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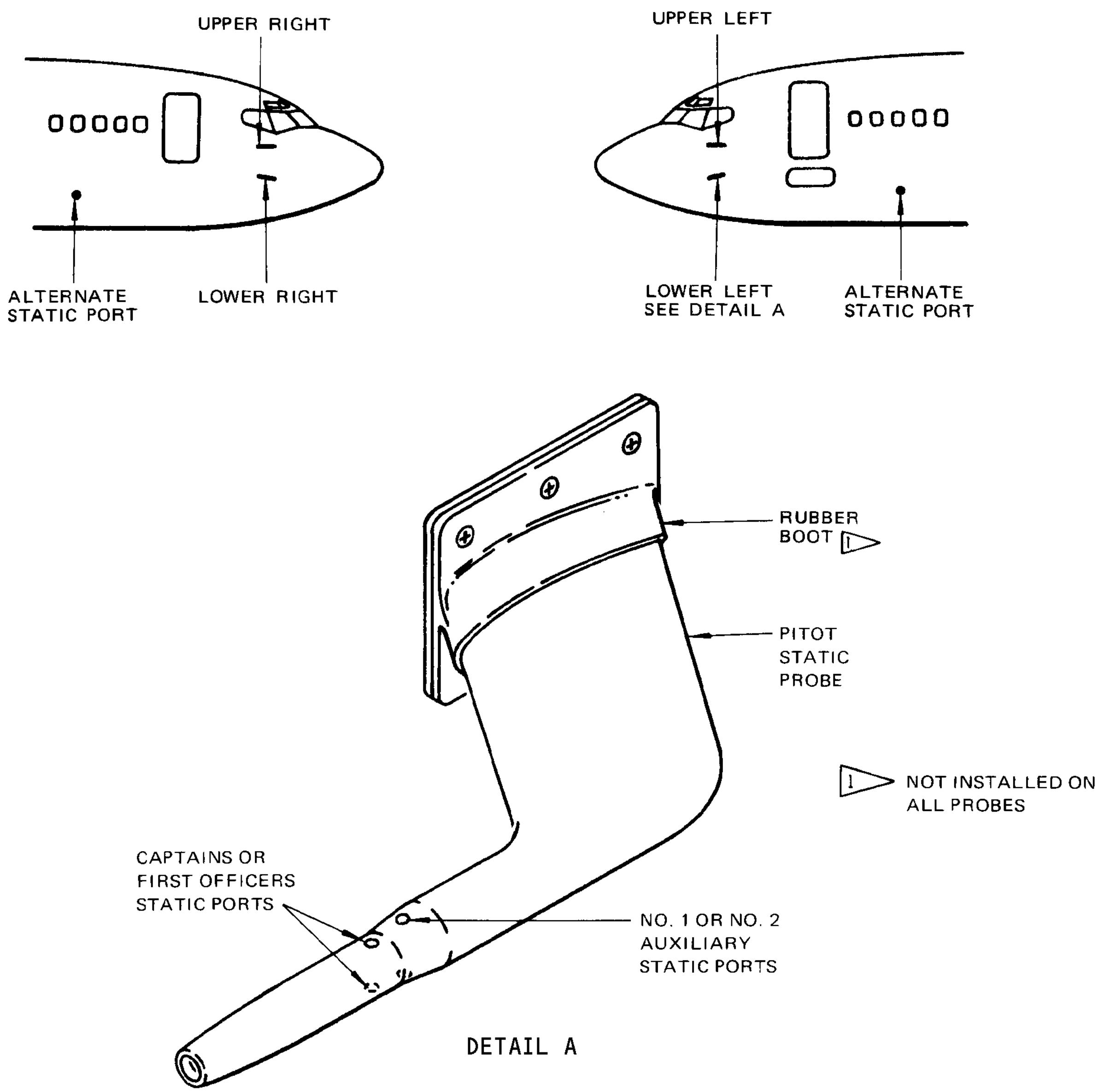
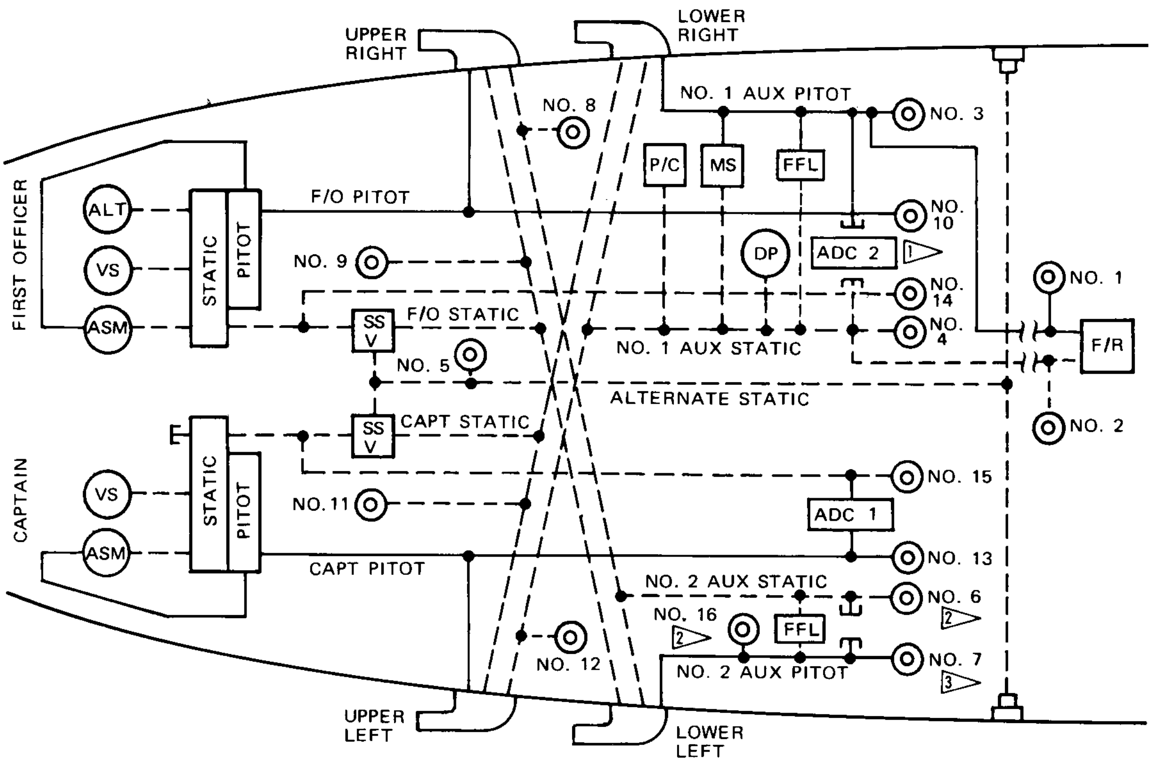
PITOT STATIC SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The pitot static system consists of four combined pitot static probes, two alternate static ports, and connecting tubing to air data instruments and drain trap and test fittings (Fig. 1 and 2). These instruments measure dynamic (pitot) and ambient (static) air pressures to determine airplane vertical speed, airspeed, mach number and altitude.
RG B. Pitot and/or static pressure is supplied to one or more of the following
RG instruments and equipment; airplane instantaneous vertical speed/rate of
RG climb indicator, altimeter, air data computer, flight recorder,
RG mach/airspeed warning switch, combined mach/airspeed indicator, cabin
RG altimeter and differential pressure indicator, air conditioning
RG pressurization control and performance management computer.
   C. Heater circuits are provided for anti-icing of the pitot static probes (Ref Chapter 30, Pitot Static Tubes and Temperature Probe Anti-Icing System).
2. Pitot System
   A. Four pitot tubes are used to sense dynamic air pressure. Two probes are mounted on each side of the fuselage below control cabin window No. 3. The captain's pitot source is from the upper left pitot static probe and the first officer's from the upper right pitot static probe. These pitot pressures feed their respective combined mach/airspeed indicators.
   B. Auxiliary No. 1 and No. 2 pitot pressures from the lower right and lower left pitot static probe feed the various indicators and equipment as shown in Fig. 2.
3. Static System
   A. Static or ambient pressures are sensed through two ports on the combined pitot static probe. Each static port is connected to a port on the opposite side of the airplane as shown in Fig. 2. Static pressure is also available from alternate static ports located at station 406 on both sides of the airplane. The four static pressure systems feed indicators and equipment as shown in Fig. 2.
   B. The static source selector valve provides a second source of static pressure for the captain's and first officer's altimeter, instantaneous vertical speed indicator and mach/airspeed indicator. Each valve is controlled by the applicable static source selector switches located on the captain's or first officer's side panel. Static air pressure is used from the appropriate static source when the switch is in NORMAL position and from the alternate static source when the switch is in ALTERNATE position.
   C. Provisions for a tail-cone static test system are installed in the empennage assembly.



Pitot Static System Component Location
Figure 1



INSTRUMENTS

- ASM — AIRSPEED / MACHMETER
- ALT — ALTIMETER
- VS — INSTANTANEOUS VERTICAL SPEED INDICATOR
- DP — CABIN ALTIMETER AND DIFFERENTIAL PRESSURE INDICATOR

DRAIN TRAP AND TEST FITTING LOCATION (FACING FORWARD)

NO. 1 AND NO. 2 – OVERHEAD AND FORWARD OF FLIGHT RECORDER

NO. 3, NO. 4, NO. 5, NO. 10, AND NO. 14 – RIGHT SIDE OF E1 RACK

NO. 6, NO. 7, NO. 13, AND NO. 15 – LEFT SIDE OF E1 RACK

NO. 8, AND NO. 9 – RIGHT FORWARD END OF NOSE WHEEL WELL

NO. 11, NO. 12, AND NO. 16 – LEFT FORWARD END OF NOSE WHEEL WELL

1 PROVISIONS ONLY FOR ADC NO. 2

2 *Passenger/Cargo Convertible Airplanes*

3 *Passenger airplanes*

EQUIPMENT

- P/C — PRESSURE CONTROLLER
- MS — AIRSPEED MACH WARNING SWITCH
- ADC — AIR DATA COMPUTER
- SS V — STATIC SOURCE SELECTOR VALVE
- F/R — FLIGHT RECORDER
- FFL — FLAP LOAD LIMITER AIRSPEED SWITCH 2
- STATIC PORT
- CAPPED PITOT OR STATIC LINE
- COMBINED PITOT STATIC PROBE
- DRAIN FITTINGS
- – – – – – STATIC LINE
- ———— PITOT LINE

Pitot Static System Schematic
Figure 2

4. Pitot Static Probe

A. The pitot static probe is designed to accurately sense pitot (total) and static (altitude) pressure over the complete operating range of the airplane. The pitot static probe consists of a probe and pressure sensing ports. A strut projects the probe several inches out from the fuselage and isolates the probe from the effects of fuselage variance. A baseplate contains the electrical and pressure fittings. Dow pins are also located on the baseplate to ensure accurate alignment of the installed probe.

B. Pitot pressure is sensed at an opening in the forward end of the probe. Static pressure ports are located in the contoured midsection of the probe; the contoured surface provides aerodynamic compensation to correct the static pressure position error caused by the airplane. The contour configuration and port locations minimize errors due to mach number and angle of attack.

C. An anti-icing heater is installed in the probe to prevent ice accumulation on the probe during flight. The heater is electrically connected to two terminal pins shielded in the baseplate of the probe. The heater is self-regulating coaxial type. The temperature-compensating characteristic provides maximum power for in-flight operation, while automatically reducing the power in still-air conditions to prevent overheating. If the heater is energized at rated voltage during ground operation, the pitot static probe will become hot, and those areas which require high heat concentration may become discolored. This discoloration is normal and has no adverse effect on the unit. Maximum performance, accuracy and life will be obtained from the pitot static probe by minimum amount of operation of the anti-icing heater in still air.

D. Do not expose the pitot static probe to contaminants; avoid solvents, lubricating oil and grease. Do not insert any object into the static ports, which could flare out the edge of the ports. If necessary to use any fluids on the pitot static probe, use fluids such as water that evaporate without leaving harmful residue.

PITOT STATIC SYSTEM - MAINTENANCE PRACTICES

1. General
   A. This section consists of procedures for flushing and draining the pitot static system. Flushing removes or assures the absence of any foreign matter in the pitot static system. Draining removes or assures the absence of any liquid collected in the pitot static system.
2. Pitot Static System Flushing
   A. Flushing the system and checking system leakage tolerance of the pitot static systems may be carried out at the same time when deemed necessary. Flushing before every system leakage check is not normally required. Replacement of pitot static instruments or seal caps does not require flushing of the pitot static system.
   B. Equipment and Materials
      (1) Dry air pressure source - 0 to 15 pounds per square inch (gage) with filter and gage for measuring pressure
   C. Prepare to Flush Pitot Static System
      (1) Disconnect following hoses or tubes from the pitot static system, cap each hose or tube, remove quick-disconnect hose fittings and restrain all hoses to prevent whipping.
         (a) Disconnect hoses to captain's and first officer's instantaneous vertical speed indicators, combined standby altimeter/airspeed indicator and cabin altimeter and differential pressure indicator.
         (b) In electronic compartment disconnect hoses to air data computers, pressure controller and flap load limiter airspeed switches.
         (c) Disconnect hoses from pitot and static system drain fittings and alternate static ports.
         (d) Remove the four pitot static probes (Ref 34-11-11, Removal/Installation). Disconnect pitot and static hoses from all four pitot tubes.
   D. Flush Pitot Static System
      (1) Flush pitot and static lines with dry filtered air (not exceeding 15 psig) for 3 minutes. Uncap each tube or hose and flush and recap in the order indicated in following steps.
         (a) Flush captain's pitot system from capped T located behind instrument panel on outboard side to:
            1) Pitot hose disconnected from mach/airspeed indicator.
            2) Pitot hose disconnected from air data computer No. 1.
            3) Pitot hose disconnected from pitot static probe (upper left).

4) Pitot line disconnected from drain fitting No. 13 located at left forward end of E1 rack.

(b) Flush first officer's pitot system from capped T located behind first officer's instrument panel on outboard side to:
1) Pitot hose disconnected from mach/airspeed indicator.
2) Pitot hose disconnected from pitot static probe (upper right).
3) Pitot line disconnected from drain fitting No. 10 located at right forward end of E1 rack.

(c) Flush auxiliary No. 1 pitot system from pitot hose disconnected from pitot static probe (lower right) to:
1) Pitot hose provisions for air data computer No. 2.
2) Pitot hose disconnected from mach/airspeed warning switch.
3) Pitot hose disconnected from flap load limiter airspeed switch No. 1 (Cargo Airplanes only).
4) Pitot line disconnected from drain fitting No. 3 located on right side of E1 rack.
5) Pitot hose disconnected from flight recorder.
6) Pitot line disconnected from drain fitting No. 1 located in front of flight recorder.

(d) Flush auxiliary No. 2 pitot system from pitot hose disconnected from pitot static probe (lower left) to:
1) Pitot hose disconnected from flap load limiter airspeed switch No. 2 (Cargo Airplanes only).
2) Pitot line disconnected from drain fitting No. 7 located on left side of E1 rack (Passenger Airplanes only).
3) Pitot line disconnected from drain fitting No. 16 located on left forward end of nose wheel well.

(e) Flush captain's static system from capped end of static manifold located forward of captain's instrument panel on inboard side to:
1) Static hose provisions for altimeter.
2) Static hose disconnected from instantaneous vertical speed indicator.
3) Static hose disconnected from mach/airspeed indicator.
4) Static hose disconnected from air data computer No. 1.
5) Disconnect alternate static system tubing from captain's static source selector valve. Place selector valve in ALTERNATE position. Flush and return selector valve to NORMAL. Do not reconnect tubing at this time.
6) Static line disconnected from drain fitting No. 11 located at forward end of nose wheel well.
7) Static line disconnected from drain fitting No. 15 located at left forward end of E1 rack.

(f) Flush captain's static system from captain's static hose disconnected from upper left pitot static probe to:
   1) Static line disconnected from drain fitting No. 11 located at forward end of nose wheel well.
   2) Static line disconnected from lower right static probe.

(g) Flush first officer's static system from capped end of static manifold located forward of first officer's instrument panel on inboard side to:
   1) Static hose disconnected from altimeter.
   2) Static hose disconnected from instantaneous vertical speed indicator.
   3) Static hose disconnected from mach/airspeed indicator.
   4) Disconnect alternate static system tubing from first officer's static source selector valve. Place selector valve in ALTERNATE position. Flush and return selector valve to NORMAL. Do not reconnect tubing at this time.
   5) Static line disconnected from drain fitting No. 9 located at forward end of nose wheel well.
   6) Static line disconnected from drain fitting No. 14 located at right side of El rack.

(h) Flush first officer's static system from first officer's static hose disconnected from upper right pitot static probe to:
   1) Static line disconnected from drain fitting No. 9.
   2) Static line disconnected from lower left pitot static probe.

(i) Flush auxiliary No. 1 static system from port on manifold located in electronics compartment forward right side to:
   1) Static hose disconnected from cabin altimeter and differential pressure indicator in P5 panel.
   2) Static hose disconnected from mach/airspeed warning switch.
   3) Static hose disconnected from air conditioning controller.
   4) Static hose provisions for air data computer No. 2.
   5) Static hose disconnected from flap load limiter airspeed switch No. 1 (Cargo Airplanes only).
   6) Static line disconnected from drain fitting No. 4 located on right side of El rack.
   7) Static hose disconnected from flight recorder.
   8) Static line disconnected from drain fitting No. 2 located forward of flight recorder.
   9) Static line disconnected from drain fitting No. 12 located at forward end of nose wheel well.

(j) Static hose disconnected from right pitot static probe.

(k) Flush auxiliary No. 1 static system from upper left pitot static probe to:
   1) Static line disconnected from lower right pitot static probe.

2) Static line disconnected from drain fitting No. 12 located at forward end of nose wheel well.

(l) Flush auxiliary No. 2 static system from static line disconnected from upper right pitot static probe to:

1) Static hose disconnected from flap load limiter airspeed switch No. 2 (Cargo Airplanes only).
2) Static line disconnected from drain fitting No. 6 located on left side of E1 rack (Passenger Airplanes only).
3) Static hose disconnected from lower left pitot static probe.
4) Static line disconnected from drain fitting No. 8 located at the forward end of nose wheel well.

(m) Flush alternate static system from line disconnected from captain's static source selector valve to:

(n) Static line disconnected from first officer's static source selector valve. Do not allow any dirt to be blown into selector valve. Reconnect line to first officer's static source selector valve.

(o) Static line disconnected from drain fitting No. 5 located on right side of E1 rack.

1) Reconnect line to captain's static source selector valve.

(p) Flush alternate static system from right side static port to:

1) Static line disconnected from left side static port.
2) Static line disconnected from drain fitting No. 5 located on right side of El rack.

E. Restore Pitot Static System to Normal

(1) Connect all pitot and static hoses or lines that were disconnected in step 2.C. to their respective components.
(2) Ensure that no deposit or roughness exists in or about the static ports and pitot probe apertures.
(3) Perform a leakage test of all systems (Ref 34-11-0, Adjustment/Test).

3. Pitot Static System Draining

A. General

(1) The pitot static system should be drained periodically per 737 Maintenance Planning Procedure Document, or when airplane has sustained ground exposure to very heavy rainfall, such as monsoonal or tropical thundersquall type rains.

B. Drain System

(1) Locate each drain fitting as shown in Fig. 2. Operate each as described in the following paragraph:

NOTE: If temperature is below freezing, turn on probe heater for 5 minutes and if necessary, warm drain line and fittings before draining.

(2) Unlock and remove drain cap on lower body of drain by upward pressure and twisting to release cap from the bayonet pins. Invert drain cap and insert the raised, hexagonal shaped portion of cap into opening now exposed in bottom of drain body. Pressing cap into drain body with hand pressure opens a spring-loaded seal and allows any collected liquid in the sump to exit, by gravity flow, from pitot-static system. Remove cap, invert it, and lock it onto base of the drain as it was originally.

PITOT STATIC SYSTEM - ADJUSTMENT/TEST

1. Pitot-Static System Test
   A. General
      (1) Make sure that the ATC transponders are not in an altitude reporting mode.

         WARNING: MAKE SURE THAT THE ATC TRANSPONDERS ARE NOT IN AN ALTITUDE REPORTING MODE WHEN YOU SIMULATE ALTITUDE. IF YOU DO NOT, YOU CAN ACCIDENTALLY CAUSE FALSE TCAS TARGETS.

      (2) The low-range leakage test should be performed when a connection (other than a quick disconnect) in the pitot-static system has been broken to permit the replacement of instrument, hoses or tubing. Quick-disconnects, when reconnected, do not require a leak check. Make sure the acutation ring is fully engaged on the lockpins, and make sure you see the colored locked ring indicator that shows a correct connection of the fitting. If after an apparent satisfactory low-range test, the behavior of the instruments is still such as to suggest leakage, a full-range test should be performed.
      (3) The full-range test must be performed after flushing procedure or each major airframe overhaul. Flushing before every system leakage check is not normally required. Both flushing to remove foreign matter and system leakage checks may be carried out at the same time when deemed necessary.

         CAUTION: THE FOLLOWING STEPS ARE REQUIRED TO PREVENT INSTRUMENT DAMAGE, AND TO ENSURE VALID READINGS.

      (4) Any application or release of vacuum or pressure should be made at a rate of climb or descent less than 5000 feet per minute for the static system and 300 knots per minute for the pitot system (appropriate gage saver restrictors should be used).
      (5) Pressure in pitot lines should always be greater than, or equal to pressure in static lines. Differential pressure should never exceed 10.00 inches of mercury.
      (6) The absolute pressure applied to the static system should never exceed the ambient absolute pressure when any instrument is connected to the static system.
      (7) The captain's, first officer's and alternate static systems must be exposed to the same absolute pressure conditions simultaneously during any leakage test unless otherwise instructed.
   B. Equipment and Materials
      (1) Dry air pressure source - 0-5 inches of mercury
      (2) Vacuum source - 0-20 inches of mercury

(3) Adapters
 (a) Static Port Test Adapter Assembly - F72823-23, or F72823-40
 (b) Pitot-Static Probe Adapter - P/N PSS22300-3-4-4, NAV AIDS USA INC.
(4) Gages
 (a) Pitot System Gage - 0 to 5 inches of mercury with accuracy of ± 0.16 (readable to ± 0.03) inches of mercury or ± 5 (readable to ± 1) knots
 (b) Static System Gage - 0 to 20 inches of mercury with accuracy of 0.10 (readable to ± 0.01) inch mercury or ± 200 (readable to ± 20) feet

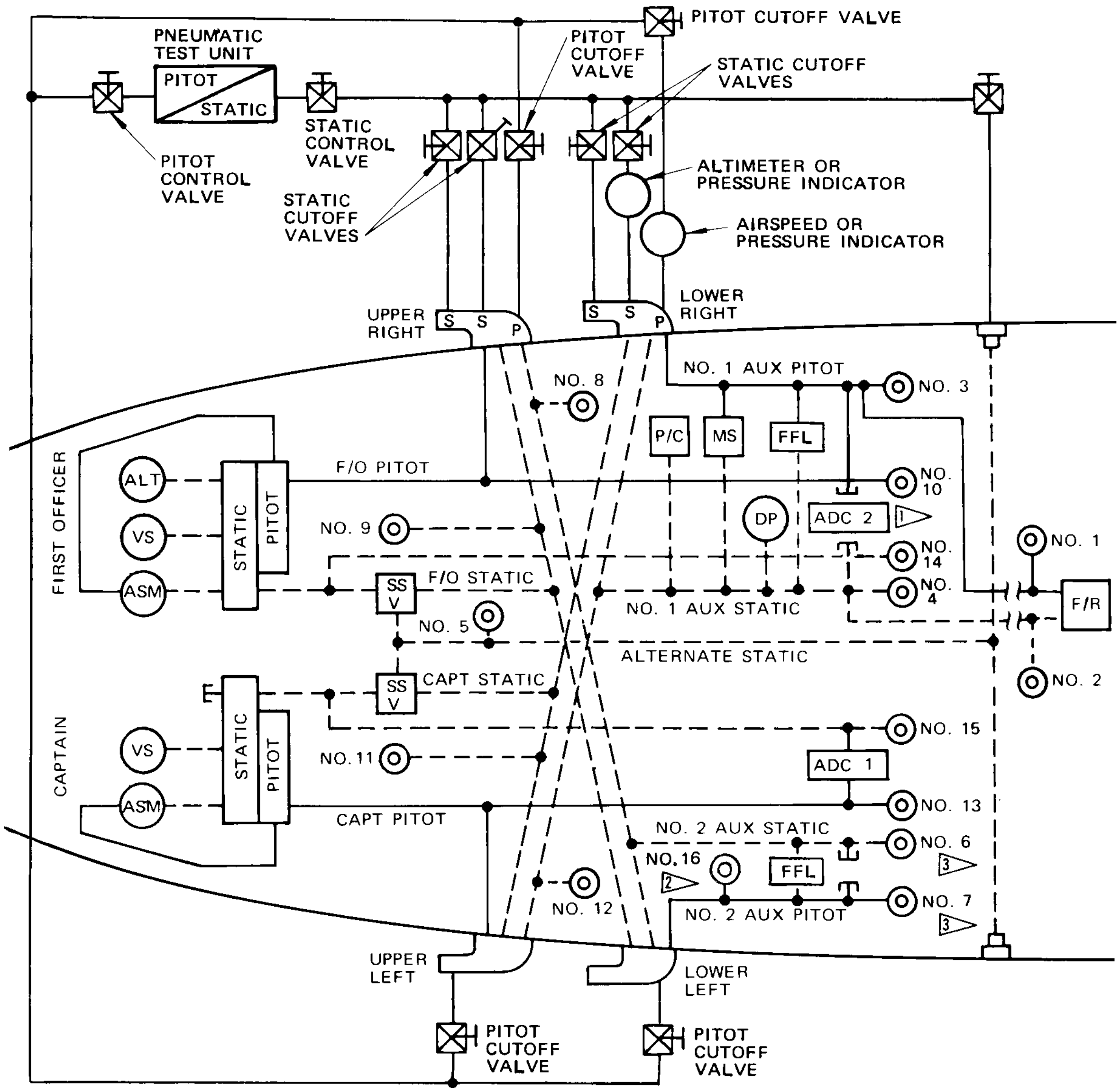
C. Prepare to Test Pitot-Static System
(1) The autopilot system should be off during this test. The flight recorder (where applicable) should be either operating or have the magazine removed and the recorder pressure bar placed in the up position.
(2) Tests should be run with drain caps connected to drain fittings.

NOTE: To test poppet valves in drains, run test with drain caps off drain fittings. Leakage test must then be rerun with drain caps connected.

(3) Pitot-static probe heaters are to remain off during this test.
(4) On airplanes with flap load relief system, open FLAP LOAD RELIEF circuit breaker on P6 panel.

WARNING: FAILURE TO THE OPEN TE FLAP LOAD RELIEF CIRCUIT BREAKER RESULT IN FLAP MOVEMENT AND INJURY TO PERSONNEL.

(5) Connect test equipment (Fig. 501) as follows:
 (a) Connect static probe adapters and cutoff valves to those static systems to be tested. Connect an external altimeter or pressure indicator between static cutoff valve and airplane system.
 (b) Connect static control valve between cutoff valves and static test unit.
 (c) Connect pitot probe adapters and cutoff valves to those pitot systems to be tested. Connect an external airspeed indicator or pressure indicator between the appropriate cutoff valve and the airplane system.
 (d) Connect pitot control valve between cutoff valves and pitot test unit.



Pitot Static System Test Connections
Figure 501 (Sheet 1)

INSTRUMENTS

- ASM AIRSPEED / MACHMETER
- ALT ALTIMETER
- VS INSTANTANEOUS VERTICAL SPEED INDICATOR
- DP CABIN ALTIMETER AND DIFFERENTIAL PRESSURE INDICATOR

EQUIPMENT

- P/C PRESSURE CONTROLLER
- MS AIRSPEED MACH WARNING SWITCH
- ADC AIR DATA COMPUTER
- SS V STATIC SOURCE SELECTOR VALVE
- F/R FLIGHT RECORDER
- FFL FLAP LOAD LIMITER AIRSPEED SWITCH [2]
- STATIC PORT
- CAPPED PITOT OR STATIC LINE
- COMBINED PITOT STATIC PROBE
- DRAIN FITTINGS
- STATIC LINE
- PITOT LINE

DRAIN TRAP AND TEST FITTING LOCATION (FACING FORWARD)

NO. 1 AND NO. 2 – OVERHEAD AND FORWARD OF FLIGHT RECORDER

NO. 3, NO. 4, NO. 5, NO. 10, AND NO. 14 – RIGHT SIDE OF E1 RACK

NO. 6, NO. 7, NO. 13, AND NO. 15 – LEFT SIDE OF E1 RACK

NO. 8 AND NO. 9 – RIGHT FORWARD END OF NOSE WHEEL WELL

NO. 11, NO. 12, AND NO. 16 – LEFT FORWARD END OF NOSE WHEEL WELL

[1] PROVISIONS ONLY FOR ADC NO. 2

[2] *Passenger/Cargo Convertible Airplanes*

[3] *Passenger Airplanes*

Pitot Static System Test Connections
Figure 501 (Sheet 2)

D. Test Pitot Systems
(1) Test captain's pitot system for leakage, observing the cautions in previous steps above.

NOTE: Accuracy of reading and recording pressure should be ± 0.03 inch of mercury or ± 1 knot.

(a) Temporarily seal pressure chamber drain hole in captain's pitot-static probe (upper left probe). Connect air pressure supply to pitot-static probe (pitot pressure portion).
(b) For low-range leakage test:
1) Apply pressure of 250 knots on captain's airspeed/mach indicator or test indicator.
2) After system has stabilized, cut off pressure between airplane system and pressure source. Read and record airspeed/mach indicator or test indicator reading.
3) Wait 1 minute. Make sure that pressure does not decrease more than 2.5 knots.
(c) For full-range leakage test:
1) Apply pressure of 4.53 ±0.16 inches of mercury (gage) or 300 ±5 knots on captain's airspeed/mach indicator or pressure indicator.
2) After system has stabilized, cut off pressure between airplane system and pressure source. Read and record airspeed/mach indicator or test indicator.
3) Wait 1 minute. Pressure should not decrease more than 0.16 inches of mercury or 5 knots of airspeed.
(2) Test first officer's pitot system for leakage.
(a) Repeat steps D.(1)(a) thru (c) above on the upper right pitot-static probe.
(3) Test auxiliary No. 1 or 2 pitot system for leakage.
(a) Repeat steps D.(1)(a) thru (c) on lower right probe (No. 1 system) or lower left probe (No. 2 system).

E. Test Static Systems
(1) Temporarily seal all static ports to which test adapters have not been applied. Method of sealing should not extend into static ports or deform or alter surface in the area when seals are removed.
(2) Place captain's and first officer's static source selector valves in NORMAL position.

CAUTION: PITOT PRESSURE SHOULD ALWAYS EQUAL OR EXCEED STATIC LINE PRESSURE AND DIFFERENCE (DIFFERENTIAL PRESSURE) SHOULD NOT EXCEED 10.00 INCHES OF MERCURY. DIFFERENTIAL PRESSURE DESCRIBED ABOVE SHOULD NOT FALL BELOW ZERO.

NOTE: Accuracy of reading and recording vacuum pressure or altimeter should be ± 0.01 inch of mercury or ± 20 feet of altitude respectively.

(3) Test captain's static system for leakage observing cautions in previous steps.
   (a) Place all static cutoff valves in open position.
   (b) Apply a vacuum to captain's static system on lower right probe (preferred) or drain fitting No. 11 (connection not shown in Fig. 501).
   (c) For low-range leakage test:
      1) Apply a vacuum to the system, so that captain's altimeter indicates an altitude of 3000 feet.
      2) After system has stabilized, shut off vacuum using appropriate static cutoff valve. Read and record altimeter or test indicator reading.
      3) Wait 1 minute. Make sure that vacuum does not decrease more than 60 feet.
   (d) For full-range leakage test:
      1) Apply a vacuum equivalent to 12.37 ±0.10 inches of mercury (absolute pressure) so that captain's altimeter indicates 22,500 ±200 feet. Pitot system should also be evacuated, if necessary, to meet the requirement under the caution note.
      2) After system has stabilized, shut off vacuum using appropriate static cutoff valve. Read and record altimeter or pressure indicator reading.
      3) Wait 1 minute. Make sure that vacuum does not decrease more than 0.21 inch of mercury or 400 feet in altitude.

         CAUTION: VACUUM SHOULD BE HELD AS A BACKUP BEHIND CUTOFF VALVE DURING CUTOFF PERIOD. FOLLOWING THIS PERIOD, CUTOFF VALVE SHOULD BE REOPENED. RELEASE OF VACUUM SHOULD BE GRADUAL AND SIMULTANEOUS FOR ALL STATIC SYSTEMS.

(4) Test first officer's static system for leakage.
   (a) Place shutoff valve in open position.
   (b) Apply a vacuum to first officer's static system on upper right probe (preferred) or drain fitting No. 9.

(c) Repeat steps above for first officer's static system and altimeter.

(5) Test auxiliary No. 1 static system for leakage.
  (a) Place shutoff valve in open position.
  (b) Apply a vacuum to auxiliary No. 1 static system on lower right probe (preferred) or drain fitting No. 12.
  (c) Repeat steps above using pressure indicator or altimeter.

(6) Test auxiliary No. 2 static system for leakage.
  (a) Place shutoff valve in open position.
  (b) Apply vacuum to upper right probe (preferred) or drain fitting No. 8.
  (c) Repeat steps above using test vacuum indicator and altimeter.

(7) Test for leakage between auxiliary static system No. 1 and captain's static system.
  (a) If captain's altimeter is the electrical type, verify that CADC No. 1 circuit breaker is closed.
  (b) Connect the vacuum source to one of the static ports on the upper left or lower right (preferred) pitot-static probe aft static pressure section or to the drain fitting for auxiliary static system No. 1.
  (c) Seal all ports of captain's static system.
  (d) Apply a vacuum equivalent to 12.37 ±0.10 inches of mercury (absolute pressure) (equivalent to an altitude of 22,500 ±200 feet).
  (e) After the system has stabilized, maintain the vacuum for 5 minutes.
  (f) Check that the captain's altimeter does not show an increase in altitude.

(8) Test for leakage between auxiliary static system No. 2 and first officer's static system.
  (a) If first officer's altimeter is the electrical type, make sure that CADC No. 2 circuit breaker is closed.
  (b) Connect the vacuum source to one of the static ports on the upper right or lower left (preferred) pitot-static probe aft static pressure section, or to the drain fitting for auxiliary static system No. 2.
  (c) Seal all ports of first officer's static system.

(d) Apply a vacuum equivalent to 12.37 ±0.10 inches of mercury (absolute pressure) (equivalent to an altitude of 22,500 ±200 feet).
(e) After the system has stabilized, maintain the vacuum for 5 minutes.
(f) Make sure that the first officer's altimeter does not show an increase in altitude.

(9) Test alternate static system for leakage.
(a) Place shutoff valve in open position.
(b) Apply vacuum to right flush to body mounted static ports (preferred) or drain fitting No. 5.
(c) Repeat steps above using test vacuum indicator and altimeter.
(d) Remove seals and adapters from captain's and first officer's static systems. Make sure that static source selector valve is in NORMAL position.
(e) Repeat step above to make sure no leakage through static source selector valve.

F. Restore Airplane to Normal Configuration

<u>WARNING</u>: FAILURE TO REMOVE THE VINYL ADHESIVE TAPE FROM THE STATIC PORTS BEFORE FLIGHT MAY CAUSE LARGE ERRORS IN AIRSPEED SENSING AND ALTITUDE SENSING SIGNALS WHICH MAY LEAD TO LOSS OF SAFE FLIGHT.

(1) Remove all seals and adapters from all pitot-static probes that were tested. Make sure that no deposits or roughness exists in or about static ports or pitot head apertures.
(2) Remove flush to body mounted adapter from alternate static port and seal if system was tested.
(3) Remove all pitot pressure chamber drain seals.
(4) Replace all drain fitting caps.
(5) Make sure that flight recorder (if installed) is returned to normal operating configuration.

PITOT STATIC PROBE - REMOVAL/INSTALLATION

1. General
   A. The removal/installation of all pitot tubes is similar.
2. Equipment and Materials
   A. Aerodynamic smoother
3. Remove Pitot Static Probe
   A. Hold pitot static probe and remove screws (6) from probe base.
   B. Pull probe away from airplane skin far enough to remove connectors.
   C. Disconnect pitot, static and heater connectors (Fig. 401).
   D. Remove probe, gasket and then shim(s) as required.
   E. Cap pitot and static lines and cover electrical connector with protective cover.
4. Prepare to Install Pitot Static Probe
   A. Check gasket and shim(s) (as required) for aging and wear.
   B. Remove aerodynamic smoother from probe base and external body skin.
5. Install Pitot Static Probe
   A. Install gasket and shim(s) (as necessary) to assure flushness between pitot-static probe and airplane skin within +0.000 or -0.020 inch (SRM 51-70).
   B. Position probe close to airplane skin and connect pitot, static and heater connectors.
   C. Place probe into position and secure to airplane with screws (6).
   D. Measure the resistance between strut of the pitot static probe and the airplane skin with an ohmmmeter.
   E. If the resistance is more than 0.010 ohm, do these steps:
      (1) Remove the pitot-static probe.
      (2) Clean the bonding surfaces, including the countersunk holes in the pitot-static probe (SWPM 20-20-00).
      (3) Replace the existing screws with new screws.
      (4) Re-install the pitot-static probe.
      (5) Measure the resistance between strut of the pitot-static probe and the airplane skin with an ohmmmeter.
      (6) If the resistance is more than 0.010 ohms, do these steps:
         (a) Remove the pitot-static probe.

