUNTING BOLT
14. COUNTERSUNK WASHER (AS REQUIRED)
15. FLAT WASHER (AS REQUIRED, MAXIMUM OF THREE)
16. CONVEX WASHER
17. CONCAVE WASHER
18. LOWER OUTBOARD MOUNTING BOLT
19. COUNTERSUNK WASHER (AS REQUIRED)
20. FLAT WASHER (AS REQUIRED, MAXIMUM OF THREE)
21. CONVEX WASHER
22. CONCAVE WASHER
23. LOWER OUTBOARD BOLT HOLE
24. FLAT WASHER
25. LOCKING NUT
26. INSPECTION HOLE
27. BAR NUT
28. LOWER INBOARD BOLT HOLE
29. FIREWALL
30. BARREL NUT
31. BARREL NUT RETAINER

2000-241-19

Engine Truss Mount Removal and Installation (LH Side)
Figure 1



Engine Illustration (LH View)
Figure 2

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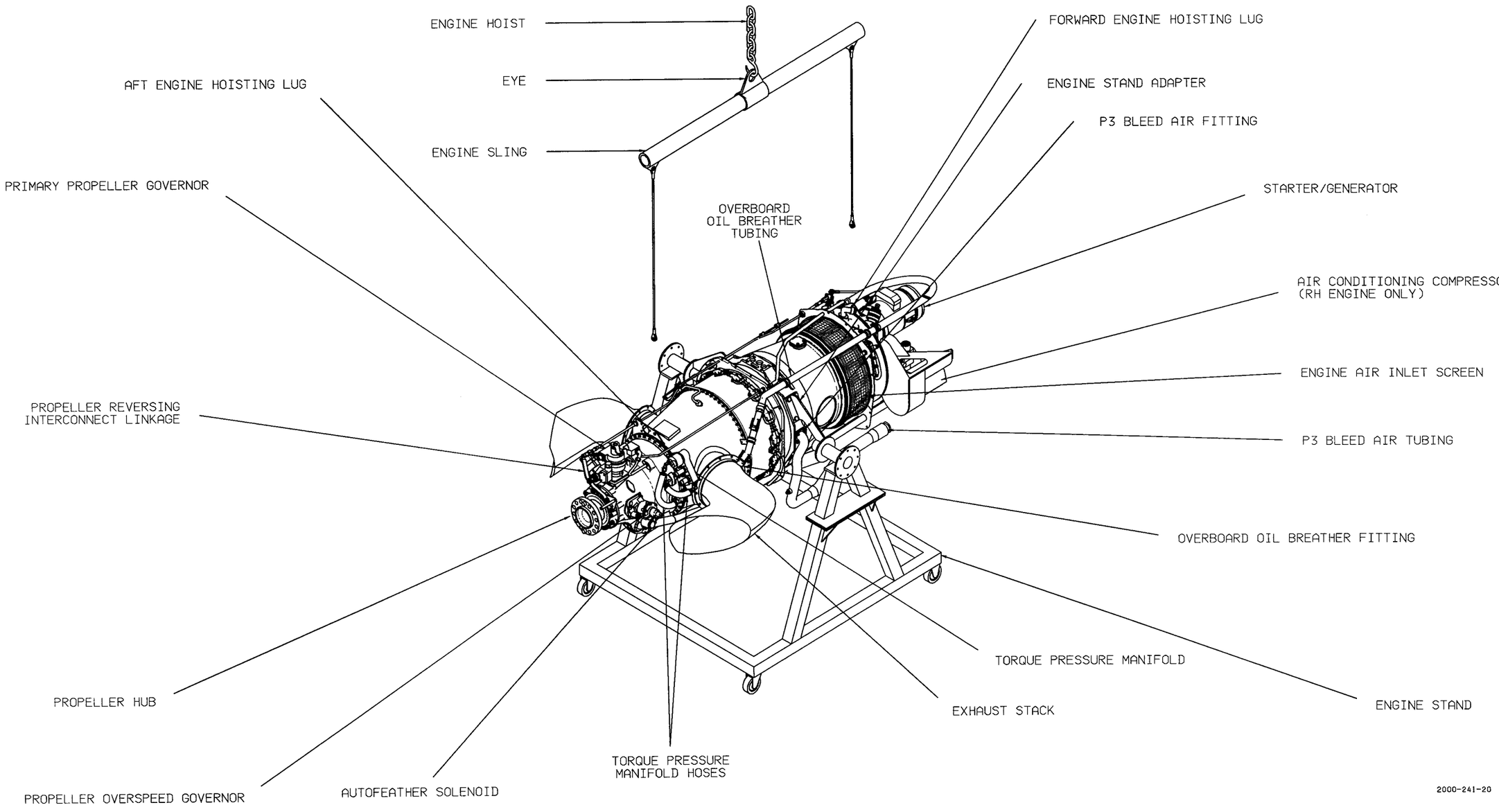

Engine Illustration (RH View)
Figure 3

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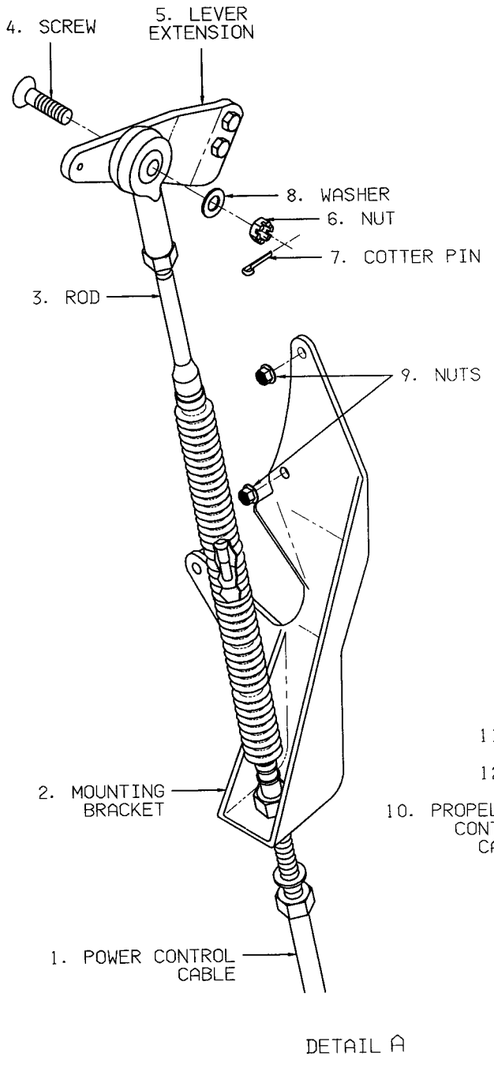
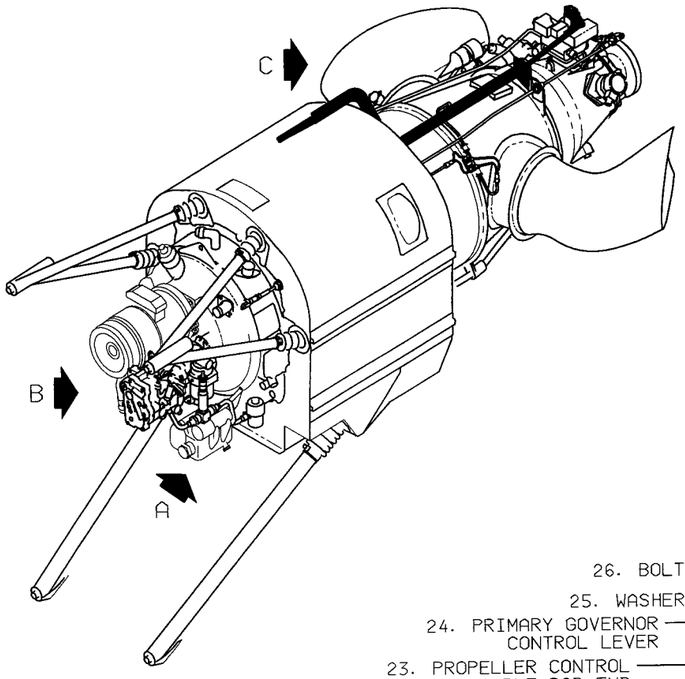
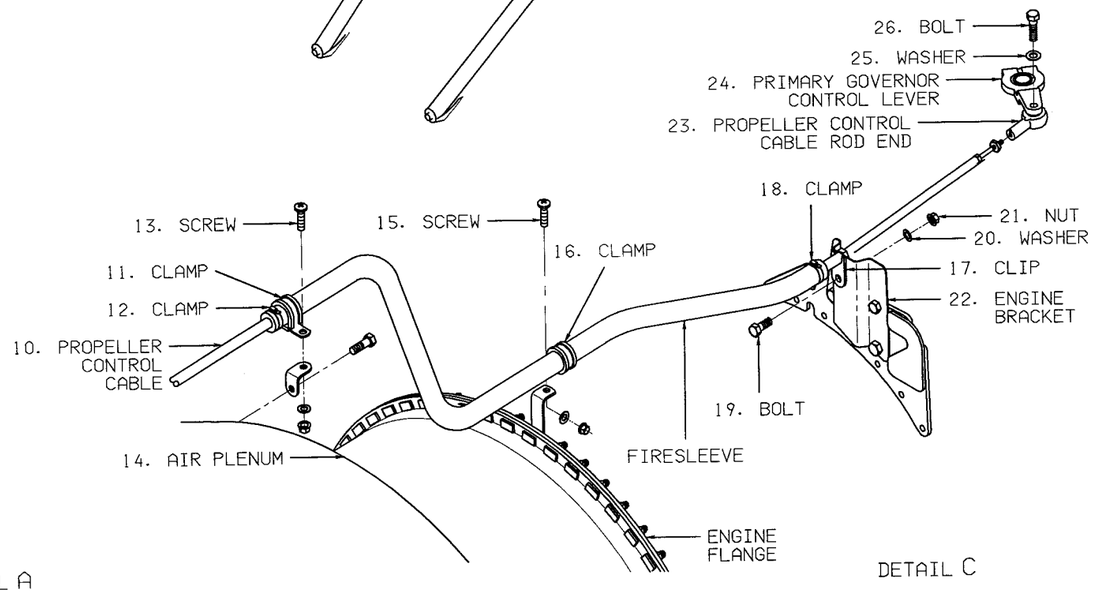
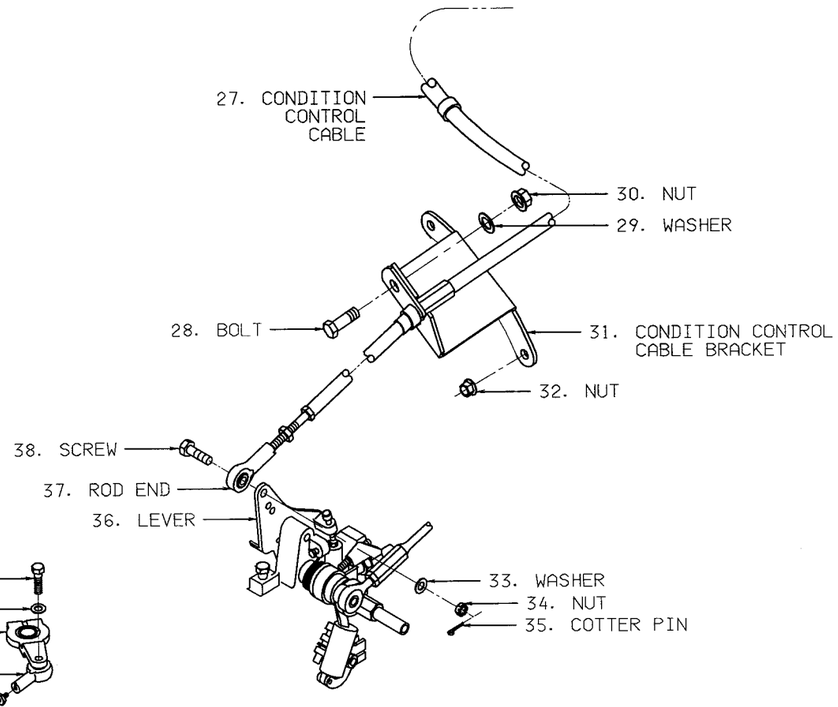


DETAIL A





DETAIL C



DETAIL B

2000-241-21

Engine Controls Removal and Installation
Figure 4

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ENGINE COWLING - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The engine cowling encloses each power plant and consists of four major cowling assemblies, three fairings, two closeouts, and a removable starter/generator air scoop. Two hinged access panels are incorporated into the engine cowling; one in the upper forward cowling and the other in the lower fairing. There are two air scoops attached to the engine cowling, a forward oil cooler intake scoop and a starter/generator air scoop. The oil cooler intake scoop is an integral part of the lower forward cowling. Air flows into the scoop and is ducted upward through the oil cooler. The air is then routed aft from the oil cooler thru an air exit duct and into the propeller wake. The starter/generator air scoop is fastened to the LH side of the lower forward cowling. Air flows into the scoop and is ducted to the starter/generator assembly.

The cowling is constructed of layers of woven graphite fibers that are impregnated with resin. The woven graphite fibers are laminated together, laid up in a mold and cured in an autoclave. Some areas of the cowling have a honeycomb core sandwiched between the layers of woven graphite. Each major cowling assembly has one layer of aluminum mesh woven with the graphite for lightning strike protection.

When removing the cowling, the proper removal sequence should be followed: upper forward cowling, upper aft cowling, lower aft cowling, fairings, starter/generator air scoop, forward oil cooler duct, and lower forward cowling. The upper forward and upper aft cowling assemblies can be independently removed by disengaging the captive fasteners that retain them.

The upper forward cowling is fastened to the lower forward cowling and has a hinged access panel (for access to the oil filler inlet) installed on the upper side of the cowling. The upper aft cowling is attached to the lower aft cowling. The lower aft cowling is fastened to the lower forward cowling and has an oil cooler air exit duct bonded to it. The lower forward cowling is mounted to studs on the engine air plenum and fastened to the nacelle. It has a removable air intake scoop for the starter/generator fastened to the LH side. The lower forward cowling has an integral oil cooler intake scoop which has an air intake duct attached to it. There are three fairings (lower, upper inboard and upper outboard) and two closeouts (inboard and outboard) attached to the lower forward cowling. The fairings and forward oil cooler air intake duct should be removed before attempting to remove the lower forward cowling.

ENGINE COWLING - MAINTENANCE PRACTICES (Effectivity: All)

ENGINE COWLING REMOVAL (Effectivity: All) (Figure 2)

a. Remove electrical power from the airplane and disconnect the battery.

b. Release the 20 fasteners that retain the upper forward cowling to the lower forward cowling.

NOTE

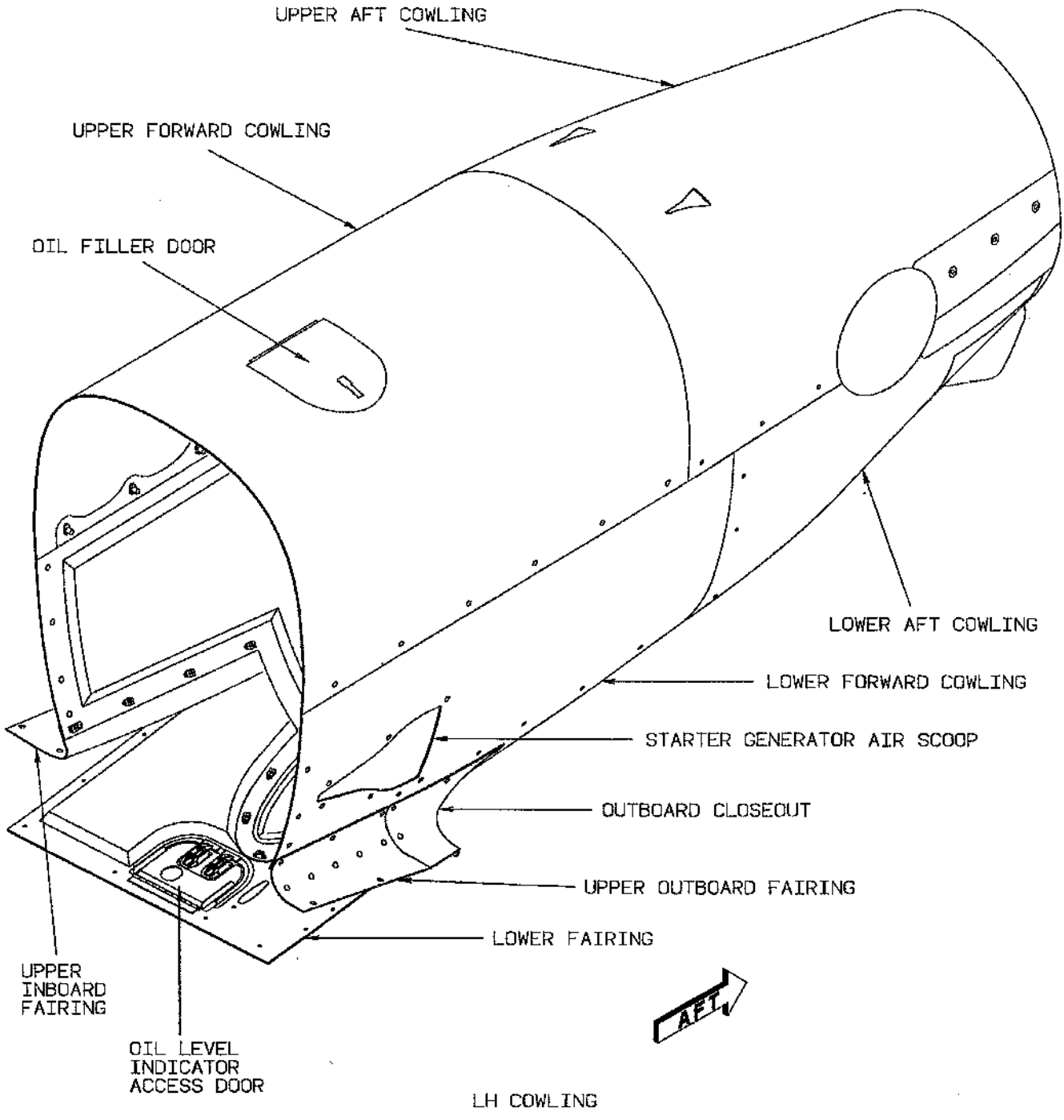
Save all attaching hardware for installation.

c. Remove the upper forward cowling by lifting the upper forward cowling from the lower forward cowling.

d. Release the 12 fasteners that retain the upper aft cowling to the lower aft cowling.

e. Remove the upper aft cowling by lifting the upper aft cowling from the lower aft cowling.

f. While supporting the lower aft cowling, release the 9 fasteners that retain the lower aft cowling.



Engine Cowling (Effectivity: All)
Figure 1

NOTE

Position the propeller so the top blade is straight up and down. If difficulty is encountered in removing the lower aft cowling, additional clearance can be obtained by removing the propeller spinner. See Chapter 61-10-00 propeller assembly.

g. Move the lower aft cowling aft approximately 2 inches. Remove the lower aft cowling by lowering it from the nacelle.

h. Release the fasteners that retain the flap position sensor cover and remove the flap position sensor cover. The flap position sensor is located adjacent to the cowling on the inboard side of the rib next to the inboard flap.

i. Release the 2 fasteners that retain the drain assembly to the lower fairing.

j. Release the 20 fasteners that retain the lower fairing to the lower forward cowling and nacelle.

k. Remove the lower fairing by sliding it aft from the nacelle.

l. Release the 9 fasteners that retain the upper outboard fairing to the cowling.

m. Remove the upper outboard fairing.

n. Release the 12 fasteners which retain the upper inboard fairing to the cowling.

o. Remove the upper inboard fairing.

p. Release the 11 fasteners which retain the outboard closeout.

q. Remove the outboard closeout.

r. Release the 11 fasteners which retain the inboard closeout.

s. Remove the inboard closeout.

t. Disconnect the starter/generator air scoop by removing the 6 fasteners that retain it to the inside of the lower forward cowling.

u. Remove the 8 bolts that secure the forward oil cooler duct to the aft oil cooler duct.

v. Remove the forward oil cooler duct from the nacelle.

w. Release the 8 fasteners that attach the lower forward cowling to the nacelle.

x. Remove the lower forward cowling by moving it aft and lowering it from the nacelle.

ENGINE COWLING INSTALLATION
(Effectivity: All)
(Figure 2)

a. Slide the lower forward cowling onto the studs on the engine air intake housing and align the holes in the lower forward cowling with the holes in the nacelle.

b. While supporting the lower forward cowling, position and install the forward oil cooler duct to the aft oil cooler duct with the 8 bolts.

WARNING

Be sure that all fasteners and bolts are properly secured. Any attachment hardware that becomes loose will go through the propeller, causing serious damage to the propeller. Hardware thrown from the propeller can damage the airplane and objects in the surrounding area.

c. Attach and secure the lower forward cowling to the nacelle with the 8 fasteners.

d. Attach the engine drain assembly to the lower fairing and secure with the 2 fasteners.

e. Install the lower fairing assembly and secure with the 20 fasteners.

f. Attach the inboard closeout to the cowling and secure with the 11 fasteners.

g. Attach the outboard closeout to the cowling and secure with the 11 fasteners.

h. Attach the upper inboard fairing to the cowling and secure with the 12 fasteners.

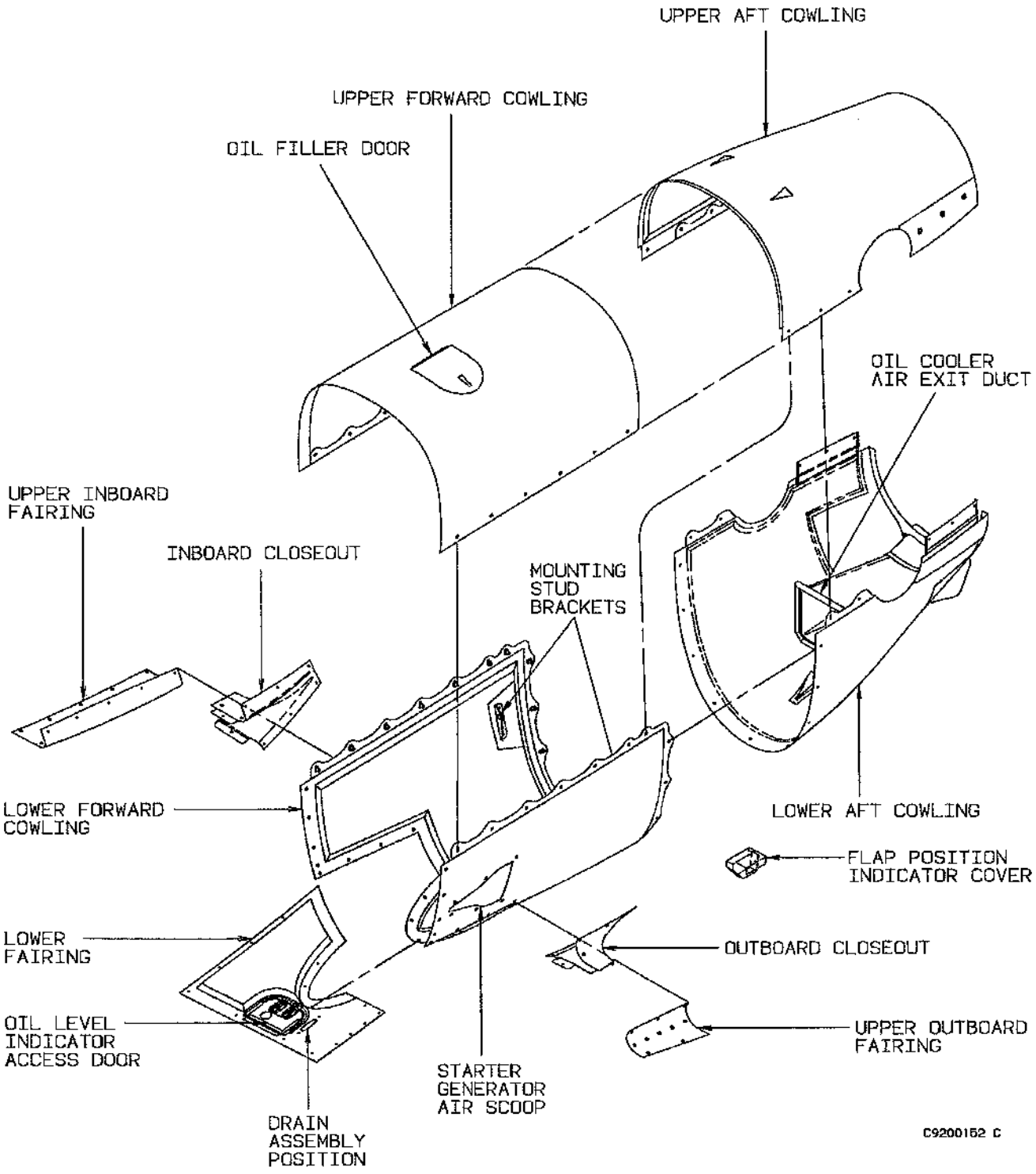
i. Attach the upper outboard fairing to the cowling and secure with the 9 fasteners.

j. Install the flap position sensor cover and secure with the fasteners. The flap position sensor is located adjacent to the cowling on the inboard side of the inboard flap.

k. Install the starter/generator air scoop on the inside of the lower forward cowling and secure with the 6 fasteners.

l. Align the holes in the lower aft cowling with the holes in the lower forward cowling.

m. Attach the lower aft cowling to the lower forward cowling and secure with the 9 fasteners.



Engine Cowling Installation - Exploded View (Effectivity: All)
Figure 2

n. Align the holes in the upper aft cowling with holes in the lower aft cowling.

o. Attach the upper aft cowling to the lower aft cowling and secure with the 12 fasteners.

p. Align the holes in the upper forward cowling with the holes in the lower forward cowling.

q. Attach the upper forward cowling to the lower aft cowling and secure with the 20 fasteners.

r. Connect the battery and restore electrical power to the airplane.

ENGINE MOUNTS - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

ENGINE TRUSS MOUNTS (Effectivity: All) (Figure 1)

The engine truss mount consists of a welded tubular framework which surrounds the forward section of the engine and attaches the engine to the firewall. The engine truss mounts are constructed of welded 4130 chrome molybdenum tubular steel that has been stress relieved, sandblasted and painted with an aluminum finish.

The engine is attached to the engine truss mount at five points on the engine case. Engine mount vibration isolators are installed between the engine and engine truss mount. The engine truss mount is bolted to the firewall at four positions: two on the aft wing spar, and two on the upper aft side of the nacelle. Access to the engine truss mounting hardware is gained by removing the engine cowling and upper aft nacelle panel, and by reaching through an access panel on the underside of the aft wing.

The engine truss mounts provide a secure structure to attach the engines to the airplane. Mounting flanges and brackets are welded to the engine truss mounts and allow attachment of the engine cowling, air plenum, air intake duct and oil cooler. The engine truss mounting bolts should be torqued to 450 to 500 inch-pounds when the engine truss mounts are installed.

ENGINE MOUNT ISOLATORS (Effectivity: All) (Figure 2)

Five engine mount isolators are used to secure the engine to the engine truss mount and dampen engine vibration to the airframe structure. Each engine mount isolator is attached to the engine with four bolts, and anchored to the engine truss mount with one bolt. Individual engine mount isolators may be replaced or overhauled without removing the engine from the engine truss mount, provided the weight of the engine is adequately supported. All engine mount isolators on an engine must be of the same manufacturer and type.

ENGINE MOUNTS - MAINTENANCE PRACTICES (Effectivity: All)

Maintenance practices for the engine mounts consist of inspection of the engine mount isolators, removal and installation of the engine mount isolators and engine truss mounts, and reconditioning of the engine truss mounts. Although it is possible to remove the engine from the engine truss mount before removing the engine truss mount from the firewall, it is recommended that the engine and engine truss mount be removed from the firewall together. This will facilitate maintenance on the engine. See Chapter 71-00-00 for engine truss mount removal and installation procedures.

ENGINE MOUNT ISOLATOR REMOVAL (Effectivity: All) (Figure 2)

a. Remove electrical power from the airplane and disconnect the battery.

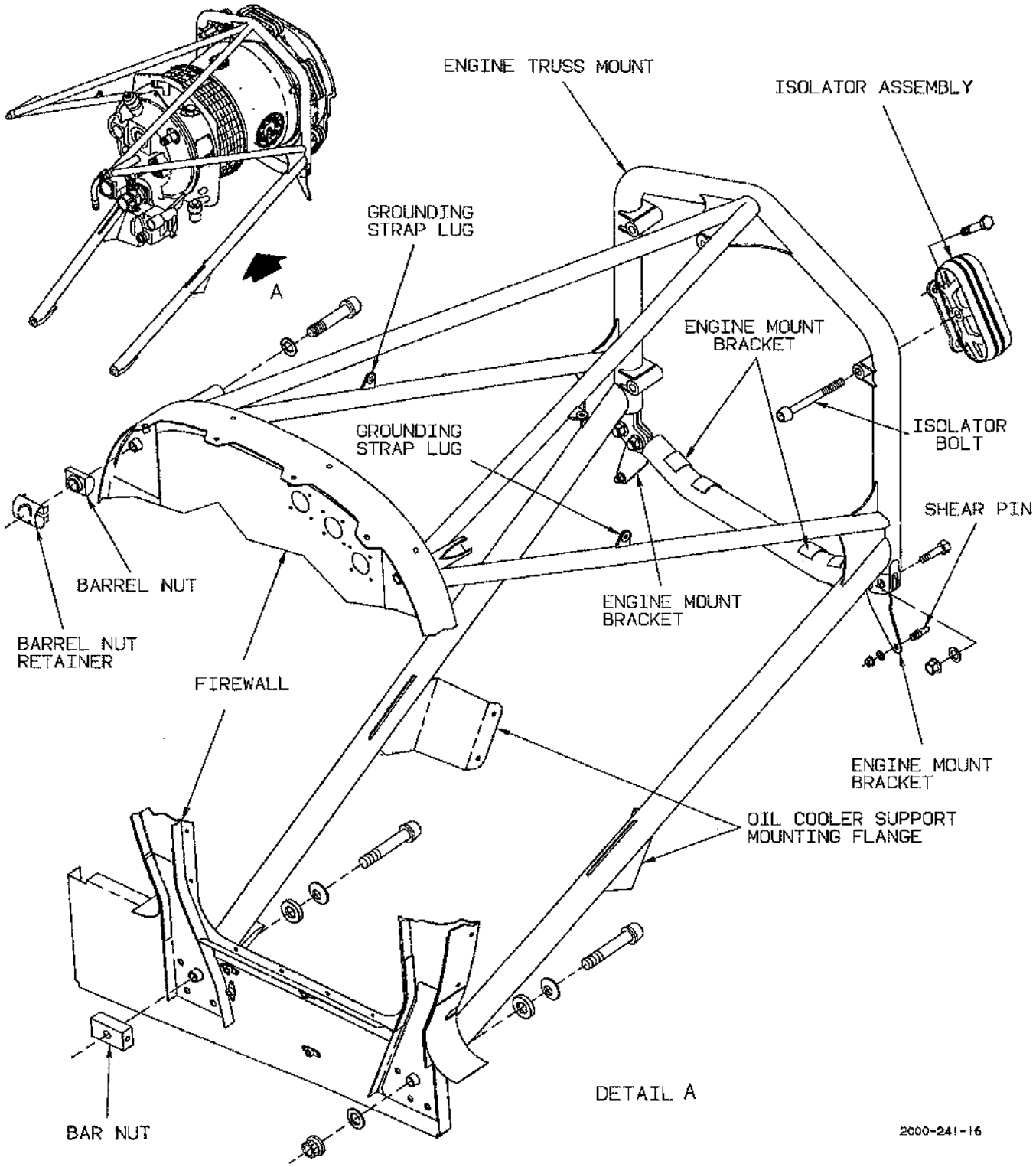
b. Remove the engine cowling as outlined in Chapter 71-10-00.

c. Remove the air plenum and air inlet duct as outlined in Chapter 71-60-00.

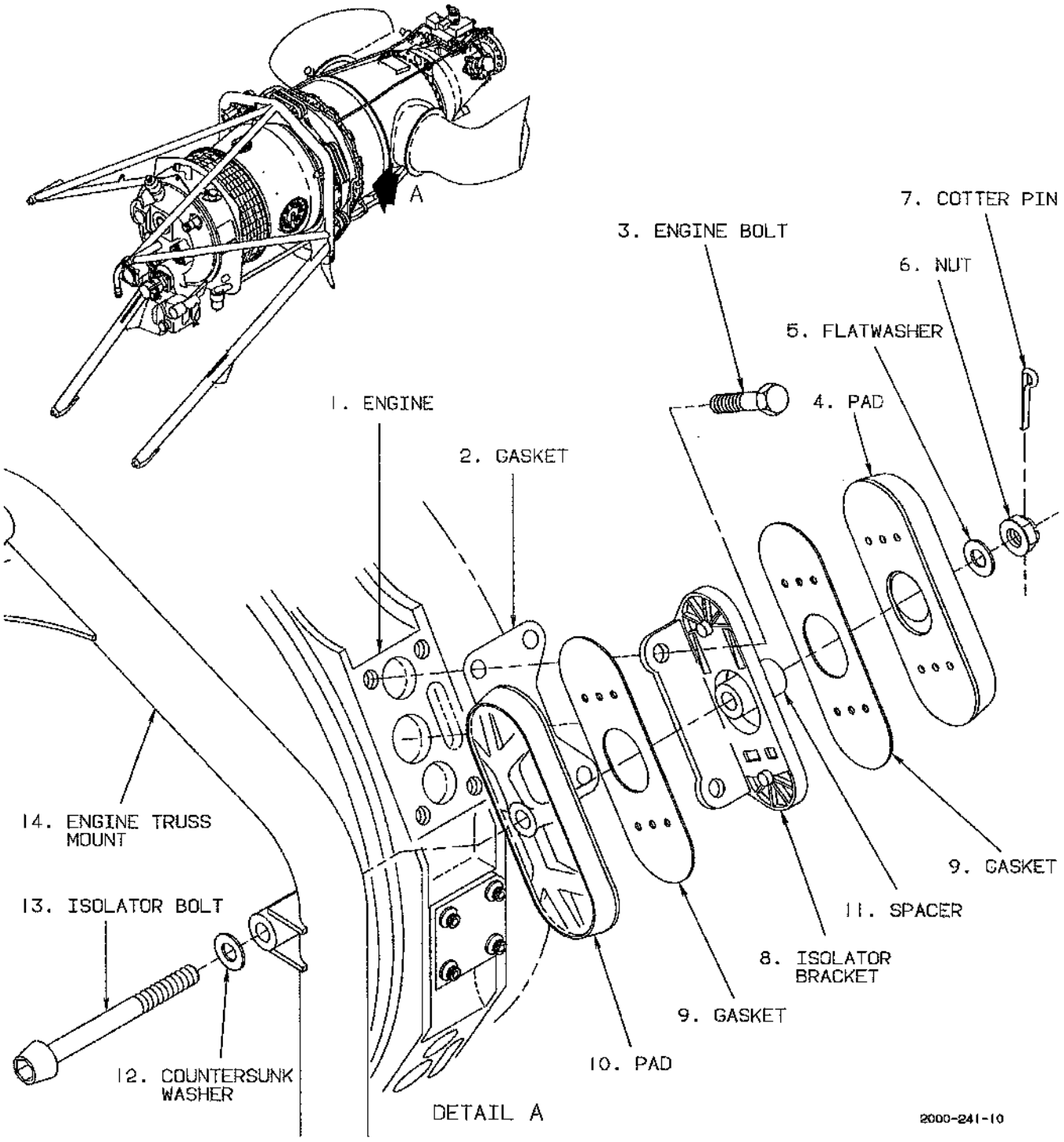
d. Attach an engine sling to the engine hoisting lugs. The forward engine hoisting lug is located on the upper forward side of the compressor inlet case. The aft engine hoisting lug is located on the upper side of the reduction gearbox to exhaust duct flange.

e. Position a suitable engine hoist directly above the eye of the engine sling.

f. Raise the engine hoist until the weight of the engine is supported by the engine sling.



Engine Truss Mount (Effectivity: All)
Figure 1



Engine Mount Isolator (Effectivity: All)
Figure 2

g. Cut the safety wire and remove the four engine bolts (3) that attach each engine mount isolator to the engine (1). Discard the gasket (2).

NOTE

Retain the engine bolts (3), spacer (11), nut (6), washers (5 and 12) and isolator bolt (13) from each engine mount isolator for installation.

h. Remove the cotter pin (7), nut (6), pads (4 and 10), spacer (11), washers (5 and 12) and isolator bolt (13) from the isolator bracket (8). Discard the gaskets (9).

i. Remove the isolator bracket (8) from the engine truss mount (14).

ENGINE MOUNT ISOLATOR INSTALLATION (Effectivity: All) (Figure 2)

CAUTION

All five isolators on an engine must have the same part number. The LH engine and RH engine on an airplane may have different isolator part numbers, but the isolator part numbers on a particular engine MUST HAVE THE SAME PART NUMBER. All isolators must have approved part numbers.

NOTE

It is often possible to overhaul, rather than replace the engine mount isolators. Frequently, all that is required to increase the service life of the isolator is to replace the rubber (elastomer) pads on the isolator. See Chapter 71-20-00 ISOLATOR INSPECTION CRITERIA for conditions governing isolator overhaul.

a. Attach the isolator bracket (8) to the engine with the four engine bolts (3). Use a new gasket (2) and torque the engine bolts (3) to 225 to 300 inch-pounds. Safety wire the engine bolts (3).

b. Adjust the position of the engine (1) so that the holes in the engine truss mount (14) line up with the holes in the isolator brackets (8).

c. Slide the spacer (11) into the isolator bracket (8) and assemble the engine mount isolator by placing a gasket (9) and pad (4 and 10) on each side of the isolator bracket (8).

d. Install the isolator bolt (13) from the forward side of the engine truss mount (14) with the countersunk washer (12) under the head of the isolator bolt (13). Slide the isolator bolt (13) into the engine truss mount (14), through the pads (4 and 10), spacer (11) and isolator bracket (8). Install the flat washer (5) and nut (6) on the isolator bolt (13) from the aft side.

e. Torque the nut (6) on the isolator bolt (13) to 450 to 500 inch-pounds. Secure the nut (6) with a new cotter pin (7).

f. After all five engine mount isolators are installed, lower the engine hoist and remove the engine sling from the engine hoisting lugs.

g. Install the air plenum and air inlet duct as outlined in Chapter 71-60-00.

h. Install the cowling as outlined in Chapter 71-10-00.

i. Connect the battery and restore electrical power to the airplane.

ENGINE TRUSS MOUNT RECONDITIONING (Effectivity: All) (Figure 1)

Normal reconditioning of the welded 4130 chrome molybdenum steel engine truss mounts is as follows:

NOTE

Some cracks may be repaired by approved heliarc welding. The Beech Aircraft Customer Support Department should be consulted to determine on an individual basis if welding is possible or feasible.

a. After removing the engine truss mount and engine from the firewall (see Chapter 71-00-00 ENGINE REMOVAL), strip the paint with paint stripper (2, Chart 1, Chapter 71-00-00), sandblast and magnaflux inspect the engine truss mount for cracks.

b. If cracks are to be welded, preheat the section to be repaired to 400 to 800 degrees Fahrenheit and post-heat after repair to 700 to 800 degrees Fahrenheit for stress relief.

c. Check the tubes for straightness. All tubes must be straight within .02 inch per foot and .05 inch through the total tube length. Measurements should be made within .50 inch of the welds.

d. Clean the engine truss mount for painting. Cover all bushings to prevent paint from adhering to the the inside of the bushing ends.

e. Prime the engine truss mount with epoxy primer (3, Chart 1, Chapter 71-00-00) per MIL-P-23377 and finish with aluminum epoxy paint (4, Chart 1, Chapter 71-00-00) per MIL-C-16173. Treat all unpainted areas per MIL-C-16173.

f. When installing the engine truss mount, refer to the ENGINE INSTALLATION sequence outlined in Chapter 71-00-00.

ISOLATOR INSPECTION CRITERIA (Effectivity: All) (Figure 3)

The following inspection criteria should be used for evaluating engine truss mount vibration isolators. If an isolator pair does not meet a single criterion for replacement, but shows a large percentage of deterioration in several places, the isolator should be changed or overhauled. Overhauling the isolator consists of replacing the elastomer pads.

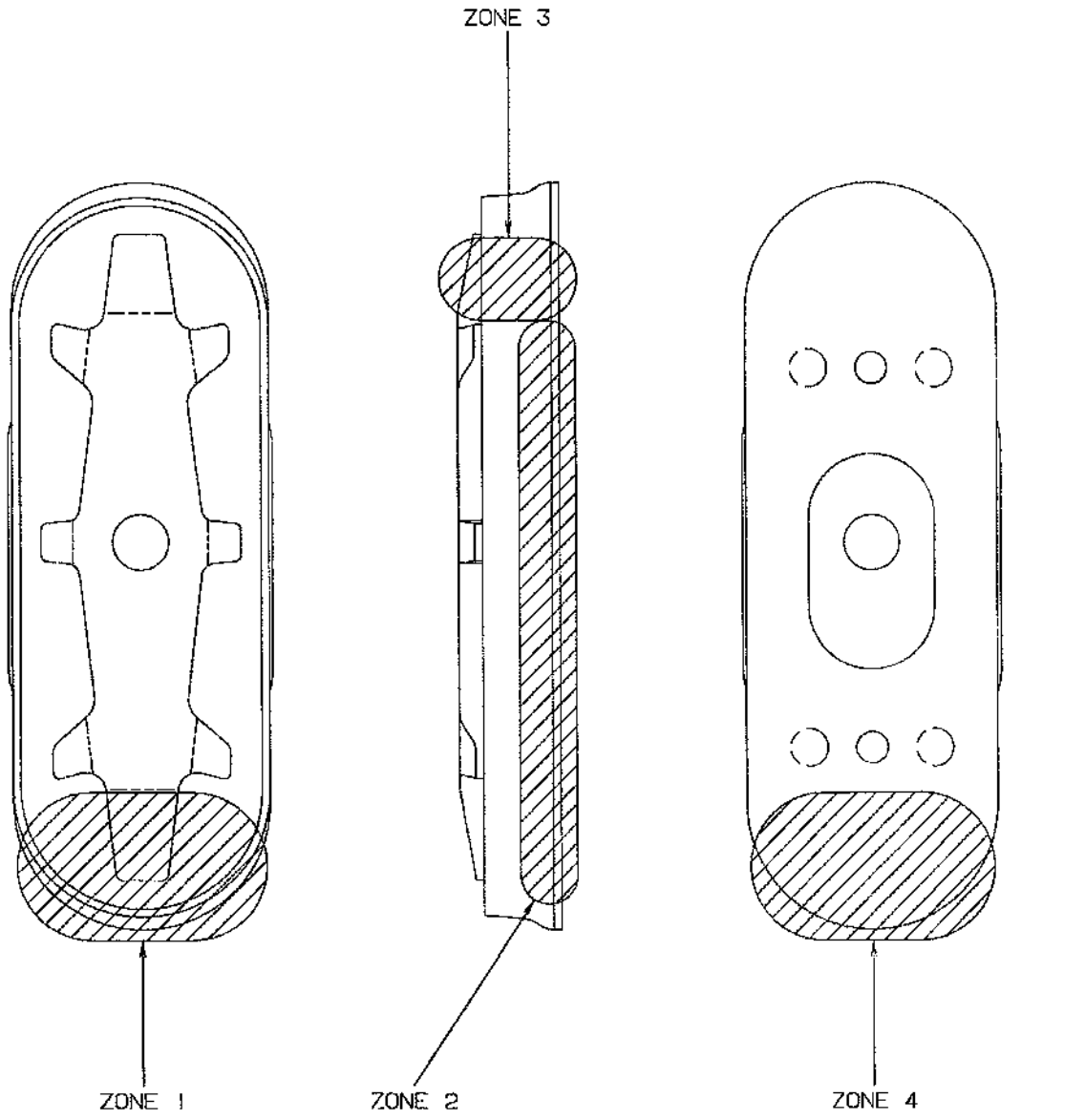
It is important to realize that deterioration due to repeated torque loadings occurs gradually. The presence of cracks in the zones listed does indicate some deterioration, but the rate of decrease in stiffness does not accelerate appreciably until a very high percentage of the elastomer/plate interface has visibly separated. The isolator must be overhauled or replaced if any one of the four following conditions exist.

a. Figure 3, Zone 1: The elastomer separation from the outer (cast) plate must not exceed 75% of the region along the radius of the isolator (shaded area).

b. Figure 3, Zone 2: The elastomer separation from the inner (stamped) plate, or cracks in the elastomer along the length of the elastomer/plate interface must not exceed 75% of this distance.

c. Figure 3, Zone 3: Any transverse tears (plate to plate) of the elastomer thickness. Zone 3 is the most likely place for this to occur.

d. Figure 3, Zone 4: If tears to any degree exist in Zones 1, 2, or 3, and if elastomer separation from the inner plate approaches 100% of the area along the radius of the isolator, the isolator must be replaced.



Engine Mount Isolator Inspection Criteria (Effectivity: All)
Figure 3

ELECTRICAL HARNESS - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The wire harness conducts electrical signals between the engine and the wiring located forward of the firewall. The wire harness connects the sensors, switches, solenoids and transmitters in the engine compartment to the control and monitoring systems in the cockpit.

The wire harness consists of multiple insulated wires that are tied together and wrapped with heat resistant tape. The tape is used to bundle the wires together and insulate them from high temperature. A metal overbraid is installed around the wires connected to the starter/generator, ignition exciter, thermocouple sensor and fire detection responder-sensor. The purpose of the metal overbraid is to provide lightning strike protection for the wires and components. The metal overbraid is grounded to positions close to the components and to the wire harness strain relief connections at the firewall.

The wire harness is located inside the engine compartment and is routed from the firewall to the aft RH side of the engine, near the propeller governor. The wire harness is connected to the No. 1 and No. 2 firewall connector plugs. The wiring is routed from the connectors at the firewall to the engine data concentrator to be digitized and sent forward to the data acquisition unit for display in the cockpit. Clamps are used to attach the wire harness to the engine truss mount, air plenum, oil cooler bracket and engine case. The wire harness runs below the air plenum to the LH side of the engine case, where it is routed aft of the exhaust stack and under the reduction gearbox case to the RH side of the reduction gearbox case. The wire harness is terminated at the aft RH side of the reduction gearbox case where it is connected to the propeller governor solenoid.

Bayonet and screw-on type plugs are used to connect the wire harness to the engine components. The wire harness is attached to the ignition exciter with a screw-on connector. The idle stop solenoid lead wire is attached to the wire harness with a spliced connection. This spliced connection uses pin and socket connectors that are wrapped with shrink tubing. The spliced connections must be disconnected before the wire harness can be removed. When the engine and engine mount are removed from the firewall together, the wire harness can easily be disconnected at the firewall with the No. 1 and No. 2 firewall plugs.

ELECTRICAL HARNESS - MAINTENANCE PRACTICES (Effectivity: All) (Figure 1)

WIRE HARNESS REMOVAL (Effectivity: All)

a. Remove electrical power from the airplane and disconnect the battery.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

NOTE

Be sure to tag all plugs and connectors prior to removal.

c. Cut the safety wire and disconnect the No. 1 and No. 2 firewall connector plugs from the firewall.

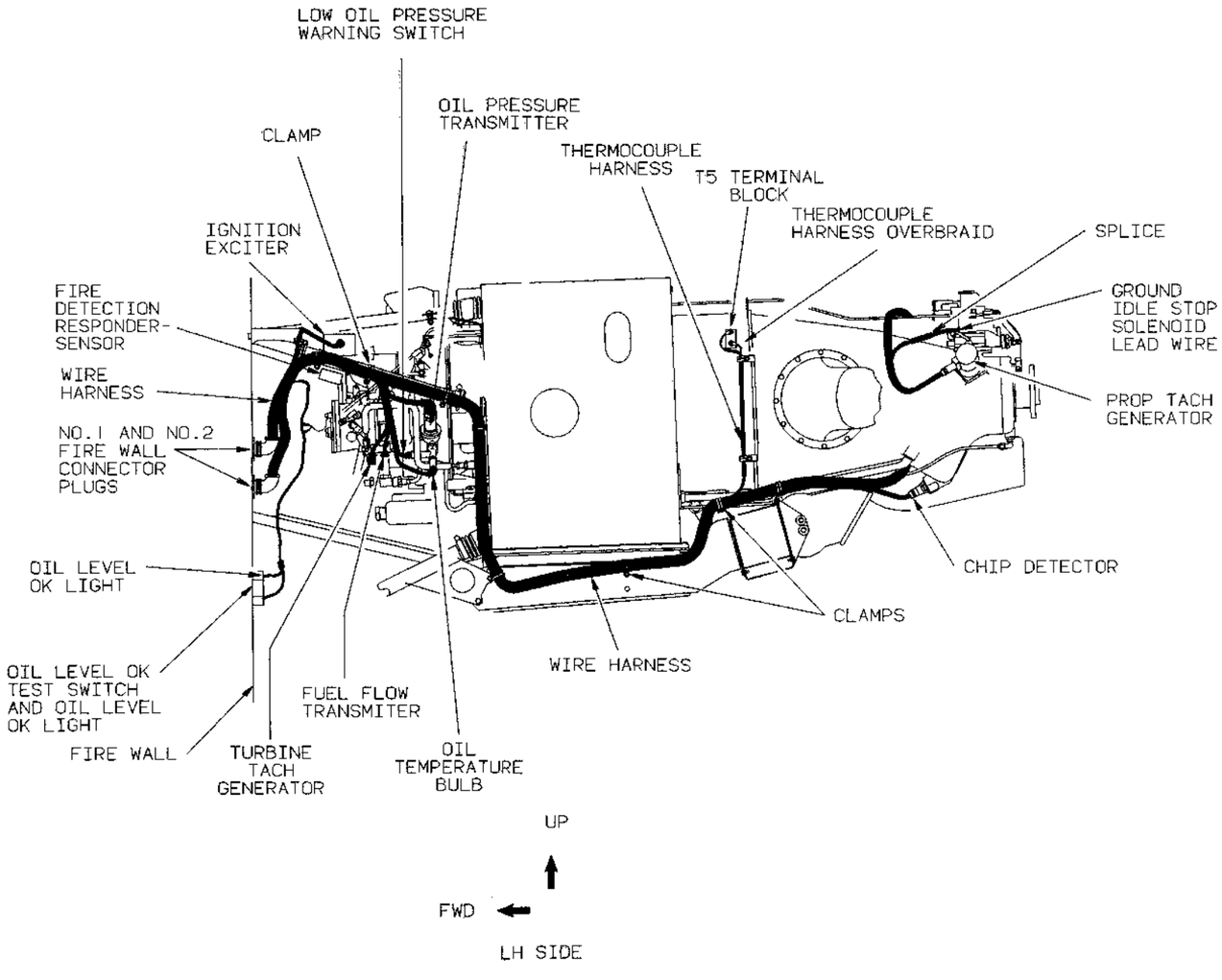
d. Cut the safety wire and disconnect the wire harness from the ignition exciter.

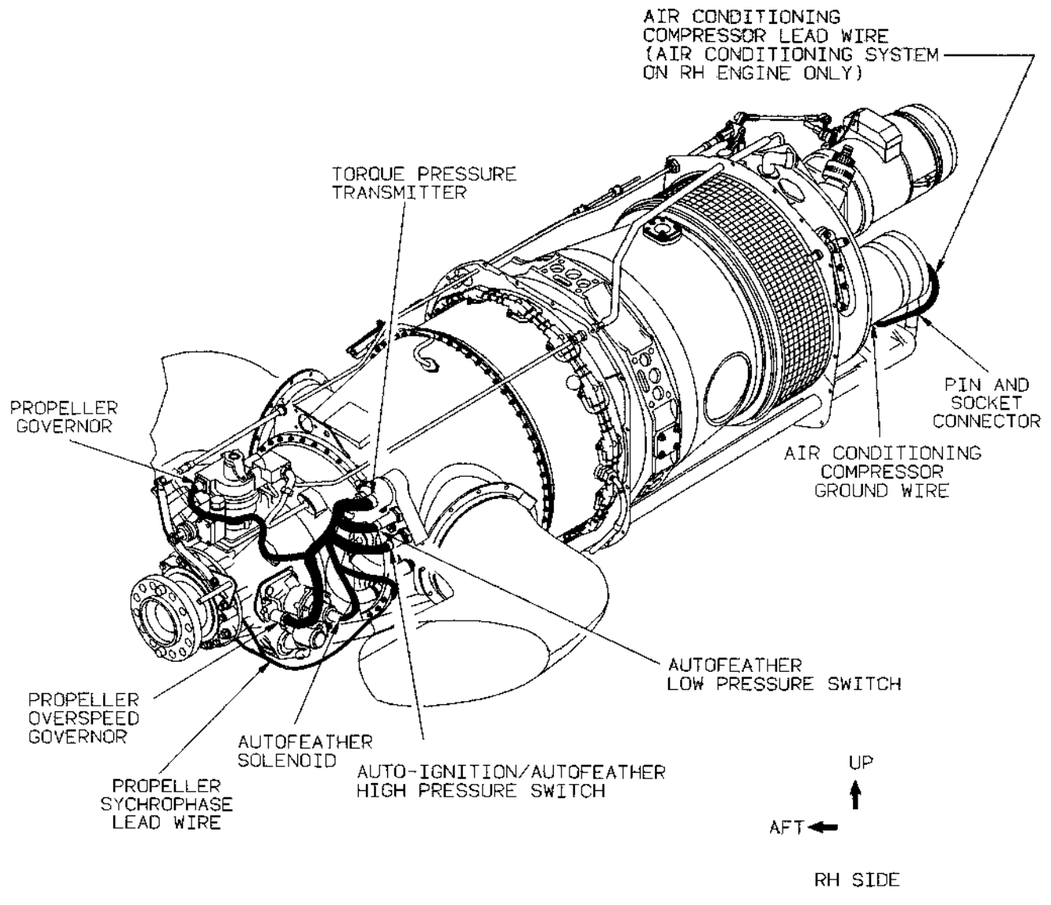
e. Cut the safety wire and disconnect the wire harness from the turbine tach generator.

f. Cut the safety wire and disconnect the wire harness from the fuel flow transmitter.

g. Cut the safety wire and disconnect the wire harness from the low oil pressure warning switch.

h. Cut the safety wire and disconnect the wire harness from the oil pressure transmitter.





Engine Electrical Wire Harness (Effectivity: All)
Figure 1

i. Cut the safety wire and disconnect the wire harness from the oil temperature bulb.

j. Cut the safety wire and disconnect the wire harness from the fire detection responder-sensor.

k. Disconnect the fire detection responder-sensor overbraid cable from the clamp located near the fire detection responder-sensor.

l. Cut the safety wire and disconnect the wire harness from the oil level OK test switch and oil level OK light. The oil level OK test switch and oil level OK light are mounted on a bracket that is secured to the firewall fuel filter mounting bracket.

m. Cut the safety wire and disconnect the wire harness from the oil level dipstick.

n. Disconnect the air conditioning compressor lead wire at the splice by disconnecting the pin and socket connector. (RH engine only.)

o. Remove the screw that retains the ground wire to the air conditioning compressor case and remove the ground wire. (RH engine only.)

p. Disconnect the thermocouple harness terminals from the T5 terminal block by unscrewing the nuts that retain them to the T5 terminal block.

q. Disconnect the thermocouple harness overbraid from the clamp located closest to the thermocouple sensor.

r. Cut the safety wire and disconnect the wire harness from the chip detector.

s. Cut the safety wire and disconnect the wire harness from the prop tach generator.

t. Remove the ground idle stop solenoid lead wire from the wire harness at the splice by disconnecting the pin and socket connector.

u. Cut the safety wire and disconnect the wire harness from the torque pressure transmitter.

v. Cut the safety wire and disconnect the wire harness from the auto-ignition/autofeather high pressure switch.

w. Cut the safety wire and disconnect the wire harness from the autofeather low pressure switch.

x. Cut the safety wire and disconnect the wire harness from the autofeather solenoid.

y. Cut the safety wire and disconnect the wire harness from the propeller overspeed governor.

z. Remove the wire harness from the magnetic pickup on the propeller synchrophase bracket.

aa. Cut the safety wire and disconnect the plug on the wire harness from the propeller governor.

ab. Remove the clamps that retain the wire harness to the firewall (2 places), engine mount (4 places), air plenum (2 places), oil cooler support (2 places) and oil cooler flanges (2 places).

ac. Remove the clamps (2 places) that retain the thermocouple harness to the gas generator case.

ad. Remove the clamps that retain the wire harness to the reduction gearbox case.

ae. Remove the wire harness from the engine.

WIRE HARNESS INSTALLATION
(Effectivity: All)
(Figure 1)

a. Remove electrical power from the airplane and disconnect the battery.

b. Connect the wire harness connectors to the No. 1 and No. 2 firewall connector plugs. Secure the connectors with safety wire.

c. Connect the wire harness to the ignition exciter. Secure the connector with safety wire.

d. Connect the wire harness to the turbine tach generator. Secure the connector with safety wire.

e. Connect the wire harness to the fuel flow transmitter. Secure the connector with safety wire.

f. Connect the wire harness to the low oil pressure warning switch. Secure the connector with safety wire.

g. Connect the wire harness to the oil pressure transmitter. Secure the connector with safety wire.

h. Connect the wire harness to the oil temperature bulb. Secure the connector with safety wire.

i. Connect the wire harness to the fire detection responder-sensor. Secure the connector with safety wire.

j. Attach the fire detection responder-sensor overbraid cable to the clamp located near the fire detection responder-sensor.

k. Connect the wire harness to the oil level OK light and to the oil level OK test switch. The oil level OK light and oil level OK test switch are mounted on a bracket which is secured to the firewall fuel filter mounting bracket.

l. Connect the wire harness to the oil level dipstick. Secure the connector with safety wire.

m. Connect the air conditioning compressor lead wire to the wire harness at the splice with the pin and socket connector. (RH engine only.)

n. Attach the air conditioning compressor ground wire to the air conditioning compressor case and secure with the screw. (RH engine only.)

o. Connect the thermocouple harness to the T5 terminal block. Attach the green wire with the alumel (large) nut and attach the gray wire with the chromel (small) nut. Torque the alumel (large) nut to 25 to 30 inch-pounds and torque the (smaller) chromel nut to 20 to 25 inch-pounds.

p. Install the thermocouple harness into the clamps and attach them to the gas generator case (2 places).

q. Connect the thermocouple harness overbraid to the clamp located closest to the thermocouple sensor.

r. Install the wire harness into the clamps and attach them to the firewall (2 places), engine mount (4 places), air plenum (2 places), oil cooler bracket (2 places) and oil cooler flanges (2 places).

s. Connect the wire harness to the chip detector. Secure the connector with safety wire.

t. Connect the wire harness to the prop tach generator. Secure the connector with safety wire.

u. Connect the wire harness to the ground idle stop solenoid lead wire at the splice with the pin and socket connector.

v. Connect the wire harness to the torque pressure transmitter. Secure the connector with safety wire.

w. Connect the wire harness to the auto-ignition/ autofeather high pressure switch. Secure the connector with safety wire.

x. Connect the wire harness to the autofeather low pressure switch. Secure the connector with safety wire.

y. Connect the wire harness to the autofeather solenoid. Secure the connector with safety wire.

z. Connect the wire harness to the propeller overspeed governor. Secure the connector with safety wire.

aa. Attach the wire harness to the magnetic pickup on the propeller synchrophase bracket.

ab. Connect the wire harness to the propeller governor. Secure the connector with safety wire.

ac. Install the wire harness into the clamps and attach the clamps to the reduction gearbox case.

ad. Install the engine cowling as outlined in Chapter 71-10-00.

ae. Connect the battery and restore electrical power to the airplane.

ENGINE AIR INTAKES - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The engine air intakes consist of an air inlet duct and air plenum which are located on the aft side of the firewall. Air flows past the heated intake at the front of the nacelle and through the ice separation vanes where ice particles are separated from the incoming air (see Chapter 30-20-00). The air is ducted through the air inlet duct to the underside of the engine where it enters the air plenum. A seal is attached to the top of the air inlet duct to provide a seal between the air inlet duct and air plenum. The air plenum is an aluminum housing which encloses the forward section of the engine, and is attached to the engine fireseal mount rings. The air plenum directs air into the engine air inlet screen.

The air plenum is constructed primarily of aluminum sheet. It consists of a one piece top cover, two side panel assemblies, six bulkhead sections, a lower flange and several closeouts, retainers and fasteners. The top cover has four channel stiffeners riveted to it and is fastened to the side panels with twenty-two Camloc fasteners. The side panels and top cover form the edges and flanges that the bulkhead panels, boot retainers and closeout panels are attached to. Aluminum is used to form both the closeout panels and the bulkhead sections. The bulkhead sections (three forward and three aft) are attached to the fireseal mount rings and provide an attach point for the oil tank drain shutoff valve. The closeout panels are attached to the bulkhead sections and shroud the engine truss mount where the engine truss mount passes through the air plenum. Boot retainers are used to attach the boots to the closeout panels. The boots form a seal between the air plenum and the engine mount. The air plenum serves as a chamber by which intake air is supplied to the engine.

The oil cooler intake scoop is an integral part of the lower forward cowling assembly. Air flows into the scoop and is ducted upward through the oil cooler. The air is then routed aft from the oil cooler through an air exit duct and into the propeller wake.

The starter/generator air intake scoop is attached to the LH side of the lower forward cowling. The scoop is fastened to the cowling with six stud nuts. Air is ducted from the scoop to the starter/generator assembly. The starter/generator intake scoop and duct ensure that the starter/generator is blast cooled during flight.

ENGINE AIR INTAKES - MAINTENANCE PRACTICES (Effectivity: All)

AIR PLENUM REMOVAL (Effectivity: All) (Figures 1 and 2)

a. Remove electrical power from the airplane and disconnect the battery.

b. Remove the engine cowling. See Chapter 71-10-00.

c. Remove the oil breather tube from the top cover. See Chapter 79-00-00.

d. Disconnect the propeller control cable and remove it from the air plenum. See Chapter 71-00-00.