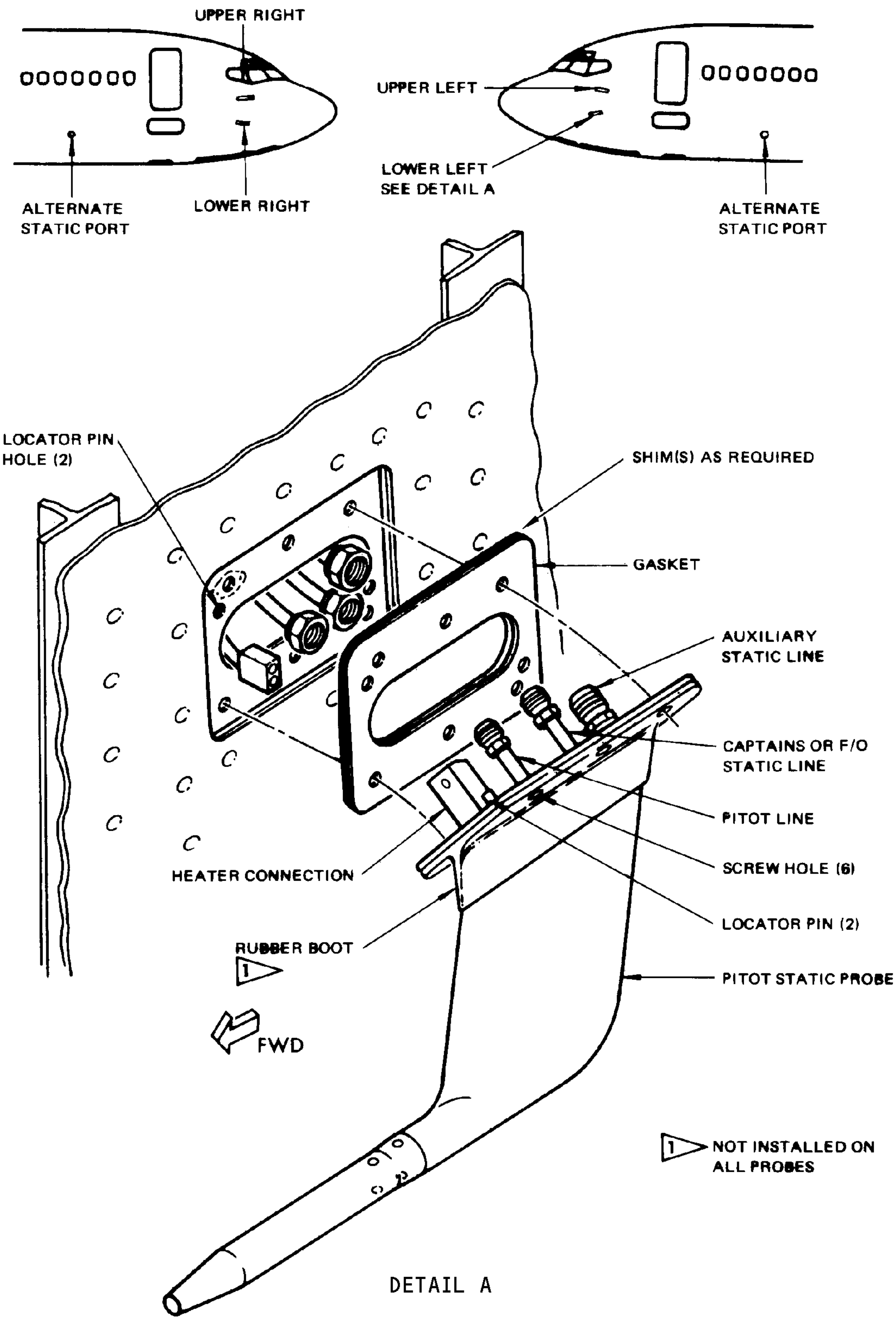
(b) Replace the nutplates and rivets that attach the pitot-static probe (SRM 51-40-02).

(c) Re-install the pitot-static probe and make sure the bonding resistance is not more than 0.010 ohm.

F. Reseal around gasket and airplane skin with aerodynamic smoother.

NOTE: It is not necessary to apply the sealant immediately if the cure time will cause a flight delay. However, you must apply the sealant as soon as possible to keep moisture out of the area between the probe and the airplane skin.

G. Perform a pitot and static test for the replaced probe per 34-11-0 A/T.



Pitot Static Probe Installation
Figure 401

PITOT-STATIC PROBE - INSPECTION/CHECK

1. General
   A. The inspection check visually inspects the pitot static probe for physical damage to the strut, head and precision-contoured surfaces and for foreign material in the static ports, drain hole and pitot opening. The pitot static probe should be replaced if it fails any of the visual inspections (AMM 34-11-11/401).
   B. Fly Back Limits
      (1) Do these steps when you see pitot-static probe deterioration that you think is more than the inspection tolerances:
         (a) You can dispatch the airplane from a base without maintenance facilities until one of these conditions occur:
            1) 24 hours elapse
            2) The airplane gets to a maintenance base.
      (2) Replace the pitot-static probe for one or more of these conditions:
         (a) The flight crew sees a cross panel airspeed difference that is more than stated below:
            1) The airspeed difference must not be more than 3 knots when the airspeed is less than 250 knots and the altitude is below 20,000 feet.
         (b) There is structural damage to the pitot-static probe since the last flight.
2. Equipment and Materials
   A. Assorted gages or wires - 0.004- to 0.020-inch, or Optical Micrometer - Model 966A1, Monocle Industries, Inc., P.O. Box 2426, Coppell, Texas 75019, or other suitable instrument, Pitot-Static Probe Tip Inspection Model 885N, Rosemount Aerospace, Inc., 14300 Judical Road, Burnsville, MN 55306-4898
3. Inspect Pitot Static Probe

   NOTE: When performing the Aerodynamic Contour Condition check or the Pitot Tip Condition check (with or without an inspection kit), it is not necessary to remove the pitot static probe.
   A. General Probe Conditions
      (1) Remove the pitot-static probe (AMM 34-11-11/401).
      (2) Visually inspect the pitot-static probe for physical damage to its precision-contoured surfaces and for foreign material in the static ports, drain hole and pitot opening. To remove the foreign material, refer to AMM 34-11-11/401.
      (3) Check for any detectable bending or twisting of the head or strut sections of the pitot-static probe.
   B. Aerodynamic Contour Condition
      (1) Visually examine the static ports as follows (Fig. 601):
         (a) Make sure that the static port edges are perpendicular to the contoured-surface.

(b) Make sure that the static ports in the probe are not dirty.
(c) Make sure that the static port edges are not damaged.

(2) Visually examne the pitot-static probe for nicks, scratches, or other surfice defects as follows:
(a) Make sure that the depth of the defect is less than 0.015 inches within a radius of 0.5 inches of the static ports.
(b) Make sure that the depth of the defect is less than 0.025 inches on the remaining head section.
(c) Make sure that the depth of the defect is less than 0.125 inches on the strut section.

(3) Replace the pitot-static probe if one of the condition is not permitted (AMM 34-11-11/401).

C. Pitot Tip Condition (WITHOUT PITOT-STATIC PROBE INSPECTION KIT)

(1) The leading edge lip of the pitot opening should be sharp. New tubes are sharpened to a 0.004 +0.002 flat. The tube must be removed if the lip has been eroded through wear, or dented by abuse, to the extent that the lip has a flatness greater than 0.015 inch. Figure 602 shows this condition.

(2) The outer surface of the lip edge must be smooth. It cannot be curled or flared outward. This condition can be detected by sliding your fingernail along the outer surfaces at the tip. The tube should be removed if the lip edge is curled or flared outward. Figure 603 illustrates this condition.

(3) The lip can have small inward dents which affect the roundness of the pitot opening, as shown in Fig. 604. The tube must be removed if the indentation exceeds 0.030 inch from the normal tip diameter. The dent may be at any location around the opening, but must not affect more than 20% of the lip circumference.

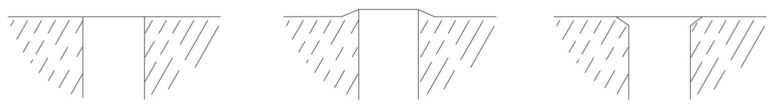
(4) The leading edge lip can have small nicks or chips as shown in Fig. 602. The tube must be replaced if two of the nicks are between 0.025 and 0.035 inch deep. The tube must also be replaced if any one nick exceeds 0.035 inch deep.

(5) Install or replace the pitot static probe (AMM 34-11-11/401).

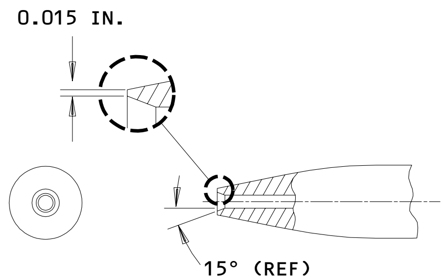
D. Pitot Tip Condition (WITH PITOT-STATIC PROBE INSPECTION KIT)

(1) Examine the pitot tip of the probe with the inspection kit.

(2) Replace the pitot-static probe if the inspection shows an out of tolerance condition (AMM 31-11-11/401).



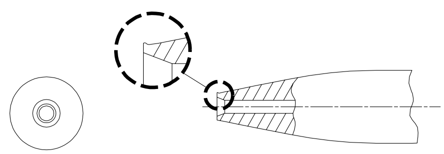
Inspection Criteria for Static Ports
Figure 601



REJECTION CRITERION:

LIP FLATNESS GREATER THAN 0.015 INCH.

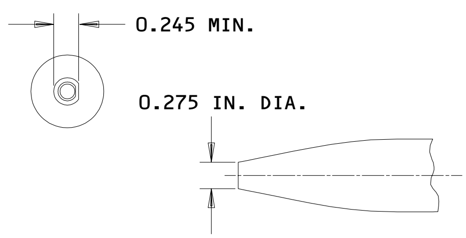
Lip Flatness
Figure 602



REJECTION CRITERION:

DETECTABLE OUTWARD FLARE OR CURL OF LIP.

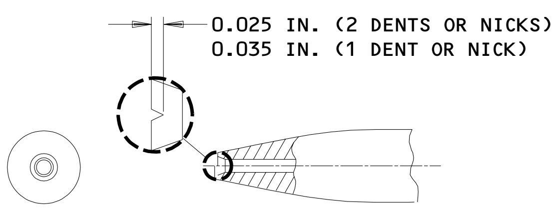
Outward Flare Of Lip
Figure 603



REJECTION CRITERIA:

1. INDENTATION EXCEEDING 0.030 IN. OF NORMAL TIP DIAMETER
2. INDENTATION AFFECTING MORE THAN 20% OF TIP CIRCUMFERENCE.

Indentation Of Pitot Opening
Figure 604



REJECTION CRITERIA:

1. ANY SINGLE NICK EXCEEDING 0.035 IN. DEEP.
2. TWO NICKS BETWEEN 0.025 IN. AND 0.035 IN. DEEP.

Chips Or Nicks In Probe Tip
Figure 602

PITOT STATIC PROBE - CLEANING/PAINTING

1. Cleaning
   A. Internal
      (1) To remove foreign materials lodged in the pitot opening, static ports or drain hole, disconnect the pitot static probe pressure lines and blow compressed dry air through the line and out the ports. Maximum air pressure to be used is 60 psi.
      (2) If the forced air method fails to remove the foreign material, remove the pitot static probe from the airplane (Ref 34-11-11, Pitot Static Probe - R/I). Soak or flush the unit with clean water, drain it and repeat (1).

          CAUTION: DO NOT USE FLUIDS THAT CONTAIN SODIUM CHLORIDE OR SULFUR COMPOUNDS TO CLEAN THE PROBE. SODIUM CHLORIDE AND SULFUR CAN CAUSE PREMATURE FAILURE OF THE PROBE.

      (3) If the forced air and soaking methods do not clean the drain hole because of its small size, cleaning may be accomplished by manually inserting first a 0.025 drill rod and then a 0.029 drill rod into the drain hole.

          CAUTION: CARE MUST BE TAKEN SO THAT THE DRAIN HOLE IS NOT ENLARGED DURING THE CLEANING PROCESS (DRAIN HOLE DIAMETER IS 0.031 +0.001 INCH).

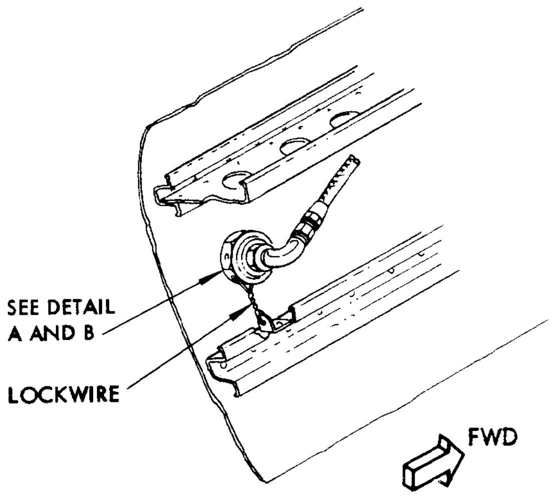
   B. External
      (1) To remove foreign material from external surfaces, wash them with clean water and dry them with a soft cloth.

          CAUTION: DO NOT PROBE INTO THE STATIC PORTS WITH ANY OBJECT. DO NOT EXPOSE PITOT-STATIC TUBES TO SOLVENTS, LUBRICATING OIL AND/OR GREASE. DO NOT SCOUR OR BUFF THE MACHINED CONTOUR SURFACE.

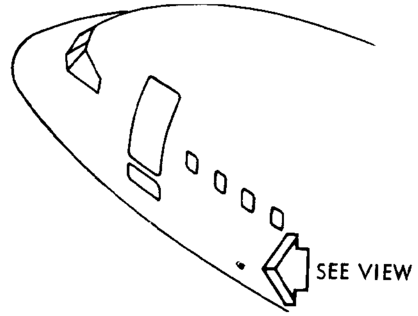
2. Painting
   A. The pitot static probe should not be painted. The data from the pitot static probe can be affected if you paint the probe.

ALTERNATE PRESSURE PORT - REMOVAL/INSTALLATION

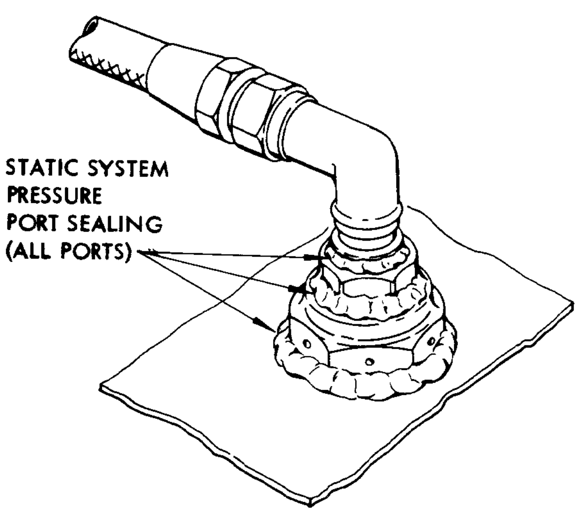
1. Equipment and Materials
   A. High Temperature Fillet Sealant - BMS 5-18A, Type II - PR-810 (or equivalent)
2. Remove Static Pressure Port (Fig. 401)
   A. Remove forward cargo compartment lining to gain access to static pressure ports (Ref Chapter 25, Equipment and Furnishings).
   B. Disconnect static hose from elbow. Cap static hose to prevent entry of foreign material.
   C. Loosen jamnut and remove elbow and O-ring from static pressure port.
   D. Remove lockwire and clean away sealant from around static pressure port.
   E. Remove static pressure port retaining nut and washer.
   F. Remove static pressure port from exterior of airplane.
3. Install Static Pressure Port
   A. Clean away any remaining sealant and prepare area for application of new sealant.
   B. Insert static pressure port in mounting hole. Adjust until face of port is as flush with airplane skin as possible and hold in place.
   C. Install static pressure port retaining nut and washer. Tighten retaining nut to torque value of 250 to 400 inch-pounds. Lockwire nut using double-twist method.
   D. Check port is flush with the skin within +0.003/-0.000 inch after installation. If necessary, shave external surfaces of static pressure port until it is flush with airplane skin within tolerance. After shaving is complete, ensure surface finish on face of static port is 63 RMS maximum. Ensure no burrs or materials are in static pressure port and all inlet holes are free from obstructions or deformation.
   E. After installation, the skin surface, for a distance of 3 inches forward and aft of the port centerline, should be flush within 0.010 inch maximum measured as the clearance between the skin and the edge of a 6-inch straightedge placed horizontally against the skin. All rivets within a 3-inch radius of the port center should be flush with the skin within 0.003 inch maximum.
   F. Alodize static pressure port with Alodine 1000 (Ref Protective Finishes in Chapter 51, Structures - General).
   G. Apply sealant to static pressure port as shown in Fig. 401, Detail A (Ref 51-30-0, Seals and Sealing).
   H. Install elbow and O-ring in static pressure port. Tighten jamnut.
   I. Remove cap from end of static hose and check for any foreign material inside hose, clean if necessary.
   J. Connect static hose and fitting to elbow. Tighten to torque value of 100 to 125 inch-pounds.
   K. Install cargo compartment lining (Ref Chapter 25, Equipment/Furnishings).
   L. Perform an alternate static system test per 34-11-0, Pitot Static System - A/T.



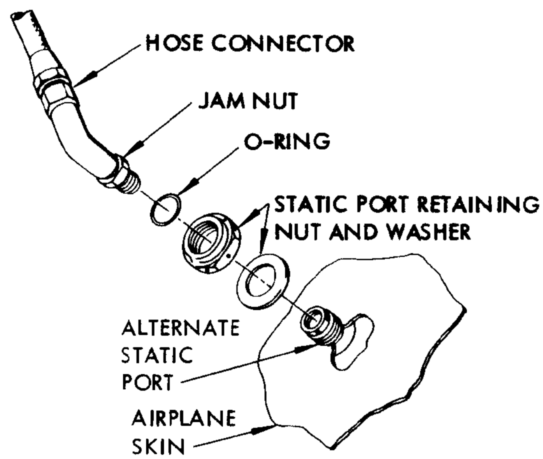
VIEW 1



STATIC PORT
LEFT SIDE SHOWN
RIGHT SIDE SIMILAR



SEALING PLAN
DETAIL A

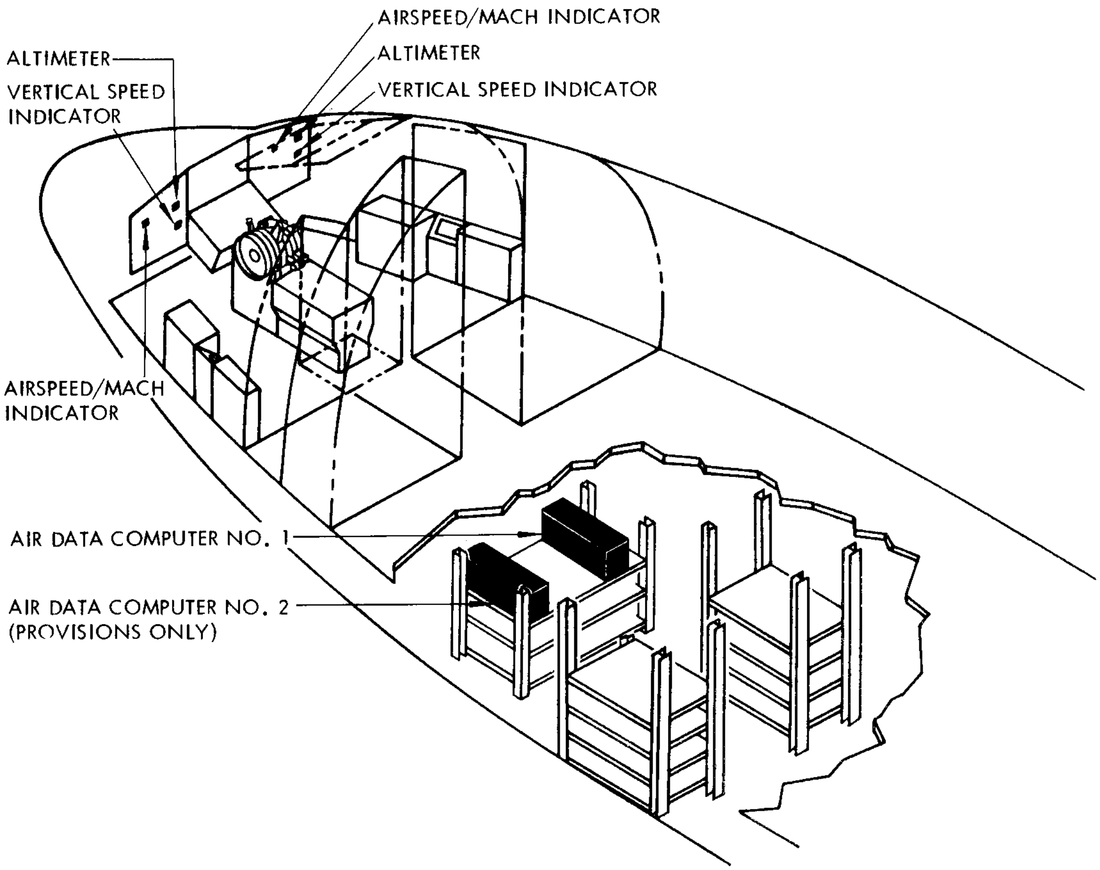


EXPLODED STATIC PORT (TYTPICAL)
DETAIL B

Alternate Static Port Installation
Figure 401

AIR DATA PRESSURE INSTRUMENTS - DESCRIPTION AND OPERATION

1. General
   A. Air data pressure instruments are those which derive their inputs from the pitot static air pressure system and the air data computer system. Refer to Pitot Static System, 34-11-0. The air data pressure instruments are composed of two airspeed/mach indicators, two instantaneous vertical speed/rate of climb indicators, two altimeters, and two air data computers.
   B. Indicated airspeed and mach number are conventionally displayed as a function of differential pitot and static air pressures on mach/airspeed combined instrument indicators.
   C. Instantaneous vertical speed/rate of climb, through a range of 0 to 6000 feet per minute, is indicated according to accelerometer and static air pressure information.
   D. Altitude throughout the range of 1000 feet below mean sea level to 50,000 feet above mean sea level may conventionally be indicated on both captain's and first officer's indicators from static air pressure information.
   E. The air data computer system electrically integrates and computes corrected altitude, mach number, and airspeed signals from pitot static pressure. This data is sent to the ATC transponder, autopilot, flight director, mach trim, and other systems as shown in figure 6. An air data computer functional system check is an integral part of the computer. The power to operate the computer is obtained from circuit breaker panel P18. Provisions are included for the installation of a second air data computer system.
2. Airspeed/Mach Indicator
   A. The airspeed/mach indicator is a combination of two flight instruments which presents an integrated display of indicated airspeed and mach number. The captain and first officer are each provided with an indicator. (See figure 2.)
   B. Indicated airspeed is displayed through the range of 60 to 420 knots. The indicator dial scale is calibrated linearly in two knot increments between 60 and 160 knots, and logarithmically in ten knot increments between 160 and 420 knots. A pointer, rotated by a mechanically connected pressure diaphragm, indicates the airspeed range through a single revolution. Two reference pointers (bugs) are provided on the bezel for pilot convenience in presetting airspeed reminders, such as V1 and V2.
   C. The mach mechanism is actuated by an evacuated capsule whose linkage is such that it rotates a subdial at an angular rate which varies linearly with the logarithm of the static pressure. This subdial is graduated in increments of 0.02 mach and reads from 0.5 to 1.0 mach.



Air Data Pressure Instrument System Component Location
Figure 1

D. The airspeed dial and mach subdial scale factors are selected so that the single pointer shows airspeed on the outer perimeter of the dial and mach number on the subdial scale through a cutout in the main dial adjacent to the high airspeed markings.

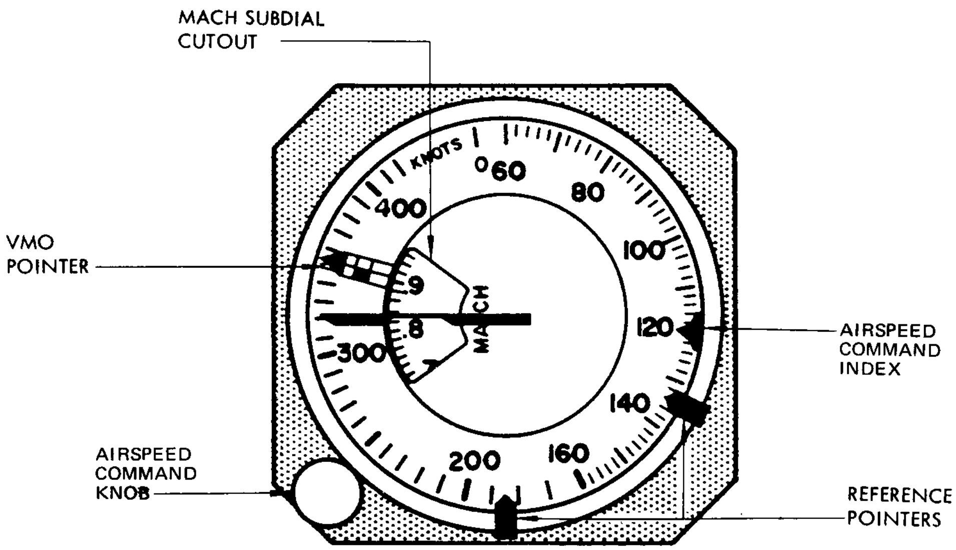
E. A manually operated airspeed command knob rotates the airspeed command index around the outer periphery of the main dial. When the auto throttle system is installed, the airspeed command knob is used to set the airspeed to be maintained. (See 22-31-0.) The VMO pointer locates and accents the maximum operating airspeed of the airplane as a function of altitude.

3. Instantaneous Vertical Speed/Rate of Climb Indicators

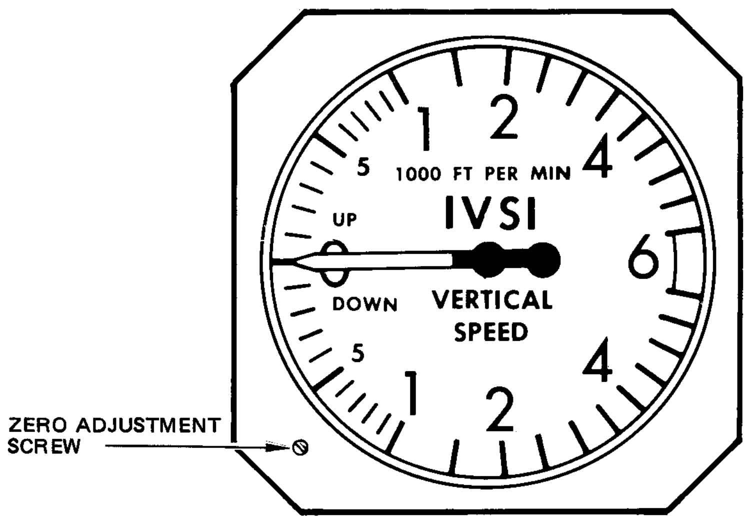
A. Instantaneous vertical speed/rate of climb indicators are provided for both the captain and first officer and are operated by static pressure applied to both the inside of the instrument case through an orifice and to the interior of a flexible diaphragm within the case. As the airplane changes altitude the pressure inside the diaphragm changes more rapidly than that inside the instrument case. The resulting differential causes the diaphragm to expand or contract, and the amount of change is proportional to the rate of change of altitude and is indicated by a radial pointer in thousands of feet per minute. An accelerometer is incorporated in the instrument also, and provides indication of instantaneous vertical speed (Fig. 3).

B. A small screw, installed on the lower left corner of the instrument, is used to adjust the pointer to zero.

NOTE: The adjustment screw range for zeroing the vertical speed indicators is from 400 ft./min. DOWN to 400 ft./min. UP. The instrument cannot be harmed by turning the zero adjustment screw too far in either direction. Recalibration of the IVSI is not necessary after the zero adjustment is accomplished, but if it cannot be zeroed, the instrument must be recalibrated.

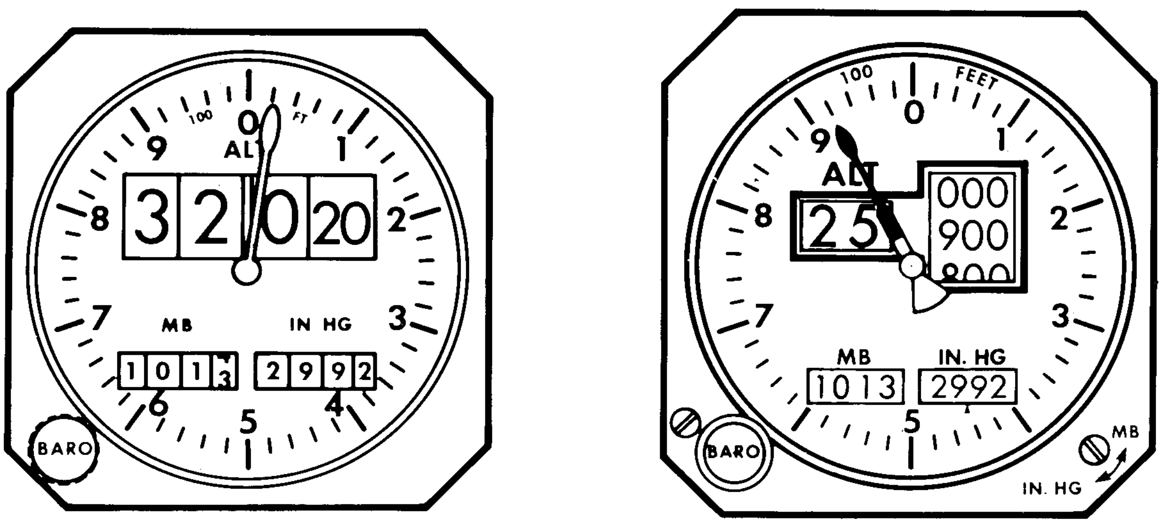


Airspeed/Mach Indicators
Figure 2



Instantaneous Vertical Speed/Rate of Climb Indicators
Figure 3

4. Altimeters
   A. Altimeters are provided for both the captain and first officer. The captain's altimeter is an electric drum-pointer type that receives altitude information from the central air data computer. The altitude is displayed in 1000-, 100-, and 20-foot increments on drum counters and in 100- and 20-foot increments by a pointer which reads against a fixed scale. The range of the altimeter is from ±1000 to 50,000 feet with altitudes below sea level displayed by the rotating pointer and a NEG flag covering the counter drums. A red warning flag will cover the counter drums in the event of a failure. The failure monitor will show the flag in the event of the following: loss of electrical power to the indicator, loss of flag power from the CADC, excessive servo null voltage, loss of input synchro signal and internal indicator malfunction. A barometric pressure setting knob is provided to adjust the pressure setting which is displayed in both millibars and inches of mercury.
   B. The first officer's altimeter is a pneumatic drum-pointer type. The range of the altimeter is from ±1000 to 50,000 feet and is displayed in 1000- and 100-foot increments on drum counters and in 100- and 20-foot increments by a pointer which reads against a fixed scale. A barometric pressure setting knob is provided to adjust the pressure setting which is displayed in both millibars and inches of mercury. Either of the barometric scales may be masked from view by means of a screwdriver slot on the face of the indicator.
   C. Instrument vibrators are contained in each altimeter to remove mechanical linkage friction errors.
5. Data Computer
   A. General
      (1) The air data computer is an electropneumatic unit, which, when supplied with the requisite electrical power, pitot and static pressures and outside air temperature inputs computes outputs for various systems.
      (2) The air data computer consists of individual plug in altitude, mach and airspeed modules, and a chassis assembly. (See figure 5.) The computer front panel has pitot and static pressure input connectors, function test select switch, module failure annunciators, module test switches, master failure monitor, and altitude reporting readout assembly. Mounted on the front panel is an airplane configuration adapter unit.
   B. Computer Modules
      (1) The altitude module receives static pressure information which the altitude E pick-off and altitude module mechanically transduces to altitude outputs. (See figure 6.) These outputs go to the autopilot system, total air temperature/engine pressure ratio limit indicator, (if installed), flight director system, and ATC transponder system. This module also sends altitude information in the form of log Ps to the mach module.



CAPTAIN FIRST OFFICER

Altimeters
Figure 4

(2) The airspeed module receives static and pitot pressure. The indicated airspeed E pick-off is positioned by the difference of the two pressures. This E pick-off signal processed in the airspeed module generates outputs to the autopilot system, and a log Qc signal to the mach transducer.

NOTE: The callout TOTAL, on the computer front panel, refers to the sum of pitot and static pressure; however, for continuity in this manual, the expression pitot pressure will apply for total pressure.

(3) The mach module uses the electrical inputs of Ps and log Qc to compute mach by means of a servomechanical rebalance variable resistor network. True airspeed and static air temperature output signal are derived from processing total air temperature and mach signals. Outputs are sent to the autopilot system electrical mach/airspeed indicators (if installed), and to the TAS/SAT indicator (if installed).

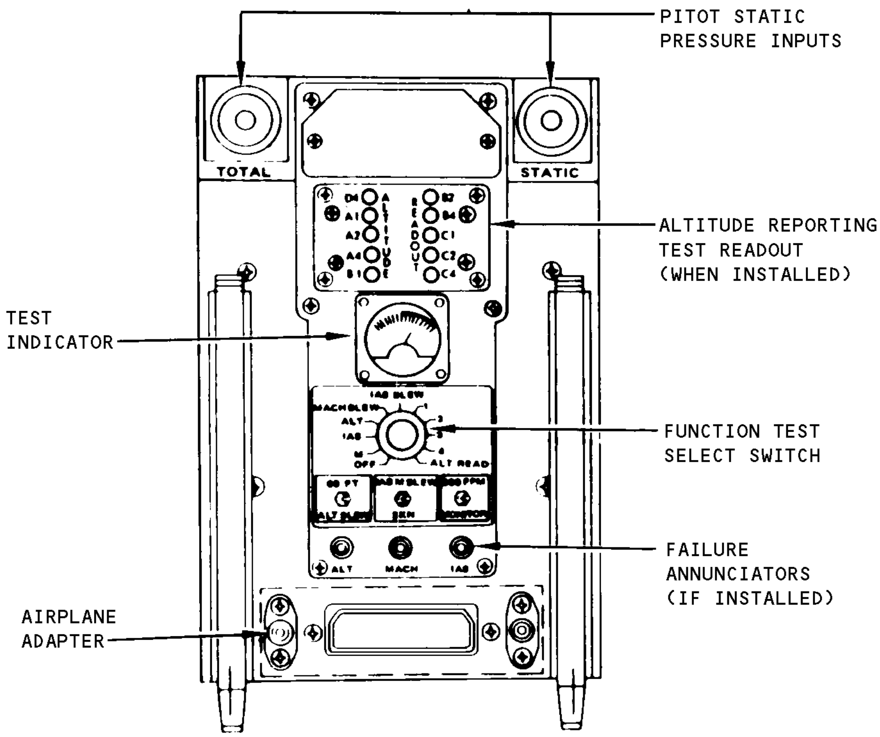
(4) The chassis assembly contains the computer power supply, computer failure, monitoring circuits and module interlock circuits. Three multipin connectors at the rear provide for connection of 115 volts ac, 26 volts ac, 28 volts dc power, and all other inputs and outputs. The chassis also provides the wiring from the external connector to each individual module and the pneumatic plumbing to the altitude and airspeed modules.

C. Computer Failure Monitoring

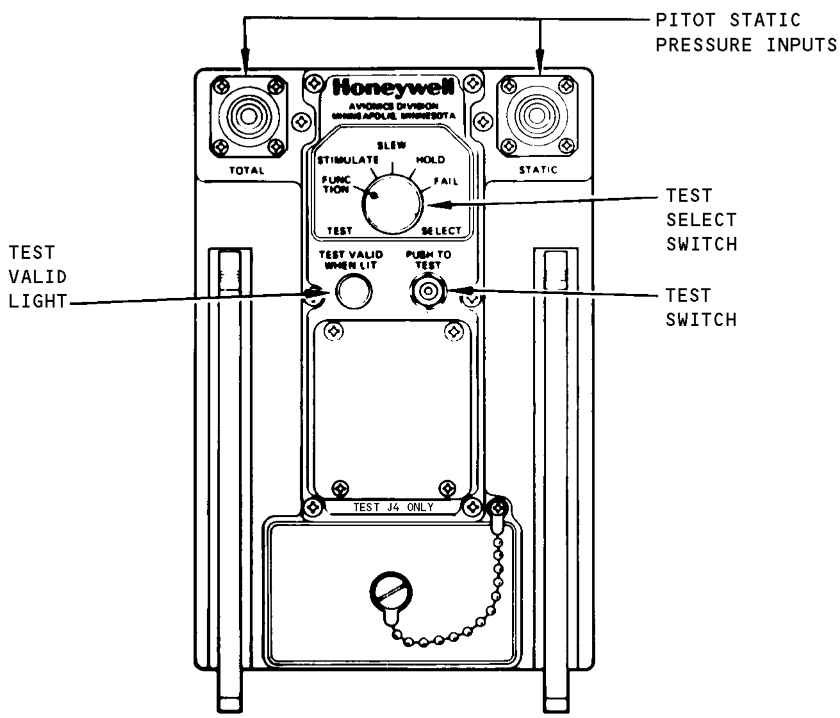
(1) The computer master failure monitor, module failure monitors, and the annunciator lights comprise the computer failure monitoring circuits. Each annunciator light (normally white) is identified with the module it monitors. When a module failure occurs, the annunciator light will illuminate red to indicate a faulty module. Each module failure monitor will also detect the loss of various input voltages or an excessively high signal voltage.

NOTE: The annunciator light will remain on until the reset switch, mounted on the chassis behind the cover, applies a reset voltage to them. The computer cover must be removed to obtain access to this switch. The fault annunciators have proven ineffective and unverified nuisance trips have compromised the credibility of the fault monitors. Action based solely on the fault annunciators can be disregarded.

(2) Each module failure monitor is designed to prevent the annunciator from being actuated when certain computer test configurations are selected.