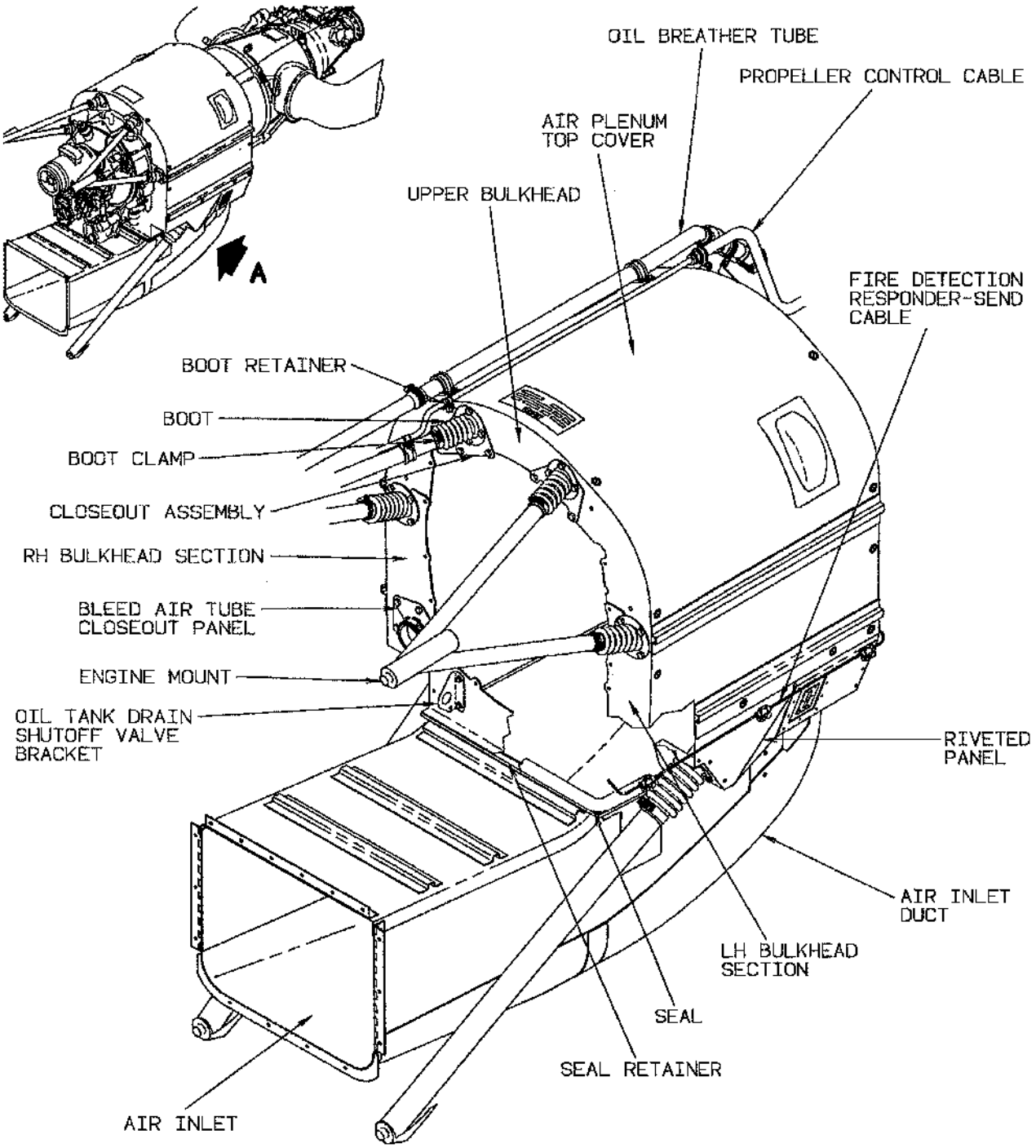
e. Remove the bond jumpers that ground the air plenum to the engine truss mount.

f. Release the Camloc fasteners that retain the top cover (1) to the side panels (5 and 6) and bulkhead sections (2,3,4,7,8 and 9). Remove the top cover (1).

g. Remove the prop seal drain tubing and combustion chamber drain tubing from the air plenum. See Chapter 71-70-00.

h. Remove the fire detection responder-sensor cable assembly from the air plenum. See Chapter 26-10-00.

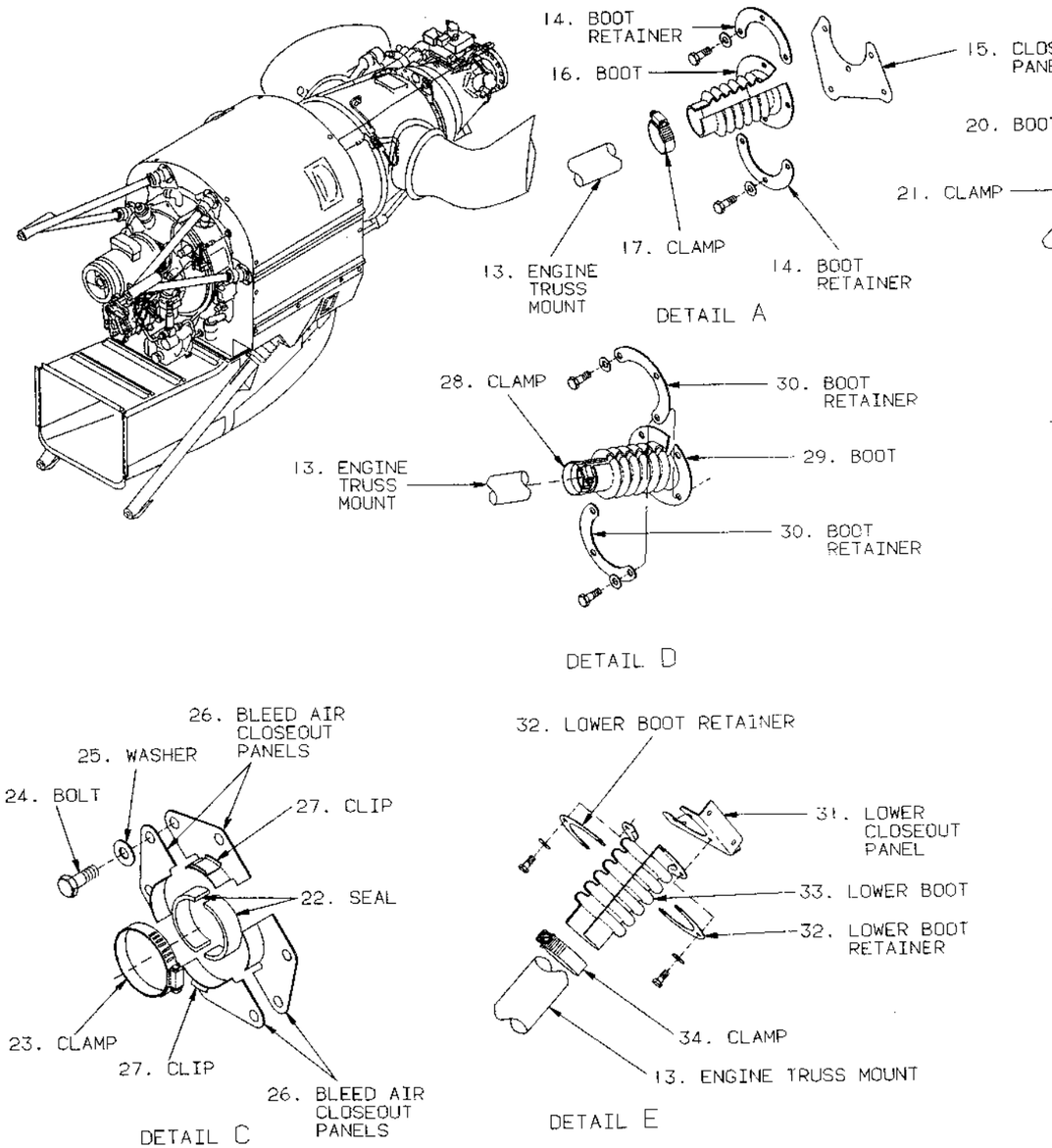
i. Remove the wire harness from the air plenum. See Chapter 71-50-00.



DETAIL A

2000-240-06

Engine Air Intakes (Effectivity: All)
Figure 1



Engine Air Intakes Installation/Removal (Effectivity: All)
Figure 2

j. Remove the oil tank drain shutoff valve bracket from the air plenum. See Chapter 79-00-00.

k. While supporting the aft end of the air inlet duct (12), remove the bolts and washers that retain the air inlet duct (12) to the engine truss mount brackets.

l. While supporting the air inlet duct (12), remove the oil cooler and oil cooler support from the air inlet duct (12). See Chapter 79-00-00.

m. Remove the bolts and washers that retain the air inlet duct (12) to the firewall.

n. Remove the air inlet duct (12) by sliding it aft and lowering it from the engine.

o. Remove the screws that attach the lower flange (10) to the LH and RH side panels (5 and 6), forward bulkhead sections (7,8 and 9) and aft bulkhead sections (2,3 and 4). The screws can be accessed from the bottom side of the lower flange (10).

p. Remove the clamps (17,21 and 28), bolts and washers that retain the closeout panels (15 and 18) and boot retainers (14,19 and 30) to the forward bulkhead sections (7,8 and 9).

q. Remove the boots (16,20 and 29), boot retainers (14,19 and 30) and closeout panels (15 and 18) from the forward bulkhead sections (7,8 and 9).

r. Remove the bolts and washers that retain the bleed air closeout panels (26) to the RH forward bulkhead section (8).

s. Remove the seal (22), clamp (23) and clips (27) from the P3 bleed air tubing. They are located forward of the bleed air closeout panels (26).

t. Remove the bleed air closeout panels (26) from the RH forward bulkhead section (8).

u. Remove the bolts and washers that retain the forward bulkhead sections (7,8 and 9) to the fireseal mount rings and remove the upper forward bulkhead section (7).

v. Remove the fasteners that retain the RH forward bulkhead section (8) to the RH side panel (5) and remove the RH forward bulkhead section (8).

w. Remove the fasteners that retain the LH forward bulkhead section (9) to the LH side panel (6) and remove the LH forward bulkhead section (9).

x. Remove the bolts and washers that retain the aft bulkhead sections (2,3 and 4) to the fireseal mount rings and remove the upper aft bulkhead section (2).

y. Remove the fasteners that retain the RH aft bulkhead section (3) to the RH side panel (5) and remove the RH aft bulkhead section (3).

z. Remove the fasteners that retain the LH aft bulkhead section (4) to the LH side panel (6) and remove the LH aft bulkhead section (4).

aa. Remove the clamps (34) that retain the lower boots (33) to the engine truss mounts (13).

ab. Remove the bolts and washers that retain the lower boot retainers (32) and lower closeout panels (31) to the LH and RH side panels (6 and 5).

ac. Remove the LH and RH side panels (6 and 5).

AIR PLENUM INSTALLATION
(Effectivity: All)
(Figure 2)

a. Attach the LH forward bulkhead section (9) to the forward fireseal mount rings with the bolts and washers.

b. Attach the RH forward bulkhead section (8) to the forward fireseal mount rings with the bolts and washers.

c. Attach the bleed air closeout panels (26) to the RH forward bulkhead section (8) with the bolts and washers.

d. Install the seal (22), clips (27) and clamp (23) on the P3 bleed air tubing at the forward side of the forward bulkhead section (8).

e. Attach the upper forward bulkhead section (7) to the forward fireseal mount rings with the bolts and washers.

f. Attach the LH aft bulkhead section (4) to the aft fireseal mount rings with the bolts and washers.

g. Attach the RH aft bulkhead section (3) to the aft fireseal mount rings with the bolts and washers.

h. Attach the upper aft bulkhead section (2) to the aft fireseal mount rings with the bolts and washers.

i. Attach the LH side panel (6) to the LH forward and LH aft bulkhead sections (9 and 4). Secure the LH side panel (6) with the fasteners.

j. Attach the RH side panel (5) to the RH forward and RH aft bulkhead sections (8 and 3). Secure the RH side panel (5) with the fasteners.

k. Install the lower boots (33) into the lower boot retainers (32). Attach the lower boot retainers (32) and lower closeout panels (31) to the LH and RH side panels (6 and 5).

NOTE

Inspect the seal (11) on the air inlet duct (12) for cracks, rips and tears. Replace the seal (11) if necessary.

l. Attach the lower flange (10) to the side panels (5 and 6) and bulkhead sections (2,3,4,7,8 and 9) with the screws.

m. Install the boot retainers (14,19 and 30) into the boots (16,20 and 29).

n. Attach the boot retainers (14,19 and 30) and closeout panels (15 and 18) to the forward bulkhead sections (7,8 and 9) with the bolts and washers.

o. Secure the boot retainers (14,19 and 30) to the engine truss mount (13) with the clamps (17,21 and 28).

p. Raise the air inlet duct (12) into position underneath the air plenum and attach it to the engine truss mount brackets with the bolts and washers.

q. Install the engine oil cooler and oil cooler support on the air inlet duct (12). See Chapter 79-00-00.

r. Attach the air inlet duct (12) to the firewall with the bolts and washers.

s. Install the prop seal drain tubing and combustion chamber drain tubing. See Chapter 71-70-00.

t. Attach the air plenum top cover (1) to the bulkhead sections (2,3,4,7,8 and 9) and side panels (5 and 6). Secure the top cover (1) with the Camloc fasteners.

u. Attach the wire harness to the air plenum. See Chapter 71-50-00.

v. Attach the oil tank drain shutoff valve bracket to the air plenum. See Chapter 79-00-00.

w. Attach the bond jumpers to the engine truss mount (13) and air plenum. Perform a resistance check on the bond jumpers. See Chapter 40-00-00.

x. Install the oil breather tube on the top cover (1). See Chapter 79-00-00.

y. Connect the propeller control cable and attach it to the air plenum. See Chapter 71-00-00.

z. Attach the fire detection responder-sensor cable assembly to the air plenum. See Chapter 26-10-00.

aa. Install the engine cowling. See Chapter 71-10-00.

ab. Connect the battery and restore electrical power to the airplane.

ENGINE DRAINS - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The engine drain system is designed to remove unwanted liquids (fuel and oil) from the engine compartment by safely draining them overboard. The engine drains are made up of a combination of flexible hoses and rigid metal tubing that run from the fuel drain, fuel vents and oil seals to the engine drain assembly. Metal tubing is used in areas where the drain tubes may be exposed to high temperatures. The engine drain assembly is attached to a fairing located at the juncture of the lower forward cowling and the lower surface of the wing. Engine drain adapters are threaded into the engine accessories, engine accessory gearbox and engine casing to provide fittings that the drain tubes and hoses can be attached to. The engine drains serve to contain the fuel and oil and reduce the chance of engine fire.

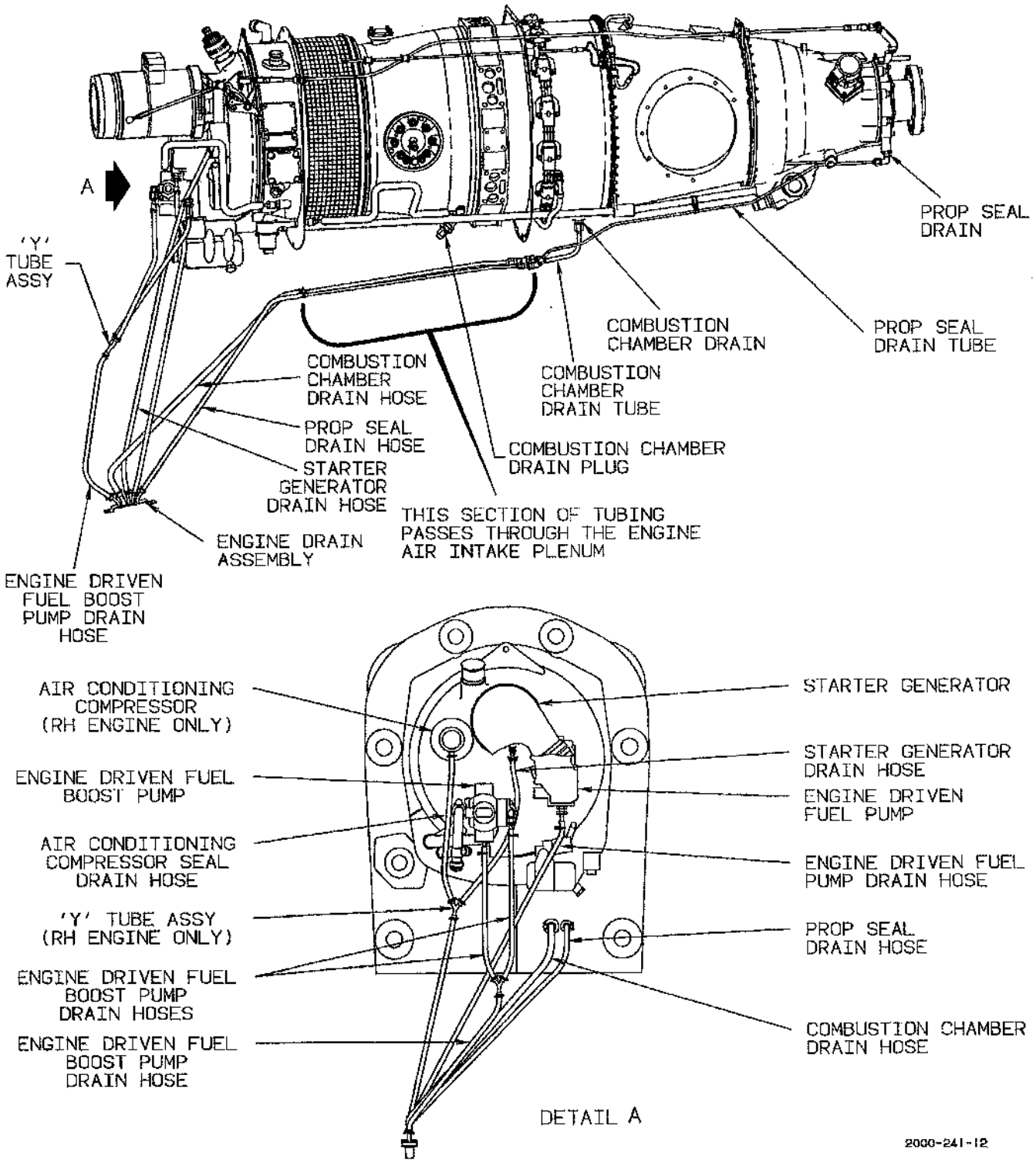
When the engine is shut down, any residual fuel that may be present in the bottom of the gas generator case and exhaust duct is drained to the combustion chamber drain valve. The combustion chamber drain valve is automatically opened when the engine is shut down (P3 pressure against the spring in the drain valve keeps the drain valve closed while the engine is operating), and the residual fuel is drained overboard through the combustion chamber drain tube. The combustion chamber drain tube is threaded onto the combustion chamber drain valve adapter at the underside of the engine. The combustion chamber drain tube runs forward from the combustion chamber drain valve and connects to a union where it passes through the aft side of the air plenum. A grommet is used to seal the forward side of the air plenum where the combustion chamber drain tube passes through. A hose is attached to the combustion chamber drain tube forward of the air plenum and runs to the engine drain assembly.

The prop seal drain tube is connected to an elbow which is threaded into the prop seal drain on the reduction gearbox housing. The prop seal drain tube is clamped to the oil transfer tube and runs to an elbow where it passes through the aft side of the air plenum. A grommet is used to seal the air plenum where the prop seal drain tube passes through. A hose is attached to the drain tube forward of the air plenum and runs to the engine drain assembly.

The engine accessory gearbox has six drain hoses (seven on the RH engine) connected to attachment points on the engine driven accessories. The engine driven fuel boost pump has two drain hoses attached to it. These two hoses are connected with hose clamps and are routed into a Y tube assembly. From the Y tube assembly, a single hose runs to the engine drain assembly. The engine driven fuel pump has one drain hose connected to it. This hose is routed directly to the engine drain assembly.

On the LH engine, the starter/generator drain hose is connected to the starter/generator seal drain and runs directly to the engine drain assembly. The LH engine does not have an air conditioning compressor, nor an air conditioning drain Y tube assembly installed on it. The air conditioning compressor is mounted on the RH engine accessory gearbox. The air conditioning compressor drain hose is connected to the air conditioning compressor drain and runs to a Y tube assembly that is connected to the starter/generator drain hose. A second starter/generator drain hose is connected to the Y tube assembly and runs to the engine drain assembly.

It will be necessary to periodically inspect and replace the engine drain system components as needed. Specific items to inspect include the Y tubes, the tube elbows and the hose connection fittings at the engine drain assembly. When the engine is to be removed from the airplane, the drain tubes, drain hoses and fittings must be removed. After the drains have been removed, compressed air and solvent can be used to clean the drain tubes and fittings.



Engine Drain Tubing (Effectivity: All)
Figure 1

ENGINE DRAINS - MAINTENANCE PRACTICES (Effectivity: All)

STARTER/GENERATOR AND AIR CONDITIONING DRAIN HOSE REMOVAL (Effectivity: All) (Figure 2)

a. Remove electrical power from the airplane and disconnect the battery.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

NOTE

Tag all tube ends before disconnecting. Save the clamps and Y tube for installation.

c. Remove the two hose clamps (2 and 4) from the air conditioning compressor mount assembly and the hose adapter (5) on the starter/generator.

NOTE

The air conditioning compressor is mounted on the RH engine only.

d. Disconnect the air conditioning compressor drain hose (1) from the air conditioning compressor mount assembly. (RH engine only.)

e. Disconnect the starter/generator drain hose (3) from the hose adapter (5) on the starter/generator.

f. Remove the hose clamp (23) from the starter/generator drain hose (22) and disconnect the starter/generator drain hose (22) from the engine drain assembly (25).

g. Remove the air conditioning and starter/generator drain hoses from the engine compartment.

h. Disconnect the air conditioning and starter/generator drain hoses from the Y tube assembly. (RH engine only.)

i. Remove the hose adapter (5) from the starter/generator by unscrewing it from the engine accessory gearbox mounting pad.

STARTER/GENERATOR AND AIR CONDITIONING DRAIN HOSE INSTALLATION (Effectivity: All) (Figure 2)

a. Screw the hose adapter (5) into the engine accessory gearbox mounting pad.

b. Connect the starter/generator drain hose (3) to the hose adapter (5) and secure with a hose clamp (4).

NOTE

The air conditioning compressor is mounted on the RH engine only.

c. Connect the air conditioning compressor drain hose (1) to the air conditioning compressor mount assembly and secure with a hose clamp (2). (RH engine only.)

d. Connect the air conditioning compressor drain hose (1) and starter/generator drain hose (3) to the Y tube assembly and secure with the two hose clamps. (RH engine only.)

e. Connect the starter/generator drain hose (22) to the Y tube assembly and route it to the engine drain assembly (25). Secure with the two hose clamps (23). (RH engine only.)

f. Slide the starter/generator drain hose (3) over the fitting on the engine drain assembly (25) and secure with a hose clamp (23). (LH engine only.)

g. Install the engine cowling as outlined in Chapter 71-10-00.

h. Connect the battery and restore electrical power to the airplane.

ENGINE DRIVEN FUEL BOOST PUMP DRAIN HOSE REMOVAL (Effectivity: All) (Figure 2)

a. Remove electrical power from the airplane and disconnect the battery.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

NOTE

Tag all tube ends before disconnecting. Save the nut and clamps for installation.

c. Remove the two hose clamps (13 and 14) and disconnect the two engine driven fuel boost pump drain hoses (11 and 12) from the engine driven fuel boost pump.

d. Remove the tube assembly (15) from the engine driven fuel boost pump by unscrewing it from the elbow (16).

e. Unscrew the elbow (16) from the engine driven fuel boost pump.

f. Remove the hose adapter (19) and O-ring (17) from the engine driven fuel boost pump by loosening the nut (18) and unscrewing the hose adapter (19). Discard the O-ring (17).

g. Remove the hose clamp (20) and disconnect the engine driven fuel boost pump drain hose (21) from the engine drain assembly (25).

h. Remove the drain hoses (11, 12 and 21) from the engine compartment and disconnect them from the Y tube assembly.

ENGINE DRIVEN FUEL BOOST PUMP DRAIN HOSE INSTALLATION (Effectivity: All)
(Figure 2)

a. Coat the threads on the elbow (16) with anti-seize compound (1, Chart 1, Chapter 71-00-00) and screw the elbow into the engine driven fuel boost pump.

b. Screw the tube assembly (15) onto the elbow (16).

c. Coat the threads on the hose adapter (19) and O-ring (17) sparingly with lubricant (5, Chart 1, Chapter 71-00-00).

d. Screw the nut (18) onto the hose adapter (19) until it is positioned past the groove between the threads.

e. Slide the O-ring (17) onto the hose adapter (19) and work the O-ring into the groove between the threads on the hose adapter.

f. Screw the nut (18) back toward the O-ring (17) until the nut forces the O-ring against the threads on the end of the hose adapter (19).

g. Thread the hose adapter (19) into the engine driven fuel boost pump and at the same time keep the nut (18) turning with the hose adapter. Screw the hose adapter in until the O-ring (17) comes in contact with the engine driven fuel boost pump. This point can be determined by a sudden increase in the amount of torque required to turn the hose adapter.

h. Place a wrench on the nut (18) to prevent it from turning and screw the hose adapter (19) into the engine driven fuel boost pump at least 1 1/2 turns, but no more than 2 1/2 turns.

i. Secure the hose adapter (19) by placing a wrench on the hose adapter and turning the nut (18) down tightly against the engine driven fuel boost pump while holding the hose adapter stationary.

j. Connect the two engine driven fuel boost pump drain hoses (11 and 12) to the tube assembly (15) and hose adapter (19). Secure them with the two hose clamps (13 and 14).

k. Connect the two engine driven fuel boost pump drain hoses (11 and 12) to the Y tube assembly and secure with the two hose clamps.

l. Slide the engine driven boost pump drain hose (21) over the lower end of the Y tube assembly and attach it to the engine drain assembly (25). Secure the hose ends with hose clamps (20).

m. Install the engine cowling as outlined in Chapter 71-10-00.

n. Connect the battery and restore electrical power to the airplane.

ENGINE DRIVEN FUEL PUMP DRAIN HOSE REMOVAL (Effectivity: All)
(Figure 2)

a. Remove electrical power from the airplane and disconnect the battery.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

NOTE

Tag all tube ends before disconnecting.
Save the nut and clamps for installation.

c. Remove the hose clamp (9) from the hose adapter (8) and disconnect the engine driven fuel pump drain hose (10).

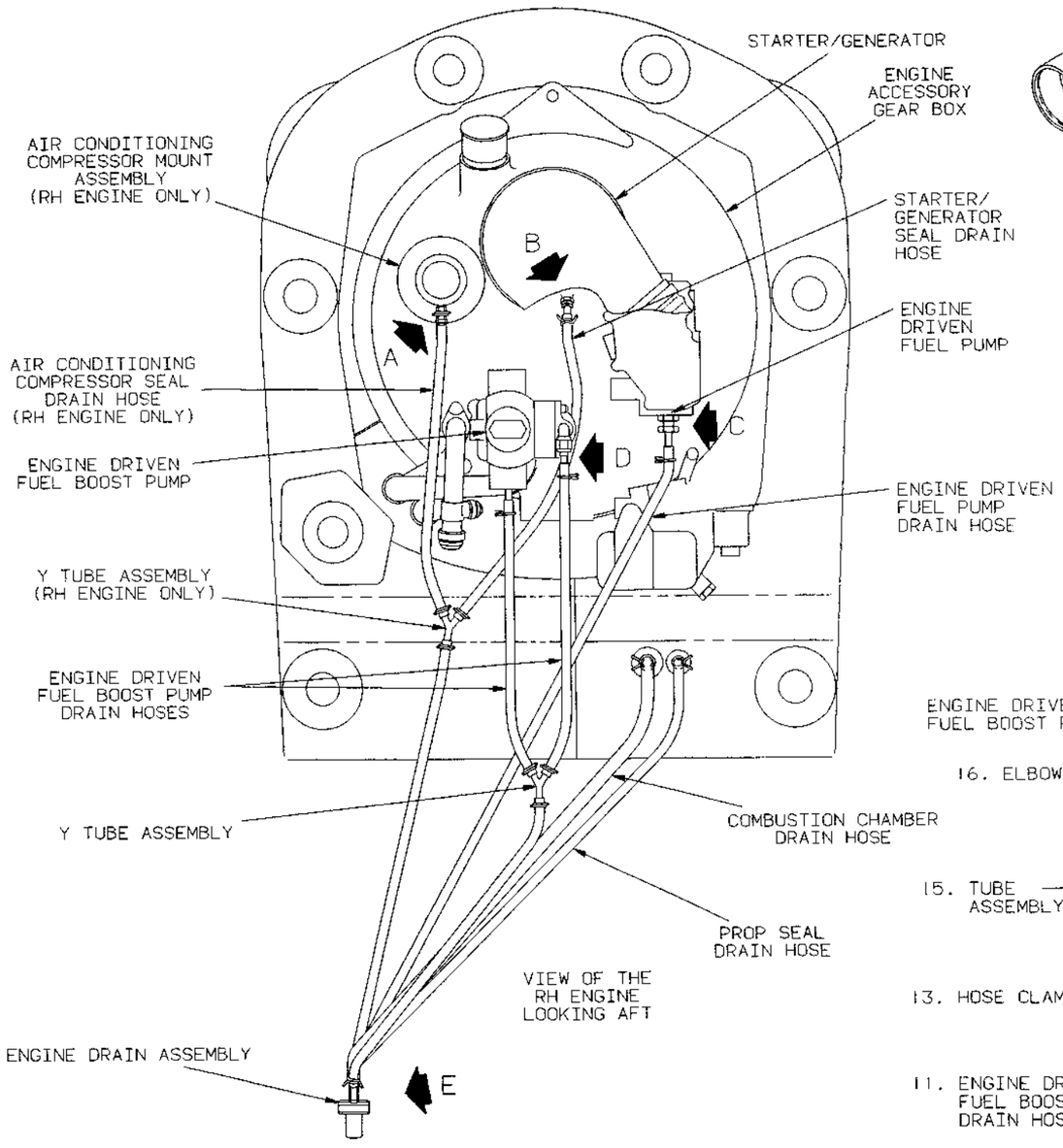
d. Remove the hose clamp (24) and disconnect the engine driven fuel pump drain hose (10) from the engine drain assembly (25).

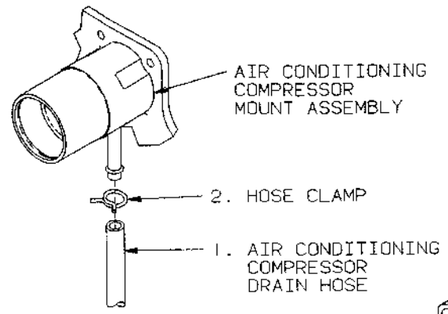
e. Remove the hose adapter (8) and O-ring (6) by loosening the nut (7) and unscrewing the hose adapter (8) from the engine driven fuel pump. Discard the O-ring (6).

f. Remove the engine driven fuel pump drain hose (10) from the engine compartment.

ENGINE DRIVEN FUEL PUMP DRAIN HOSE INSTALLATION (Effectivity: All)
(Figure 2)

a. Coat the threads of the hose adapter (8) and O-ring (6) sparingly with lubricant (5, Chart 1, Chapter 71-00-00).





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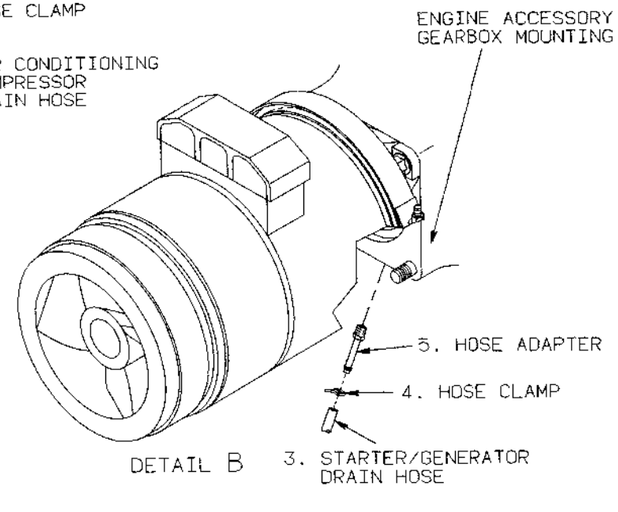

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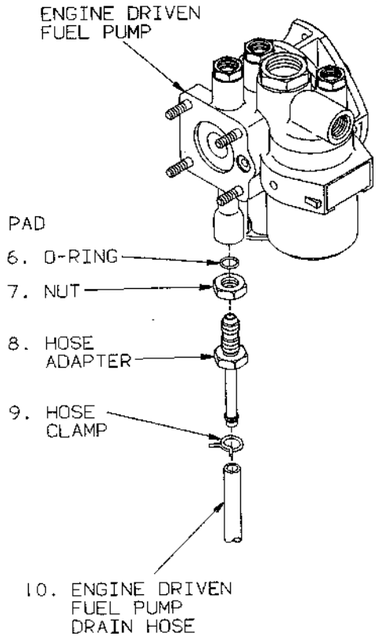

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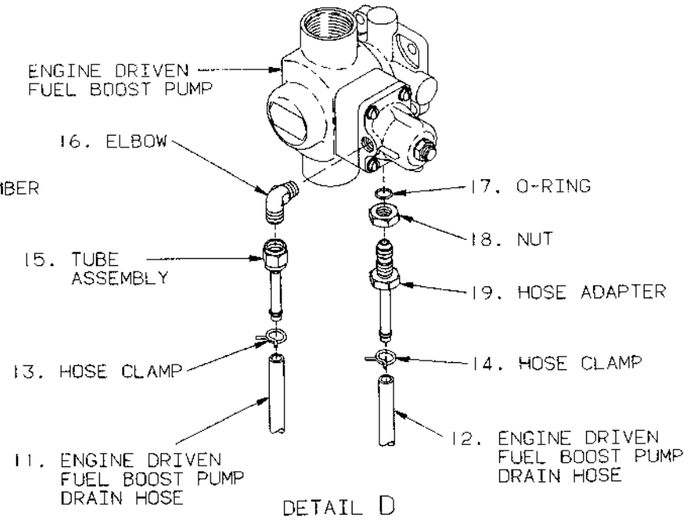

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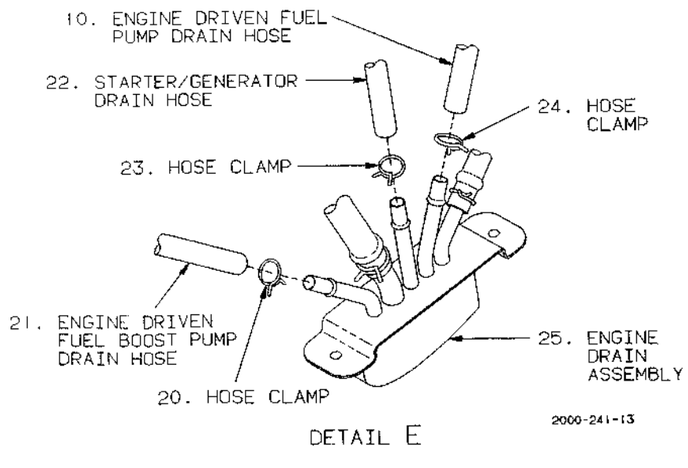


DETAIL E

Engine Drain Hoses Removal/Installation (Effectivity: All)
Figure 2

b. Screw the nut (7) onto the hose adapter (8) until it is positioned past the groove between the threads.

c. Slide the O-ring (6) onto the hose adapter (8) and work the O-ring into the groove between the threads.

d. Screw the nut (7) back toward the O-ring (6) until the nut forces the O-ring against the threads on the end of the hose adapter (8).

e. Thread the hose adapter (8) into the engine driven fuel pump and at the same time keep the nut (7) turning with the hose adapter. Screw the hose adapter in until the O-ring (6) comes in contact with the engine driven fuel pump. This point can be determined by a sudden increase in the amount of torque required to turn the hose adapter.

f. Place a wrench on the nut (7) to prevent it from turning and screw the hose adapter (8) into the engine driven fuel pump at least 1 1/2 turns, but no more than 2 1/2 turns.

g. Secure the hose adapter (8) by placing a wrench on the hose adapter and turning the nut (7) down tightly against the engine driven fuel pump while holding the hose adapter stationary.

h. Slide the engine driven fuel pump drain hose (10) over the hose adapter (8) and secure with a hose clamp (9).

i. Connect the engine driven fuel pump drain hose (10) to the engine drain assembly (25) and secure with a hose clamp (24).

j. Install the engine cowling as outlined in Chapter 71-10-00.

k. Connect the battery and restore electrical power to the airplane.

PROP SEAL DRAIN TUBING REMOVAL (Effectivity: All) (Figure 3)

a. Remove electrical power from the airplane and disconnect the battery.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

NOTE

Tag all tube ends before disconnecting. Save the nuts, washers bolts and clamps for installation.

c. Disconnect the prop seal drain tube (14) from the elbow (35).

d. Remove the clamp (31) which is mounted to the oil transfer tube (28) on the LH side of the engine.

e. Unscrew the prop seal drain tube (14) from the elbow (15) on the aft side of the air plenum (16), and remove the prop seal drain tube (14).

f. Loosen the nut (34) on the elbow (35).

g. Remove the elbow (35), O-ring (33), and nut (34) from the prop seal drain (32) and retain them for installation. Discard the O-ring (33).

h. Remove the air plenum top cover as outlined in Chapter 71-60-00.

i. Disconnect the prop seal drain tube (19) from the elbow (15) on the inside of the air plenum (16).

j. Unscrew the nut (18) from the elbow (15) on the inside of the air plenum (16). Remove the elbow (15) from the aft side of the air plenum (16).

k. Remove the hose clamp (22) from the prop seal drain hose (3) and disconnect the prop seal drain hose from the prop seal drain tube (19).

l. Remove the prop seal drain tube (19) from the grommet (20) in the air plenum (21).

m. Remove the hose clamp (4) from the prop seal drain hose (3) at the engine drain assembly (5) and disconnect the prop seal drain hose from the engine drain assembly.

n. Remove the prop seal drain hose (3) from the engine compartment.

PROP SEAL DRAIN TUBING INSTALLATION (Effectivity: All) (Figure 3)

a. Screw the nut (34) onto the elbow (35) until the nut is positioned past the groove between the threads on the elbow.

b. Lubricate the O-ring (33) sparingly with lubricant (5, Chart 1, Chapter 71-00-00) and slide the O-ring into the groove on the elbow (35). Screw the nut (34) back towards the O-ring until it touches the O-ring.

c. Screw the elbow (35) and nut (34) simultaneously into the prop seal drain (32) until the O-ring (33) contacts the prop seal drain.

d. Position the elbow (35) in the prop seal drain (32) so the open end of the elbow points forward. The elbow can be screwed in not more than 3/4 of a turn or backed out not more than 1/4 of a turn.

e. Secure the elbow (35) by holding the elbow in place and tightening the nut (34) against the prop seal drain (32).

f. Install the elbow (15) onto the aft side of the air plenum (16) and secure it with the washer (17) and nut (18).

g. Connect the prop seal drain tube (14) to the elbow (35).

h. Attach the clamp (31) to the prop seal drain tube (14) and secure the clamp (31) to the oil transfer tube on the LH side of the engine case.

i. Connect the prop seal drain tube (14) to the elbow (15) on the aft side of the air plenum (16).

j. Route the prop seal drain tube (19) through the grommet (20) on the forward side of the air plenum (21).

k. Connect the prop seal drain tube (19) to the elbow (15) on the inside of the air plenum (16).

l. Attach the prop seal drain hose (3) to the prop seal drain tube (19) and secure with the hose clamp (22).

m. Connect the prop seal drain hose (3) to the engine drain assembly (5) and secure with hose clamp (4).

n. Install the air plenum top cover as outlined in Chapter 71-60-00.

o. Install the engine cowling as outlined in Chapter 71-10-00.

p. Connect the battery and restore electrical power to the airplane.

COMBUSTION CHAMBER DRAIN TUBING REMOVAL (Effectivity: All) (Figure 3)

a. Remove electrical power from the airplane and disconnect the battery.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

NOTE

Tag all tube ends before disconnecting. Save the nuts, washers and clamps for installation.

c. Unscrew and disconnect the combustion chamber drain tube (12) from the drain valve adapter (13).

d. Unscrew the combustion chamber drain tube (12) from the union (11) on the air plenum (16) and remove the combustion chamber drain tube (12).

e. Remove the air plenum top cover as outlined in Chapter 71-60-00.

f. Unscrew the combustion chamber drain tube (8) from the union (11) on the inside of the air plenum (16).

g. Unscrew the nut (9) from the union (11) on the inside of the air plenum (16). Remove the union (11) and washer (10) from the aft side of the air plenum (16).

h. Remove the hose clamp (6) and disconnect the combustion chamber drain hose (2) from the combustion chamber drain tube (8).

i. Remove the combustion chamber drain tube (8) from the grommet (7) in the air plenum (21).

j. Remove the hose clamp (1) from the combustion chamber drain hose (2) at the engine drain assembly (5).

k. Disconnect the combustion chamber drain hose (2) from the engine drain assembly (5).

l. Remove the combustion chamber drain hose (2) from the engine compartment.

COMBUSTION CHAMBER DRAIN TUBING INSTALLATION (Effectivity: All) (Figure 3)

a. Install the union (11) onto the aft side of the air plenum (16) with the washer (10) and nut (9).

b. Connect the combustion chamber drain tube (12) to the union (11) on the air plenum (16).

c. Screw the combustion chamber drain tube (12) onto the drain valve adapter (13).

d. Route the combustion chamber drain tube (8) through the grommet (7) in the forward side of the air plenum (21).

e. Connect the combustion chamber drain tube (8) to the union (11) on the inside of the air inlet plenum (16).

f. Attach the combustion chamber drain hose (2) to the combustion chamber drain tube (8) and secure with the hose clamp (6).

g. Connect the combustion chamber drain hose (2) to the engine drain assembly (5) and secure with the hose clamp (1).

h. Install the air plenum top cover as outlined in Chapter 71-60-00.

i. Install the engine cowling as outlined in Chapter 71-10-00.

j. Connect the battery and restore electrical power to the airplane.

COMBUSTION CHAMBER DRAIN CAP REMOVAL (Effectivity: All) (Figure 3)

a. Remove electrical power from the airplane and disconnect the battery.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

c. Remove the air plenum top cover as outlined in Chapter 71-60-00.

d. Unscrew the combustion chamber drain cap (23) from the drain valve adapter (24).

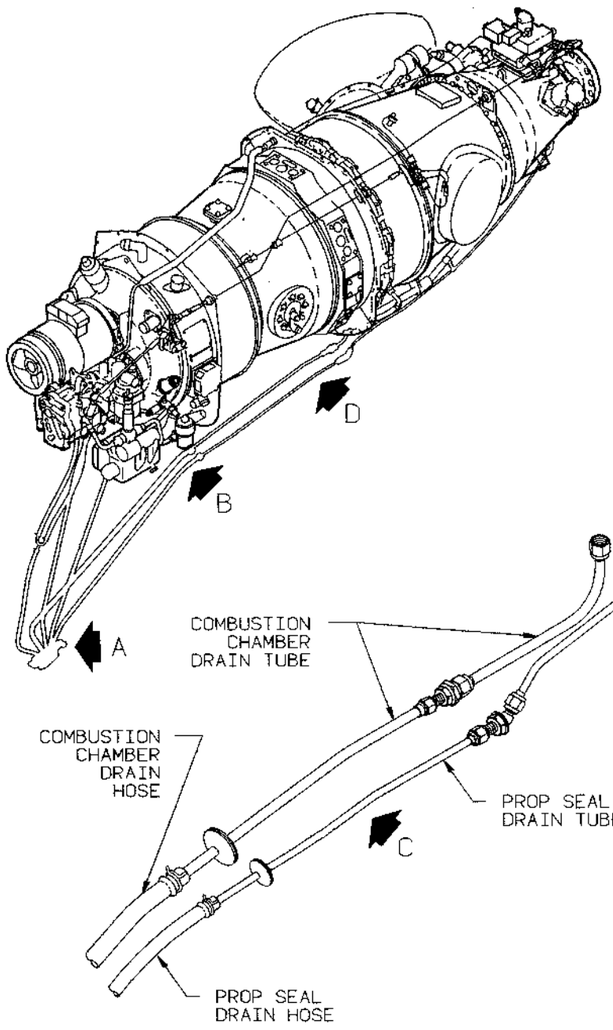
COMBUSTION CHAMBER DRAIN CAP INSTALLATION (Effectivity: All) (Figure 3)

a. Screw the combustion chamber drain cap (23) onto the drain valve adapter (24).

b. Install the air plenum top cover as outlined in Chapter 71-60-00.

c. Install the engine cowling as outlined in Chapter 71-10-00.

d. Connect the battery and restore electrical power to the airplane.



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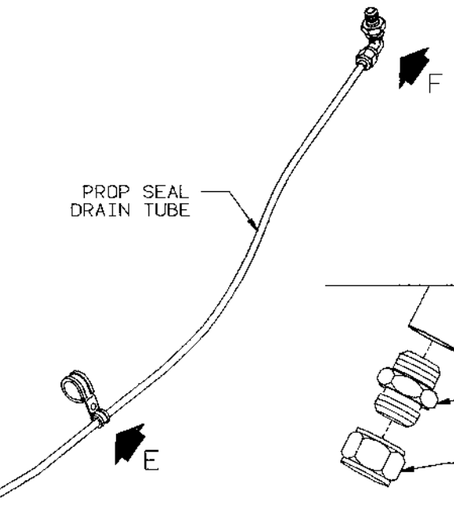

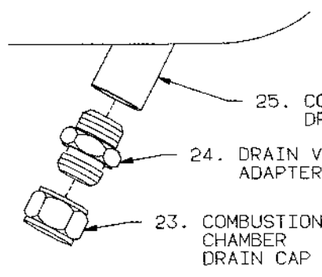

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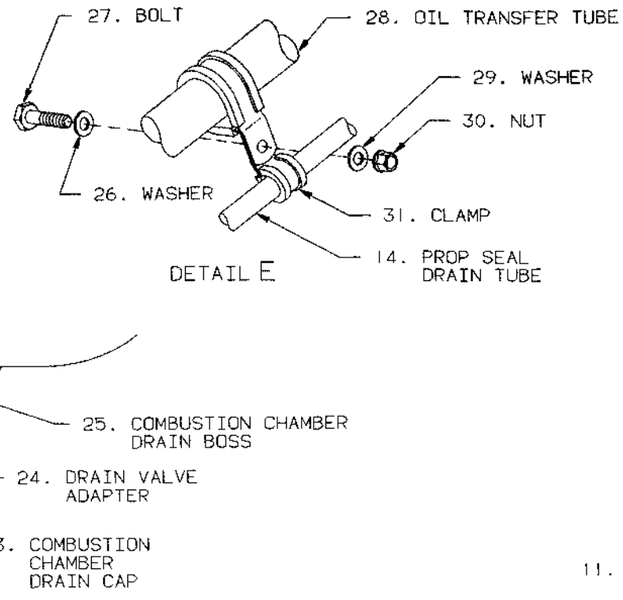

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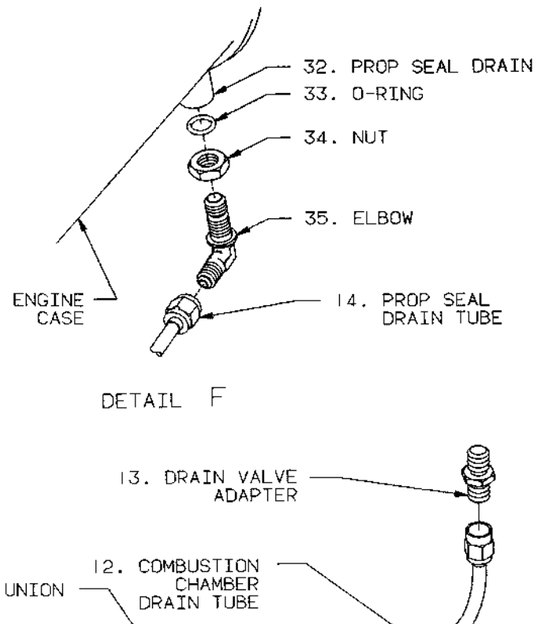

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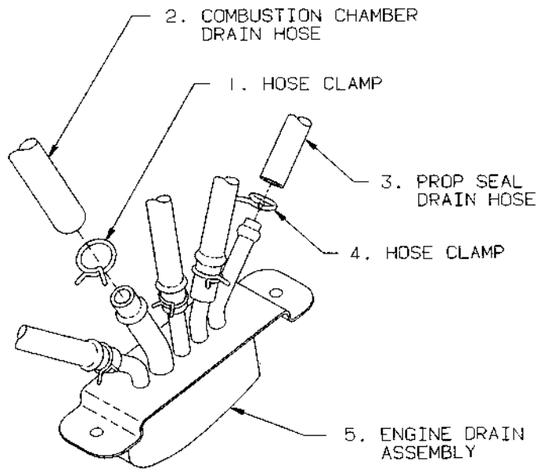

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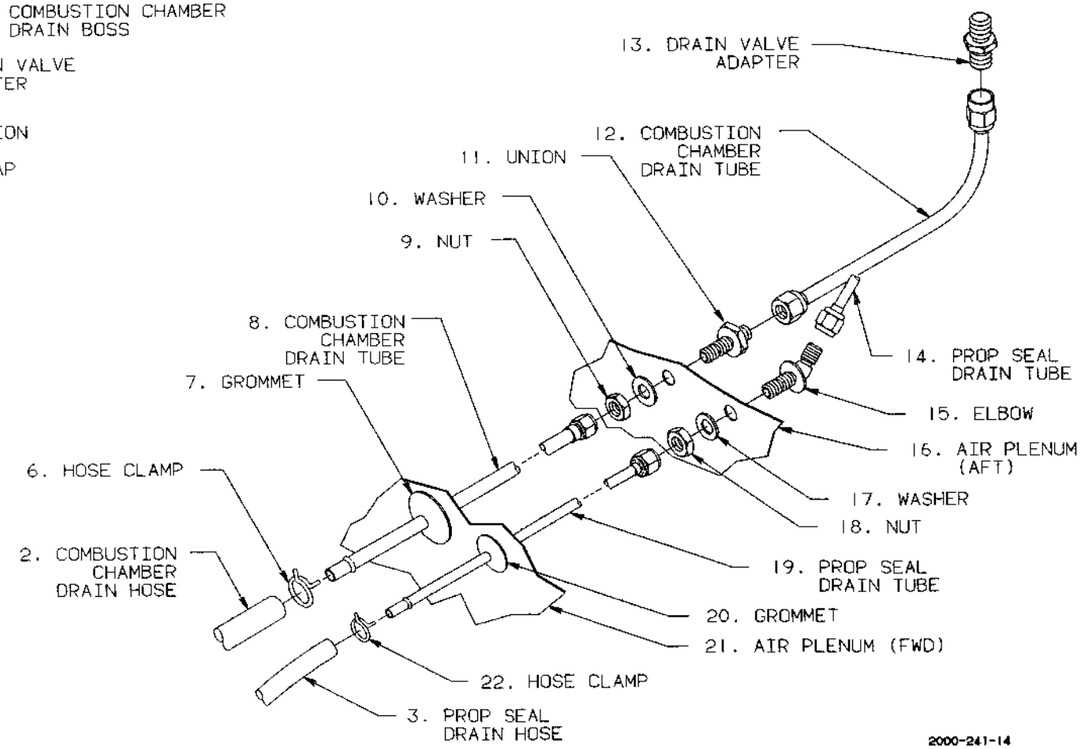

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DETAIL C

Engine Drain Tubing Removal/Installation (Effectivity: All)
Figure 3

CHAPTER 72 - ENGINE

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STARSHIP 1
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CHAPTER 72 - ENGINE

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General - Maintenance Practices (Effectivity: All) 1

GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

GENERAL - MAINTENANCE PRACTICES (Effectivity: All)

NOTE

Refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual for detailed information on these subjects.

CHAPTER 73 - ENGINE FUEL AND CONTROL

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CHAPTER 73 - ENGINE FUEL AND CONTROL

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Fuel Manifold Purge System (Effectivity: All) 1
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Fuel Distribution - Description and Operation (Effectivity: All) 1
Engine-Driven Fuel Boost Pump (Effectivity: All) 1
Main Fuel Filter (Effectivity: All) 1
Fuel Tubing and Hose Assemblies (Effectivity: All) 2
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Fuel Distribution - Maintenance Practices (Effectivity: All) 3
Engine-Driven Fuel Boost Pump Removal (Effectivity: All) 3
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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

This chapter covers the left and right fuel distribution and fuel manifold purge systems from the inboard forward firewalls to the engine. It does not include the engine fuel systems and components covered in the Pratt & Whitney PT6A-67A Engine Maintenance Manual.

Fuel flow indicating is part of the Integrated Engine Indicating System described in Chapter 77-40-00. The left and right engine fuel flow is indicated to the operator through a visual readout of the fuel flow (in pounds-per-hour) on the Engine Instrument, Crew Alerting System (EICAS) display.

FUEL DISTRIBUTION (Effectivity: All)

The section on fuel distribution describes the left and right fuel systems to and from the engine accessory compartment in the engine nacelle area. Included are the fuel supply to the engine-driven fuel boost pump, the motive fuel flow from the engine-driven fuel boost pump, and the fuel/air purge system for purging the engine fuel pump during engine ignition and starting.

NOTE

For continuation of the engine fuel supply, motive fuel flow and the engine fuel pump purge systems, refer to the applicable sections of Chapter 28, Fuel.

FUEL MANIFOLD PURGE SYSTEM (Effectivity: All)

The fuel distribution section also contains a description of the engine fuel manifold purge system, which is an independent system and fully contained in each engine compartment. The purpose of the system is to purge (evacuate) the engine fuel manifold during engine shutdown.

RECOMMENDED MATERIALS (Effectivity: All)

The recommended materials listed in Chart 1 as meeting federal, military or supplier specifications are provided for reference only and are not specifically prescribed by Raytheon Aircraft Company. Any product conforming to the specification listed may be used. The products included in this chart have been tested and approved for aviation usage by Raytheon Aircraft Company, by the supplier, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the chart. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the supplier of the product to be used.

CHART 1
RECOMMENDED MATERIALS (Effectivity: All)

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
1. Solvent, Cleaning (Mineral Spirits)	PD680, Type III		Obtain locally

FUEL DISTRIBUTION - DESCRIPTION AND OPERATION (EFFECTIVITY: ALL) (FIGURE 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

Components in the fuel distribution systems are the same for the left and right systems and include the engine-driven fuel boost pumps, the main fuel filters and necessary tubing and hose assemblies.

The self-contained purge system for the engine fuel manifold is connected to the fuel flow divider and purge valve assembly of the left and right engines. Each system consists of a purge tank, two check valves, an air filter and connecting hose assemblies.

ENGINE-DRIVEN FUEL BOOST PUMP (EFFECTIVITY: ALL)

The engine-driven fuel boost pump is a self-priming, positive displacement, vane-type fuel pump. The pump incorporates a pressure relief valve and a bypass valve enclosed within a valve cover. The pressure relief valve is an adjustable spring-loaded poppet-type valve which operates from a flexible diaphragm. The pressure relief valve is factory adjusted and does not require additional adjustment. The bypass valve is a lightly spring-loaded poppet-type valve with a fixed cracking pressure. The bypass valve allows fuel from the standby fuel boost pump to pass through the engine-driven fuel boost pump during engine ignition and starting, and/or in the event of an engine-driven fuel boost pump failure.

The purpose of the engine-driven fuel boost pump is twofold. It provides motive fuel flow for the primary and transfer jet fuel pumps in the fuel tanks and delivers fuel to the engine fuel pump for engine operation. Each engine-driven fuel boost pump is mounted on the drive pad of the engine external scavenge oil pump. Rotation of the engine-driven fuel boost pump is clockwise facing the pump spline.

MAIN FUEL FILTER (EFFECTIVITY: ALL)

The purpose of the main fuel filters is to filter out impurities from the fuel prior to the fuel entering the engine-driven fuel boost pump and engine fuel components. The filter consists of a bypass valve, a contamination indicator, a filter bowl, a filter element and a drain valve.

The filter element filtration is 20 microns nominal. Fuel flow entering the filter passes through the filter element. When the element becomes restricted from contamination and the differential pressure across the filter reaches 1.4±0.2 psid, the bypass valve will open and the red contamination indicator will extend. The contamination indicator can be reset after the contamination has been removed.

A fuel drain valve is installed in the bottom of the left and right filter bowl and extends through a cutout (hole) in a hinged access door (411BB left/421BB right) on the engine lower fairing. The purpose of the drain valve is to allow the draining of moisture and sediments, and/or to take fuel samples for evaluation. The drain valve is opened by rotation of the valve cross-pin lever and can be locked in the open position.

FUEL TUBING AND HOSE ASSEMBLIES (EFFECTIVITY: ALL)

Fuel is carried to and from the engines through tubing and hose assemblies. The airplane fuel system enters the engine nacelle area through the inboard forward firewall area of each engine.

Engine fuel is routed from the inboard forward firewall to the main fuel filter through a tube assembly. A hose assembly connects the main fuel filter to the engine-driven fuel boost pump.

NOTE

Fuel to the engine is either from the standby fuel boost pump or from motive fuel originating at the engine-driven fuel boost pump through the primary fuel jet pump and returning back to the inboard forward firewall.

Tubing routes fuel from the engine-driven fuel boost pump through a tee fitting to the oil-to-fuel heater. The motive fuel flow hose between the tee fitting and the inboard forward firewall incorporates a reducer fitting.

The left and right engine fuel pumps are purged of air during engine starts. A purge hose is connected between each engine fuel pump and the respective inboard forward firewall. Purged fuel/air is allowed to flow through the hose when the purge valve located in the wing is energized open during the engine ignition and start cycle.

NOTE

For description and maintenance of the standby fuel boost pump, primary fuel jet pump and the purge valve, refer to Chapter 28-20-00.

ENGINE FUEL MANIFOLD PURGE SYSTEM (EFFECTIVITY: ALL)

The engine fuel manifold purge system is designed to ensure that any residual fuel in the fuel manifolds is consumed during engine shutdown.

The fuel manifold purge tank is mounted on the left side of each engine oil cooler assembly with brackets at the forward and aft oil cooler flanges. The tank is secured at each bracket with clamps. Connected to the inlet of the purge tank is a check valve and an in-line air filter. An aft hose assembly connects the air filter to a boss on the engine bleed air discharge tube located on the right side of the engine. An elbow fitting at the purge tank outlet permits connecting a forward hose assembly between the tank and a check valve at the engine flow divider and purge valve assembly. The check valve at the flow divider and purge valve assembly is to prevent fuel from entering the purge tank if the soft seat check valve in the flow divider and purge valve assembly fails.

Fuel distribution begins at the engine inboard forward firewall and the engine fuel supply tube connection. During engine start, fuel flow through the engine fuel supply line is supplied by the standby fuel boost pump. Fuel flows through the main fuel filter, engine-driven fuel boost pump, oil-to-fuel heater, and the engine fuel pump. The wing-mounted purge valve is energized open and fuel purges the engine fuel pump of air. Purged fuel/air is routed to and through the inboard forward firewall and the purge valve, then routed into the aft fuel tank.

After engine start, the ENGINE IGNITION AND START switch is positioned OFF, the purge valve deenergizes closed and the standby fuel boost pump is deactivated. The engine-driven fuel boost pump is then providing fuel flow to the engine, as well as, motive fuel flow for operation of the primary jet fuel pump. Refer to Chapter 28-20-00 for operation of the primary jet fuel pump.

During engine operation, engine bleed air is directed through an air filter and check valve into a purge tank from the bleed air discharge tube. The bleed air is maintained in the purge tank by the check valve at the tank inlet and by the flow divider and purge valve assembly located under the engine.

At engine shutdown, fuel flow is cut off at the fuel control unit. Without fuel pressure at the flow divider, a soft seat check valve between the flow divider and purge valve opens. Compressed bleed air is released from the purge tank through the flow divider into the engine fuel manifold. The compressed air forces residual fuel from the manifold, out the fuel nozzles into the combustion chambers.

FUEL DISTRIBUTION - MAINTENANCE PRACTICES (EFFECTIVITY: ALL)

ENGINE-DRIVEN FUEL BOOST PUMP REMOVAL (EFFECTIVITY: ALL) (FIGURES 1 AND 2)

Removal of the left and right engine-driven fuel boost pump is typical. The left/right firewall fuel shutoff valve must be closed prior to disconnecting any fuel tubing/hoses to prevent gravity flow of fuel from the fuel tanks.

CAUTION

Observe all fire precautions and safety practices when working on the fuel system. Refer to Chapter 28-00-00.

a. Gain access to the engine-driven fuel boost pump by removing the engine cowling as outlined in Chapter 71-10-00.

CAUTION

The engine fire extinguisher system is ARMED when the ENG FIRE F/W VALVE PUSH CLOSED light/switch is actuated. DO NOT ACTUATE THE EXTINGUISHER PUSH light/switch with the system ARMED. Deployment of fire extinguishing agent will result.

b. Close the firewall fuel shutoff valve by applying electrical power to the airplane and actuating the left/right ENG FIRE F/W VALVE PUSH CLOSED light/switch. Confirm that the light/switch illuminates CLOSED. DO NOT ACTUATE THE EXTINGUISHER PUSH LIGHT/SWITCH.

c. Remove electrical power from the airplane.

d. At the inboard forward firewall (Figure 1), place a container under the fuel line connections. Disconnect the motive fuel flow hose from the inboard forward firewall fitting, and the engine fuel supply hose from the main fuel filter and drain the residual fuel from the hoses. After draining the fuel, reconnect the hoses.

e. Disconnect the engine fuel supply hose from the inlet elbow fitting of the pump.

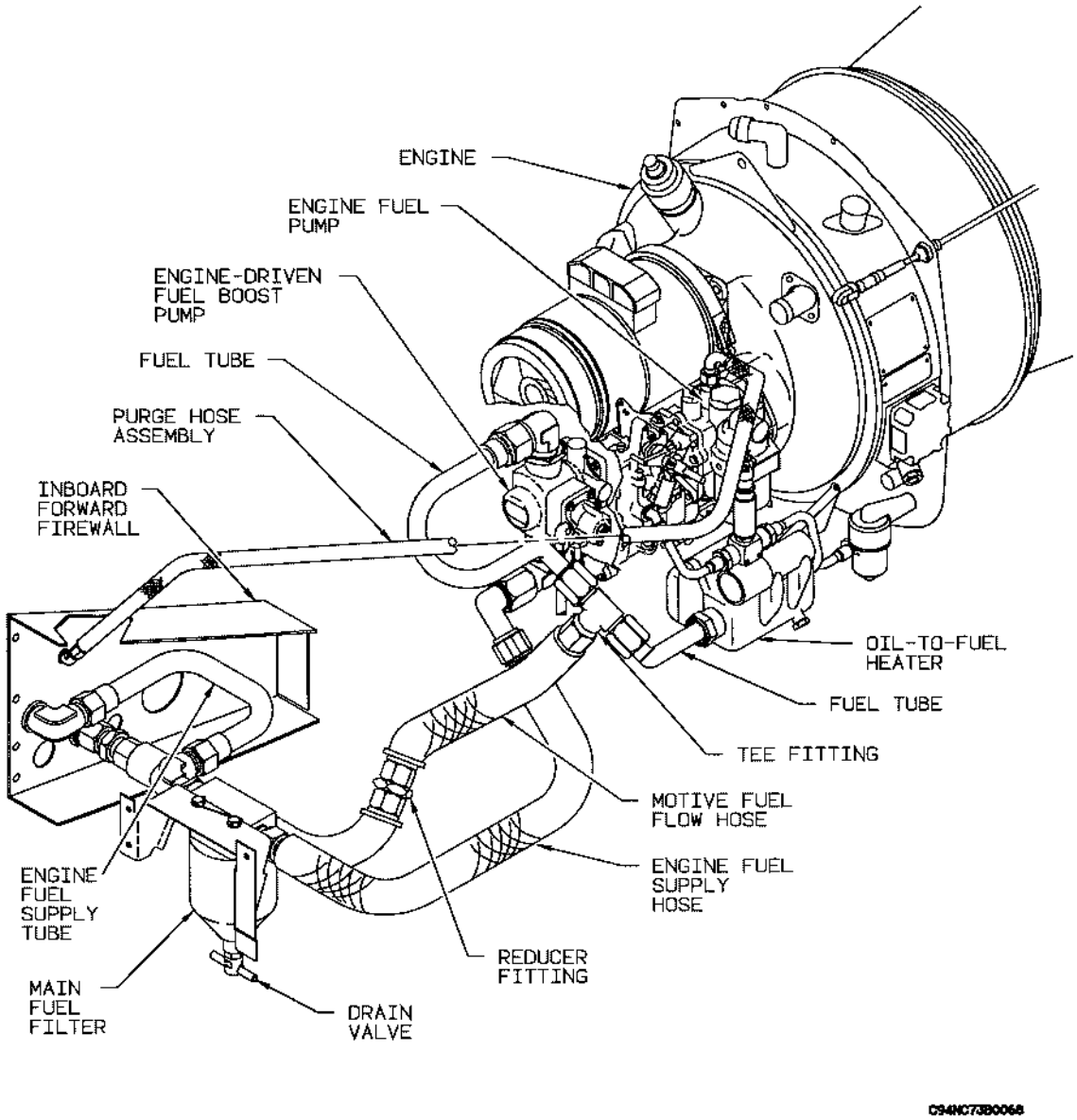
f. Disconnect and remove the motive fuel flow tube from the outlet elbow fitting of the pump and the tee fitting.

g. Cap or cover the fuel tube/hose openings to prevent entry of foreign matter.

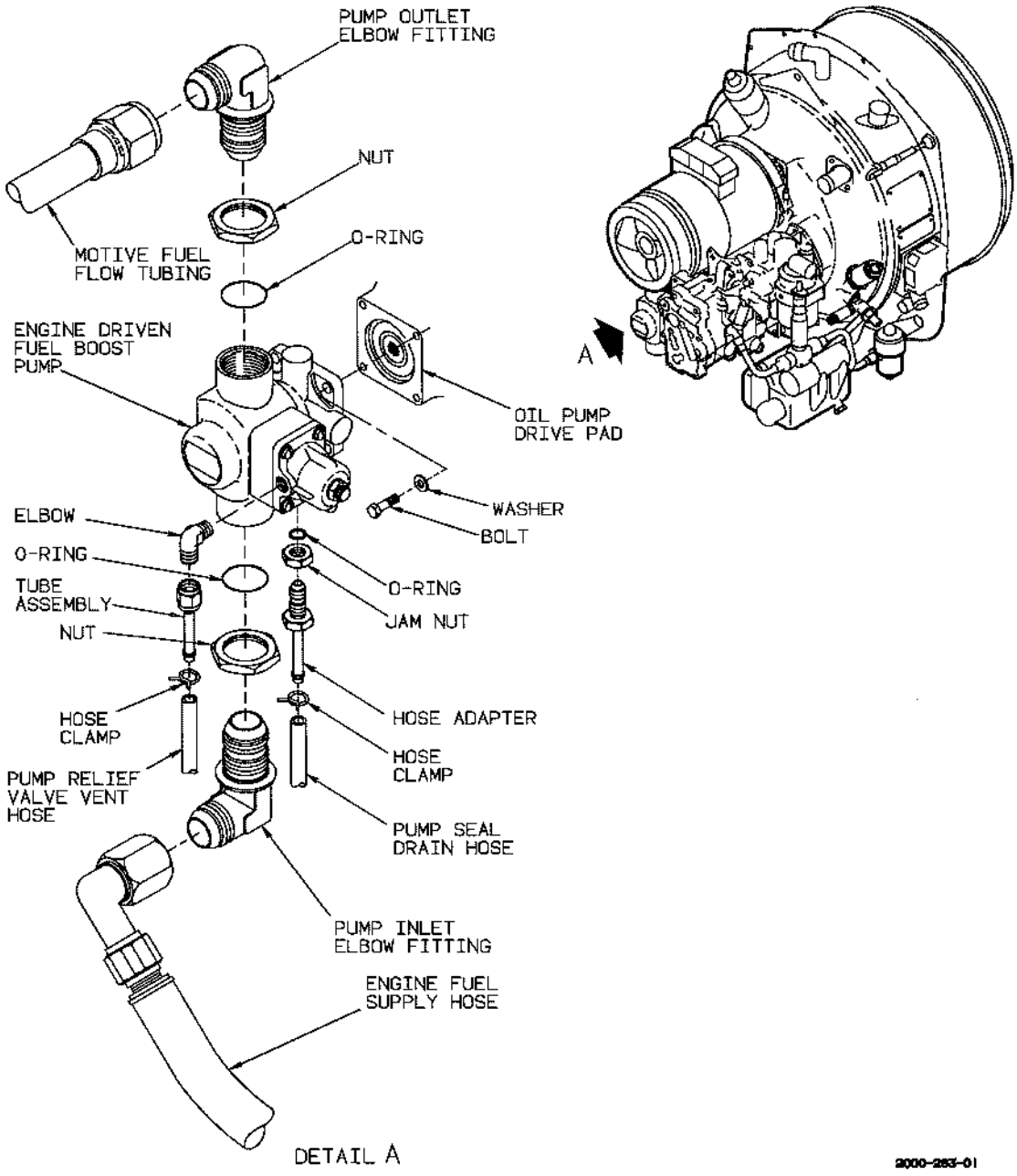
h. Disconnect the pump relief valve vent hose from the tube assembly and the pump seal drain hose from the hose adapter by removing the hose clamps.

i. Remove the four mounting bolts and washers securing the engine-driven fuel boost pump to the drive pad on the engine external oil scavenge pump.

j. Remove the engine-driven fuel boost pump.



Fuel Distribution Components (Effectivity: All)
Figure 1



Engine-Driven Fuel Boost Pump Installation (Effectivity: All)
Figure 2

ENGINE-DRIVEN FUEL BOOST PUMP INSTALLATION (EFFECTIVITY: ALL) (FIGURES 1 AND 2)

a. If the engine-driven fuel boost pump to be installed is a new pump, accomplish the following:

1. Remove the pump inlet and outlet elbow fittings, the pump seal drain hose adapter, the pump relief valve vent tube assembly and elbow fitting from the removed engine-driven fuel boost pump.

2. On the new engine-driven fuel boost pump, position and install the pump inlet and outlet elbow fittings with new O-rings. Install the pump relief valve tube elbow fitting and tube assembly in the vent port. Install the pump seal drain hose adapter with a new O-ring and secure with jam nut.

b. Position the pump on the external oil scavenge pump drive pad and secure the pump with four mounting bolts and washers. Torque the mounting bolts to 40-50 inch-pounds. Safety wire the mounting bolts.

c. Connect the pump seal drain hose to the hose adapter with a hose clamp. Connect the pump relief valve vent hose to the tube assembly with a hose clamp.

d. Connect the motive fuel flow tubing to the pump outlet elbow fitting and the tee fitting.

e. Connect the engine fuel supply hose to the pump inlet elbow fitting.

f. Apply electrical power to the airplane and actuate the left/right ENG FIRE F/W VALVE PUSH CLOSED light/switch to open the firewall fuel shutoff valve. Confirm that the light/switch extinguishes.

g. Operate the left/right standby fuel boost pump to pressurize the fuel lines.

h. Check all fuel lines and fittings for fuel leakage.

i. Turn the standby fuel boost pump OFF and remove electrical power from the airplane.

j. Install the engine cowling as outlined in Chapter 71-10-00.

MAIN FUEL FILTER ASSEMBLY REMOVAL (EFFECTIVITY: ALL) (FIGURE 3)

The left and right main fuel filters are mounted on a bracket attached to the engine firewall at the nacelle centerline.

The left/right firewall fuel shutoff valve must be closed prior to disconnecting the fuel tube/hose to prevent gravity flow of fuel from the fuel tanks. Removal of the left and right main fuel filters is typical.

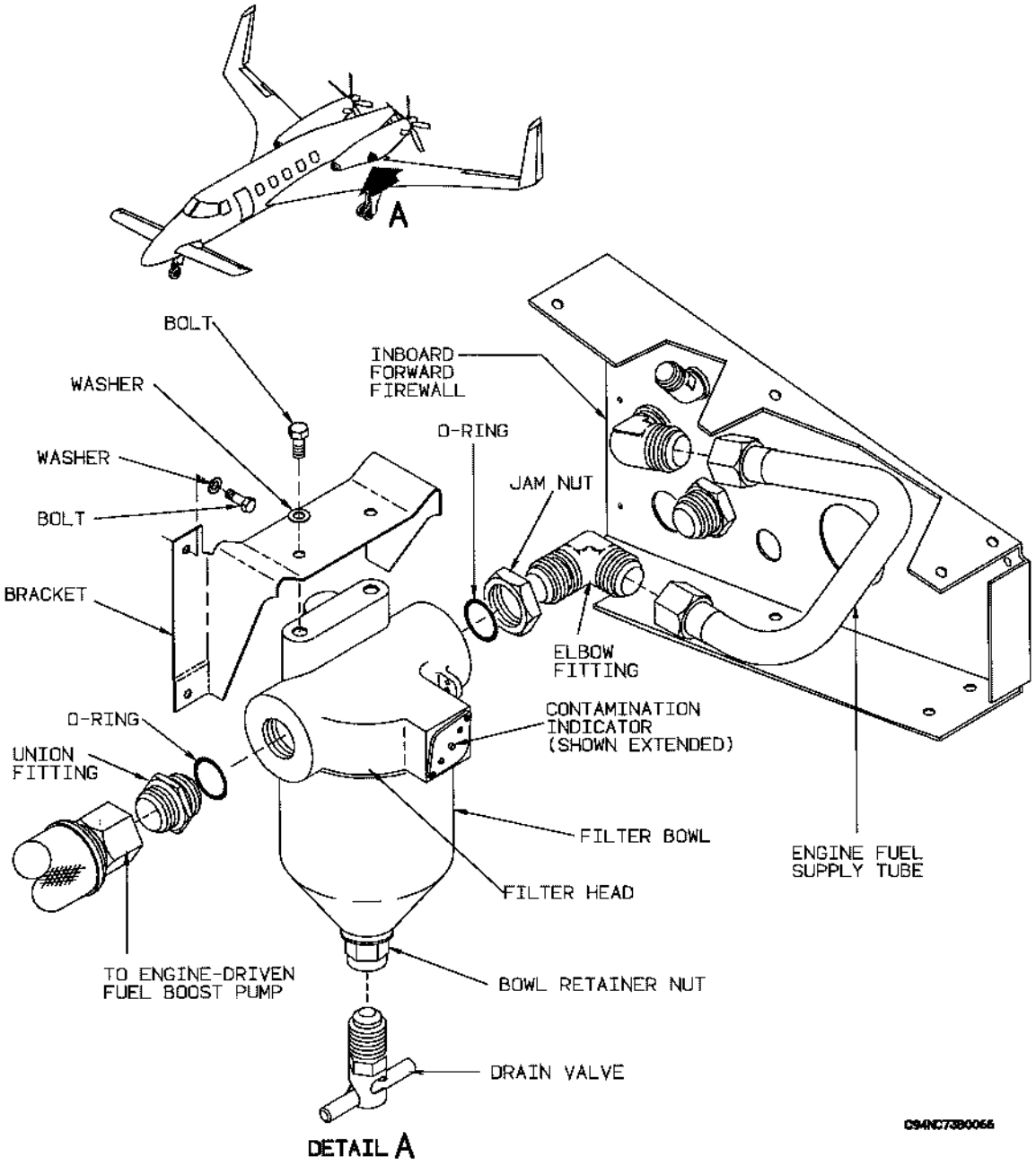
CAUTION

Observe all fire precautions and safety practices when working on the fuel system. Refer to Chapter 28-00-00.

NOTE

For scheduled periodic inspection and cleaning of the fuel filter and element, refer to Chapter 5-20-00 for the cleaning intervals.

a. Gain access to the filter assembly through the hinged access door (411BB or 421BB) on the engine lower fairing or remove the engine lower fairing.



Main Fuel Filter Assembly Installation (Effectivity: All)
Figure 3

CAUTION

The engine fire extinguisher system is ARMED when the ENG FIRE F/W VALVE PUSH CLOSED light/switch is actuated. Do not actuate the EXTINGUISHER PUSH light/switch with the system ARMED. Deployment of the fire extinguishing agent will result.

b. Close the firewall fuel shutoff valve by applying electrical power to the airplane and actuating the left/right ENG FIRE F/W VALVE PUSH CLOSED light/switch. Confirm that the light/switch illuminates CLOSED. DO NOT ACTUATE THE EXTINGUISHER PUSH LIGHT/SWITCH.

c. Remove electrical power from the airplane.

d. With an appropriate container under the main fuel filter drain valve, open the filter drain valve and drain all the residual fuel from the filter assembly.

e. Disconnect the fuel hose from the union fitting (filter outlet) and the engine fuel supply tube from the elbow fitting (filter inlet).

f. Cut safety wire and remove the two mounting bolts and washers securing the filter to the bracket.

g. Remove the filter assembly from the airplane.

h. If the filter assembly is not to be installed immediately, cover the open ends of the fuel lines to prevent entry of foreign matter.

MAIN FUEL FILTER ASSEMBLY INSTALLATION (EFFECTIVITY: ALL) (FIGURE 3)

a. If a new filter assembly is to be installed, remove the union and elbow fittings from the removed filter assembly and install them on the new filter assembly with new O-rings. Position the elbow fitting as it was installed on the removed filter.

b. Connect the fuel hose to the union fitting (filter outlet) and the engine fuel supply tube to the elbow fitting (filter inlet). Check that the contamination indicator is not extended and the drain valve is closed. If the contamination indicator is extended refer to MAIN FUEL FILTER SERVICING in this section.

c. Position the filter assembly to the bracket and secure with two bolts and washers. Safety wire the bolt heads together.

d. Open the firewall fuel shutoff valve by applying electrical power to the airplane and actuating the left/right ENG FIRE F/W VALVE PUSH CLOSED light/switch OFF. Confirm that the light/switch extinguishes.

e. Operate the left/right standby fuel boost pump to pressurize the engine fuel supply line.

f. Check for fuel leakage at the fuel filter assembly.

g. Turn the standby fuel boost pump OFF and remove electrical power from the airplane.

h. Install the engine lower fairing or close the hinged access door (411BB or 421BB).

MAIN FUEL FILTER DRAIN VALVE REMOVAL (EFFECTIVITY: ALL) (FIGURE 3)

The main fuel filter drain valve is threaded into the bottom of the fuel filter bowl. The left/right firewall fuel shutoff valve must be closed prior to draining fuel from the filter or removal of the fuel drain valve to prevent gravity flow of fuel from the fuel tanks. Removal of the left and right filter drain valve is typical.

a. Gain access to the main fuel filter through the hinged access door on the engine lower fairing.

CAUTION

The engine fire extinguisher system is ARMED when the ENG FIRE F/W VALVE PUSH CLOSED light/switch is actuated. Do not actuate the EXTINGUISHER PUSH light/switch with the system ARMED. Deployment of the fire extinguishing agent will result.

b. Close the firewall fuel shutoff valve by applying electrical power to the airplane and actuating the left/right ENG FIRE F/W VALVE PUSH CLOSED light/switch. Confirm that the light/switch illuminates CLOSED. DO NOT ACTUATE THE EXTINGUISHER PUSH LIGHT/SWITCH.

c. Remove electrical power from the airplane.

d. With a container under the fuel filter, lock the filter drain valve OPEN. Drain all residual fuel from the filter and the attaching fuel hose and tube.

e. With a wrench on the hex portion of the drain valve, rotate the valve counterclockwise from the filter bowl.

MAIN FUEL FILTER DRAIN VALVE INSTALLATION (EFFECTIVITY: ALL) (FIGURE 3)

a. Thread the drain valve into the bottom of the filter bowl and tighten. Check that the drain valve is closed.

b. Open the left/right firewall fuel shutoff valve by applying electrical power to the airplane and actuating the left/right ENG FIRE F/W VALVE PUSH CLOSED light/switch OFF. Confirm that the light/switch extinguishes.

c. Operate the left/right standby fuel boost pump to pressurize the engine fuel supply line.

d. Check the filter drain valve for fuel leakage.

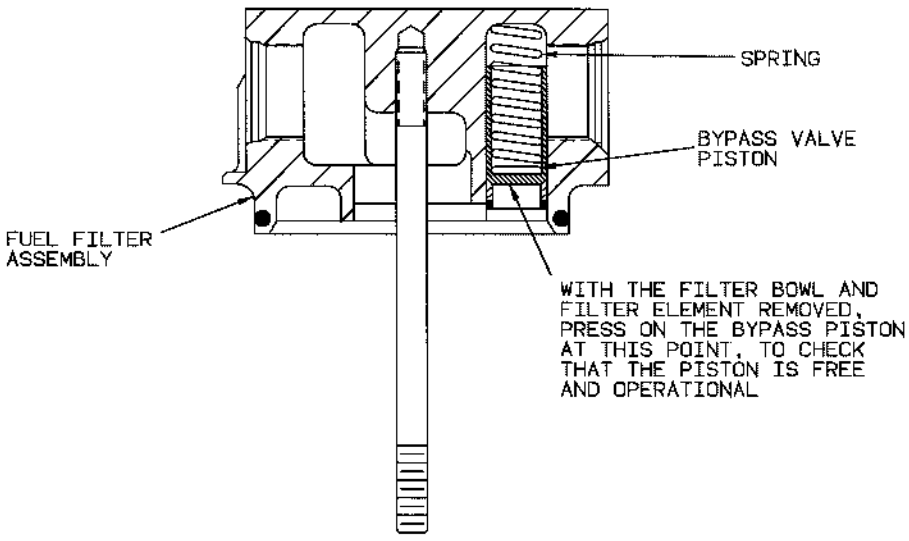
e. Turn the standby fuel boost pump OFF and remove electrical power from the airplane.

f. Close the hinged access door on the engine lower fairing.

MAIN FUEL FILTER SERVICING (EFFECTIVITY: ALL) (FIGURE 4)

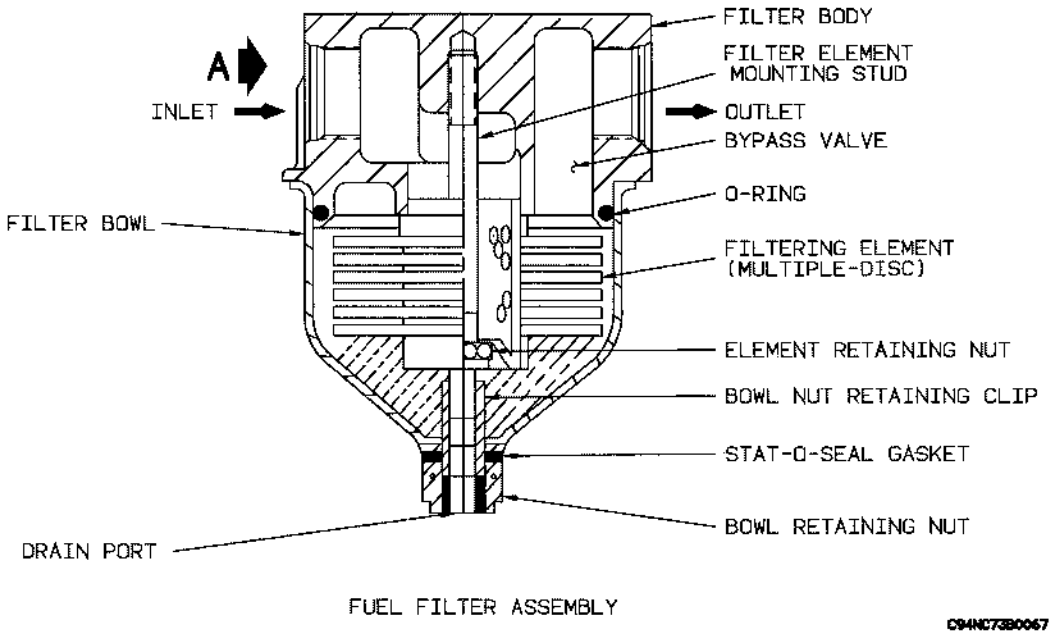
The main fuel filter can be serviced while installed in the airplane. The left/right firewall fuel shutoff valve must be closed prior to opening the fuel filter to prevent gravity flow of fuel from the fuel tanks.

a. Gain access to the main fuel filter through the hinged access door (411BB or 421BB) on the engine lower fairing.



SHOWN WITH FILTER ELEMENT AND BOWL REMOVED

DETAIL A



FUEL FILTER ASSEMBLY

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Main Fuel Filter Cutaway View (Effectivity: All)
Figure 4

CAUTION

The engine fire extinguisher system is ARMED when the ENG FIRE F/W VALVE PUSH CLOSED light/switch is actuated. Do not actuate the EXTINGUISHER PUSH light/switch with the system ARMED. Deployment of extinguishing agent will result.

b. Close the left/right firewall fuel shutoff valve by applying electrical power to the airplane and actuating the ENG FIRE F/W VALVE PUSH CLOSED light/switch. Confirm that the light/switch illuminates CLOSED. DO NOT ACTUATE THE EXTINGUISHER PUSH LIGHT/SWITCH.

c. Remove electrical power from the airplane.

d. With a container under the filter, lock the filter drain valve OPEN. Drain all residual fuel from the filter bowl.

e. Cut safety wire and remove the bowl retaining nut and filter bowl.

f. Remove the filter element retaining nut and filter element.

g. Clean the filter element, bowl and interior surface of the filter body with solvent, (1, Chart 1, 73-00-00). If the contamination indicator is extended, reset it by pressing on it.

CAUTION

Do not apply a force greater than 5 pounds to the piston of the bypass valve. Avoid scratching or contaminating the sealing surface when applying force to the piston.

h. Check that the bypass valve piston is free to move by applying a slight finger force to the piston or by using a blunt object. The bypass piston is springloaded and should be free to move in and out (approximately 3/16 inch to 1/4 inch) with approximately 1/2 pound of force. If the piston does not move freely, the fuel filter assembly needs to be replaced.

i. Position the filter element on the mounting stud and install the retaining nut to snug up the filter element, then retighten the retaining nut 1/4 turn clockwise.

j. Install a new O-ring in the groove around the filter body.

k. Install the filter bowl and bowl retaining nut and torque to 80-120 inch-pounds. Safety wire the filter bowl retaining nut to the filter body.

l. Open the firewall fuel shutoff valve by applying electrical power to the airplane and actuating the left/right ENG FIRE F/W VALVE PUSH CLOSED light/switch OFF. Confirm that the light/switch extinguishes.

m. Operate the left/right standby fuel boost pump to pressurize the engine fuel supply line.

n. Check the fuel filter for fuel leakage.

o. Turn the standby fuel boost pump OFF and remove electrical power from the airplane.

p. Close the hinged access door (411BB or 421BB) on the engine lower fairing.

ENGINE FUEL MANIFOLD PURGE SYSTEM REMOVAL (EFFECTIVITY: ALL) (FIGURE 5)

Removal of the left and right fuel manifold purge tanks is typical. The check valve and/or air filter can be removed and installed without removal of the purge tank. When the purge tank is to be removed or replaced, the check valve and air filter can be left connected to the purge tank during tank removal.

An additional check valve, installed on the engine flow divider and purge valve assembly is identical to the check valve on the purge tank inlet. For a new engine and the initial installation of the check valve on the flow divider and purge valve assembly, remove and discard the fitting supplied with the engine.

a. Gain access to the purge tank by removing the engine cowling as outlined in Chapter 71-10-00.

b. Disconnect the hose assembly from the air filter.

c. Disconnect the hose assembly at the tank outlet elbow.

d. Remove the two clamps securing the tank to the two mounting brackets.

e. Remove the tank, check valve and air filter together.

f. If the purge tank is not to be installed immediately, plug or cover the hose ends to prevent entry of foreign matter.

ENGINE FUEL MANIFOLD PURGE SYSTEM INSTALLATION (EFFECTIVITY: ALL) (FIGURE 5)

a. If the purge tank to be installed is a new tank, accomplish the following:

1. Remove the air filter and check valve from the removed purge tank inlet, and the elbow fitting from the outlet.

2. Install the elbow fitting and a new O-ring on the new purge tank outlet as it was positioned on the removed tank. Secure the elbow fitting with the jam nut.

3. Install the check valve and a new O-ring on the tank inlet, (flow arrow pointing toward the purge tank).

4. Install the air filter and a new O-ring on the check valve.

b. Position the purge tank to the two mounting brackets and secure the purge tank to the forward and aft brackets with two clamps.

NOTE

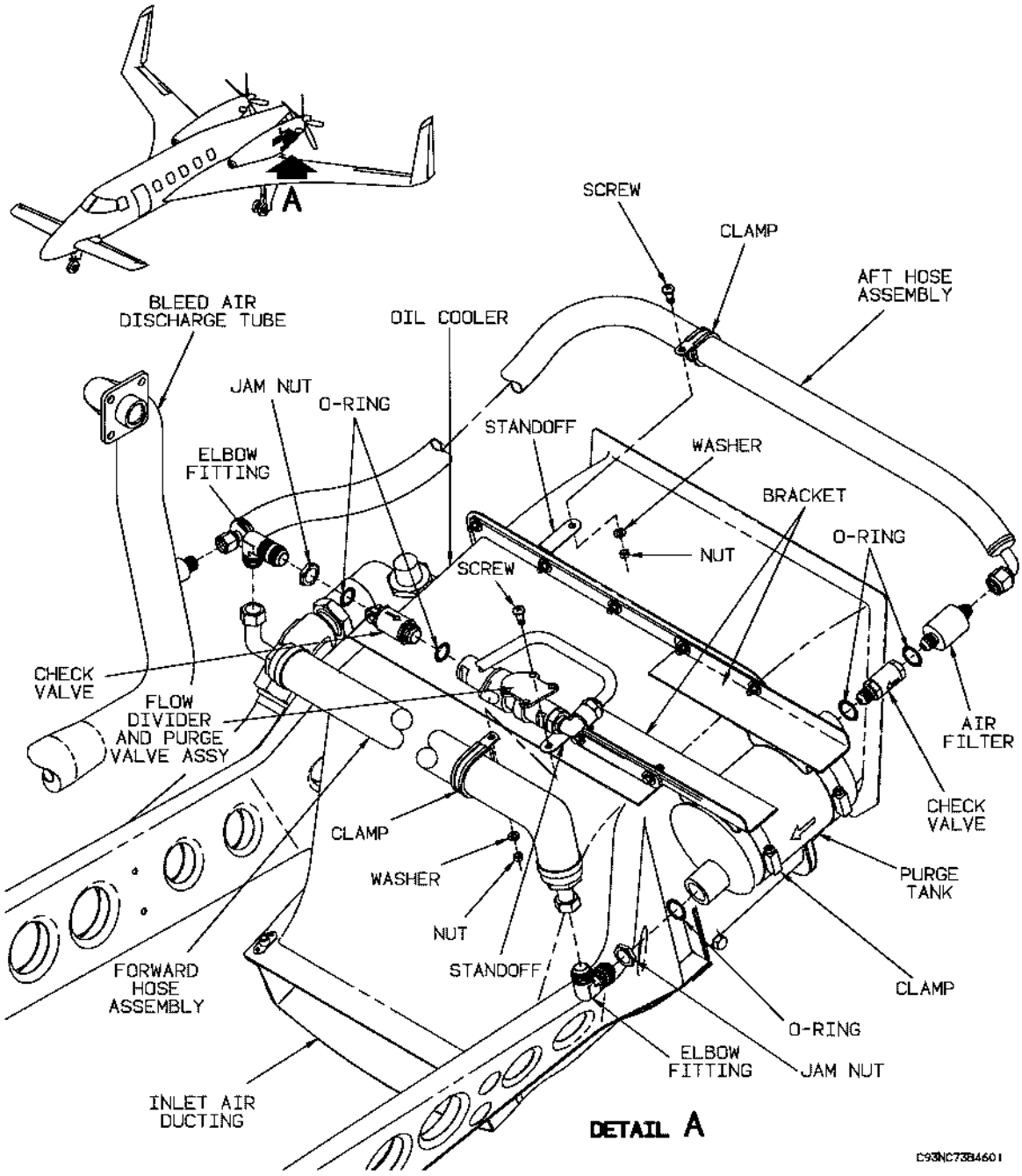
The forward and aft tank brackets are secured to the upper flanges of the oil cooler with existing oil cooler flange bolts.

NOTE

If this is the initial manifold purge system installation for a new engine, remove and discard the fitting supplied with the engine from the flow divider and purge valve assembly.

c. Install a check valve and new O-ring in the place of the discarded fitting, (flow arrow toward the flow divider and purge valve assembly).

d. Install an elbow fitting and new O-ring on the check valve. Secure the elbow fitting with the jam nut.



Engine Fuel Manifold Purge System Installation (Effectivity: All)
Figure 5

e. Connect the forward hose assembly between the elbow fitting and the purge tank outlet elbow fitting.

f. Connect the aft hose assembly to the air filter.

g. Install the engine cowling as outlined in Chapter 71-10-00.

MANIFOLD PURGE TANK CLEANING (EFFECTIVITY: ALL)

Cleaning the manifold purge tank is accomplished at specified intervals. Refer to Chapter 5-20-00 for the intervals.

a. Refer to ENGINE FUEL MANIFOLD PURGE SYSTEM REMOVAL in this section and remove the purge tank from its brackets.

b. Remove the check valve and air filter from the tank inlet and discard the O-rings.

c. Remove the elbow fitting from the tank outlet and discard the O-ring. Note the position of the elbow fitting for reinstallation.

d. Flush the purge tank with solvent (1, Chart 1, 73-00-00)

e. Blow dry the purge tank with dry shop air.

f. Install the elbow fitting with a new O-ring on the tank outlet as it was positioned prior to removal. Secure the elbow fitting with the jam nut.

g. Install the check valve and air filter with new O-rings on the tank inlet.

h. Refer to ENGINE FUEL MANIFOLD PURGE SYSTEM INSTALLATION in this section and install the purge tank.

MANIFOLD PURGE SYSTEM AIR FILTER CLEANING (EFFECTIVITY: ALL)

The manifold purge system air filter is located at the purge tank between the check valve and hose assembly. Inspection and cleaning of the air filter is accomplished at specified intervals, refer to Chapter 5-20-00 for the intervals. Inspection of the filter consist of checking for rust and corrosion. A filter that shows signs of rust and/or corrosion must be replaced.

a. Disconnect the aft hose assembly from the filter inlet.

b. Remove the filter from the check valve by holding the check valve with a wrench and rotating the filter counterclockwise.

c. Clean the filter with solvent (1, Chart 1, 73-00-00).

d. Blow dry the filter with dry shop air.

e. Install the filter and a new O-ring on the check valve.

f. Connect the aft hose assembly to the air filter.

FUEL PURGE SYSTEM CHECK VALVE INSPECTION, CLEANING AND LEAKAGE TEST (EFFECTIVITY: ALL) (FIGURES 6 AND 7)

NOTE

This procedure applies to both of the check valves located adjacent to each of the engine fuel purge tanks.

CAUTION

Do not disassemble check valves.

WARNING

Perform the following procedures in a protected area to avoid hazard to personnel.

Refer to Chapter 5-20-00 for inspection and cleaning intervals.

WARNING

Wear rubber gloves and eye protection when using solvents.

a. Remove both check valves.

b. Inspect for any foreign material and/or corrosion.

c. Using solvent (1, Chart 1, 73-00-00), pressure flush the check valve to remove carbon particles or sludge residue. This can be accomplished using a set up similar to that in Figure 6.

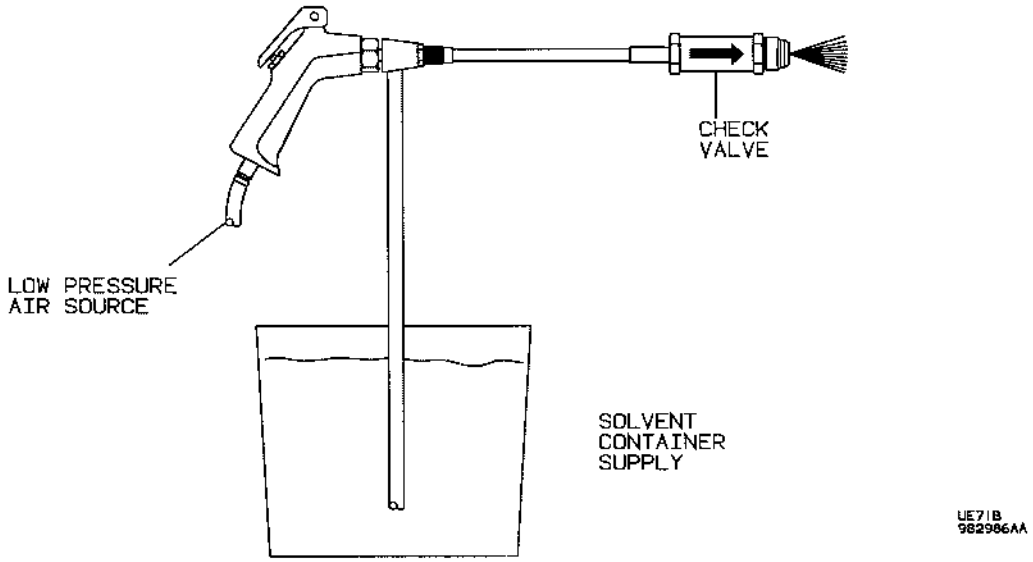
d. It is permissible to blow dry the check valve using shop air (80 to 120 psi) after assuring that all of the solvent is drained out.

e. After cleaning as described in step c, use a test set up as shown in Figure 7 to test the check valve for leakage.

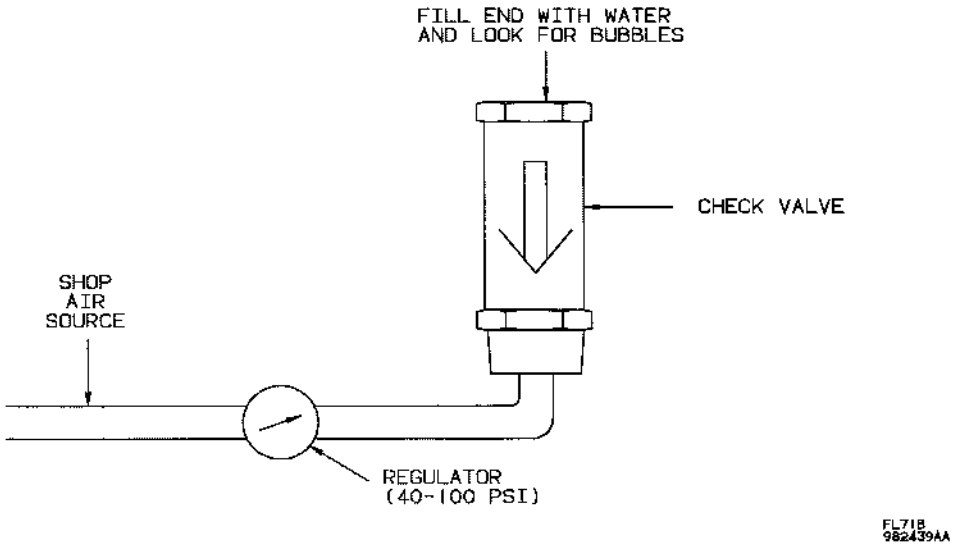
f. Apply a filtered regulated air source (40 to 100 psi) to the upstream side of the check valve.

g. Fill the downstream side of check valve with water and look for signs of leakage (bubbles). No leakage is allowed. If leakage is detected, repeat steps c through g. Replace the check valve if leakage is detected the second time.

h. It is permissible to blow dry the check valve with shop air (80 to 120 psi).



Check Valve Pressure Flush (Effectivity: All)
Figure 6



Check Valve Leakage Test (Effectivity: All)
Figure 7

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CHAPTER 74 - IGNITION

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CHAPTER 74 - IGNITION

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

IGNITION SYSTEM (Effectivity: All) (Figures 1, 2 and 3)

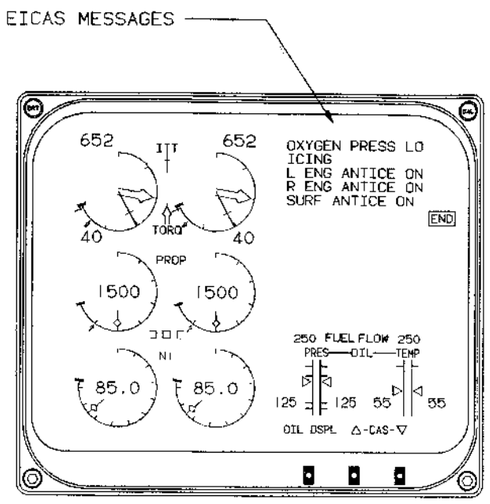
The Starship 1 ignition system components consist of two double-pole, three position ignition/start switches on the center subpanel, placarded IGNITION AND ENGINE START, LEFT-RIGHT, ON-OFF-STARTER ONLY; two igniter power circuit breakers on the left circuit breaker panel in the cockpit, placarded IGNITER POWER, LEFT-RIGHT; two ignition power relays mounted in the logic relay panel A129 on the aft fuselage equipment shelf, and the spark igniters, ignition exciters, and cable assemblies mounted in each engine compartment. The system is designed for the pilot to control both cranking and ignition operations with the left and right ignition/start switches. When the switches are placed in the ON position the system will initiate combustion, and once combustion is self-sustaining the ignition system is shut off by the pilot.

The ignition power relays, K3 (LH) and K13 (RH), receive triple-fed bus voltage from the ignition/start switches and auto-ignition circuit. They are responsible for distributing power from the igniter power circuit breakers to the ignition exciters.

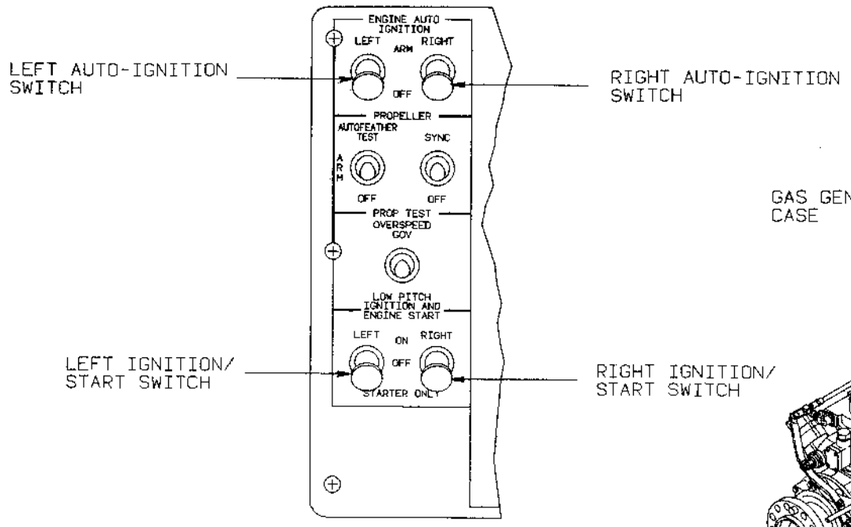
Each engine has its own ignition exciter mounted with clamps and a bracket on the engine truss mount. They are sealed units with the internal components encased in epoxy resin, and are designed to provide quick light-ups over a broad temperature range. The triple-fed bus provides 27.5 vdc under normal conditions, however the ignition exciters are capable of operating between 9 and 30 vdc. During the starting sequence, when the ignition exciter is energized, a capacitor on the high voltage side of the output transformer is progressively charged until the voltage potential reaches an adequate level to ionize the spark gap on the spark igniters. The signal is then divided and stepped-up before leaving the ignition exciter. The unit is designed to isolate a faulty spark igniter from the rest of the system, preventing a total failure of the system due to a problem in the opposite spark igniter. The network will allow the capacitor to discharge if both spark igniters are inoperative or if the input voltage is switched off.

When the ignition exciter receives a normal DC input, between 9 and 30 vdc, through the supply cable, it will emit a pulsed, high voltage output through two shielded, high tension, flexible cable assemblies, responsible for routing the electrical charge to the spark igniters. The cable assemblies are routed to the underside of the engine, where they pass through two fire seals by way of two ignition cable flanges bolted to the fireseals. Each cable assembly has coupling nuts to facilitate an easy and reliable connection.

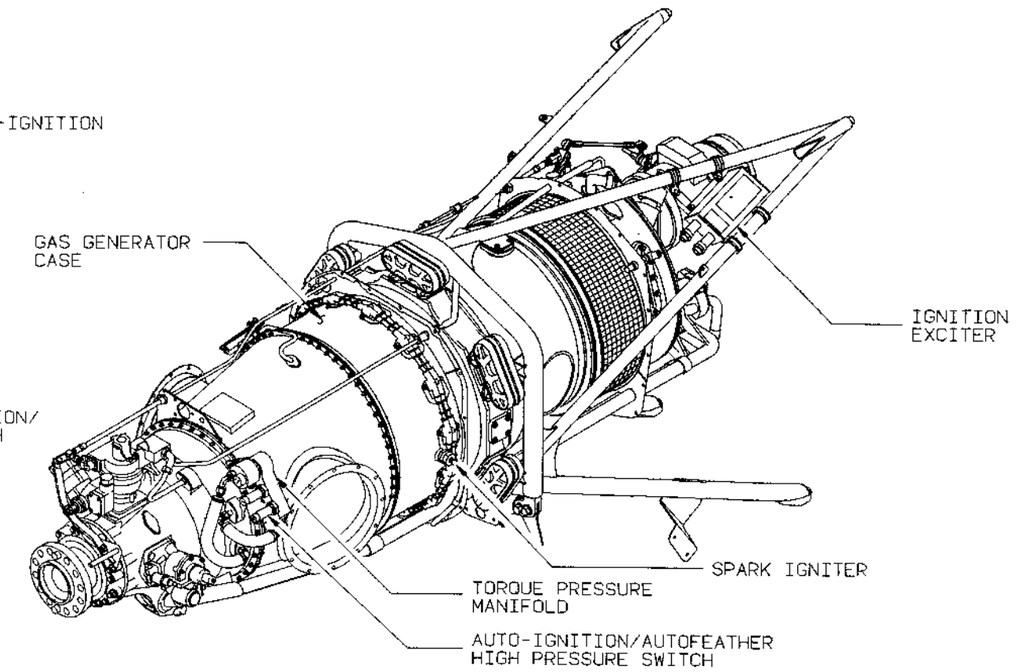
The two spark igniters are mounted at the four and nine o'clock positions on the gas generator case of each engine. They are double-ended, air-cooled, threaded plugs with a central positive electrode surrounded by a ring shaped semi-conducting material. The electrical signal generated by the ignition exciter is conducted by the cable assemblies to the spark igniters, where it builds a potential across the electrodes. As the potential increases to a preset threshold the air gap between the electrodes is ionized and a spark occurs during discharge. The spark will always occur between the central electrode and the outer ring of the unit. A copper gasket is used to provide an efficient seal between the spark igniter and the gas generator case.



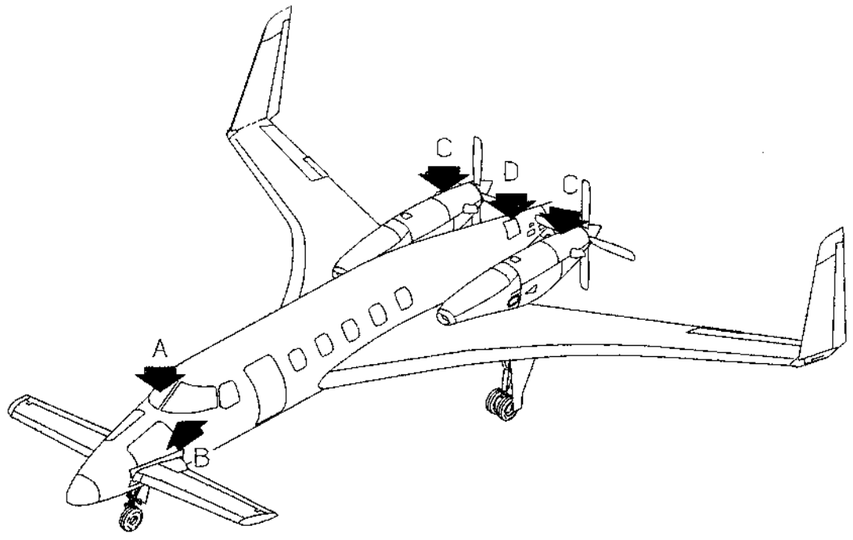
DETAIL A
EICAS DISPLAY

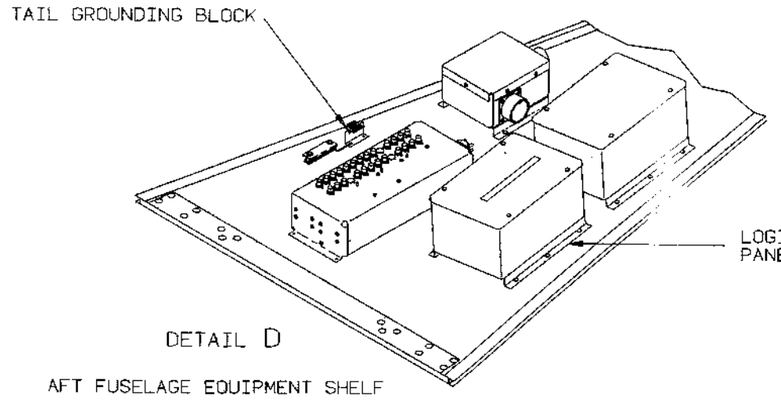


DETAIL B
CENTER SUBPANEL



DETAIL C
ENGINE





DETAIL D
AFT FUSELAGE EQUIPMENT SHELF

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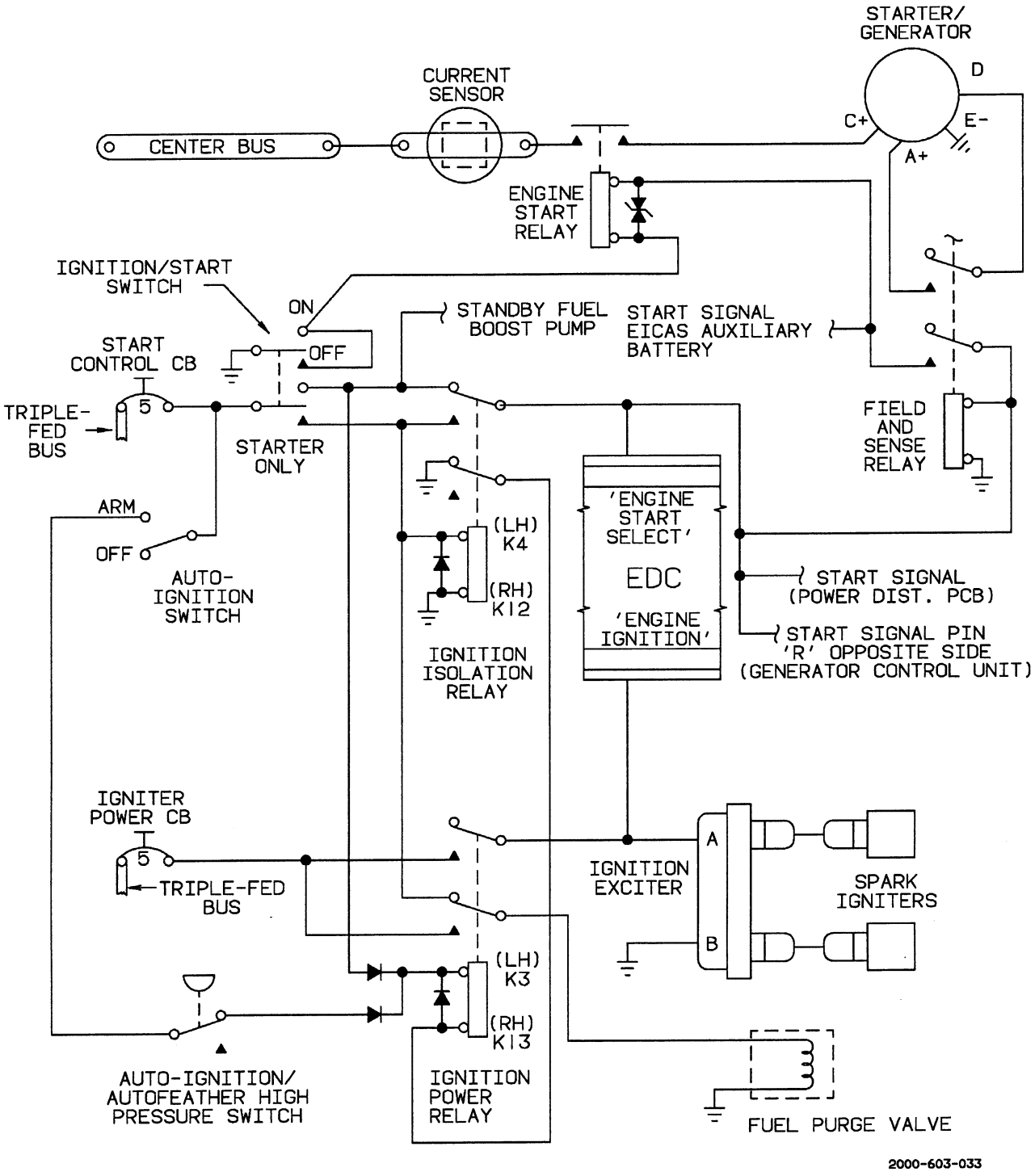
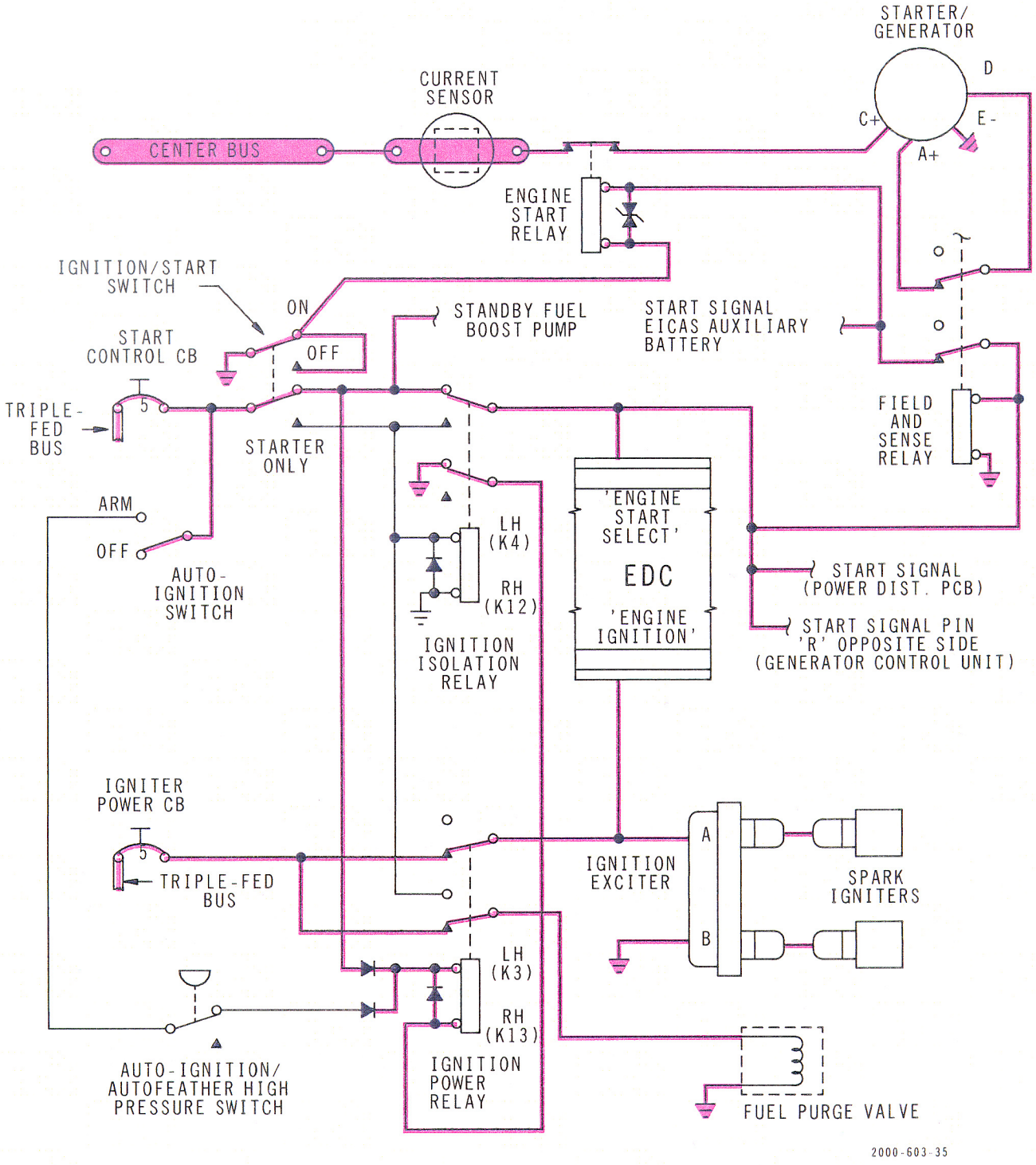
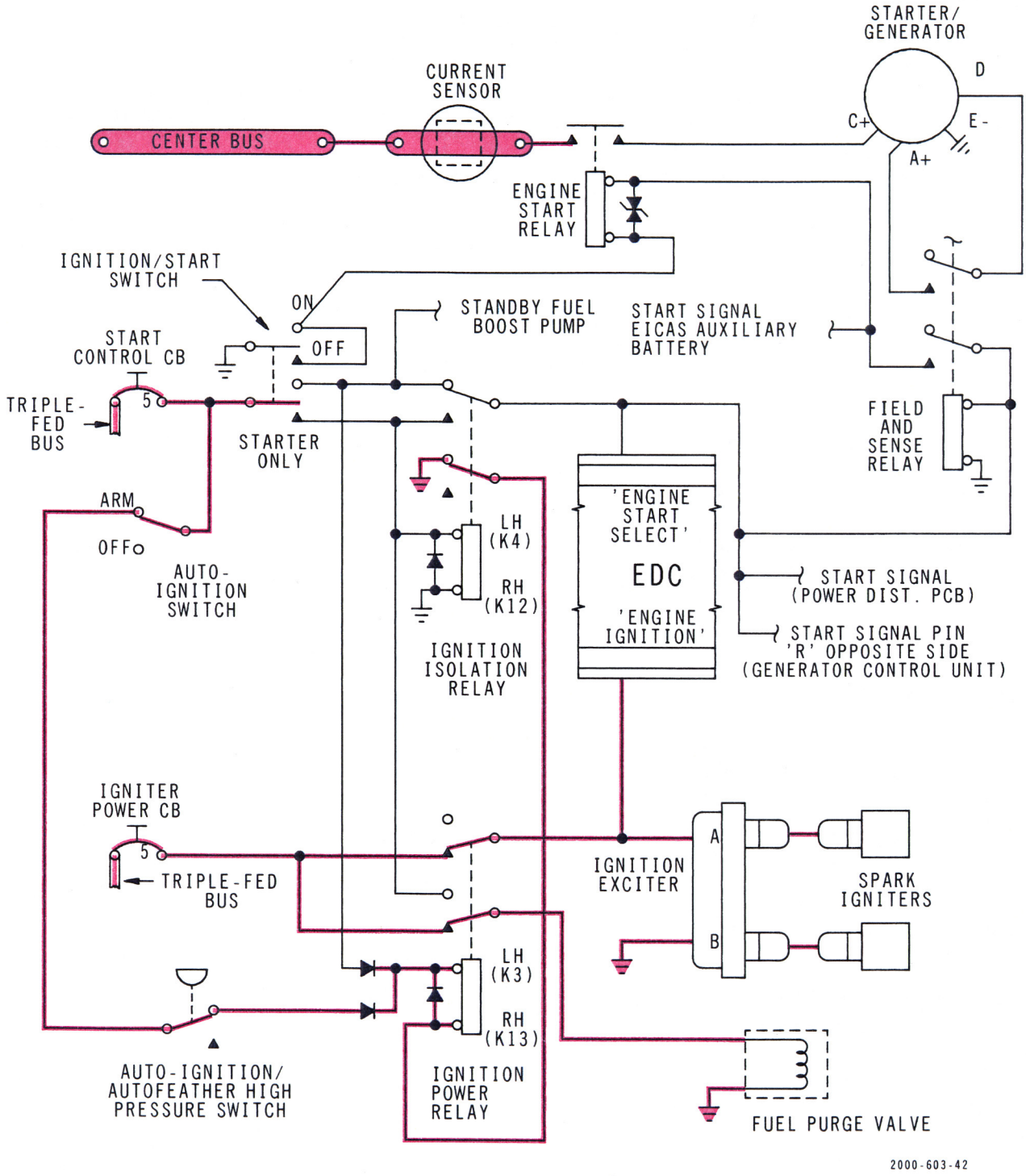
Engine Ignition System (Effectivity: All)
Figure 1



Ignition System Schematic (OFF) (Effectivity: All)
Figure 2



Ignition System Schematic (ON) (Effectivity: All)
Figure 3



Ignition System Schematic (Auto-Ignition On) (Effectivity: All)
Figure 4

IGNITION CIRCUIT (Effectivity: All)
(Figures 2 and 3)

NOTE

Both left and right sides of the ignition system are identical, but independent of each other. Due to the inherent relationship between the cranking and ignition systems they are shown together in the functional schematics.

The start control circuit breaker is powered by the triple-fed bus and supplies 28 vdc to one set of the ignition/start switch contacts. When the ignition/start switch is placed in the ON position, 28 vdc is provided through the ignition isolation and field and sense relays to energize the engine start relay, connecting the center bus to terminal C of the starter/generator, thereby motoring the engine. Refer to Chapter 80-00-00 for detailed information on the starting system. The ignition/start switch will also energize the ignition power relay, which is grounded through the ignition isolation relay. The igniter power circuit breaker provides 28 vdc from the triple-fed bus through the ignition power relay to energize the fuel purge valve and ignition exciter. A lead is spliced into the wiring between the ignition power relay and ignition exciter to provide a signal to the Engine Data Concentrator (EDC), initiating the green L IGNITION ON or R IGNITION ON advisory message on the Engine Instrument, Crew Alerting System (EICAS) display. The ignition exciter takes the input voltage and converts it to a pulsed, high voltage output and sends it through the shielded ignition cables to the spark igniter. When voltage reaches the level needed to ionize the air in the spark igniter spark gap, the ignition circuit is complete.

AUTO-IGNITION SYSTEM (Effectivity: All) (Figures 1 and 4)

The auto-ignition system is installed in the airplane to provide automatic engine ignition when engine power falls below 17% of maximum allowable torque. The auto-ignition system should be ARMED when encountering turbulence, precipitation, or icing conditions.

The auto-ignition switches are single-pole, single-throw, two position switches located on the center subpanel and placarded ENGINE AUTO-IGNITION, LEFT-RIGHT, ARM-OFF. The auto-ignition switch draws 28 vdc from the triple-fed bus through the start control circuit breaker. When placed in the ARM position, 28 vdc is routed through the switch to the auto-ignition/autofeather high pressure switch mounted on the torque pressure manifold adjacent to the engine reduction gearbox. The auto-ignition/autofeather high pressure switch will close when decreasing oil pressure in the torque pressure manifold reaches 3.1 ± .6 psi (17% of maximum allowable engine torque). When the contacts close, voltage is supplied to the ignition power relay, bypassing the ignition/start switch and automatically powering the ignition exciter. The Engine Instrument, Crew Alerting System (EICAS) green R IGNITION ON or L IGNITION ON advisory message will be displayed to indicate when the respective side is being powered. As combustion is resumed and torque begins to increase, the contacts on the auto-ignition/autofeather high pressure switch will reopen at 17% of maximum allowable torque, removing 28 vdc from the ignition circuit. The EICAS messages will no longer be displayed.

CAUTION

The auto-ignition switches should be placed in the OFF position whenever the engines are intentionally operated below 17% of maximum allowable torque. This will prevent unnecessary operation of the system and prolong the operational service life of the spark igniters.

SPECIAL TOOLS AND RECOMMENDED MATERIALS (Effectivity: All)

The special tools and recommended materials listed in Charts 1 and 2 as meeting federal, military or vendor specifications are provided for reference only and are not specifically prescribed by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
SPECIAL TOOLS AND EQUIPMENT (Effectivity: All)

TOOL NAME	PART NO.	VENDOR	USE
1.Multimeter	N/A	Obtain locally	Perform voltage and resistance checks

CHART 2
RECOMMENDED MATERIALS (Effectivity: All)

MATERIALS	SPECIFICATION	PRODUCT	VENDOR
1.Solvent,Dry Cleaner or White Spirits	PD680 or British Specification 245	Stoddard Solvent (Mineral Spirits)	Obtain Locally
2.Cleaner		Methyl Alcohol	Obtain Locally

GENERAL - TROUBLESHOOTING (Effectivity: All) (Figures 2, 3 and 4)

IGNITION SYSTEM (Effectivity: All)

The BEECHCRAFT Starship 1 Wiring Diagram Manual should be used to trace out specific wire routes and locate pin and plug numbers. Many times a loose or faulty pin connection will cause a fault. Ensure that there are no loose, damaged, or corroded pins on any connectors in the system. Continuity checks should be made with power removed from the specific connector or terminal. When making voltage checks ensure that the battery switch is in the ON position and the ignition/start switch is in the specified position. Battery voltage on the triple-fed bus can be approximated by pressing the TRIP FED volts switch on the center subpanel. The reading displayed on the voltmeter should be used as a reference when performing voltage checks in Chart 3. The steps in Chart 3 should be followed sequentially unless otherwise specified. An example would be to proceed to the next step when there are no directions immediately following a NO condition. Certain precautions should be observed when working with or around ignition related components and systems.

WARNING

Before attempting any maintenance on the ignition system, wait at least six minutes for any residual voltage to dissipate. Attempts to remove ignition components before the waiting period has elapsed may cause injury to personnel. Residual voltage in the ignition exciter is unpredictable; therefore, even after waiting for six minutes, further precautions must be taken. Always use insulated tools; never touch the outputs with bare hands; and disconnect the coupling nuts at the ignition exciter first.

Before using Chart 3 determine that the ignition exciter is not receiving a voltage input at pin A of the input connector. If battery voltage is at pin A and the system is not functional, check for continuity from pin B on the ignition exciter to the left or right nacelle ground block. If the system still does not function, check the cable assemblies for continuity with an ohmmeter. Refer to the Pratt & Whitney Engine Maintenance Manual for further troubleshooting of the ignition system. Refer to IGNITION EXCITER, SPARK IGNITER, AND CABLE ASSEMBLY FUNCTIONAL CHECK to perform a functional check of the ignition system after performing troubleshooting and maintenance operations.

CHART 3
TROUBLESHOOTING - IGNITION (Effectivity: All)

NO POWER TO THE IGNITION EXCITER (1)

STEP 1	Press the TRIP FED volts button on the center subpanel. Is a reading slightly less than battery voltage displayed on the voltmeter?	**YES**	Proceed to Step 2.
	NO		
	Refer to Chapter 24-50-00 to troubleshoot a possible triple-fed bus isolation.		
STEP 2	Is the igniter power circuit breaker open?	**YES**	Restore power to the system by closing the igniter power circuit breaker.
	NO		
STEP 3	Is the start control circuit breaker open?	**YES**	Restore power to the system by closing the start control circuit breaker.
	NO		
STEP 4	Is triple-fed bus voltage at pin 16 (LH) or pin 28 (RH) in the logic relay panel connector? (See Note 2.)	**YES**	Locate and correct break in circuit between logic relay panel connector and ignition exciter.
	NO		
STEP 5	Is triple-fed bus voltage from the igniter power circuit breaker at pin 12 (LH) or pin 24 (RH) in the logic relay panel connector?	**YES**	Proceed to Step 6.
	NO		
	Locate wiring break between the logic relay panel connector and igniter power circuit breaker. If no break is evident suspect a faulty igniter power circuit breaker.		
STEP 6	Is triple-fed bus voltage at pin 8 (LH) or pin 20 (RH) in the logic relay panel connector? (See Note 2.)	**YES**	Proceed to Step 10.
	NO		
STEP 7	Is triple-fed bus voltage at pin 26 (LH) or pin 21 (RH) in the center subpanel connector?	**YES**	Proceed to Step 8.
	NO		
	Locate break in wiring between the start control circuit breaker and center subpanel connector.		
STEP 8	Is there continuity between pins 24 and 26 (LH) or between pins 19 and 21 (RH) in the center subpanel connector? (See Note 2.)	**YES**	Proceed to Step 9.