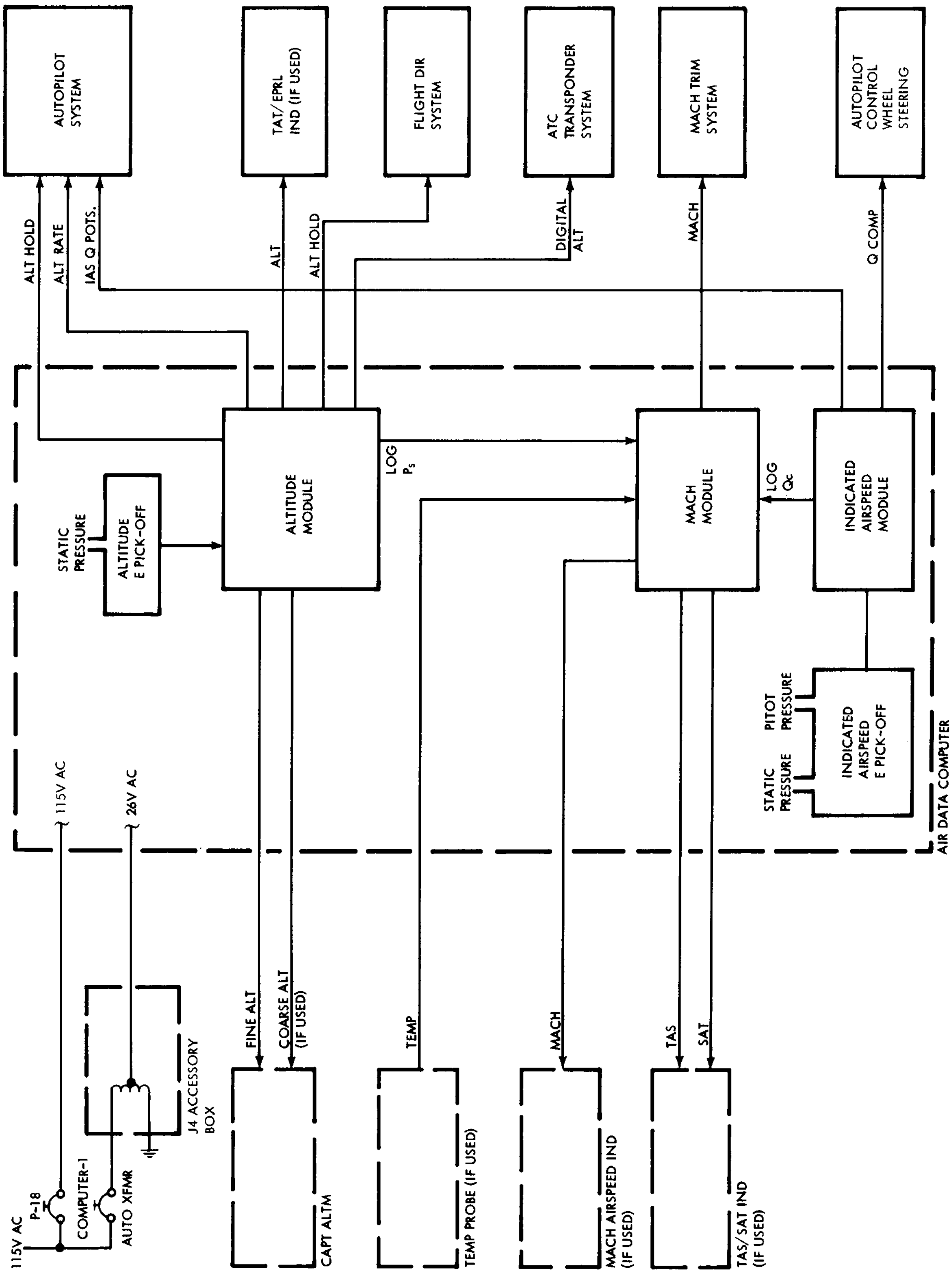


ANALOG COMPUTER FRONT PANEL



DIGITAL COMPUTER FRONT PANEL

Air Data Computer
Figure 5



Air Data Computer System Diagram
Figure 6

D. Computer Altitude Reporting Readout Assembly

(1) This assembly (if installed) gives a visual display of the absence or presence of a pulse of the digital altitude signals sent to the ATC transponder system. Each light on the front of the computer is marked with a letter/number code to reflect the altitude encoder pulse assignment. By comparing each light with a pulse assignment chart the exact altitude can be determined.

E. Airplane - Adapter

(1) This adapter is used with the computer to characterize selected outputs to meet specific airplane requirements and to adapt the computer to the particular airplane for which it is installed. Interlocking circuits prevent the use of the wrong adapter in a particular airplane. Two turnlock fasteners secure the adapter to the computer.

6. <u>Digital Air Data Computer</u>

A. General

(1) The digital air data computer (DADC) is a solid-state device that uses pitot static pressures and air temperature inputs to compute air data information required by airplane displays and systems. In addition to the above inputs, 115-volt ac and 26-volt ac are provided to a multipin connectors at the rear of the computer.

(2) The computer is of modular construction and consists of two pressure transducers, 12 plug-in printed circuit cards, a power supply and a chassis assembly. The front panel (Fig. 5) provides two switches for self-test actuation, a self-test indicator light, a fault isolation test connector for off-aircraft trouble shooting and pitot static pressure input connectors. Three electrical connectors mounted on the back of the chassis provide interconnection with the airplane wiring.

(3) The TEST SELECT switch manually selects built-in test to be performed as follows:
FUNCTION - transmits fixed values on all the computer signal output lines.
SLEW - sets altitude rate to 600 feet per minute.
HOLD - activates hold circuits.
FAIL - activates all failure warning flags.
The PUSH TO TEST pushbutton switch initiates the built-in test and the TEST VALID WHEN LIT indicator light comes on within 2 seconds after manually activated test is successfully completed and remains on if there is no failure.

B. Functional Description

(1) The computer uses primary inputs of static pressure (Ps), total pressure (Pt), and air temperature (Ti) to compute air data information. Other inputs required by the computer are power, reference voltages, enabling and command signals, and programmed inputs to inform the computer what airplane it is installed in. The computer contains a microcomputer that operates on the input information to form output parameters of analog and digital voltages representing altitude, airspeed, air temperature, and derivations thereof.

(2) Two plug-in circuit card assemblies each contain a microprocessor that controls an arithmetic processing unit for making air data calculations. Both circuit card assemblies have large scale integration (LSI) integrated circuits that contain read only memory (ROM) for program storage, random access memory (RAM) for scratch pad memory, and peripheral integrated circuits (IC's) having I/O ports. All of the LSI integrated circuits are tied together by common buses. Calculations of the computer are performed on a time-shared basis.

(3) Two pressure inputs and one temperature input are converted into 16-bit data words. Various reference, enabling, and command signals are received from the airplane and are used by the computing circuitry (using the pressure and temperature data) to provide outputs that drive airplane systems and indicators. The computer uses solid-state electronics to perform functions that were formerly carried out by mechanical devices. Synchros and potentiometers are "synthesized" using solid-state electronics. Also developed are hold signals, rate signals, and failure warning signals.

(4) The computer contains a digital information transfer system (DITS) that generates a serial, bipolar output. An air traffic control (ATC) code is also generated. The computer also contains an extensive, self-test monitoring system.

(5) A static source error correction (SSEC) program is stored in the instructional program. The SSEC program is used by the computer to compensate for differences in airplane type. The program to be used is selected by airplane program jumpers that inform the microcomputer which type of airplane the computer is in.

(6) Outputs to the air data instruments from the air data computer(s) and air data computer interface to other systems are shown in Fig. 6.

AIR DATA PRESSURE INSTRUMENTS - ADJUSTMENT/TEST

1. Air Data Pressure Instruments Test
   A. General
      (1) All relevant systems should be fully operational before performing these tests.
      (2) The following tests include procedures to check operation and accuracy of air data system instruments, and include:
         (a) Instruments pneumatically operated from direct pitot and static pressures.
         (b) Air data computer(s).
         (c) Instruments electrically operated from air data computer output signals initiated by pitot and static pressures and total air temperature inputs (if used).
         (d) Perform leakage test and restore systems to normal after any one of the above tests is performed.
      (3) The tests are to be made after the airplane has been on the ground at least 4 hours, except for digital air data computers.
      (4) The systems must be continuously energized for at least 30 minutes prior to tests, except for digital air data computers.
      (5) Head correction must be applied to pneumatic, servoed/pneumatic, or servoed altimeter test readouts to compensate for any difference in height between test set and altimeter being tested. If height difference exists, use appropriate test chart plus altimeter head correction test chart (Fig. 502). Determine height difference and read head correction for pressure altitude applied to altimeter. Subtract correction from altimeter reading if altimeter is above test set. Add correction to altimeter reading if altimeter is below test set. Height corrections are explained as follows:
         (a) Pneumatic altimeter - Height difference between altimeter and test set.
         (b) Servoed/pneumatic or servoed altimeter - Height difference between air data computer and test set.
      (6) Completely test one air data computer at a time if No. 1 and 2 are installed. Any or all of the air data indicator instrument scale error tests may be performed when air data computers are tested one at a time.
      (7) If two air data computers or their associated servoed indicator are to be tested, check that proper pitot and static connections are being used.

FL B. Equipment and Materials

(1) Air pressure and vacuum source - 5 to 40 inches of mercury absolute, and gage for measuring differential pressure, accurate within ± 0.010 inch of mercury. Range should be at least 10 inches of mercury

(2) Air pressure and vacuum source - 5 to 30 inches of mercury absolute

(3) Pressure gage - accurate to within ± 0.032 inch of mercury

(4) Stopwatch

(5) Mercury thermometer - accuracy ± 0.5°C

(6) Appropriate gage saver restrictors

(7) Adequate lengths of hose to couple pressure source with pitot tubes and vacuum source with static ports or drains

(8) Pitot-static probe adapter - P/N AD-953-7 and flush static port adapter 33410LH-125-4, Nav-Aids Limited, 2955 Diab Street, Quebec, Canada H4S1M1, or equivalents

C. Prepare to Test Air Data Pressure Instruments

(1) To prevent damage to equipment and instruments, check following precautions before applying pressures to pitot-static system.

(a) Apply or release vacuum or pressure at rate of climb or descent less than 3000 feet per minute for static system and 300 knots per minute for pitot system between test points (appropriate gage-saver restrictors should be used). At each test point, pressure should be reduced slowly to desired level without overshoot.

(b) Pressure in pitot lines should always be greater than or equal to pressure in static lines. Differential pressure should never exceed 10.00 inches of mercury or fall below zero.

(c) On airplanes with a digital air data system installed, the system will be damaged if static pressure exceeds pitot pressure more than 2.00 inches of mercury.

(d) Absolute pressure applied to static system should never exceed ambient absolute pressure when any instrument is connected to static system.

(e) Captain's, first officer's, and alternate static systems must be exposed to same absolute pressure conditions simultaneously during any test unless otherwise instructed.

(f) Do not change setting of static source selector valve while vacuum is on static system. Make sure that selector valve remains in NORMAL position unless test procedures specify otherwise.

(g) Make sure that pitot-static probe heaters remain off during tests.

(h) Inspect each pitot-static probe head both before and after these tests for evidence of damage to circular leading edge of its barrel.

(i) Do not apply pressure in excess of 31 inches of mercury absolute to static lines.

<u>CAUTION</u>: IF THE ABOVE PROCEDURES ARE NOT OBSERVED, EQUIPMENT AND INSTRUMENTS WILL BE DAMAGED.

(j) Make sure that instrument vibrator circuit breakers are closed (if installed) during testing.

(2) Disconnect cabin differential pressure indicator from static pressure source and cap pressure source.

(3) Make sure that flight recorder (if installed) is either operating or that its magazine is removed and recorder pressure bar is up.

(4) Connect pitot pressure source and static pressure source to captain's or first officer's pitot-static probe and static source to the alternate static port.

NOTE: System drain fittings may be used for pitot and static pressure connections but are not the preferred connections.

(5) Seal all unused static ports in such a way that removal of seal will be complete leaving no deposits or roughness in or about ports. Seal should be properly marked to prevent removal while test is in progress.

(6) Place thermometer adjacent to total air temperature probe and shield from direct wind and sunlight.

(7) On airplanes with flap load relief system, open FLAP LOAD RELIEF circuit breaker on P6 panel.

WARNING: FAILURE TO OPEN THE FLAP LOAD RELIEF CIRCUIT BREAKER MAY RESULT IN FLAP MOVEMENT AND INJURY TO PERSONNEL.

(8) Provide electrical power.

(9) Set barometric scale setting knob on captain's and first officer's altimeter to 29.92 inches of mercury before starting tests. Adjustment is to be made in the increasing reading direction without overshoot.

D. Test Air Data Pressure Instruments

(1) Test Pneumatic-Operated Instruments

(a) Perform following series of tests for pneumatic instruments per conditions shown in Fig. 501. Perform servoed instruments test below for instruments receiving air data computer inputs (if installed). Apply head correction above, when applicable.

CAUTION: DURING TESTS, DO NOT ALLOW MACH/AIRSPEED WARNING CLACKER TO OPERATE MORE THAN 5 MINUTES PER SYSTEM OR DAMAGE WILL RESULT FROM OVERHEATING. AFTER 5 MINUTES OF OPERATION, A MINIMUM COOLING PERIOD OF 15 MINUTES SHOULD BE ALLOWED.

1) Establish pitot and static pressure conditions in captain's, first officer's, auxiliary and alternate system lines per Fig. 501 for increasing and decreasing altitudes.

(2) Test Air Data Computer(s)

NOTE: Refer to A.(3), (4), and (5) when two computers are to be tested. Test below is written for computer No. 1. For computer No. 2 use applicable switches and circuit breakers.

(a) Failure annunciators on computer may be reset with following provisions.

NOTE: Operators may, at their discretion, disregard the maintenance fault annunciators, cover them with opaque adhesive tape, or continue their use.

1) Failure annunciators did not trip in flight (except during emergency descent).
2) Pressure limits per C.(1) above were not exceeded.
3) If test equipment used has automatic means of preventing excessive back pressure of more than 0.10 inch mercury.
4) Failure annunciator does not trip during air data computer test below.

(b) Reset tripped failure annunciators as follows (tripped annunciators appear red, reset annunciators appear white):
1) Remove computer from airplane rack and remove dust cover. Replace computer in airplane rack.
2) Make sure that 115-volt and 26-volt power is applied to computer.
3) Press annunciator reset button (located on right-hand side of computer chassis). Make sure that tripped annunciators change from red to white.

NOTE: Do not reset annunciators with magnets.

(c) Open the following circuit breakers on panel P18:
1) AIR DATA COMPUTER NO. 1 115 volts ac
2) AIR DATA COMPUTER NO. 1 26 volts ac

TEST INPUT VALUES	UNITS	PANEL	TEST 1	TEST 2	TEST 3	TEST 4	TEST 5
STATIC PRESSURE	IN. HG ABS		29.92 [1]	29.92 [1]	29.92 [1]	29.92 [1]	24.90
DIFFERENTIAL PRESSURE	IN. HG ABS		0.39	0.58	0.82	3.10	5.68
PITOT PRESSURE	IN. HG ABS		30.31	30.50	30.74	33.02	30.58
STATIC SOURCE SELECTOR VALVE			NORMAL	NORMAL	NORMAL	NORMAL	NORMAL
INSTRUMENT READINGS							
PNEUMATIC ALTIMETER	FEET	F/O	0 ±20	0 ±20	0 ±20	0 ±20	5000 ±38
COMBINED MACH AIRSPEED INDICATOR (MACH)	MACH	CAPT & F/O					.55 ±.03
(AIRSPEED)	KNOTS	CAPT & F/O	90 ±3	110 ±2	130 ±2	250 ±3.5	334 ±4
MAXIMUM ALLOWABLE DIFFERENCE BETWEEN AIRSPEED INDICATORS	KNOTS	CAPT & F/O	6	4	4	7	8
(AIRSPEED LIMIT)	KNOTS	CAPT & F/O	350 +1/-6	350 +1/-6	350 +1/-6	350 +1/-6	352 +1/-6
VERTICAL SPEED INDICATOR [2]	FT/MIN	CAPT & F/O					

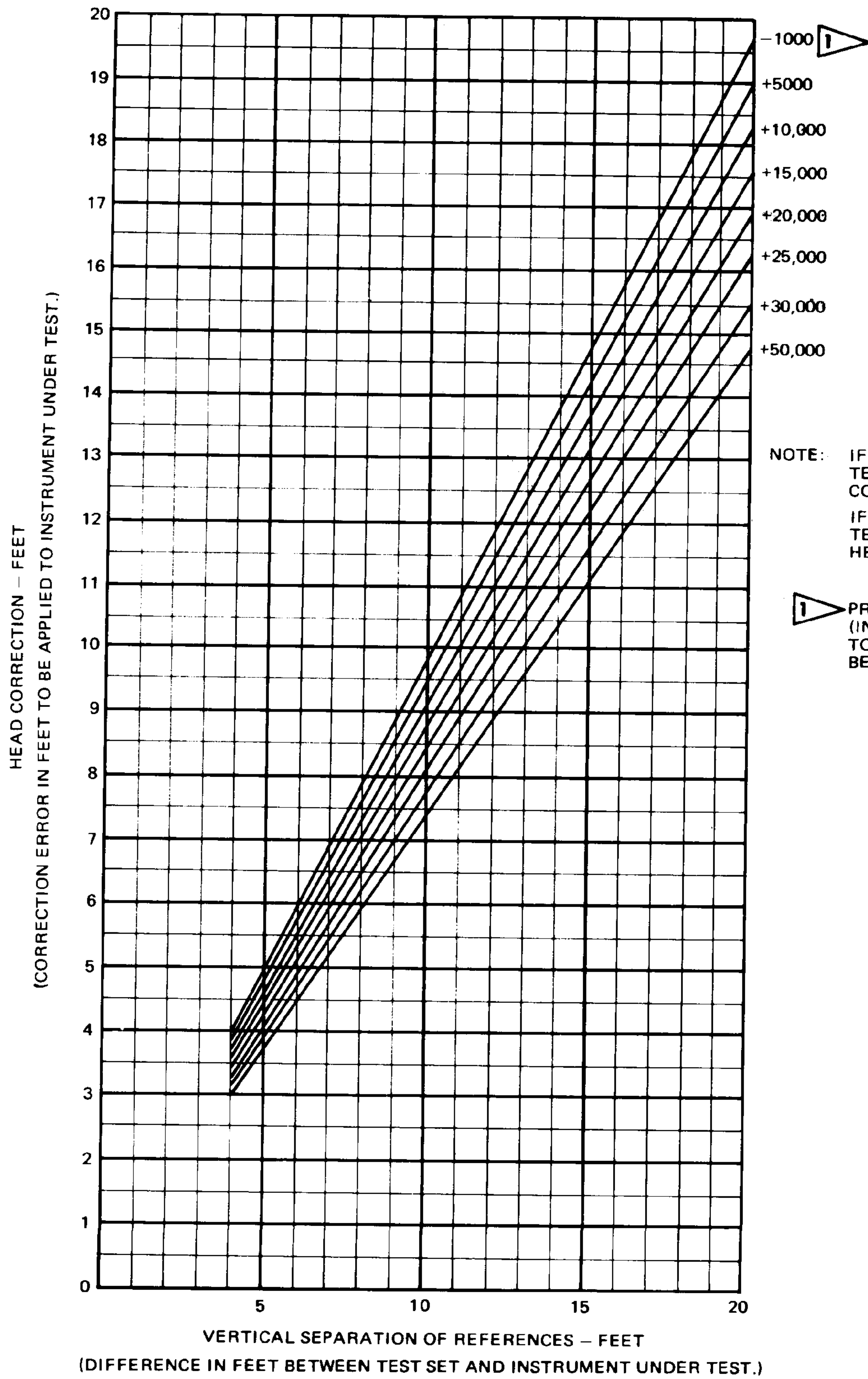
Pneumatic Instruments Test Chart
Figure 501 (Sheet 1)

TEST INPUT VALUES	UNITS	PANEL	TEST 6	TEST 7	TEST 8	TEST 9	TEST 10
STATIC PRESSURE	IN. HG ABS		20.58	13.75	8.88	8.49	5.538
DIFFERENTIAL PRESSURE	IN. HG ABS		5.67	6.22	4.65	4.99	3.637
PITOT PRESSURE	IN. HG ABS		26.25	19.97	13.53	13.48	9.175
STATIC SOURCE SELECTOR VALVE			NORMAL	NORMAL	NORMAL	NORMAL	NORMAL
INSTRUMENT READINGS							
PNEUMATIC ALTIMETER	FEET	F/O	10,000 ±80	20,000 ±130	30,000 ±180	31,000 ±185	40,000 ±235
COMBINED MACH AIRSPEED INDICATOR (MACH)	MACH	CAPT & F/O	.60 ±.02	.75 ±.02	.80 ±.02	.84 ±.02	.88 ±.02
(AIRSPEED)	KNOTS	CAPT & F/O	333 ±4	349 ±4	304 ±4	314 ±4	270 +0/-4
MAXIMUM ALLOWABLE DIFFERENCE BETWEEN AIRSPEED INDICATORS	KNOTS	CAPT & F/O	8	8	8	8	8
(AIRSPEED LIMIT)	KNOTS	CAPT & F/O	355 +1/-6	363 +1/-6	321 +1/-6	314 +1/-6	248 +4/-0
VERTICAL SPEED INDICATOR [2]	FT/MIN	CAPT & F/O					

[1] THIS STATIC PRESSURE POINT SHOULD BE ESTABLISHED BY DECREASING THE STATIC PRESSURE TO 28.00 INCHES OF MERCURY ABS AND THEN SLOWLY INCREASING TO 29.92 INCHES MERCURY ABS BEING CAREFUL NOT TO OVERSHOOT.

[2] WITH THE NORMAL AND AUXILIARY STATIC SYSTEMS CONNECTED TOGETHER, REDUCE THE STATIC PRESSURE AT A CONSTANT RATE OF ASCENT OF 2000 FEET/MINUTE (AS INDICATED ON THE CAPTAIN'S AND F/O'S VERTICAL SPEED INDICATOR) BETWEEN THE ALTITUDES OF 2000 AND 4000 FEET. THE TIME REQUIRED SHOULD BE BETWEEN 52 AND 71 SECONDS. AN INCREASE OF THE STATIC PRESSURE AT THE SAME RATE BETWEEN INDICATED ALTITUDES OF 4000 TO 2000 FEET SHOULD REQUIRE THE SAME INTERVAL (EQUIVALENT TO A TOLERANCE OF 300 FEET/MINUTE). INDICATOR SHOULD BE TAPPED LIGHTLY ON CASE BEFORE CHECKING READING. (DO NOT TAP GLASS DIAL.)

Pneumatic Instruments Test Chart
Figure 501 (Sheet 2)



NOTE: IF ALTIMETER IS ABOVE TEST SET SUBTRACT HEAD CORRECTION.

IF ALTIMETER IS BELOW TEST SET ADD HEAD CORRECTION.

1 PRESSURE ALTITUDE (IN FEET) APPLIED TO ALTIMETER BEING TESTED.

Altimeter Head Correction Test Chart
Figure 502

473853

3) CAPTS AUTO TRANSFORMER

(d) Place function test select switch in IAS SLEW position and operate IASM SLEW test switch for 10 seconds. Make sure that test indicator pointer does not move.

(e) Close AIR DATA COMPUTER NO. 1 115-volt ac circuit breaker on panel P18. Operate IASM SLEW test switch for 10 seconds. Make sure that panel meter reads in green area.

(f) Place function test select switch in ALT MONITOR position. Make sure that panel meter reads near zero.

(g) Close all other air data circuit breakers not closed by above steps. Make sure that panel meter reads in green area.

(h) Operate MONITOR test switch. Make sure that panel meter reads near zero.

(i) Place function test select switch in RATE 1 position and operate ALT SLEW test switch. Make sure that panel meter reads in green area.

(3) Electric Altimeter Test

(a) General

1) These tests are for the electric altimeter which receives an electrical input from the air data computer.

2) Air data computer should be tested per step 2) or known to be functioning properly before this test. The system must have been continuously energized for at least 30 minutes prior to this test.

3) Use Fig. 503 below as directed in each test procedure. Apply head correction above, when applicable.

(b) Test Electric Altimeter

1) Open CAPT ALTM 26V AC circuit breaker on P18 panel.

2) Set barometric setting on captain's altimeter to 29.92 inches of mercury or 1013 millibars.

3) Observe that altimeter failure warning flag is in view.

4) Close CAPT ALTM 26V AC circuit breaker on panel P18. Check that altimeter flag is removed from view.

INPUTS						
Test Point	Altitude (Feet)	Static Press (In. Hg)	Diff Press (In. Hg)	Pitot Press (In. Hg)	Mach Input	Airspeed Input
1	0	29.921	3.100	33.021	0.3779	250
2	10,000	20.577	6.286	26.863	0.6290	350
3	20,000	13.750	5.368	19.073	0.7034	325
4	35,000	7.041	4.534	11.575	0.8735	300
5	40,000	5.538	3.637	9.175	0.8808	270

OUTPUTS			
Test Point	Altitude (Feet)	Altimeter Tolerance	Altimeter Differences *[1]
1	0	±30 ft	±40 ft
2	10,000	±40 ft	±90 ft
3	20,000	±50 ft	±140 ft
4	35,000	±90 ft	±220 ft
5	40,000	±110 ft	±220 ft

*[1] Defines the maximum allowable differences between the captain's electric altimeter and the first officer's pneumatic altimeter.

5) Ensure static source selector valve on captain's static system is in NORMAL position and that captain's altimeter warning flag is not in view.
6) Apply pressure to airplane pitot-static systems to point 1 in Fig. 503. Make sure that altimeter readings are within tolerance shown. Use head correction as necessary.
7) Repeat step 6) for test points 2 thru 5 of Fig. 503.

(4) Test Digital Air Data Computer(s)

<u>NOTE</u>: Refer to A.(5), (6), and (7) when two computers are to be tested. Test below is written for computer No. 1. For computer No. 2 use applicable switches and circuit breakers.

(a) The following steps list all self-tests. Perform those steps for self-test readouts available.
1) Rotate self-test switch to FUNCTION and press push-to-test switch. Switch must be held depressed approximately 35 seconds. Verify following values on indicators, as applicable, when TEST VALID WHEN LIT light on computer front panel comes on.
a) Altitude 10,000 ±40 feet (not applicable to servo-pneumatic altimeters)

b) Mach 0.785 ±.01
c) Airspeed 440 ±2 knots
d) True airspeed 477.8 ±6 knots (if applicable)
e) Static air temperature -29.4 ±2°C (if applicable)

2) Rotate self-test switch to SLEW and press push-to-test switch. Verify that TEST VALID WHEN LIT light comes on.
3) Rotate self-test switch to HOLD and press push-to-test switch. Verify that TEST VALID WHEN LIT light comes on.
4) Rotate self-test switch to FAIL WARN and press push-to-test switch. Verify that test VALID WHEN LIT light comes on.

(5) Test Air Data Pressure Instrument System for Leakage
(a) Perform leakage test on each pitot-static system: captain's, first officer's, and auxiliary. Apply vacuum of 22,500 ±200 feet altitude to static system and pressure of 300 knots equivalent airspeed to pitot system. Check that leakage rate does not exceed 400 feet per minute and 5 knots per minute respectively.
(b) Reconnect cabin altimeter and differential pressure indicator to auxiliary static pressure system.

(6) Restore Air Data Pressure Instrument and Pitot-Static Systems to Normal
(a) Remove external hoses and static port seals, ensuring that no deposit or roughness exists in or about static ports.
(b) Ensure that flight recorder (if installed) is returned to normal operating configuration.
(c) If no longer required, remove electrical power from airplane.

AIR DATA COMPUTER-REMOVAL/INSTALLATION

1. Remove Air Data Computer

CAUTION: TO AVOID COMPUTER DAMAGE, PITOT AND STATIC SYSTEM PRESSURES MUST BE AT AMBIENT BEFORE PROCEEDING.
TO AVOID ELECTRIC FLIGHT RECORDER DAMAGE, FLIGHT RECORDER CIRCUIT BREAKERS MUST BE OPENED BEFORE PROCEEDING.

A. Open air data computer circuit breakers on panel P6 or P18.
B. Disconnect pitot and static pneumatic hoses from front panel.
C. Loosen cam-lock handles and remove computer.

2. Install Air Data Computer

A. Insert computer into case being careful to avoid stress and/or damage to electrical connectors on rear chassis panel.
B. Tighten cam-lock handles

CAUTION: TO AVOID COMPUTER DAMAGE, PITOT AND STATIC SYSTEM PRESSURES MUST BE AT AMBIENT BEFORE PROCEEDING.

C. Connect pitot and static system pneumatic hoses to front panel.
D. Close air data computer circuit breakers on panel P6 or P18.
E. If any connection in the pitot static system (other than a quick disconnect) has been broken, perform a low range leak test of the applicable pitot static system (Ref 34-11-0, A/T).

3. Test Digital Air Data Computer

A. Provide electrical power.
B. Perform self-test as follows:

NOTE: TEST VALID WHEN LIT light should come on and remain on while PUSH TO TEST switch is pressed. Switch must be held for approximately 1/2 minute for the FUNCTION switch position to allow instruments to stabilize. For other switch positions light should come on in approximately 2 seconds.

(1) Rotate self-test switch on front of computer to FUNCTION and press push-to-test switch. Verify that TEST VALID WHEN LIT light comes on.
(2) Repeat step 3.B.(1) with self-test switch set to SLEW, HOLD, and FAIL WARN.

C. Remove electrical power if no longer required.

4. Test Analog Air Data Computer (Honeywell)

A. Provide electrical power.

B. The following steps list all self-tests. Perform those steps for self-test readout available.
   (1) Set the ADC panel rotary switch to ALT RATE No. 1.
      (a) Set the ADC panel 60 FT/ALT SLEW switch to ALT SLEW. The ADC panel meter needle shall move into the green area of the scale.
      (b) Release the 60 FT/ALT SLEW switch. The meter needle shall move to the lower 5 percent of the scale after a short time delay during which the servo loop re-nulls.
   (2) Set the ADC panel rotary switch to IAS SLEW.
      (a) Set the ADC panel IAS-M SLEW/5KN switch to IAS-M SLEW. The panel meter needle shall move into the green area of the scale.
      (b) Release the switch. The meter needle shall drop to the lower 5 percent of the scale when the servo has had time to re-null.
   (3) Set the ADC panel rotary switch to MACH SLEW.
      (a) Repeat steps (2)(a) and (2)(b). Results shall be the same as in the steps except that the meter needle shall delay for no more than six seconds before going to the green area of the scale. (Typically the needle will delay about two and one-half seconds.)
   (4) Observe that ALT, MACH, and IAS failure annunciators at bottom of front panel show white centers.

NOTE: In test (5), (6) and (7) the ALT, IAS, and MACH failure annunciators should be observed to check that no red center appears to indicate a failure.

   (5) Set the ADC panel rotary switch to ALT MONITOR. The panel meter needle shall be in the green area of the scale.
      (a) Set the 600 FPM/MONITOR panel switch to MONITOR and hold it at that position. The meter needle shall drop to the lower 5 percent of the scale.
      (b) Release the switch. The meter needle shall return to the green area of the scale.
   (6) Set the ADC panel rotary switch to IAS MONITOR. The panel meter needle shall be in the green area of the scale.
      (a) Repeat steps (5)(a) and (5)(b). Results shall be the same as in the steps.
   (7) Set the ADC panel rotary switch to M MONITOR (if mach module installed). The panel meter needle shall be in the green area of the scale.
      (a) Repeat steps (5)(a) and (5)(b). Results shall be the same as in the steps.
C. Remove electrical power if no longer required.

AIR DATA PRESSURE INDICATORS - REMOVAL/INSTALLATION

1. Remove Indicator
   A. Open applicable indicator circuit breakers on panel P6 or P18.

      NOTE: If servoed or servo-pneumatic indicator, also open applicable air data computer circuit breakers.

   B. Loosen instrument mounting clamp and slide indicator out of instrument panel.

      CAUTION: DRAW INDICATOR OUT OF INSTRUMENT PANEL WITH CARE TO AVOID STRESS AND/OR DAMAGE TO CABLE AND PNEUMATIC HOSES (IF ATTACHED) AT BACK OF CASE.

   C. Disconnect electrical connector on rear of indicator case.

      CAUTION: TO AVOID DAMAGE TO PNEUMATIC OR SERVO-PNEUMATIC INSTRUMENTS, PITOT AND STATIC SYSTEM PRESSURES MUST BE AT AMBIENT BEFORE PROCEEDING.

   D. Disconnect pitot static system hoses from rear of indicator case (if applicable).
2. Install Indicator
   A. Connect electrical cable to connector on rear of indicator case.

      CAUTION: TO AVOID DAMAGE TO PNEUMATIC OR SERVO-PNEUMATIC INSTRUMENTS, PITOT AND STATIC SYSTEM PRESSURES MUST BE AT AMBIENT BEFORE PROCEEDING.

   B. Connect pitot static system hoses to rear of indicator case (if applicable). Quick disconnects, when reconnected, do not require a leak check. A visual check of the quick disconnect for full mating and to assure connection is locked and in the sealed position is required. If behavior of instrument is such as to suggest leakage, a leak test should be performed (Ref 34-11-0, A/T).
   C. Slide indicator into position in instrument panel and secure with instrument mounting clamp.

      CAUTION: INSERT INDICATOR INTO PANEL WITH CARE TO AVOID STRESS AND/OR DAMAGE TO CABLE AND PNEUMATIC HOSES (IF ATTACHED).

D. Close applicable circuit breakers opened in step 1.A.

NOTE: Place mode switch on STBY (BARO) or OFF position on servo-pneumatic indicators.

3. Test Indicator
   A. Provide electrical power.
   B. Check that panel light circuit breakers on panel P6 are closed.
   C. Check that indicator illumination lights are on.

   NOTE: For servoed or servo-pneumatic indicators, air data computer should be known to be functioning properly before this test.

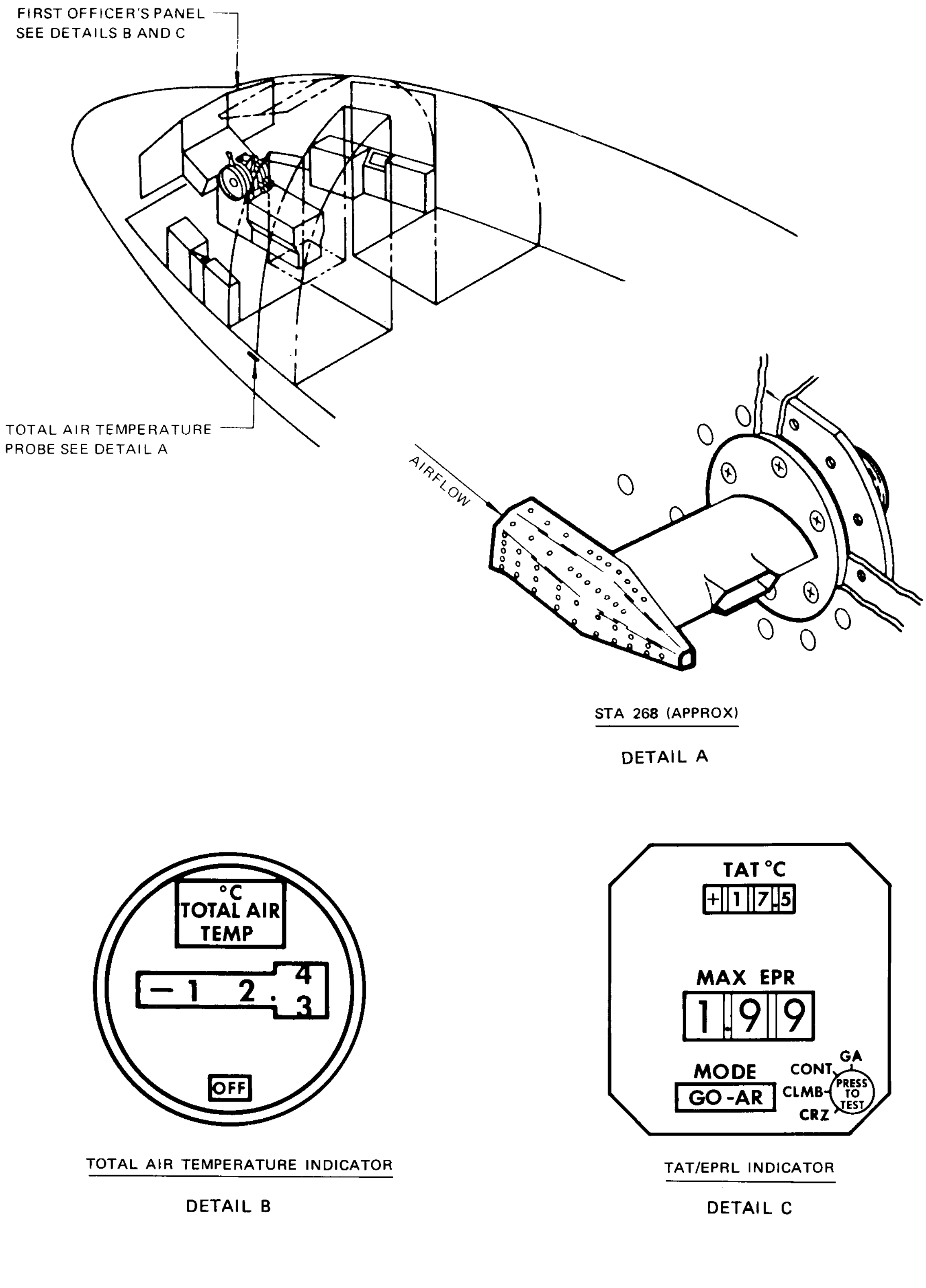
   D. Check that indicator failure warning flags are out of view.

   NOTE: Place mode switch on servo-pneumatic indicator in CADC or ON position.