CHART 3
TROUBLESHOOTING - IGNITION (Effectivity: All) *(Continued)*

NO POWER TO THE IGNITION EXCITER (1)

NO

Faulty ignition/start switch or a faulty connection at the switch terminals. Correct the fault.

STEP 9	Is there continuity between pin 24 (LH) or pin 19 (RH) in the center subpanel connector and pin 8 (LH) or pin 20 (RH) of logic relay panel connector? (See Note 2.)	**YES**	Proceed to Step 10.

NO

Locate break in wiring and restore continuity to the circuit.

STEP 10	Is there continuity between pin 12 (LH) or pin 24 (RH) in the logic relay panel connector and pin 1B in the ignition power relay connector?	**YES**	Proceed to Step 11.

NO

Restore continuity to wiring inside logic relay panel.

STEP 11	Is there continuity between pin 8 (LH) or pin 20 (RH) in logic relay panel connector and pin X1 in the ignition power relay connector?	**YES**	Proceed to Step 13.

NO

STEP 12	Is diode A129CR4 (LH) or A129CR2 (RH) in the logic relay panel A129 open?	**YES**	Remove and replace the faulty diode.

NO

Restore continuity to wiring inside of logic relay panel.

STEP 13	Is there continuity between the tail grounding block on the aft fuselage equipment shelf and pin X2 in the ignition power relay connector?	**YES**	Proceed to Step 18.

NO

STEP 14	Is there continuity between pin 10 (LH) or pin 22 (RH) and the tail grounding block on the aft fuselage equipment shelf?	**YES**	Proceed to Step 15.

NO

Restore continuity to ground.

STEP 15	Is there continuity between pin 10 (LH) or pin 22 (RH) and the ignition isolation relay connector pin 3A?	**YES**	Proceed to Step 16.

CHART 3
TROUBLESHOOTING - IGNITION (Effectivity: All) *(Continued)*

NO POWER TO THE IGNITION EXCITER (1)

	NO		
	Restore continuity to the connection inside logic relay panel.		
STEP 16	Is there continuity between pins 3A and 2A on the ignition isolation relay?	**YES**	Proceed to Step 17.
	NO		
	Faulty relay contacts. Replace relay.		
STEP 17	Is there continuity between pin 2A in ignition isolation relay connector and pin X2 in the ignition power relay connector?	**YES**	Proceed to Step 18.
	NO		
	Restore continuity to wiring inside logic relay panel.		
STEP 18	Is there continuity between pin 16 (LH) or pin 28 (RH) in logic relay panel connector and pin 2B in the ignition power relay connector?	**YES**	Proceed to Step 19.
	NO		
	Restore continuity to connection inside logic relay panel. System should be functional.		
STEP 19	Is there continuity between pin 16 (LH) or pin 28 (RH) and the exciter input connector pin A?	**YES**	Suspect a faulty ignition power relay. Remove and replace as required.
	NO		
	Restore continuity to the connection between the logic relay panel and ignition exciter. If the exciter still does not receive an input at pin A remove and replace the ignition power relay.		

1. Voltage checks should be made with the battery switch in the ON position. Continuity checks should be made with power removed from the circuit being checked.

2. The ignition/start switch should be in the ON position for this check.

GENERAL - MAINTENANCE PRACTICES (Effectivity: All)

NOTE

Refer to Chapter 24-50-00 for logic relay panel removal and installation.

SPARK IGNITER REMOVAL (Effectivity: All)

a. Ensure that both the auto-ignition and ignition/start switches are in the OFF position and pull the igniter power circuit breaker.

CAUTION

Ensure that all power is removed from the airplane before starting maintenance.

b. Refer to Chapter 71-10-00 and remove the engine cowling.

WARNING

Before attempting any maintenance on the ignition system, wait at least six minutes for any residual voltage to dissipate. Attempts to remove ignition components before the waiting period has elapsed may cause injury to personnel. Residual voltage in the ignition exciter is unpredictable; therefore, even after waiting for six minutes, further precautions must be taken. Always use insulated tools; never touch the outputs with bare hands; and disconnect the coupling nuts at the ignition exciter first.

CAUTION

Ensure that cable braiding, ferrules, and spark igniter remain stationary during coupling disconnection to prevent damage to the cable assembly.

c. Remove safety wire from the cable assembly at the ignition exciter output coupling, and disconnect the cable assembly.

d. Remove the safety wire between the cable assembly and spark igniter, and between the spark igniter and transfer tube locking plate.

e. Remove the cable assembly from the spark igniter.

f. Loosen the spark igniter and remove from the gas generator case.

SPARK IGNITER INSTALLATION (Effectivity: All)

a. If the same spark igniter is to be installed, discard the old copper gasket and replace with a new one.

CAUTION

Whenever a spark igniter has been dropped, there is a chance that it has been damaged in a way that is not detectable during ordinary testing procedures. Always replace a spark igniter that has been dropped. Never use thread lubricants when installing spark igniters.

b. Align the spark igniter with threads in the gas generator case and tighten until finger tight.

c. Torque spark igniter to 300 inch-pounds.

d. Loosen spark igniter to 0 inch-pounds, then retorque to 200 to 240 inch-pounds.

e. Safety wire the spark igniter to the fuel transfer tube locking plate.

CAUTION

Ensure that cable braiding, ferrules, and spark igniter remain stationary during coupling connection to prevent damage to the cable assembly.

f. Mate the cable assembly to the spark igniter and tighten couplings 1/4-turn past finger tight.

g. Safety wire the cable assembly coupling to the spark igniter.

h. Install engine cowling using procedures outlined in Chapter 71-10-00.

i. Restore electrical power to the airplane.

IGNITION EXCITER REMOVAL
(Effectivity: All)
(Figure 5)

a. Ensure that both the auto-ignition and ignition/start switches are in the OFF position and pull the igniter power circuit breaker.

CAUTION

Ensure that all power is removed from the airplane before starting maintenance.

b. Remove engine cowling using procedures outlined in Chapter 71-10-00.

WARNING

Before attempting any maintenance on the ignition system, wait at least six minutes for any residual voltage to dissipate. Attempts to remove ignition components before the waiting period has elapsed may cause injury to personnel. Residual voltage in the ignition exciter is unpredictable; therefore, even after waiting for six minutes, further precautions must be taken. Always use insulated tools; never touch the outputs with bare hands; and disconnect the coupling nuts at the ignition exciter first.

c. Remove safety wire on both ignition cable couplings (1) and the supply cable (8).

CAUTION

Ensure that cable braiding and ferrules remain stationary during coupling disconnection to prevent damage to the cable assembly.

d. Loosen the supply cable connector and remove from the ignition exciter input (7).

e. Loosen both ignition cable couplings and remove from the ignition exciter outputs (2).

f. Remove five bolts (9), washers (4), and nuts (3) from clamps (6 and 12) and mounting bracket (5) on ignition exciter (10).

g. Remove mounting bracket (5) and slide clamps (6 and 12) to the side to remove the ignition exciter (10).

IGNITION EXCITER INSTALLATION
(Effectivity: All)
(Figure 5)

a. Inspect clamps (6 and 12) for elongated holes and deterioration of the insulated cover. Replace any unserviceable clamps.

b. Align the ignition exciter (10) with the mounting bracket (5) and clamps (6 and 12).

c. Install the five bolts (9) with heads facing outboard of the engine.

d. Install five washers (4) and nuts (3).

CAUTION

Ensure that cable braiding and ferrules remain stationary during coupler connection to prevent damage to the cable assembly.

e. Mate the two ignition cable couplings (1) with the ignition exciter outputs (2), and tighten 1/4-turn past finger tight.

f. Safety wire ignition cable couplings (1) so they prevent each other from rotating loose.

g. Mate power supply cable connector (8) with the ignition exciter input (7), and tighten 1/4-turn past finger tight.

h. Secure supply cable connector (8) to ignition exciter (10) with safety wire.

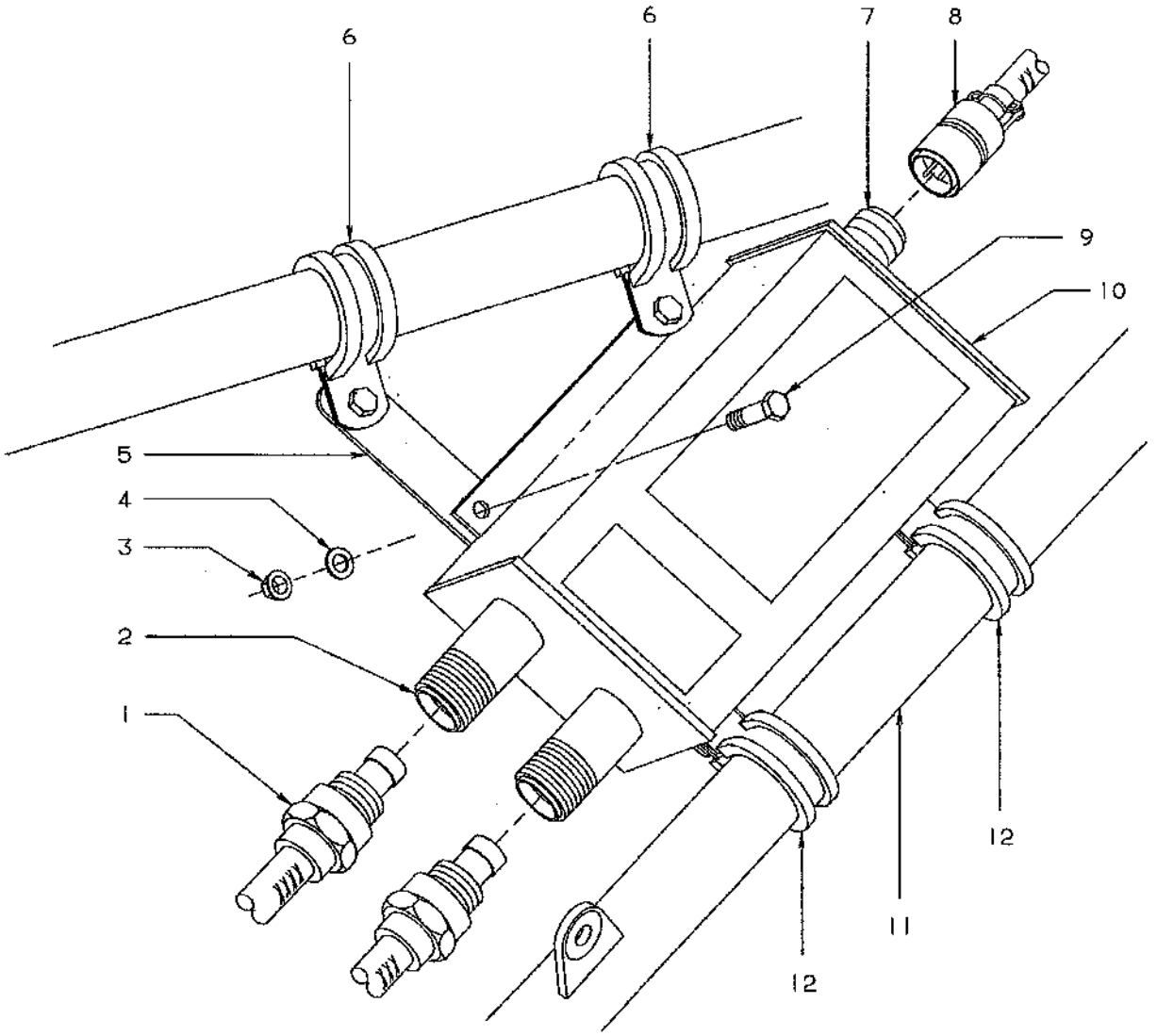
i. Install engine cowling using procedures outlined in Chapter 71-10-00.

j. Restore electrical power to the airplane.

AUTO-IGNITION/AUTOFEATHER HIGH PRESSURE SWITCH REMOVAL
(Effectivity: All)
(Figure 6)

NOTE

If engine has been operating allow sufficient time for oil pressure to diminish before removing pressure switch.



1. IGNITION CABLE COUPLING
2. IGNITION EXCITER OUTPUT
3. NUT
4. WASHER
5. MOUNTING BRACKET
6. CLAMPS
7. IGNITION EXCITER INPUT
8. SUPPLY CABLE CONNECTOR
9. BOLT
10. IGNITION EXCITER
11. ENGINE TRUSS MOUNT
12. CLAMPS

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Ignition Exciter Installation (Effectivity: All)
Figure 5

a. Remove engine cowling using procedures outlined in Chapter 71-10-00.

b. Ensure that the battery, generator and external power switches are in the OFF positions.

c. Remove safety wire securing electrical connector to the auto-ignition/autofeather high pressure switch.

d. Unscrew the electrical connector from auto-ignition/autofeather high pressure switch.

e. Remove safety wire connecting the auto-ignition/autofeather high pressure switch and autofeather low pressure switch.

f. Remove the auto-ignition/autofeather high pressure switch from the torque pressure manifold with a wrench and discard the O-ring.

g. Install a protective plug in the torque pressure manifold to prevent oil contamination.

AUTO-IGNITION/AUTOFEATHER HIGH PRESSURE SWITCH INSTALLATION (Effectivity: All) (Figure 6)

a. Remove the protective plug from torque pressure manifold and ensure that no foreign material is on threads or in opening.

b. Install a new O-ring on the auto-ignition/autofeather high pressure switch.

c. Align threads on the auto-ignition/autofeather high pressure switch with threaded mount on the torque pressure manifold, then tighten until finger tight.

d. Torque auto-ignition/autofeather high pressure switch to 30 inch-pounds maximum.

e. Safety wire the auto-ignition/autofeather high pressure switch to the autofeather low pressure switch so neither switch can rotate loose.

f. Align electrical connector keyway with auto-ignition/autofeather high pressure switch and tighten connector finger tight.

g. Safety wire electrical connector to auto-ignition/autofeather high pressure switch.

h. Start the engine using procedures outlined in Section IV of the BEECHCRAFT Starship 1 Pilot's Operating Handbook and allow oil pressure to build in the torque pressure manifold.

i. Shut the engine down using procedures outlined in Section IV of the BEECHCRAFT Starship 1 Pilot's Operating Handbook.

j. Inspect the torque pressure manifold at the auto-ignition/autofeather high pressure switch base for any signs of oil leakage. If leakage is evident check torque or install a new O-ring if required.

k. Install engine cowling using procedures outlined in Chapter 71-10-00.

l. Perform an operational check of the auto-ignition system as outlined in this chapter under AUTO-IGNITION SYSTEM OPERATIONAL CHECK.

AUTO-IGNITION SYSTEM OPERATIONAL CHECK (Effectivity: All)

a. Start both engines as outlined in Section IV of the BEECHCRAFT Starship 1 Pilot's Operating Handbook.

b. Set both power levers to IDLE.

c. The Engine Instrument, Crew Alerting System (EICAS) green L IGNITION ON and R IGNITION ON advisory messages should no longer be displayed after the ignition/start switches are placed in the OFF position.

CAUTION

Avoid operating the auto-ignition system for an extended period of time. This will prolong the operational service life of the igniters.

d. Place the LH auto-ignition switch in the ARM position. The green L IGNITION ON advisory message should appear on the EICAS display.

e. Advance the LH power lever from IDLE. Note when the EICAS green advisory message L IGNITION ON is no longer displayed. The message should blank from the EICAS display when the engine reaches 17% of maximum allowable torque.

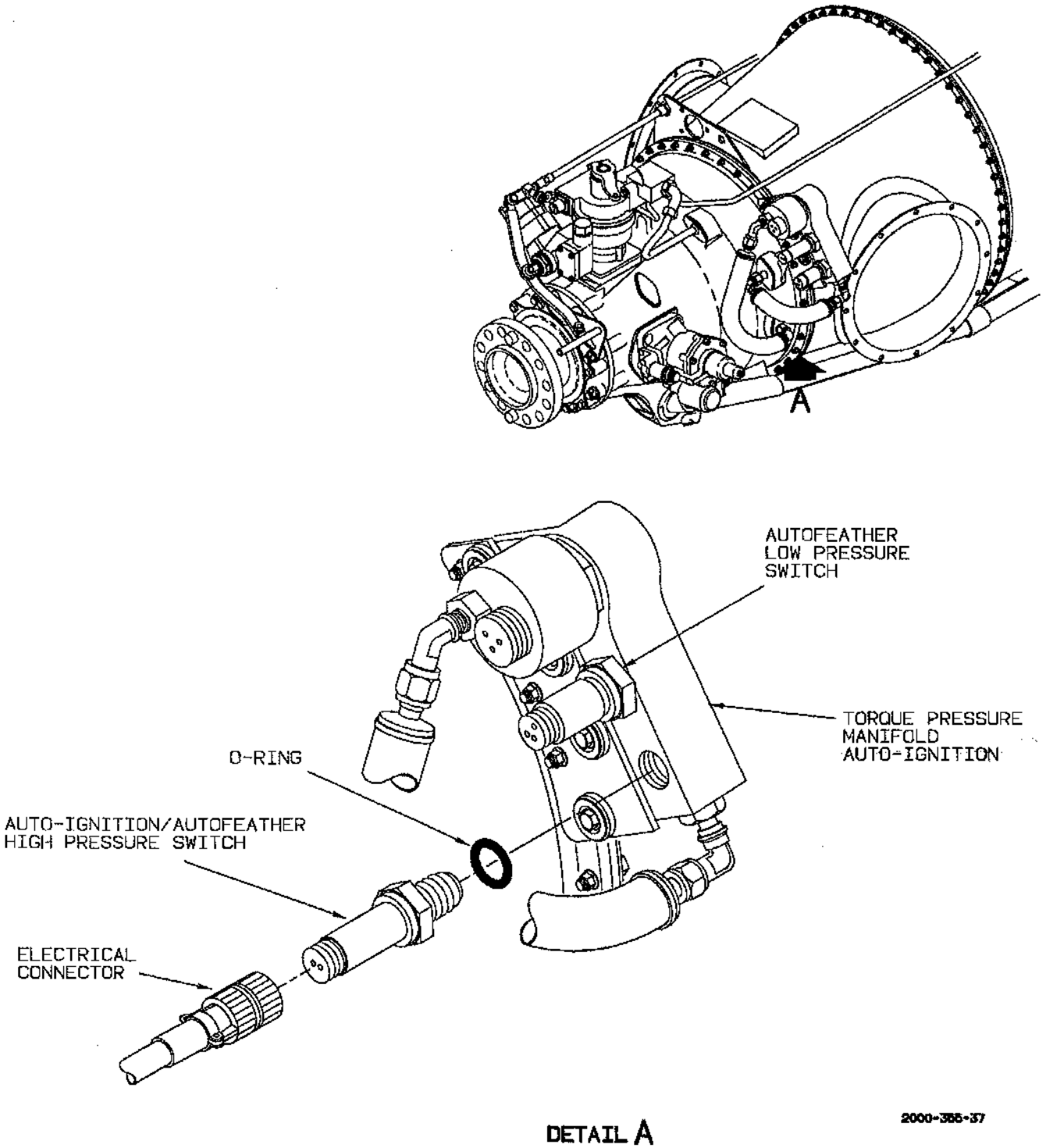
f. Place the LH auto-ignition switch in the OFF position.

g. Repeat Steps d, e and f for the RH engine auto-ignition system.

h. Shut the engines down using procedures outlined in Section IV of the BEECHCRAFT Starship 1 Pilot's Operating Handbook.

NOTE

If the auto-ignition system performs as described, the system is working properly. If the auto-ignition/autofeather high pressure switch does not actuate at 17% of maximum allowable torque, perform a



Auto-Ignition/Autofeather High Pressure Switch Installation (Effectivity: All)
Figure 6

functional check of the switch using procedures outlined in this chapter under AUTO-IGNITION/AUTOFEATHER HIGH PRESSURE SWITCH FUNCTIONAL CHECK. Check the spark igniters, cable assemblies, and ignition exciter for other possible faults using procedures outlined in this chapter under IGNITION EXCITER, SPARK IGNITER, AND CABLE ASSEMBLY FUNCTIONAL CHECK.

AUTO-IGNITION/AUTOFEATHER HIGH PRESSURE SWITCH FUNCTIONAL CHECK (Effectivity: All)

a. The following items are required to perform this check:

1. Regulated supply of dry, filtered, compressed air at no less than 50 psi.

2. Air pressure gage calibrated in .01 psi increments.

3. Plumbing necessary to connect the pressure supply to the switch.

4. Multimeter (1, Chart 1).

b. Remove the switch from the torque pressure manifold using procedures outlined in this chapter under AUTO-IGNITION/AUTOFEATHER HIGH PRESSURE SWITCH REMOVAL.

c. Connect the pressure source to the switch.

d. Connect the multimeter to pins A and B; the normally closed contacts of the switch.

e. With no pressure applied to the switch, check for continuity across the normally closed contacts. If no continuity is indicated replace the switch.

f. Apply pressure to the switch at a slow steady rate until 50 psi is indicated on the pressure gage. There should be no continuity across the normally closed contacts; if there is, replace the switch.

g. Reduce the pressure at a slow steady rate, noting at what pressure continuity is indicated. The switch should actuate between 2.5 and 3.7 psi; if not, replace the switch.

h. Slowly increase the pressure, noting at what pressure there is no continuity across the normally closed contacts. The difference between the decreasing and increasing actuation points shall be a maximum of 30% of the actual decreasing point.

NOTE

To find the maximum increasing point for the decreasing point noted in step g, find the value denoted by step g in Chart 4 under the column heading "Actuate On Decrease PSI," then read across Chart 4 to the value denoted under the column heading of "Actuate On Increase PSI." The increasing actuation pressure noted in step h must not be higher than this value; if it is, replace the switch.

i. Repeat the check four times to ensure consistency.

CAUTION

Bleed the air pressure from the auto-ignition/autofeather high pressure switch before disconnecting the switch.

j. Disconnect the multimeter from the switch.

k. Install the switch in the torque pressure manifold using procedures outlined in this chapter under AUTO-IGNITION/AUTOFEATHER HIGH PRESSURE SWITCH INSTALLATION.

CHART 4
AUTO-IGNITION/AUTOFEATHER HIGH SWITCH ACTUATION POINT
(Effectivity: All)

Actuate On Decrease PSI	Actuate On Increase PSI
2.5	3.2
2.6	3.38
2.7	3.51
2.8	3.64
2.9	3.77
3.0	3.90
3.1	4.03
3.2	4.16
3.3	4.29
3.4	4.42
3.5	4.55
3.6	4.68
3.7	4.68

IGNITION EXCITER, SPARK IGNITER AND CABLE ASSEMBLY FUNCTIONAL CHECK (Effectivity: All)

After any maintenance has been performed, or when there is a question of proper system operation, a functional check should be performed as follows:

a. Refer to Chapter 71-10-00 and remove the engine cowling if not already removed.

b. Ensure that both the ignition/start and the auto-ignition switches are in the OFF position.

WARNING

Before attempting any maintenance on the ignition system, wait at least six minutes for any residual voltage to dissipate. Attempts to remove ignition components before the waiting period has elapsed may cause injury to personnel. Residual voltage in the ignition exciter is unpredictable; therefore, even after waiting for six minutes, further precautions must be taken. Always use insulated tools; never touch the outputs with bare hands; and disconnect the coupling nuts at the ignition exciter first.

c. Remove safety wire from cable coupling at the ignition exciter.

CAUTION

Ensure that cable braiding and ferrules remain stationary during coupler disconnection to prevent damage to the cable assembly.

d. Disconnect one of the ignition cables from the ignition exciter.

CAUTION

Before placing the ignition/start switch in the ON position, perform the engine clearing procedures outlined in Section IV of the BEECHCRAFT Starship 1 Pilot's Operating Handbook to remove unwanted fuel from the gas generator case.

e. Place the battery switch in the ON position.

f. Place the ignition/start switch in the ON position.

g. Listen for a snapping sound emitting from the gas generator case about once every second.

h. The spark igniters should spark at the rate of .8 to 1.0 spark per second with 9 vdc applied, and up to a rate of 1.4 to 4.0 sparks per second with 30 vdc applied. The voltage input to the ignition exciter will vary depending on the load that the starter/generator pulls from the airplanes electrical system; therefore the system is designed to operate over a 9 to 30 vdc range and a continuous 28 vdc is not required. If the snapping sound does not occur for either spark igniter, check for power to the ignition exciter. If the ignition exciter is not receiving power during the ignition sequence, isolate the fault using Chart 3 in this chapter. If the ignition exciter is receiving power, remove and replace the unit using procedures outlined in this chapter and perform the operational check sequence again. If a snapping sound is heard during only one of the checks, check the ignition cable for continuity with an ohmmeter. If the cable is faulty refer to the Pratt & Whitney Engine Maintenance Manual for cable replacement procedures. A good ignition exciter and cable assembly indicates a faulty spark igniter.

i. Place the ignition/start switch in the OFF position.

j. Place the battery switch in the OFF position.

WARNING

Before attempting any maintenance on the ignition system, wait at least six minutes for any residual voltage to dissipate. Attempts to remove ignition components before the waiting period has elapsed may cause injury to personnel. Residual voltage in the ignition exciter is unpredictable; therefore, even after waiting for six minutes, further precautions must be taken. Always use insulated tools; never touch the outputs with bare hands; and disconnect the coupling nuts at the ignition exciter first.

k. Reconnect the cable assembly to the ignition exciter by tightening 1/4-turn past finger tight.

l. Repeat the procedure for the other spark igniter on the same engine.

m. Safety wire the ignition exciter cable couplings so neither can rotate loose.

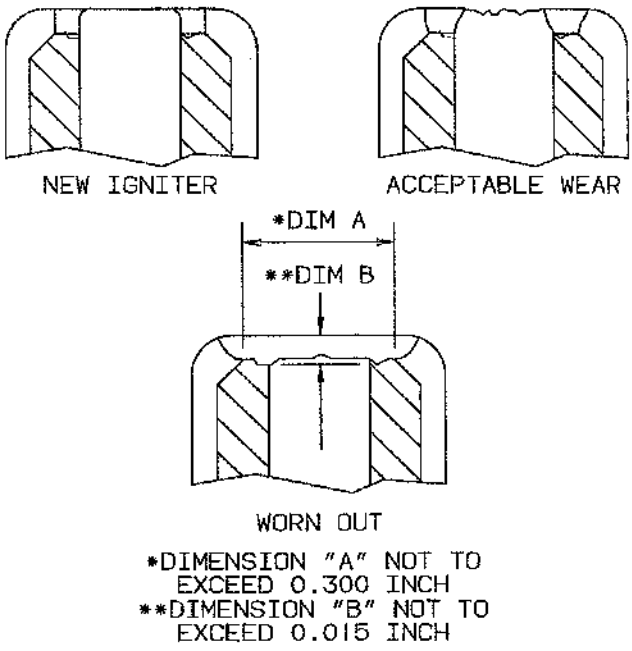
NOTE

Always perform an inspection, and cleaning if needed, of the spark igniters when a functional check indicates a fault.

n. Refer to Chapter 71-10-00 and install the engine cowling if the functional check results are good.

SPARK IGNITER CLEANING AND INSPECTION (Effectivity: All) (Figure 7)

a. Remove engine cowling using procedures outlined in chapter 71-10-00.



Spark Igniter Wear Limits (Effectivity: All)
Figure 7

CAUTION

Whenever a spark igniter is dropped there is a chance that the spark igniter has been damaged in a way that is not detectable during ordinary test procedures. Always replace a spark igniter that has been dropped with a new one. Never use lubricants on the spark igniter threads.

b. Remove the spark igniter using procedures outlined under SPARK IGNITER REMOVAL in this chapter.

NOTE

The firing end of the spark igniters should never be cleaned. Carbon formed on the electrodes or gap area should remain intact to aid in spark igniter operation.

c. Using a felt swab soaked in PD-680 solvent (1, Chart 2), or methyl alcohol (2, Chart 2), clean the inside surface of the connection terminal of the spark igniter.

d. Dry the spark igniter with dry, filtered, compressed air.

e. Inspect the air cooling holes for foreign material, and remove any that may have become lodged in the cooling holes.

f. Inspect the exterior of the firing end of the spark igniter for chafing wear. Wear is not to exceed a depth of 0.015-inch. Refer to Figure 7 for an illustration of acceptable wear limits.

g. Inspect the electrode and ensure that erosion does not exceed the specified limits.

h. If spark igniter wear exceeds the limits specified in Figure 7, replace with a new spark igniter.

i. Install the spark igniter as outlined under SPARK IGNITER INSTALLATION in this chapter. After cleaning and inspecting one spark igniter, it is recommended that an operational check be performed prior to cleaning and inspecting the other spark igniter. It is also recommended that the cable assemblies not be safety wired until an operational check of both spark igniters has been performed.

CHAPTER 75 - AIR

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Starter/Generator Cooling Assembly Installation (Effectivity: All) 1

ACCESSORY COOLING - DESCRIPTION AND OPERATION (EFFECTIVITY :ALL)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

STARTER/GENERATOR COOLING (Effectivity: All) (Figure 1)

The starter/generators are individually blast cooled during flight by identical cooling systems. Outside air is ducted through an air inlet assembly attached to the left side of each nacelle lower forward cowling. Each nacelle contains a cooling hose and a pair of cooling adapters directing outside air from the inlet to the starter/generator, where it flows through an internal shaft driven fan, across the windings, and out the opposite side of the unit. The internal shaft driven fan provides sufficient air flow for cooling during ground operation.

ACCESSORY COOLING - MAINTENANCE PRACTICES (Effectivity: All)

STARTER/GENERATOR COOLING ASSEMBLY REMOVAL (Effectivity: All) (Figure 1)

a. Refer to Chapter 71-10-00 and remove the engine cowling.

b. Loosen the adapter clamp and slide it onto the starter/generator.

c. Remove the cooling cap adapter from the starter/generator, then pull the cooling adapter down to access the upper hose clamp.

d. Remove the upper hose clamp and cooling adapter from the cooling hose.

e. Loosen the six captive stud nuts attaching the air inlet to the nacelle lower forward cowling.

f. Remove the cooling hose and air inlet assembly from the nacelle.

g. Remove the cooling hose from the air inlet assembly by removing the remaining hose clamp.

STARTER/GENERATOR COOLING ASSEMBLY INSTALLATION (Effectivity: All) (Figure 1)

a. Attach the cooling hose to the air inlet assembly with the lower hose clamp.

b. Align the air inlet assembly with inlet cutout on the nacelle lower forward cowling.

c. Thread the six captive stud nuts into the six receptacles, then tighten the captive stud nuts until the air inlet is secured.

d. Attach the cooling adapter to the cooling hose with the upper hose clamp.

e. Align the the cooling adapter with the lower side of the starter/generator.

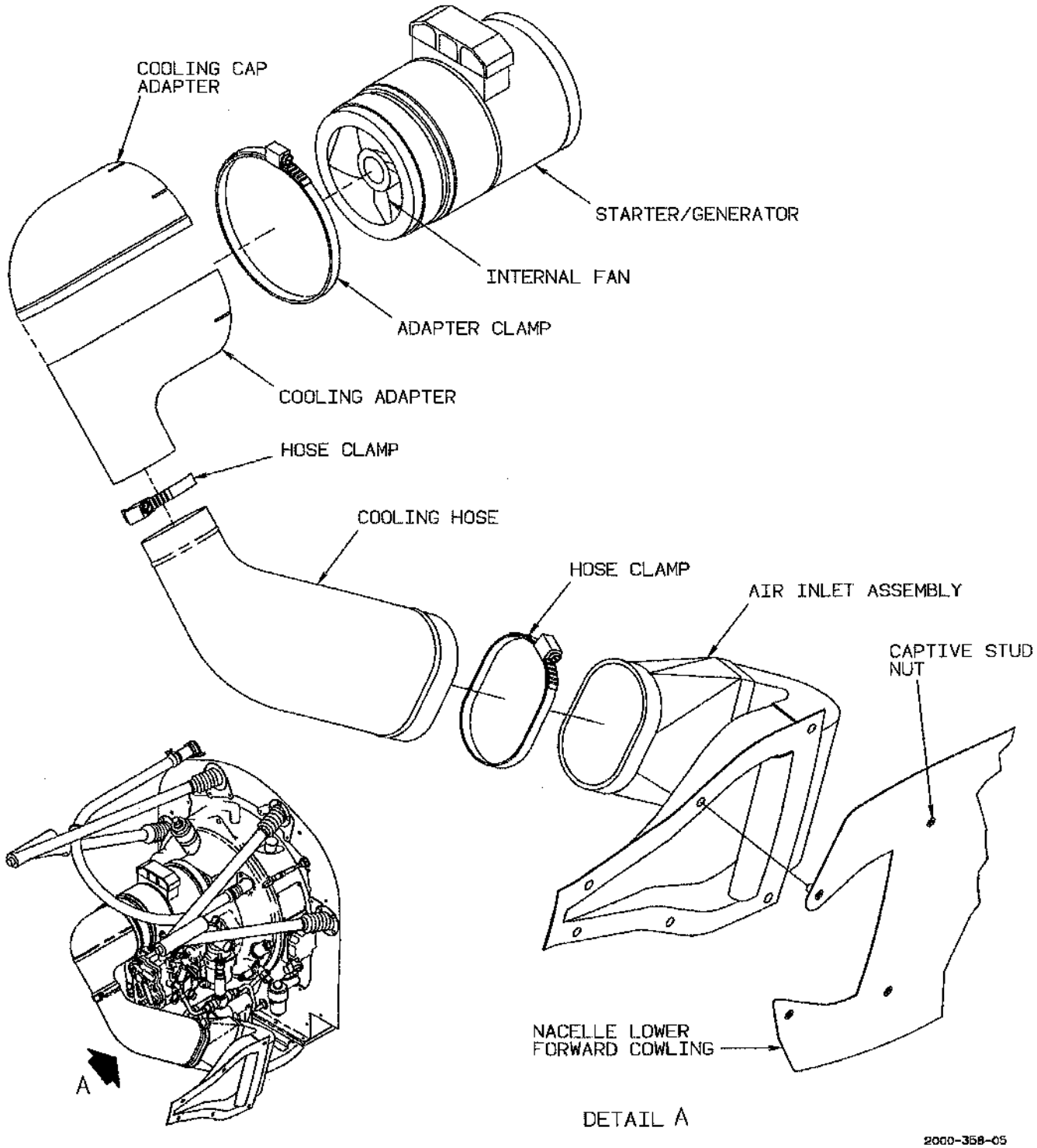
f. Align the cooling cap adapter with the starter/generator and cooling adapter edges.

g. Slide the adapter clamp over the cooling adapter and cooling cap adapter edges.

h. Ensure that the alignment of both adapters and the adapter clamp provide a good seal on the starter/generator.

i. Tighten the adapter clamp.

j. Refer to Chapter 71-10-00 and install the engine cowling.



Starter/Generator Cooling Assembly Installation (Effectivity: All)
Figure 1

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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

ENGINE CONTROLS (Effectivity: All) (Figures 1 and 2)

The engine controls are used to control engine fuel delivery and propeller pitch. The engine controls consist of the control levers (mounted on the center pedestal), the control cables (which run from below the pedestal thru the fuselage, underneath the cabin floor and into the wings) and the associated bellcranks, linkages and cambox hardware that connect the control cables to their respective (engine and propeller) components.

The engine controls are operated by three sets (pairs) of control levers. The power levers are used to control the amount of power that the engine generates. The propeller levers are used to change the pitch and speed of the propellers. The condition levers control fuel flow at the fuel control unit, and limit N1 (gas generator) speed at idle.

POWER CONTROLS (Effectivity: All) (Figures 1 and 2)

The power levers are located on the LH side of the pedestal and regulate the amount of power delivered to the propellers. The power (N1) is measured as a percentage of maximum gas generator speed. Detents in the pedestal control mount provide positive power lever selections for IDLE, GND FINE and REVERSE.

N1 power is increased as the power levers are moved forward and is decreased as the power levers are moved aft. The power levers will contact the IDLE stop when they are moved fully aft in the POWER range. While the power levers are at the IDLE stop detent, they can be lifted up into the GND FINE detent. When the GND FINE detent is selected, propeller blade angle is flattened. GND FINE is used for aerodynamic braking during ground roll. Both the IDLE setting and the GND FINE range should produce an N1 of 66.0%. There should be no change in N1 when the power levers are moved through the idle detent and into the GND FINE range.

Propeller reversing is also controlled by the power levers. The REVERSE detent is selected by lifting the power levers up and pulling them aft from the GND FINE range. As the power levers are moved through the GND FINE detent into the REVERSE range, the Beta valve position (at the propeller) is changed to a negative pitch. The Beta valve causes the propeller blades to move from the GND FINE blade angle to the reverse pitch blade angle (-8°). When the power lever is pulled fully aft in the REVERSE range (maximum REVERSE position), the maximum reverse power stop is contacted and an interconnect through the Beta cable resets maximum propeller speed.

The power levers are mounted on a shaft inside the pedestal. Friction locks are installed on the pedestal control mount next to each power lever. Go around switches are located in the top of each power lever. The electrical lead for the go around switch is routed down the power lever to a receptacle at the forward side of the pedestal. Pushrods are connected to the lower end of the power levers. The pushrods are connected to cam assemblies (small bellcranks) which pivot around a shaft in the lower quadrant of the pedestal. The aft end of the cam assembly is connected to a rod end which is attached to an adapter that connects the rod end to the power cable.

The power cable consists of a wire rope (cable) encased in a flexible covered conduit. The power cable runs the full length (a single length cable) from the pedestal to the engine. The conduit is threaded at each end and is secured to brackets. The forward end of the conduit is secured to a bracket mounted on the lower fuselage aft of the pedestal, while the aft end of the conduit is secured to a bracket mounted to the forward engine flange. Clamps are used to attach the power cable to the airplane structure at several places under the cabin floorboards and inside the aft wing.

Cable heaters are attached to the power control cables to prevent cable freeze up in the fuselage and aft wing. The cable heaters are activated by the LEFT and RIGHT VENT/CABLE switches which are located on the LH inboard subpanel. The cable heaters are wired through 7.5 amp circuit breakers labeled CABLE HEATER, LEFT and RIGHT located in the ICE PROTECT section of the auxiliary circuit breaker panel.

A small tube is bonded into the lower side of the fuselage where the power cable passes through the fuselage and into the aft wing. The power cable conduit passes through a protective grommet at the firewall. The aft end of the power cable is attached to an adapter assembly and rod end which actuates the power control extension lever on the cambox assembly.

PROPELLER CONTROLS (Effectivity: All)
(Figures 1 and 2)

The propeller levers are located in the pedestal to the right of the power levers. The propeller levers are used to control the pitch and speed of the propeller blades. Markings on the pedestal indicate the forward (labeled TAKEOFF LANDING AND REVERSE, and HIGH RPM) and aft (FEATHER) positions of the propeller levers.

Propeller speed is increased as the propeller levers are moved forward (HIGH RPM), and is decreased as they are moved aft. The propeller levers will contact a detent at the rear of the lever travel to prevent inadvertent movement into the FEATHER range. Pulling the propellers back through the detent will cause the propeller blades to feather.

Pushrods are attached to the lower ends of the control levers. Each pushrod is attached to the forward end of a cam assembly (small bellcrank) which pivots on a shaft in the lower quadrant of the pedestal. A rod end on the end of the propeller cable is connected to the aft end of the cam assembly.

The propeller control cable consists of two separate cable assemblies (forward and aft) connected by a bellcrank. Each propeller cable assembly is constructed of a wire rope (cable) which slides inside a conduit. The conduit is threaded at one end and has provisions for attachment to the bellcrank bracket at the other end. The cables are fitted with a 16° threaded swivel at each end.

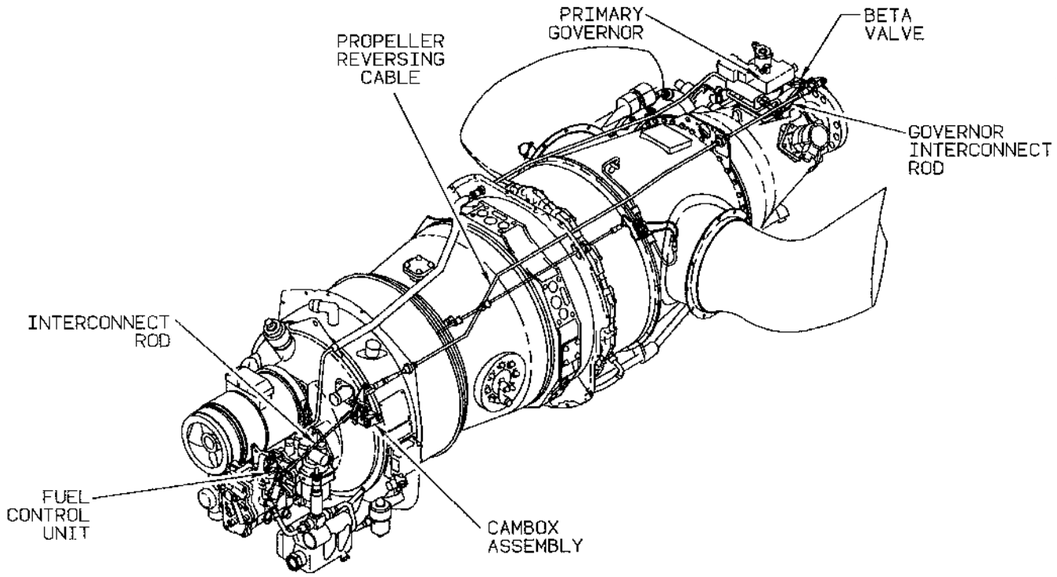
The forward propeller cable is routed aft from the pedestal under the cabin floorboards to a bellcrank mounted at the lower side of the fuselage. The propeller bellcrank penetrates through the fuselage and the aft propeller cable is attached to the bellcrank outboard of the fuselage. The aft propeller cable is routed through the aft wing to the firewall where it passes through a protective grommet into the engine compartment. The aft propeller cable is attached to the engine truss mount, runs aft along the top of the air plenum to a bracket near the propeller governor, where the conduit is secured. The aft propeller cable is attached to the primary governor control arm at the aft end of the engine.

CONDITION CONTROLS (Effectivity: All)
(Figures 1 and 2)

The condition levers control fuel delivery to the engine by activating the fuel control unit. Markings on the pedestal indicate preset fuel conditions of RUN, START and FUEL CUTOFF.

The condition controls are connected to the fuel control units in nearly the same configuration as the propeller controls are connected to the primary governors. The condition cables and propeller cables share the same design (two cables with bellcrank), construction and routing, with the exception that the condition bellcrank penetrates the fuselage forward of the propeller bellcrank. The aft end of the condition cable conduit is secured to a bracket mounted to a flange at the front of the engine. A rod end is attached to the aft condition control cable. The rod end is attached to and actuates a lever on the fuel control unit.

Placing the condition lever in the FUEL CUTOFF position causes the fuel condition cam to move against the cutoff stop and actuate a lever which depresses the fuel cutoff valve on the fuel control unit. When the condition lever is placed in the RUN position, IDLE and GND FINE speeds are limited to approximately 66% N1. Gas generator speed is controlled by the amount of fuel metered into the engine. Metering the fuel is a function of valve position in the fuel control unit and P3 bleed air pressure. For more information on operation of the fuel control unit, see the Pratt & Whitney PT6A-67A Engine Maintenance Manual.



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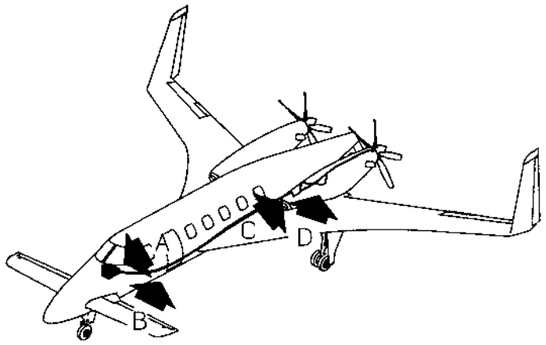
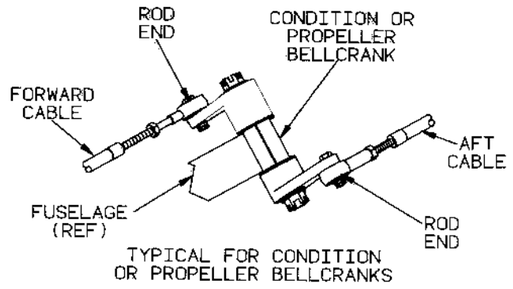
PT6A-67A Engine and Major Controls (Effectivity: All)
Figure 1





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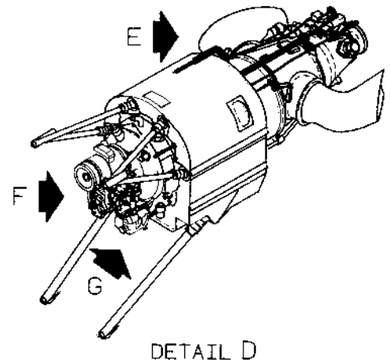

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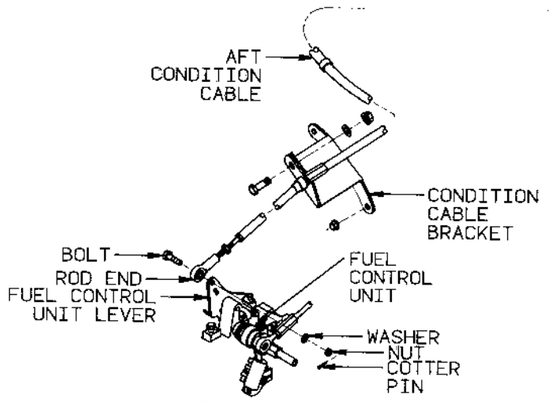

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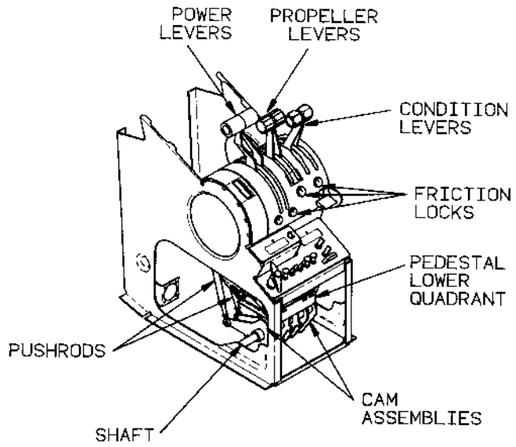

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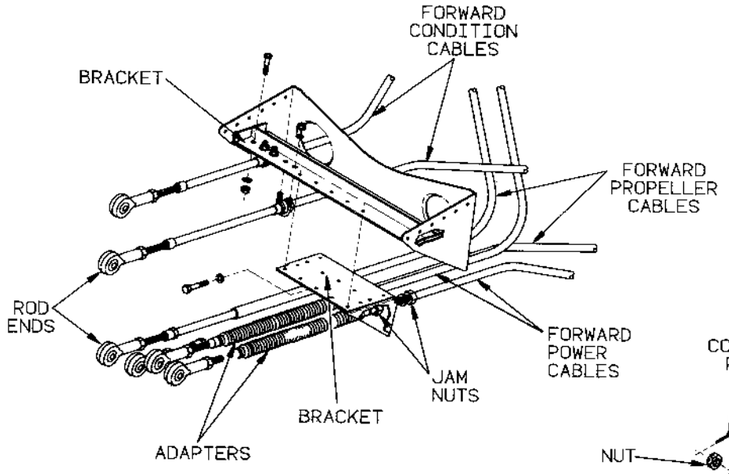
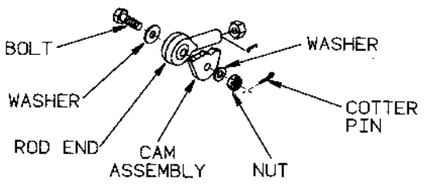
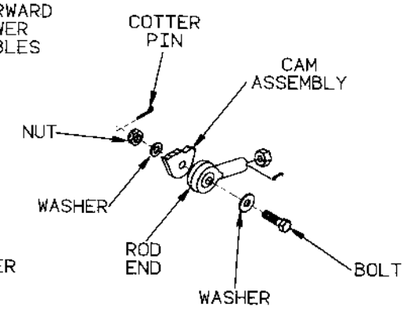
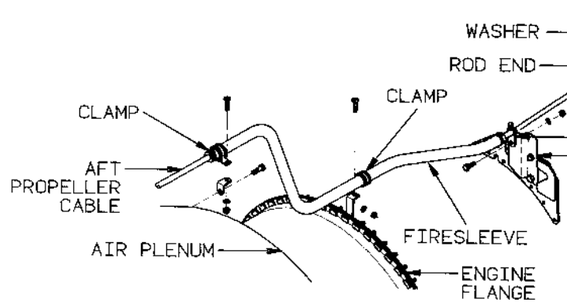






(EXCEPT CONDITION CABLES)

DETAIL B



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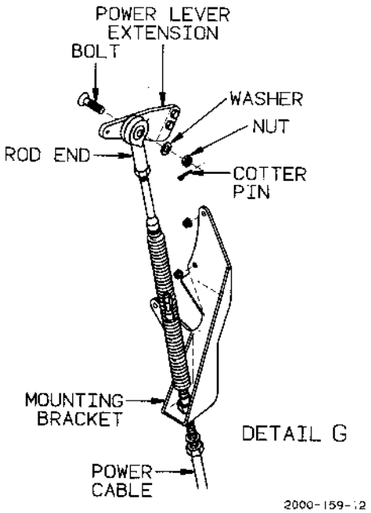


DETAIL G

2000-159-12

Engine Control Cables Routing (Effectivity: All)
Figure 2

RECOMMENDED MATERIALS (Effectivity: All)

The recommended materials listed in Chart 1 as meeting federal, military or vendor specifications are provided for reference only and are not specifically required by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor, or by compliance with the applicable specifications. GENERIC OR LOCALLY MANUFACTURED PRODUCTS WHICH CONFORM TO THE REQUIREMENTS OF THE SPECIFICATION MAY BE USED EVEN THOUGH NOT INCLUDED IN THE CHARTS. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 1
RECOMMENDED MATERIALS (Effectivity: All)

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
1. Cleaner	TT-I-735 or MIL-I-10428	Ispropyl Alcohol	Procure locally
2. Cleaner	O-T-634	1, 1, 1 Trichloroethane	Procure locally

ENGINE CONTROLS - MAINTENANCE PRACTICES (Effectivity: All)

The following rigging instructions produce nominal settings of the operation parameters of the engine. Any time an engine is removed, the fuel control unit is changed, the propeller governor is replaced or any other time the adjustments of one of these units is disturbed, the engine controls rigging procedure should be performed, all or in part.

These rigging procedures yield nominal results; a ground check and final adjustment must be performed any time a rigging procedure is completed. Resultant operating parameters of each engine should be recorded and final adjustments made to restore the engine to its original performance parameters.

The following final adjustments to the engine operating parameters are permitted to field technicians:

a. START position of 54% ±1% N1.

b. RUN IDLE and GND FINE of 66.0% ±.5% N1.

c. Maximum N1 of 104%.

d. Maximum N2 of 1700 RPM.

e. Maximum REVERSE N1 of 84% ±1%.

CONDITION CONTROLS RIGGING (Effectivity: All) (Figures 2 and 3)

a. Adjust the aft condition cable, and attach it to the condition bellcrank as follows:

1. Remove the pedestal side panels and disconnect the condition cable rod end from the cam assembly located on the shaft in the lower pedestal quadrant. See Chapter 53-20-00 for removal of the cockpit floorboards, if necessary.

2. Remove the upper aft engine cowling as outlined in Chapter 71-10-00.

3. Disconnect the aft condition cable rod end from the arm on the fuel control unit by removing the cotter pin, nut, washer and bolt.

4. Remove the aft wing fairings as outlined in Chapter 53-50-00.

5. Locate the condition bellcrank from outside the fuselage and disconnect the rod end from the condition bellcrank.

6. Adjust the rod end to the mid-adjustment position on the condition cable end. Verify that cable threads are visible through the inspection hole in the rod end.

7. Set the aft condition cable to the mid-travel position by sliding the cable inside the conduit.

8. Rotate the condition bellcrank until the centerline of the arm is horizontal.

NOTE

Do not move the condition bellcrank from this position while rigging the condition cables to the bellcrank.

9. Adjust the forward condition conduit mounting point in the bracket so that the bolt hole in the rod end aligns with the hole in the condition bellcrank. Secure the conduit to the bracket with the jam nuts.

10. Attach the rod end to the condition bellcrank by installing the bolt, washer, nut and cotter pin.

b. Adjust the forward condition cable, and attach it to the condition bellcrank as follows:

1. Gain access to the condition cable bellcrank by removing the necessary aft cabin floorboards as outlined in Chapter 53-20-00.

2. Locate the condition bellcrank from inside the fuselage and disconnect rod end from the condition bellcrank.

3. Adjust the forward condition cable rod end to the mid-adjustment position on the condition cable end. Verify that condition cable threads are visible through the inspection hole in the rod end.

4. Set the forward condition cable to the mid-travel position by sliding the cable inside the conduit.

NOTE

Do not allow the condition bellcrank to move from the position set in step (a), while rigging the condition cables to the bellcrank.

5. Adjust the forward condition conduit mounting point in the bracket so that the bolt hole in the rod end aligns with the hole in the condition bellcrank. Secure the conduit to the bracket with the jam nuts.

6. Attach the rod end to the condition bellcrank by installing the bolt, washers, nut and cotter pin.

c. Connect the aft condition cable to the engine as follows:

1. Ensure the condition lever is in the START position and that a 3/16 inch diameter rig pin is installed in the rigging hole at the fuel control unit.

2. Position the rod end at the mid-adjustment point on the threads located on the aft end of the aft

condition cable. Verify that threads are visible through the inspection hole in the rod end.

3. Slide the aft condition cable until the bolt hole in the rod end aligns with the bottom hole in the fuel control unit condition lever.

4. Install the bolt through the bottom hole in the fuel control unit condition lever from the RH side, with the rod end on the RH side of the condition lever. Secure with the washer, nut and cotter pin.

d. Attach and adjust the condition cable at the pedestal as follows:

1. Locate the ground idle catch at the mid adjustment position.

2. Move the condition lever to the start position against the ground idle catch.

3. Adjust the rod end on the forward end of the forward condition cable so that the bolt hole in the rod end aligns with the hole in the cam assembly (in the lower quadrant of the pedestal). Verify that cable threads are visible through the inspection hole in the rod end.

NOTE

It may be necessary to adjust the rod end at the engine condition lever on the fuel control unit to obtain proper cable length.

4. Attach the rod end to the cam assembly at the pedestal and secure with the bolt, washers, nut and cotter pin. Insert the bolt into the LH rod end from the LH side, and insert the bolt into the RH rod end from the LH side. Position the RH rod end on the RH side of the cam assembly, and position the LH rod end on the LH side of the cam assembly.

e. Remove the 3/16-inch rig pin from the engine condition lever on the fuel control unit.

f. Pull the condition lever back to the FUEL CUTOFF position. With the engine condition lever on the fuel control unit fully against the stop, check the clearance between the condition lever and the aft edge of the pedestal slot. There should be a minimum of .15-inch between the condition lever and the aft edge of the slot.

g. Advance the condition lever to the RUN position. With the engine condition lever on the fuel control unit fully against the stop, check the clearance between the condition lever and the forward edge of the pedestal slot. There should be a minimum of .15-inch between the condition lever and the forward edge of the slot.

h. Install a key on each rod end and secure with safety wire. Secure the jam nuts on the brackets at the engine and the condition bellcranks with safety wire. Ensure that all cotter pins are properly installed.

i. Install the cockpit and cabin floorboards as outlined in Chapter 53-20-00.

j. Install the aft wing fairings as outlined in Chapter 53-50-00.

k. Perform the FINAL ENGINE CONTROLS RIGGING procedures as outlined in this Chapter, before returning the airplane to service.

POWER CONTROLS RIGGING
(Effectivity: All)
(Figures 2 and 6)

a. Remove the engine cowling as outlined in Chapter 71-10-00.

b. Disconnect the power cable rod end from the the power lever extension at the cambox.

c. Remove the pedestal side panels. Remove the cockpit floorboards (as necessary), as outlined in Chapter 53-20-00.

d. Disconnect the power cable rod end from the cam assembly, located aft of the pedestal.

e. Adjust the power cable and attach to the engine as follows:

1. Ensure that the engine power lever is in the idle position and that a 3/32-diameter rig pin has been installed in the rigging hole at the cambox assembly.

NOTE

If the rig pin will not fit into the rigging hole at the cambox assembly, move the power lever slightly forward of the idle detent.

2. Position the rod end on aft end of the power cable at the mid-adjustment position. Verify that cable threads are visible through inspection hole in the rod end.

3. Adjust the power cable conduit mounting point to the mid-adjustment position in the power cable bracket and secure with the jam nuts.

4. Slide the power cable in the conduit so that the bolt hole in the rod end aligns with the hole in the engine power lever extension.

5. Attach the rod end to the engine power lever extension by installing the bolt, washer, nut and cotter

pin. Insert the bolt into the power lever extension from the RH side with the rod end located on the LH side of the power lever extension.

f. Adjust and attach the power cable at the pedestal as follows:

1. Inspect the pedestal to ensure that all hardware inside the pedestal is rigged properly.

2. Pull the power lever back to the IDLE position.

3. Adjust the power conduit mounting point to the mid-adjustment position in the bracket located aft of the pedestal under the floorboards. Secure the conduit to the bracket with the jam nuts.

4. Adjust the rod end so that bolt hole in the rod end aligns with the hole in the cam assembly. Verify that cable threads are visible through the inspection hole in the rod end.

NOTE

It may be necessary to adjust the rod end at the engine power lever inside the pedestal to obtain the proper cable length.

5. Attach the rod end to the cam assembly with the bolt, washers, nut and cotter pin. Insert the bolt into the LH rod end from the RH side, and insert the bolt into the RH rod end from the LH side. Position the RH rod end on the LH side of the cam assembly, and position the LH rod end on the RH side of the cam assembly.

g. Remove the rig pin from the rigging hole in the cambox assembly (at the engine flange).

h. Install a key on each rod end and secure with safety wire. Secure the jam nuts on the brackets at the engine and the pedestal with safety wire. Ensure that all cotter pins are properly installed.

i. Install the engine cowling as outlined in Chapter 71-10-00.

j. Install the cockpit floorboards as outlined in Chapter 53-20-00.

k. Perform the FINAL ENGINE CONTROLS RIGGING procedures as outlined in this Chapter, before returning the airplane to service.

PROPELLER CONTROLS RIGGING
(Effectivity: All)
(Figures 2 and 6)

a. Remove the pedestal side panels and cockpit floorboards as outlined in Chapter 53-20-00.

b. Disconnect the forward propeller cable rod end from the cam assembly on the shaft in the lower pedestal quadrant.

c. Remove the upper aft engine cowling as outlined in Chapter 71-10-00 and disconnect the aft propeller cable rod end from the primary governor control arm.

d. Remove the aft wing fairings as outlined in Chapter 53-50-00 to gain access to the propeller bellcrank.

e. Locate the propeller bellcrank from outside the fuselage and disconnect the aft propeller cable rod ends from the bellcrank by removing the cotter pin, nut, washer and bolt from the rod end.

f. Rig the aft propeller cable to the propeller bellcrank as follows:

1. Position the rod end at the mid-adjustment point on the threads located on the forward end of the aft propeller cable. Verify that the propeller cable threads are visible through the inspection hole in the rod end.

2. Set the aft propeller cable to the mid-travel position by sliding the aft propeller cable inside the conduit.

3. Rotate the propeller cable bellcrank until the centerline of the arm is vertical.

4. Adjust the conduit mounting point in the bracket so that the bolt hole in the rod end aligns with the hole in the propeller cable bellcrank. Secure the conduit to the bracket with the jam nuts.

5. Attach the rod end to the bellcrank by installing the bolt, washer, nut and cotter pin.

NOTE

Do not move the propeller cable bellcrank from this position while rigging the propeller cables to the bellcrank.

g. Rig the forward propeller cable to the propeller bellcrank as follows:

1. Gain access to the propeller bellcrank by removing the aft cabin floorboards, as outlined in chapter 53-20-00.

2. Locate the propeller cable bellcrank from inside the fuselage and disconnect the forward propeller cable from the bellcrank by removing the cotter pin, nut, washer and bolt from the rod end.

3. Position the rod end at the mid-adjustment point on the threads located on the aft end of the forward propeller cable. Verify that the propeller cable threads are visible through the inspection hole in the rod end.

4. Set the forward propeller cable to the mid-travel position by sliding the forward propeller cable inside the conduit.

5. Check the propeller cable bellcrank to make sure that the arm is still in the vertical position.

6. Adjust the conduit mounting point in the bracket located forward of the bellcrank so that the bolt hole in the rod end aligns with the hole in the propeller cable bellcrank. Secure the conduit with the jam nuts.

7. Attach the rod end to the propeller bellcrank with the bolt, washer, nut and cotter pin.

h. Rig the aft propeller cable to the primary governor control arm as follows:

1. Check the primary governor control arm to make sure that it is installed properly. The primary governor control arm should form a 55° angle with the centerline of the engine (angle is to the LH side of engine) when the governor stop plate is full forward against the stop.

2. Rotate the primary governor control arm forward approximately 36.5° from the maximum speed stop. This should be the mid-travel position for the control arm (an approx. 91.5° angle with the engine centerline).

3. Set the aft propeller cable at mid-travel position by sliding the aft propeller cable inside the conduit.

4. Adjust the rod end on the aft end of the aft propeller cable so that it aligns with the bolt hole in the primary governor control arm. Verify that cable threads are visible through the inspection hole in the rod end.

5. Attach the rod end to the primary governor control arm with the bolt and washers.

6. Set the minimum speed stop by rotating the primary governor control arm forward until there is a .20-inch gap between the feathering valve adjustment bolt and the governor stop plate. Reference Chapter 61-20-00 Figure 4, in this maintenance manual.

i. Rig the forward propeller cable at the pedestal as follows:

1. Remove the cockpit floorboards aft of the pedestal to gain access to the propeller cable, as outlined in Chapter 53-20-00.

2. Pull the propeller control lever back to the detent (minimum governor speed setting), located forward of the feathering range.

3. Set the forward propeller cable to the mid-travel position by sliding the forward propeller cable inside the conduit.

4. Adjust the rod end on the forward end of the propeller cable so that the bolt hole in the rod end aligns with the hole in the cam assembly. Verify that cable threads are visible through the inspection hole in the rod end.

5. Attach the rod end to the cam assembly with the bolt, washers, nut and cotter pin. Insert the bolt into the LH rod end from the RH side, and insert the bolt into the RH rod end from the LH side. Position the RH rod end on the LH side of the cam assembly, and position the LH rod end on the RH side of the cam assembly.

j. Install a key on each rod end and secure with safety wire. Secure the jam nuts on the brackets at the propeller bellcrank with safety wire. Safety wire the bolt that attaches the rod end to the propeller control lever. Ensure that all cotter pins are properly installed.

k. Install the cockpit and cabin floorboards, as outlined in Chapter 53-20-00.

l. Install the aft wing fairings, as outlined in Chapter 53-50-00.

m. Install the engine cowling, as outlined in Chapter 71-10-00.

n. Perform the FINAL ENGINE CONTROLS RIGGING procedures located in this chapter, before returning the airplane to service.

FINAL ENGINE CONTROLS RIGGING
(Effectivity: All)
(Figures 3, 4, 5, 6, and 7)

a. Disconnect the propeller reversing cable from the cambox assembly (by removing the pin and pulling the rod end on the propeller reversing cable from the cam) to prevent damage to the carbon block in the following steps (Figure 6).

b. Set the rod ends on the interconnect rod located between the fuel control unit and the cambox so that the bolt holes measure 8.25 inches from center-to-center (see Figure 6).

c. Connect the forward end of the interconnect rod to the upper bolt hole in the fuel control unit arm (see Figure 6).

d. Connect the aft end of the interconnect rod to the cambox lever. Verify that the bolt in the rod end is installed in the hole located next to the top hole (see Figure 6).

e. Measure the deadband as follows (Figure 6):

1. Pull power lever back and up into the reverse range until the fuel control unit lever moves off of the deadband stop. Slowly move the linkage (power lever) forward until the deadband stop screw contacts the stop. The screw should be against the stop such that a piece of paper between the screw and the stop is held tightly and any further motion in the reverse direction will release the paper.

2. Use a protractor to measure the angle of the upper fuel control unit arm.

3. Advance the power lever until the stop screw is about to lift off the stop, but will still grip a piece of paper tightly.

4. Measure the angle of the upper fuel control unit arm with a protractor again. The difference between the two angular measurements represents the deadband travel and should be between 11.5° and 13.5°. If not, adjust the deadband stop screw to obtain the correct deadband. Turning the deadband screw in will widen the deadband .8° per 1/2-turn.

f. Position the deadband so that the deadband stop screw contacts the stop immediately aft of the GND FINE detent on the power lever at the pedestal, see step (1) above. The deadband stop screw should be about to lift off the stop at or immediately forward of the IDLE detent at the pedestal, see step (3) above (Figure 6).

g. Once set, the deadband may be repositioned between the GND FINE and IDLE by adjusting the serrated washer on the upper fuel control unit arm. A movement of one serration will result in a .6° change in the position of the arm. Fine adjustments can be made by lengthening or shortening the interconnect rod (see Figure 6).

NOTE

If the interconnect rod length is changed, the interconnect rod on the opposite engine should be changed to match before rigging the other engine. If the opposite engine has already been rigged, its interconnect rod length should not be changed.

If it is not possible to position the deadband between IDLE and GND FINE, the deadband can be increased slightly. However, the deadband should not extend more than 1° forward of the IDLE detent or more than 1° aft of the GND FINE detent, as measured on the cambox.

h. Set the Beta valve by sliding the propeller reversing cable to the full aft position and adjusting the aft rod end so that the flat surface of the clevis on the Beta valve is flush with the aft surface of the conical cap nut on the Beta valve. This adjustment should be made with the propeller solenoids de-energized (see Figure 7).

CAUTION

DO NOT place the power lever in the REVERSE range after the propeller reversing cable is connected to the cambox unless the engine is running.

NOTE

When lengthening the propeller reversing cable by adjusting the rod ends, at least one thread must be visible inside the clevis area.

i. Connect the forward rod end of the propeller reversing cable to the center hole in the Beta cam on the cambox (see Figure 6).

WARNING

After the Beta valve is adjusted, make certain that the flat surface of the clevis on the Beta valve is flush with the aft surface of the conical cap nut. Misadjustment of the Beta valve can cause unplanned feathering of the propeller, resulting in a possible hazard to airplane operation and overtorque damage to the engine.

j. Adjust the governor interconnect rod so that there is a .01-inch gap between the upper and lower halves (see Figure 7).

NOTE

Three complete turns of either the upper or lower half of the governor interconnect rod is equivalent to .10 inch.

k. Install safety wire on the propeller reversing cable, saddle bracket and torque adjusting screw. Be sure

that all safety wire and cotter pins are properly installed. Check that all clevis end nuts are properly tightened and safetied, as outlined in the Pratt & Whitney PT6A-67A Engine Maintenance Manual.

l. Adjust the low pitch throttle switch (ground idle stop switch) as outlined in THROTTLE SWITCH ADJUSTMENT, Chapter 61-40-00.

CAUTION

Before the first engine start, ensure that the fuel cutoff system is operating properly as outlined in FUEL CUTOFF CHECK, located in this Chapter. Be sure that the feather dump valve plunger on each propeller governor is fully compressed when the propeller control levers are placed in the FEATHER position.

m. Run both engines until the oil temperatures are well into the operating range. Note engine torques at 1500 rpm (reference the Starship 1 Pilot's Operating Handbook). Reference the GROUND PERFORMANCE WORKSHEET in Chart 2. Use Chart 2 to record the outlined engine data as final control adjustments are made in the following steps.

CAUTION

It is possible to get a "hung" start after making adjustments on the deadband or on the idle speed. If the engine shows a tendency to "hang" around 40% to 50% N1, advancement of the power lever approximately one inch should help. This can occur because the idle speed may be set too low, especially if large adjustments have been made.

WARNING

DO NOT MAKE ANY ADJUSTMENTS TO THE ENGINE, PROPELLER, ASSOCIATED LINKAGES OR ADJUSTMENT SCREWS WHILE THE ENGINE IS RUNNING. Ensure that the engine cowling is installed and that the engine oil temperature is within the operating range when taking readings. If adjustments are made outside the hangar, the propeller sling should be installed to prevent the propeller from windmilling as outlined in Chapter 10-20-00. Exercise caution when working on or near a hot power plant. Always remove all electrical power from the airplane and disconnect the battery before working on the engine or propeller.

n. Set the engine start position at 54.0% ±1% N1 on both engines. Verify that both engines are within .1% (see Figure 6).

1. Pull the power lever back to IDLE and set the condition lever to START.

2. Loosen the fuel control unit lever clamp screw.

3. There are two start speed adjustment screws located on the fuel control unit lever clamp. Increase the start speed by loosening the upper screw and tightening the lower screw an equal amount. To decrease the start speed, loosen the lower screw and tighten the upper screw an equal amount. The idle screws will adjust N1 2% per flat.

NOTE

Adjusting the start speed may affect the maximum forward and maximum reverse N1 speeds sufficiently to require resetting.

4. After the adjustments have been made, tighten the fuel control unit lever clamp screw and run the engines to check the start speed.

NOTE

A start speed adjustment will not affect the deadband. It merely turns the fuel control unit input shaft without disturbing the linkage geometry; however, it will shift the entire schedule so that the maximum forward and reverse stops will have to be reset.

o. Set the propeller flight idle low pitch blade angle to absorb a specific engine torque as follows:

NOTE

Reference Chapter 61-10-00 PROPELLER ASSEMBLY for more information on the propeller hardware.

1. With the propeller in the feathered position, adjust the Beta feedback collar home position as follows (Figure 3):

NOTE

The distance from the forward surface of the propeller flange to the aft surface of the feedback collar must be .985-inch.

a) Hold the Beta rod and loosen the lock nut at the Beta feedback collar.

b) Rotate the Beta rod making sure the aft-adjusting nut rotates with the Beta rod. Rotating the Beta rod clockwise will move the collar aft. Rotating the Beta rod counterclockwise will move the collar forward.

c) After the feedback collar home position has been adjusted, hold the Beta rods in place and torque the forward lock nuts.

2. Set the propeller flight idle low pitch blade angle to absorb a specific percentage of engine torque per Figure 4 within ±1.3% indicated torque. Calculate the "flight idle torque percentage" in Figure 4 by finding the intersection of the current outside air temperature (in degrees Celsius, found on the lower horizontal axis of the graph) with the altitude arc (located directly above the temperature). Draw a line from the temperature-altitude intersection straight across to the vertical axis of the graph and read the "flight idle torque percentage".

a) Start the engines and allow the oil temperature to increase into the operating temperature range.

b) Place the power levers at the IDLE stop and ensure that the condition levers are in the RUN position.

c) Place the propeller levers in the HIGH RPM position. Advance the propeller levers until propeller rpm is at 1600 RPM. Hold the PROP TEST switch in the LOW PITCH position. Adjust the propeller speed to 1500 RPM. Record the engine torque while holding the PROP TEST switch in the LOW PITCH position.

d) Adjust engine torque as indicated on the Engine Instrument, Crew Alerting System (EICAS) TORQ display (green) to match the "flight idle torque percentage" taken from Figure 4 to within ±1.3% at 1500 propeller rpm as follows (see Figure 3):

1) Shut down the engines as outlined in the Starship 1 Pilot's Operating Handbook. Remove the engine cowling as outlined in Chapter 71-10-00 to gain access to the Beta rod lock nuts.

2) Hold the Beta rod and loosen the lock nut at the Beta feedback collar.

3) Hold the aft adjusting nut and turn the Beta rod in or out. Turning the Beta rod in will increase torque, turning the Beta rod out will decrease torque. Each 1/2-turn of the Beta rod is equal to a 1% change in engine torque.

NOTE

When adjusting the Beta rods, set the distance between the aft edge of the aft adjusting nut and the aft end of the Beta rod to 1.15-inch. Use the 1.15-inch adjustment as a starting point, then adjust the Beta rods as required.

4) After adjusting the flight idle torque, hold the Beta rods in place and torque the lock nuts.

3. Make minor adjustments by adjusting the aft clevis or the torque adjusting screw. Each one-half turn of the aft clevis is equal to an approximate 1.2% change in engine torque. Turn the rod end in to decrease the propeller blade angle and engine torque. Turn the rod end out to increase propeller blade angle and engine torque. Make fine adjustments by turning the torque adjusting screw. Each one-half turn of the torque adjusting screw equals an approximate 1% change in engine torque (Figure 7).

NOTE

Set the torque adjusting screw so that there is .55-inch between the aft edge of the locknut (which is safety wired to the aft edge of the aft clevis) and the aft end of the torque adjusting screw. Use the .55-inch adjustment as an inital starting point and make further adjustments as necessary. Do not set the aft end of the torque adjusting screw any closer than .45-inch to the locknut. If adjustments of less than .45-inch are required, adjust the aft clevis instead.

p. Set the propeller ground idle low pitch blade angle to absorb a specific engine torque.

1. Set the ground idle engine torque to a specific percentage of torque per Figure 5 within ±1.3%. Calculate the ground idle torque percentage in the graph in Figure 5 by intersecting the outside air temperature with the proper altitude arc. Draw a line straight across the graph from the temperature-altitude intersection to the vertical axis and read the "ground idle torque percentage".

a) Start the engines and allow the oil temperature to increase into the operating range.

b) Position the airplane crosswind, 90° to the wind direction to eliminate variation in parameters due to changing wind velocity.

c) Place the power levers at the IDLE stop and ensure that the condition levers are in the RUN position.

d) Place the propeller levers in the HIGH RPM position. Advance the propeller levers until propeller rpm is at 1500 RPM and record the engine torque.

e) Adjust engine torque as indicated on the Engine Instrument, Crew Alerting System (EICAS) TORQ display (green) to match the "ground idle torque percentage" taken from Figure 5 to within ±1.3% at 1500 propeller rpm as follows:

1) Shut down the engines as outlined in the Starship 1 Pilot's Operating Handbook. Remove the engine cowling as outlined in Chapter 71-10-00 to gain access to the ground idle solenoid.

2) Loosen the two attachment bolts that secure the ground idle solenoid (see Figure 7).

3) Adjust the ground idle solenoid forward to decrease propeller blade angle and engine torque, and adjust to the aft to increase blade angle and engine torque (see Figure 7).

4) Tighten the two ground idle solenoid attachment bolts after the adjustments are complete (see Figure 7).

q. Check for throttle mismatch by running the engines full travel in forward and reverse. Confirm that the power lever handles are within 1/8 inch of each other when both engines are at the same N1 speed. Minor mismatch should be rigged out by adjusting the serrated washer and interconnect rod at the fuel control unit. For best results, adjust both engines by moving each power lever toward the other for half of the necessary adjustment (see Figure 6).

NOTE

It may be necessary to slightly widen the deadband to cover both the IDLE and the GND FINE stops. If this is done, the IDLE may need to be reset. If the power lever handles do not match after the above adjustments have been made, perform the following:

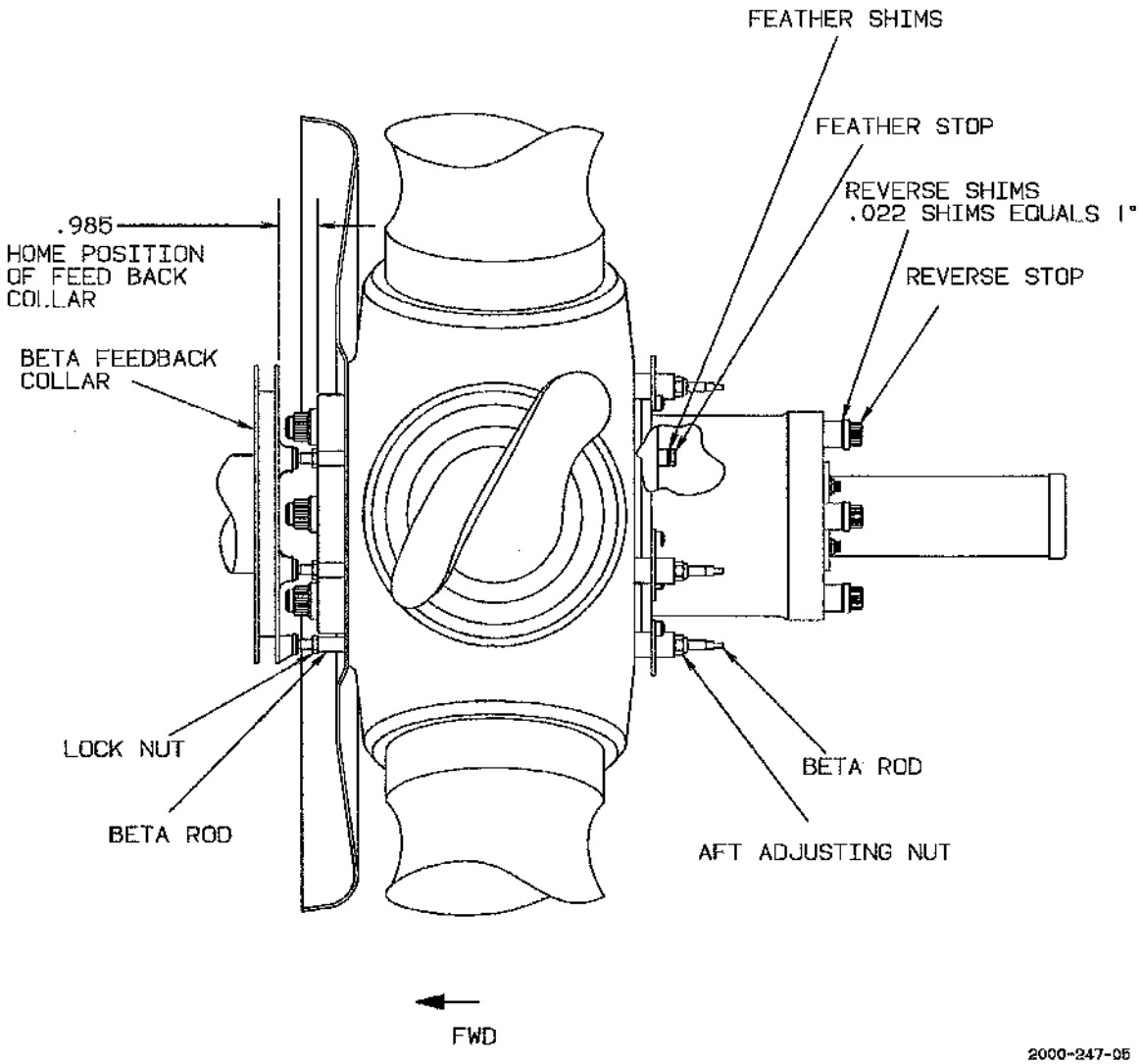
1. Ensure that the power control cables have been properly rigged as outlined in POWER CONTROLS RIGGING. Be sure that the cambox rigging holes are properly aligned. Ensure that the interconnect rods are the same length and that all rod end bolts are installed the proper holes (Figure 2).

2. Ensure that the cables in the control pedestal are correctly rigged, that the control rods in the pedestal are identical left and right (Figure 2).

NOTE

If an inconsistent amount of power lever mismatch is present (for example, if the power levers are together at idle and maximum, but mismatched in between), one fuel control unit or the other may be suspect and should be tested or replaced.

3. If all rigging is correct and the power levers are still mismatched, inspect the fuel control unit and the engine for faults.



Propeller Beta Rods (Effectivity: All)
Figure 3

NOTE

When attempting to match a new engine to and old one, it may be necessary to alter the adjustment of the cables in the pedestal or to alter the engine-to-engine rigging of the camboxes in order to match the power levers.

Fuel flow rates and ITT will not match between engines with differing time in service.

r. Set the RUN position to 66.0% ±.5% N1. The run idle screw will adjust .5% per flat. Adjust the N1 speed as follows (see Figure 6):

1. Place the power levers at the IDLE detent and advance the condition lever to the RUN position. Check the N1 speed with the bleed air OFF, less than 50% generator load, and the air conditioning compressor off.

NOTE

Do not set the run idle roller so far out toward the run idle select cam that the deadband stop screw does not ride against the deadband stop surface. If this happens, adjust the run idle stop screw to allow more rotation of the condition lever shaft, then readjust the run idle roller.

2. If the engine does not respond properly in step (1), check the condition levers to see if they are mismatched. If they are not, adjust the run idle roller position by loosening the lock nuts and moving the roller toward the run idle select cam to increase, or away from the cam to decrease N1 speed.

3. If the condition levers are mismatched, ensure that the control cable rod ends are installed properly, then adjust the run idle stop by turning the run idle stop screw in to increase, or out to decrease N1 speed. Each flat of the run idle stop screw changes N1 speed approximately .5%.

s. Set maximum reverse power to 84% ±1% N1 (see Figure 6).

1. Move the power lever to the full reverse position.

2. Confirm that N1 is 84% N1 ±1%. If not, adjust the maximum reverse N1 speed by turning the maximum reverse stop screw in to increase, or out to decrease N1 speed. The maximum reverse stop screw will adjust N1 .9% per flat.

t. Set maximum forward power to 104% N1. The maximum N1 stop screw will adjust .45% per flat. Maximum forward power should be checked in flight under conditions which will not exceed engine torque and temperature limits (see Figure 6).

u. Check the feathered blade angle. The propeller blades feathered blade angle should be visually checked during flight. If the engine rotates during shutdown with the propeller in the feathered condition, the feathered blade angle should be adjusted. Counterclockwise propeller rotation (as viewed from the aft) requires an increase in blade angle, while clockwise propeller rotation will require a reduction in blade angle. The feathered blade angle can be adjusted by adding or removing shims. Adding shims will decrease blade angle, while removing shims will increase blade angle (see Figure 3). If changes in the feathered blade angle are required, refer to the McCauley Service and Overhaul Manual P/N 890119.

v. Reverse blade angle can be adjusted by adding or removing shims. Adding reverse shims will increase the reverse blade angle, and removing reverse shims will decrease the reverse blade angle (see Figure 3). Reference the McCauley Service and Overhaul Manual P/N 890119 for details concerning propeller reverse blade angle.

w. Perform the ENGINE OPERATING PARAMETERS CHECK as outlined in Chapter 77-00-00, before returning the airplane to service.

FUEL CUTOFF CHECK (Effectivity: All)

a. Disconnect the engine fuel line at the fuel flow divider. Reference Chapter 73-10-00. Place a container under the fuel line to catch any fuel that may be pumped out of the line during this check.

b. Turn the Master-Battery switch ON.

c. Motor the engine by placing the ENGINE START switch in the START ONLY position.

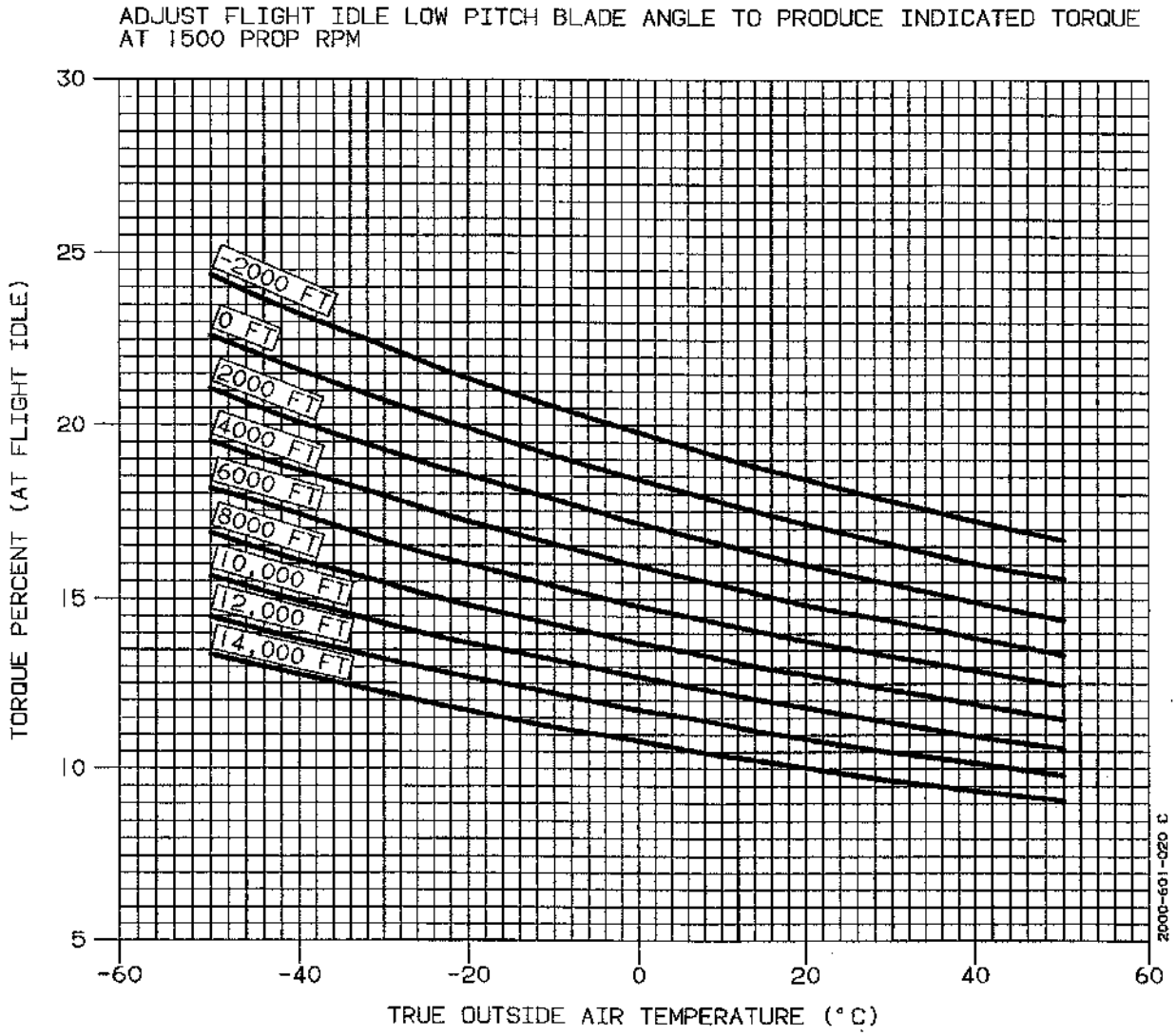
d. Move the condition lever forward and confirm that fuel flow is present.

e. Place the condition lever in the FUEL CUTOFF position and confirm that the fuel flow has stopped.

f. Connect the fuel line to the fuel flow divider.

POWER CABLE HEATER CHECK (Effectivity: All)

The power cable heaters should be checked to see that the heating elements are functioning as follows:



FIND INTERSECTION OF FIELD ELEVATION ARC AND AIR TEMPERATURE.
READ TORQUE PERCENTAGE ON LEFT SIDE OF GRAPH.

Flight Idle Torque Adjustment (Effectivity: All)
Figure 4

a. Remove the RH and LH aft wing fairings as outlined in Chapter 53-50-00 to gain access to the power cables.

b. Turn the Master-Battery switch ON.

c. Ensure that the CABLE HEATER, LEFT and RIGHT circuit breakers, (located in the ICE PROTECT section of the auxiliary circuit breaker panel) are pushed in.

d. Turn the LEFT and RIGHT VENT/CABLE switches (located on the LH inboard subpanel), ON.

e. After approximately two minutes, feel the power cables to see if they are warm. Both the LH and RH power cable heaters should be warm to the touch.

f. Turn the LEFT and RIGHT VENT/CABLE switches OFF.

g. Turn the Master-Battery switch OFF.

h. Install the RH and LH aft wing fairings as outlined in Chapter 53-50-00.

ENGINE CONTROL CABLES TEST AND REPAIR PROCEDURES (Effectivity: All)

The engine control cables are pre-lubricated by the cable manufacturer and the ends are sealed to reduce the possibility of ingress by moisture or contaminants into the cable housings. If problems are encountered with binding, stiff operation or freezing of the cables, the following procedures should be followed:

NOTE

The procedures which follow are for the condition and propeller cables only. The following equipment is required for testing, purging and repair of the engine control cables:

1. A regulated vacuum source.
2. A regulated dry nitrogen source.
3. A compressed air source.
4. Snap ring pliers.
5. Dual Manifold Gage Assembly

LEAK CHECK PROCEDURE (Effectivity: All)

a. Visually inspect the cable assemblies for damage, look for crimps, cuts, abrasions, unusally tight bends or installation damage.

b. Disconnect the suspect cable from the engine as outlined under CONDITION or PROPELLER CONTROLS RIGGING, as applicable, located in this chapter. If the suspect cable is the forward propeller cable or forward condition cable, remove the pedestal side panels and cockpit floorboards as outlined in Chapter 53-20-00. Disconnect the cable at the cam assembly.

c. Using snap ring pliers, remove the snap ring and O-ring from the engine side of the cable. Remove the snap ring and O-rings from the pedestal side of the cable if the suspect cable is a forward cable.

d. Connect the manifold gage assembly to the control cable, and connect the vacuum source to one side of the manifold.

e. Apply vacuum (20-in. HG) to the cable, and close the valve which controls the vacuum side of the manifold, thereby trapping the vacuum inside the cable housing.

f. Monitor the vacuum gage; there should be no loss of vacuum pressure during a two minute period.

g. Loss of vacuum pressure will indicate either a leaking O-ring at the opposite end of the cable, or a leak in the cable housing cover.

INSPECTION AND REPAIR OF DAMAGED CABLES (Effectivity: All)

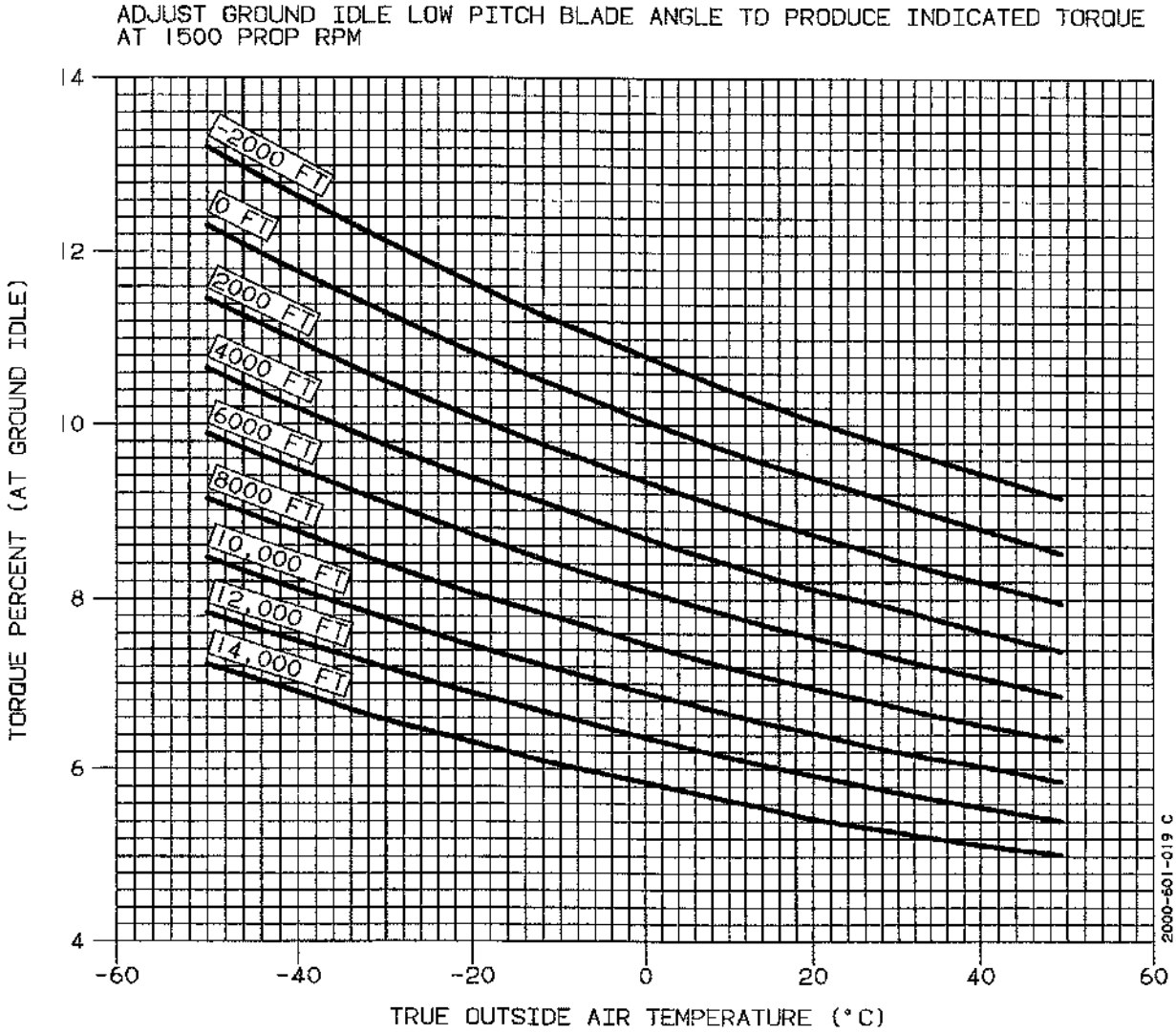
This procedure can be used to repair cuts or cracks in the outer teflon covering of the cable only. A repaired cable that exhibits any additional signs of leaking or freezing should be retested and purged. If freezing still persists, the cable should be replaced.

a. Visually examine the cable to determine the extent of the damage. The cable assembly must be discarded if the housing has been crushed, the binder or strand wires cut through or if the binder or strand wires have been contaminated (by petroleum products, dirt, etc).

NOTE

The purpose of this repair is to seal the damaged area and reduce the possibility of entry of moisture or other other contaminants into the cable housing, and to protect the assembly from further damage or wear from abrasion.

b. If examination indicates that damage is confined to the teflon cover, proceed with the following steps:



FIND INTERSECTION OF FIELD ELEVATION ARC AND AIR TEMPERATURE.
READ TORQUE PERCENTAGE ON LEFT SIDE OF GRAPH.

Ground Idle Torque Adjustment (Effectivity: All)
Figure 5

NOTE

If there is reason to believe that a substantial quantity of moisture (water vapor or condensation) has accumulated in the cable assembly, it should be baked dry at 125° Celsius for four hours.

1. Clean the damaged area, plus three inches on either side, using a clean cloth which has been dampened with 100% alcohol (1, Chart 1) or trichlorethane (2, Chart 1). Wipe the area dry with a clean dry cloth.

2. Wrap the area two inches either side of the damaged area with pressure sensitive teflon tape (.005-inch thick X 3/4-inch wide extruded type tape - may be locally obtained). Wrapping should be performed so that the tape overlaps itself by one-half of its width.

NOTE

Be sure to select teflon tape with a temperature range suitable for use in the area being repaired.

3. In areas subject to abrasion damage (clamp points and areas where the cable passes through bulkheads or firewalls), a second layer of teflon tape should be applied. This layer should be applied in the opposite direction of the first layer.

NOTE

If additional protection is required, heat-shrinkable teflon tubing may be applied over the teflon tape. The largest size possible should be used to achieve maximum shrinkage.

c. Heat shrink-shrinkable teflon tubing may be applied over the teflon to provide additional protection. The thickness of the covered area over the existing cable housing should be equal to, or greater than .010-inch. Maximum shrinkage may be obtained by using the largest size of tubing possible. The heat-shrink tubing should extend at least .25-inch beyond the tape wrapped area.

d. Heat the tubing above the crystalline melt point of the tubing (327°F) with a forced air heat gun. As the tubing begins to shrink, remove the heat (20% to 30% shrinkage takes place as the tubing cools).

e. Apply teflon heat-shrink tubing to the high abrasion areas when installing new control cables.

f. After the repairs are completed, perform the following NITROGEN CABLE PURGING procedure.

NITROGEN CABLE PURGING (Effectivity: All)

a. With the vacuum source connected to the cable, remove the snap ring and O-ring from the opposite end of the cable. Reference Chapter 53-50-00 for removal of the aft wing fairings and reference Chapter 53-20-00 for removal of the aft cabin floorboards as necessary.

b. Connect a regulated nitrogen source to the cable and adjust the regulator output pressure to 18 psig.

c. Purge the cable with 18 psig for at least 15 minutes. Install a new O-ring, and snap rings at the opposite end of the cable.

d. Apply 10 psig dry nitrogen pressure to the cable, trap the pressure in the cable and monitor the pressure gage for a two minute period. There should be no observed pressure loss.

e. Remove the gage manifold assembly and install a new O-ring, washer and snap ring on the end of the cable.

f. Rig the cable to the airplane as outlined under CONDITION or PROPELLER CABLE RIGGING, as applicable, located in this chapter.

CHART 2
GROUND PERFORMANCE WORKSHEET

AIRSPLANE S/N:_____ ENGINE S/N: LH_____ RH:_____

TECHNICIAN:_____ DATE:_____ FIELD ELEVATION:_____

OUTSIDE AIR TEMP:_____ PRESSURE ALTITUDE:_____

FUEL CUTOFF CHECK: LH_____ RH_____

ENGINE START COMMENTS: LH_____

RH_____

ENGINE OIL TEMP: LH_____ RH_____

GROUND PERFORMANCE CHECKS

START N1: (TARGET 53-55%)	BEFORE ADJUSTMENT LH_____ RH_____	AFTER ADJUSTMENT LH_____ RH_____
RUN-IDLE N1: (TARGET 65.5-66.5%)	BEFORE ADJUSTMENT LH_____ RH_____	AFTER ADJUSTMENT LH_____ RH_____
MAX FORWARD N1: (TARGET 104%)	BEFORE ADJUSTMENT LH_____ RH_____	AFTER ADJUSTMENT LH_____ RH_____
MAX REVERSE N1: (TARGET 83-85%)	BEFORE ADJUSTMENT LH_____ RH_____	AFTER ADJUSTMENT LH_____ RH_____
MAX N2: (TARGET 1700 RPM)	BEFORE ADJUSTMENT LH_____ RH_____	AFTER ADJUSTMENT LH_____ RH_____

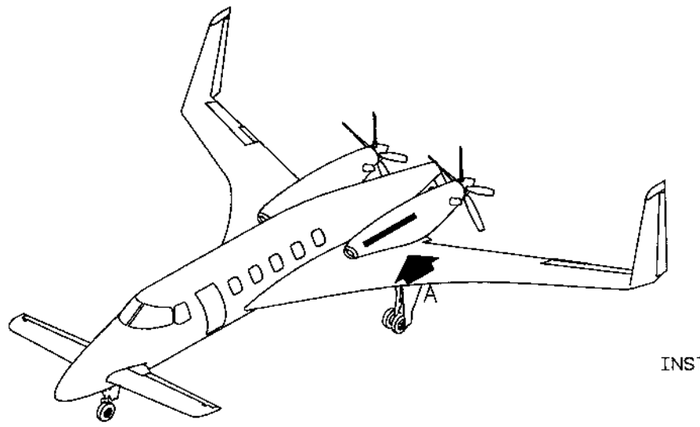
FLIGHT IDLE TORQUE - HOLD PROP TEST SWITCH AT IDLE, IN RUN @ 1500 N2

1500 RPM TORQUE FROM FIGURE 4: (± 1.3% INDICATED TORQUE)	BEFORE ADJUSTMENT LH_____ RH_____	AFTER ADJUSTMENT LH_____ RH_____

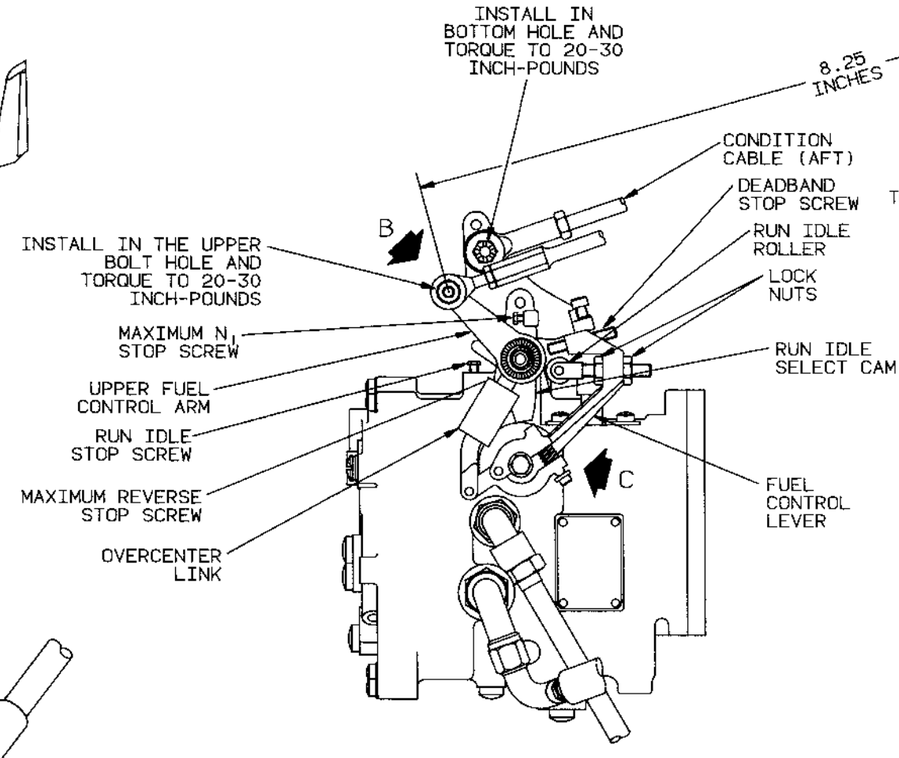
GROUND IDLE TORQUE - AT IDLE, IN RUN @ 1500 N2

1500 RPM TORQUE FROM FIGURE 5: (± 1.3% INDICATED TORQUE)	BEFORE ADJUSTMENT LH_____ RH_____	AFTER ADJUSTMENT LH_____ RH_____

BT01566

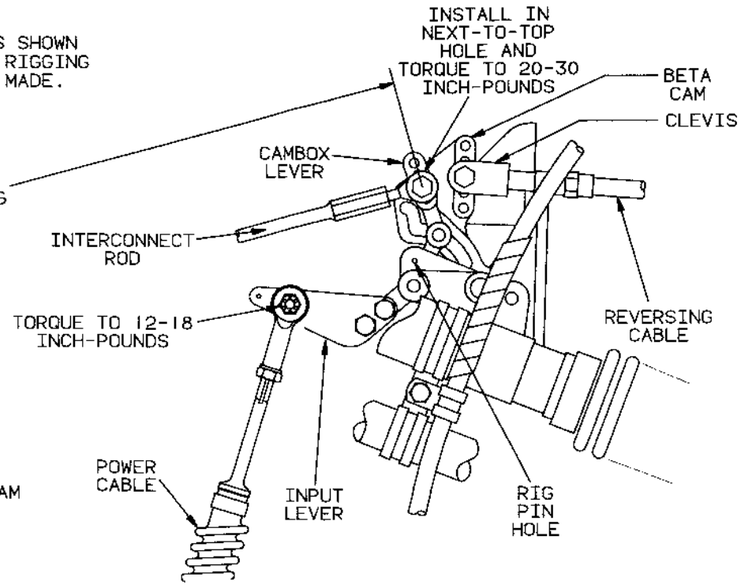


NOTE: TORQUE TO VALUES SHOWN AFTER ALL FINAL RIGGING ADJUSTMENTS ARE MADE.

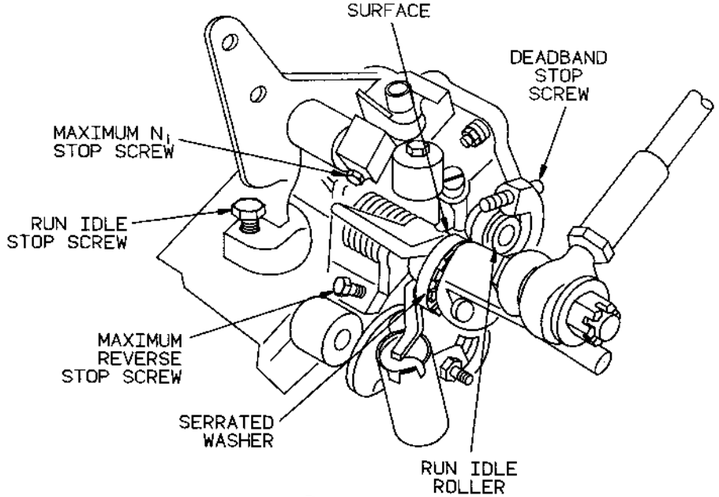


FUEL CONTROL UNIT

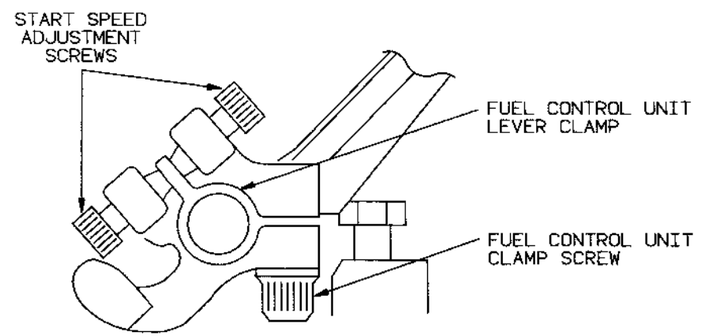
DETAIL A



CAMBOX



DETAIL B



DETAIL C

2000-159-11

Fuel Control Unit and Cambox Rigging (Effectivity: All)
Figure 6

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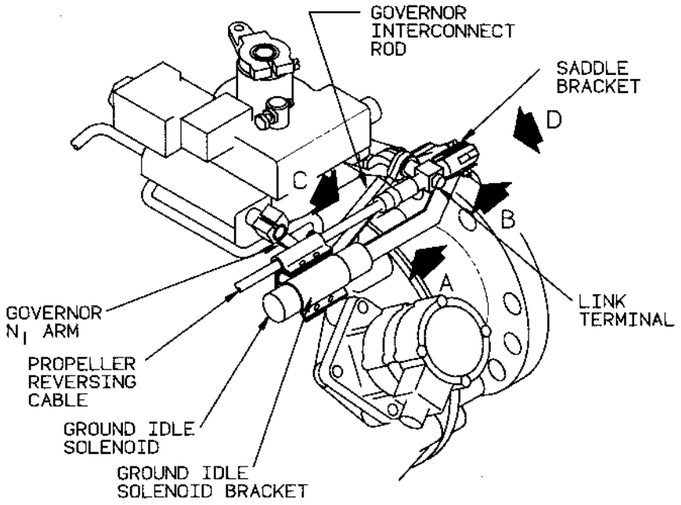
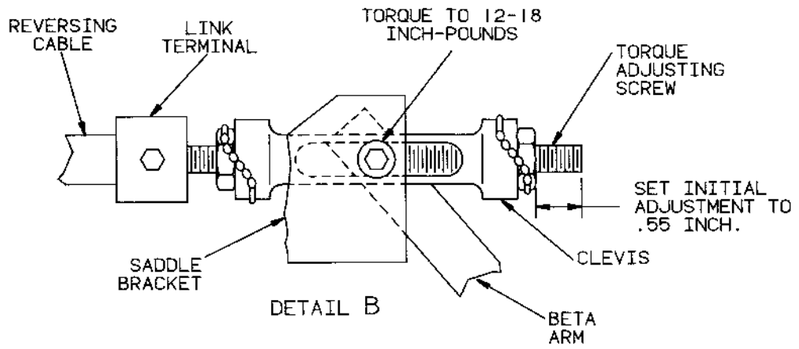
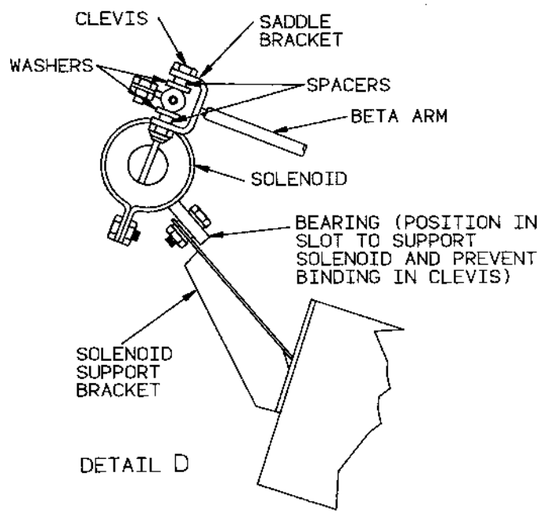
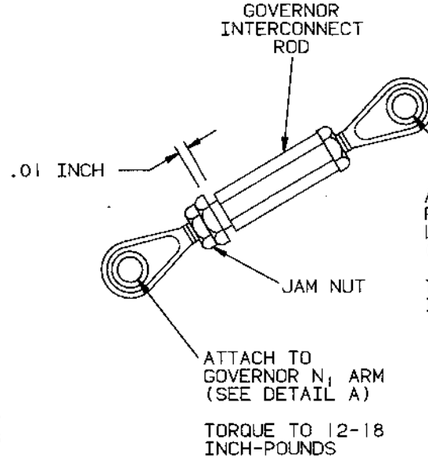
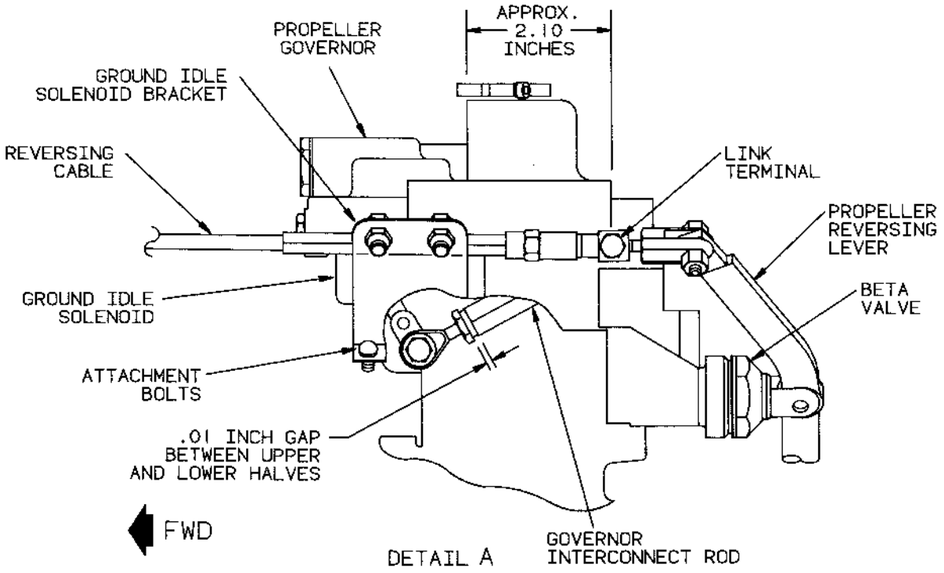




NOTE: TORQUE TO VALUES SHOWN AFTER ALL FINAL RIGGING ADJUSTMENTS ARE MADE.

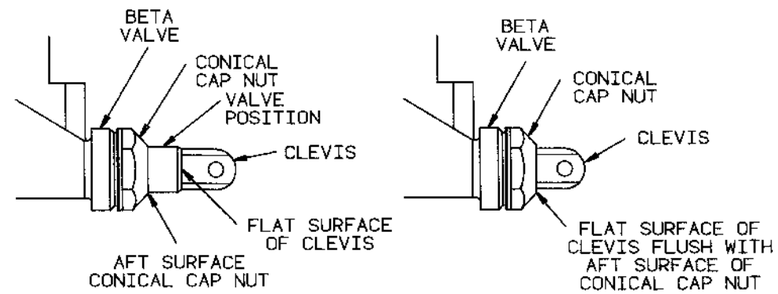






WARNING

MISADJUSTMENT OF THE BETA VALVE CAN CAUSE UNPLANNED FEATHERING OF THE PROPELLER, RESULTING IN A POSSIBLE HAZARD TO AIRPLANE OPERATION AND OVERTORQUE DAMAGE TO THE ENGINE.



VALVE SHOWN IN FULL AFT (UNADJUSTED) POSITION FOR CLARITY

VALVE SHOWN IN ADJUSTED POSITION

BETA ARM NOT SHOWN FOR CLARITY

BETA VALVE ADJUSTMENT

2000-159-10

Propeller Reversing and Beta Valve Adjustment (Effectivity: All)
Figure 7

77 - ENGINE INDICATING

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Beechcraft
STARSHIP 1
MAINTENANCE MANUAL

CHAPTER 77 - ENGINE INDICATING

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GENERAL MAINTENANCE PRACTICES (Effectivity: All)

The Starship 1 engine indicating system is part of the Automatic Data Acquisition System (ADAS). This system utilizes two Engine Data Concentrators (EDC), a Data Acquisition Unit (DAU) and ARINC 429-6 low speed data buses to compile, encode and transmit engine parameters from the various engine sensing devices to the avionics interface and the display units in the instrument panel. When only one engine performance parameter is suspect, it may be assumed that the problem is in the engine indicating system. When the fault is in the engine itself, it is common for at least two of the performance parameters to be outside of the expected limits. In either case, the engine should never be automatically condemned without a thorough investigation of the engine indicating system.

The ENGINE OPERATING PARAMETER CHECKS in this section should be performed to determine if a problem exists with engine performance or the engine indicating system.

The ENGINE DATA SIMULATION TEST in Chapter 77-40-00 is used to check the engine indicating system downstream of the EDC's by inputing known values into the EDC and watching the engine displays in the cockpit for display of the correct parameters.

The EDC TORQUE CALIBRATION in Chapter 77-40-00 should be accomplished anytime the torque indicating system requires adjustment.

ENGINE OPERATING PARAMETERS CHECK (Effectivity: All) (Figure 1)

NOTE

Engine parameter check data may be used by maintenance personnel to evaluate the effects of component replacement, inaccuracies in the engine displays or progressive hot section deterioration; but, such data should never be used as the sole criteria for determining the airworthiness of an engine.

The graphs for engine operating parameter checks are presented as torque versus ambient temperature and pressure so that engine parameters can be checked over a wide range of conditions without overtorquing the reduction gearbox. Periodic engine checks must be carried out and changes in engine parameters noted.

All forms of engine deterioration will be accompanied by an increase in interstage turbine temperature and fuel flow at a given power setting. Compressor deterioration will, in most cases, be due to dirt deposits that cause an increase in gas generator speed to obtain a given power setting. This form of deterioration can be remedied by compressor washing as outlined in Pratt & Whitney PT6A-67A Engine Maintenance Manual. Hot-section deterioration will cause a decrease in the gas generator speed, an increase in fuel flow and an increase in ITT for a given power setting.

Make a copy of the data recording sheet provided in Chart 1 and record the engine performance data obtained during the following procedure.

CAUTION

To prevent damage to the engine as a result of centrifugal force bending the propeller blade closeouts until they make contact with the oil cooler exit duct, the propeller spinners must be installed to perform a ground engine run.

NOTE

The engine cowling must be installed in order to ensure consistency of engine check perameters.

a. Determine the field barometric pressure.

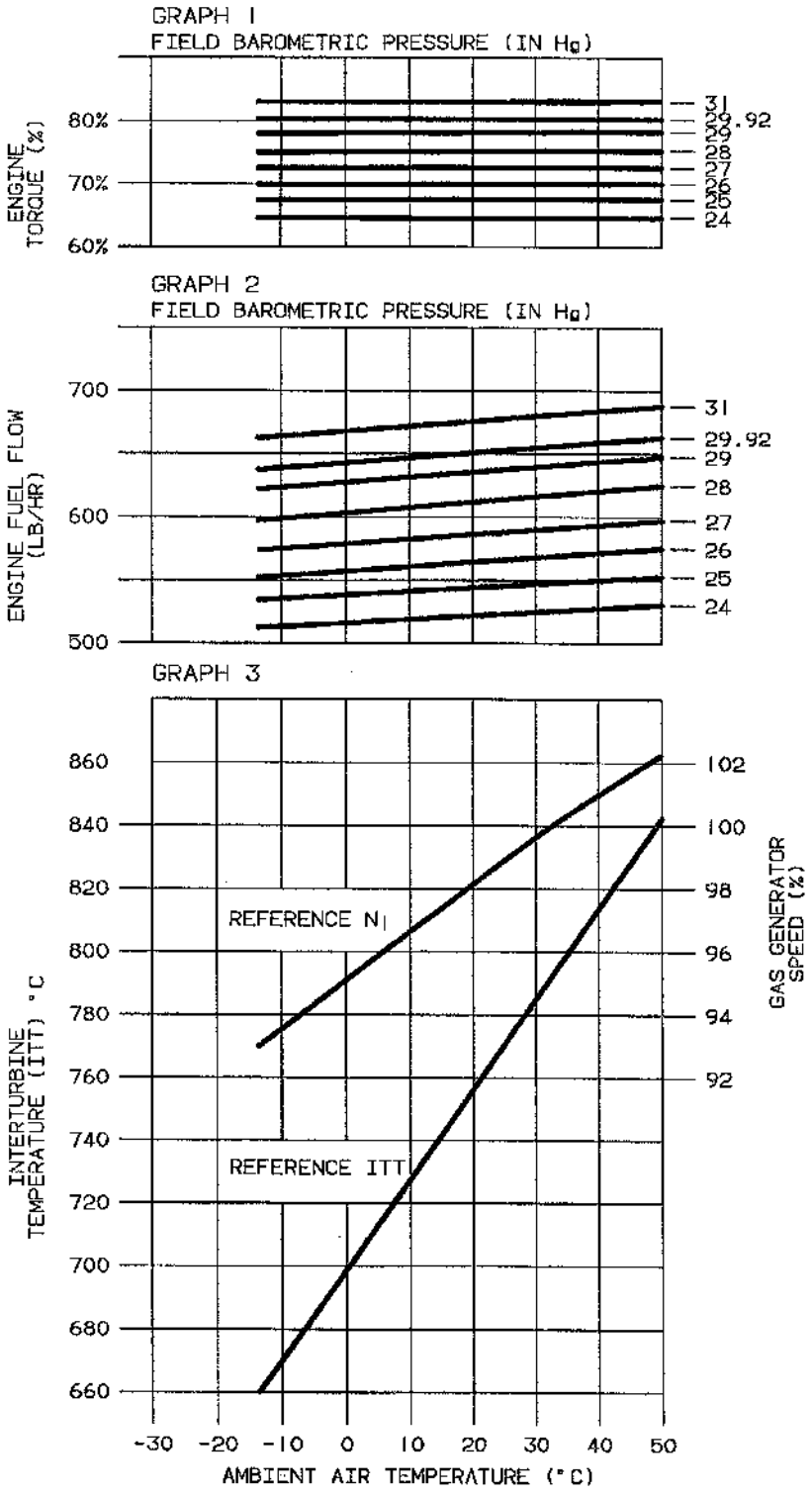
NOTE

This pressure may be obtained by setting the altimeter to zero and reading the pressure from the Kollsman window or by contacting the local Flight Service for an uncorrected pressure reading. Do not use the sea level pressure normally reported by the tower.

b. Record the outside air temperature in degrees Celsius.

c. Determine the target torque reading from Graph 1 of Figure 1 by plotting temperature against field barometric pressure.

d. Start the engines with procedures outlined in Section IV of the Starship 1 Airplane Flight Manual and allow the engine oil to reach normal operating temperature.



MODEL 2000

PT6A-67A
GROUND PERFORMANCE CHECK CHART
STATIC INSTALLED NP - 1700 RPM
EQUIV. 1000 SHP AT 1000 FT ALT

NOTE:

THIS GRAPH MUST NOT BE USED TO ACCEPT/REJECT AN ENGINE

2000-601-018 C

PT6A-67A Operating Parameters Check Graphs (Effectivity: All)
Figure 1

e. Position the airplane with the nose into the wind to eliminate variation in parameters due to changes in wind velocity.

f. Place the generator switch for the engine being checked in the OFF position.

g. Place the rotary BLEED AIR VALVES selector knob on the copilot's inboard subpanel in the OFF position.

h. Ensure that the environmental system temperature controls are set to prevent operation of the air conditioner compressor.

i. Ensure that the engine air inlet ice vane switches are in the OFF position.

j. Place the propeller levers in the HIGH RPM position.

k. Advance the power levers until a torque indication equal to the target torque value specified on Graph 1 in Figure 1 is attained.

l. Check that the engine propeller is governing at 1700 rpm.

m. Record the N_1 readings, ITT indications and the fuel flows for the engine being checked.

n. Shut down the engines according to procedures outlined in Section IV of the Starship 1 Pilot's Operating Manual.

o. Determine the reference fuel flow from Graph 2 in Figure 1 by plotting field barometric pressure against the outside air temperature.

p. Determine the reference N_1 speed from Graph 3 in Figure 1 based on outside air temperature.

q. Determine the reference ITT value by plotting the outside air temperature against the ITT curve shown on Graph 3 in Figure 1.

The graphs in Figure 1 are based on a typical new installation. Minor differences between engines and installations may produce results that are slightly different from the reference values represented by the graphs. Should an engine parameter begin to drift progressively up or down during subsequent checks, a trend toward engine deterioration may be assumed. The real value of the graphs in Figure 1 is in giving the technician an approximation of where the engine parameters should be for a typical new installation.

Detailed record keeping of engine performance parameters is all-important in detecting early trends of change in performance. For that purpose, a good system of engine condition trend monitoring should be employed by maintenance personnel so that preventive maintenance measures may be accomplished well in advance of the development of more serious conditions. Ensure that all data is recorded for future reference on a worksheet like the one shown in Chart 1.

CHART 1
ENGINE OPERATING PARAMETERS CHECK DATA WORKSHEET
(Effectivity: All)

Field Barometric Pressure _____	OAT _____
Torque From Graph 1 _____	Indicated LH _____ RH _____
Propeller Speed _____ (Target 1700 RPM)	Indicated LH _____ RH _____
Fuel Flow From Graph 2 _____	Indicated LH _____ RH _____
Engine N_1 From Graph 3 _____	Indicated LH _____ RH _____
ITT From Graph 3 _____	Indicated LH _____ RH _____
Engine Shutdown Comments:	LH _____ RH _____

SPECIAL TOOLS AND RECOMMENDED MATERIALS (Effectivity: All)

The special tools and recommended materials listed in Charts 2 and 3 as meeting federal, military or vendor specifications are provided for reference only and are not specifically prescribed by Beech Aircraft Corporation. Any product conforming to the specification listed may be used. The products included in these charts have been tested and approved for aviation usage by Beech Aircraft Corporation, by the vendor, or by compliance with the applicable specifications. Generic or locally manufactured products which conform to the requirements of the specification listed may be used even though not included in the charts. Only the basic number of each specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the vendor of the product to be used.

CHART 2
SPECIAL TOOLS (Effectivity: All)

TOOL NAME	PART NO.	VENDOR	USE
1. Engine Data Simulator	01-357-00	J.C. Air Industrial Airport, KS 66031	Test the engine indicating system
2. Deadweight Tester	2311F	Barfield Instrument Miami, FL 33152	Calibrate torque on the EDC's

CHART 3
RECOMMENDED MATERIALS (Effectivity: All)

MATERIAL	SPECIFICATION	PRODUCT	VENDOR
1. Plastilube Grease		Plastilub No. 3	Parr/Koppers Cleveland, OH 44102
2. Lubricating Grease gasoline and oil resistant	MIL-G-6032	Grease	Lehigh Cemicals Naddix Div., Tennaco Chemicals Inc., Chestertown, MD 21620

INTEGRATED ENGINE INSTRUMENT SYSTEMS - DESCRIPTION AND OPERATION (Effectivity: All) (Figures 1, 2 and 3)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The following provides information on the engine sensing devices and interface to the Automatic Data Acquisition System (ADAS). For detailed information on the ADAS system refer to Chapter 31-40-00.

The Engine Data Concentrators (EDC) provide the digital conversion, concentration and transmission of the engine operating parameters and several airplane system discretes. The EDC's are mounted (one per side) in the aft wing between the nacelle and the fuselage. Each EDC has two channels (A and B) that share an internal interconnect, but function independently of each other. Separation of the channels ensures that a failure on one channel does not prevent transmission of the engine data from the EDC via the opposite channel. If a fault exists on one channel, the EDC has the capability to detect the failure, transmit the appropriate diagnostic code to the self-diagnostic system for display during the self-diagnostic routine, and continue processing engine data for output on the remaining operational channel. Refer to the Collins Starship 1 Avionics Systems Manual P/N 523-0774076-00111A for detailed information on using the self-diagnostics routine. Should a fault occur in one of the engine sensing devices that causes an erroneous signal, or if the signal is not received by the EDC, the EDC will only transmit the data used to display the engine parameter scale and the actual engine parameters will be displayed as zero or blanked from the scale. The only adjustable EDC engine parameter is engine torque. Each EDC has two externally accessible torque calibration screws placarded OFFSET and SLOPE, allowing an adjustment of the factors used by each EDC to calculate torque from oil pressure.

The individual engine parameters are received by the EDC's as analog inputs directly from the respective engine parameter sensing devices. After the analog engine data is received by the EDC's, it is digitally converted and output from both channels to separate ARINC 429-6 low speed data buses. The ARINC 429-6 low speed data buses transmit the digital engine data to both channels of the Data Acquisition Unit (DAU). The DAU is located just aft of the pedestal under the cockpit floor and has two identical channels (A and B) to maintain separation of left and right engine data. After the DAU receives the digital engine data, it is compiled and transmitted through the instrument panel avionics interface to the Engine Instrument and Crew Alerting System (EICAS) display unit. In the normal mode, the engine data is displayed on the EICAS display. In the reversionary mode, the engine data is reverted from the EICAS display to the Multifunction Display (MFD). The switch placarded EICAS REV-NORM on the center reversionary panel selects either the normal or the reversionary display mode. If both the EICAS display and MFD are inoperable, ITT, Torq, Prop rpm and N_1 rpm data can be digitally displayed on either RTU 1 or RTU 2 by placing the respective RTU reversionary switch (located on the center reversionary panel) in the ENG DATA position. Refer to the Collins Starship 1 Avionics Systems Manual P/N 523-0774076-00111A for detailed information on the EICAS engine displays. Refer to Figure 1 to locate the engine parameter sensing devices that are discussed in the following paragraphs.