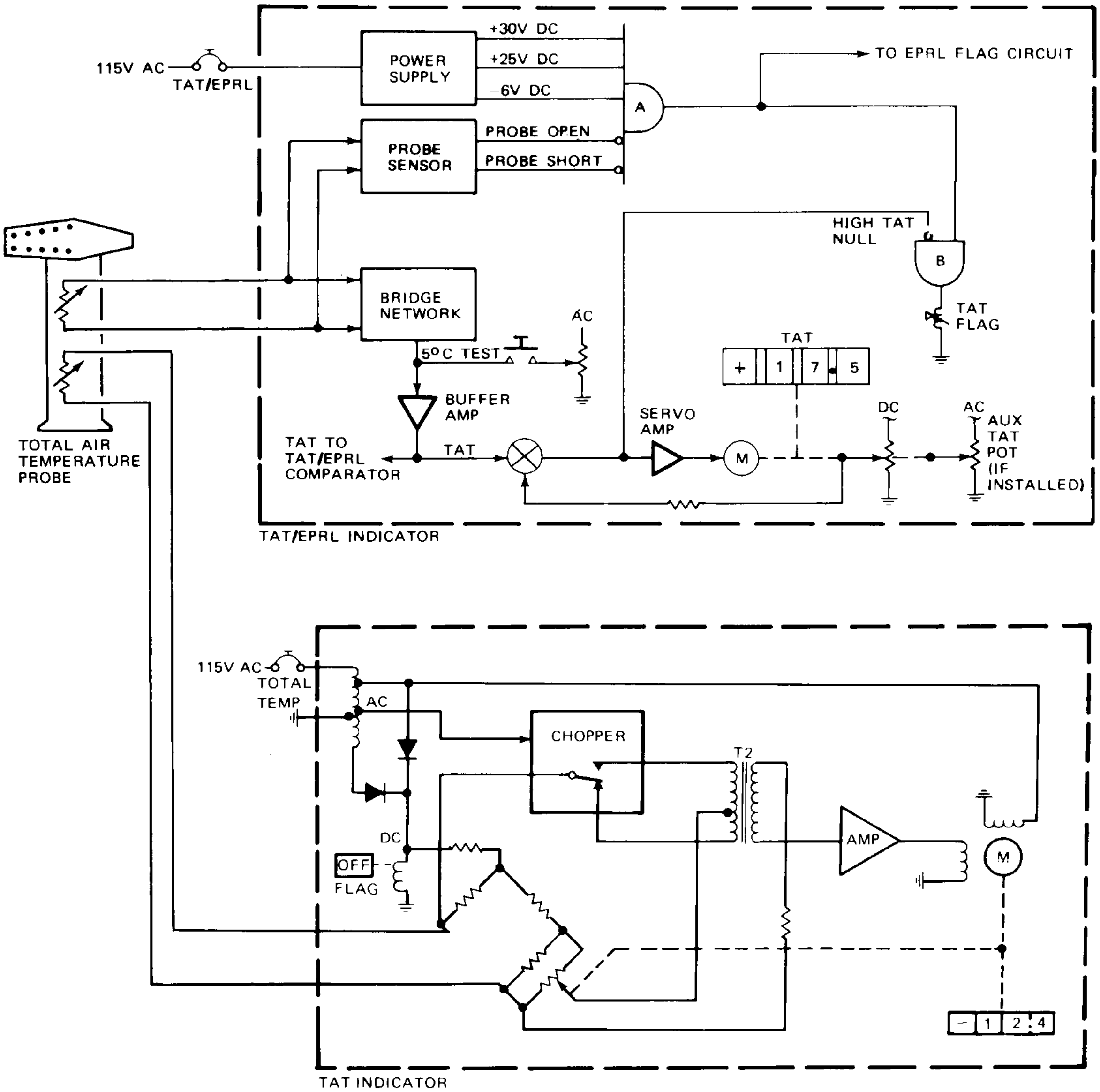
   E. Remove electrical power if no longer required.

AIR DATA TEMPERATURE INDICATING SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The temperature of the outside air is used to calculate air density, which is required for correct adjustment of the airplane engine. The air temperature indicating system consists of a total air temperature indicator, a total air temperature/engine pressure ratio limit indicator and a dual element temperature probe.
2. Total Air Temperature Probe
   A. The outside dual element air temperature probe contains noninductive resistive transducer sensing elements and an anti-icing heater. Temperature variations of the slipstream airflow through the probe and sensor causes the element resistance to vary, thus providing air temperature data to the total air temperature indicators (Fig. 1 and 2).
   B. In flight, heating of the probe assembly is controlled by a switch on the pilots' overhead panel (Ref Chapter 30, Pitot Tubes and Temperature Probe Anti-Icing System).
3. Total Air Temperature Indicator
   A. The total air temperature sensor resistance is electrically included as one leg of a dc bridge circuit (Fig. 2). A change in air temperature will cause a subsequent change in resistance and unbalance the bridge. When the bridge is unbalanced the error signal is sent through the chopper and transformer to the servo-amplifier. The servo-amplifier drives the motor which positions the digital readout scale and the feedback potentiometer in the bridge circuit. Power to operate the indicator comes from a 115-volt ac circuit breaker on panel P6.
   B. An OFF flag will appear when electrical power is lost or disconnected. The indicator has a digital readout scale from -70°C to +70°C.
4. Total Air Temperature/Engine Pressure Ratio Indicator
   A. This indicator uses temperature and pressure to compute and display accurate total air temperature (TAT) and engine pressure ratio limit (EPRL). Refer to Chapter 77 for a description of the EPRL part of the indicator. Total air temperature is displayed through the range of -70°C to +65°C. Power to operate the indicator comes from the 115-volt ac circuit breaker on P6 panel.
   B. The TAT circuit develops total air temperature from the temperature probe and bridge network. The bridge signal is amplified and sent to the TAT servo which drives the indicator. The buffer amplifier also sends temperature to the EPRL section of the indicator. (See figure 2.)



Air Data Temperature Indicating System Component Location
Figure 1



Air Data Temperature Indicating System Schematic
Figure 2

C. The TAT failure monitor consists of AND gates A and B. The output of AND gates A and B will become zero causing the TAT warning flag to come into view if the following occurs; loss of 30 volt dc, 25 volt dc, -6 volt dc, or if the TAT probe circuit is open or short circuited. AND gate B output will become zero if there is a high null in the TAT servo. When the self test switch is pressed, a 5°C TAT input is applied to the buffer amplifier. The TAT indicator should read +5 (+ 0.5)°C if this part of the indicator is working properly.

AIR DATA TEMPERATURE INDICATING SYSTEM - ADJUSTMENT/TEST

1. Air Data Temperature Indicating System Test
   A. Equipment and Materials
      (1) Thermometer accurate to ±0.5°C
   B. Prepare to Test Total Air Temperature System

      CAUTION: THIS TEST SHOULD EITHER BE PERFORMED BEFORE CHECKING HEATERS IN TEMPERATURE PROBE OR SUFFICIENT TIME SHOULD BE ALLOWED TO ENSURE THAT TEMPERATURE PROBE HAS RETURNED TO AMBIENT TEMPERATURE.

      (1) Apply external electrical power.
      (2) Place thermometer adjacent to total air temperature probe. Shield from direct sunlight and wind.
      (3) Close total air temperature circuit breaker on P6.
   C. Test Total Air Temperature System

      NOTE: For TAT values of (-60)°C or less and 60°C or greater, the indicator will show horizontal bars.

      (1) Close TAT/EPRL circuit breaker on panel P6. Check that TAT indicator flag goes out of view.
      (2) Total air temperature indicator should read ambient temperature of probe within ± 4°C.
      (3) Apply heat to probe and check that indicator shows increasing reading.
      (4) Apply cold to temperature probe and check that indicator shows decreasing temperature.
      (5) Press PUSH TO TEST switch with mode selector switch in GA, CONT, CLIMB, and CRZ positions. Check that TAT indicator reads +5±0.5°C for each position.
      (6) Open circuit breaker. Check that TAT flag on TAT/EPRL comes into view.
   D. Return Airplane to Normal Configuration
      (1) Restore total air temperature system to normal and open circuit breaker.
      (2) Determine if there is further need for external power, if not remove external power.

TOTAL AIR TEMPERATURE PROBE - REMOVAL/INSTALLATION

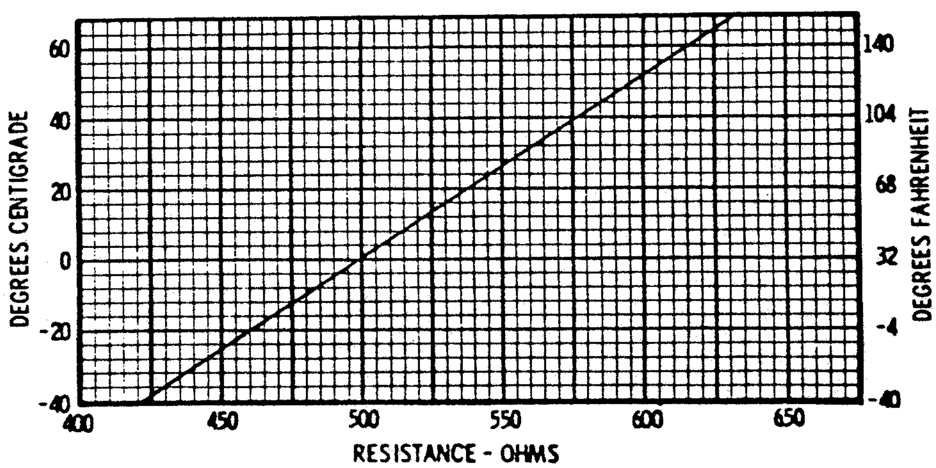
1. Remove Temperature Probe
   A. Check that TOTAL TEMP circuit breaker is open and attach a DO-NOT-CLOSE tag.
   B. Support probe and remove six mounting screws (Fig. 401).
   C. Pull probe away from airplane to obtain access to electrical connector. Remove electrical connector from probe.
   D. Remove gasket.
2. Install Temperature Probe
   A. Install gasket (Fig. 401).

      NOTE: Check to see that gasket is not damaged; replace if necessary.

   B. Test probe per Total Air Temperature Probe - Adjustment/Test (AMM 34-14-21/501).
   C. Connect electrical connector to probe.
   D. Position probe so that widest orifice is pointed forward and install six mounting screws.
   E. Measure the resistance between strut of the pitot static probe and the airplane skin with an ohmmmeter.
   F. If the resistance is more than 0.010 ohm, do these steps:
      (1) Remove the pitot-static probe.
      (2) Clean the bonding surfaces, including the countersunk holes in the pitot-static probe (SWPM 20-20-00).
      (3) Replace the existing screws with new screws.
      (4) Re-install the pitot-static probe.
      (5) Measure the resistance between strut of the pitot-static probe and the airplane skin with an ohmmmeter.
      (6) If the resistance is more than 0.010 ohms, do these steps:
         (a) Remove the pitot-static probe.
         (b) Replace the nutplates and rivets that attach the pitot-static probe (SRM 51-40-02).
         (c) Re-install the pitot-static probe and make sure the bonding resistance is not more than 0.010 ohm.
   G. Remove the DO-NOT-CLOSE tag and close the TOTAL TEMP circuit breaker.
   H. Test probe per Air Data Temperature Indicating System - Adjustment/Test (AMM 34-14-0/501).
   I. Seal around the TAT probe and the airplane skin using BMS 5-95 class c sealant.

TOTAL AIR TEMPERATURE PROBE - ADJUSTMENT/TEST

1. Total Air Temperature Probe Test
   A. Equipment and Materials
      (1) Ohmmeter - low excitation voltage type
   B. Test Total Air Temperature Probe
      (1) Look into the air inlet scoop and exhaust ports and check for foreign particles which block or partially block the free passage of air. If foreign particles exist, see Total Air Temperature Probe - Cleaning/Painting.
      (2) Remove electrical connector from temperature probe.
      (3) Check the temperature sensor by measuring the resistance between pins 3 and 4.
      (4) The measured resistance at a given ambient temperature should read as indicated on the temperature - resistance graph taking into account the tolerance of the ohmmeter. (See figure 501.)
      (5) Measure the resistance at a given ambient temperature probe case. This resistance should be greater than 10 megohms.
      (6) Check the heater element by measuring the resistance between pins 1 and 6. The resistance should be approximately 20 ohms at ambient temperature between 60 and 90°F. It should be somewhat less at low ambient temperature and somewhat higher at high ambient temperature.
      (7) Measure the resistance from either pin 1 or 6 to the temperature probe case. This resistance should be greater than 5 megohms.
      (8) Replace electrical connector on temperature probe.



Temperature - Resistance Graph
Figure 501

TOTAL AIR TEMPERATURE PROBE - CLEANING/PAINTING

1. Total Air Temperature Probe Cleaning
   A. Equipment and Materials
      (1) Compressed air source
      (2) Syringe and water or dry cleaning solvent
   B. Cleaning Total Air Temperature Probe

      CAUTION: CARE MUST BE USED IN REMOVING FOREIGN MATERIAL THAT MAY BE LODGED WITHIN THE AIR INLET SCOOP TO PREVENT DAMAGE TO THE SENSING ELEMENT.

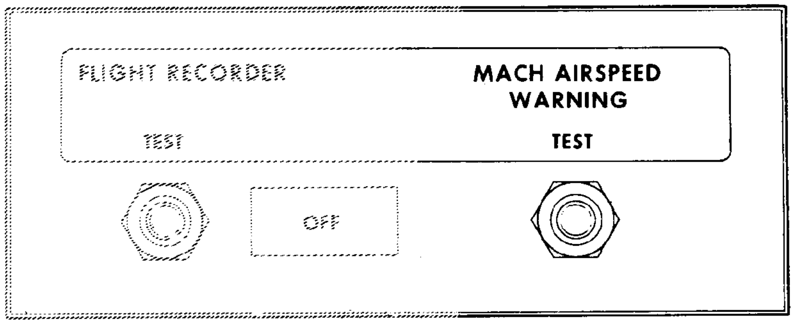
      (1) To remove foreign material, blow compressed air in a reverse direction through the air exit hole.
      (2) If this method fails, use a syringe and flush the interior of the air inlet scoop with water or a dry cleaning solvent. If foreign particles still exist, it may be necessary to remove the temperature probe from the airplane for additional cleaning.
      (3) With the temperature probe removed from the airplane, soak the air inlet scoop and strut with water or a dry cleaning solvent.

         CAUTION: DO NOT USE FLUIDS THAT CONTAIN SODIUM CHLORIDE OR SULFUR COMPOUNDS TO CLEAN THE PROBE. SODIUM CHLORIDE AND SULFUR CAN CAUSE PREMATURE FAILURE OF THE PROBE.

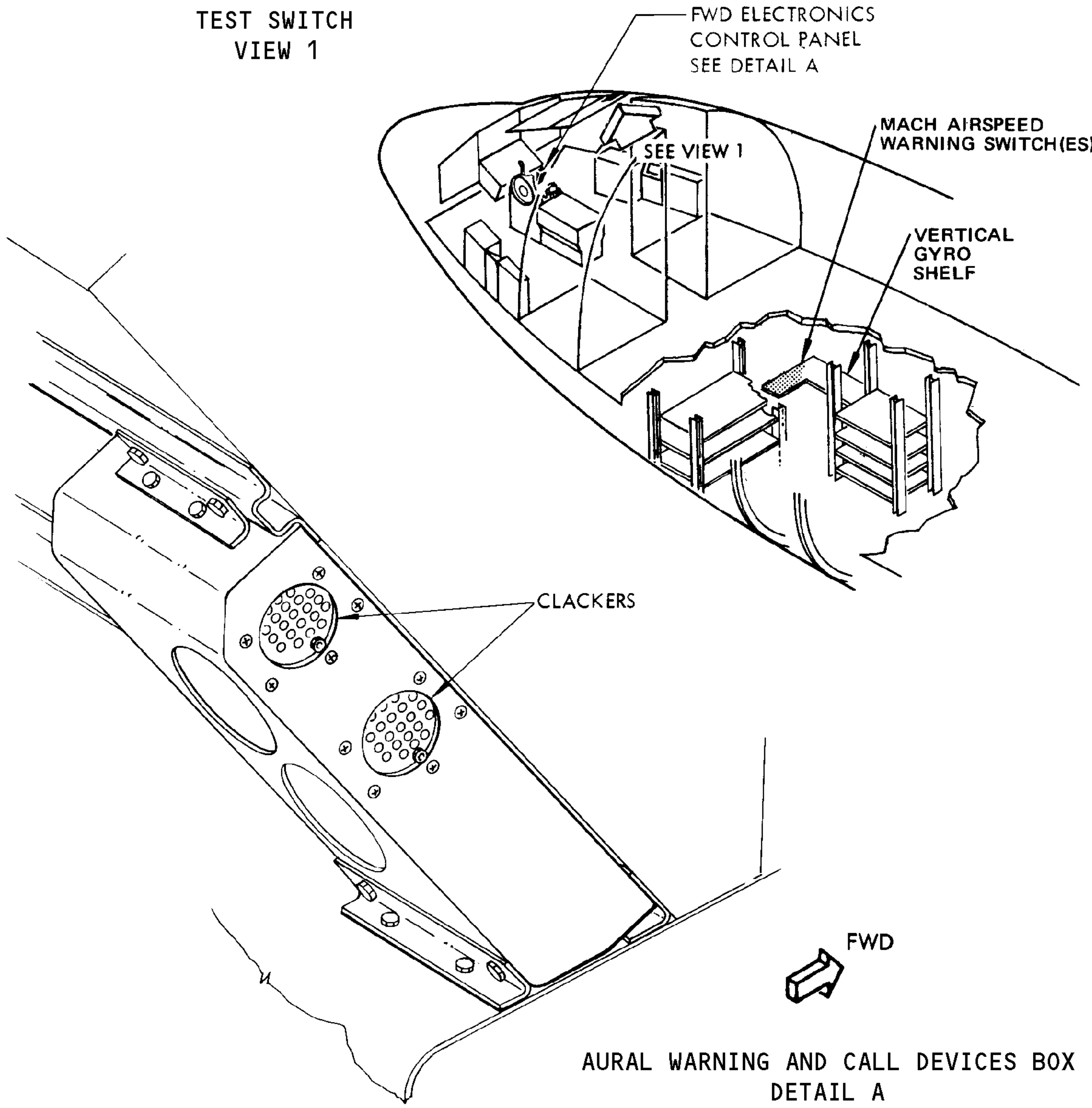
      (4) Move the probe back and forth in the liquid, completely flushing interior of the scoop and strut.
      (5) When the foreign material has become dislodged, blow compressed air through the scoop in a reverse direction.
      (6) After cleaning and before installation, perform test as indicated in Total Air Temperature Probe - Adjustment/Test.

MACH AIRSPEED WARNING SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The mach airspeed warning system is provided to alert the flight crew when the airplane approaches a critical speed. The warning is accomplished by a pitot static operated switch and an aural warning (See figures. 1 and 2.) A push-to-test switch is provided on the overhead panel (P5) to check operation of the warning circuit.
2. Mach Airspeed Warning Switch
   A. The switch assembly, under conditions of high mach airspeed will trigger an electronic clacker located in the aural warning devices unit, providing an aural warning signal. The clacker will sound any time the indicated airspeed exceeds the maximum operating airspeed or mach number.
   B. The warning switch bellows is controlled by pitot and static pressure inputs and is mechanically connected to a set of contacts. When a predetermined airspeed or mach number is exceeded, the contacts open and 28 volts dc is removed from the switch relay. With the relay de-energized, a ground is applied to the clackers in the aural warning and call devices box. The clackers will remain energized until the airplane airspeed or mach number is reduced.
3. Mach Airspeed Aural Warning Clackers
   A. The aural warning and call devices box on the control stand has two aural clackers. Upon initiation by the mach airspeed warning switch, the clackers provide a distinct warning signal which cannot be mistaken for the aural warnings associated with or activated by other systems. Refer to Chapter 31, Aural Warning and Call Devices.

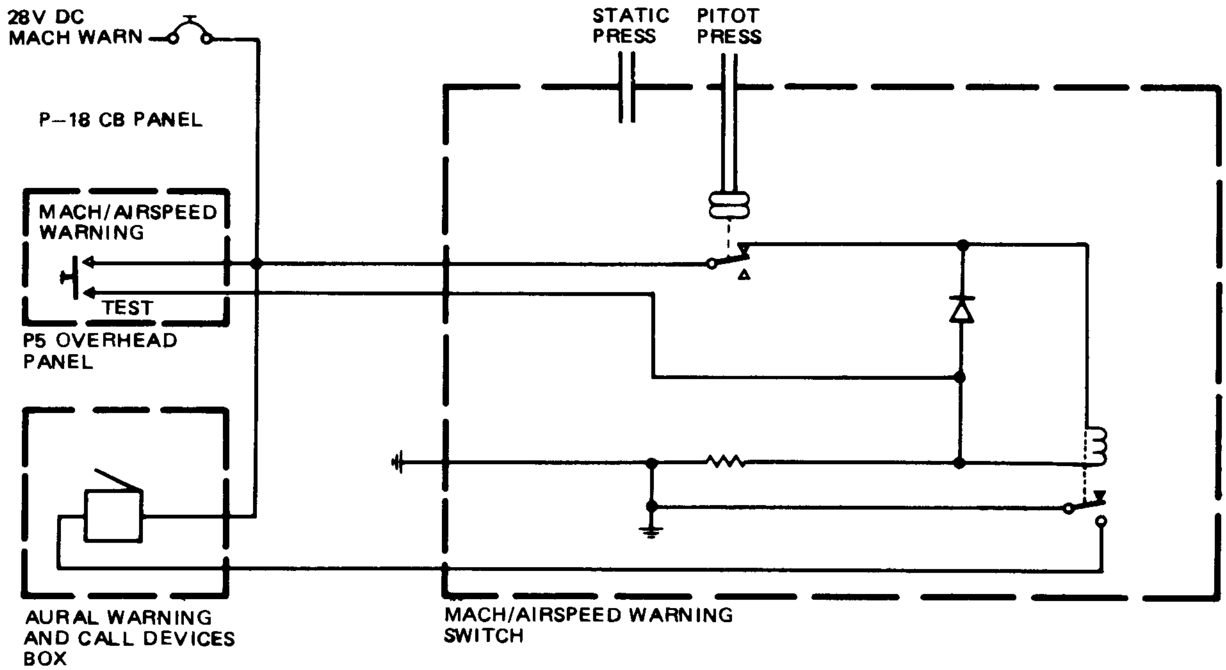


MACH/AIRSPEED WARNING
TEST SWITCH
VIEW 1



AURAL WARNING AND CALL DEVICES BOX
DETAIL A

Mach/Airspeed Warning System Component Location
Figure 1



Mach/Airspeed Warning System Circuit
Figure 2

MACH AIRSPEED WARNING SYSTEM - ADJUSTMENT/TEST

1. Mach Airspeed Warning System Test
   A. Equipment and Materials
      (1) Air pressure and vacuum source - 5 to 40 inches of mercury absolute, with adapters for connection to pitot and static sources on pitot static tube or to test fittings
      (2) Gage for measuring differential pressure accurate within ± 0.010 inch of mercury with a range of at least 10 inches of mercury
   B. Prepare to Test Mach Airspeed Warning System
      (1) When applying pitot pressure, note following:
         (a) Pitot-static tube heaters should remain OFF during these tests.
         (b) Pitot line pressure over static line pressure should not exceed 9 inches of mercury. Static line pressure must never exceed pitot line pressure.
      (2) When applying static pressure, note following:

CAUTION: CABIN ALTITUDE DIFFERENTIAL PRESSURE INDICATOR SHOULD BE DISCONNECTED FROM ITS STATIC PRESSURE SOURCE AND SOURCE SHOULD BE CAPPED.

         (a) Static source selector valves should be in NORMAL position and should not be changed while vacuum is on static system.
         (b) Do not apply pressure in excess of 31 inches of mercury absolute to static lines.
         (c) Apply or release vacuum at a rate of climb or descent of approximately 3000 feet per minute or less between test points. At each test point, pressure should be reduced slowly to desired level without overshoot.
         (d) Seal static port on pitot-static probe surface in such a way that removal of seal will be complete, leaving no deposits or roughness in or about static ports.
         (e) Apply pressure to system using pitot-static probes by performing steps (f) thru (h) or using system drain fittings by performing steps (i) thru (l).
         (f) Connect vacuum source with adapter to No. 1 auxiliary static port on lower right pitot-static probe.
         (g) Seal No. 1 auxiliary static port on upper left pitot-static probe.
         (h) Connect positive pressure source with adapter to No. 1 auxiliary pitot port on lower right pitot-static probe.
         (i) Connect vacuum source with adapter to No. 4 drain fitting located on the right side of E1 electronics rack.
         (j) Seal No. 1 auxiliary static port on lower right and upper left pitot-static probes.

(k) Connect positive pressure source with adapter to No. 3 drain fitting located on right side of E1 electronics rack.
(l) Seal No. 1 auxiliary pitot tube port on lower right pitot-static probe.
(m) Connect differential pressure gage into pitot and static test lines.
(n) Disconnect cabin differential pressure indicator and cap static pressure tube.

C. Test Mach Airspeed Warning Systems

CAUTION: WARNING DEVICE (CLACKER) SHOULD NOT BE ENERGIZED FOR MORE THAN 5 MINUTES OR DAMAGE MAY RESULT. ALLOW A MINIMUM COOLING PERIOD OF 15 MINUTES BEFORE ENERGIZING SYSTEM AGAIN.

(1) Close MACH WARN circuit breaker. Push MACH AIRSPEED WARNING TEST switch located on flight recorder and mach airspeed warning test module (P5 overhead panel). Clackers should come on.
(2) Release test switch. Clackers should go off.
(3) Apply pressure as shown in Table I. Clackers should come on within tolerances indicated and go off when differential pressure is reduced below lower tolerance value.

CAUTION: DIFFERENTIAL PRESSURE SHOULD BE BETWEEN 0 AND 10 INCHES OF MERCURY AT ALL TIMES.

(4) Release vacuum at specified rate.

TABLE I

ALT (FEET) (REF)	STATIC PRESSURE IN. HG ABSOLUTE	PITOT PRESSURE IN. HG ABSOLUTE	DIFF PRESSURE IN. HG ABSOLUTE	INDICATED AIRSPEED KNOTS
Sea Level	29.921	36.207-36.441	6.286-6.520	350.0-356.0
20,000	13.750	20.562-20.806	6.812-7.056	363.4-369.4
30,000	8.885	14.104-14.252	5.219-5.367	320.7-324.9
35,000	7.041	11.173-11.295	4.132-4.254	287.0-291.0

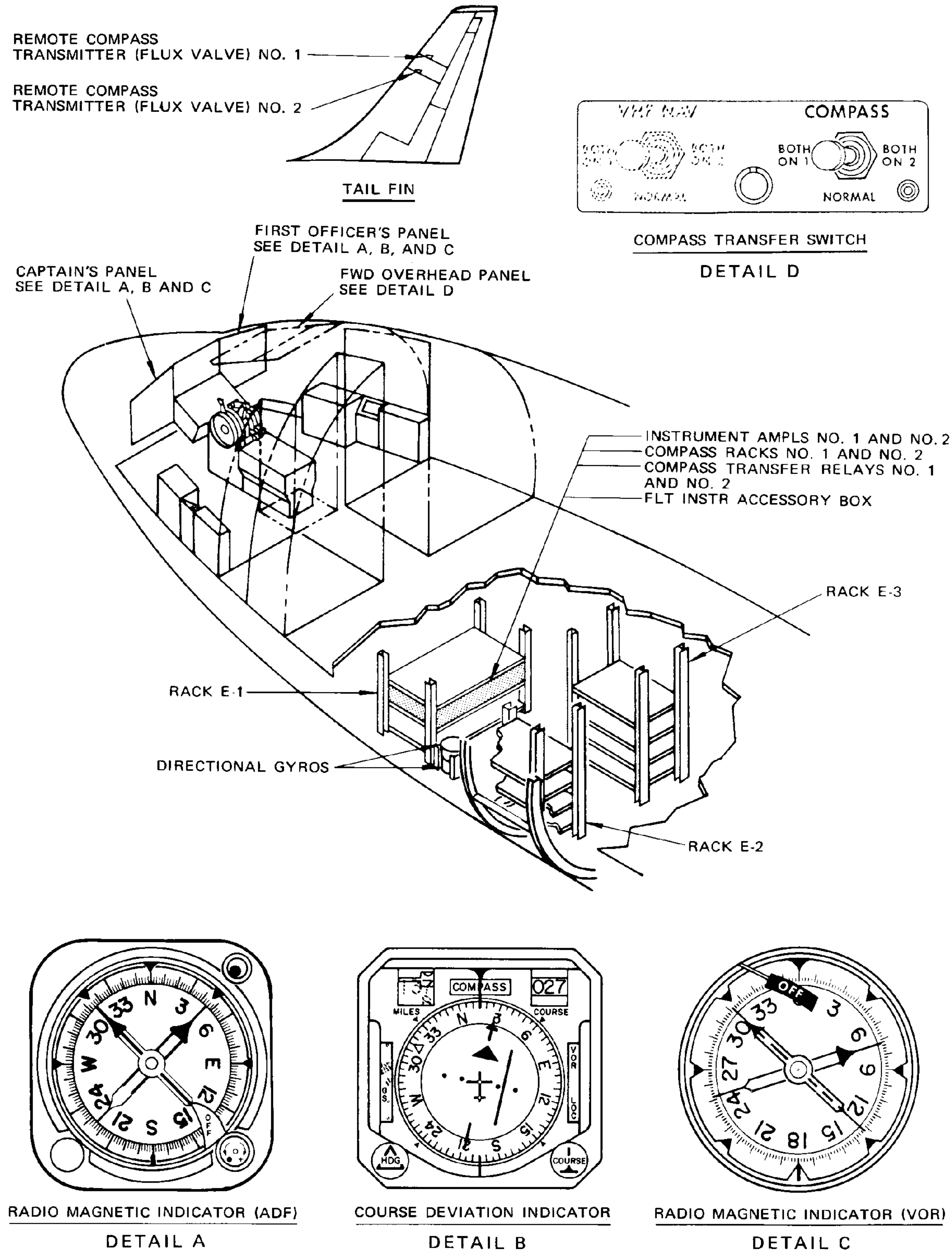
D. Restore Airplane to Normal Configuration

(1) Restore mach airspeed warning and pitot-static systems to normal.

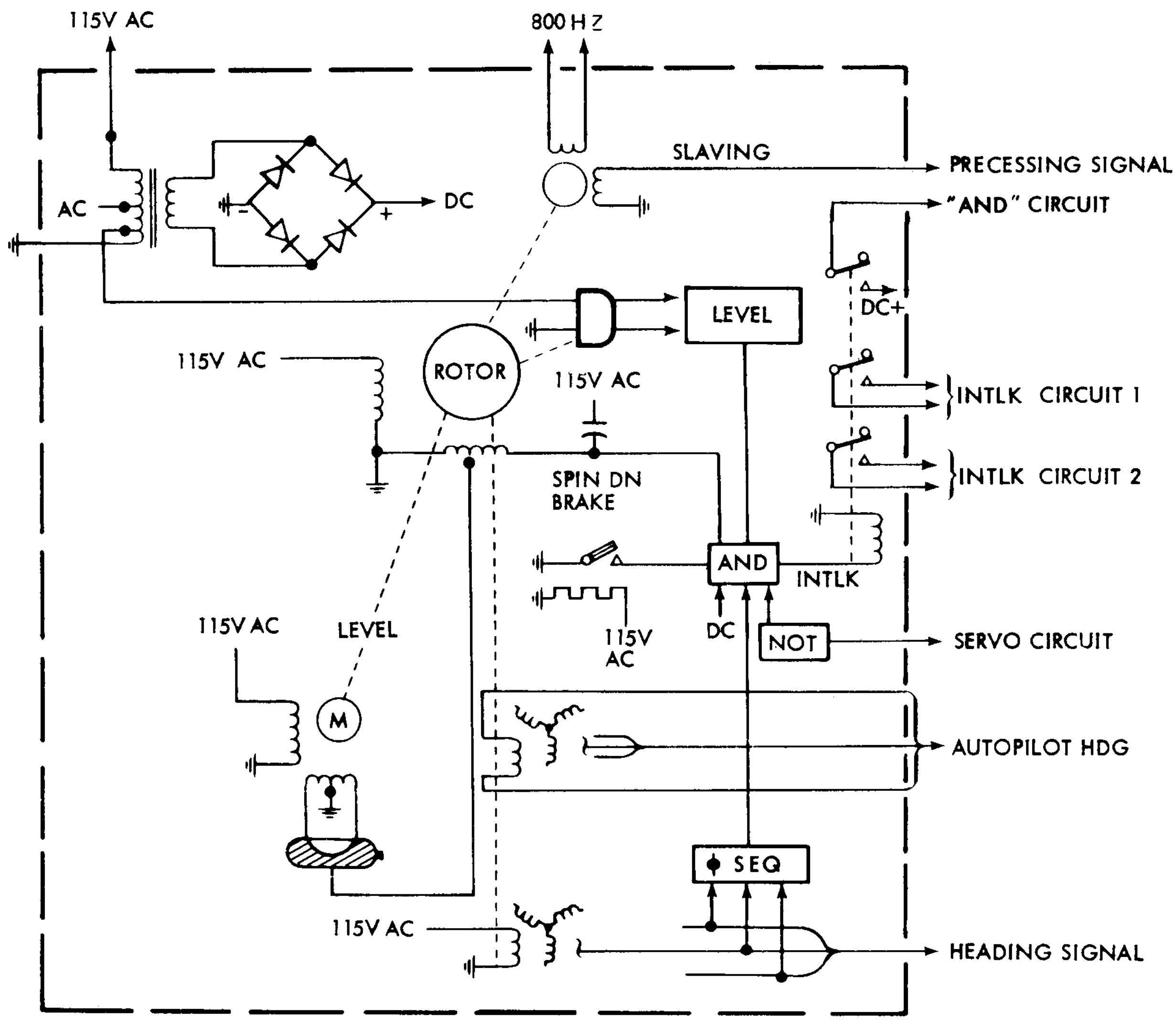
(2) If no longer required, remove electrical power from airplane.

COMPASS SYSTEMS - DESCRIPTION AND OPERATION

1. General
    A. The compass system is designed primarily to furnish the captain and first officer (F/O) with information concerning the airplane's magnetic heading at all times during flight. Heading information is displayed on the airplane's ADF radio magnetic directional indicators (ADF RMIs), VOR radio magnetic directional indicators (VOR RMIs) and course deviation indicators (CDIs).
    B. Two compass systems are installed. Each consists of the following components; remote compass transmitter (flux valve), directional gyro, ADF RMI, VOR RMI, CDI, servo-amplifier, and slaving amplifier. Location of the components is shown in figure 1.
    C. Each ADF RMI feeds heading information to a VOR RMI and a CDI. Each CDI feeds heading information to the autopilot, flight recorder, navigation systems, flight director systems, and comparator unit.
    D. A compass transfer switch is provided to connect the captain's ADF RMI to the F/O's ADF RMI in the event of a malfunction in compass system No. 1. here are two switch positions; COMPASS NO. 1 and COMPASS NO. 2.
2. Remote Compass Transmitter (Flux Valve)
    A. The flux valve contains a pendulous magnetic detector which senses the direction of the horizontal component of the earth's magnetic field and utilizes this reference to generate an ac signal representative of the airplane's magnetic heading.
3. Deleted
4. Directional Gyros
    A. The directional gyro provides the heading reference for the compass system. Its gimbals may be torqued by two torquemotors, the slaving torquemotor being used to precess the gyro to the magnetic reference generated in the flux valve and the leveling torquemotor to level the gyro with respect to gravity. The former motor is driven by the slaving amplifier and the latter is controlled by means of an electrolytic switch. (See figure 2.)
    B. A spin-down brake in the directional gyro prevents nutation of the gyro by holding the outer gimbal steady for 10 seconds after power is applied. At shutdown, the brake is reapplied to the gimbal 30 seconds after power is removed. This prevents the gyro from spinning around its vertical axis. The 30-second time delay of the spin-down brake at shutdown also prevents momentary power interruptions (such as changing inverters) from introducing errors into the heading information.
    C. A logic circuit monitoring the gyroscope wheel speed, gyroscope leveling, nutation brake operation, heading synchro excitation and compass system signal null are used to operate two malfunction relays. The contacts of these relays are used in the flag warning circuits.
    D. Heading information for autopilot system is obtained from either one of the gyros through the compass transfer switch.



Compass System Component Location
Figure 1



Directional Gyro Schematic
Figure 2

5. ADF Radio Magnetic Indicator (ADF RMI)
   A. Heading information from the directional gyro is displayed on a rotating compass dial which is read against a fixed index at the top of the indicator. Synchro transmitters in the captain's ADF RMI repeat this information to the captain's CDI and to the F/O's VOR RMI. The F/O's ADF RMI provides a similar service for the F/O's CDI and captain's VOR RMI and when the compass transfer switch is in No. 2 position feeds the captain's CDI.
   B. An annunciator is provided to indicate alignment of the compass card with the magnetic reference. It takes the form of a small bar which will point to a cross or a dot whenever slaving of the gyro is taking place. (See figure 1.) An indication midway between cross and dot signifies that the system is synchronized.
   C. The synchronizing knob on the face of the RMI is provided for rapid synchronization of the system. This is done by turning the knob in the direction indicated by the annunciator (dot or cross).
   D. The warning flag on the RMI will appear when power is lost to the system or in the event of a malfunction in the system.
   E. The RMI pointers display information from the ADF systems.
6. VOR Radio Magnetic Indicator (VOR RMI)
   A. Heading information is fed to each VOR RMI from the opposite system's ADF RMI. Heading information is displayed on the indicator's compass dial. For a more detailed description of this indicator, refer to 34-31-0, VOR/GS Navigation System.
   B. The RMI pointers display information from the VOR systems.
7. Course Deviation Indicator (CDI)
   A. One of the synchros in the captain's CDI receives heading information from the captain's ADF RMI. This synchro is used in conjunction with the servo-amplifier to control the read-out of the CDI compass card. The other two synchros have their stators connected in parallel and are fed from another synchro in the captain's ADF RMI. These two synchros are equipped with set course and set heading knobs in order that their rotors may be preset accordingly. The outputs are fed to the autopilot. The same synchros in the F/O's CDI stators are similarly connected to the F/O's ADF RMI. The outputs from the two preset synchros are fed to the flight director system, and autopilot system. Refer to Chapter 34-26-02, Flight Director System Autopilot System, Chapter 22.
   B. The captain's CDI flag gets its dc power from circuits in the airplane's accessory box. The power is fed via a monitor circuit in Compass Rack No.1. If either the power circuits or the monitor circuit fails, the flag will appear. The F/O's CDI flag is also monitored in this way.
8. Rack Mountings
   A. The compass racks provide "plug-in" mountings for the servo-amplifiers and slaving amplifiers. (See figure 1.) Each rack is wired into its own compass system.