ENGINE TORQUEMETER (Effectivity: All)

The EDC provides a regulated voltage excitation signal to the torque transducer on the torque pressure manifold and reads the return voltage to determine engine torque. Resistance is read by the EDC and automatic adjustments up to ±5 percent are made as necessary by the EDC to correct sensor errors. Torque signals from the torque transducer are digitized by the EDC and transmitted along an ARINC 429-6 low

speed data bus through the Data Acquisition Unit (DAU) to the Engine Instrument, Crew Alerting System (EICAS) display. The left and right engine torque meters are located in the upper left corner of the EICAS display.

INTERSTAGE TURBINE TEMPERATURE (Effectivity: All)

Interstage Turbine Temperature data from the engine's T_5 terminal block is passed through a thermocouple wire harness as an input signal to the EDC. The input signal is digitized by the EDC and transmitted along an ARINC 429-6 low speed data bus through the Data Acquisition Unit (DAU) to the Engine Instrument, Crew Alerting System (EICAS) display. The left and right engine ITT meters are located in the upper left corner of the EICAS display.

PROPELLER TACHOMETER (Effectivity: All)

The propeller tachometer generator located on the engine reduction gearbox provides a propeller RPM signal to the EDC as an input. The input signal is digitized by the EDC and transmitted along an ARINC 429-6 low speed data bus through the Data Acquisition Unit (DAU) to the Engine Instrument, Crew Alerting System (EICAS) display. The left and right propeller tachometers are located in the center left side of the EICAS display.

N_1 TACHOMETER (Effectivity: All)

The turbine tachometer generator on the engine accessory gearbox provides an N_1 speed signal to the EDC as an input. The input signal is digitized by the EDC and transmitted along an ARINC 429-6 low speed data bus through the Data Acquisition Unit (DAU) to the Engine Instrument, Crew Alerting System (EICAS) display. The left and right engine N_1 tachometers are located in the lower left corner of the EICAS display.

FUEL FLOWMETER TRANSMITTER (Effectivity: All)

The fuel flowmeter transmitter is electrically powered with 5 ±0.5 vdc from the Engine Data Concentrator (EDC) and mechanically powered by fuel flow turning a six blade, free spinning rotor in the fuel flowmeter transmitter. At engine ignition and start, fuel flow from the engine fuel control unit to the fuel flow divider passes through the fuel flowmeter transmitter and rotates the fuel flowmeter transmitter's rotor, which sends a pulsating frequeny to the EDC as an input. The input signal is digitized by the EDC and transmitted along an ARINC 429-6 low speed data bus through the Data Acquisition Unit (DAU) to the Engine Instrument, Crew Alerting System (EICAS) display. The left and right fuel flow is displayed above the oil information in the lower right corner of the Engine Instrument, Crew Alerting System (EICAS) display.

ENGINE OIL TEMPERATURE BULB (Effectivity: All)

The engine oil temperature is monitored by an oil temperature bulb mounted on the accessory gearbox housing. A constant voltage is supplied to the resistance-type temperature bulb by the EDC. The input signal from the bulb to the EDC is digitized by the EDC and transmitted along an ARINC 429-6 low speed data bus through the Data Acquisition Unit (DAU) to the Engine Instrument, Crew Alerting System (EICAS) display, where the engine oil temperature is indicated on the oil temperature gage. The left and right engine oil temperature gages are located in the lower RH corner of the EICAS display.

ENGINE OIL PRESSURE TRANSDUCER (Effectivity: All)

The engine oil pressure is monitored by an oil pressure transducer mounted on the accessory gearbox housing of the engine. Electrical power for the transducers is through the L and R OIL PRESS circuit breakers on the LH circuit breaker panel in the pilot's console. The engine oil pressure is sensed by the transducer and converted by the transducer into an electrical signal that is transmitted to the Engine Data Concentrator (EDC). The input signal to the EDC will range from 0 to 5 vdc, with 0 volts being equal to 0 psi and 5 volts being equal to 200 psi. The EDC digitizes the signal and transmits it along an ARINC 429-6 low speed data bus through the Data Acquisition Unit (DAU) to the Engine Instrument, Crew Alerting System (EICAS) display. The left and right engine oil pressure gages are located in the lower RH corner of the EICAS display.

PROPELLER SYNCHRONIZERS (Effectivity: All)

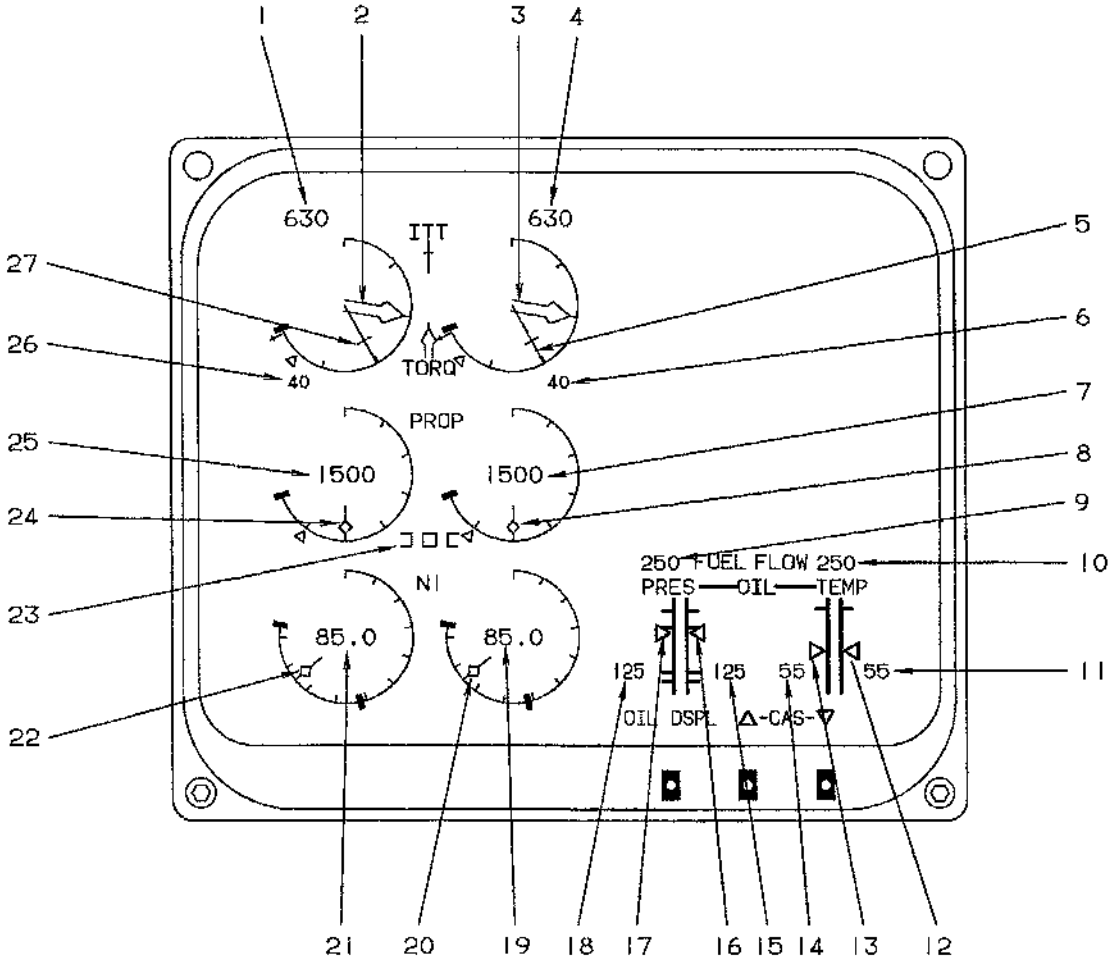
The propeller synchronizer display consists of a series of square white boxes which are located on the EICAS screen below the propeller display. If the white boxes

are motionless, the propellers are synchronized. When the white boxes move to the right, the RH propeller RPM is higher. If the white boxes are moving to the left, the LH propeller RPM is higher. The white boxes will disappear from the EICAS display if either propeller signal fails.

The synchronizer display works independant of the synchronizer system discussed in Chapter 61-22-00. The signals generated by the propeller tachometers are sampled in the DAU and processed by a comparator circuit to generate the signal which drives the display. For more detailed information and maintenance procedures on the synchronizer system refer to Chapter 61-22-00 in this maintenance manual.

ENGINE INDICATING - TROUBLESHOOTING (Effectivity: All)

The first indication that a check of the engine indicating system needs to be performed would be when one of the airplane's engines has a shift in engine operating parameters. Because the EDC's and DAU are Line Replaceable Units (LRU), a failure of either of these engine indicating system components will be displayed on the MFD when the self-diagnostics routine is selected. Refer to the Starship 1 Collins Avionics Systems Manual P/N 523-0774076-00111A for detailed information on using the self-diagnostic system. If no fault is indicated for the EDC's or DAU, and the engine indicating system is suspect, a check of the engine indicating system using the engine data simulator (1, Chart 3, Chapter 77-00-00) should be performed to isolate any potential faults in the system. Usually, when only one engine performance parameter is suspect, it may be assumed that the problem is in the engine indicating system. When the fault is in the engine itself, it is common for at least two of the performance parameters to be outside of the expected limits. In either case, the engine should never be automatically condemned without a thorough investigation of the engine indicating system.



1. L ENGINE ITT DIGITAL DISPLAY
2. L ENGINE TORQUE POINTER
3. R ENGINE TORQUE POINTER
4. R ENGINE ITT DIGITAL DISPLAY
5. R ENGINE ITT POINTER
6. R ENGINE TORQUE DIGITAL DISPLAY
7. R ENGINE PROP RPM DIGITAL DISPLAY
8. R ENGINE PROP RPM POINTER
9. L ENGINE FUEL FLOW DIGITAL DISPLAY
10. R ENGINE FUEL FLOW DIGITAL DISPLAY
11. R ENGINE OIL TEMP DIGITAL DISPLAY
12. R ENGINE OIL TEMP POINTER
13. L ENGINE OIL TEMP POINTER
14. L ENGINE OIL TEMP DIGITAL DISPLAY
15. R ENGINE OIL PRESS DIGITAL DISPLAY
16. R ENGINE OIL PRESS POINTER
17. L ENGINE OIL PRESS POINTER
18. L ENGINE OIL PRESS DIGITAL DISPLAY
19. R ENGINE N_1 RPM DIGITAL DISPLAY
20. R ENGINE N_1 RPM POINTER
21. L ENGINE N_1 RPM DIGITAL DISPLAY
22. L ENGINE N_1 RPM POINTER
23. PROPELLER SYNCROPHASE INDICATOR
24. L ENGINE PROP RPM INDICATOR
25. L ENGINE PROP RPM DIGITAL DISPLAY
26. L ENGINE TORQUE DIGITAL DISPLAY
27. L ENGINE ITT POINTER

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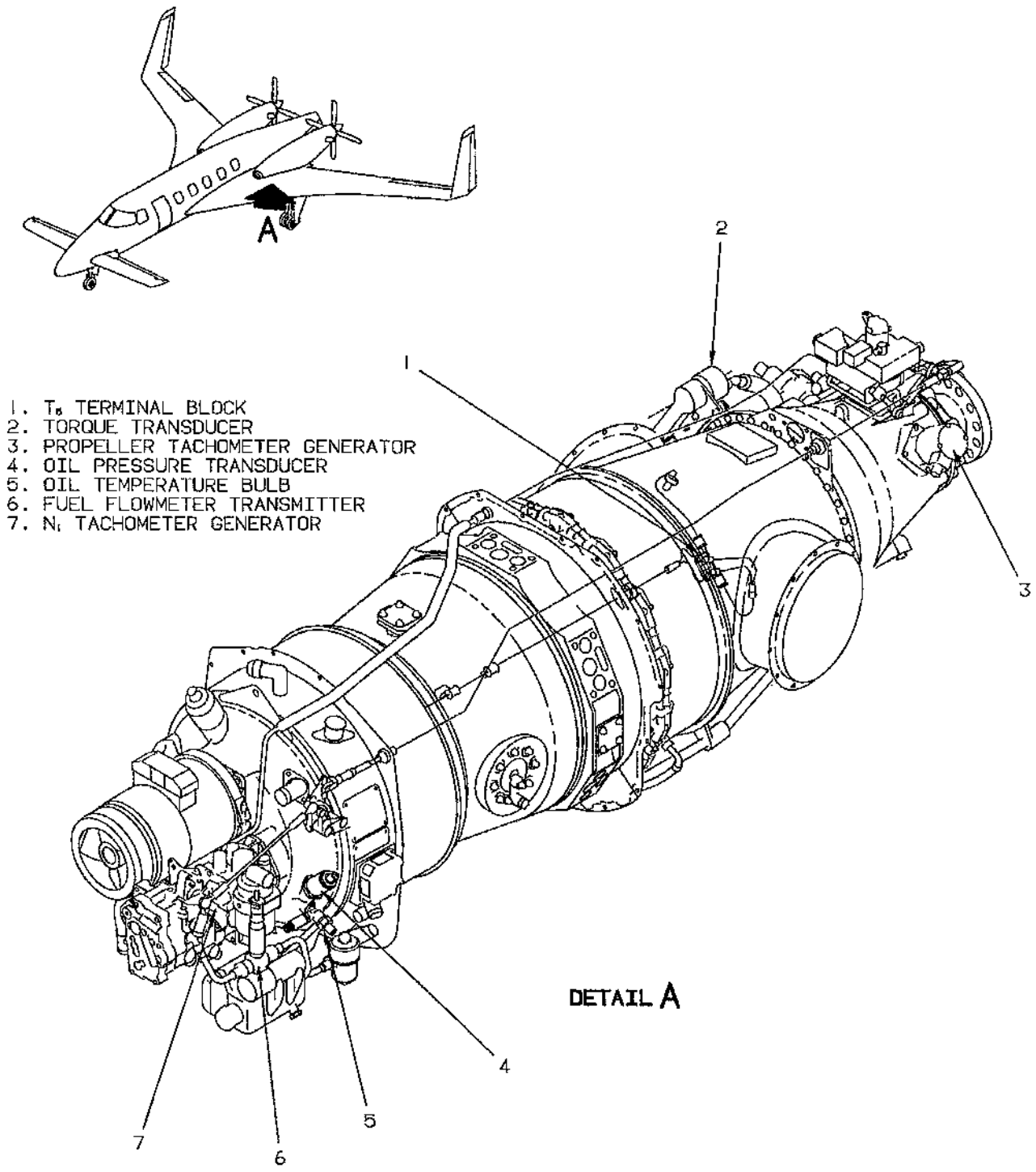
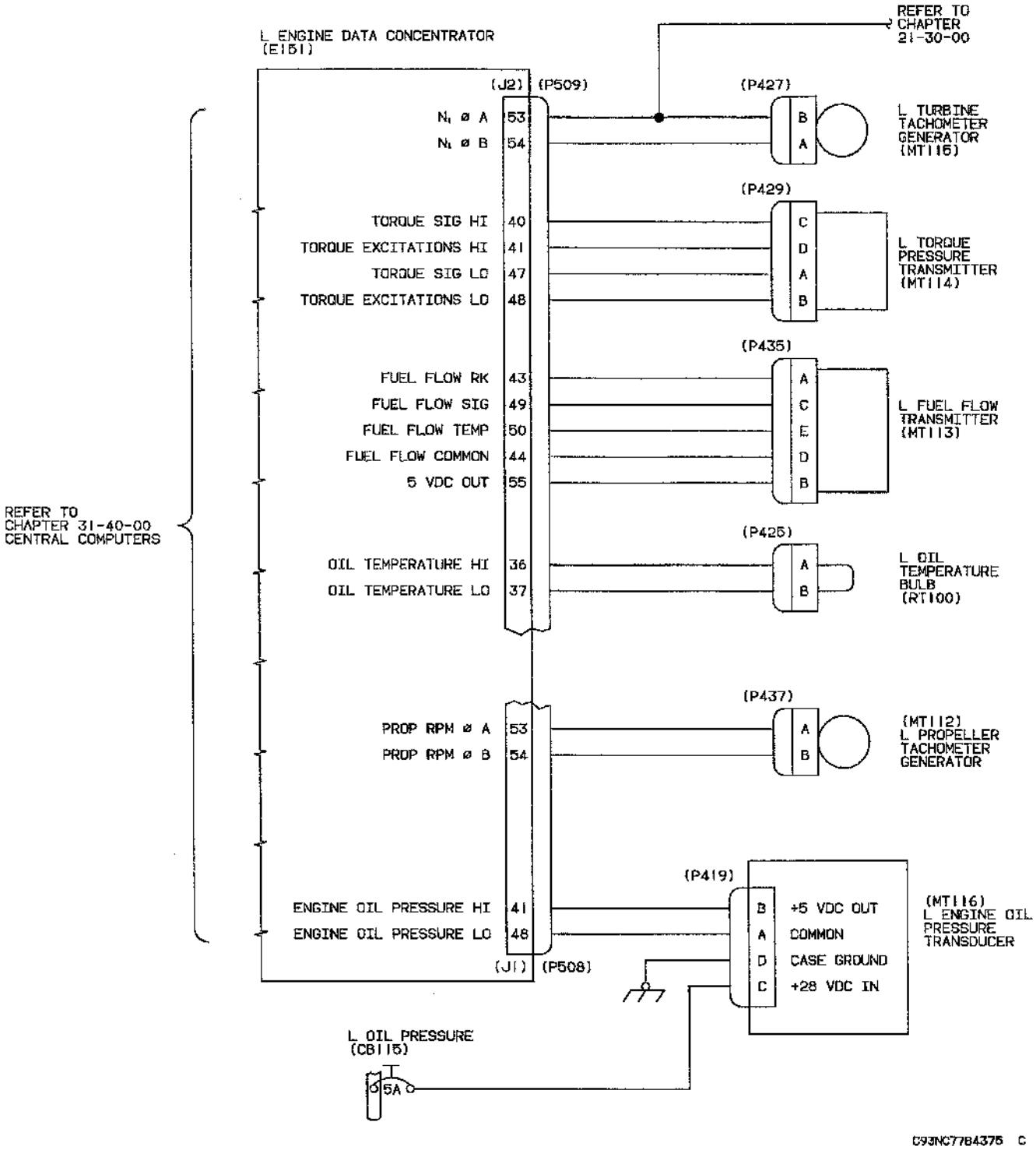
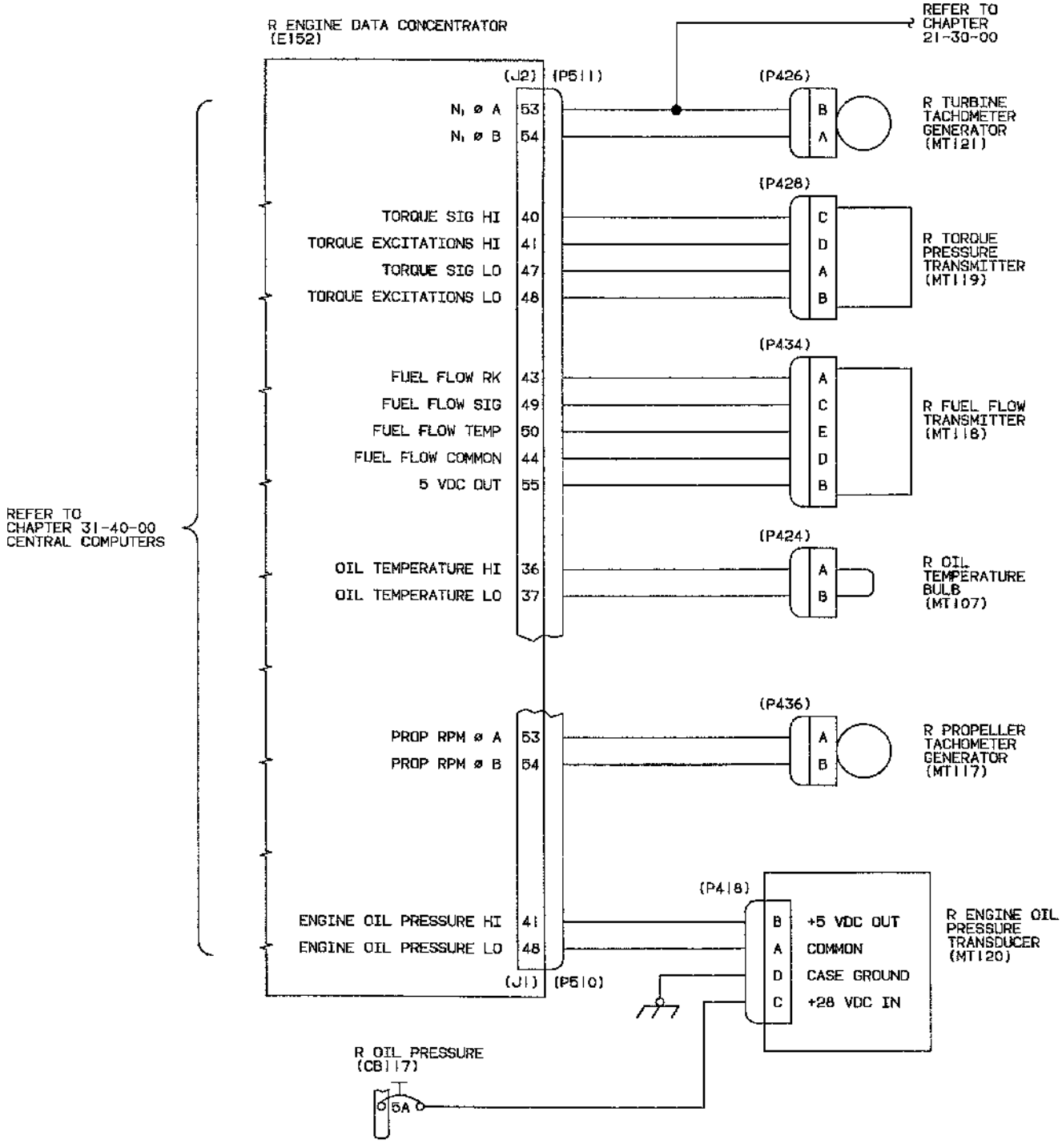
Engine Indicating EICAS Display (Effectivity: All)
Figure 1



Location Of Engine Parameter Sensing Devices (Effectivity: All)
Figure 2



Engine Indicating Functional Schematic Diagram (Effectivity: All)
(Sheet 1 of 2)
Figure 3



Engine Indicating Functional Schematic Diagram (Effectivity: All)
(Sheet 2 of 2)
Figure 3

ENGINE INDICATING - MAINTENANCE PRACTICES (Effectivity: All)

Most of the engine maintenance procedures outlined in this chapter are supplemental to the procedures outlined in the Pratt and Whitney PT6A-67A Engine Maintenance Manual and to the instructions provided with the test equipment used to check the system. Factors affecting the accuracy of the equipment, such as calibration, frequency of certification or misuse, must be taken into consideration before any equipment may be used to check and calibrate the engine indicating system.

ENGINE TORQUE TRANSDUCER REMOVAL (Effectivity: All) (Figure 4)

a. Remove electrical power from the airplane.

b. Refer to Chapter 71-10-00 and remove the required engine cowlings.

c. Remove the safety wire and disconnect the electrical connector (3) from the torque transducer (2).

d. Disconnect the B-nut (4) from the elbow (5).

NOTE

If the torque transducer is not being replaced proceed to step g, otherwise perform all steps.

e. Loosen the jamnut (7).

f. Remove the jamnut (7) preformed packing (6), and elbow (5) from the torque transducer (2).

g. Remove the torque transducer (2) and preformed packing (1) from the manifold (8).

ENGINE TORQUE TRANSDUCER INSTALLATION (Effectivity: All) (Figure 4)

a. Install a new preformed packing (1) on the torque transducer (2).

b. Install the torque transducer (3) into the manifold (8).

NOTE

If a new torque transducer is being installed perform all steps, otherwise proceed to step e.

c. Install a new preformed packing (6) and the jamnut (7) on the elbow (5).

d. Install the elbow (5) into the torque transducer (2) and tighten the jamnut (7).

e. Install the B-nut (4) on the elbow (5).

f. Connect the electrical connector (3) to the torque transducer (2) and safety wire.

g. Refer to Chapter 77-10-00 and install the required engine cowlings.

h. Verify proper system operation by performing the ENGINE OPERATING PARAMETERS CHECK in this chapter.

ENGINE PROPELLER TACHOMETER GENERATOR REMOVAL (Effectivity: All) (Figure 5)

a. Remove electrical power from the airplane.

b. Refer to Chapter 71-10-00 and remove the required engine cowlings.

c. Remove safety wire and disconnect the electrical connector (4) from the propeller tachometer generator (2).

d. Remove the nuts (5) and washers (6) from the studs (8).

e. Move the brackets (3 and 7) off the studs (8) and carefully position them out of the way.

f. Remove the propeller tachometer generator (2) and the gasket (1) from the airplane.

ENGINE PROPELLER TACHOMETER GENERATOR INSTALLATION (Effectivity: All) (Figure 5)

a. Lubricate the shaft of the propeller tachometer generator (2) with plastilub No. 3 (1, Chart 3, 77-00-00).

b. Position the propeller tachometer generator (2) and the gasket (1) on the studs (8).

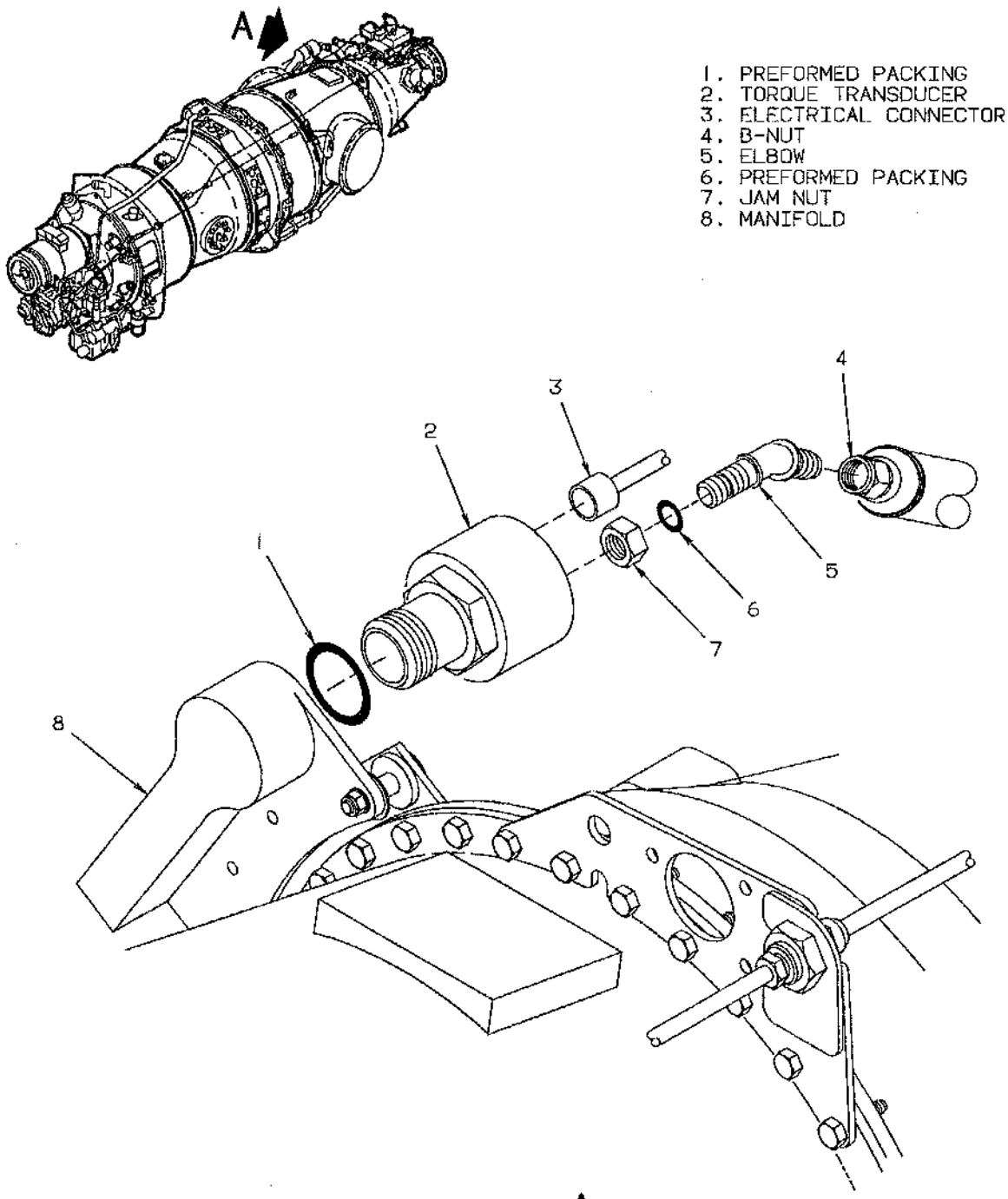
c. Position the brackets (3 and 7) on the studs (8).

d. Install the washers (6) and nuts (5) on the studs (8) and torque to 30 to 40 inch-pounds.

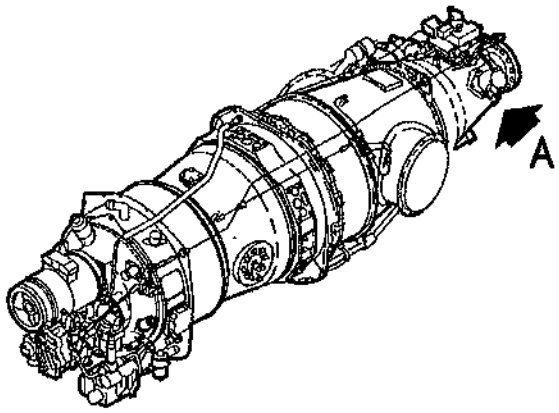
e. Install the electrical connector (4) on the propeller tachometer generator (2) and secure with safety wire.

f. Refer to Chapter 71-10-00 and install the required engine cowlings.

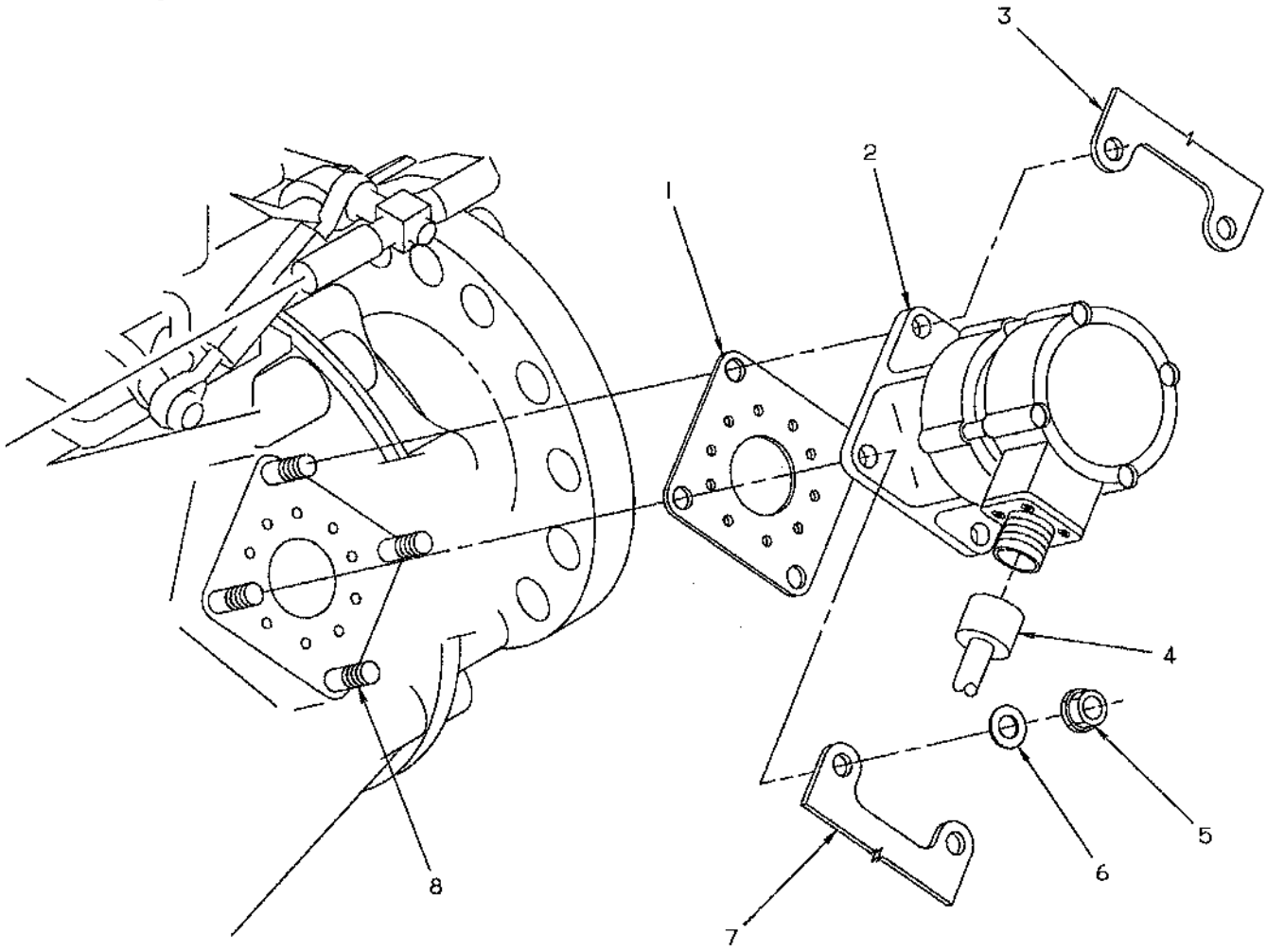
g. Verify proper system operation by performing the ENGINE OPERATING PARAMETERS CHECK in this chapter.



Engine Torque Transducer Installation (Effectivity: All)
Figure 4



1. GASKET
2. PROPELLER TACHOMETER GENERATOR
3. BRACKET
4. ELECTRICAL CONNECTOR
5. NUT
6. WASHER
7. BRACKET
8. STUD



DETAIL A

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Engine Propeller Tachometer Generator Installation (Effectivity: All)
Figure 5

ENGINE N_1 TACHOMETER GENERATOR REMOVAL (Effectivity: All) (Figure 6)

a. Remove electrical power from the airplane.

b. Refer to Chapter 71-10-00 and remove the required engine cowlings.

c. Remove safety wire and disconnect the electrical connector (4) from the N_1 tachometer generator (1).

d. Remove the nuts (5) and washers (6) from the studs (2).

e. Remove the N_1 tachometer generator (1) and the gasket (3) from the airplane.

ENGINE N_1 TACHOMETER GENERATOR INSTALLATION (Effectivity: All) (Figure 6)

a. Lubricate the shaft of the N_1 tachometer generator (1) with plastilub No. 3 (1, Chart 3, 77-00-00).

b. Position the N_1 tachometer generator (1) and the gasket (3) on the studs (2).

c. Install the washers (6) and nuts (5) on the studs (2) and torque to 30 to 40 inch-pounds.

d. Install the electrical connector (4) on the N_1 tachometer generator (1) and secure with safety wire.

e. Refer to Chapter 71-10-00 and install the required engine cowlings.

f. Verify proper system operation by performing the ENGINE OPERATING PARAMETERS CHECK in this chapter.

FUEL FLOWMETER TRANSMITTER REMOVAL (Effectivity: All) (Figure 7)

a. Refer to Chapter 71-10-00 and remove the required engine cowlings.

b. Remove electrical power from the airplane.

c. Remove safety wire and disconnect the electrical connector(1) from the fuel flowmeter transmitter(4).

d. Disconnect the fuel lines (2 and 3) from the fuel flowmeter transmitter (4).

e. Remove the fuel flowmeter transmitter (4) from the airplane.

FUEL FLOWMETER TRANSMITTER INSTALLATION (Effectivity: All) (Figure 7)

a. Position the fuel flowmeter transmitter (4) so that the flow arrow on the fuel flowmeter transmitter is pointing away from the fuel control unit.

b. Connect the fuel lines (2 and 3) to the fuel flowmeter transmitter (4).

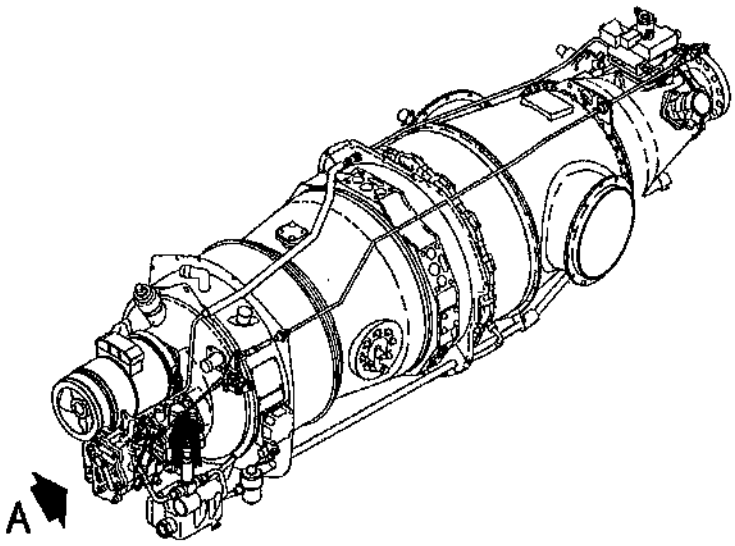
c. Connect the electrical connector (1) to the fuel flowmeter transmitter (4) and safety wire.

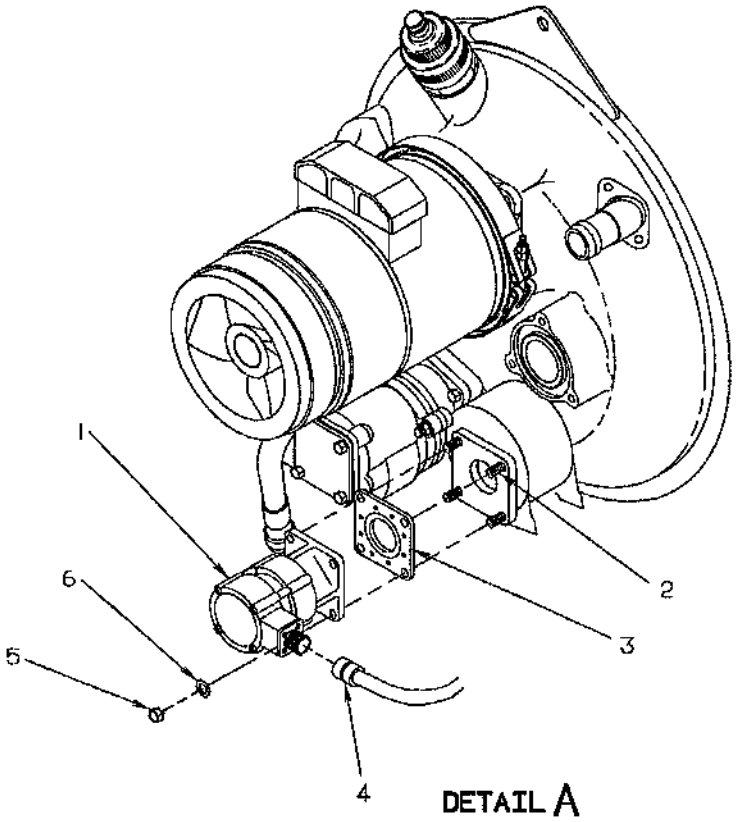
d. Operate the engine and verify that there is fuel flow displayed on the Engine Instrument, Crew Alerting System (EICAS) display.

e. Check the fuel line connections at the fuel flowmeter transmitter for fuel leakage.

f. Refer to Chapter 71-10-00 and install the required engine cowlings.

1. N_1 TACHOMETER GENERATOR
2. STUDS
3. GASKET
4. ELECTRICAL CONNECTOR
5. NUT
6. WASHER





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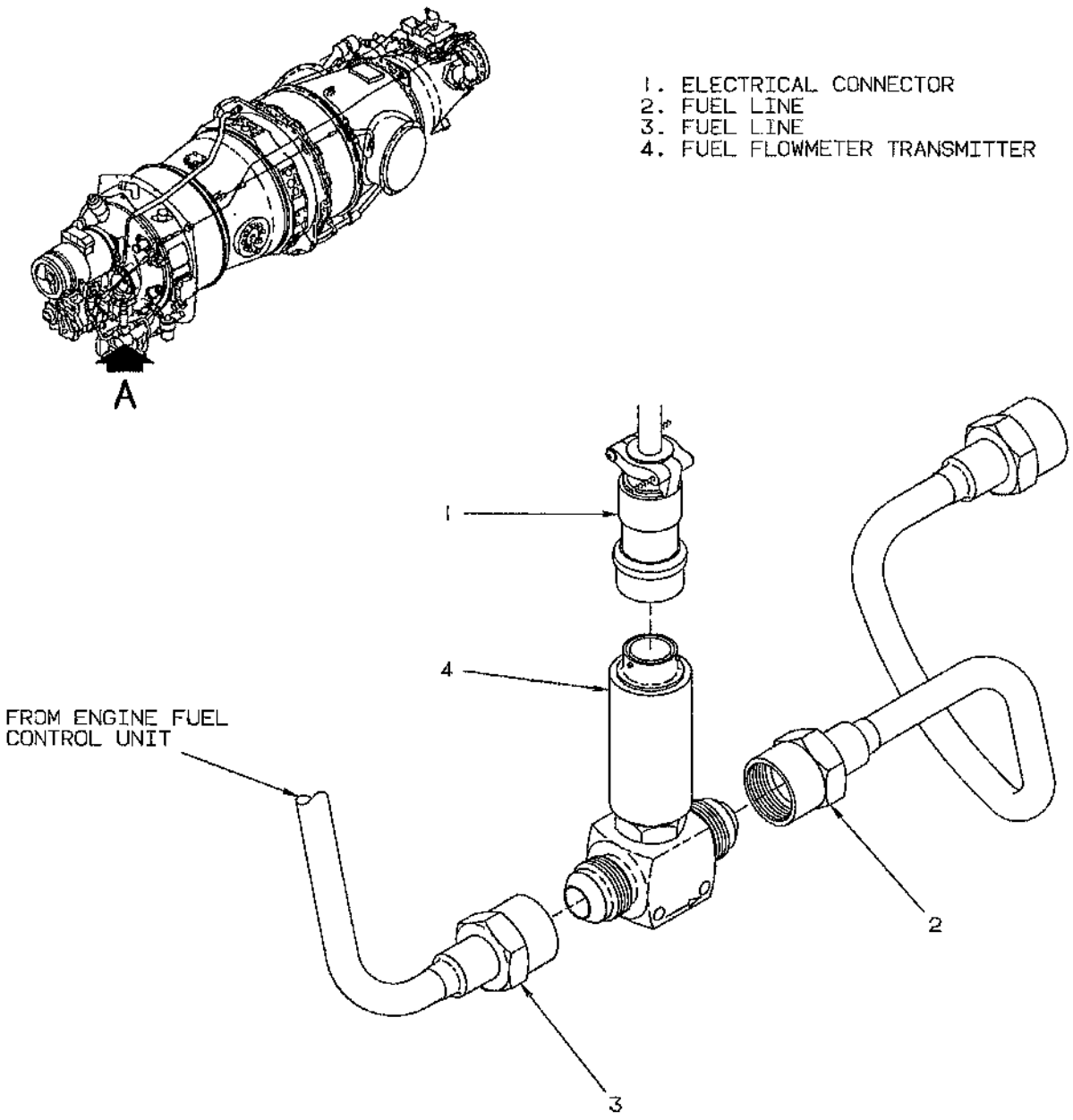
Engine N_1 Tachhometer Generator Installation (Effectivity: All)
Figure 6



Fuel Flowmeter Transmitter Installation (Effectivity: All)
Figure 7

OIL PRESSURE TRANSDUCER REMOVAL (Effectivity: All) (Figure 8)

a. Remove electrical power from the airplane.

b. Refer to Chapter 71-10-00 and remove the required engine cowlings.

c. Remove the safety wire and disconnect the electrical connector (5) from the transducer (4).

d. Remove the safety wire and remove the transducer (4) from the reducer bushing (2).

e. If required, remove the safety wire, reducer bushing (2) and preformed packing (1) from the accessory gearbox.

OIL PRESSURE TRANSDUCER INSTALLATION (Effectivity: All) (Figure 8)

a. If required, install the reducer bushing (2) as follows:

1. Install a new preformed packing (1) on the reducer bushing (2).

2. Apply a thin coat of lubricant (2, Chart 3, 77-00-00) to the external threads of the reducer bushing (2).

3. Install the reducer bushing (2) into the accessory gearbox housing. Torque the reducer bushing (2) to 40 inch-pounds and safety wire.

b. Install a new preformed packing (3) on the transducer (4).

c. Apply a thin coat of lubricant (2, Chart 3, 77-00-00) to the external threads of the transducer (4).

d. Install the transducer (4) into the reducer bushing (2) and torque the transducer to 40 inch-pounds and safety wire.

e. Connect the electrical connector (5) to the transducer (4), and safety wire.

f. Motor the engine with the starter long enough to obtain an oil pressure reading. Shut the engine down and check for oil leaks at the transducer.

CAUTION

Observe the starter operating limits of 30 seconds ON, 5 minutes OFF, 30 Seconds ON, 5 minutes OFF, 30 seconds ON, then 30 minutes OFF.

g. Refer to Chapter 71-10-00 and install the required engine cowlings.

OIL TEMPERATURE BULB REMOVAL (Effectivity: All) (Figure 8)

a. Remove electrical power from the airplane.

b. Refer to Chapter 71-10-00 and remove the required engine cowlings.

c. Remove the safety wire and disconnect the electrical connector (9) from the temperature bulb (8).

d. Remove the safety wire, temperature bulb (8), and gasket (7) from the tee (6).

OIL TEMPERATURE BULB INSTALLATION (Effectivity: All) (Figure 8)

a. Install a new gasket (7) on the temperature bulb (8).

b. Install the temperature bulb (8) into the tee (6) and safety wire.

c. Connect the electrical connector (9) to the temperature bulb (8) and safety wire.

d. Motor the engine with the starter long enough to obtain an oil pressure reading. Shut the engine down and check for oil leaks at the temperature bulb.

CAUTION

Observe the starter operating limits of 30 seconds ON, 5 minutes OFF, 30 Seconds ON, 5 minutes OFF, 30 seconds ON, then 30 minutes OFF.

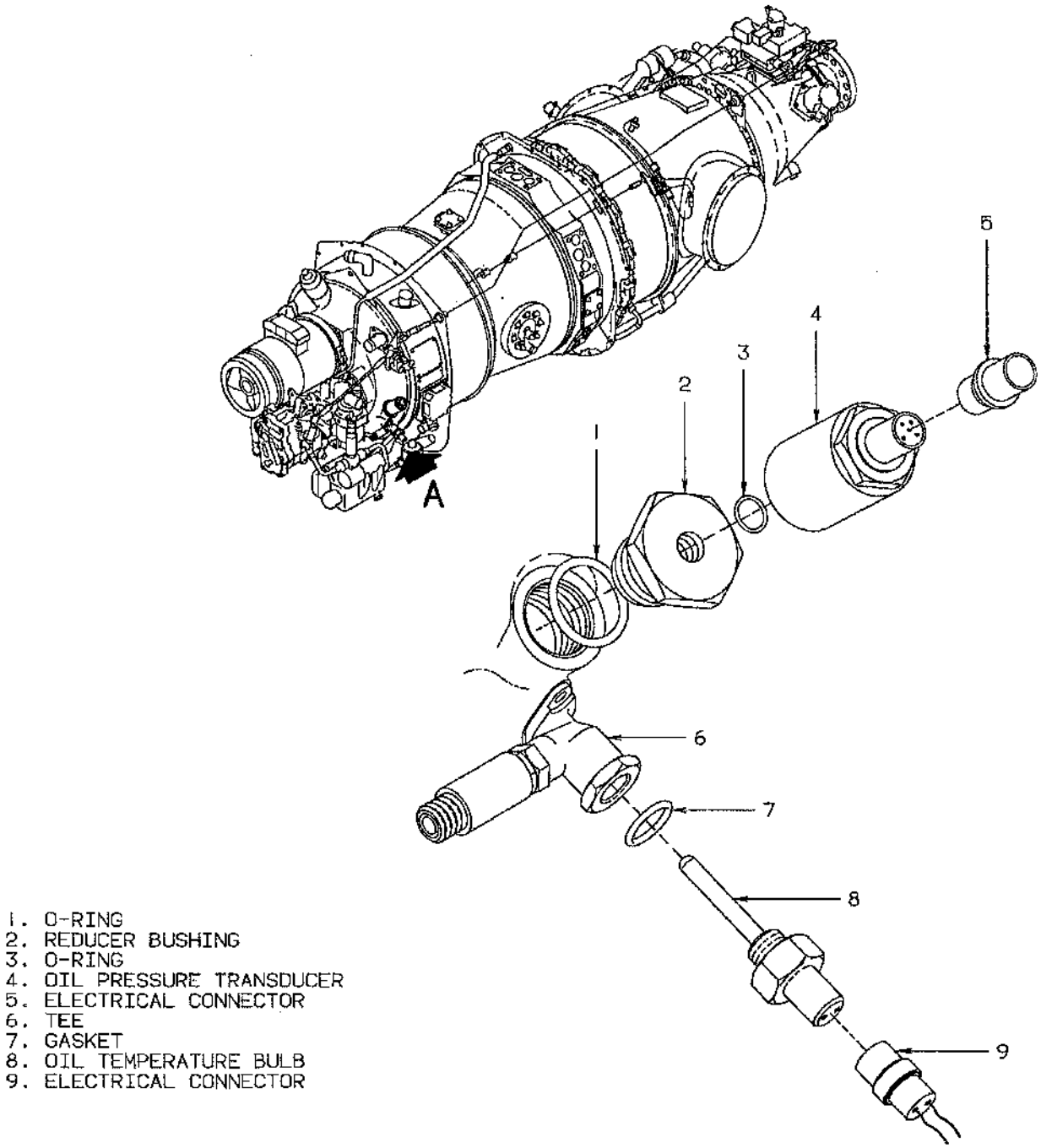
e. Refer to Chapter 71-10-00 and install the required engine cowlings.

OIL PRESSURE TRANSDUCER FUNCTIONAL TEST (Effectivity: All) (Figure 9)

a. The following items are required to perform this test:

1. Regulated supply of dry, filtered, compressed nitrogen at no less than 200 psi.

2. Pressure gage.



1. O-RING
2. REDUCER BUSHING
3. O-RING
4. OIL PRESSURE TRANSDUCER
5. ELECTRICAL CONNECTOR
6. TEE
7. GASKET
8. OIL TEMPERATURE BULB
9. ELECTRICAL CONNECTOR

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Oil Pressure Transducer and Temperature Bulb Installation (Effectivity: All)
Figure 8

3. Plumbing necessary to connect transducer to pressure supply.

4. Digital voltmeter calibrated in .01-volt increments.

5. 28 vdc power supply.

b. Remove the transducer as instructed under OIL PRESSURE TRANSDUCER REMOVAL in this chapter.

c. Connect the nitrogen pressure source and pressure gage to the transducer.

d. Connect the voltmeter, resistor and power supply to the transducer as shown in Figure 9.

e. Apply pressure to the transducer at a slow steady rate until 200 psi is indicated on the pressure gage. As the pressure rises, record the output voltage for each input pressure interval listed in Chart 1. The output voltage must fall within the range shown in Chart 1 for each specific interval.

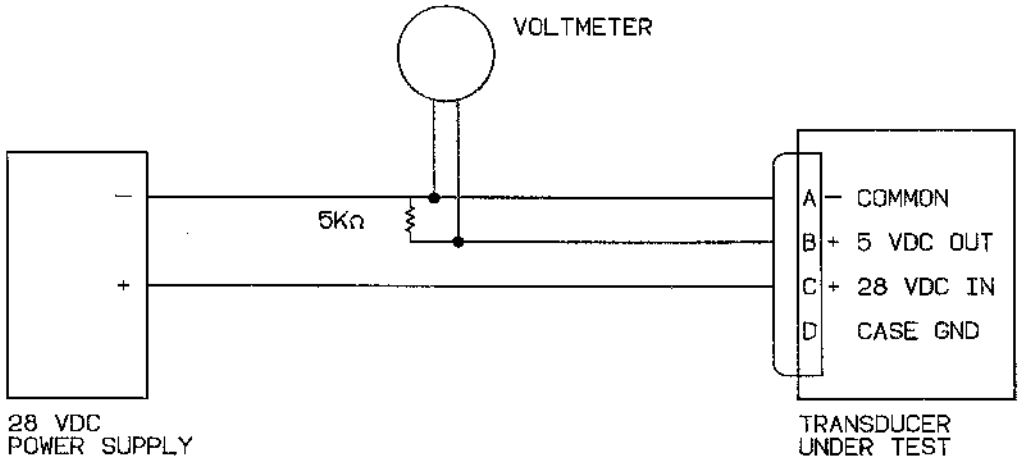
f. Slowly reduce the pressure, as the pressure lowers record the output voltage for each input pressure setting listed in Chart 1. The output voltage must fall within the range shown in the chart for each specific setting.

g. Disconnect all of the test equipment from the transducer.

CAUTION

Bleed the pressure down to 0 psi before disconnecting the transducer from the pressure source.

h. Install the transducer in the airplane as instructed in OIL PRESSURE TRANSDUCER INSTALLATION in this chapter. Replace any transducer that fails to meet the requirements of this test procedure.



Oil Pressure Transducer Functional Test Schematic (Effectivity: All)
Figure 9

CHART 1
OIL PRESSURE TRANSDUCER OUTPUT VOLTAGES (Effectivity: All)

Input Pressure In PSI	Output Voltage In VDC
0	-.10 to .10
25	.53 to .73
50	1.15 to 1.35
75	1.78 to 1.98
100	2.40 to 2.60
125	3.03 to 3.23
150	3.65 to 3.85
175	4.28 to 4.48
220	4.90 to 5.10

ENGINE DATA SIMULATION TEST (Effectivity: All)

This test should be performed when a fault is suspected in the engine indicating system. The engine data simulator (1, Chart 2, 77-00-00) contains a set of known values used to test the engine indicating system. Use the following procedures to check the system.

a. Obtain the engine data test simulator (1, Chart 2, 77-00-00).

b. Connect power to the engine data test simulator.

c. Refer to Chapter 57-00-00 and remove the applicable center section access door (left 551AB or right 651AB) to gain access to the applicable EDC.

d. Connect the engine data test simulator interface cable to the input connector on the EDC for the side of the system being tested.

e. Apply 28 vdc external power and place the battery switch in the ON position to power the avionics.

f. Turn on the EICAS display with the EICAS integrated avionics switch on the center subpanel.

g. Turn on the engine data test simulator and apply the engine data signals to the EDC.

h. Observe the EICAS engine displays for the proper engine parameters. If the engine parameters are within limits, suspect that the fault is occurring in the indicating system before the signal reaches the EDC.

i. Perform the check for the opposite side of the engine indicating system as required.

j. Turn off the EICAS display and the battery switch and remove 28 vdc external power.

k. Turn off the engine data simulator and disconnect its power source.

l. Disconnect the engine data simulator interface cable from the EDC and reconnect the indicating system connector.

m. Refer to Chapter 57-00-00 and install the applicable center section access door (left 551AB or right 651AB).

T_5 TERMINAL BLOCK THERMOCOUPLE RESISTANCE CHECK (Effectivity: All)

If a problem exists with the ITT portion of the engine indicating system, refer to the latest revision of the PT6A-67A Engine Maintenance Manual for detailed information used in performing a resistance check of the T_5 terminal block block thermocouple wire harness.

EDC TORQUE CALIBRATION (Effectivity: All)

This calibration procedure does not require starting the engines. Engine oil pressure is measured by an oil pressure transducer mounted on the torque pressure manifold. An electrical signal is sent to the EDC from the transducer indicating the engine oil pressure. The EDC converts this information into a percentage of maximum engine torque. One psi of oil pressure is equal to approximately 83.63 foot-pounds of torque. For example, 44.34 psi of oil pressure from the torque pressure manifold equates to 3,708 foot-pounds of torque, which converts to 100 percent of maximum allowable engine torque. The deadweight tester is capable of applying a precise amount of oil pressure and therefore may be used as a standard to calibrate the EDC's. Use the following procedures and Chart 2 to calibrate the torque indicating circuit in the EDC's.

a. Obtain the deadweight tester (2, Chart 2, 77-00-00).

b. Refer to Chapter 71-10-00 and remove the required engine cowlings.

c. Disconnect the torque pressure manifold oil supply line from the engine and connect it to the pressure port on the deadweight tester.

d. Apply 28 vdc external power and place the battery switch in the ON position to power the avionics.

e. Turn on the EICAS display with the EICAS integrated avionics switch on the center subpanel and observe that the engine torque scales are displayed.

f. Apply approximately 5 psi of oil pressure to the torque pressure manifold and monitor the applicable EICAS engine torque display. The indicated torque should correlate with the 5 psi applied to the torque pressure manifold.

g. Increase the oil pressure in 5 psi increments while observing the scale on the EICAS engine torque display. Record the indicated torque each time the oil pressure is increased to the next increment.

NOTE

The EDC has two separate torque adjustments. The SLOPE adjustment will change the torque curve throughout the entire torque range. The OFFSET adjustment will change the actual torque values across the entire torque range. Use the information recorded during the previous steps to determine which adjustments need to be made, if any.

h. Adjust the SLOPE and/or OFFSET screw(s) on the face of the EDC as required until the actual torque value and torque curve are within acceptable limits. Torque system accuracy shall be within ±1.0 percent over the entire torque range and +0.5/-0.0 at 100 percent of engine torque.

i. Ensure that the torque adjustment screws retain enough locking pressure to retain their settings.

j. Perform the torque calibration procedure for the opposite EDC if necessary.

k. Turn off the EICAS display and the battery switch and remove 28 vdc external power.

l. Disconnect the torque pressure manifold oil supply line from the deadweight tester and reconnect it to the engine.

m. Refer to Chapter 71-10-00 and install the required engine cowlings.

CHART 2
EDC TORQUE CALIBRATION
(Effectivity: All)

PSI	FT-LBS Torque	% Torque
1	83.63	2.255
5	418.15	11.3
10	836.30	22.6
15	1,254.45	33.8
20	1,672.60	45.1
25	2,090.75	56.4
30	2,508.90	67.7
35	2,927.05	78.9
40	3,345.2	90.2
44.34	3,708.0	100.0

CHAPTER 78 - EXHAUST

LIST OF PAGE EFFECTIVITY

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78-Contents	1	Feb 25/94
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	2	Feb 25/94

CHAPTER 78 - EXHAUST

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Exhaust Stack Installation (Effectivity: All) 1

GENERAL - DESCRIPTION AND OPERATION (Effectivity: All)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

L-shaped exhaust stacks are mounted directly to the engine exhaust ducts for channeling hot engine exhaust gases past the cowlings and away from the engine. An overboard breather tube is plumbed from each engine oil tank to the respective RH exhaust stack. Refer to Chapter 79-00-00 for more information on the oil breather tube.

GENERAL - MAINTENANCE PRACTICES (Effectivity: All)

EXHAUST STACK REMOVAL (Effectivity: All) (Figure 1)

a. Remove all electrical power from the airplane.

b. Remove the upper and lower sections of the aft engine cowling from the respective engine as outlined in Chapter 71-10-00.

c. Refer to Chapter 79-00-00 and remove the engine oil breather tube from the RH exhaust stack as necessary.

NOTE

Before removing any clamps that secure the fire detection sensor cable around the exhaust stack, note all clamp locations to assist in proper placement of the sensor cable during exhaust stack installation. Refer to Chapter 26-10-00 for a detailed illustration of fire detection sensor cable routing.

d. Remove the bolts (1), washers (2) and nuts (3) securing the exhaust stack to the engine exhaust duct, then remove the exhaust stack. Note the location of the slotted hole on the exhaust stack to aid in reinstallation.

EXHAUST STACK INSTALLATION (Effectivity: All) (Figure 1)

a. Align the exhaust stack with the engine exhaust duct and ensure that the slotted hole on the exhaust stack is forward.

b. Secure the exhaust stack by installating the bolts (1), washers (2) and nuts (3).

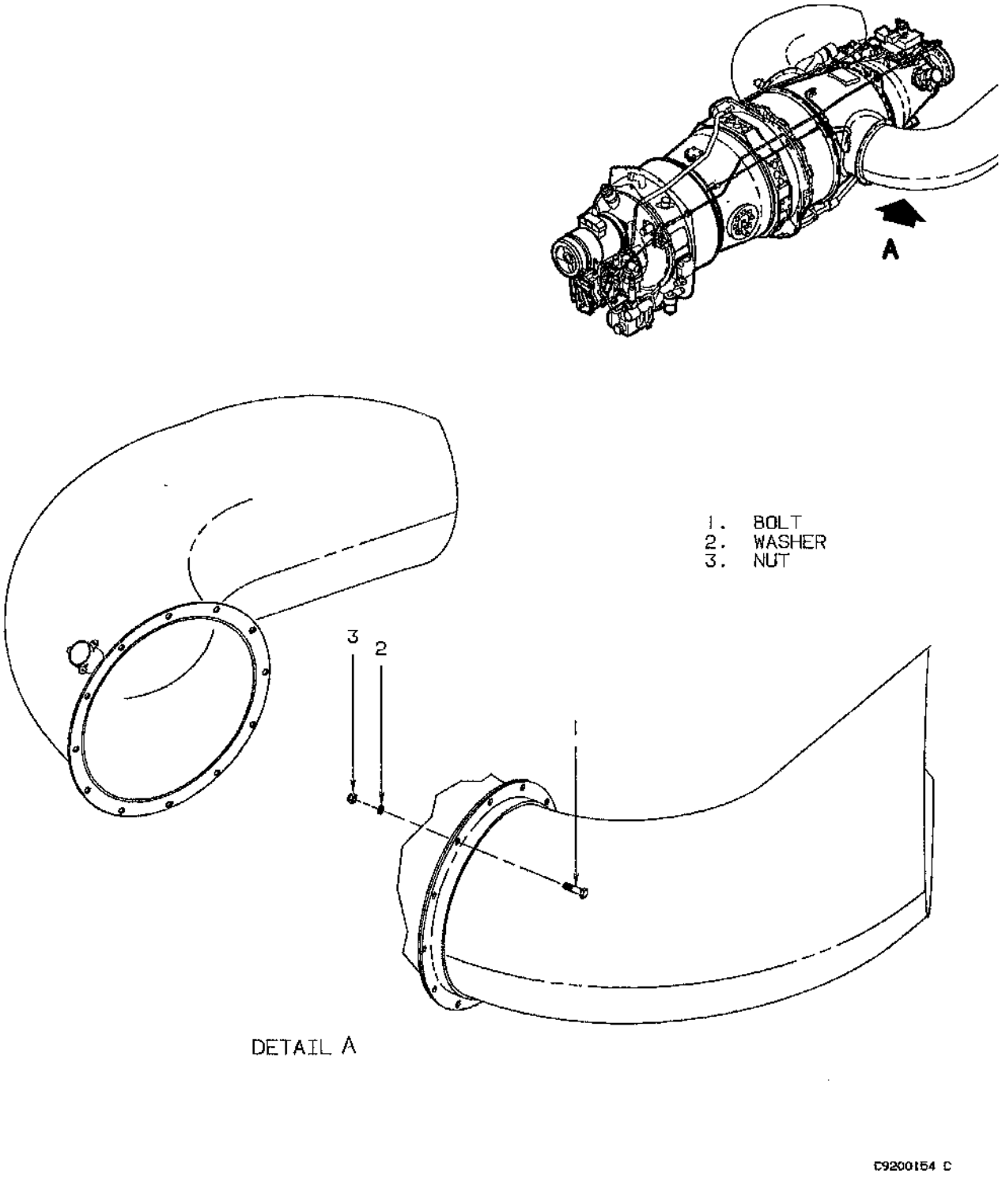
NOTE

Verify proper positioning of the sensor cable clamps and ensure that the fire detection sensor cable is properly positioned in the clamp liners. Refer to Chapter 26-10-00 for fire detection sensor cable installation procedures.

c. Refer to Chapter 79-00-00 and reinstall the engine oil breather tube on the RH exhaust stack if it was removed.

d. Reinstall the upper and lower sections of the aft engine cowling as outlined in Chapter 71-10-00.

e. Restore electrical power to the airplane.



Exhaust Stack Installation (Effectivity: All)
Figure 1

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CHAPTER 79 - OIL

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GENERAL - DESCRIPTION AND OPERATION

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The engine oil system is a closed system consisting basically of a pressure system, a scavenge system and a breather system, all of which are supplied as part of the engine by the engine manufacturer. Added to the closed engine system is an oil cooling system, an overboard breather line and an indicating system. Refer to Chapter 79-30-00 for information on the indicating system. All of the oil system components are mounted within or on the engine with the exception of the oil cooler which is mounted below the engine.

NOTE

Refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual for detailed information relating to the system components supplied by the engine manufacturer.

The oil supply for the engine is contained in an oil tank which is an integral part of the compressor inlet case for the engine. Oil is added to the tank through a filler neck located at the 11 o'clock position on the accessory gearbox housing and accessed through a hinged access door on the top surface of the upper forward engine cowling. A cap and electric dipstick extends down through the filler neck into the oil tank. The dipstick is marked in US quarts to indicate the amount of oil required to fill the engine oil tank to the full mark. The tank has a capacity of 2.5 gallons (10 quarts) of which 1.5 gallons (6 quarts) are usable, with a total system capacity of 16.3 quarts. The oil system is serviced with the oil type specified in the latest revision of Pratt & Whitney Service Bulletin No. 14001. The tank is vented through a breather boss on the accessory gearbox housing; a breather tube from this boss is routed aft along the engine to a boss on the RH exhaust stack.

The oil is picked up from the oil tank and routed through the pressure side of the oil system by an internal pressure pump mounted at the bottom of the oil tank. A pressure regulating valve and a cold pressure relief valve are mounted at the top of the pump housing. A filter assembly, consisting of a filter element and a filter bypass valve downstream of the oil pump, filters the oil before it is delivered to the engine components. The filter assembly is accessible through a cover at the 3 o'clock position on the compressor inlet case.

Oil is returned to the oil tank by two double-element scavenge pumps, one mounted within the accessory gearbox and another mounted externally on the accessory gearbox. The rear element of the external scavenge pump forces the oil from the reduction gearbox to the oil cooler inlet port. The oil cooler incorporates a thermostatic control valve which will allow some oil to bypass the cooler core when the oil temperature is below 160° F. At 160° F the valve will direct all of the oil flow through the cooler core; oil from the cooler is then returned to the oil tank through a hose from the oil cooler outlet port to a fitting at the 12 o'clock position on the compressor inlet case. The rear element of the internal pump scavenges oil from the accessory gearbox sump and forces it through the oil-to-fuel heater. Oil from the heater is returned to the oil tank by one of two paths determined by the temperature of the oil as it leaves the heater. A thermostatic bypass/check and diverter valve positioned between the heater and oil tank will prevent oil that is too hot from returning directly to the oil tank without first flowing to the oil cooler. This valve will divert the hot oil to join the oil flowing to the oil cooler from the external scavenge pump.

The oil system can be drained at the following locations: the reduction gearbox, the oil tank, the accessory gearbox and the oil cooler. The oil tank is drained through a drain tube and shutoff valve mounted at the 6 o'clock position on the compressor inlet case. To operate the drain remove the cap and open the shutoff valve.

CAUTION

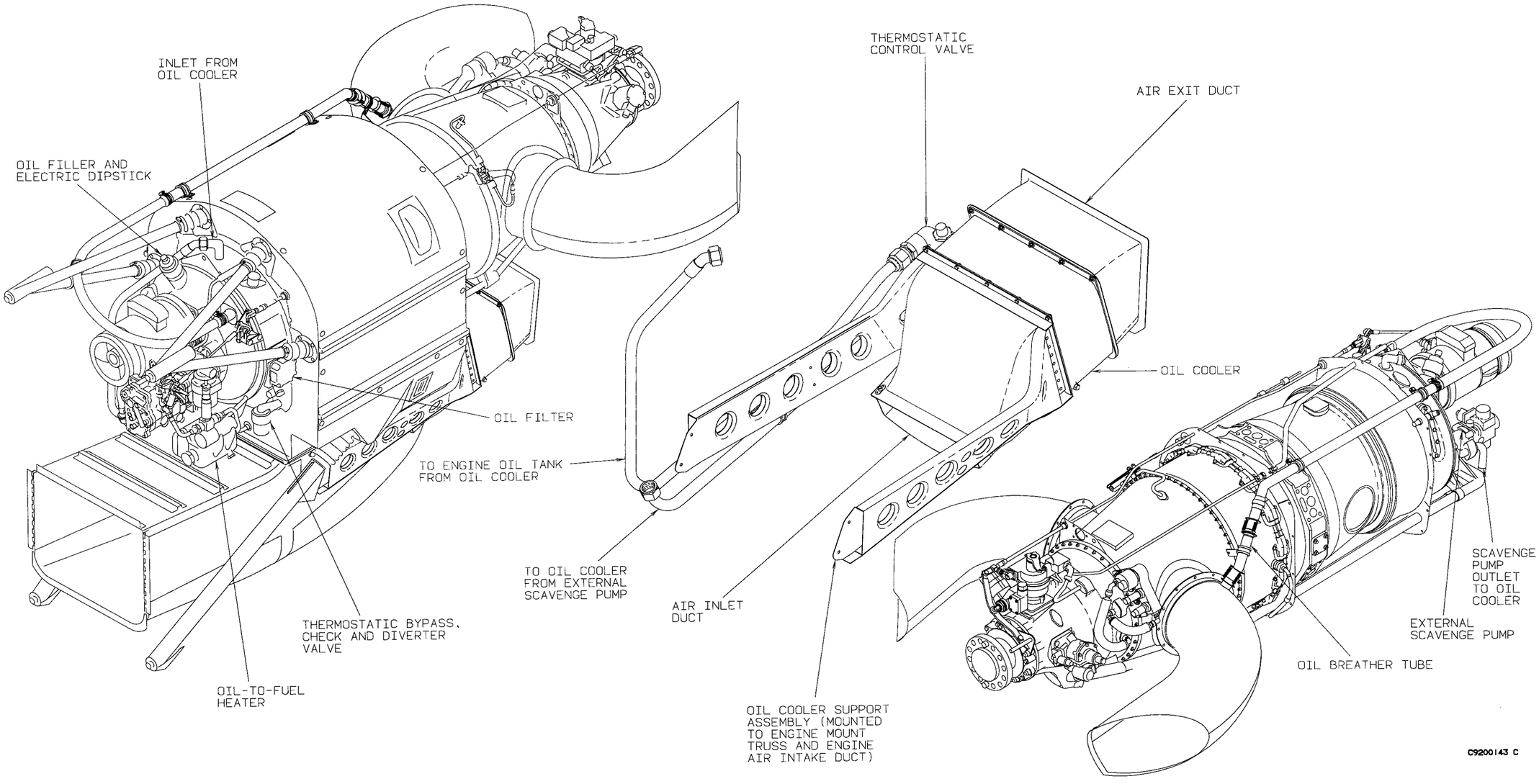
The shutoff valve is safety wired in the closed position; after the draining operation is complete, always make sure that the valve is in the closed position and safety wired.

RECOMMENDED MATERIALS

The recommended materials listed in Chart 1 as meeting federal, military or supplier specifications are provided for reference only and are not specifically recommended by Raytheon Aircraft Company. Any product conforming to the product listed may be used. The products included in this chart have been tested and approved for aviation usage by Raytheon Aircraft Company, by the supplier or by compliance with the applicable specification. Generic or locally manufactured products that conform to the requirements of the specification listed may be used even though not included herein. Only the basic number of each product specification is listed. No attempt has been made to update the listing to the latest revision. It is the responsibility of the technician or mechanic to determine the current revision of the applicable specification prior to usage of the product listed. This can be done by contacting the supplier of the product to be used.

CHART 1
RECOMMENDED MATERIALS

MATERIAL	SPECIFICATION	PRODUCT	SUPPLIER
Lubricant	MIL-G-6032		



Engine Oil System
Figure 1

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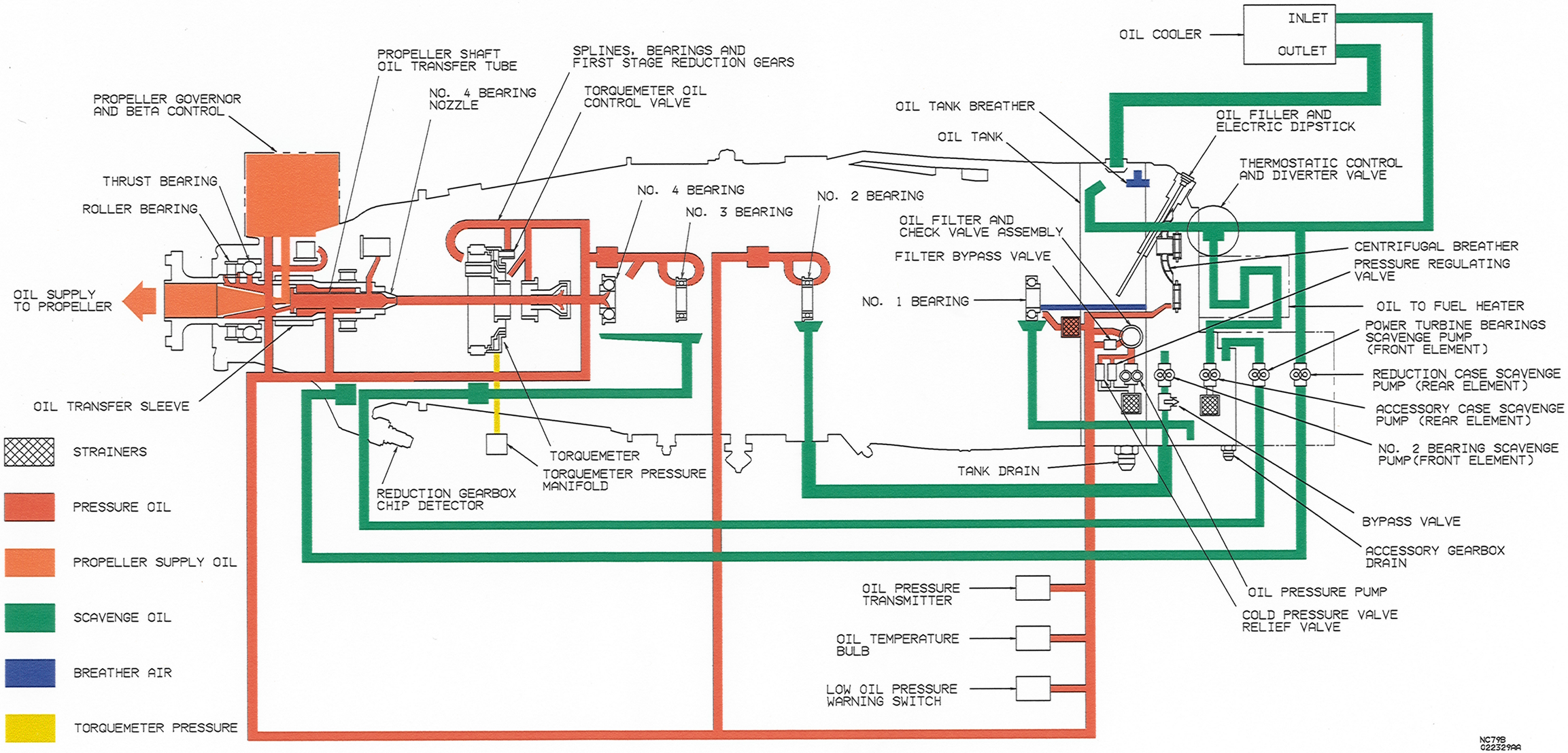


Engine Oil System Schematic
Figure 2

GENERAL - MAINTENANCE PRACTICES

OIL SYSTEM SERVICING

Servicing the engine oil system primarily involves maintaining the engine oil at the proper level, inspecting and cleaning the engine oil filter and changing the engine oil when the oil becomes contaminated. Refer to Chapter 12 for the oil system filling procedure. The oil filter should be cleaned and inspected at each engine oil change and every 100 hours, then replaced every 1000 hours. Refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual for the proper oil filter inspection and replacement procedures.

ENGINE OIL CHANGING

CAUTION

When changing to a different brand of oil, completely drain the engine oil system as indicated in the procedure below. Remove the oil filter and immerse it in the new brand of oil to be used. Install the oil filter and drain plugs. Fill the system to the proper level, and ground-run the engines for 20 minutes to thoroughly circulate the new brand of oil throughout the system. Completely drain the engine oil system and again remove the oil filter and immerse it in the new brand of oil. Fill the engine oil system as indicated below. This procedure will thoroughly purge the system of the old oil to prevent chemical interaction between it and the new brand of oil.

a. Remove the engine cowling as outlined in Chapter 71-10-00.

b. Provide suitable containers, funnels, and drain hoses to facilitate oil drainage and to prevent unnecessary oil spillage.

c. Remove the dust cap (10) from the oil tank drain shutoff valve (9) at the 6 o'clock position of the compressor inlet case. Remove the safety wire and open the shutoff valve to drain the engine oil tank.

d. Remove the drain plug (1) located at the 6 o'clock position on the rear face of the accessory gearbox housing.

e. Remove the drain plug (5) from the oil cooler.

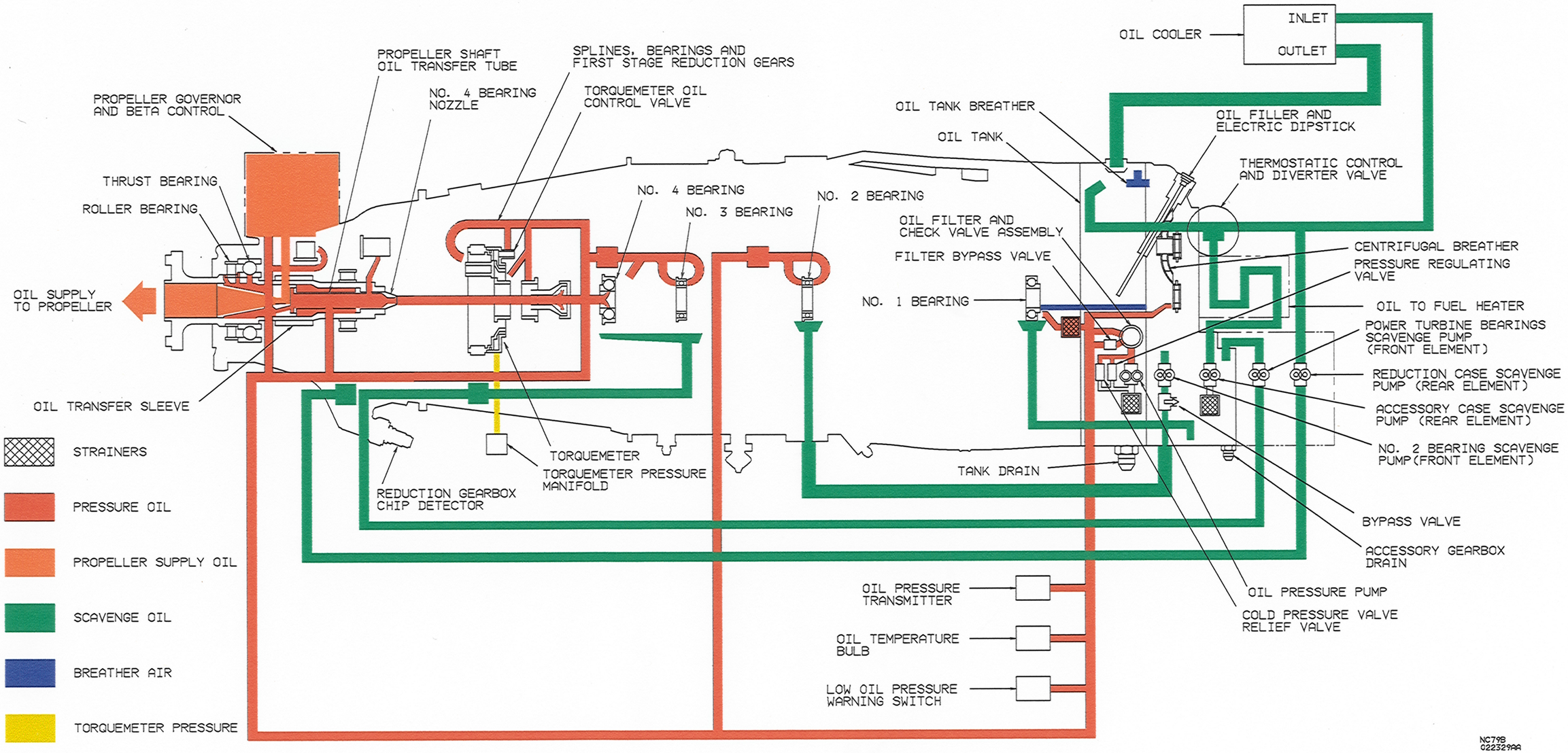
f. Disconnect the hoses from the inlet and outlet ports of the oil cooler.

g. Remove the oil filter element as detailed in the Pratt & Whitney PT6A-67A Engine Maintenance Manual.

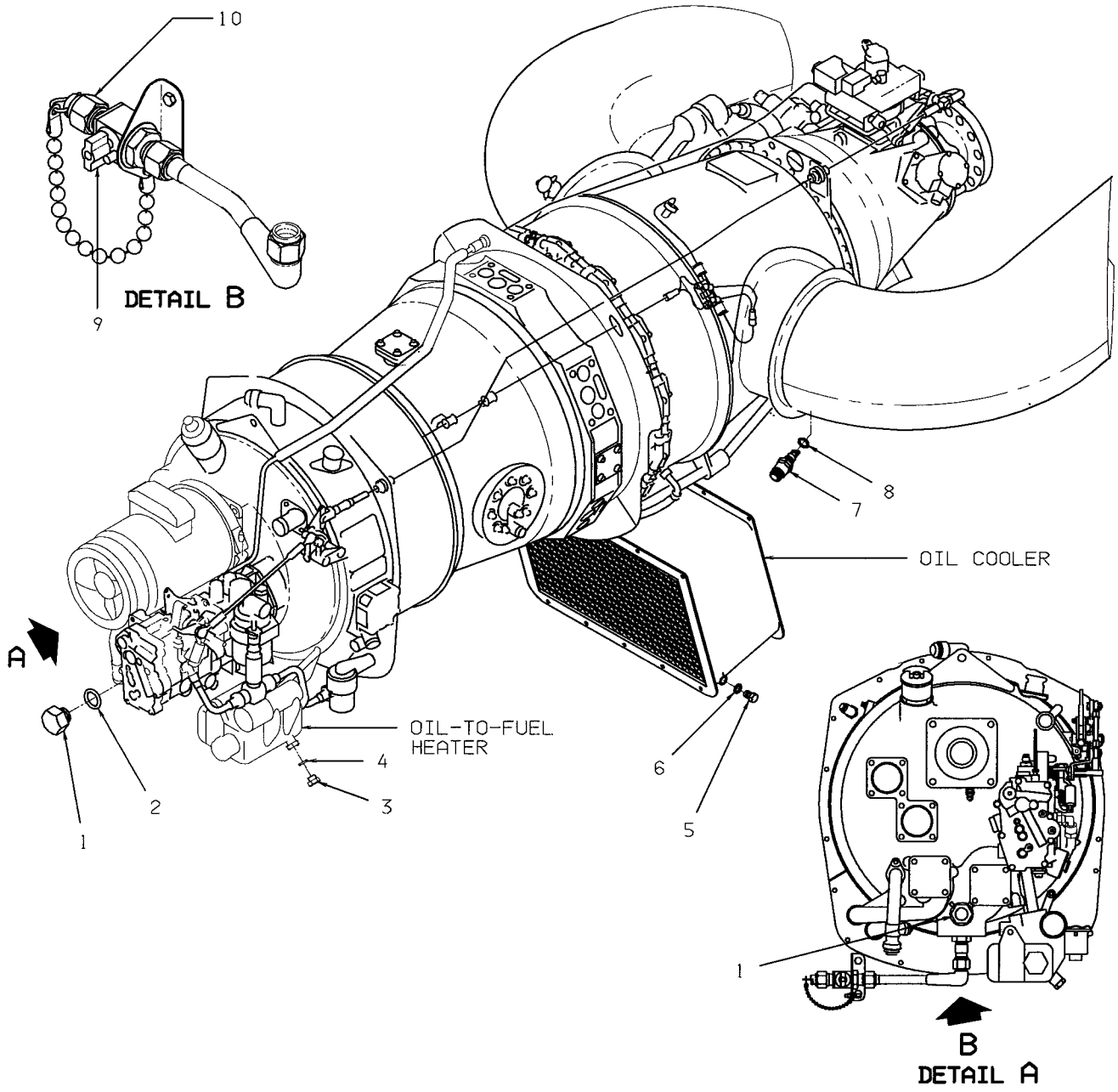
h. Remove the chip detector (7) at the 6 o'clock position on the reduction gearbox front case.

i. Remove the drain plug (3) from the oil-to-fuel heater.

j. With all the drain plugs removed, motor the engine with the starter only (no ignition) to permit the scavenge pumps to clear the engine of oil.



Engine Oil System Schematic
Figure 2



1. ACCESSORY GEARBOX DRAIN PLUG
2. O-RING
3. OIL-TO-FUEL HEATER DRAIN PLUG
4. O-RING
5. OIL COOLER DRAIN PLUG
6. O-RING
7. CHIP DETECTOR
8. O-RING
9. OIL TANK DRAIN SHUTOFF VALVE
10. DUST CAP

C9200364 C

Engine Oil System Drains
Figure 3

CAUTION

Limit motoring to the time required to accomplish the above due to the limited lubrication available to the engine during this operation. To prevent damage to the fuel control unit, leave the condition lever in the FUEL CUT OFF position while motoring the engine.

Observe the starter operating limits of 30 seconds ON, 5 minutes OFF, 30 seconds ON, 5 minutes OFF, 30 seconds ON, then 30 minutes OFF.

k. Install the oil filter element as detailed in the Pratt & Whitney PT6A-67A Engine Maintenance Manual.

l. Install new O-rings on the drain plugs and the chip detector.

m. Install the chip detector (7) in the reduction gearbox case. Torque to 45-55 inch-pounds and safety wire.

n. Install the drain plug (1) on the rear face of the accessory gearbox housing. Torque to 215-240 inch-pounds and safety wire.

o. Install the drain plug (5) in the oil cooler and safety wire.

p. Connect the hoses to the inlet and outlet ports of the oil cooler.

q. Install the drain plug (3) in the oil-to-fuel heater and torque to 65-70 inch-pounds, then safety wire.

r. Place the shutoff valve (9) in the closed position and safety wire. Install the dust cap on the shutoff valve.

s. Fill the engine with the correct amount and type of oil as specified in latest revision of Pratt & Whitney Service Bulletin No. 14001.

NOTE

The oil tank capacity of each engine is 2.5 gallons (10 quarts), of which only 1.5 gallons (6 quarts) are usable. Additionally, approximately 6 quarts of oil are required to fill the lines and oil cooler, giving a total system capacity of 16.3 quarts. However, because of the residual oil trapped in the system, no more than 13 quarts should be added during an oil change.

t. Motor the engine over, with the starter only, long enough to get an oil pressure reading.

CAUTION

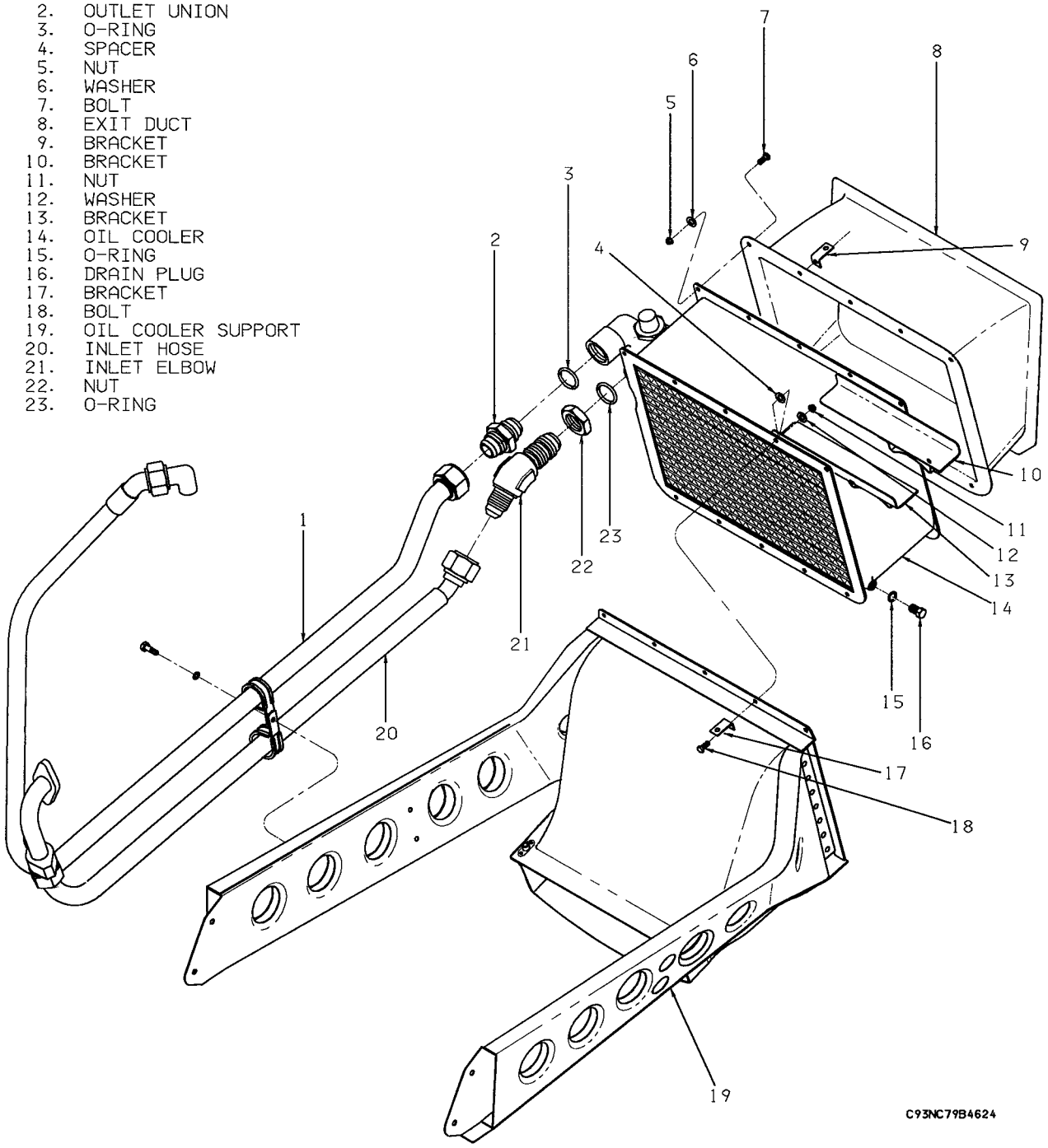
Observe the starter operating limits of 30 seconds ON, 5 minutes OFF, 30 seconds ON, 5 minutes OFF, 30 seconds ON, then 30 minutes OFF.

u. Check the engine for oil leaks.

v. Check the engine oil level and add oil as required to fill the tank to the proper level. Refer to the latest revision of Pratt & Whitney Service Bulletin No. 14001 for the correct type of oil.

w. Install the engine cowling as outlined in Chapter 71-10-00.

1. OUTLET HOSE
2. OUTLET UNION
3. O-RING
4. SPACER
5. NUT
6. WASHER
7. BOLT
8. EXIT DUCT
9. BRACKET
10. BRACKET
11. NUT
12. WASHER
13. BRACKET
14. OIL COOLER
15. O-RING
16. DRAIN PLUG
17. BRACKET
18. BOLT
19. OIL COOLER SUPPORT
20. INLET HOSE
21. INLET ELBOW
22. NUT
23. O-RING



Oil Cooler Installation
Figure 4

OIL COOLER REMOVAL

CAUTION

To prevent contamination of the oil system, cap or plug all openings in the system as the components are disconnected or removed.

a. Remove the engine cowling as detailed in Chapter 71-10-00.

b. Place a drip pan under the engine to catch any oil spillage.

c. Cut the safety wire and remove the oil cooler drain plug (16) and drain the oil into a suitable container.

d. Disconnect the oil inlet and outlet hoses (1 and 20) from the elbow (21) and union (2), and drain the oil into a suitable container.

NOTE

The fuel purge system tank is mounted on the oil cooler, refer to Chapter 73 for the removal procedure.

e. Remove the bolts (7), washers (6), nuts (5) and bracket (9) holding the exit duct (8) to the oil cooler.

NOTE

If the oil cooler is removed for replacement, remove the outlet union (2) and inlet elbow (21) from the oil cooler. Note the position of the inlet elbow (21) so it can be installed on the new oil cooler in the same position. Install the union and elbow in the new oil cooler with new O-rings (3 and 23).

OIL COOLER INSTALLATION

a. If the oil cooler being installed is the one that was removed, install a new O-ring (15) on the drain plug (16). Install the drain plug in the oil cooler.

NOTE

The fuel purge system tank is mounted to the oil cooler, refer to Chapter 73 for the installation procedure.

b. Position the oil cooler (14) on the mounting flange of the oil cooler support (19) and align the brackets (13 and 17), attach with the bolts (18), washers (12), spacers (4) and nuts (11).

c. Position the exit duct (8) and brackets (9 and 10) with the oil cooler and secure with the bolts (7), washers (6) and nuts (5).

d. Connect the inlet and outlet hoses (1 and 20) to the elbow (21) and the union (2).

e. Refer to Chapter 12-10-00 and check the oil level; add oil as necessary to replace any that may have been lost during this procedure.

f. Motor the engine over with the starter only (no ignition) long enough to obtain an oil pressure reading.

CAUTION

Observe the starter operating limits of 30 seconds ON, 5 minutes OFF, 30 seconds ON, 5 minutes OFF, 30 seconds ON, then 30 minutes OFF.

g. Check the area around the oil cooler inlet elbow, outlet union and the drain plug for leaks.

h. If no leaks are found, install the engine cowling as detailed in Chapter 71-10-00.

OIL COOLER SUPPORT REMOVAL

CAUTION

To prevent contamination of the oil system, cap or plug all openings in the system as the components are disconnected or removed.

a. Remove the engine cowling as detailed in Chapter 71-10-00.

b. Place a drip pan under the engine to catch any oil spillage.

c. Disconnect the oil inlet and outlet hoses from the engine and drain the oil into a suitable container.

d. Cut the safety wire and remove the oil cooler drain plug and drain the oil into a suitable container.

NOTE

The oil cooler and fuel purge system tank are mounted to the oil cooler support. The oil cooler support may be removed from the airplane with these components mounted to the support if desired. If the fuel purge tank is to be removed with the oil cooler support, refer to Chapter 73 and disconnect the hoses from the engine. The tank removal procedure can also be found in Chapter 73.

e. Support the oil cooler support from below.

f. Remove the screws (1) and washers (2) securing the forward end of the support to the engine truss mount and air plenum.

g. Remove the screws (3) and washers (4) securing the aft end of the support to the air plenum.

h. Lower the support away from the engine.

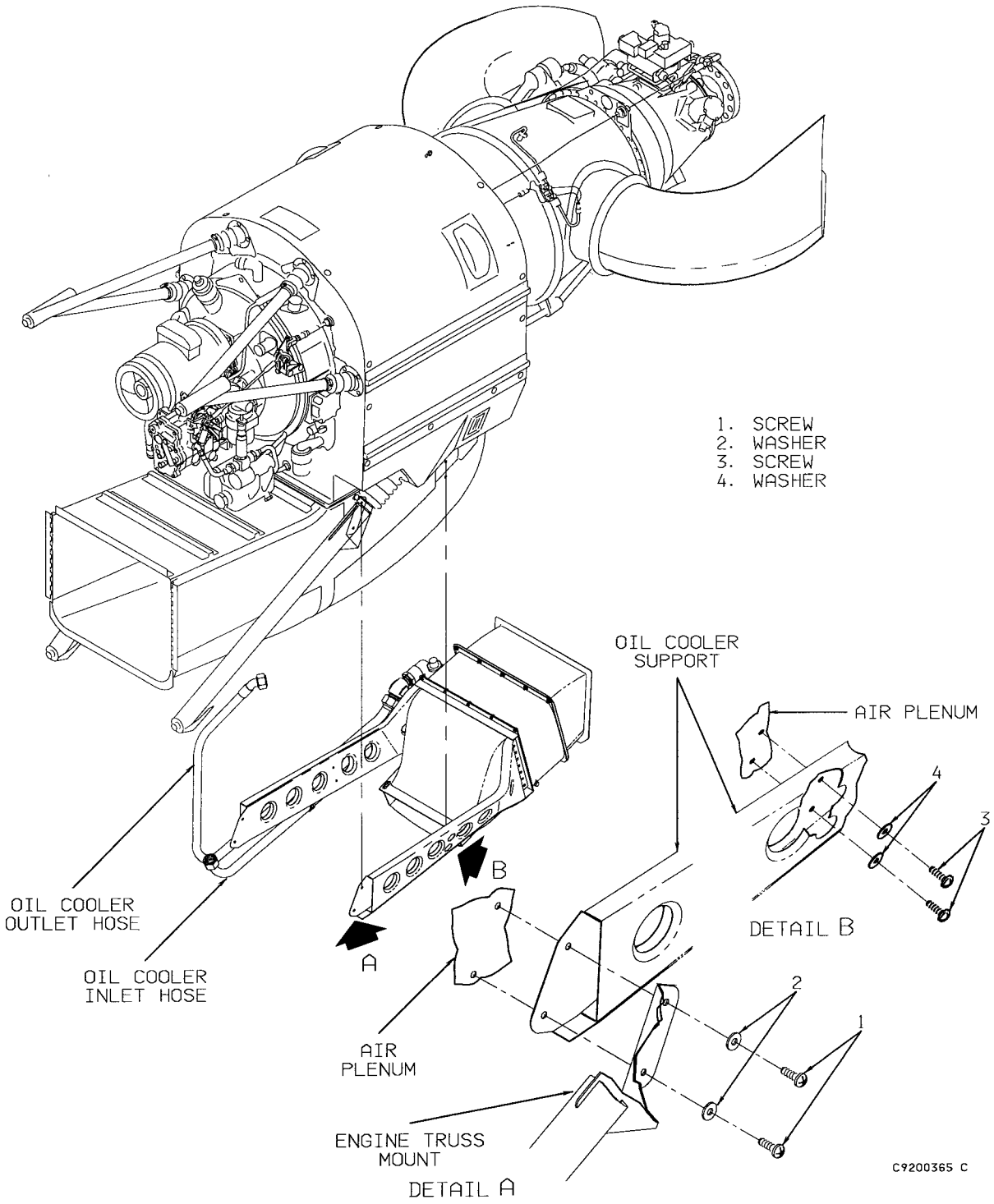
OIL COOLER SUPPORT INSTALLATION

a. Position the oil cooler support on the air intake plenum and engine truss mount and support from below.

b. Secure the forward end of the support to the engine truss mount and air plenum with screws (1) and washers (2).

c. Secure the aft end of the support to the air plenum with screws (3) and washers (4).

d. Install a new O-ring on the drain plug. Install the drain plug in the oil cooler and safety wire.



Oil Cooler Support Installation
Figure 5

e. Connect the oil cooler inlet and outlet hoses to the engine.

f. Refer to Chapter 73 and connect the fuel purge tank hoses to the engine. If necessary, refer to Chapter 73 and install the fuel purge tank.

g. Refer to Chapter 12, check the oil level and add oil as necessary to replace any that may have been lost during this procedure.

h. Motor the engine over with the starter only (no ignition) long enough to obtain an oil pressure reading.

CAUTION

Observe the starter operating limits of 30 seconds ON, 5 minutes OFF, 30 seconds ON, 5 minutes OFF, 30 seconds ON, then 30 minutes OFF.

i. Check the area around the oil cooler inlet and outlet elbows and the drain plug for leaks.

j. If no leaks are found, install the engine cowling as detailed in Chapter 71-10-00.

OIL BREATHER TUBE REMOVAL

a. Remove all electrical power from the airplane.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

c. Loosen the clamps (1 and 7) and remove hose (2) from tube (11) and the overboard breather port on the engine accessory gearbox housing.

d. Remove nut (3), washer (4), bolt (5) and clamps (6) that secure hose (2) to the engine truss mount.

e. Unclamp the clamshell assembly (12) from tubes (11 and 15). To unclamp the clamshell assembly all three latch pawls must be lifted simultaneously.

f. Slide sleeve (13) onto tube (15) far enough to expose the flange on tube (11).

g. Remove the bolts (8) and washers (9) attaching tube (11) to the air plenum. Remove tube (11) and clamps (10).

h. Remove the sleeve (13) from tube (15).

i. Remove the springs (16) that secure tube (15) and tube (23) together. Remove tube (15) from tube (23).

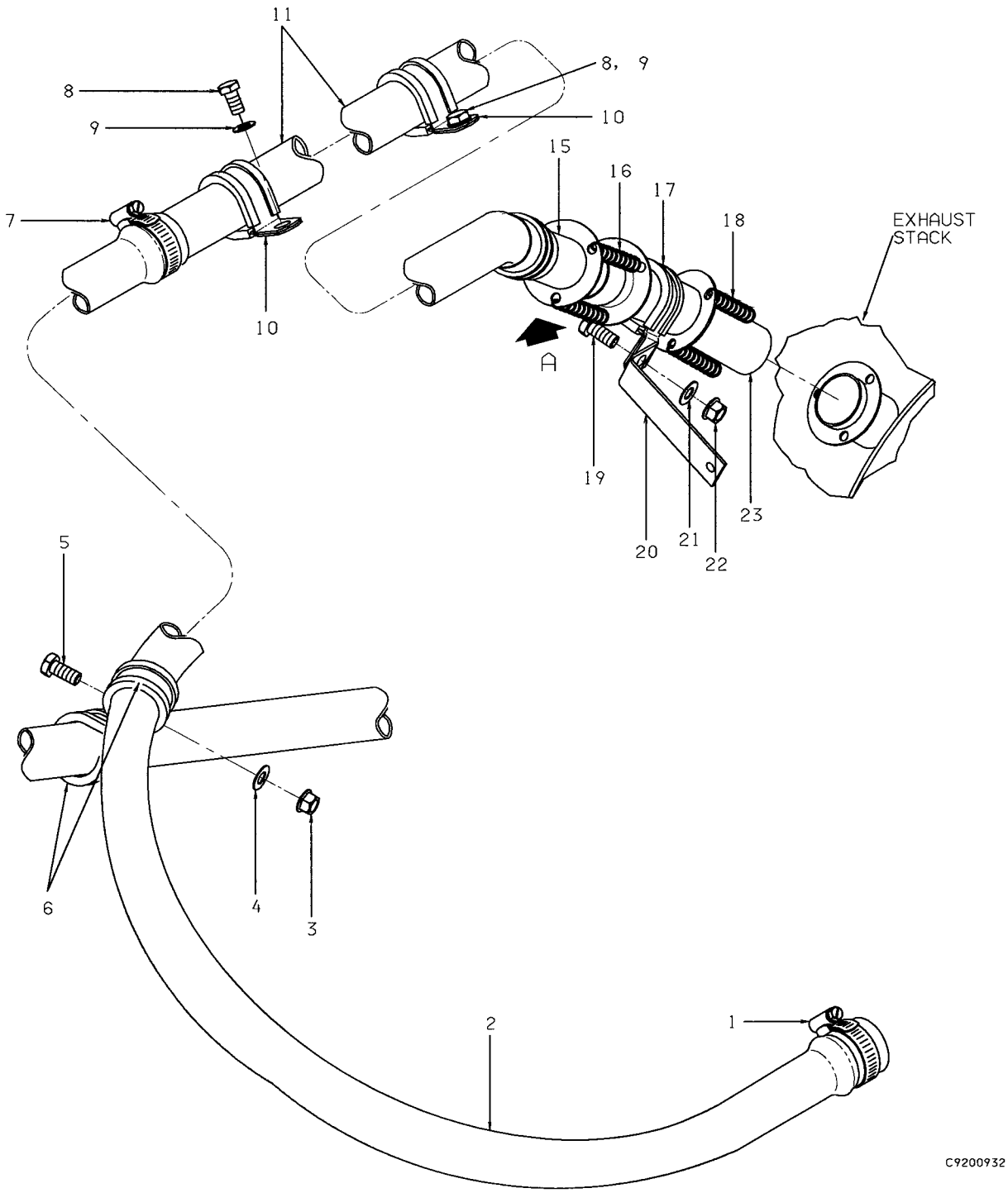
j. Remove the nut (22), washer (21) and bolt (19) securing clamp (17) to bracket (20).

k. Remove the springs (18) that secure tube (23) to the exhaust stack. Remove tube (23) from the exhaust stack.

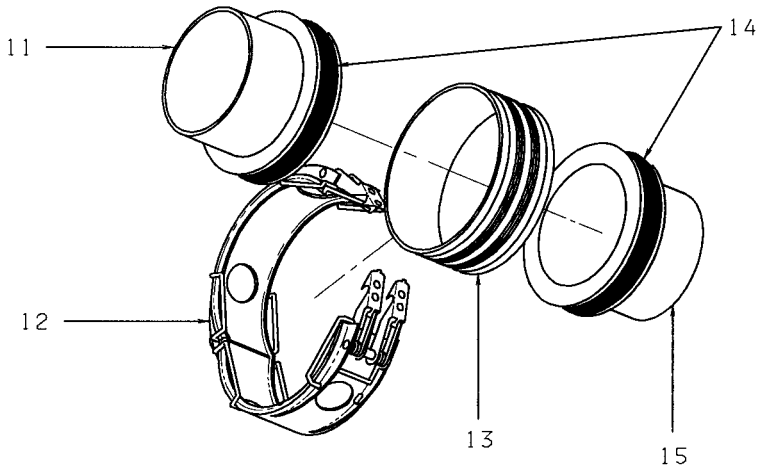
OIL BREATHER TUBE INSTALLATION

a. Install tube (23) into the exhaust stack. Align the spring attach points on the tube with those on the exhaust stack and install the springs (18) to secure.

b. Position clamp (17) to the bracket (20) and install the bolt (19), washer (21) and nut (22).



Oil Breather Tube Installation (Sheet 1 of 2)
Figure 6



DETAIL A

1. CLAMP
2. HOSE
3. NUT
4. WASHER
5. BOLT
6. CLAMPS
7. CLAMP
8. BOLTS
9. WASHER
10. CLAMP
11. TUBE
12. CLAMSHELL ASSEMBLY
13. SLEEVE
14. O-RING
15. TUBE
16. SPRINGS
17. CLAMP
18. SPRINGS
19. BOLT
20. BRACKET
21. WASHER
22. NUT
23. TUBE

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Oil Breather Tube Installation (Sheet 2 of 2)
Figure 6

c. Install a new O-ring (14) into the flange of tube (15). Lightly lubricate the O-ring with engine oil.

d. Install tube (15) into tube (23). Align the spring attach points on tube (15) with those on tube (23) and install springs (16) to secure.

e. Lightly lubricate the I.D. of sleeve (13) with engine oil. Slide the sleeve over the flange of tube (15).

f. Install a new O-ring (14) into the flange of tube (11). Lightly lubricate the O-ring with engine oil.

g. Install tube (11) onto the air plenum with clamps (10), bolts (8) and washers (9). Align tube (11) with tube (15) with a gap approximately 1/4-inch wide between the two tubes. Do not fully tighten bolts (8) at this time.

h. Slide the sleeve (13) from tube (15) onto the flange of tube (11). The sleeve must span both tube flanges completely.

i. Install the clamshell assembly (12) over the sleeve (13) and snap shut.

j. Tighten bolts (8) to secure tube (11) to the air plenum.

k. Install the engine cowling as instructed in Chapter 71-10-00.

l. Install hose (2) to the overboard breather port on the engine accessory gearbox housing and to tube (11) with clamps (1 and 7). Secure the hose (2) to the engine truss mount with clamps (6), bolt (5), washer (4) and nut (3).

m. Restore electrical power to the airplane.

OIL TANK DRAIN SHUTOFF VALVE REMOVAL

a. Remove the engine cowling as detailed in Chapter 71-10-00.

b. Place a drip pan under the engine to catch any oil spillage.

c. Remove the dust cap (1) from the shutoff valve (2). Remove the safety wire and open the shutoff valve to drain the engine oil tank.

d. Remove the drain tube (8) from the union (9) and the shutoff valve.

CAUTION

To prevent contamination of the oil system after the drain tube (8) is removed, cap the union (9).

e. Remove nut (7) from the shutoff valve (2) and remove the shutoff valve and washers (3) from the bracket (6).

OIL TANK DRAIN SHUTOFF VALVE INSTALLATION

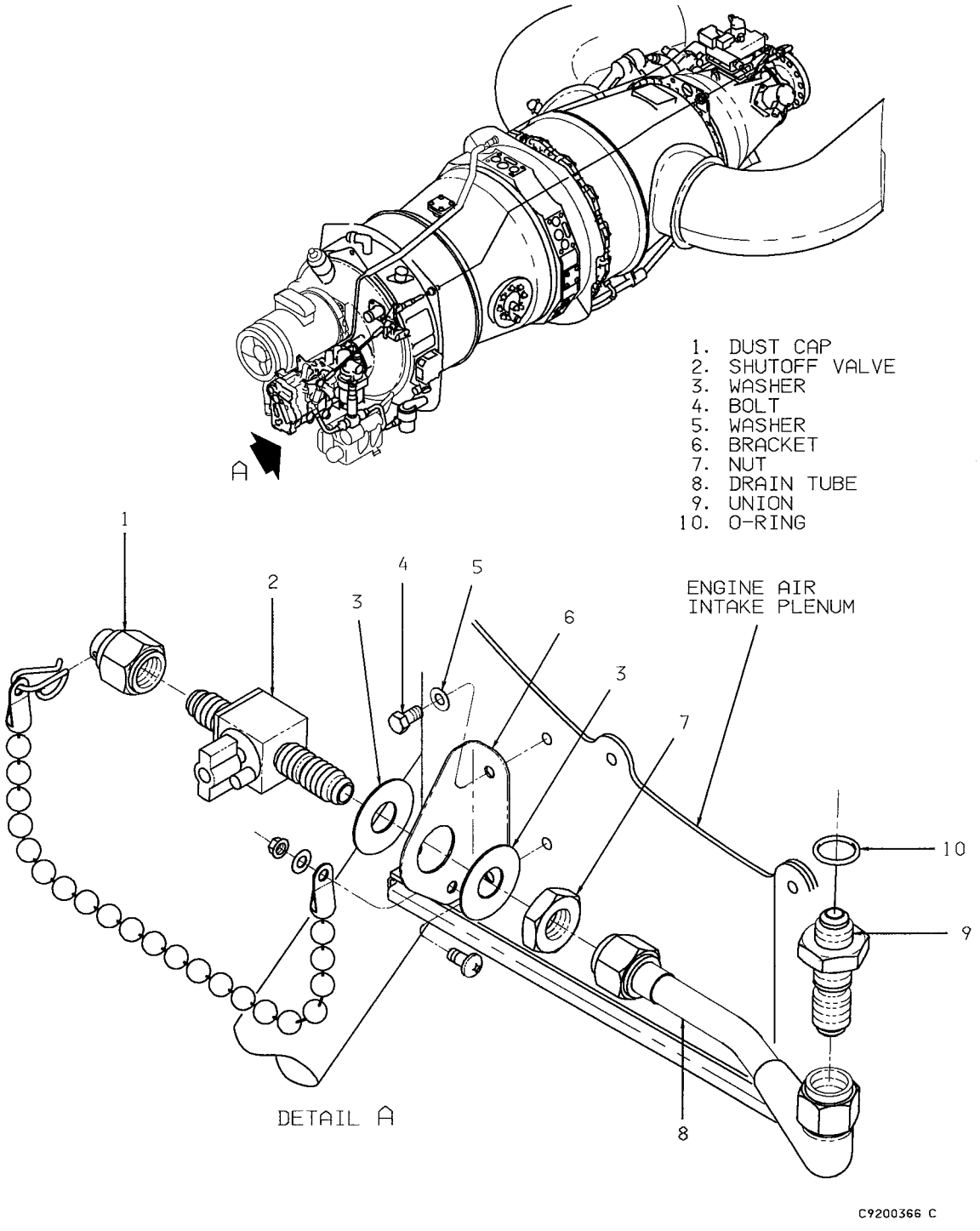
a. Install the shutoff valve (2) and washers (3) into the bracket (6).

b. Install nut (7) on the shutoff valve (2) to secure the shutoff valve to the bracket (6).

c. Connect the drain tube (8) to the shutoff valve (2) and union (9).

d. Place the shutoff valve in the closed position and safety wire.

e. Install the dust cap (1) on the shutoff valve (2).



Oil Tank Drain Shutoff Valve
Figure 7

f. Refer to Chapter 12-10-00 and check the oil level and add oil as necessary to replace the oil drained from the tank.

g. Motor the engine over with the starter only (no ignition) long enough to obtain an oil pressure reading.

CAUTION

Observe the starter operating limits of 30 seconds ON, 5 minutes OFF, 30 seconds ON, 5 minutes OFF, 30 seconds ON, then 30 minutes OFF.

h. Check the area around the shutoff valve for leaks.

i. If no leaks are found, install the engine cowling as detailed in Chapter 71-10-00.

INDICATING - DESCRIPTION AND OPERATION

WARNING

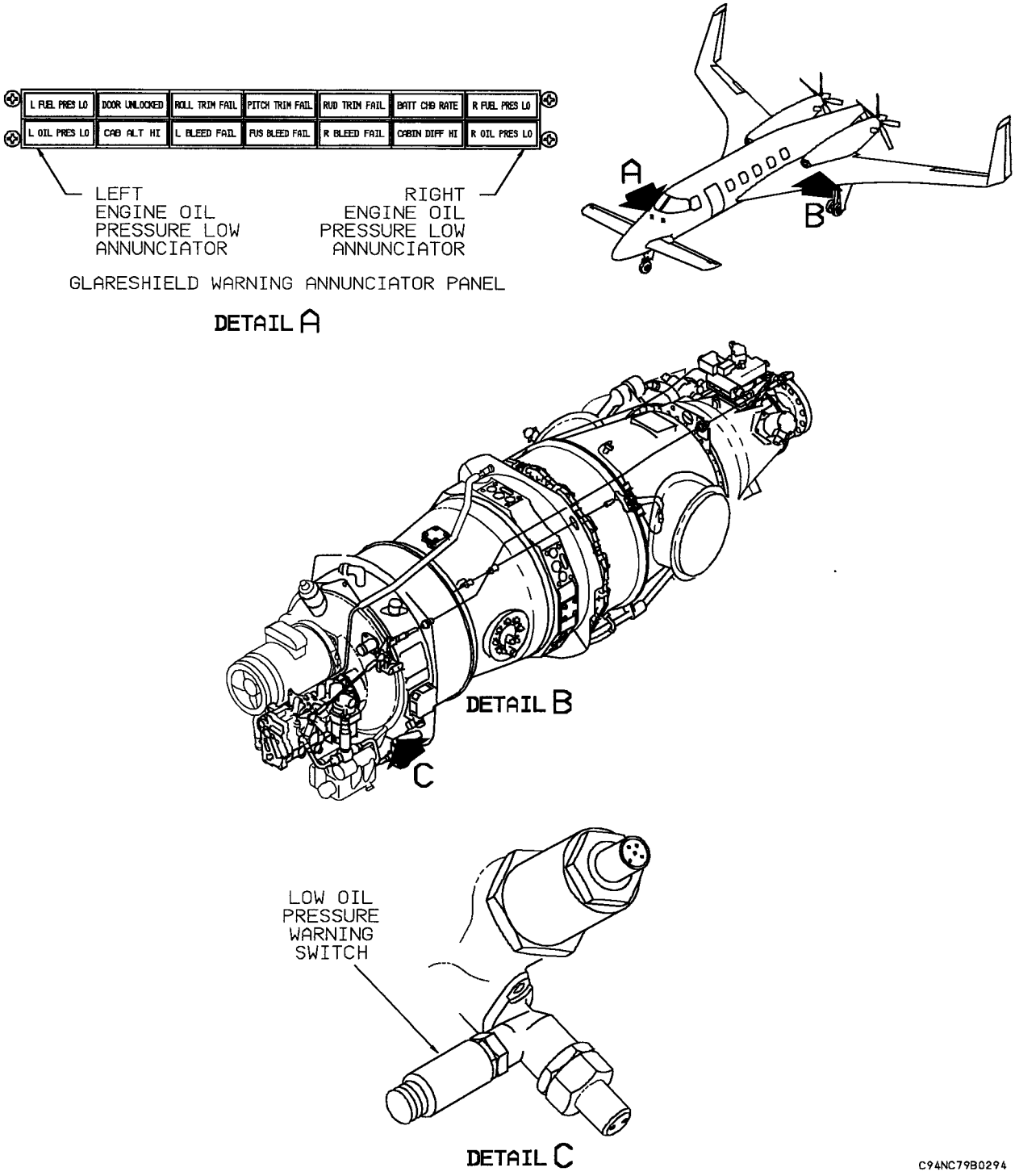
Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

LOW OIL PRESSURE WARNING SYSTEM

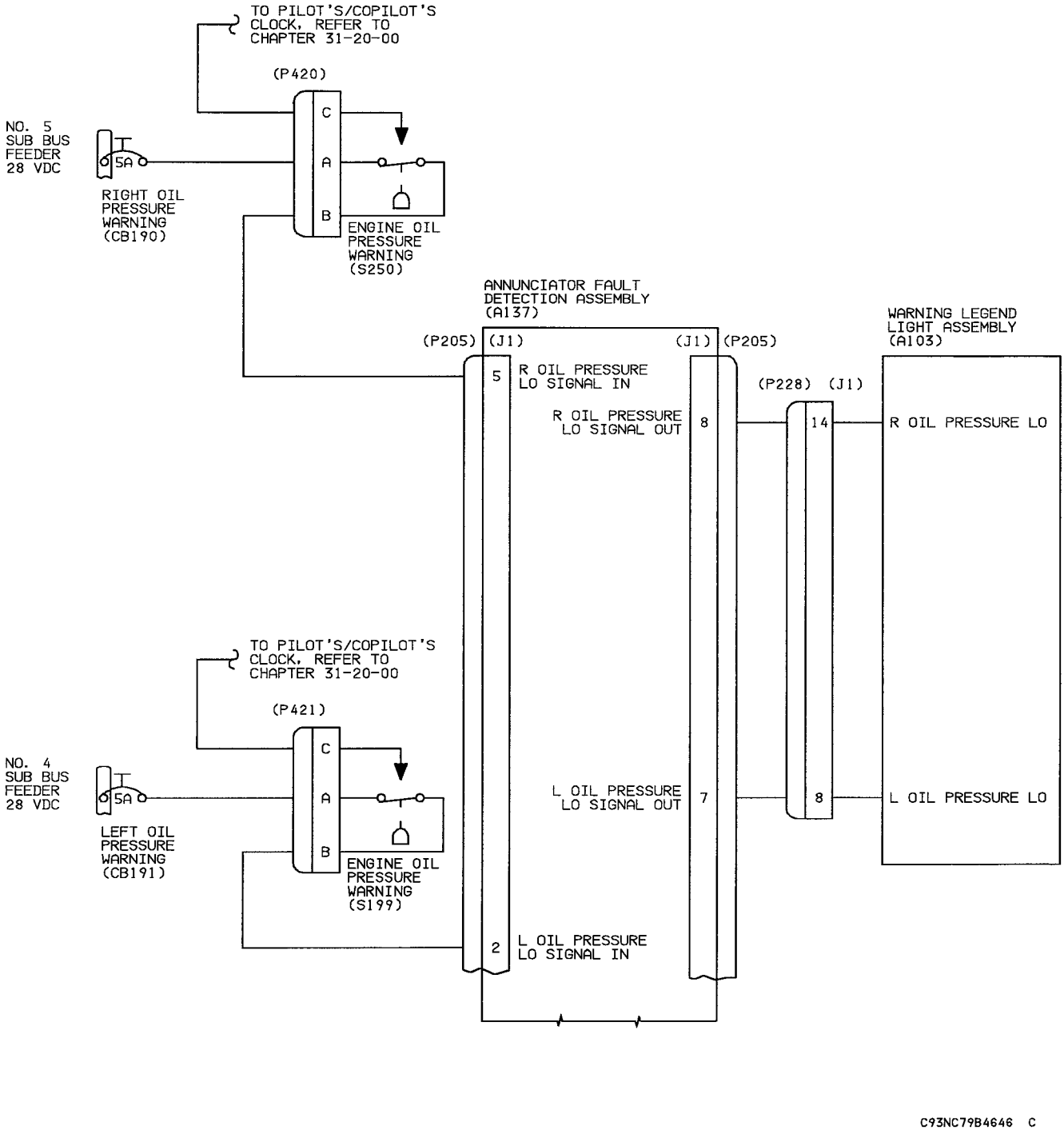
When the engine oil pressure in either engine drops below approximately 40 psi, the crew is notified by the low oil pressure warning system. Refer to Figures 1 and 2. The system consists of a low oil pressure switch mounted on the accessory gearbox housing of each engine, 5-amp L and R OIL PRESS WARNING circuit breakers on the LH circuit breaker panel in the pilot's console, and L and R OIL PRESS LO annunciators in the glareshield warning annunciator panel. The oil pressure warning switches close at a decreasing pressure of 40 ± 2 psi and open at an increasing pressure that is a maximum of 30 per cent above the closing pressure. When one of the switches closes, a circuit is completed between the appropriate circuit breaker and the warning annunciator panel to illuminate the L or R OIL PRESS LO annunciator. Refer to Chapter 31 for additional information on the operation of the warning annunciator system. The switches also serve to provide a 28 vdc signal to the pilot's and copilot's clocks when the engine oil pressure is sufficient to open the switches. Refer to Chapter 31-20-00 for details on the operation of the clocks.

OIL SYSTEM CHIP DETECTORS

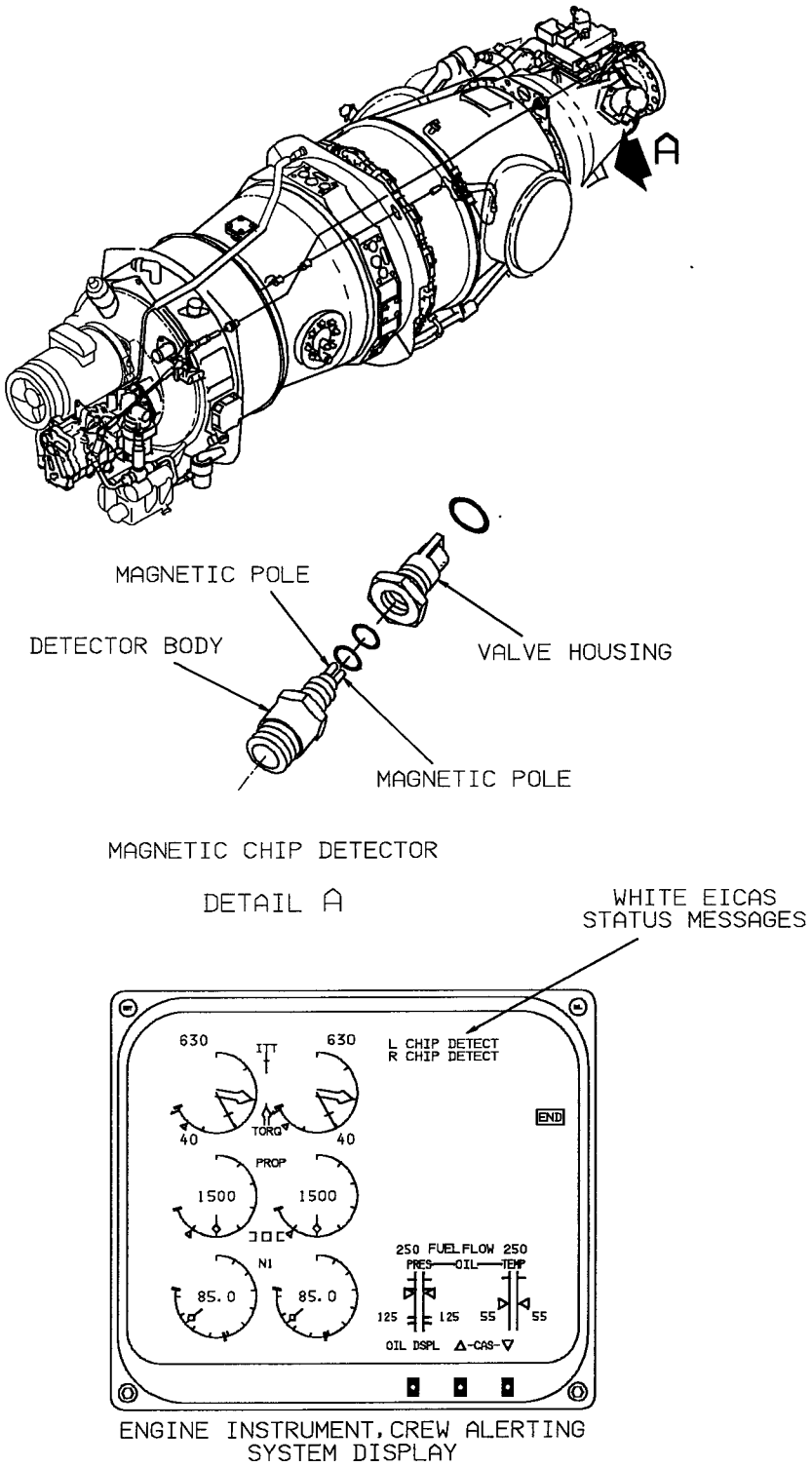
When the lubricating oil in one or both engines is contaminated with metal particles, a white "L CHIP DETECT" or "R CHIP DETECT" status message will be displayed on the Engine Instrument, Crew Alerting System (EICAS) display on the instrument panel. Refer to Figures 3 and 4. The presence of metal particles in the oil is disclosed by a magnetic chip detector installed in the reduction gearbox housing of each engine. The chip detector consists of two parts, the valve housing and the detector body. The valve housing is threaded to accept the detector body and is installed into an adapter mounted to the gearbox housing. The valve housing contains a spring-loaded valve which is self-closing when the detector body is removed, allowing removal of the chip detector body without the loss of engine oil from the gearbox housing. The chip detector body has two magnetic poles on one end (which are in contact with the engine oil when the detector is installed in the engine) and an electrical connector on the other end. Electrical power to the chip detectors is through the 5-amp LEFT and RIGHT CHIP DETR circuit breakers on the LH circuit breaker panel in the pilot's console. Metal particles in the engine oil are attracted to the magnetic poles. When enough particles are present to bridge the gap between the poles, an electrical circuit through the chip detector will be completed to the respective Engine Data Concentrator (EDC). The EDC digitizes the 28 vdc signal and transmits it along an ARINC 429 low speed data bus, through the Data Acquisition Unit (DAU), to the EICAS display where the white "L CHIP DETECT" or "R CHIP DETECT" status message is displayed. Refer to Chapter 31 for detailed information on operation of the Data Acquisition Unit and Engine Instrument, Crew Alerting System.