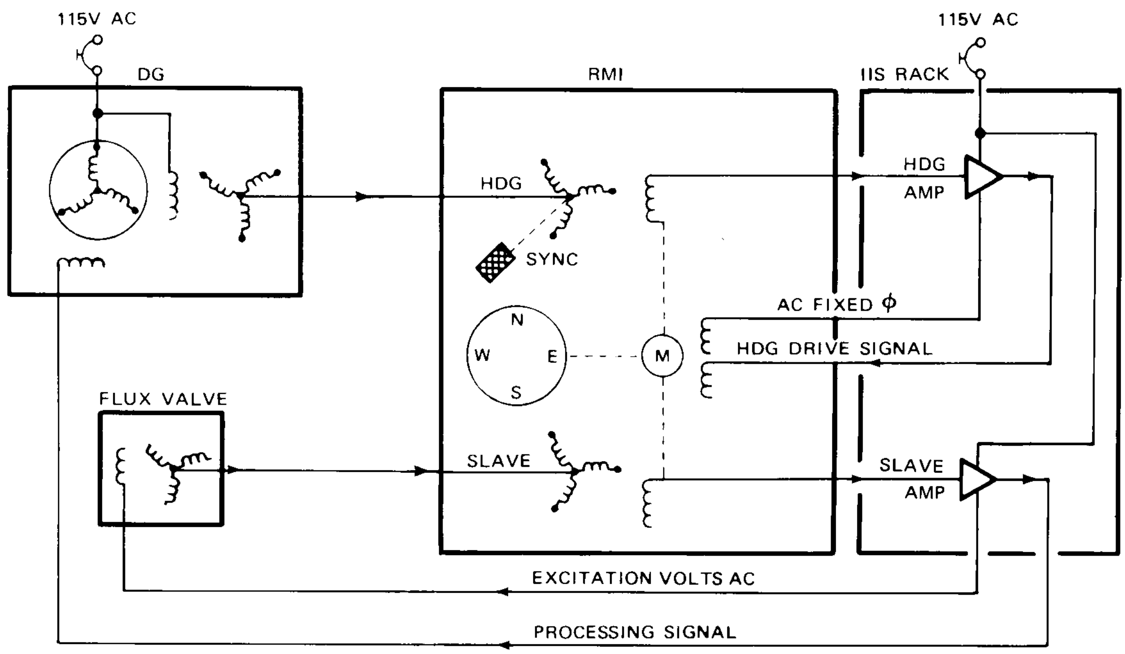
9. Operation

A. Both systems are made operational by closing the system circuit breakers. After a brief period of time the gyros will run up and the compass cards will reflect the magnetic heading of the airplane. (See figure 3.)

B. Assume that the system is activated and that the airplane heading is magnetic north. This heading will be displayed on the RMI and CDI compass cards. Assume now that the airplane turns 10 degrees to the left. The synchro in the directional gyro responds immediately to the change in direction, and puts out a signal which is now related, in both phase and amplitude, to the new heading. This signal is fed to the servo-amplifier via the heading synchro in the RMI. (Refer to figure 3.) The servo-amplifier then drives the servomotor in the RMI (which is geared to the compass card) towards the new heading, and at the same time turns the rotor of the servocontrol synchro in such a direction as to null out the error in the servo circuit. Thus, when the motor stops, the signal from the synchro in the directional gyro has been nulled out in the RMI servocontrol synchro, and the compass card in the RMI reflects the new airplane heading.

C. The sensing element in the flux valve will also reflect this change in heading, in the form of a signal which is fed to the slaving amplifier via the slaving synchro in the RMI. However, the error in the slaving amplifier circuit is also nulled out by the servomotor in the RMI since both synchro rotors are driven simultaneously by the same servomotor.

D. Random drift in the gyro is overcome by means of the flux valve and servo slaving circuits. Assume there are no error signals in the system and that the airplane is headed towards magnetic north once more. The gyro now drifts slightly off course. This results in a signal being generated in the gyro synchro which is passed to the servo-amplifier via the heading synchro in the RMI. As in the case of the heading change, the servo-amplifier drives the servomotor to null out the error in the heading synchro and automatically turns the rotor of the slaving synchro in the RMI along with it. However, in this case the stator of the slaving synchro is still referenced to north by the flux valve; hence, an error signal is now generated in the slaving synchro which is amplified by the slaving amplifier and fed to the torquemotor to precess the gyro back to a northerly heading. The error in the gyro synchro is thus nulled out and the system returns to its original zero error condition with the gyro again "pointing" north.

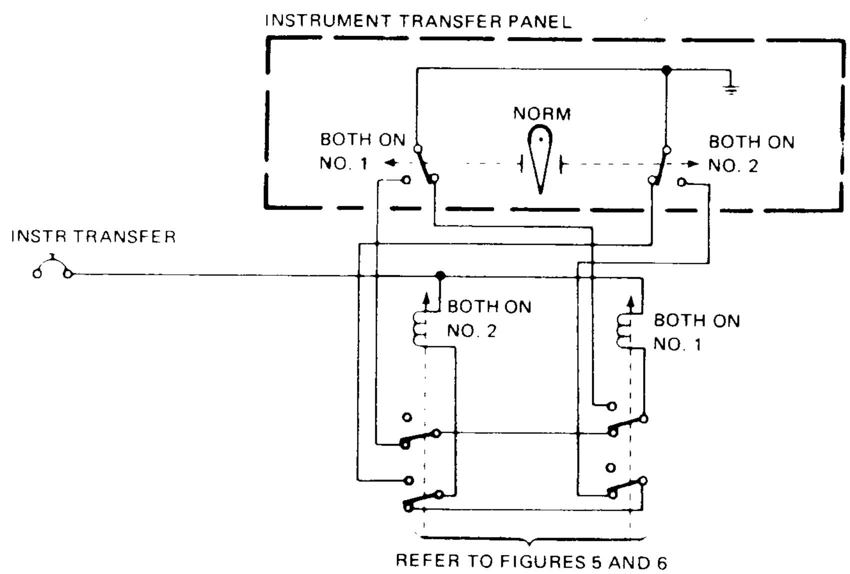


Basic Compass System Circuit
Figure 3

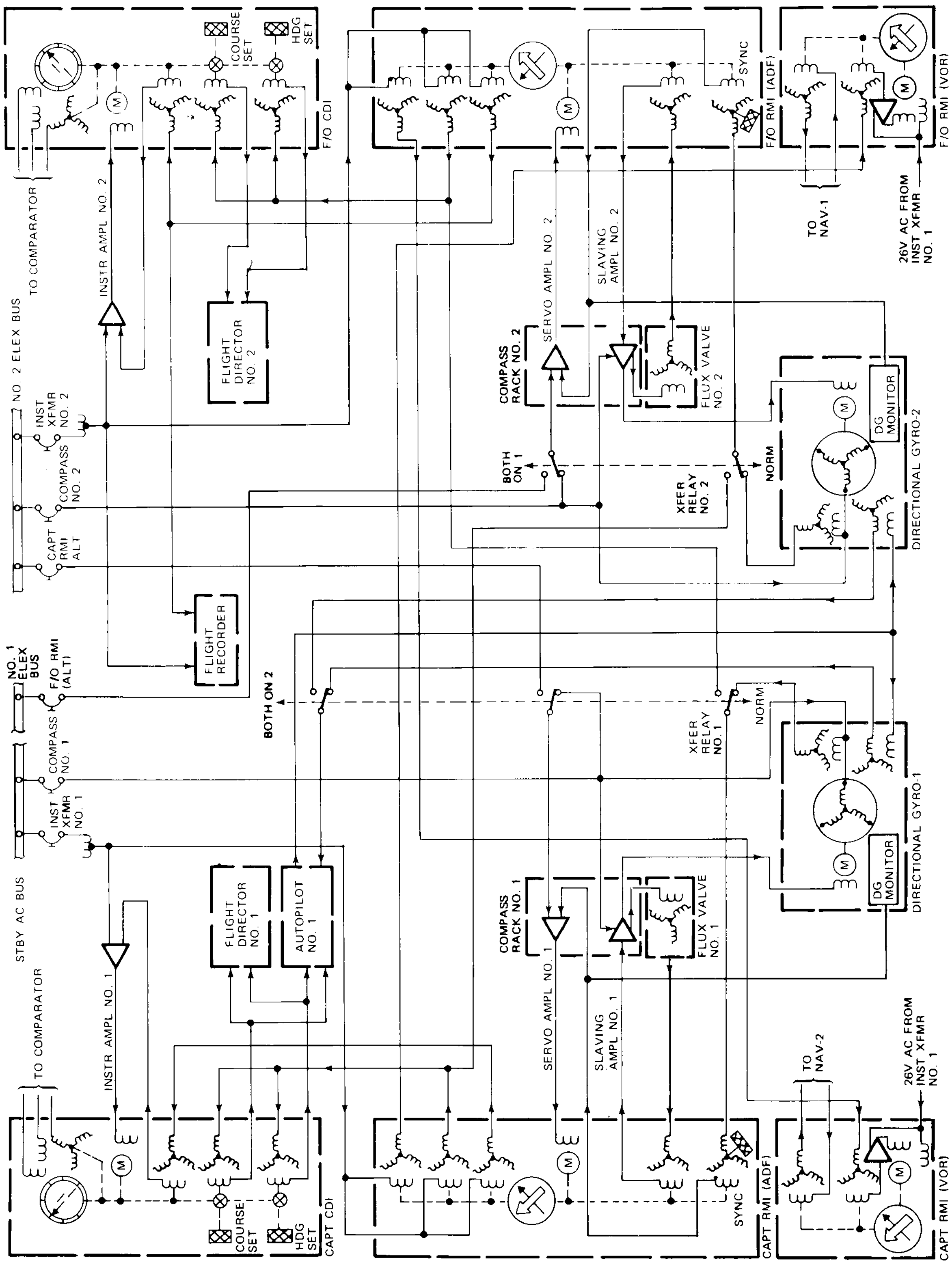
E. In practice, when the compass system is first activated, some disagreement will exist between the airplane heading and that of the directional gyro. The system will immediately start to precess the gyro to take out this error, but the precession rate for the slaving circuits is a relatively slow one, and the time lag involved could not always be tolerated. Hence, a "manual" method of synchronizing the system has been devised. The method used is to introduce an artificial error signal into the heading servo circuit, which rapidly produces the desired results. Assume that the system has been activated and that the aircraft heading is magnetic north, whereas the compass cards are 10 degrees "out of line." An annunciator indication of a dot or cross, will show that the slaving circuits are operating, and will also serve to indicate which way the HDG/SYNC knob should be turned to synchronize the system. When the knob is turned, it turns the stator of the heading synchro, thus generating an error voltage in the rotor which is amplified by the servo-amplifier and drives the servomotor and compass cards to the correct airplane heading (and nulls out the error in the heading synchro as before). Since the servo-amplifier circuit is able to respond far more rapidly to an error signal than the slaving circuit, the "manual" method of synchronizing the system is preferred.

F. The operation of the compass transfer switch and relays is illustrated in figures 4, 5 and 6. Figure 4 shows how the relay coils are connected with the dc supply through the relay contacts. The switch is shown in NORMAL position. Assume that the transfer switch is now set in BOTH ON NO. 2 position. This switch puts a ground on the lower end of the BOTH ON NO. 2 latching relay coil via BOTH ON NO. 2 and BOTH ON

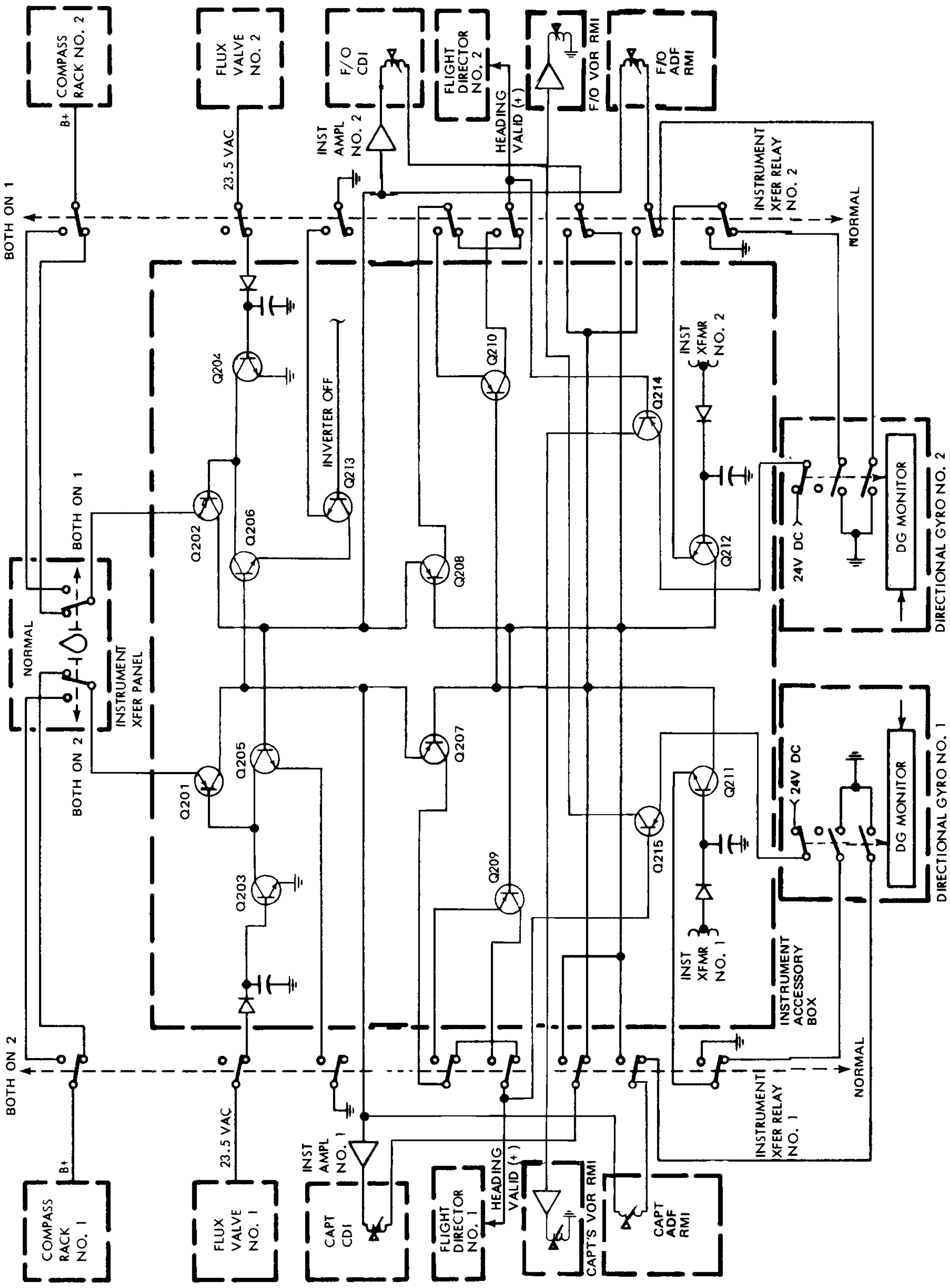
G. No. 1 relay contacts. Since the other end of the coil is connected to 28 volts dc, it will now latch over to the alternate set of contacts and remain until energized once more. Note that the ground is removed from the coil by the transfer of BOTH ON NO. 2 relay contacts. In order to ground the coil once more and switch the relay back again, the switch must be returned to NORMAL position. The switching to BOTH ON NO. 1 position is carried out in a similar manner. The switching contacts shown in Fig. 5 are a continuation of those in Fig. 4., but they take care of the transfer of signals within the system. In the normal configuration, the captain's RMI receives its signals direct from directional gyro No. 1 and the first officer's RMI receives its signals from directional gyro No. 2. AC power for each system servo-amplifier is taken from COMPASS-1 and COMPASS-2 circuit breakers respectively. When the transfer relay is switched over to BOTH ON NO. 2, the first officer's system receives its signals and power supplies as before but the captain's system now receives its signals from a synchro in the first officer's RMI and the power supply for No. 1 servo-amplifier is taken from the captain's RMI (ALT) circuit breaker. A similar transfer takes place on switching to BOTH ON NO. 1. The autopilot normally gets its heading reference from No. 1 directional gyro, but in the BOTH ON NO. 2 position this signal is taken from directional gyro No. 2.



Compass Switching Circuit
Figure 4



Compass System Schematic
Figure 5



Compass Warning Flag Logic Circuits
Figure 6

H. The compass system contains flag warning circuits (figure 6) which are used to detect and display malfunctions. Malfunctions are displayed by warning flags on the captain's and first officer's ADF and VOR Radio Magnetic Indicators (RMI's) and Course Deviation Indicators (CDI's). A heading valid signal is also sent to the captain's and first officer's Flight Director Systems. When all components are operating normally, the warning flags are not in view. The flag circuitry monitors the No. 1 and No. 2 compass racks, flux valves, instrument transformers, and Directional Gyro (DG) monitors. Power for the circuits is provided by full wave rectifiers in the compass racks.

I. With all components operating normally and the compass transfer switch in NORMAL position (as shown in figure 6), the circuits to hold the captain's CDI and RMI's out of view are as follows. Positive voltage to the flag coil comes from compass rack No. 1 through transfer relay No. 1, compass transfer switch, and Q201 to the captain's ADF RMI. It also goes to instrument amplifier No. 1 and then to the CDI. Ground to the captain's ADF RMI flag coil comes from DG monitor No. 1 through transfer relay No. 1. Ground to the captain's CDI flag coil comes from DG monitor No. 1, through transfer relay No. 1, Q211, and transfer relay No. 1. Positive voltage and ground to the first officer's CDI and ADF RMI flag coils comes from compass rack No. 2 and DG monitor No. 2 respectively and go to the first officer's flag coils in the same manner described above for the captain's flag coils. Positive voltage to the captain's VOR RMI flag coil comes from directional gyro No. 2 and transistor Q214 when heading valid signal No. 2 is present to Q214. Positive voltage to the first officer's VOR RMI flag is similarly applied.

J. From the circuits described above, if there is a malfunction in compass rack No. 1, the positive voltage to Q201 is lost and the warning flags will appear in the captain's CDI and ADF RMI and first officer's VOR RMI. If there is a malfunction in flux valve No. 1, positive voltage to Q203 is interrupted resulting in a loss of ground to Q201 which causes the same flags to appear. If there is a malfunction in DG monitor No. 1, the ground to the captain's CDI and RMI flag coils is interrupted causing the warning flag to appear. If a malfunction occurs in the instrument transformer No. 1, the positive voltage to Q211 is lost and the ground is removed from the captain's CDI flag coil. Positive voltage to Q215 is also lost removing the positive voltage on the first officer's VOR RMI flag coil. A malfunction in the No. 1 compass rack, flux valve, instrument transformer, or DG monitor will interrupt the heading valid signal to flight director system No. 1.

K. With all components operating normally and the compass transfer switch in BOTH ON NO. 2 position, the circuits to hold the captain's CDI and RMI's flag out of view that differ than when in NORMAL position are as follow. (Circuits to hold the first officer's CDI and ADF RMI flag coils do not change.) Flux valve No. 2 supplies a positive voltage to Q204 which places a ground on Q202. The positive voltage output of Q202 is applied to Q205 which sends a ground to Q201 to replace the ground from Q203. The ground to the captain's CDI and ADF RMI flag coils come from DG monitor No. 2 through transfer relay No. 2, Q212, and transfer relay No. 1. This same ground from Q212, switches Q209 on to control heading valid No. 1 which controls the positive voltage to the first officer's VOR RMI flag coil.

L. From the above description, if there is a malfunction in No. 2 compass rack, DG monitor, or flux valve, all warning flags will come into view. If there is a malfunction in instrument transformer No. 2, all warning flags will come into view except the first officer's ADF RMI. A malfunction in the No. 2 compass rack, flux valve, instrument transformer, or DG monitor will interrupt the heading valid signals to flight director systems No. 1 and No. 2.

COMPASS SYSTEM - TROUBLESHOOTING

1. General
   A. The location of troubles in the compass system can best be determined by a systematic functional test. Use of unit electrical schematics, block diagrams, and electronic wiring diagrams of the compass system will aid in troubleshooting analysis. Troubleshooting the compass system is best accomplished by substituting components where possible.
   B. The flux valve (remote compass transmitter) is extremely sensitive to magnetic fields and will malfunction if it is near magnetic materials which distort the earth's magnetic field.

      CAUTION: DO NOT CHECK THE FLUX VALVE WITH AN OHMMETER. DIRECT CURRENT IN THE WINDINGS MAY MAGNETIZE THE CORES.

   C. Before proceeding with the troubleshooting procedures, recheck that the compass transfer switch is in normal position and all compass system circuit breakers are closed. Make sure that cable connectors are properly mated and securely connected. Make sure that 400-cycle ac synchro excitation is present to all units. When substituting known good units in place of suspected malfunctioning units, leave replacement units in the circuit until trouble is cleared and then replace original good units into the operating system.
   D. The following troubleshooting procedures are for compass system No. 1. If trouble is encountered in compass system No. 2, use the same procedure substituting, in the procedure, with an equivalent compass system No. 2 component.

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Capt's CDI compass card does not repeat ADF capt's RMI compass card	Instrument amplifier No. 1 capt's CDI, capt's ADF RMI	Rotate capt's ADF RMI synch knob. Capt's CDI compass card does not respond to RMI readings	Replacement instrument amplifier No. 1
			Replace capt's CDI
			Replace capt's ADF RMI
Capt's ADF RMI compass card will not drive or gives erratic readings	Servo amplifier No. 1 or capt's ADF RMI	Rotate capt's ADF RMI synch knob, capt's ADF RMI compass card does not respond.	Replace servo amplifier No. 1.
			Replace capt's ADF RMI
	Directional gyro No. 1	Capt's RMI compass card responds to synch knob rotation but gives erratic readings	Replace directional gyro No. 1
Capt's ADF RMI will not synchronize		Rotate capt's ADF RMI synch knob toward dot or plus	
	Servo amplifier No. 1	Capt's RMI compass card does not respond to synch knob rotation	Replace servo amplifier No. 1
	Capt's ADF RMI	Capt's ADF RMI compass card rotates but does not return to synchronized position	Replace capt's ADF RMI
Capt's ADF RMI and CDI do not indicate airplane heading		Rotate capt's ADF RMI synch knob	
	Servo amplifier No. 1 or capt's ADF RMI	If capt's ADF RMI and CDI compass cards do not respond to synch knob rotation	Replace servo amplifier No. 1
			Replace Capt's ADF RMI or F/O's CDI

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Capt's ADF RMI and CDI do not indicate airplane heading	Slaving amplifier No. 1 or directional gyro No. 1	Capt's ADF RMI and CDI compass cards follow synch knob rotation but will not return to synchronized position	Replace slaving amplifier No. 1
			Replace directional gyro No. 1
		If above isolation procedure failed to correct trouble, replace components in order listed	Directional gyro No. 1
			Flux valve No. 1
	Damaged airplane wiring	If replacement of components failed to correct trouble, check airplane wiring	Refer to electrical wiring diagrams

CAUTION: DISCONNECT FLUX VALVE BEFORE MAKING CONTINUITY CHECKS TO THE FLUX VALVE AIRPLANE WIRING.

COMPASS SYSTEM - MAINTENANCE PRACTICES

1. General
   A. This procedure describes the manner in which the airplane compass systems are "swung". There are two main systems, the captain's and first officer's, plus a standby compass. A method of compensation is also included for airplanes fitted with remote compensator units.

      NOTE: Disregard flux valve compensator adjustments on airplanes without remote magnetic compensator.

   B. The procedure is written so that the swing for the standby compass system is described separately from that for the main systems. However, all systems may be swung simultaneously, if so desired.
   C. The process of "ground swinging" an airplane compass consists of accurately aligning the airplane with at least four cardinal headings in turn, and observing the deviation between known heading and compass heading, in each case. The results are then tabulated, and after all necessary adjustments have been made to reduce the deviations to specified limits, another circuit is made for the purpose of compiling the compass correction card. This card remains with the compass.
   D. Deviation errors fall into two categories, "index error" and "single cycle error", each of which is summarized below.
      (1) An index error exists when the direction error remains constant, both in direction and value at each checkpoint.
      (2) A single cycle error exists when the direction error is equal, but opposite in sign, at opposite checkpoints on the compass.
   E. The airplane initial compass swing will normally require at least two complete circuits around the compass rose, once to take out the index error and the single cycle error, and once to make the compass correction chart. Passenger cabin seats and cockpit seats shall be installed prior to swinging.
2. Swing Main Compass System
   A. General
      (1) The tests described under Compass System - Adjustment/Test, should be carried out prior to this procedure, to make sure that a serviceable compass system is being swung.
         (a) Replacement of an RMI or CDI will not necessitate performing a compass swing if the unit has been properly tested at the component level.
         (b) Replacement of a flux valve does not require a compass swing on the compass rose. To make sure that flux valve is operating properly, turn airplane 360 degrees and observe RMIs. Make sure that one system closely follows the other.

(2) Index errors may be eliminated or reduced by adjusting the differential synchro in the remote magnetic compensator (if installed).

(3) Single cycle errors may be eliminated or reduced by adjusting the E-W and N-S potentiometers on the compensator (if installed).

B. Equipment and Materials

(1) Compass rose having the following features.

(a) A flat, obstruction free area, large enough to permit towing the airplane in a circle around a surveyors transit, and still leave 25 feet between its outer wing tip and the edge of the area (Fig. 201). The direction of the horizontal component of the earth magnetic field in the area should be constant within 0.2 degree throughout the test area.

(b) A baseplate located in the center of the area to identify the position for the transit.

(2) Two heading markers, whose magnetic bearing from the baseplate has been established

(3) Two tractor

(a) Control over an area extending 180 feet in all directions, from the rose center, which may be used to exclude any type of traffic during a compass swing.

(4) Surveyors transit or equivalent (accurate to 1 minute or less)

(5) Compass Fixture Set - TE 65-73701 or equivalent (Fig. 201)

(6) Nonmagnetic tools for adjusting standby compass

(7) One compass correction card (AN5823) for each compass

(8) 3-phase, 115-volt, 400-cps external power supply (5 KVA capacity)

C. Prepare for Compass Swing

(1) Assemble and attach compass swing target fixture to airplane as follows:

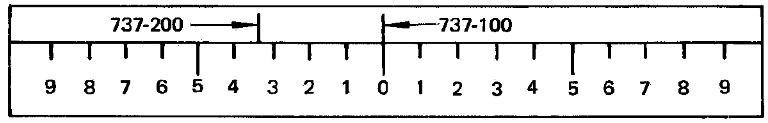
(a) Remove apex bolt connecting upper and lower torsion links of nose gear. Move upper torsion link down to release spring tension in piston position spring cartridge. Remove upper bolt of piston position spring cartridge and release from position.

<u>CAUTION</u>: THE PISTON POSITION SPRING CARTRIDGE IS SPRING LOADED TO EXTENDED POSITION. THEREFORE, UPPER TORSION LINK SHOULD BE HELD WHILE APEX BOLT IS BEING REMOVED.

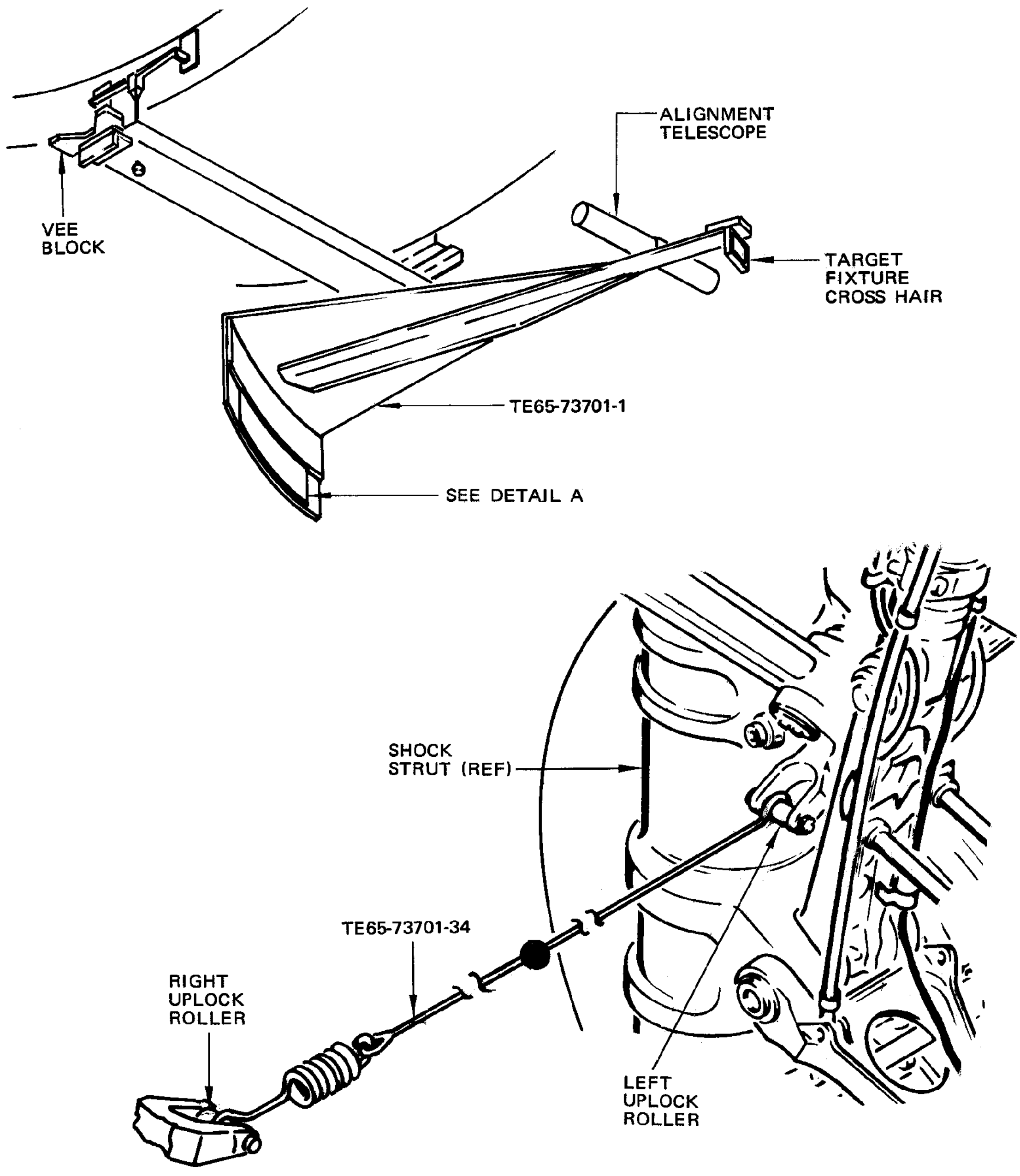
(b) Adjust sliding scale on fixture so that its 0 position is located on -100 or -200 mark (to agree with airplane type body). Lock scale in position.

(c) Position fixture under electronics compartment hatchway with vee block pointing forward.

(d) Raise fixture and engage guide pin in latchpin hole in the forward striker plate inside the hatchway.



DETAIL A



Compass Swing Target Fixture
Figure 201

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(e) Attach vee block to aft side of nose gear shock strut. Secure vee block in position with its pin.
(f) Attach target ball to the uplock rollers on the main gear struts with cable attachments. When in position, cable should be taut and ball situated to left of airplane centerline (Fig. 201, 202).
(g) Make sure that lockscrews on target fixture are loose, to permit movement of fixture from side to side using vee block as a pivot point.
(h) Sight through telescope and adjust fixture in azimuth until ball is centered in telescope sight. Tighten lockscrew in fixture below guide pin to ensure that crosshair and 0 mark on scale will remain at a fixed angle to centerline of airplane.

(2) Fit ground downlocks to landing gear and tow airplane around compass rose at least once, then position airplane on compass rose to meet following conditions:
(a) Airplane heading within 2 degrees of magnetic north
(b) Heading roughly tangential to a circle drawn about transit base.
(c) Compass swinging fixture readily seen from transit base (Fig. 202).

(3) Provide external electrical power to airplane.

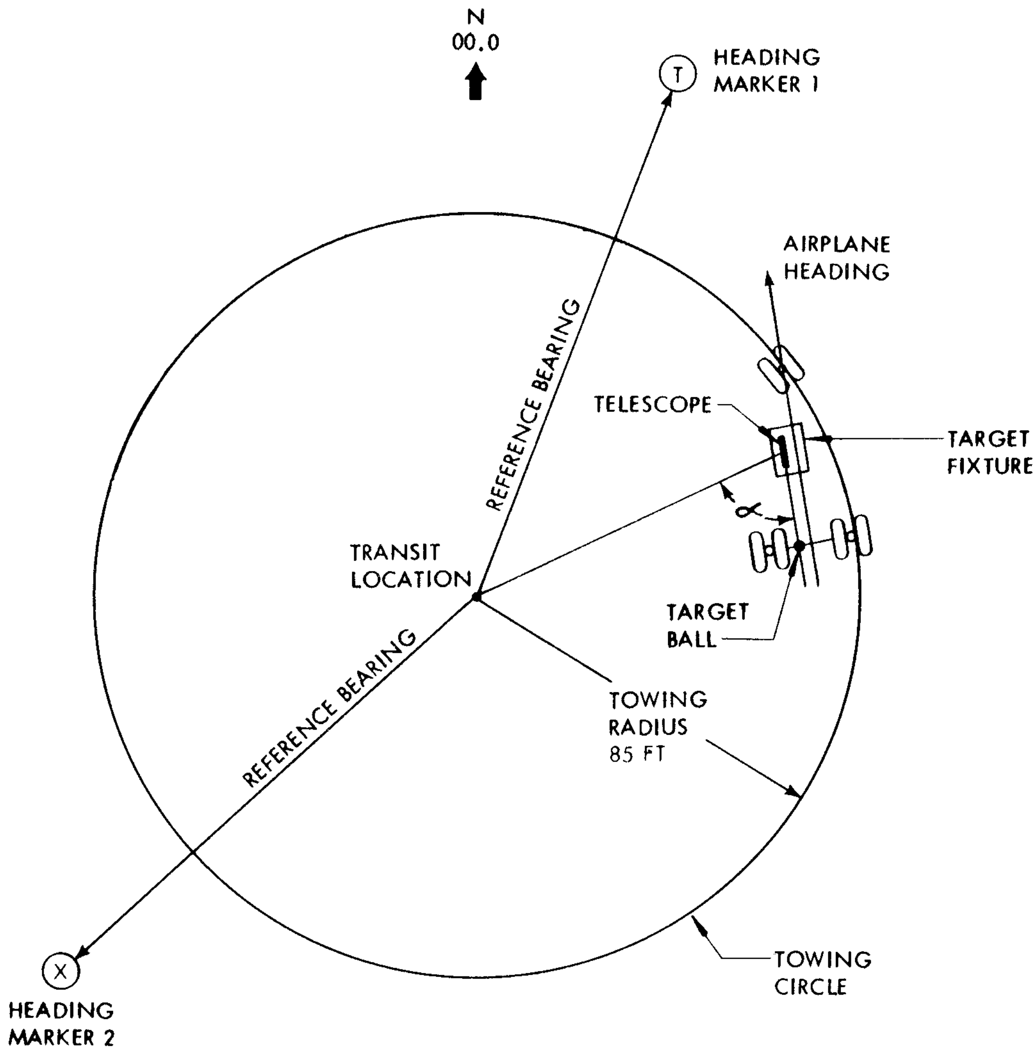
(4) Energize both compass systems by closing circuit breakers listed in Compass Systems - Adjustment/Test. Set compass transfer switches, if fitted, to normal configuration for the systems.

(5) Set up transit on baseplate in center of compass rose and calibrate as follows:
(a) Level off transit and sight on marker No.1. Unlock transit lower motion and set scale to read magnetic bearing. of marker minus 66 degrees for 737-200 airplanes and 67 degrees, 58 minutes for 737-100 airplanes. Lock lower motion.

NOTE: If bearing thus calculated results in a minus quantity; add 360 degrees to it in order to validate it.

(b) Check transit calibration with marker No. 2. Transit should read marker's magnetic heading minus 66 degrees for 731-200 airplanes and 67 degrees 58 minutes for 737-100 airplanes. Correct as necessary.
(c) Repeat steps (a) and (b) until transit is correctly calibrated.

NOTE: The transit is purposely calibrated to read magnetic heading minus 66 degrees for 737-200 airplanes and 67 degrees 58 minutes for 737-100 airplanes, since it is used to measure airplane heading through the target fixture.



Target Fixture and Transit Arrangement
Figure 202

(6) Place radio power switches to ON. Select a localizer frequency on VHF NAV control panel.

(7) Set flight director MODE selector to VOR-LOC and weather radar MODE selector to STBY.

(8) Make sure that landing gear downlocks are in place. Energize landing gear lever latch by placing a mild steel slug against sensitive surface of air sensing sensor (S106). Do not move landing gear lever while solenoid is energized. The solenoid shall be energized at least 20 minutes before proceeding with test.

D. Swing Compass

(1) Set up remote magnetic compensator as follows (if installed):

(a) Set compensator transmission control to 0, and turn slotted shaft in center of transmission control fully clockwise then 1/4 turn counterclockwise.

(b) With a multimeter set to DC, connect DC probe to E-W terminals TP2 (+ve) and TP5 (-ve) of compensator. Adjust E-W control until multimeter indicates a null.

(c) Connect DC probe to N-S terminals TP1 (+ve) and TP5 (-ve) and adjust N-S control until multimeter indicates a null.

(2) Make sure that compass transfer switch is in NORMAL.

(3) Using sync knobs, set captain's and first officer's master compass cards to airplane heading. The annunciators shall indicate a null.

(4) Determine true magnetic heading of airplane and North Heading Deviation for each compass.

(a) Sight transit on crosshair of target fixture and record transit angle T°.

(b) Sight through fixture crosshair (with transit) to fixture scale. Read and record fixture-scale angle F°, which is negative if read to left of zero mark, and positive if read to right.

(c) Calculate true magnetic heading of airplane thus:

$$\boxed{MHn = T° - F°}$$

(5) Read and record heading (CH) shown by each compass system main instrument (i.e., two instruments receiving information direct from DG No. 1 and DG No. 2) and STBY compass.