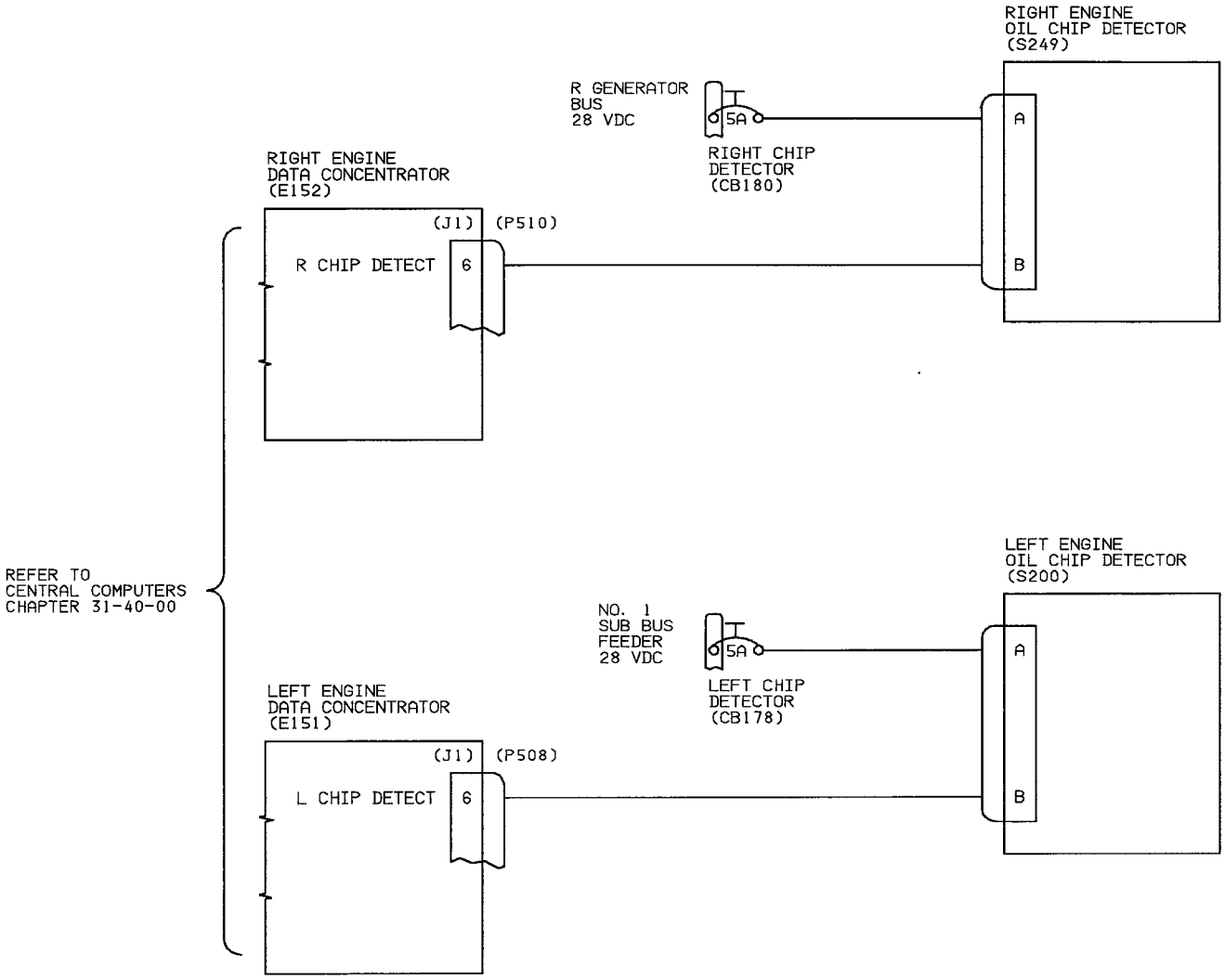
Engine Low Oil Pressure Indicating System
Figure 1



Engine Low Oil Pressure Indicating System Schematic
Figure 2



Engine Oil Chip Detector System
Figure 3



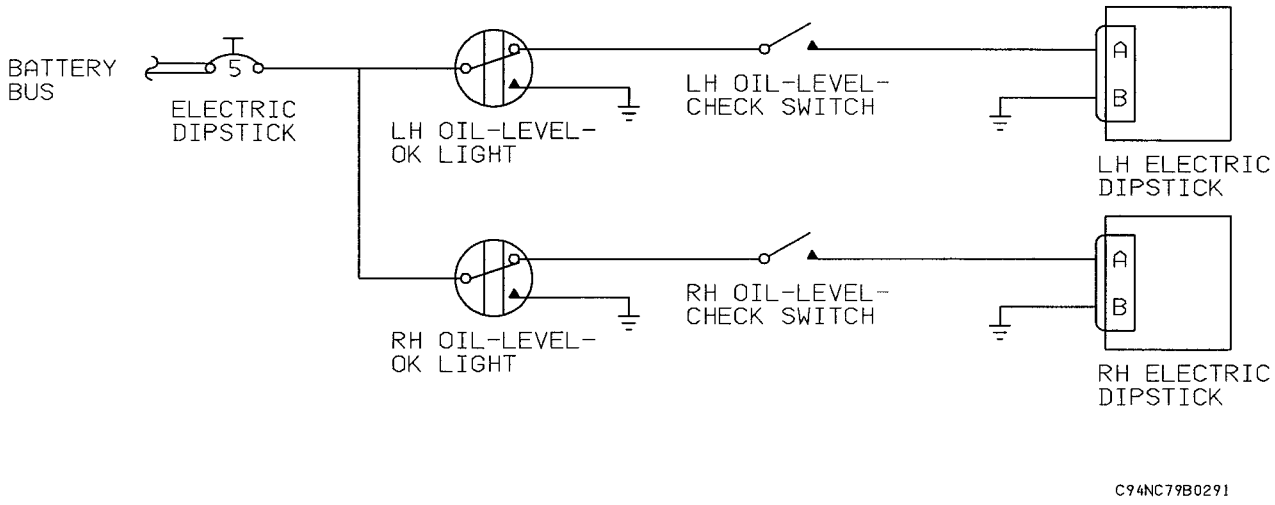
Engine Oil Chip Detector System Schematic
Figure 4

CAUTION

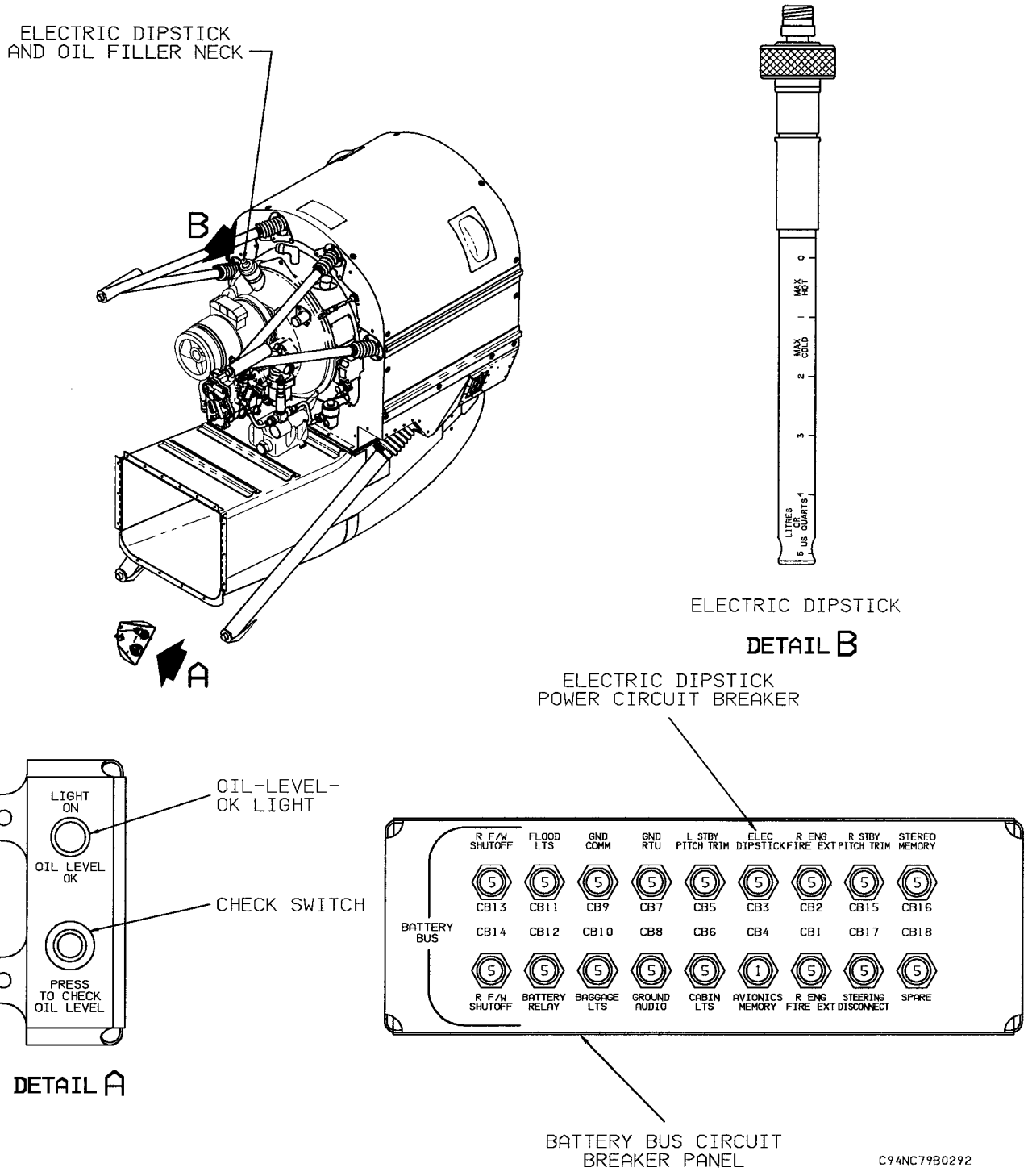
Anytime the oil system is contaminated by metal particles, the oil cooler must be replaced and the oil system flushed to prevent engine damage. In addition, all airplane components and associated plumbing utilizing the engine oil system, such as propeller governors, must be flushed until free of contamination.

OIL LEVEL INDICATING SYSTEM

An oil level indicating system installed in the airplane allows maintenance personnel to determine from the ground if oil needs to be added to the engines. Refer to Figures 5 and 6. The system consists of an OIL LEVEL OK light, an oil level check switch and an electric oil level indicator (dipstick) for each engine. If the engine oil level is adequate for safe operation of the engine, the OIL LEVEL OK light will illuminate when the oil level check switch is actuated. The green press-to-test OIL LEVEL OK light and the momentary-push button oil-level-check switch are mounted on a bracket attached to the engine fuel filter bracket on the aft side of the firewall. The light and switch are accessible through a door in the lower surface of the lower forward engine cowling. The electric dipstick is located in the engine oil tank filler neck at the eleven o'clock position on the accessory gearbox housing and is accessible through a door in the upper surface of the upper forward engine cowling. Power for both OIL LEVEL OK lights is from a 5-amp circuit breaker, placarded ELEC DIPSTICK, on the "hot" battery bus circuit breaker panel. If the oil level in the oil tank is within the acceptable range, the electric dipstick will complete the circuit to illuminate the OIL LEVEL OK light when the oil level indicator test switch is actuated, The electric dipstick is also marked in US quarts to indicate the amount of oil required to fill the engine oil tank under both hot and cold conditions. Since the OIL LEVEL OK light does not indicate the amount of oil in the engine, only that the level is within the normal range, the electric dipstick should be used as a conventional dipstick when adding oil to the engine. A sight gage is also provided at the nine o'clock position on the compressor inlet case of the engine.



Engine Oil Level Indicating System Schematic
Figure 5



Engine Oil Level Indicating System
Figure 6

INDICATING - TROUBLESHOOTING

The steps in Charts 1, 2 and 3 should be followed sequentially unless otherwise specified. An example would be to proceed to the next step when there are no directions immediately following a NO condition.

CHART 1
TROUBLESHOOTING - ENGINE OIL TEMPERATURE BULB

ENGINE OPERATING NO OIL TEMPERATURE

STEP 1	Is there any continuity between the temp bulb and the engine Data Concentrator?	**YES**	Replace Temperature Bulb
	NO		
	Restore continuity		

CHART 2
TROUBLESHOOTING - OIL PRESSURE INDICATING SYSTEM

ENGINE OPERATING, NO OIL PRESSURE INDICATED

STEP 1	Is the L or R OIL PRESSURE circuit breaker open?	**YES**	Reset circuit breaker.
	NO		
STEP 2	Is the engine oil level low?	**YES**	Replenish as instructed in Chapter 12-10-00.
	NO		
STEP 3	Is there continuity between the L or R OIL PRESSURE circuit breaker and the appropriate transducer?	**YES**	Go to Step 4.
	NO		
	Restore Continuity.		
STEP 4	Is there continuity between the transducer and the engine data concentrater?	**YES**	Go to Step 5.
	NO		
	Restore Continuity		
STEP 5	Perform Functional check of transducer as outlined under OIL PRESSURE TRANSDUCER FUNCTIONAL TEST in this chapter. Is the transducer functioning properly?		

CHART 3
TROUBLESHOOTING - CHIP DETECTORS

STEP 1	Is the L or R CHIP DETECT circuit breaker open?	**YES**	Reset circuit breaker.
	NO		
STEP 2	Is there continuity between the L or R CHIP DETECT circuit breaker and the appropriate chip detector?	**YES**	Go to Step 3.
	NO		
	Restore continuity.		
STEP 3	Is there continuity between the chip detector and the Engine Data Concentrator?	**YES**	Go to Step 4.
	NO		
	Restore continuity.		
STEP 4	Inspect chip detector as outlined under CHIP DETECTOR INSPECTION in this chapter. Does the chip detector meet the inspection requirement.		
	NO		
	Replace chip detector.		

INDICATING - MAINTENANCE PRACTICES

OIL-LEVEL-OK LIGHT REMOVAL

a. Remove all electrical power from the airplane.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

c. Disconnect the electrical wiring from the light (1). See Figure 7.

d. Remove the nut (3) and washer (2) that retain the light to the bracket (7).

e. Remove the light from the bracket.

OIL-LEVEL-OK LIGHT INSTALLATION

a. Insert the light (1) into the bracket (7). See Figure 7.

b. Install the washer (2) and nut (3) to retain the light to the bracket.

c. Connect the electrical wiring to the light.

d. Install the engine cowling as detailed in Chapter 71-10-00.

e. Restore electrical power to the airplane.

OIL-LEVEL-TEST SWITCH REMOVAL

a. Remove all electrical power from the airplane.

b. Remove the engine cowling as outlined in Chapter 71-10-00.

c. Disconnect the electrical wiring from the switch (4). See Figure 7.

d. Remove the nut (6) and washer (5) that retain the switch to the bracket (7).

e. Remove the switch from the bracket.

OIL-LEVEL-TEST SWITCH INSTALLATION

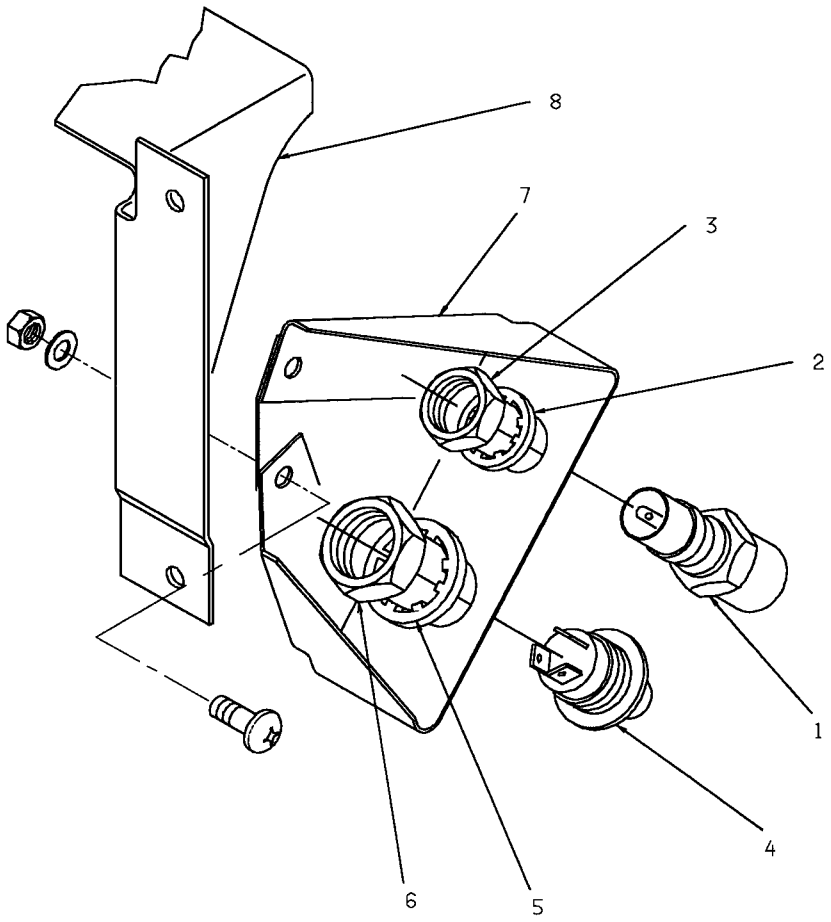
a. Insert the switch (4) into the bracket (7). See Figure 7.

b. Install the washer (5) and nut (6) to retain the switch to the bracket.

c. Connect the electrical wiring to the switch.

d. Install the engine cowling as detailed in Chapter 71-10-00.

e. Restore electrical power to the airplane.



1. OIL-LEVEL-OK LIGHT
2. WASHER
3. NUT
4. OIL-LEVEL-CHECK SWITCH
5. WASHER
6. NUT
7. LIGHT AND SWITCH MOUNTING BRACKET
8. FUEL FILTER MOUNTING BRACKET

C94NC79B0293

Oil Level OK Light and Test Switch Installation
Figure 7

LOW OIL PRESSURE WARNING SWITCH REMOVAL

a. Remove all electrical power from airplane.

b. Remove the engine cowling as outlined in chapter 71-00-00.

c. Disconnect the electrical connector from switch. See Figure 8.

d. Remove switch and discard O-ring.

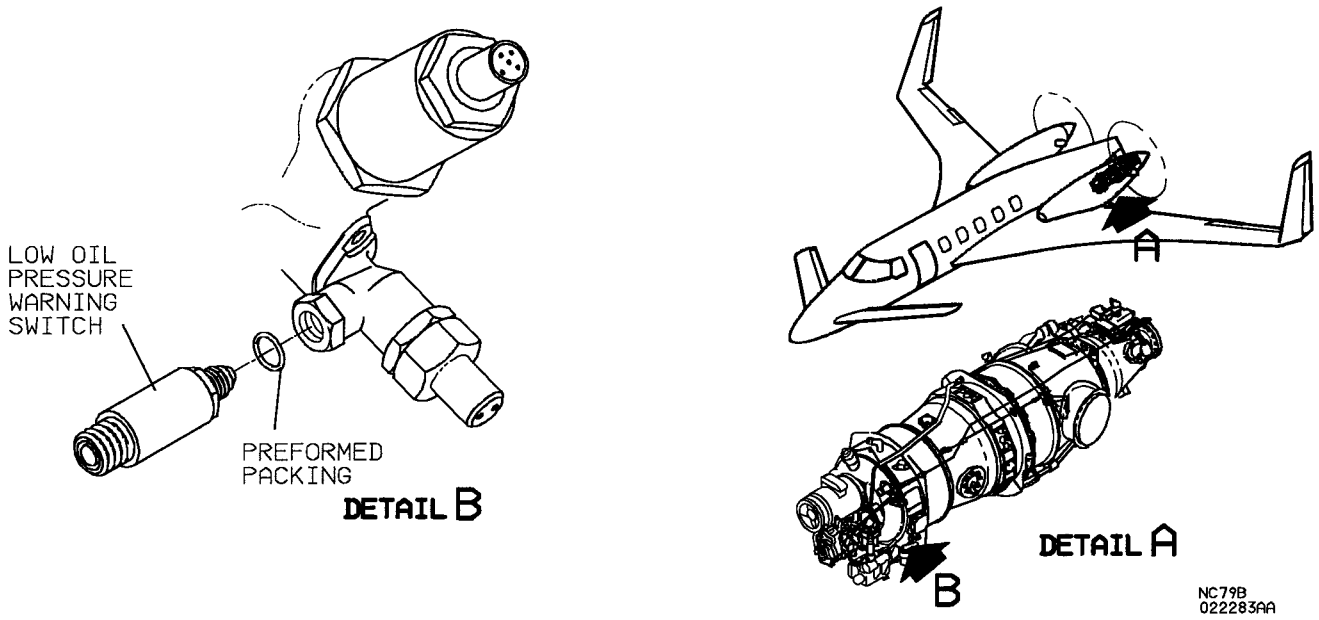
LOW OIL PRESSURE WARNING SWITCH INSTALLATION

a. Install new O-ring on the switch. See Figure 8.

b. Apply thin coat of lubricant (1,Chart 1, 79-00-00) to external threads of switch. Install switch and O-ring into engine. Torque switch to 40 inch pounds.

c. Install electrical connector.

d. Install cowling as outlined in chapter 71-00-00.



Low Oil Pressure Warning Switch
Figure 8

e. Restore electrical power to airplane.

LOW OIL PRESSURE WARNING SWITCH FUNCTIONAL TEST

a. The following items are required to perform this test:

1. Regulated supply of dry, filtered, compressed air at no less than 50 psi.
2. Air pressure gage calibrated in .01 psi increments.
3. Plumbing necessary to connect switch to pressure supply.
4. Continuity tester.

b. Remove the switch from the airplane as instructed under LOW OIL PRESSURE WARNING SWITCH REMOVAL in this chapter.

c. Connect the pressure source and pressure gage to the switch.

d. Connect the continuity tester to the normally closed contacts of the switch, pins "A" and "B".

e. With no pressure applied to the switch, check for continuity. If no continuity is indicated replace the switch.

f. Apply pressure to the switch at a slow steady rate until 50 psi is indicated on the pressure gage. There should be no continuity across the normally closed contacts; if there is, replace the switch.

g. Reduce the pressure at a slow steady rate, noting at what pressure continuity is indicated. The switch should actuate between 38 and 42 psi; if not, replace the switch.

h. Slowly increase the pressure, noting at what pressure there is no longer continuity. The difference between the decreasing and increasing actuation points shall be a maximum of 30% of the actual decreasing point.

NOTE

To find the maximum increasing point for a specific decreasing point refer to Chart 4; locate the decreasing point, noted in step "g", under ACTUATE ON DECREASE PSI and read across under ACTUATE ON INCREASE PSI, the increasing actuation pressure noted in step "h" must not be higher than this value. If it is, replace the switch.

CHART 1
LOW OIL PRESSURE WARNING SWITCH ACTUATION POINTS

ACTUATE ON DECREASE PSI	ACTUATE ON INCREASE PSI
42.0	54.60
41.5	53.95
41.0	53.30
40.5	52.65
40.0	52.00
39.5	51.35

CHART 1
LOW OIL PRESSURE WARNING SWITCH ACTUATION POINTS

ACTUATE ON DECREASE PSI	ACTUATE ON INCREASE PSI
39-0	50.70
38-5	50.05
38-0	49.40

i. Repeat the check four times to ensure consistency.

j. Disconnect the test equipment from the switch.

CAUTION

Bleed the air pressure down to 0 psi before disconnecting the air supply from the switch.

k. Install the switch in the airplane as instructed under LOW OIL PRESSURE WARNING SWITCH INSTALLATION in this chapter. Replace any switch that does not meet the requirements of this test procedure.

CHIP DETECTOR REMOVAL

The chip detector consists of two parts, the body (2) and the valve housing (5). These parts are not procurable individually. If one part of the chip detector requires replacement, replace both parts; do not mix parts from one chip detector with those from another.

a. Remove all electrical power from the airplane.

b. Remove the lower aft engine cowling as outlined in Chapter 71-10-00.

c. Disconnect the electrical connector (1) from the chip detector body (2). See Figure 9.

d. Cut the safety wire and remove the chip detector body (2) from the valve housing (5). The spring-loaded valve in the valve housing will close when the chip detector body is removed to prevent the loss of any oil from the gearbox.

e. If required, remove the valve housing (5) as follows:

1. Place a suitable container under the gearbox to catch the oil that will be released when the valve housing is removed.

2. Cut the safety wire and remove the valve housing from the adapter (7).

CHIP DETECTOR INSTALLATION

The chip detector consists of two parts, the body (2) and the valve housing (5). These parts are not procurable individually. If one part of the chip detector requires replacement, replace both parts; do not mix parts from one chip detector with those from another. See Figure 9.

a. If required, install the valve housing (5) as follows:

1. Install a new O-ring (6) on the valve housing.

2. Install the valve housing into the adapter (7). Torque the valve housing to 175 to 200 inch-pounds and safety wire.

b. Install a new O-ring (3) on the chip detector body (2).

c. Install the detector body (2) into the valve housing (5). Torque the detector body to 45 to 55 inch-pounds and safety wire.

CAUTION

Due to the size of wrench required to fit the wrench flats on the chip detector, overtightening can result in the fracturing of the chip detector body with just "normal" hand-wrenching pressure. Therefore, only tighten the chip detector body with a calibrated torque wrench, and apply no more than 45 to 55 inch-pounds of torque to the detector body.

d. Connect the electrical connector (1) to the chip detector.

e. Refer to Chapter 12-10-00 and check the oil level in the engine oil tank and add oil as required.

CHIP DETECTOR INSPECTION

The chip detector consists of two parts, the body (2) and the valve housing (5). These parts are not procurable individually. If one part of the chip detector requires replacement, replace both parts; do not mix parts from one chip detector with those from another.

a. Remove the chip detector body (2) as described under CHIP DETECTOR REMOVAL in this chapter.

b. Inspect the chip detector for metal deposits on the magnetic poles. If any metal material is found, refer to the Pratt & Whitney PT6A-67A Engine Maintenance Manual for the proper action to be taken.

c. With a continuity tester connected to the electrical pins on the chip detector body, place a steel bar across the magnetic poles and check for continuity. If there is no continuity, replace the chip detector.

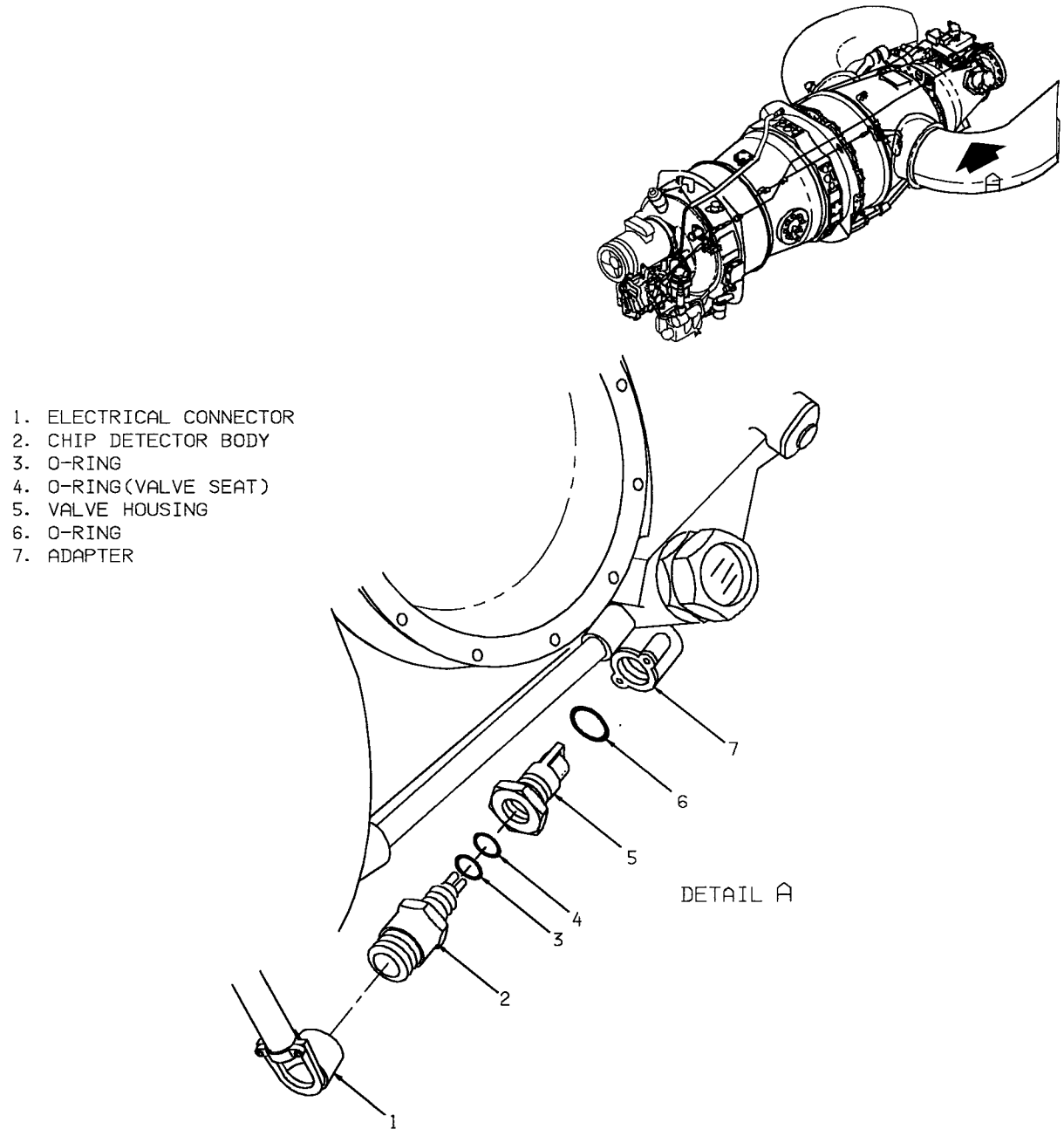
d. Inspect the threads on the chip detector body and valve housing and replace any chip detector with damaged threads.

e. Check the electrical connector for bent, broken or burned pins.

f. Check for signs of excessive oil leakage in the valve housing, which would indicate a deteriorated or damaged valve seat O-ring (4) or a valve that is not functioning properly. If required, remove the valve housing as instructed under CHIP DETECTOR REMOVAL in this chapter; the valve seat O-ring (4) can then be replaced. If the valve is sticking or has a broken spring, replace the complete chip detector. Install the valve housing or new chip detector as instructed under CHIP DETECTOR INSTALLATION in this chapter.

g. Install a new O-ring (3) on the chip detector body and install the chip detector body in the valve housing as instructed under CHIP DETECTOR INSTALLATION in this chapter.

h. Refer to Chapter 12-10-00 and check the oil level in the engine oil tank and add oil as required.



Chip Detector Installation
Figure 9

Beechcraft
STARSHIP 1
MAINTENANCE MANUAL

CHAPTER 80 - STARTING

LIST OF PAGE EFFECTIVITY

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CHAPTER 80 - STARTING

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General - Troubleshooting (Effectivity: All) 5
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GENERAL - DESCRIPTION AND OPERATION (Effectivity: All) (Figure 1)

WARNING

Anytime the airplane is on the ground (whether on jacks or on the wheels), the nose and main landing gear MUST be pinned in the down and locked position. The only exceptions to this would be during landing gear operational checks, during the removal or installation of the landing gear components, and during taxiing operations prior to takeoff or after landing. When any work is being performed in the nose gear wheel well, the nose gear doors MUST be pinned in the open position. Refer to Chapters 32-10-00 and 32-20-00 for the proper lock pin installation procedures.

The starting system for the Starship 1 consists of two 28-volt, 300-amp DC starter/generators, two engine start relays, two ignition/start switches, two field and sense relays, two start control circuit breakers and two ignition isolation relays. The left and right start control circuit breakers power the left and right ignition/start switches, and are placarded ENGINES, START CONTROL, LEFT-RIGHT on the left circuit breaker panel in the cockpit. The ignition isolation relays are mounted in the logic relay panel which is located on the aft fuselage equipment shelf. The field and sense relays are mounted on a bracket in the wing center section. The engine start relays are located in the left and right nacelles and mounted inside the left and right generator power relay panels respectively. Each starter/generator is mounted on its respective engine accessory gearbox. The ignition/start switches are double pole, three position switches located on the center subpanel placarded IGNITION AND ENGINE START, LEFT-RIGHT, ON-OFF-STARTER ONLY. The ignition system provides spark for combustion when the ignition/start switch is placed in the ON position only, while the starter/generator motors the engine in both the ON and STARTER ONLY positions. For more information on the ignition system refer to Chapter 74-00-00.

Engine starting is accomplished with three separate power sources. The battery is normally used to start the first engine, then either starter/generator is used to cross-start the opposite engine. An auxiliary ground power unit capable of providing 600 amperes continuous output, and 750 amperes intermittent output, may be used for engine starts. Battery voltage must be above 20 vdc for external power to be applied to the airplane. Refer to Chapter 24-40-00 for detailed information on the external power system.

STARTER/GENERATORS (Effectivity: All)

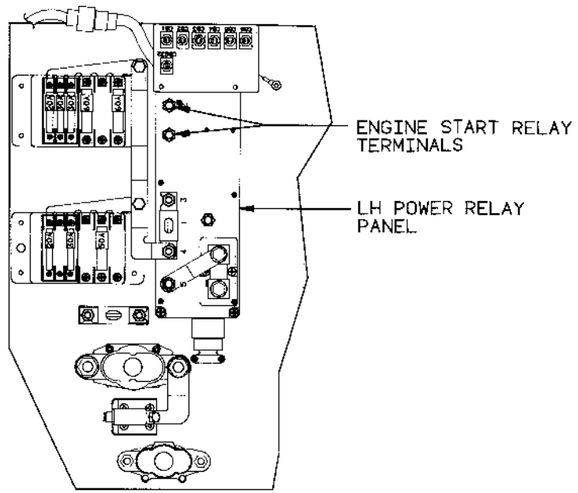
The Starship 1 uses a starter/generator to produce torque for engine starts. When functioning as a starter, a series starter winding is used instead of the shunt field winding used during generator operation. Refer to Chapter 24-30-00 for detailed information on generator operation of the starter/generator, including an illustration of mounting and associated hardware.

A quick disconnect mounting adapter is bolted to the engine accessory gearbox mounting pad and provides the unit with a pin aligned mount. The starter/generator mates with the engine gearbox by means of a splined drive shaft, providing a direct torque transfer for both of the starter/generator's functions. The starter/generator utilizes a splined drive shaft with a shear torque of 1600 inch-pounds. If a condition occurs causing the starter/generator drive shaft torque to exceed 1600 inch-pounds, the splined drive shaft will shear, minimizing damage to the starter/generator and engine components.

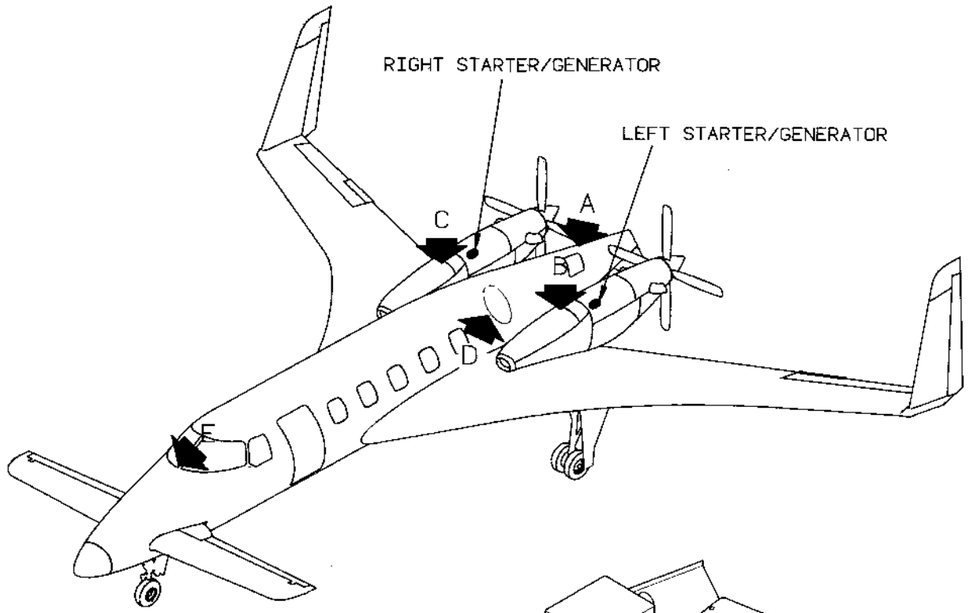
The starter/generator is blast cooled during flight by a scoop located on the LH side of the lower forward engine cowling. The scoop is directly connected to an air duct routed to the cooling cap at the forward end of the starter/generator. During ground operation, the starter/generator is dependent upon a shaft driven internal fan to provide sufficient cooling. Refer to Chapter 75-20-00 for detailed information on starter/generator cooling.

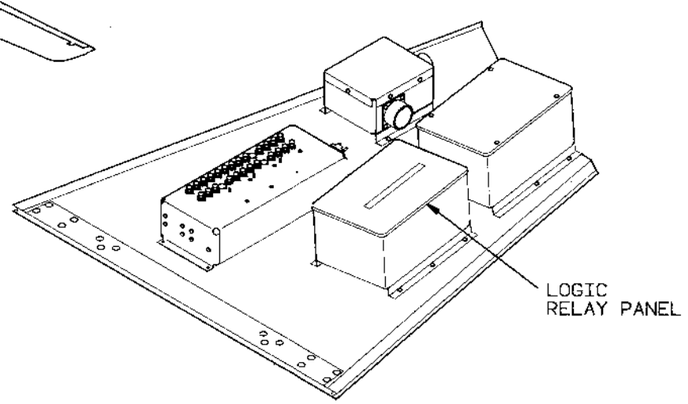
CAUTION

Starter operating time limits are as follows: 30 seconds ON, 5 minutes OFF, 30 seconds ON, 5 minutes OFF, 30 seconds ON, and 30 minutes OFF.

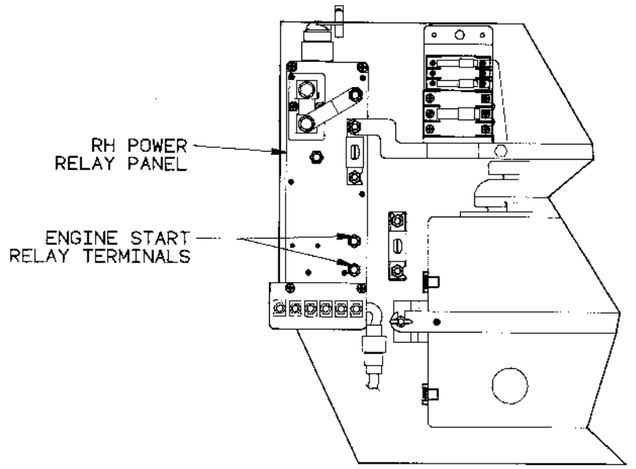


DETAIL B
LH NACELLE ELECTRICAL EQUIPMENT SHELF

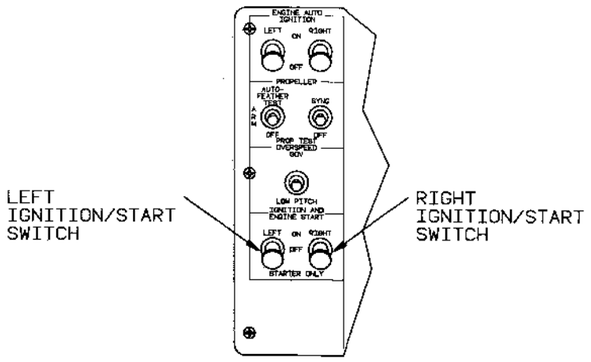




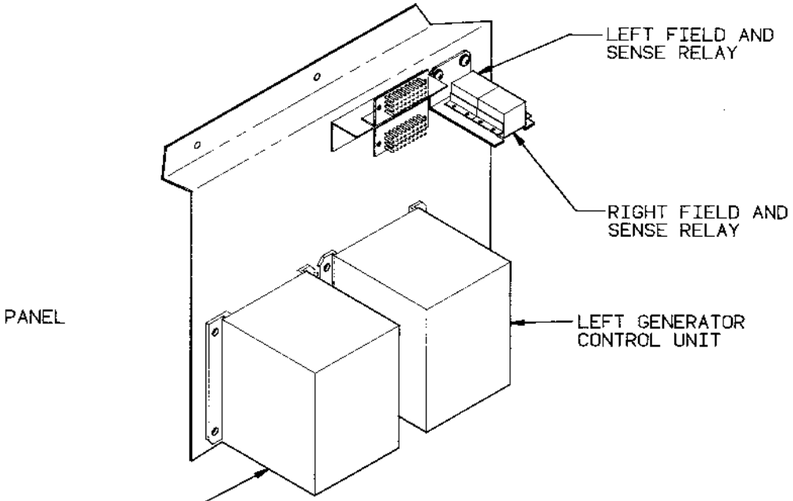
DETAIL A
AFT FUSELAGE
EQUIPMENT SHELF



DETAIL C
RH NACELLE ELECTRICAL
EQUIPMENT SHELF



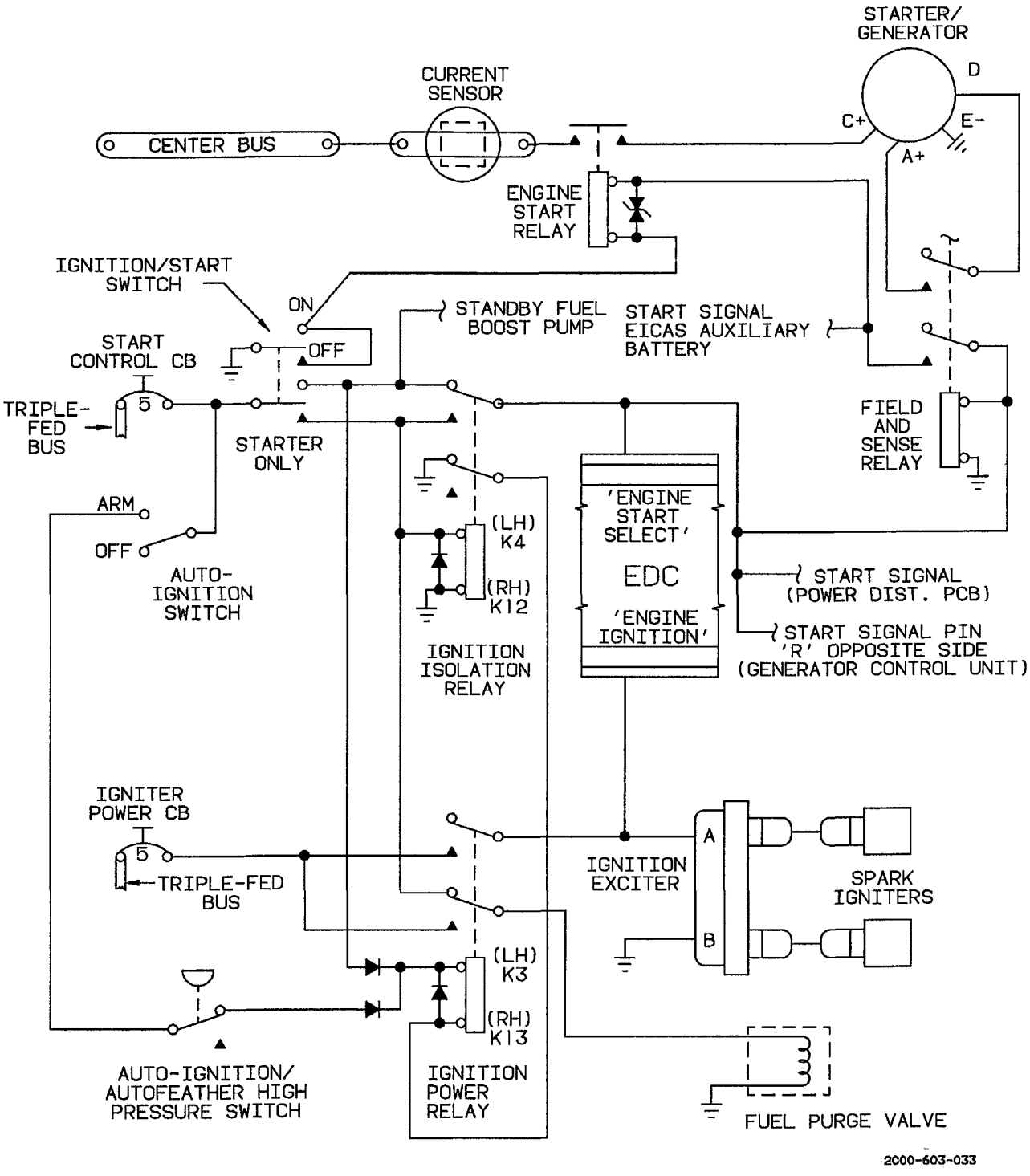
DETAIL E
CENTER SUBPANEL



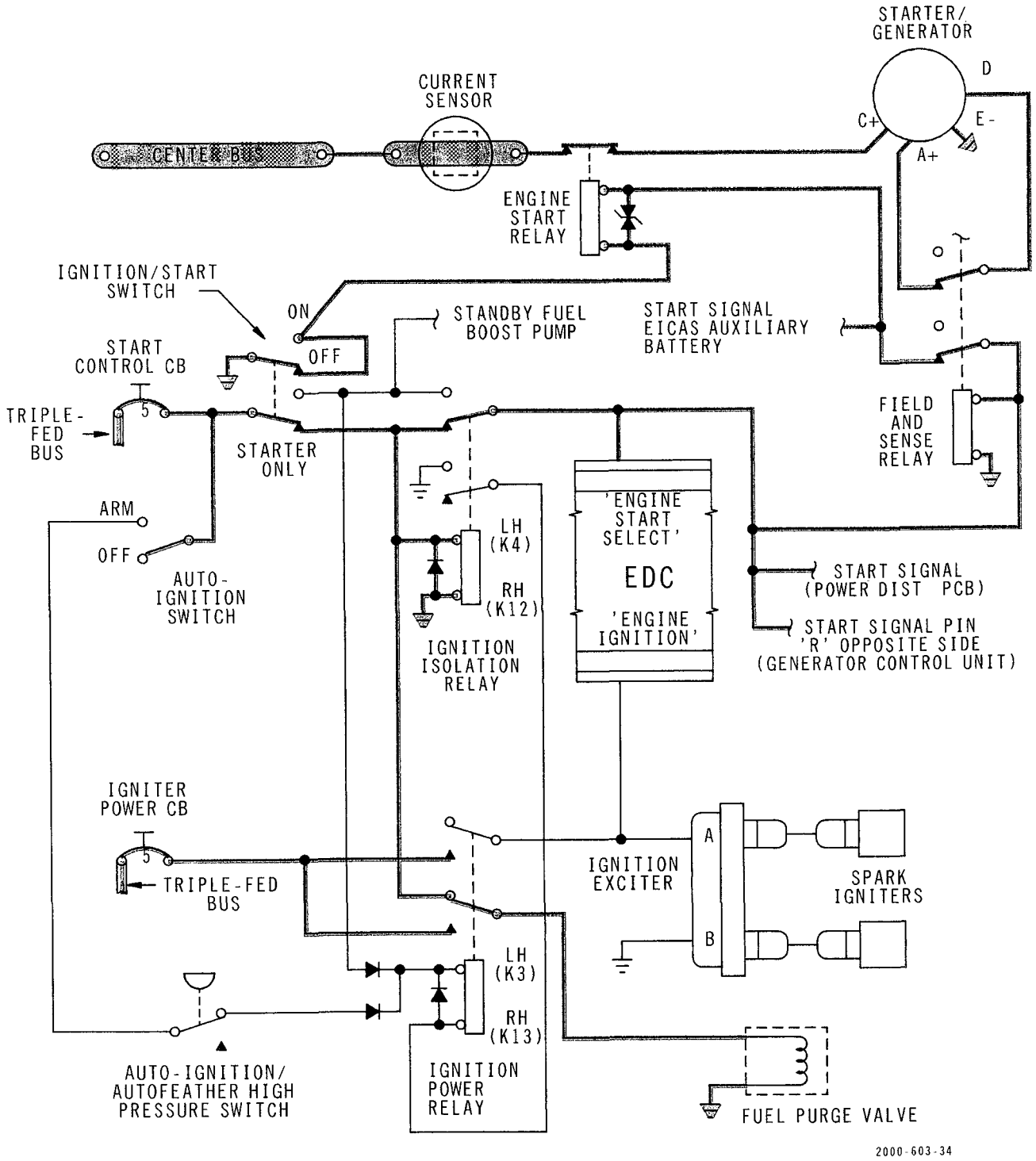
DETAIL D
WING CENTER SECTION
ELECTRICAL EQUIPMENT PANEL

2000-355-32

Engine Starting System (Effectivity: All)
Figure 1



Starting System Schematic (OFF) (Effectivity: All)
Figure 2



Starting System Schematic (STARTER ONLY) (Effectivity: All)
Figure 3

STARTING CIRCUIT (Effectivity: All) (Figures 2 and 3)

NOTE

Both the left and right sides of the starting system are identical, but are individual systems independent of each other. Due to the inherent relationship between the starting and ignition systems, they are shown together in Figures 2 and 3.

The start control circuit breakers are powered by the triple-fed bus and supply voltage to one set of the ignition/start switch contacts. When the ignition/start switch is placed in the STARTER ONLY position, 28 vdc energizes the ignition isolation relay and flows to one set of contacts on both the ignition isolation and ignition power relays, energizing the fuel purge valve through the ignition power relay contacts. The 28 vdc to the ignition isolation relay energizes the field and sense relay and provides signals to the Engine Data Concentrator (EDC), power distribution PCB, and opposite side generator control unit (GCU). The signal to the EDC tells it that an engine start has been selected. The signal to the power distribution PCB will cause the PCB to disable the power relay panel current sensors, allowing the bus tie relays to remain closed during high current operation. The start signal is also applied to pin "R" on the opposite GCU, activating the current limiting feature of that GCU. This circuit will limit on-line starter/generator output to 250 amperes, protecting the on-side current limiter from overload during cross-starts. Energizing the field and sense relay shorts the starter/generator shunt field through one set of closed contacts, preventing generator operation and transients from entering the GCU during engine starts. Voltage is applied through a separate set of field and sense relay contacts to energize the Engine Instrument, Crew Alerting System (EICAS) standby power relay on the EICAS auxiliary battery rack, providing EICAS auxiliary battery power to operate the EICAS display during starter operation. Refer to Chapter 24-32-00 for detailed information on auxiliary power. The 28 vdc from the same set of field and sense relay contacts, and a ground from the ignition/start switch will simultaneously be applied to the engine start relay coil, energizing the relay and connecting the center bus to terminal "C" of the starter/generator. The starter/generator series winding will be energized, producing torque to motor the engine.

NOTE

The starting circuit is the same with the ignition/start switch in the ON position or the STARTER ONLY position with two exceptions: the ignition isolation relay is not energized, and the ignition power relay is energized, routing power to the ignition exciter. Refer to Chapter 74-00-00 for a schematic with the ignition/start switch in the ON position.

GENERAL - TROUBLESHOOTING (Effectivity: All)

Before utilizing the troubleshooting charts, ensure that the auxiliary ground power unit is set at 28 vdc and that the battery has a charge of 20 vdc or more. Many times a loose or damaged pin connection will cause a fault. Ensure that all connectors are clean and free of corrosion and repair or replace any damaged or loose pins. Refer to the BEECHCRAFT Starship 1 Wiring Diagram Manual for specific pin and plug locations. Voltage checks should be made with the ignition/start switch in the specified position and the battery switch ON. Continuity checks should be made with power removed from the specific connector or terminal. The steps in the troubleshooting charts should be followed sequentially unless otherwise specified. An example would be to proceed to the next step when there are no directions immediately following a NO condition.

NOTE

If a problem cannot be isolated with the charts in this chapter refer to Chapter 24-30-00 for detailed information on the generator control units, power distribution PCB, and other DC generation components which work in unison with the starter/generator.

CHART 1
TROUBLESHOOTING - STARTING SYSTEM (Effectivity: All)

STARTER ROTATES BUT ENGINE DOES NOT

STEP 1	Remove the starter/generator from the engine with procedures outlined in Chapter 24-30-00. Are there any signs of damage or excessive wear to the drive shaft splines?	**YES**	Refer to the BEECHCRAFT Starship 1 Component Maintenance Manual for repair procedures and proceed to Step 3.
	NO		
STEP 2	Is the starter/generator drive shaft sheared or damaged?	**YES**	Refer to the BEECHCRAFT Starship 1 Component Maintenance Manual for repair procedures and proceed to Step 3.
	NO		
STEP 3	Are there any signs of damage to the splines in the engine accessory gearbox?	**YES**	Refer to the Pratt & Whitney Engine Maintenance Manual for further troubleshooting information.
	NO		
	Recheck each component for damage. If no damage is evident refer to the Pratt & Whitney Engine Maintenance Manual to further troubleshoot the engine components.		

CHART 2
TROUBLESHOOTING - STARTING SYSTEM (Effectivity: All)

STARTER/GENERATOR FAILS TO OPERATE (1)

STEP 1	Push the TRIP FED volts button on the center subpanel. Is a reading slightly less than battery voltage displayed?	**YES**	Proceed to Step 2.
	NO		
	Refer to Chapter 24-50-00 for information on bus power distribution and schematics to help isolate potential faults.		
STEP 2	Is the start control circuit breaker open?	**YES**	Close circuit breaker.
	NO		
STEP 3	Does the start control circuit breaker have continuity across its contacts?	**YES**	Proceed to Step 4.
	NO		
	Replace the start control circuit breaker. System should be functional.		
STEP 4	Press the CTR BUS volts button on the center subpanel. Is a reading slightly less than battery voltage displayed?	**YES**	Proceed to Step 5.

CHART 2
TROUBLESHOOTING - STARTING SYSTEM (Effectivity: All) *(Continued)*

STARTER/GENERATOR FAILS TO OPERATE (1)

NO

Refer to Chapter 24-50-00 for information on bus power distribution and schematics to help isolate potential faults.

STEP 5	Is pin "E" of the starter/generator well grounded?	**YES**	Proceed to Step 6.

NO

Restore continuity to ground.

STEP 6	Is there battery voltage at pin "C" of the starter/generator? (See Note 2.)	**YES**	Refer to the BEECHCRAFT Starship 1 Component Maintenance Manual for troubleshooting or return the unit to the overhaul shop for repair if required.

NO

Check for continuity between the starter/generator terminal "C" and terminal 2 on the generator power relay panel. If the connection is good proceed to Step 7, if not, restore continuity.

STEP 7	Is there battery voltage at terminal 1 of the generator power relay panel?	**YES**	Proceed to Step 8.

NO

Correct the open circuit between the center bus and terminal 1 of the generator power relay panel.

STEP 8	Is there battery voltage at terminal 6 of the generator power relay panel?	**YES**	Proceed to Step 9.

NO

Internal fault of the generator power relay panel. Check the current sensor and internal connections for continuity.

STEP 9	Is there battery voltage at terminal 2 of the generator power relay panel? (See Note 2.)	**YES**	The system should be functional. If a problem still exists recheck the connection between the starter/generator and terminal 2 on the generator power relay panel. If good, troubleshoot the starter/generator using the BEECHCRAFT Starship 1 Component Maintenance Manual.

NO

STEP 10	Is there battery voltage at pin "Z" in the generator power relay panel connector? (See Note 2.)	**YES**	Proceed to Step 11.

CHART 2
TROUBLESHOOTING - STARTING SYSTEM (Effectivity: All) *(Continued)*

STARTER/GENERATOR FAILS TO OPERATE (1)

NO

Refer to Chart 3, NO POWER TO ENGINE START RELAY COIL, in this chapter to troubleshoot the fault.

STEP 11	Is there continuity between pin 18(RH), 23(LH) in center subpanel connector and pin "a" in the power relay panel connector?	**YES**	Proceed to Step 12.

NO

Locate the break and restore continuity to the circuit.

STEP 12	Is there continuity between pins 17(RH), 22(LH) and the LH subpanel ground block?	**YES**	Proceed to Step 13.

NO

Locate the break and restore continuity to ground.

STEP 13	Is there continuity between pins 17(RH), 22(LH) and pins 18(RH), 23(LH) in the center subpanel connector? (See Note 2.)	**YES**	System should be functional, if not repeat the steps to isolate a fault not detected the first time.

NO

Faulty ignition/start switch or a wiring fault from the connector to the switch itself. Check for continuities at the switch to determine which fault applies and correct the problem by replacing the switch with procedures outlined in Chapter 39-10-00 or by repairing the wiring.

1. Voltage checks should be performed with the opposite engine OFF and the battery switch in the ON position. Continuity checks should be made with power removed from the circuit being checked.

2. Perform the check for both ignition/start switch positions; ON and STARTER ONLY. If only one of the switch positions proves to be inoperative during the check suspect a malfunction with that set of ignition/start switch contacts. Refer to the BEECHCRAFT Starship 1 Wiring Diagram Manual for pin numbers to check the switch for continuity. Refer to Chapter 39-10-00 for switch replacement if necessary.

CHART 3
TROUBLESHOOTING - STARTING SYSTEM (Effectivity: All)

NO POWER TO ENGINE START RELAY COIL (1)

STEP 1	Is there battery voltage at pin "1C" in the field and sense relay connector? (See Note 2.)	**YES**	Correct open circuit between field and sense relay connector and generator power relay panel connector.

NO

CHART 3
TROUBLESHOOTING - STARTING SYSTEM (Effectivity: All) *(Continued)*

NO POWER TO ENGINE START RELAY COIL (1)

STEP 2	Is there battery voltage at pin "X1" in field and sense relay connector? (See Note 2.)	**YES**	Check the relay coil ground. If open, restore continuity, if not, perform a field and sense relay check as outlined in Chapter 24-30-00. If the check is good proceed to Step 3, if not replace the relay.
	NO		
STEP 3	Is there battery voltage at pin "2C" in the field and sense relay connector? (See Note 2.)	**YES**	Locate and correct break in circuit between pin "X1" in the field and sense relay connector and the logic relay panel connector.
	NO		
STEP 4	Is there battery voltage at pin 25(RH), 13(LH) of the logic relay panel connector? (See Note 2.)	**YES**	Correct the open circuit between logic relay panel connector and pin "2C" in the field and sense relay connector.
	NO		
STEP 5	Is there continuity between pin 25(RH), 13(LH) in logic relay panel connector and pin "2B" in the ignition isolation relay connector? (See Note 2.)	**YES**	For troubleshooting with the ignition/start switch in the ON position proceed to Step 12. For the STARTER ONLY position proceed to Step 6.
	NO		
	Locate and correct wiring fault inside logic relay panel.		
STEP 6	Is there battery voltage at pin 9(LH), 21(RH) of logic relay panel connector?	**YES**	Proceed to Step 7.
	NO		
	Proceed to Step 11.		
STEP 7	Is there continuity to the tail grounding block from pins 11(LH), 23(RH) of logic relay panel connector?	**YES**	Proceed to Step 8.
	NO		
	Restore continuity to ground.		
STEP 8	Is there continuity between pin 11(LH), 23(RH) in logic relay panel connector and pin "X2" in the ignition isolation relay connector?	**YES**	Proceed to Step 9
	NO		
	Restore continuity to ignition isolation relay coil ground.		

CHART 3
TROUBLESHOOTING - STARTING SYSTEM (Effectivity: All) *(Continued)*

NO POWER TO ENGINE START RELAY COIL (1)

STEP 9	Is there continuity between pin 9(LH), 21(RH) in logic relay panel connector and pin "X1" in the ignition isolation relay connector?	**YES**	Proceed to Step 10.
	NO Restore continuity to the connection inside logic relay panel.		
STEP 10	Is there continuity between pin 9(LH), 21(RH) in the logic relay panel connector and pin "1B" in the ignition isolation relay connector?	**YES**	Replace the faulty ignition isolation relay.
	NO Restore continuity to the connection inside logic relay panel.		
STEP 11	Is there battery voltage at pin 20(RH), 25(LH) of the center subpanel connector?	**YES**	Correct open circuit between ignition/start switch connector and logic relay panel connector.
	NO Proceed to Step 15.		
STEP 12	Is there continuity between pin 8(LH), 20(RH) in logic relay panel connector and pin "3B" in the ignition isolation relay connector?	**YES**	Proceed to Step 13.
	NO Restore continuity to the connection inside logic relay panel.		
STEP 13	Is there continuity between pins 8(LH), 20(RH) in logic relay panel connector and pins 24(LH), 19(RH) in center subpanel connector?	**YES**	Proceed to Step 14.
	NO Locate break and restore continuity to the circuit.		
STEP 14	Is there continuity between pins 24 and 26 (LH), or pins 21 and 19 (RH) in the center subpanel connector? (ON position)	**YES**	System should be working properly with the ignition/start switch in the ON position, if not repeat steps until problem is identified.
	NO Problem is either a faulty connection at the ignition/start switch or a faulty ignition/start switch. Replace or repair as required.		

CHART 3
TROUBLESHOOTING - STARTING SYSTEM (Effectivity: All) *(Continued)*

NO POWER TO ENGINE START RELAY COIL (1)

STEP 15	Is there continuity between pins 25 and 26 (LH), or pins 21 and 20 (RH) in the center subpanel connector? (STARTER ONLY position)	**YES**	System should be working properly with the ignition/start switch in the STARTER ONLY position, if not repeat steps until problem is identified.
	NO		
	Problem is either a faulty connection at the ignition/start switch or a faulty ignition/start switch. Replace or repair as required.		

1. Voltage checks should be made with the ignition/start switch in the specified position and the battery switch ON. Continuity checks should be made with power removed from the circuit being checked.

2. Perform the check for both ignition/start switch positions; ON and STARTER ONLY.

GENERAL - MAINTENANCE PRACTICES (Effectivity: All)

Refer to Chapter 24-30-00 for starter/generator removal and installation, brush replacement, and field and sense relay removal and installation. Refer to Chapter 24-50-00 for logic relay panel removal and installation.