<u>NOTE</u>: Allow time for dials to settle before taking readings.

(6) Repeat steps (4) and (5) for a westerly heading.

(7) Repeat steps (4) and (5) for a southerly heading.

(8) Correct for N-S single cycle error (coefficient C) in system No. 1 as follows:

(a) Calculate deviation north (Dn).

$$Dn = MHn - CHn$$

(b) Calculate deviation south (Ds).

$$Ds = MHs - CHs$$

(c) Calculate single cycle error (C).

$$C = \frac{Dn - Ds}{2}$$

<u>NOTE</u>: It is important that plus and minus signs are taken into account when making calculations.

(d) AIRPLANES WITH REMOTE COMPENSATORS;
adjust N-S control on No. 1 remote magnetic compensator until main instrument in No. 1 system reads CHs - C.

(9) Repeat step (8) for No. 2 system and its remote magnetic compensator.

(10) Repeat steps (4) and (5) for an easterly heading.

(11) Correct for E-W single cycle error (coefficient B) in system No. 1 as follows:

(a) Calculate deviation west (Dw).

$$Dw = MHw - CHw$$

(b) Calculate deviation east (De),

$$De = MHe - CHe$$

(c) Calculate single cycle error (B).

$$B = \frac{De - Dw}{2}$$

NOTE: It is important that plus and minus signs are taken into account when making calculations.

(d) For airplanes with remote magnetic compensator, adjust E-W control on No. 1 remote magnetic compensator to cause main instrument in No. 1 system to read CHe - B.

(12) Repeat step (11) for No. 2 system and its remote magnetic compensator.

(13) Correct for index error (coefficient A) in system No. 1 as follows:

(a) Calculate index error (A)

$$A = \frac{Dn + Ds + Dw + De}{4}$$

(b) Remove index error by rotating index control on remote magnetic compensator clockwise if error is positive, or counterclockwise if error is negative. The amount of rotation as read on RMDI shall equal calculated index error.

(c) Record remote compensator voltage settings.

NOTE: Do not select lower than 2.5 vdc range on voltmeter except when using DVM/VTM as meter loading may cause reading errors.

1) With a multimeter set to DC, measure and record the N-S voltage at TP1 (+vc) and TP5 (-ve) on the compensator.
2) With a multimeter set to DC, measure and record the E-W voltage at TP2 (+ve) and TP5 (-ve) on compensator.
3) Affix label with the recorded voltages on the face of the compensator.

(14) Tow airplane to northerly heading again and commence 12-point residual error swing.
(a) Index error cannot be removed from standby magnetic compass.
(15) Repeat step (13) for No. 2 system.
(16) Record magnetic heading at each 30-degree compass heading and actual reading of master compass card for each system. Calculate residual deviation, residual deviation equals magnetic heading minus RMI reading.

NOTE: Check transit calibration against heading markers before taking heading reading at each point.

(17) Make sure that residual deviation does not exceed 2 degrees on each 30-degree heading.

E. Restore Airplane to Normal
(1) Remove all items of test equipment.
(2) If there is no further need for external power, remove external power.

3. Swing Standby Compass System

NOTE: The standby compass may be calibrated using one of two different procedures. One of the procedures is contained in the following paragraphs. This procedure tows the aircraft to swing the compass. The other procedure is contained in AMM 34-24-11/201, Magnetic Standby Compass (Compass Swing). It uses a compass calibrator to swing the compass.

A. General
(1) Compensation for single-cycle errors only, is provided on the standby compass.

B. Equipment and Materials
(1) All items listed in par. 2.B.

C. Prepare for Compass Swing
(1) Repeat par. 2.C.(1), (2), (3), and (5) thru (7).

CAUTION: BE SURE THAT THE GROUND DOWN LOCKS ARE IN PLACE BEFORE PROCEEDING WITH STEP (2), OR THE LANDING GEAR WILL BE IN DANGER OF COLLAPSE.

(2) Energize the landing gear lever latch solenoid after observing above caution by grounding terminal A of latch solenoid. Allow at least 20 minutes before commencing swing.

CAUTION: DO NOT MOVE LANDING GEAR LEVER AFTER COMPLETING STEP (2).

NOTE: It is necessary to energize the landing gear latch solenoid since it is energized during flight, and its magnetic field can affect the accuracy of the standby compass system. When the solenoid is energized it frees the landing gear lever from its latching mechanism. It is then possible for the lever to be moved to UP.

D. Swing Compass

(1) Adjust the compensators on the compass to align the dots.
(2) Allow the system to settle and note compass heading.
(3) Calculate airplane heading as in paragraph 2.D.(4).
(4) Record true magnetic heading and standby compass heading.
(5) Tow the airplane to within 2 degrees of a westerly heading, and repeat steps (2) thru (4) above.
(6) Tow the airplane to within 2 degrees of a southerly heading, and repeat steps (2) thru (4) above.
(7) Calculate the N-S single-cycle error as in par. 2.D.(8), and adjust compass N-S control accordingly.
(8) Tow the airplane to within 2 degrees of an easterly heading, and repeat steps (2) thru (4) above.
(9) Calculate the E-W single-cycle error as in par. 2.D.(11), and adjust compass E-W control accordingly.
(10) Tow airplane to northerly heading once more and commence 8-point residual error swing.
(11) Record magnetic heading at each 45 degrees compass point (TO FLY heading), and the actual reading of the standby compass (TO STEER heading).
(12) Make sure that the difference between TO FLY headings and TO STEER headings does not exceed ± 8 degrees (FAA requirement) or ± 4 degrees (CAA requirement) for any of the 8 points calibrated. Check also that the difference between the maximum positive and the maximum negative error does not exceed 10 degrees (FAA requirement) or 7 degrees (CAA requirement).
(13) Complete a compass correction card for the system.

(14) See CAUTION below and restore airplane to normal as in par. 2.E.

CAUTION: BEFORE COMPLETING STEP (14) MAKE SURE LANDING GEAR LEVER IS IN DOWN POSITION AND GROUNDING JUMPER IS REMOVED FROM TERMINAL A OF LANDING GEAR LEVER LATCH SOLENOID.

COMPASS SYSTEM - ADJUSTMENT/TEST

1. Compass System Tests
   A. General
      (1) The following tests should be carried out prior to compass compensation procedure which is detailed in Compass System - Maintenance Practices.
      (2) The tests are designed to check out the serviceability of the system and not its accuracy.
   B. Prepare System for Test
      (1) Provide electrical power.
      (2) Make sure that the following circuit breakers on P18 are closed:
         (a) INST TRANS
         (b) COMPASS-1
         (c) INST XFMR-1
         (d) CAPT RMDI ALT
      (3) Make sure that the following circuit breakers on P6 are closed:
         (a) COMPASS-2
         (b) INST XFMR-2
         (c) F/O RMDI ALT
      (4) Place compass transfer switch in NORMAL position.
      (5) Allow 10 minutes for systems to stabilize. Synchronize system No. 1 by pushing in and rotating captain's ADF RMI SYNC knob in direction indicated by annunciator. Synchronize system No. 2 using first officer's ADF RMI SYNC knob.
   C. Test Compass Systems
      (1) Fast Synchronization Tests
         (a) Rotate captain's ADF RMI SYNC knob as fast as possible. Make sure that warning flags appear on captain's RMI, CDI, and F/O's VOR RMI.
         (b) Rotate first officer's ADF RMI SYNC knob as fast as possible. Make sure that warning flags appear on F/O's RMI, CDI, and captain's VOR RMI.
         (c) Resynchronize both systems.
         (d) Rotate captain's ADF RMI SYNC knob in direction of dot. Make sure that compass card rotates clockwise and annunciator rotates toward plus sign. Rotate knob through synchronization in direction of plus sign. Make sure that compass card rotates counterclockwise and annunciator rotates toward dot.
         (e) Repeat step (d) using F/O's ADF RMI SYNC knob.
      (2) Slaving Tests
         (a) Use captain's ADF RMI SYNC knob to set heading 3 to 4 degrees away from synchronized reading. Make sure that compass card returns to synchronized reading and annunciator to null in less than 5 minutes.
         (b) Repeat step (a) using first officer's ADF RMI.

(3) Repeater Operation
  (a) Using captain's ADF RMI SYNC knob, rotate compass card through 360 degrees in 15-degree increments. Make sure that captain's CDI and F/O's VOR RMI compass cards follow smoothly and repeat each setting within ± 1 degree.
  (b) Using F/O's ADF RMI SYNC knob, rotate compass card through 360 degrees in 15-degree increments. Make sure that F/O's CDI and captain's VOR RMI compass cards follow smoothly and repeat each setting within ± 1 degree.
  (c) Place compass transfer switch in BOTH ON 1 position.
  (d) Using captain's ADF RMI SYNC knob, rotate compass card through 360 degrees in 15-degree increments. Make sure that captain's CDI, VOR RMI and first officer's CDI, VOR RMI, and ADF RMI compass cards follow smoothly and repeat each setting within ± 1 degree.
  (e) Using first officer's ADF RMI SYNC knob, rotate compass card through 360 degrees in 15-degree increments. Make sure that first officer's CDI, VOR RMI and captain's CDI, ADF RMI, and VOR RMI compass cards follow smoothly and repeat each setting within ± 1 degree.
  (f) Place compass transfer switch in BOTH ON 2 position.
(4) Compass Warning Tests
  (a) Check system malfunction warning test using following table. Open circuit breakers listed one at a time and close at completion of each section. Make sure that flags appearing upon opening of a circuit breaker disappear when that circuit breaker is closed.

STEP	COMPASS TRANSFER SWITCH POSITION	OPEN CIRCUIT BREAKER	FLAG APPEARS ON
1	NORMAL	COMPASS-1	CAPT ADF RMI, CDI; F/O VOR RMI
2	NORMAL	COMPASS-2	CAPT VOR RMI; F/O CDI, ADF RMI
3	NORMAL	INSTR XFMR-1	CAPT CDI; F/O VOR RMI
4	NORMAL	INSTR XFMR-2	F/O CDI; CAPT VOR RMI
5	BOTH ON NO. 1	COMPASS-1	CAPT VOR RMI, ADF RMI, CDI; F/O VOR RMI, ADF RMI, CDI
6	BOTH ON NO. 1	F/O RMI (ALT)	CAPT VOR RMI; F/O CDI, ADF RMI
7	BOTH ON NO. 1	INSTR XFMR-1	CAPT VOR RMI, CDI; F/O CDI, VOR RMI, ADF RMI
8	BOTH ON NO. 1	INSTR XFMR-2	F/O CDI; CAPT VOR RMI
9	BOTH ON NO. 2	COMPASS-2	F/O RMI, CDI; CAPT RMI & CDI
10	BOTH ON NO. 2	F/O RMI (ALT)	F/O & CAPT VOR RMI, ADF RMI, CDI
11	BOTH ON NO. 2	INSTR XFMR-1	CAPT CDI; F/O VOR RMI
12	BOTH ON NO. 2	INSTR XFMR-2	F/O VOR RMI, CDI; CAPT CDI, VOR RMI, ADF RMI
13	BOTH ON NO. 1	INVERTER CONTROL	CAPT VOR RMI; F/O CDI, ADF RMI

(b) Place compass transfer switch in normal position.

(5) If no longer required, remove electrical power from airplane.

REMOTE COMPASS TRANSMITTER - REMOVAL/INSTALLATION

1. General
   A. The remote compass transmitter (flux valve) is extremely sensitive to magnetic fields. It should never be exposed to strong fields in its vicinity. Perform demagnetication of area near flux valves shown in Fig. 401 when necessary (Ref 34-24-0, Maintenance Practices).
   B. Care should be exercised in handling and transporting the flux valve since the sensing element is mounted to allow 30 degrees of movement in the pitch and roll axes.
   C. The flux valve is preindexed by the vendor to the plate upon which it is mounted, thus permitting flux valve replacement without system recalibration. The vendor adjusted screws should not be removed or loosened for flux valve replacement purposes.
2. Equipment and Materials
   A. Maintenance stand of sufficient height to reach the flux valve access panels in the tail fin.
3. Remove Flux Valve

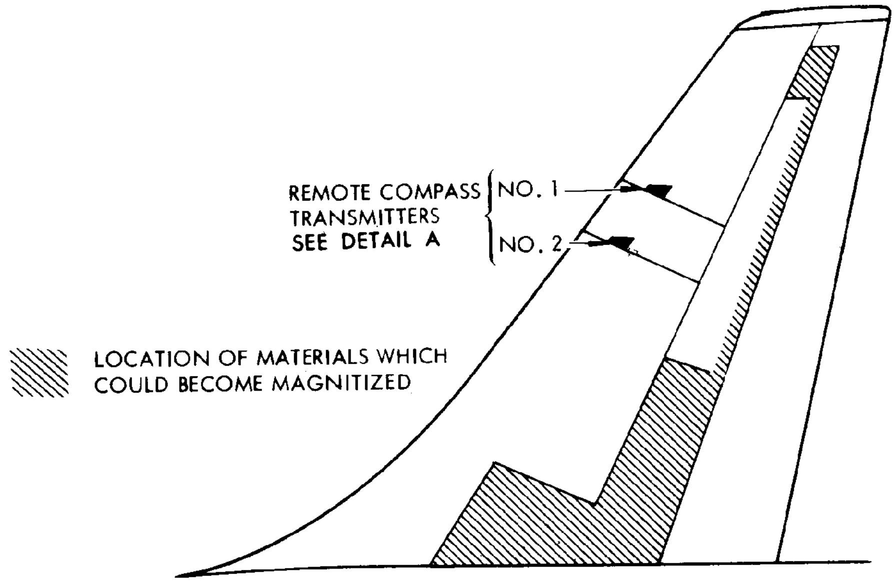
   CAUTION: USE NONMAGNETIC TOOLS AND FASTENERS WHEN WORKING WITHIN 3 FEET OF FLUX VALVE OR COMPASS SYSTEM CALIBRATION MAY BE DISTURBED. DO NOT MEASURE FLUX VALVE CONTINUITY WITH AN OHMMETER. CURRENT FLOW THROUGH SENSING COILS MIGHT DAMAGE UNIT.

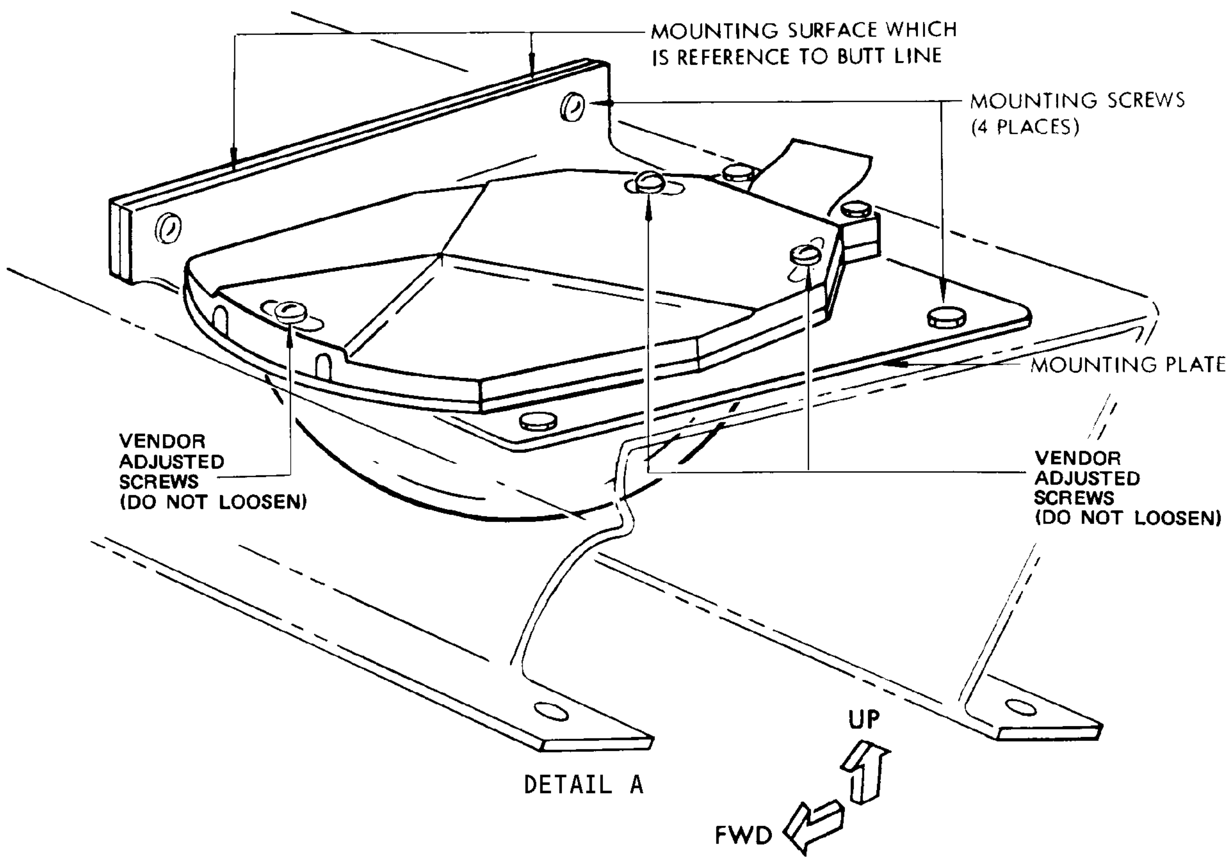
   A. Remove electrical power from the compass system by opening the applicable circuit breakers on the P6 and P18 circuit breaker panels.
   B. Remove access panel to gain access to flux valve and mounting. Bag screws and attach to access panel (Ref Chapter 12, Access Doors and Panels).
   C. Remove flux valve connector.
   D. Remove four mounting screws holding mounting plate to bracket in airplane (Fig. 401).

      CAUTION: DO NOT DISTURB THE VENDOR ADJUSTED SCREWS OR REINDEXING (A BENCH ADJUSTMENT) WILL BE REQUIRED.

   E. Bag screws separately from panel screws and attach to access panel.
   F. Remove flux valve (with mounting plate).
4. Install Flux Valve

   CAUTION: USE NONMAGNETIC TOOLS AND FASTENERS WHEN WORKING WITHIN 3 FEET OF FLUX VALVE OR COMPASS SYSTEM CALIBRATION MAY BE DISTURBED. DO NOT MEASURE FLUX VALVE CONTINUITY WITH AN OHMMETER. CURRENT FLOW THROUGH SENSING COILS MIGHT DAMAGE UNIT.



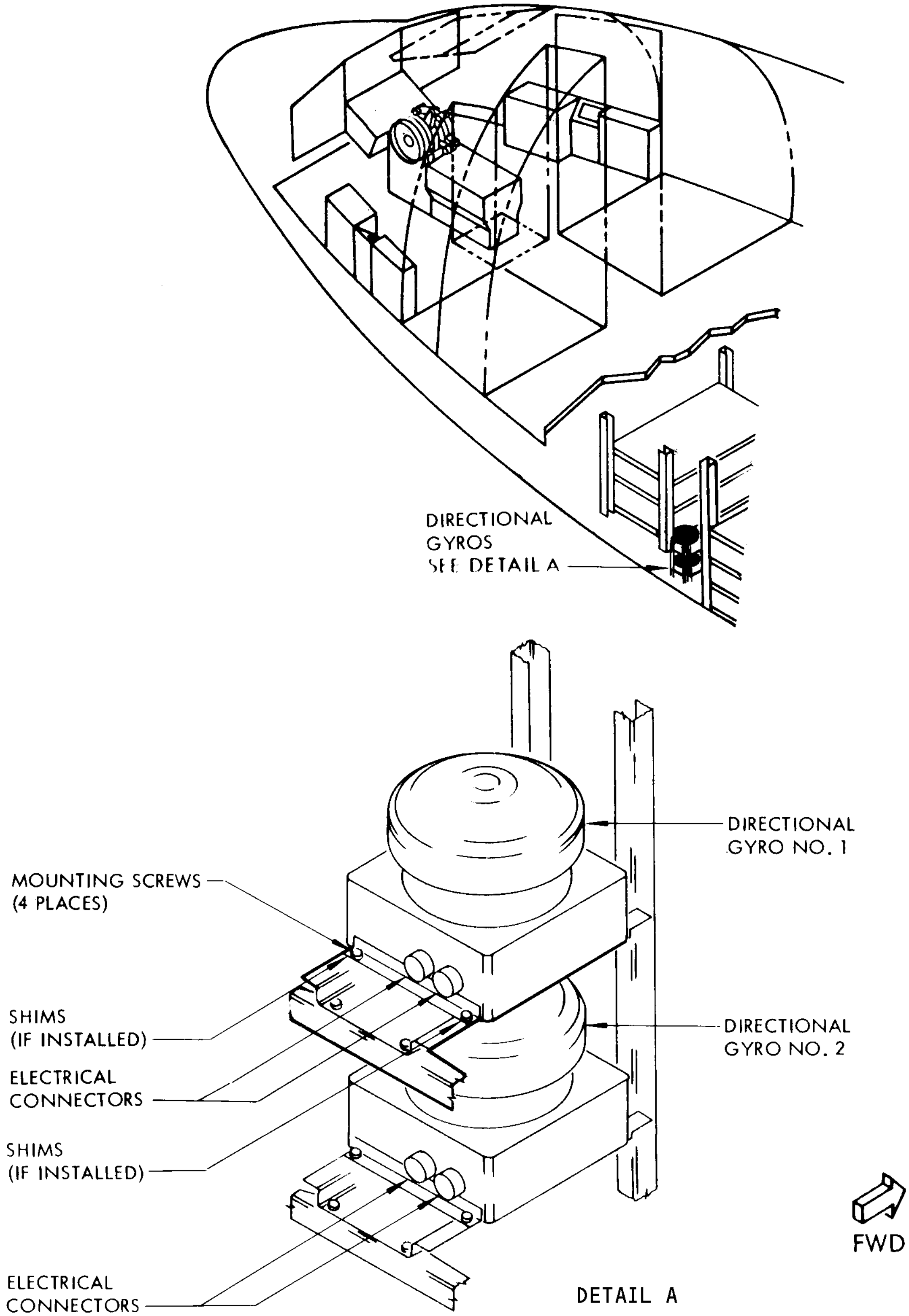


Remote Compass Transmitter Installation
Figure 401

A. Remove electrical power from the compass system by opening the applicable circuit breakers on the P6 and P18 circuit breaker panels.
B. Place flux valve, complete with mounting plate, on airplane bracket as shown in Fig. 401.
C. Line up four mounting holes and insert mounting screws.
D. Hand tighten all four screws.
E. Tighten down the two screws which secure the mounting plate to the butt line referenced surface on the airplane bracket.
F. Tighten down the remaining two screws.
G. Connect electrical connector and tighten.
H. Replace access panels (all screws and hardware within 36 inches of flux valve must be of nonmagnetic material).
I. Apply electrical power to the system by closing the applicable circuit breakers on the P6 and P18 circuit breaker panels.
J. Perform operational check of compass system (Ref 34-21-0, Adjustment/Test).

DIRECTIONAL GYRO - REMOVAL/INSTALLATION

1. General
   A. A gyro should not be removed from its seating position until its rotor has come to rest. At least 3 minutes should be allowed after switch off, in order to meet this requirement.
   B. Particular care should be exercised in transporting a gyro from place to place. It is a delicate instrument, which will not withstand knocks or rough handling. Use approved containers for transportation whenever possible.
   C. Where necessary, laminated shim stock is used to seat each gyro within required tolerances relative to the airplane's axes. Extreme care should be exercised when removing the gyros to ensure that-the shims are not dislodged; and, if they are, to return them to their original positions when replacing the gyro. (If installed, the shims are bonded to the mounting surface in the airplane.)
2. Remove Directional Gyro
   A. Remove power from compass systems. Open all related circuit breakers.
   B. Disconnect electrical connectors from gyro to be removed (Fig. 4ol).
   C. Remove four screws attaching gyro to mounting plate.
   D. Bag screws and attach to gyro or mounting.
   E. Remove gyro.
3. Install Directional Gyro
   A. Place directional gyro on mounting plate as shown in Fig. 401.
   B. Align gyro to holes in mounting surfaces and install four mounting screws. Tighten screws.
   C. Connect electrical connectors and tighten.
   D. Connect external electrical power or use APU power.
   E. Check that all compass system and instrument transformer circuit breakers are closed.
   F. Place compass transfer switch to NORMAL (if installed).
   G. Allow 10 minutes for the system to stabilize and check that the RMI and CDI (HSI) compass warning flags are out of view.
   H. Rotate captain's RMI SYNC knob in direction of dot; compass card should rotate clockwise and annunciator should rotate toward plus sign. Rotate knob through synchronization in direction of plus sign; compass card should rotate counterclockwise and annunciator should rotate toward dot.
   I. Repeat step H using first officer's RMI SYNC knob.
   J. Disconnect electrical power.



Directional Gyro Installation
Figure 401

REMOTE COMPASS COMPENSATOR - REMOVAL/INSTALLATION

1. General
   A. The remote compensator is installed behind E2 rack, E and E compartment. Each unit is held in place by four quick-release fasteners and has one electrical cable attached to the rear panel through a connector.
   B. Replacement of a remote compensator will not seriously degrade system performance if the compensating voltage settings are properly transferred to the replacement unit.
2. Equipment and Materials
   A. Voltmeter - Simpson model 260
3. Remove Remote Compensator
   A. Open Compass 1 (Panel P18) or Compass 2 (Panel P6) c/b, depending on system being disturbed.
   B. Gain access to the Remote Compensator through the Forward Cargo compartment and remove the compensator cover. Check the voltage levels recorded on the labels on the face of the compensator.
   C. Carefully record these values on new labels and affix to the replacement unit.

      NOTE: Loss of these voltage values will require that a complete new airplane compass swing be performed.

   D. Note position of INDEX pointer on compensator to be removed.
   E. Loosen quick-release fasteners and remove unit.
   F. Disconnect electrical connector at rear of unit.
4. Install Remote Compensator
   A. Check that voltage levels are recorded on labels on compensator to be installed. Use data from N-S and E-W labels on previously installed unit if necessary, and affix new labels.

      NOTE: Failure to correctly transfer these voltages would make a complete compass swing procedure necessary.

   B. Attach electrical connector on rear panel of unit and install with quick-release fasteners.
   C. Close Compass 1 (Panel P18) or Compass 2 (Panel P6) circuit breaker.
   D. Using the voltmeter, measure dc voltage between terminals TP1 (+ve and TP5 (-ve) on the compensator.

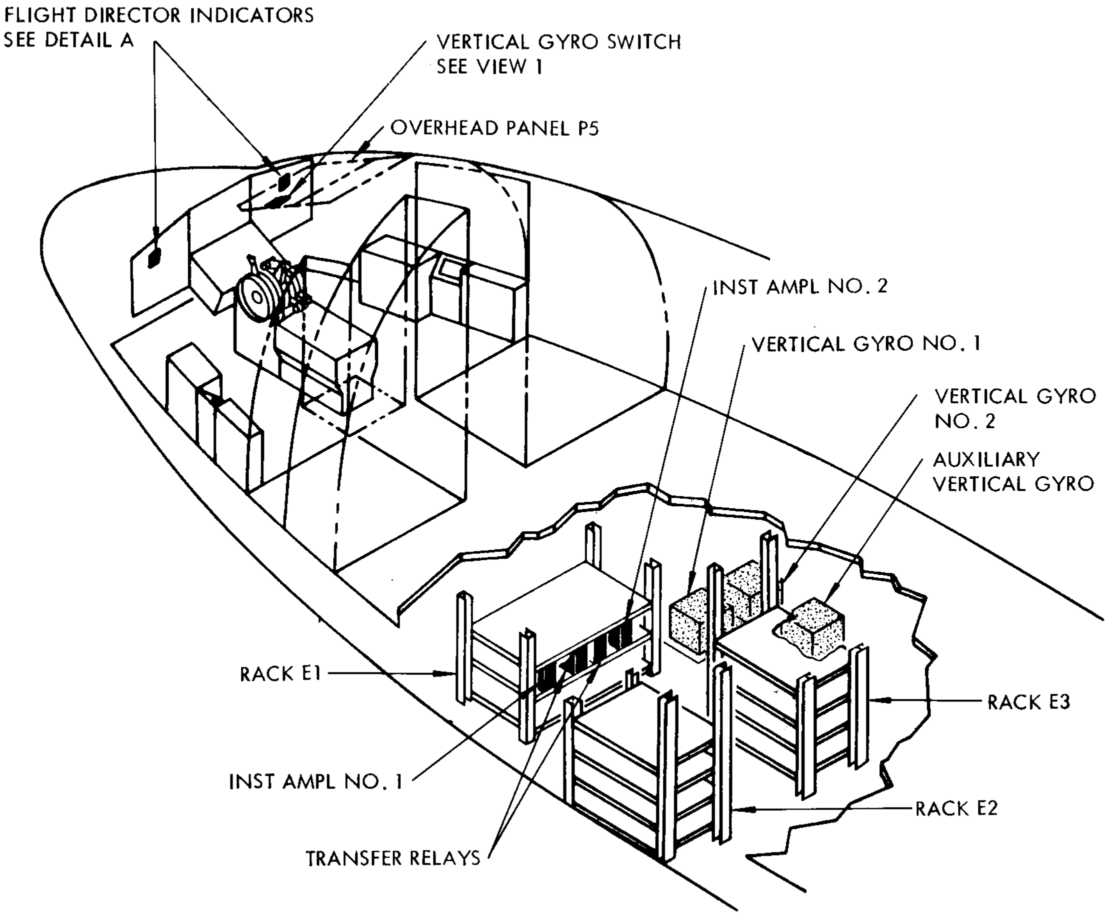
      NOTE: Do not select lower than 2.5vdc range on voltmeter except when using DVM/VTM as meter loading may cause reading errors.

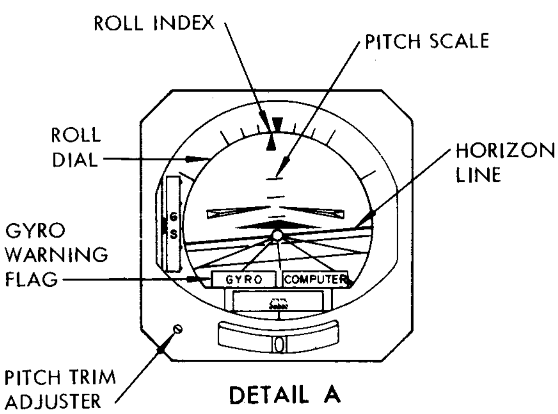
   E. Adjust N-S control until voltage level equals the value on the label (within 0.1 volt dc).

F. Repeat operations (d) and (e), using terminals TP2 (+ve) and TP5 (-ve), and adjusting the E-W control.
G. Set INDEX pointer to the position noted on the compensator removed previously.
H. Refit compensator cover and replace cargo compartment access panel.
I. Close Compass 1 (Panel P18) or Compass 2 (Panel P6) cb and perform Compass System - Adjustment/Test (Ref 34-21-0) on applicable system.

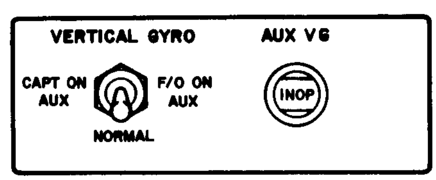
ATTITUDE REFERENCE SYSTEMS - DESCRIPTION AND OPERATION

1. General
   A. The attitude reference system is designed primarily to furnish the captain and first officer with information concerning the airplane's attitude in both PITCH and ROLL axes at all times during flight. It also provides pitch and roll displacement signals to the autopilot, flight director system, comparator and weather radar antenna stabilization system. Two systems are installed; system No. 1 feeding information to the captain's FDI, steering computer No. 1, and autopilot system; while system No. 2 feeds information to the F/O FDI, steering computer No. 2, and weather radar.
   B. Each system consists of the following components; vertical gyro, roll servo-amplifier, pitch servo-amplifier, warning flag amplifier and flight director indicator. The three amplifiers, plus a small power supply for the servomotors, are contained in the flight instrument amplifier unit, and the location of all components is shown in figure 1.
   C. DC power which is required to withdraw the warning flag from view in each FDI is generated in the warning flag amplifier for that system. The warning flag will appear when power is lost to the flight instrument amplifier, or if the flag amplifier loses its dc input from the roll amplifier, or in the event of a malfunction in either pitch or roll servo-amplifier circuits.
   D. A third auxiliary vertical gyro is provided as a backup gyro for either system.
   E. A vertical gyro transfer switch located on the P-5 overhead panel is used to energize either of the two gyro transfer relays and thus bring the auxiliary gyro into use in place of one of the others. There are three switch conditions: F/O ON AUX - NORMAL - CAPT ON AUX. (See figure 1.)
2. Vertical Gyro
   A. The gyro has 360 degrees of freedom about the roll axis, and + 85 degrees of freedom about the pitch axis. Initial erection of the gyro is accomplished by internal circuits, and no manual caging device is used. The erection system maintains verticality within 1/4 degree during unaccelerated flight and within 1 degree during maneuvers; or in turbulent air.
3. Flight Instrument Amplifier Rack
   A. The roll and pitch servo-amplifiers and the warning flag amplifiers are all plug-in type units which plug into the appropriate amplifier rack.





DETAIL A

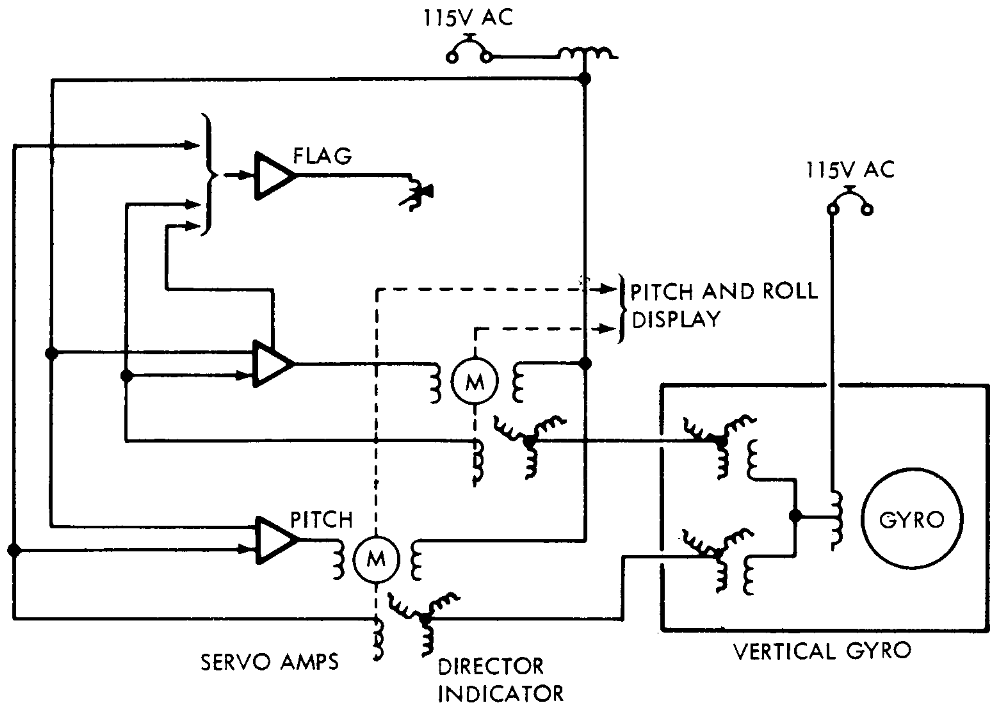


INSTRUMENT TRANSFER PANEL

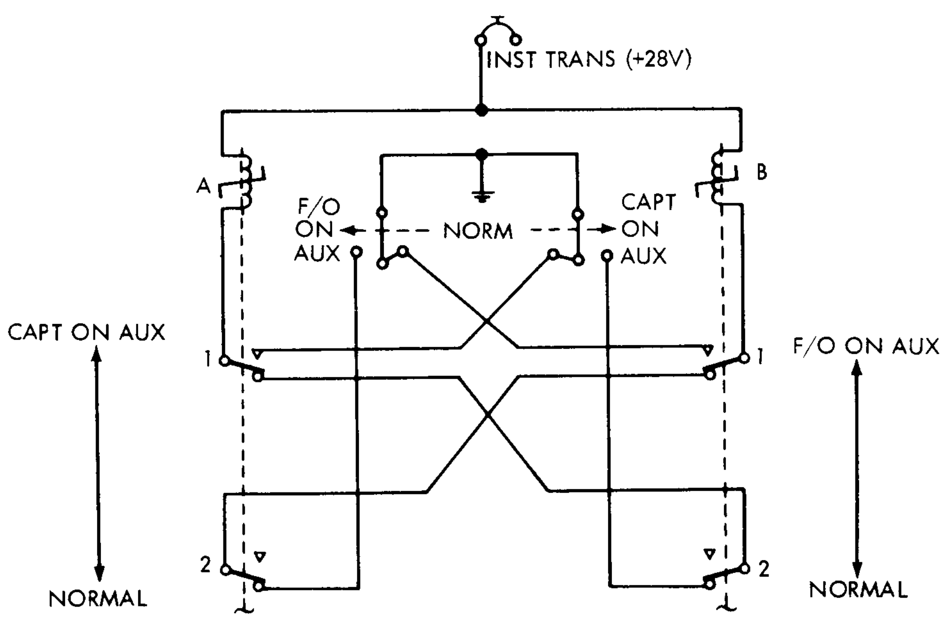
VIEW 1

Attitude Reference System Component Location
Figure 1

4. Flight Director Indicator (FDI)
    A. Each artificial horizon (contained in the FDI) consists of a moving tape assembly which displays the horizon line, pitch scale and roll index. The tape assembly is coupled to the rotors of the roll and pitch synchros and to the roll and pitch servomotors. When the servomotors turn, the synchros and the tape assembly are positioned simultaneously. Airplane attitude in roll is read from the roll index against the roll scale, and airplane attitude in pitch is read from the pitch scale against the miniature airplane symbol. Adjustment of the tape in pitch is provided for, through the pitch trim adjuster at the lower left corner of the indicator.
    B. A gyro warning flag is provided in each FDI to indicate loss of power, or malfunction of the gyro or of either servo-amplifier circuit in the system.
5. Operation
    A. Both systems are made operational by closing the system circuit breakers. After a brief period of time the gyros will erect and each FDI should indicate a level condition in both pitch and roll axes.
    B. Each system operates as follows: when the airplane is displaced in the roll plane, the gyro will transmit an ac signal which is not only related in amplitude to the degree of displacement, but also phase oriented to differentiate between left and right roll. The signal is passed to the roll servo-amplifier where it is amplified to drive the servomotor, which in turn drives the horizon tape. This ac signal is fed via the roll synchro on the roll motor shaft to ensure that the motor stops as soon as it has turned sufficiently to null out the error signal. Pitch displacement signals are handled in a similar way, and the electromechanical design is such that the horizon tape is able to respond to both pitch and roll signals simultaneously. (See figure 2.)
    C. The operation of the transfer switch and relays is illustrated in figures 2 and 3. Figure 2 shows how the relay coils are connected with the dc supply through the relay contacts. The switch is shown in its NORMAL setting. Assume that the transfer switch is now set to CAPT ON AUX. This switch puts a ground on the lower end of latching relay coil A via contacts B2 and Al; and since the other end of that coil is connected to 28 volts dc, it will now latch over to the alternate set of contacts and remain there until energized once more. Note that the ground is removed from the coil by the transfer of contacts at Al. In order to ground the coil once more and switch the relay back again, it will be necessary to
    D. return the switch to NORMAL. The switching to F/O ON AUX is carried out in a similar way.



Basic Attitude Reference System Schematic
Figure 2

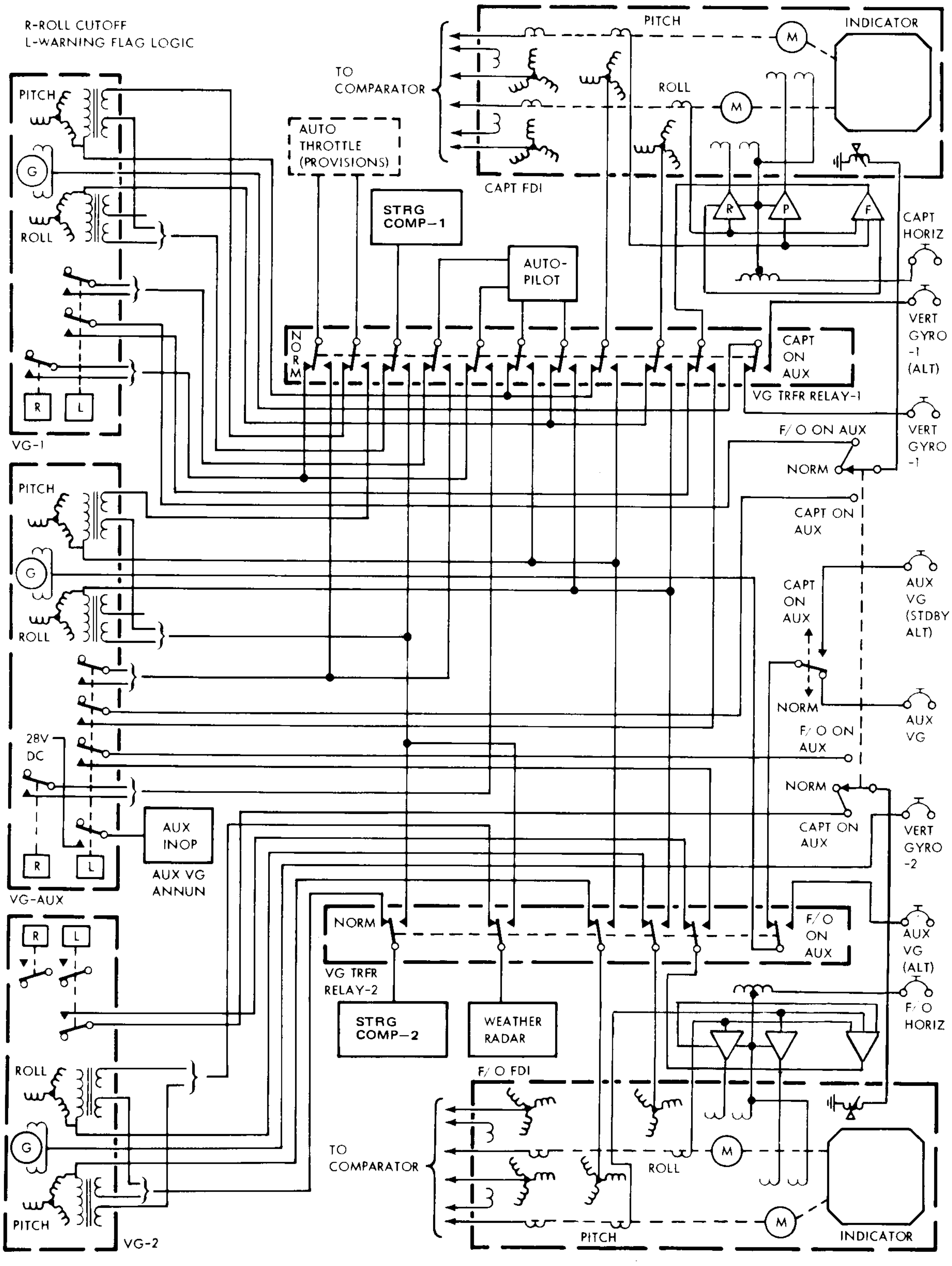


Transfer Switching Circuit
Figure 2

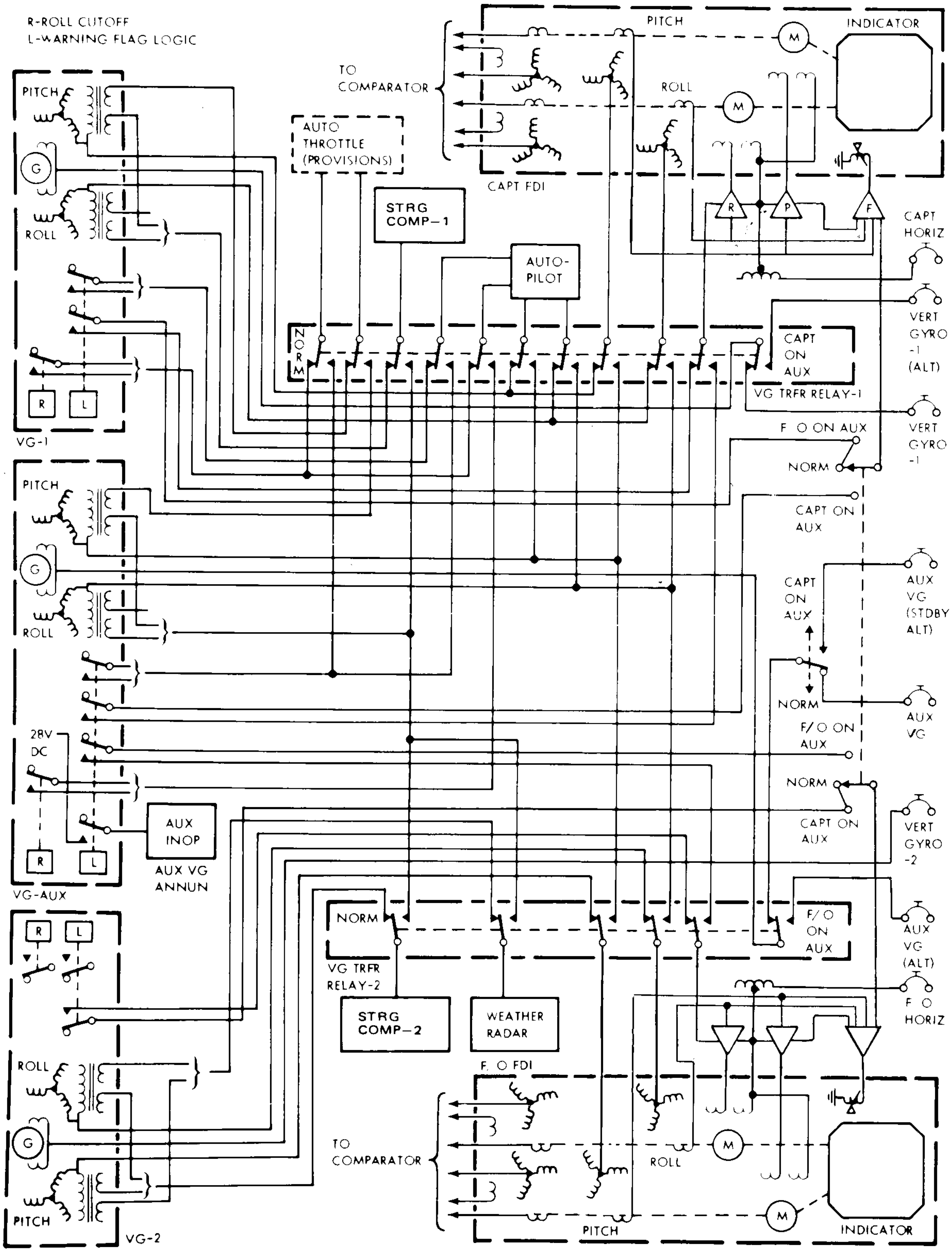
E. The switching contacts shown in figure 3 are a continuation of those in figure 2, but the y take care of the transfer of signals within the system. In the NORMAL configuration, the captain's FDI receives its pitch and roll signals direct from gyro No. 1, but in the CAPT ON AUX position, it will be seen that these signals are taken from the auxiliary vertical gyro. The dc signal from the flag amplifier is fed to the flag coil via the contacts of the vertical gyro switch, the flag warning logic relay, and the transfer relay. In NORMAL it is derived from its associated flag amplifier, through the contacts of the vertical gyro relay switch and the flag warning relay in vertical gyro No. 1. In the CAPT ON AUX position, the dc source is still the same flag amplifier but the transfer relays now route it through the contacts of the flag warning relay in the auxiliary gyro. The steering computer also receives its pitch and roll signals via the contacts of the transfer relay. The transfer of signals from NORMAL to F/O ON AUX follows the same pattern of operation as that for the captain's system. The weather radar input signals may be switched from vertical gyro No. 2 to auxiliary gyro on the F/O's side only. The autopilot derives signals from vertical gyro No. 1 or the auxiliary gyro. The ac power source for the auxiliary gyro is the AUX V/G circuit breaker when the systems are in NORMAL. But when the transfer switch is set to F/O ON AUX, ac power is taken from the AUX V/G (ALT) circuit breaker. When the transfer switch is set to CAPT ON AUX, ac power is derived from the AUX V/G (STDBY ALT) circuit breaker.

F. The annunciator which is fitted to the forward overhead panel will operate whenever the auxiliary gyro suffers a malfunction or loses its power supply. The dc power for its shutter is obtained from a rectifier circuit inside the gyro and it is fed to the shutter coil (to withdraw it from view) via the contacts of the flag warning logic relay in the gyro. In the event of gyro failure the annunciator power is cut off and the shutter falls back into place.

G. Normal operation of the servo loop will ensure that the rotor that feeds the servo-amplifier is always at or near a NULL. This signal is also fed to a null sensing circuit in the flag amplifier where it is used as a reference from the servo loop circuit. In the absence of the null sensing signal, the flag amplifier will fail and the warning flag will appear. The flag amplifier receives these NULL signals from both pitch and roll servo circuits so that a malfunction in either servo loop will trigger the flag. Roll and pitch information is also supplied to comparator circuits in the navigation warning system (refer to 34-29-0) from the three windings of an additional transolver in the captain's and F/O's FDI.



Attitude Reference System Schematic
Figure 3 (Sheet 1)



Attitude Reference System Schematic
Figure 3 (Sheet 2)

ATTITUDE REFERENCE SYSTEM - TROUBLESHOOTING

1. General
   A. A prerequisite for accurately trouble shooting any system is a good knowledge of normal system operation. Refer to vertical gyro system description and operation for review when required.
   B. To maintain integrity of the vertical gyro system and other associated systems that can be affected when a component is changed, a complete functional test as specified in adjustment/test procedure is required.
   C. When a malfunction occurs in the vertical gyro system and cannot be quickly isolated by substitution of a known serviceable component(s), then a thorough check should be made to determine if correct power is present, at the connector to the component. Refer to electronic wiring diagrams for pin number(s).

      CAUTION: OBSERVE CORRECT POLARITY TO AVOID DAMAGE TO TEST EQUIPMENT.

   D. If trouble persists after voltage check as in C. above, then check all related connectors for security of connection.
2. Troubleshooting Chart
   A. The troubleshooting chart should be used as a guide to help analyze possible source and/or determine the order in which to make tests or substitutions.

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Attitude sphere malfunctioning and gyro flag in view	Vertical gyro inoperative or instrument amplifier failure	Check that appropriate circuit breakers are closed. Check dc voltage output from instrument amplifiers (refer to Wiring Schematic for terminals). Check valid dc voltage output through vertical gyro interlock	Replace defective vertical gyro and/or instrument amplifier
Gyro flag in view but attitude sphere normal	Malfunction of VG switch and/or vertical gyro transfer relay	Check Attitude Director Indicator (sometimes referred to as FDI or HDI). Check for continuity between vertical gyro relay and vertical gyro transfer switch	Replace defective component
Gyro flag does not appear when appropriate circuit breakers are opened	Sticking flag	Check attitude indicator(s)	Replace defective indicator
Sluggish attitude sphere movement	Attitude indicator or vertical gyro not functioning properly	Perform adjustment test	Replace or repair faulty components
Attitude sphere malfunctioning but gyro flag not in view	Attitude indicator	Check vertical gyro(s). Check appropriate vertical gyro relay contacts	Replace defective component

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Attitude sphere out of tolerance with flag removed from view	Vertical gyro(s) not erecting properly	Perform adjustment test	Replace defective component
Pitch trim adjustment (where provided) not effective but attitude sphere appears normal	Attitude indicator	Check attitude indicator(s)	Replace defective indicator

ATTITUDE REFERENCE SYSTEMS - ADJUSTMENT/TEST

1. General
   A. The following procedures describe two methods for testing the attitude reference systems in the airplane. The first procedure is an operational test in which only onboard equipment is used. The second procedure is a system test using the tilt tables to check system tolerances.
   B. The system test should be made prior to or concurrent with the adjustment/test of autopilot, weather radar, flight director, and navigation warning systems. If these systems are to be tested, performing those portions of test that require use of tilt table will save setup time.
2. Operational Test
   A. Prepare for Operational Test
      (1) Remove mounting bolts from vertical gyros.
      (2) Connect electrical power to airplane and close GND PWR switch on panel P5. Energize No. 1 and 2 electronics buses by closing ac and dc circuit breakers on panel P6 and placing master switches in ON position.
      (3) Make sure that the following circuit breakers on panel P6 are closed:
         (a) VERT GYRO-2
         (b) AUX VERT GYRO (ALT)
         (c) F/O'S HORIZON
      (4) Make sure that the following circuit breakers on panel P18 are closed:
         (a) VERT GYRO-1
         (b) VERT GYRO-1 (ALT)
         (c) AUX VERT GYRO (STBY ALT)
         (d) CAPT'S HORIZON
         (e) AUX VERT GYRO
         (f) INSTR TRANS
   B. Roll and Pitch Servo Test
      (1) Place VERTICAL GYRO transfer switch in NORMAL position. Make sure that both ADIs indicate airplane roll and pitch attitude 1 minute after power is applied.
      (2) Tilt vertical gyro No. 1 until captain's ADI indicates 10 degrees pitch down (attitude assembly moves toward top of indicator).

         NOTE: ADI should follow displacements of vertical gyro smoothly with no sluggishness or overshoot.

      (3) Tilt vertical gyro No. 1 until captain's ADI indicates 10 degrees pitch up.
      (4) Tilt vertical gyro No. 1 until captain's ADI indicates 20 degrees right roll (attitude assembly rotates counterclockwise).
      (5) Tilt vertical gyro No. 1 until captain's ADI indicates 20 degrees left roll.
      (6) Repeat steps (2) thru (5) using vertical gyro No. 2 and first officer's ADI.

(7) Place VERTICAL GYRO transfer switch in CAPT ON AUX position and repeat steps (2) thru (5) using auxiliary gyro and captain's ADI.
(8) Place VERTICAL GYRO transfer switch in F/O ON AUX position and repeat steps (2) thru (5) using auxiliary gyro and first officer's ADI.
(9) Return vertical gyros to level and install mounting bolts.

C. Attitude Warning Flag Circuit Test
(1) Check warning circuits as tabulated below.
(2) Circuit breakers shall be opened one at a time and closed at completion of each test.
(3) Make sure that flag appears when breaker is opened, disappears when breaker is closed.

TEST	VG TRANSFER SWITCH POSITION	OPEN CIRCUIT BREAKER	GYRO FLAG APPEARS ON ADI	AUX VG INOP ANNUNCIATOR
1	NORMAL	VERT GYRO-1	Captain's	Out of View
2	NORMAL	CAPT HORIZ	Captain's	Out of View
3	NORMAL	VERT GYRO-2	F/O's	Out of View
4	NORMAL	F/O HORIZ	F/O's	Out of View
5	NORMAL	AUX VERT GYRO	Nowhere	In View
6	CAPT ON AUX	AUX VERT GYRO (STBY ALT)	Captain's	In View
7	F/O ON AUX	AUX VERT GYRO (ALT)	F/O's	In View

(4) Place VERTICAL GYRO switch in NORMAL position.

D. Vertical Gyro Transfer Relay Monitor Test
(1) Check transfer relay monitor as tabulated below.

(2) Open instrument transfer circuit breaker for test and reclose at the completion of each test.

TEST	VG GYRO SWITCH POSITION	CIRCUIT BREAKER	REPOSITION VERTICAL GYRO SWITCH TO	FLAG APPEARS ON ADI
1	NORMAL	INSTR TRANS	CAPT ON AUX F/O ON AUX	Captain's F/O's
2	CAPT ON AUX	INSTR TRANS	NORMAL F/O ON AUX	Captain's Captain's & F/O's
3	F/O ON AUX	INSTR TRANS	NORMAL CAPT ON AUX	F/O's Captain's & F/O's

E. Remove electrical power from airplane.

3. System Test

A. Equipment and Materials

(1) Calibrated tilt tables (3 required if an auxiliary vertical gyro is installed)

B. Prepare for System Test

(1) Remove vertical gyros from mountings and install on calibrated tilt tables (oriented as installed in airplanes).

(2) Connect jumper cables between gyros and their respective airplane connector.

(3) Adjust tilt tables to 0 degree roll and pitch.

(4) Connect electrical power to airplane and close GND PWR switch on panel P5. Energize No. 1 and 2 electronics buses by closing ac and dc circuit breakers on panel P6 and placing master switches in ON position.

(5) Make sure that the following circuit breakers on panel P6 are closed:

(a) VERT GYRO-2

(b) AUX VERT GYRO (ALT)
(c) F/O HORIZ

(6) Make sure that the following circuit breakers on panel P18 are closed:
(a) VERT GYRO-1
(b) VERT GYRO-1 (ALT)
(c) AUX VERT GYRO (STBY ALT)
(d) CAPT HORIZ
(e) AUX VERT GYRO
(f) INSTR TRANS

C. Roll and Pitch Servo Test

(1) Place VERTICAL GYRO transfer switch in NORMAL position. Make sure that both ADIs indicate 0.0 ±2 degrees roll and pitch attitude 1 minute after power is applied.
(2) Adjust each ADI for zero pitch attitude using adjustment screw on lower left corner of ADIs.
(3) Tilt vertical gyro No. 1 until captain's ADI indicates 10 degrees pitchdown (attitude assembly moves toward top of indicator). Make sure that the tilt table indicates 10 ±2 degrees pitchdown.

NOTE: ADI should follow displacements of vertical gyro smoothly with no sluggishness or overshoot.

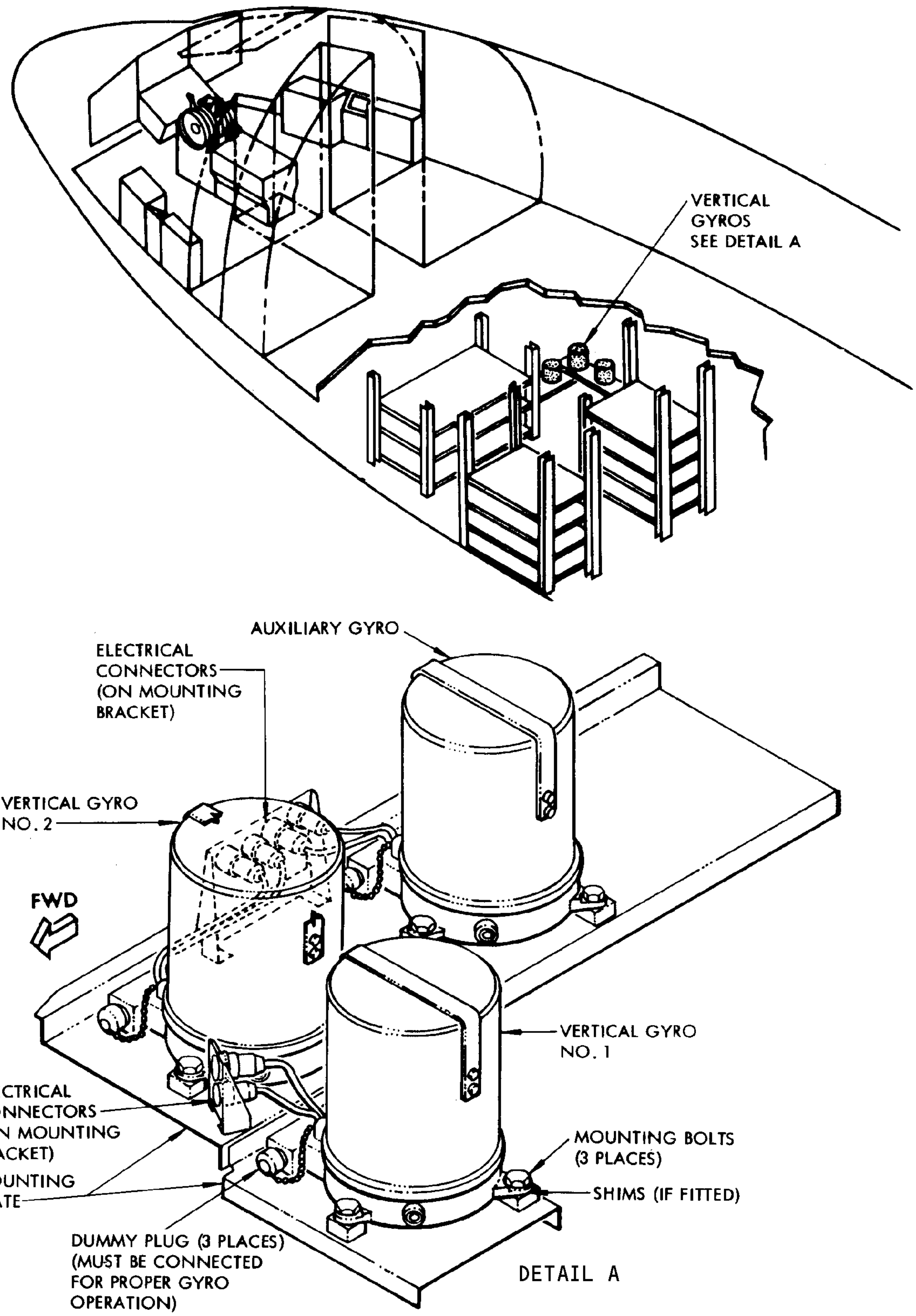
(4) Tilt vertical gyro No. 1 until captain's ADI indicates 10 degrees pitchup. Make sure that the tilt table indicates 10 ±2 degrees pitchup.
(5) Tilt vertical gyro No. 1 until captain's ADI indicates 20 degrees right roll (attitude assembly rotates counterclockwise). Make sure that the tilt table indicates 20 ±3 degrees right roll.
(6) Tilt vertical gyro No. 1 until captain's ADI indicates 20 degrees left roll. Make sure that the tilt table indicates 20 ±3 degrees left roll.
(7) Repeat steps (3) thru (6) using vertical gyro No. 2 and first officer's ADI.
(8) Place VERTICAL GYRO transfer switch in CAPT ON AUX position and repeat steps (3) thru (6) using auxiliary gyro and captain's ADI.
(9) Place VERTICAL GYRO transfer switch in F/O on AUX position and repeat steps (3) thru (6) using auxiliary gyro and F/O's ADI.
(10) Return vertical gyros to level attitude.

D. Restore Airplane to Normal

(1) Open all attitude reference system circuit breakers.
(2) Allow 3 minutes for gyros to run down. Install in airplane and connect electrical cables.
(3) Close system circuit breakers and make sure that system indicates attitude of airplane.
(4) Determine whether there is further need for electrical power, if not, remove power.

VERTICAL GYRO - REMOVAL/INSTALLATION

1. General
   A. A gyro should not be removed from its seating position until its rotor has come to rest. At least 3 minutes should be allowed after switch off, to meet this requirement.
   B. Particular care should be exercised in transporting a gyro from place to place. It is a delicate instrument which will not withstand knocks or rough handling. Use approved containers for transportation wherever possible.
   C. Where necessary, laminated shim stock is used to seat each gyro within required tolerances relative to the airplane's axes. Prior to incorporation of SB 34-1171, (on airplanes with shims bonded to the mounting brackets), extreme care should be exercised when removing the gyros to ensure that the shims are not dislodged; and, if they are, to return them to their original positions when replacing the gyro. After incorporation of SB 34-1171, the shims are riveted to the mounting brackets.
2. Remove Vertical Gyro
   A. Remove power from the attitude system.
   B. Disconnect electrical connectors and remove gyro cables from mounting bracket (Fig. 401).
   C. Remove three bolts attaching gyro to mounting plate.
   D. Bag bolts and attach to gyro or mounting.
   E. Remove gyro.
3. Install Vertical Gyro
   A. Place vertical gyro on mounting plate as shown in Fig. 401. Arrow on top of unit should be pointing forward.
   B. Align holes in mounting surfaces and install three mounting bolts. Tighten bolts.
   C. Install gyro cables on mounting bracket and connect electrical connectors.
   D. Connect external electrical power or use APU power.
   E. Check that all vertical gyro, F/O's HORIZ and CAPT's HORIZ circuit breakers are closed.
   F. Place AUX VERT GYRO switch in NORMAL position.
   G. Allow 1 minute for gyros to stabilize and check that ADI's (FDI's) indicate airplane attitude and that GYRO flags are out of view.
   H. Disconnect electrical power.



Vertical Gyro Installation
Figure 401

TURN-AND-BANK INDICATION - DESCRIPTION AND OPERATION

1. Turn-and-Bank Indicator
   A. The turn portion of the indicator is an electrically driven rate-gyro linked to the pointer. Pointer deflection is proportional to rate of turn.
   B. The bank portion of the indicator contains a ball-type inclinometer. Fluid in the inclinometer tube damps out any vibration between ball and tube, thereby ensuring a smooth movement of the ball. With the airplane level, the ball should be centered between the two indices shown on figure 1. If the ball is not centered, the clamping ring holding the indicator (rear of panel) may be loosened and the indicator rotated until the ball has centered.
   C. A power failure warning flag is also included in the indicator.
2. General
   A. On AR ALL EXCEPT LV-JMW thru LV-JMZ, LV-JTD, LV-JTO, LV-JND and LV-JNE, the bank indicator is installed as an integral part of the flight director indicator (34-26-0) on the captain's and first officer's instrument panel. On LV-JMW thru LV-JMZ, LV-JTD, LV-JTO, LV-JND and LV-JNE, a separate turn-and-bank indicator is installed on the captain's and first officer's instrument panels.
   B. The indicators are of the direct reading type. Indications are used for reference in keeping the airplane level in straight flight, and in establishing a desired rate of turn, and proper angle of bank for that turn. For a pictorial view of the indicator, see Fig. 1.
3. Operation
   A. Power for the indicators is obtained from the TURN AND SLIP CIRCUIT breakers at the P18 load control center.
   B. When the pointer is off center, it indicates that the airplane is turning in the direction shown by the pointer.
   C. When the airplane is in a turn, the ball is acted upon by two forces, the force of gravity and centrifugal force. During a coordinated turn, both forces are such that the resultant force holds the ball centered in the tube. The centered ball indicates the proper lateral attitude for any rate of turn.

TURN-AND-BANK INDICATION - DESCRIPTION AND OPERATION

1. General

A. On AR ALL EXCEPT LV-JMW thru LV-JMZ, LV-JTD, LV-JTO, LV-JND and LV-JNE, the bank indicator is installed as an integral part of the flight director indicator (34-26-0) on the captain's and first officer's instrument panel. On LV-JMW thru LV-JMZ, LV-JTD, LV-JTO, LV-JND and LV-JNE, a separate turn-and-bank indicator is installed on the captain's and first officer's instrument panels.

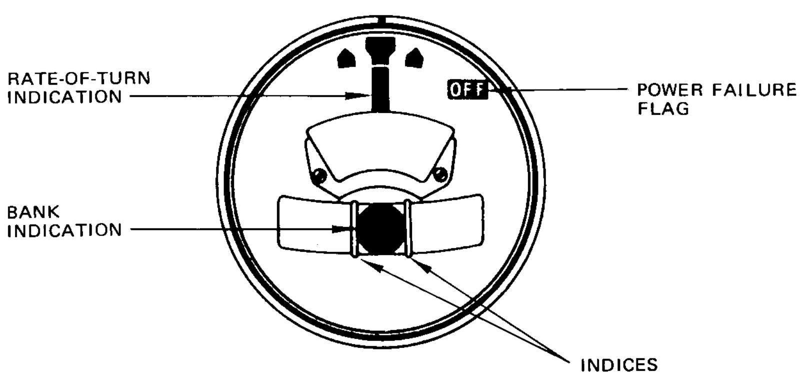
B. The indicators are of the direct reading type. Indications are used for reference in keeping the airplane level in straight flight, and in establishing a desired rate of turn, and proper angle of bank for that turn. For a pictorial view of the indicator, see Fig. 1.

2. Turn-and-Bank Indicator

A. The turn portion of the indicator is an electrically driven rate-gyro linked to the pointer. Pointer deflection is proportional to rate of turn.

B. The bank portion of the indicator contains a ball-type inclinometer. Fluid in the inclinometer tube damps out any vibration between ball and tube, thereby ensuring a smooth movement of the ball. With the airplane level, the ball should be centered between the two indices shown on figure 1. If the ball is not centered, the clamping ring holding the indicator (rear of panel) may be loosened and the indicator rotated until the ball has centered.

C. A power failure warning flag is also included in the indicator.



Turn-and-Bank Indicator
Figure 1

TURN AND BANK INDICATION - MAINTENANCE PRACTICES

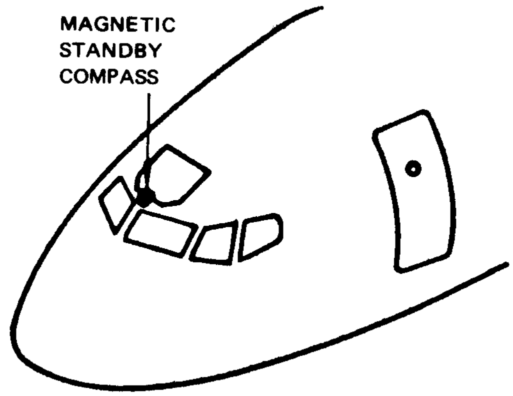
1. General
   A. During attitude director indicator installation it is extremely important that the ball in the turn and bank inclinometer is centered when the airplane is level laterally.
   B. To preclude improper (unlevel) installation, accomplish the following prior to indicator clamp screw tightening:
      (1) Level the airplane laterally using a spirit level across the captain's and copilot's inboard seat tracks.
      (2) Locate the indicator so that the ball is centered; then tighten the clamp adjustment screw.

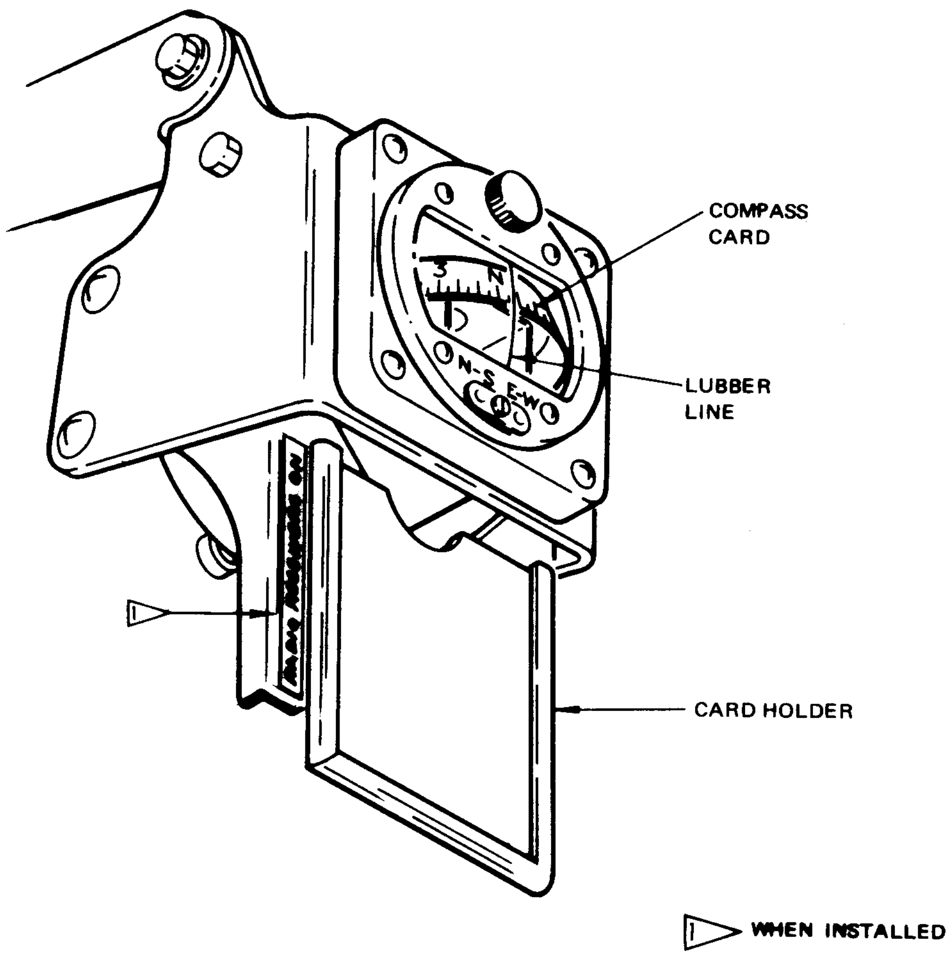
         NOTE: The spirit level bubble and the turn and bank indicator ball move in opposite directions during airplane leveling.

   C. The following alternate procedure may be used when it is known that the indicator being changed is defective and there are no flight reports on the second indicator on the opposite panel:
      (1) Locate the replacement indicator so that the ball is positioned in the same position as the ball in the indicator on the opposite panel.

MAGNETIC STANDBY COMPASS - DESCRIPTION AND OPERATION

1. General
   A. The magnetic standby compass is installed forward of the forward overhead panel. The standby compass is a self-contained unit and indicates the magnetic heading of the airplane. It is used primarily in cases of emergency when all other compass systems fail. For a pictorial view of the magnetic standby compass, see figure 1.
2. Magnetic Standby Compass
   A. The standby compass is a highly sensitive instrument of liquid-filled design. The card assembly, which indicates magnetic direction, is illuminated by a bulb located at the top of the compass.
   B. The standby compass for its operation depends upon the effect of the earth's magnetic field. The card assembly is so suspended that the magnets which are attached to the card assembly are free to align themselves with the horizontal component of the earth's magnetic field. The magnetic heading of the airplane is read from the card assembly against the lubber line.
   C. At the bottom of the standby compass under a cover plate are two compensators decaled N-S and E-W. The purpose of the compensators is to compensate for magnetic disturbances in the area of standby compass.
   D. The correction card holder mounted below the standby compass holds the correction card which is prepared during the compass swing. The compass card lists the heading for the pilot to steer to maintain a desired heading.
   E. The standby compass (figure 1) is shown in the swing-down position for normal operation. For better visibility, the pilot may place the standby compass in the stow position. To stow the standby compass, fold the card holder forward under the standby compass and then push the compass assembly aft and up against the airplane frame.





Standby Compass Location Diagram
Figure 1

MAGNETIC STANDBY COMPASS - MAINTENANCE PRACTICES

1. General
   A. The maintenance practices included in this section are general maintenance instructions that do not definitely fall within a specific category. Any maintenance practices that fall within a specific category will be provided in the applicable page blocks.
   B. Standby compass maintenance practices consist of compass swinging procedures (Ref 34-24-11, Maintenance Practices or Ref 34-24-11, Magnetic Standby Compass - Maintenance Practices (Compass Swing)) and demagnetization. Demagnetization of certain parts of the airplane may be necessary after it has been exposed to heavy static or has been struck by lightning. Evidence of magnetization may show up in the magnetic standby compass, in the gyro compass system, or in both in the form of excessive errors in the compass readouts. When necessary, a handheld compass such as the standby compass may be used to detect the presence of any unduly large magnetic fields in the airplane structure.
   C. If such magnetic fields are found, it will be necessary to demagnetize the area in the immediate vicinity using a degausser. Suitable degaussers are:
      (1) Luma Electric Equipment Co. Degausser, Model 85 (60 Hz 115 volt) 1607 Coining Dr., Toledo, Ohio 43612
      (2) Electro-Matic Products Co. Degausser, Models A1-3 & A1-4 (60 Hz 115 volt) 2235 North Knox Avenue, Chicago, Illinois 60639
      (3) Printz Electric Company Degausser, Model JD (50/60 Hz 120 volt), 20854 W. 8 Mile Rd., Southfield, Michigan 48075
2. Precautions
   A. Do not permit degausser to remain within 10 feet of any part of airplane UNLESS IT IS ENERGIZED.
   B. DO NOT operate degausser in vicinity of magnetic compass or remote compass transmitter.
   C. Care should be taken to ensure that degausser is not actually in contact with surface when it is used. (To be effective, however, it should be used as close to surface as possible).
3. Demagnetization of the Nose Gear Strut Assembly
   A. Energize degausser at a minimum distance of 10 feet from airplane and move it slowly from side of airplane towards nose gear assembly.
   B. When degausser is almost touching surface of strut, move it up and down in parallel strokes while gradually working around it, and over entire length. Include steering cylinders and supporting braces.
   C. Withdraw degausser to approximately 1 inch from strut and repeat process working around strut in opposite direction.
   D. Withdraw degausser to point at least 10 feet from airplane before turning it off.

E. Check strut for residual magnetism with small compass. Check that there is no compass deflection when it is held 3 feet from strut and moved about in this vicinity.

F. Repeat degaussing process until area is satisfactorily demagnetized.

NOTE: The following additional check should be made where practical:

G. Jack up nose of airplane.

H. Check airplane magnetic compass for deviation when nose gear is retracted. Check that there is none.

I. Lower gear.

J. Remove jack when strut is found to be properly demagnetized.

4. Demagnetization of Remote Compass Transmitter Area

A. Remove remote compass transmitter (flux valve) from mounting and place carefully in safe place. Ref 34-21-11, Remote Compass Transmitter - Removal/Installation.

B. Energize degausser at a minimum distance of 10 feet from airplane and move it toward remote compass transmitter area.

C. Demagnetize all steel bolts and screws in area.

D. Withdraw degausser to point at least 10 feet from airplane before turning it off.

E. Check area for residual magnetism with handheld compass and repeat stop C. as necessary.

NOTE: If a non-preindexed flux valve is not installed in its original index position, it will be necessary to swing compass to verify system accuracy.

F. Perform operational check of compass system (Ref 34-21-0, Compass System - Adjustment/Test).

MAGNETIC STANDBY COMPASS - MAINTENANCE PRACTICES (COMPASS SWING)

1. General
   A. This procedure contains one task. The procedure uses a standby compass calibrator instead of the usual tow-around procedure to do a swing of the standby compass.
   B. The task has two parts. The first part is to do a swing of the standby compass through four compass points (N, E, S, W). Use this procedure for compass calibration. The second part is to do a swing of the standby compass through 12 compass points that are approximately 30 degrees apart. Use this procedure to measure the remaining errors and to make sure the standby compass heading is accurate. Use this data to make the compass correction card. This card must stay with the standby compass.
   C. Do not park vehicles less than 250 feet from the airplane during the compass swing.
   D. Make sure the radio receivers are on during the compass swing.
   E. You can use the APU during the compass swing.
2. Standby Compass Calibrator Procedure

   NOTE: The standby compass may be calibrated using one of two different procedures. One of the procedures is contained in the following paragraphs. This procedure uses a compass calibrator to swing the compass. The other procedure is contained in Section 34-21-0, Compass System Maintenance Practices as a portion of the compass swing for the primary aircraft compass. It tows the aircraft to swing the compass.

   A. Equipment
      (1) Tow tractor
      (2) Nonmagnetic tools to adjust the standby magnetic compass
      (3) One Compass Correction Card - AN5823-1
      (4) Standby Compass Calibrator Kit, Honeywell - 2591553-901
      (5) Nonmagnetic tripod
   B. References
      (1) AMM 9-10-00/201, Towing
      (2) AMM 24-22-00/201, Manual Control (Apply Power)
   C. Calibrator Adjustment (Fig. 201)
      (1) Do this procedure to calibrate the standby compass calibrator (SCC) to the magnetic field at the location of the compass swing area:
         (a) Make sure that there are no vehicles or airplanes less than 250 feet away.
         (b) Make sure that there are no buildings less than 250 feet away.
         (c) Put a nonmagnetic tripod at the center of the compass swing area.

(d) Attach a master magnetic compass to the SCC with two mounting screws.

NOTE: You can use an accurate standby magnetic compass for a master magnetic compass. Make sure that the N-S and E-W adjustment screws are at neutral.

(e) Put the SCC/master compass assembly on the tripod.

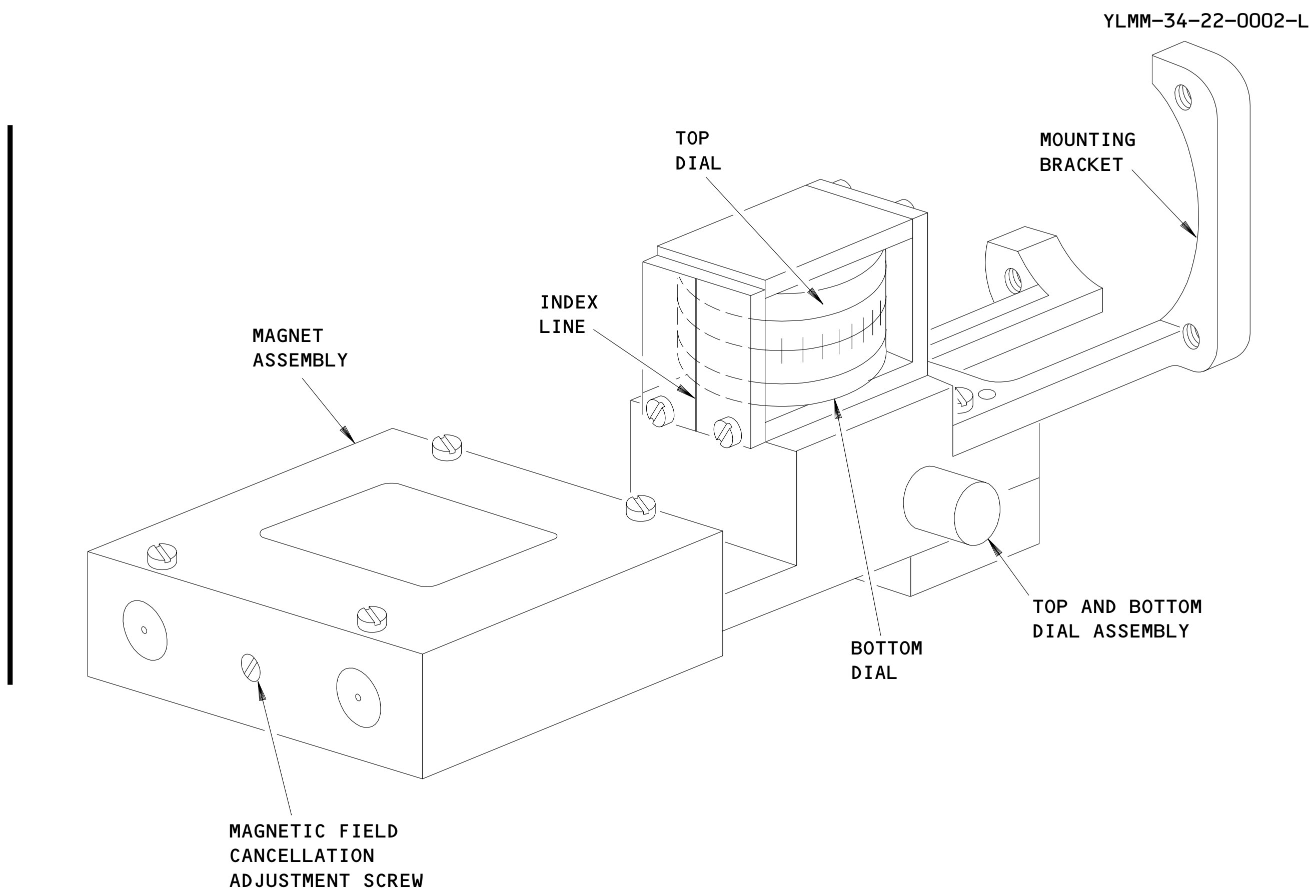
NOTE: Make sure the assembly is level.

(f) Turn the assembly until the master magnetic compass displays an indication of magnetic north (N).
(g) Turn the top and bottom SCC dials to show an indication of E at the index line.
(h) Make a record of the heading shown on the standby magnetic compass.
(i) Turn the top and bottom SCC dials to show and indication of W at the index line.
(j) Make a record of the heading shown on the standby magnetic compass.
(k) Turn the magnetic field cancellation adjustment screw on the SCC to decrease the heading errors in each direction (E and W) to a minimum.
(l) Continue to adjust the SCC dials for each direction (E and W). Then use the field cancellation adjustment screw until the errors are at a minimum.

NOTE: Continue to do this procedure until the error in each direction is at a minimum. Do not move the field cancellation adjustment screw before the compass swing is completed.

D. Prepare for a Compass Swing of the Standby Compass

NOTE: Ferromagnetic parts installed near the standby compass can cause compass heading errors. Make sure no ferromagnetic parts are near the captain's and the first officer's window frames. If found, replace the parts with nonmagnetic, corrosion-resistant steel (CRES) parts.



Standby Compass Calibrator
Figure 201

(1) Tow the airplane to the compass swing area (AMM 9-10-00/201).

NOTE: The compass swing area must be a level area with a smooth surface. It must be sufficiently strong to hold the weight of the airplane. The area must be large enough to tow or taxi the airplane. Make sure all vehicles other than the tow vehicle are not less than 250 feet from the airplane. The horizontal component of the earth's-magnetic field must be constant (+ 1 degree) in the test area. Measure the horizontal. component if magnetic material (such as a new building) is less than 600 feet from the compass swing area. Do not use the compass swing area if the horizontal component is not constant (+ 1 degree).

(2) Supply the electrical power (AMM 24-22-00/201).
(3) Energize all of the electronic equipment, radios, and control cabin lighting for the usual conditions that occur in flight.
(4) Make sure that the N-S and E-W adjustment screws on the standby compass are at neutral.

CAUTION: USE TOOLS THAT ARE NOT MAGNETIC. MAGNETIC TOOLS CAN CAUSE COMPASS DIFFERENCES.

(5) Make sure all vehicles but the tow truck are not less than 250 feet from the airplane.

E. Standby Compass Adjustment

NOTE: The steps that follow are for the four-point calibration swing.

(1) Turn the airplane to a magnetic heading of 0 degrees.
(2) Remove and keep the lower left and upper right mounting screws on the standby magnetic compass.
(3) Use the two mounting screws to install the SCC on the face of the standby magnetic compass.
(4) Turn the top and bottom SCC dials to show an indication of E at the index line.
(5) Make a record of the heading shown on the standby magnetic compass.
(6) Turn the top and bottom SCC dials to show an indication of W at the index line.
(7) Make a record of the heading shown on the standby magnetic compass.
(8) Turn the E-W adjustment screw on the standby magnetic compass until the error in the last two recorded values is at a minimum.

NOTE: Divide the error in each of the two directions as equally as possible.

(9) Turn the top and bottom SCC dials to show an indication of N at the index line.
(10) Make a record of the heading shown on the standby magnetic compass.
(11) Turn the top and bottom SCC dials to show an indication of S at the index line.
(12) Make a record of the heading shown on the standby magnetic compass.
(13) Turn the N-S adjustment screw on the standby magnetic compass until the error in the last two recorded values is at a minimum.

NOTE: Divide the error in each of the two directions as equally as possible.

(14) Continue to adjust the SCC dials for each pair of directions (E-W, N-S). Then turn the E-W and N-S adjustment screws until the errors are at a minimum.

NOTE: Continue to do this procedure until the error in each pair of directions is at a minimum. Start with E-W and turn the E-W adjustment screw. Then do N-S and turn the N-S adjustment screw.
When the errors are at a minimum, do the steps that follow for the 12-point accuracy swing.

(15) Adjust the SCC dials to indicate these magnetic headings:, 0, 30, 60, 90, 120, 150, 180, 210, 240, 270, 300, and 330 degrees.
(16) Make a record of the magnetic heading (MH) and standby compass heading (CH).
(17) Calculate and make a record of the difference for each 30 degree heading in the steer column on the compass correction card as follows:
D = MH - CH
(18) During these steps, make sure each difference Is less than + 8 degrees. Make sure the difference between the maximum positive and maximum negative value is 10 degrees or less.
(19) CAA CERTIFICATION:
During these steps, make sure each difference is less than + 5 degrees. Make sure the difference between the maximum positive and maximum negative value is 10 degrees or less.
(20) Put the SCC dials back to the N-N position.

NOTE: Do this step to keep the magnetic strength-high.

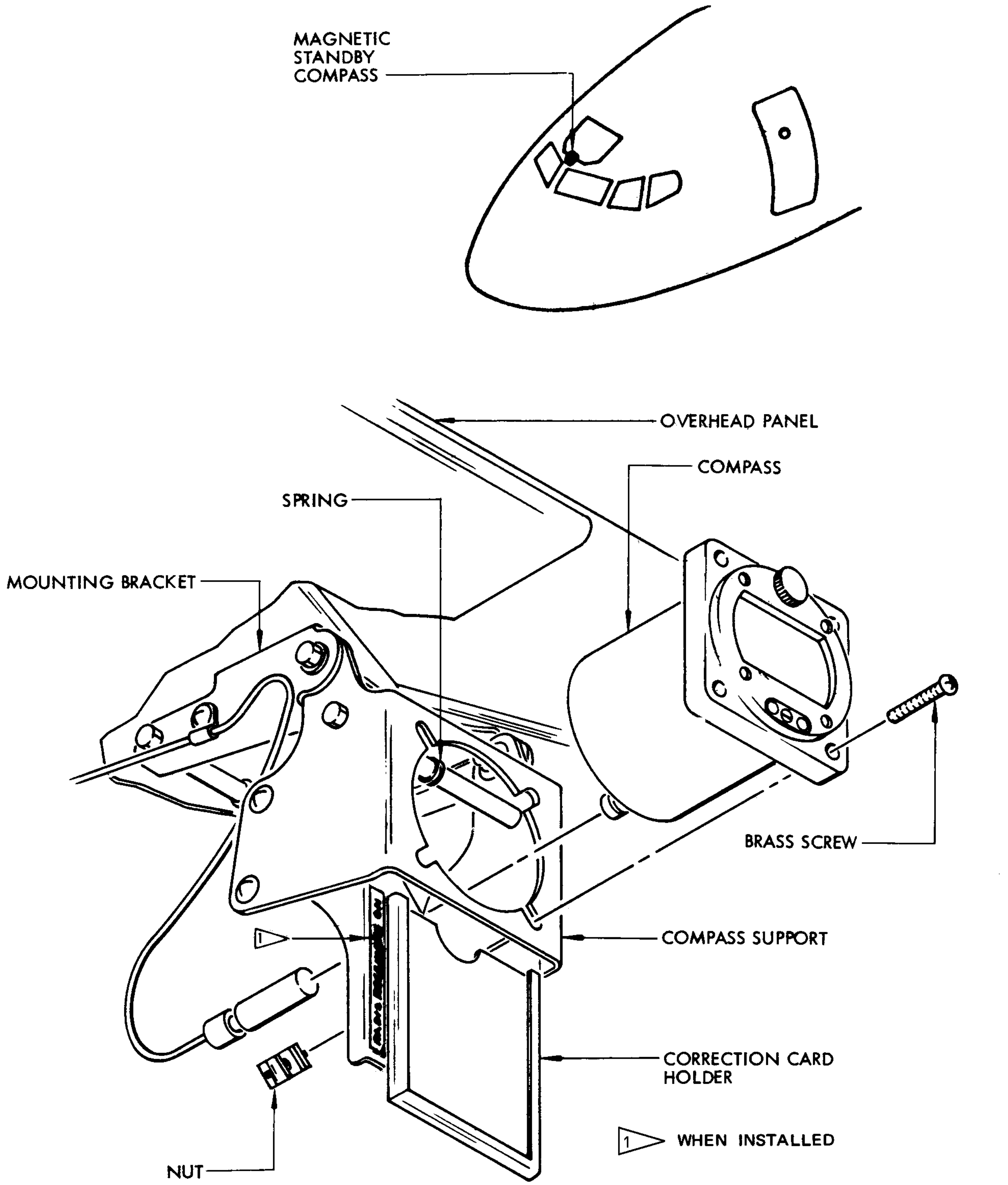
F. Put the airplane back to its usual condition.
(1) Remove the electrical power if it is not necessary (AMM 24-22-00/201).

MAGNETIC STANDBY COMPASS - REMOVAL/INSTALLATION

1. Remove Magnetic Compass
   A. Support compass and remove four brass screws and nuts holding compass to mount. (See figure 401.)
   B. Remove compass.
2. Install Magnetic Compass
   A. Insert compass in cutout in support. (See figure 401.)
   B. Install brass mounting screws and nuts.

      CAUTION: DO NOT USE STEEL SCREWS TO ATTACH COMPASS TO MOUNT.

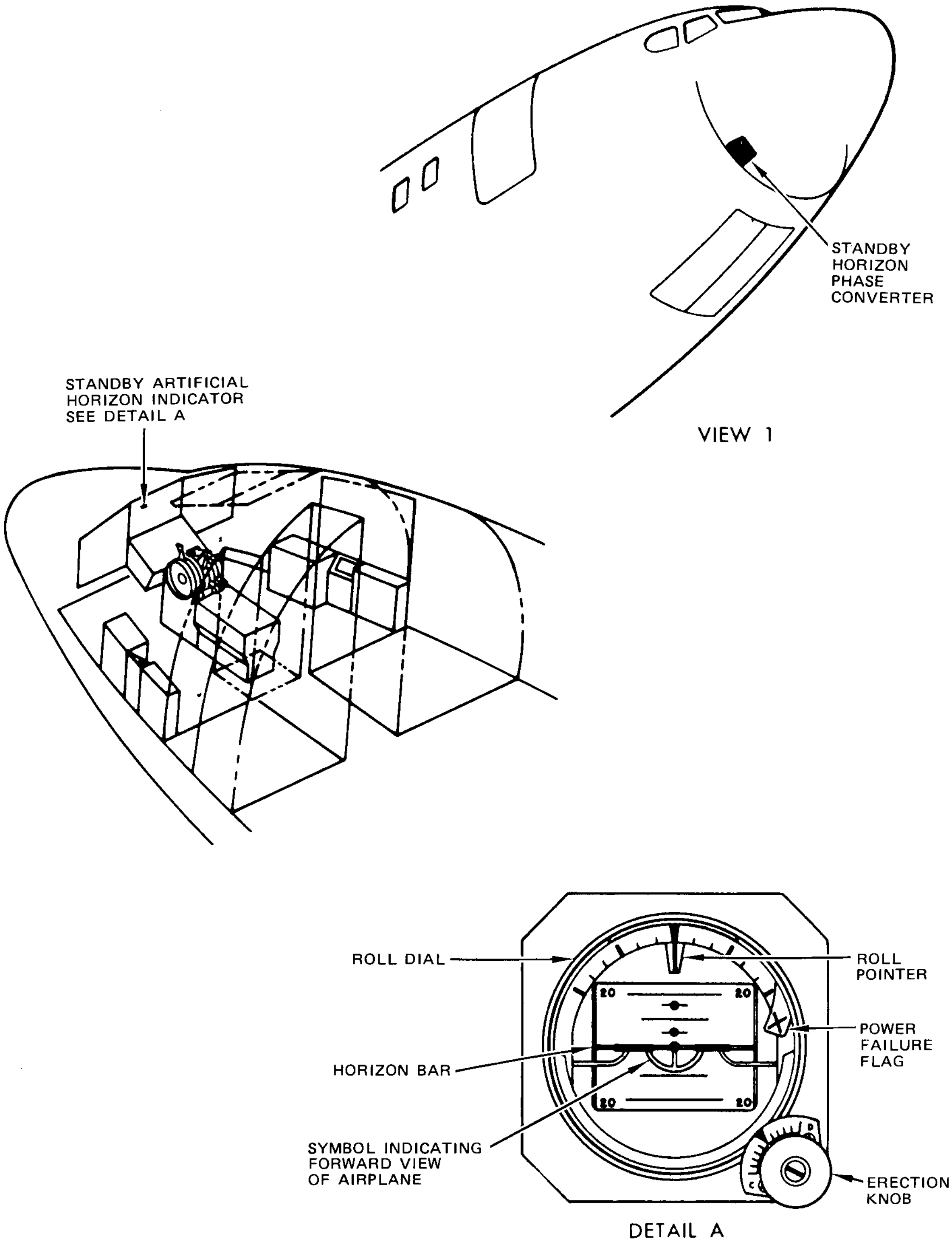
   C. Perform compass swing of magnetic standby compass. (Refer to 34-21-0, Compass System Maintenance Practices or 34-24-11, Magnetic Standby Compass Maintenance Practices.))



Magnetic Standby Compass Installation
Figure 401

STANDBY ARTIFICIAL HORIZON SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The standby artificial horizon indicator is fitted to the airplane as a backup system for the attitude reference system. It functions as a completely independent system which provides its own.
   B. On AR LV-JMW thru LV-JMZ, LV-JND and LV-JNE, the phase converter is located in the J6 junction box.
      On AR LV-JTD, LV-JTO, LV-LEB and on, the phase converter is located on a bracket in the forward electronics compartment (Sta 211, WL 180, RB L 20).
   C. The power supply for the system is provided from the STDBY HORIZ circuit breaker, situated on the P18 load control center.
2. Standby Artificial Horizon Indicator
   A. The standby artificial horizon indicator contains an electrically-driven gyro and the necessary mechanism for its erection, plus a power failure motor which operates the flag. The gyro has 360 degrees freedom of movement in roll, and is limited in pitch to 90 degrees in climb and 80 degrees in dive.
   B. The face of the instrument is shown in Fig. 1. The power failure flag will show in the event of power failure or incorrect phase rotation.
3. Standby Attitude Phase Converter
   A. The standby attitude phase converter is a transformer coupled RC phase shift network which converts single phase ac to three-phase ac for the gyro motor.
4. Operation
   A. The gyro warning flag should disappear from view when power is applied and the gyro should reach operational speed within 60 seconds. The normal rate of erection is 2.7 degrees per minute but a rapid erection device is fitted to bring about erection within a few seconds. The mechanism for rapid erection is actuated by pulling gently on the knob which is fitted to the front panel of the instrument. This operation should only be attempted when the airplane is in straight and level flight. When at rest the knob may be used to adjust the horizon line to zero pitch by turning the knob clockwise or counterclockwise. The gyro takes approximately 13 minutes to come to rest after power has been switched off.



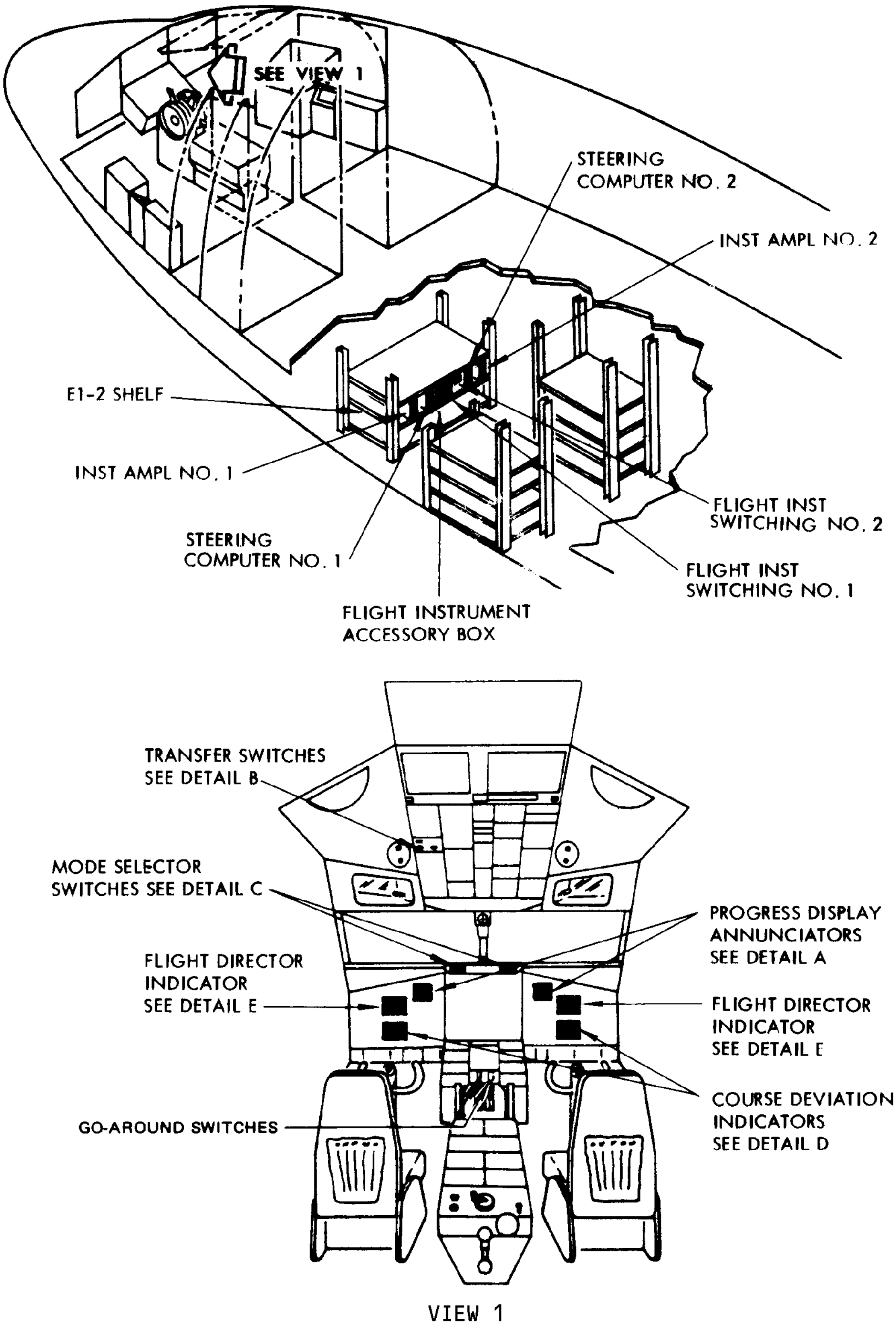
Standby Artificial Horizon System Component Location
Figure 1

STANDBY ARTIFICIAL HORIZON SYSTEM - ADJUSTMENT/TEST

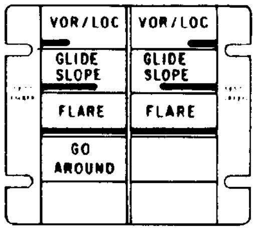
1. Standby Artificial Horizon System Test
   A. General
      (1) The standby artificial horizon indicator is a backup for the attitude reference system.
   B. Test
      (1) Close STBY HORIZ circuit breaker located on load control panel P18.
      (2) After the instrument has been energized for 30 seconds pull the caging knob, the "X" warning flag which is visible on the right hand side of the instrument face when the system is not operating, should disappear.
      (3) Five minutes after power has been on check that the indication of pitch and roll is within ± 1 degree of airplane's attitude as compared with aircraft primary attitude reference system on inclinometer.
      (4) Pull STBY HORIZ circuit breaker and observe that "X" flag comes into view.

FLIGHT DIRECTOR SYSTEMS - DESCRIPTION AND OPERATION

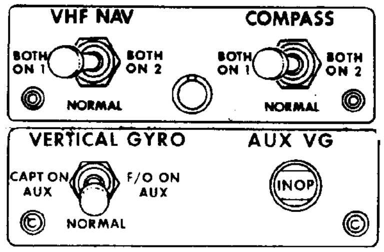
1. General
   A. The flight director systems (Collins FD-108) furnish the means to select a desired flight path, along with lateral (bank) and vertical (pitch) steering commands, which if followed, will enable intercept and tracking of the desired flight path. The desired flight path may be a magnetic heading, go around after aborted approach, VOR course, or localizer (LOC)/glide slope (GS) approach beams. In addition, altitude information obtained from the air data computer is converted to pitch steering command information and utilized in holding the airplane at a desired altitude.
   B. Two systems are installed, captain's (No. 1) and first officer's (No. 2). Each is comprised of the following units: approach horizon (flight director) indicator (FDI), course deviation indicator (CDI), flight instrument amplifier, steering computer, flight director control panel, and flight director annunciator. A single air data computer supplies altitude signals to both systems. Location of the components is shown in figure 1. The FDI, annunciator and CDI are shown in figure 1.
   C. Each flight director computer receives input signals from its associated VOR/GS navigation system (GS, or VOR/LOC deviation signals), and attitude reference system (bank and pitch displacement/error signals), and attitude reference system (bank and pitch displacement/error signals). (See 34-22-0, Attitude Reference Systems and 34-31-0, VOR/GS Navigation Systems.) Selected heading error and course datum error signals are also fed to each computer from its associated CDI. These error signals are developed with reference to airplane magnetic heading (34-21-0, Compass Systems) when the desired magnetic heading, or course is selected. The selected heading, or course error signals, in combination with the radio deviation, bank, or pitch signals are fed to the computer, which then develops the bank and pitch steering signals. The bank and pitch steering commands are displayed by the steering command display system (V-pointers) in the FDI. The pointers are servo-operated and move to indicate the desired attitude in bank and pitch. The airplane is then maneuvered so as to align the pointers against the miniature airplane symbol. When the pointers are aligned with the airplane symbol, the real airplane will be in the correct attitude to intercept, or remain on the desired flight path. An overall simplified schematic of the systems is shown in figure 2.
   D. Signal and/or power failure in a system, including failure of the steering command display, along with similar types of failure in any one of the data supplying systems (VOR/LOC and GS navigation, attitude reference, or compass systems), is monitored through a warning flag circuit in the computer, and a warning then displayed by the COMPUTER warning flag in the FDI.
   E. Deleted



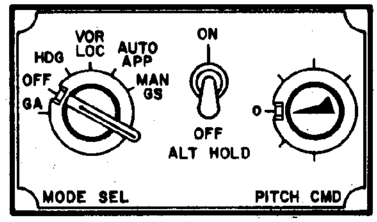
Flight Director System Component Location
Figure 1 (Sheet 1)



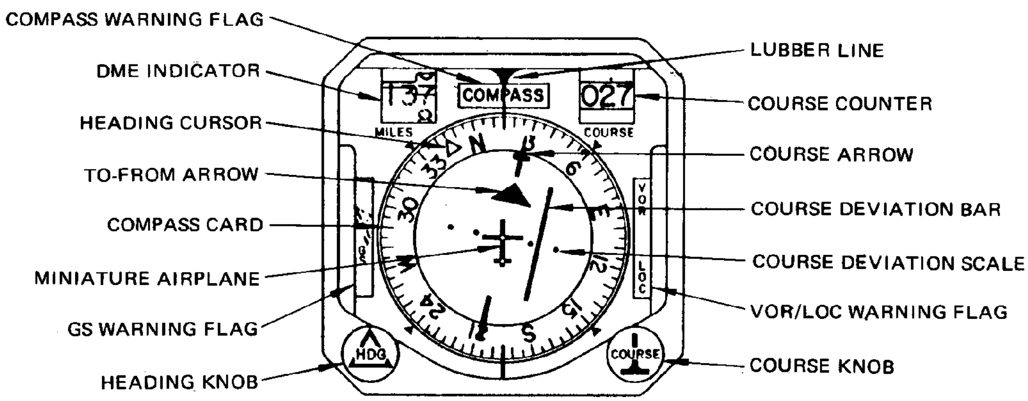
APPROACH PROGRESS
DISPLAY ANNUNCIATOR
DETAIL A



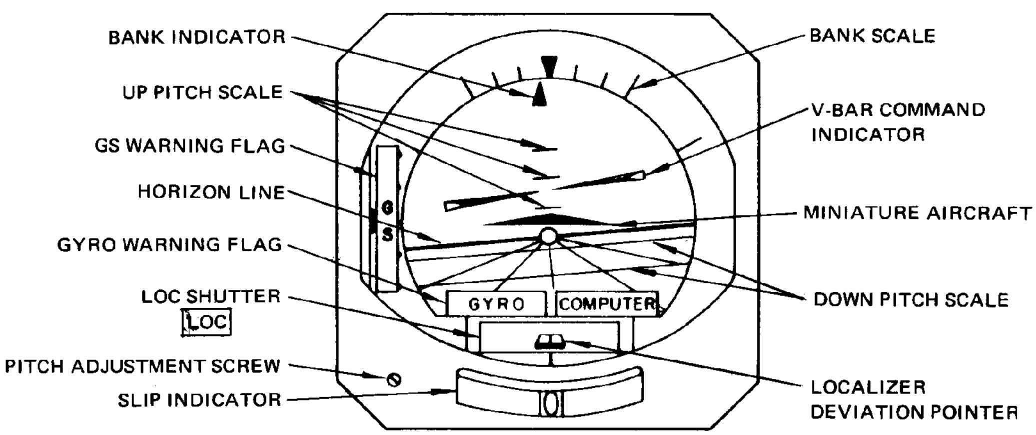
TRANSFER SWITCHES
DETAIL B



MODE SELECTOR SWITCH
DETAIL C



COURSE DEVIATION INDICATOR
DETAIL D



FLIGHT DIRECTOR INDICATOR
DETAIL E

Flight Director System Component Location
Figure 1 (Sheet 2)

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2. Flight Director Indicator

A. The flight director indicator (Fig. 1) furnishes a pictorial display of airplane attitude in bank and pitch, along with the bank and pitch steering commands. The bank and pitch steering commands are superimposed over an artificial horizon on a moving tape. The horizon line displays the pitch attitude of the airplane, and is read relative to the miniature airplane symbol. The bank marker index, connected by gearing to the tape, is read against the bank scale to give airplane bank angle. The tape is positioned by the bank and pitch servos, controlled by the bank and pitch attitude signals fed from the associated attitude reference system.

B. The steering command display includes the two V-pointers, the-localizer deviation pointer and the glide slope deviation pointer.

C. The two V-pointers flank the miniature airplane symbol and form a spread "V." They are servo-positioned, and controlled by the bank and pitch steering command signals fed from the computer (Fig. 3). The airplane is maneuvered in response to the commands displayed by the pointers. When the pointers are aligned alongside the miniature airplane symbol, the real airplane will be in the correct attitude to intercept, or remain on the selected flight path.

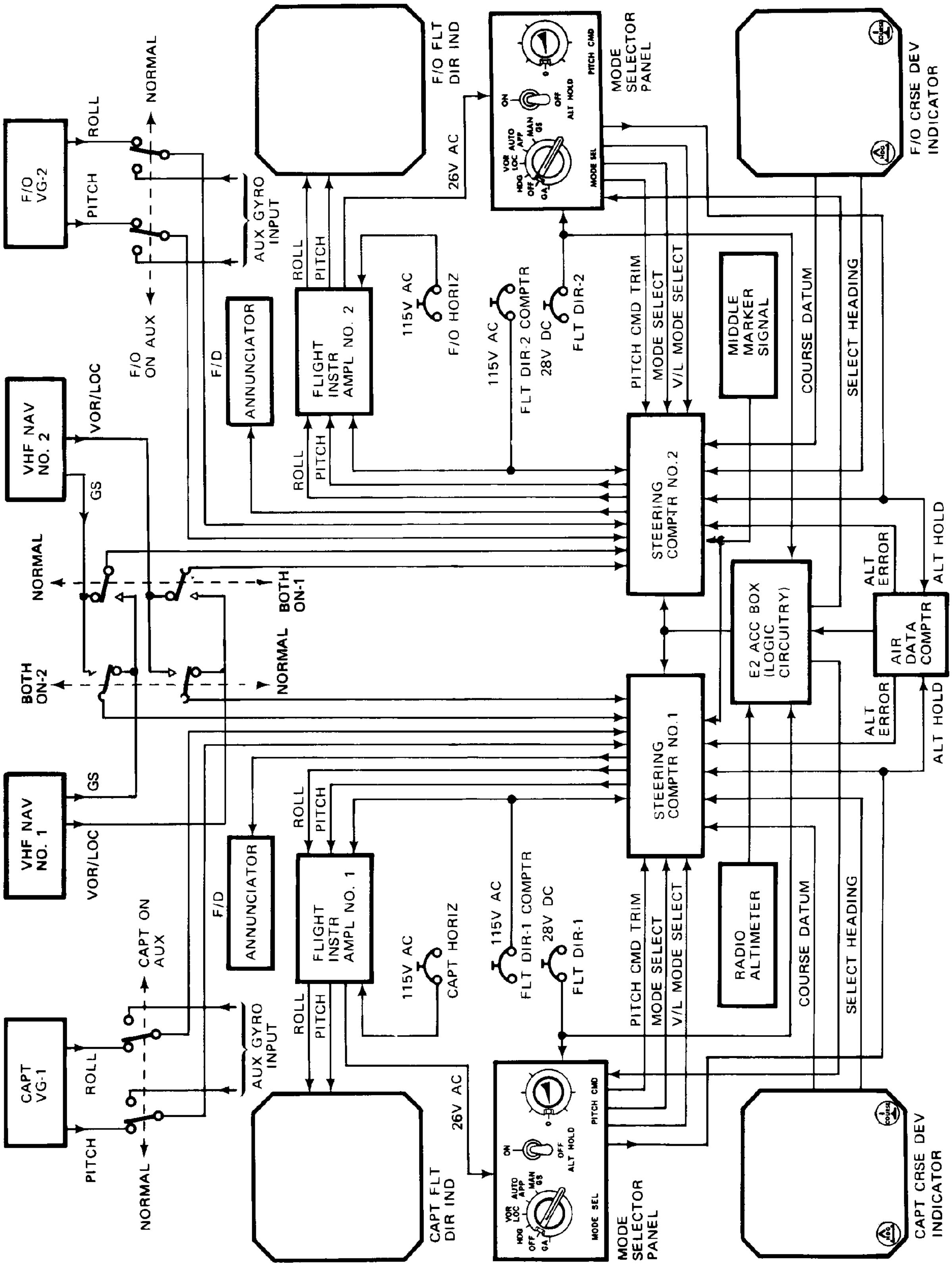
D. The localizer deviation pointer at the lower part of the display indicates deviation from the localizer beam and is in view only after capture of the glide slope. Pointer deviation to the left indicates that the localizer beam is left of the airplane.

E. The localizer warning flag covers the localizer deviation pointer when the LOC receiver is not in use or in the event of failure of the LOC receiver or signal.

F. A horizontal pointer at the left side of the FDI indicates airplane deviation from the center of the glide slope beam. Pointer deflection above the center reference mark indicates that the center of the glide slope beam is above the airplane. The glide slope pointer is biased out of sight when the VOR/GS receiver is tuned to a VOR frequency.

G. A warning flag marked GS covers the glide slope pointer and scale in the event of failure of GS receiver or signal. It is biased out of sight when the glide slope deviation signal is valid or when the VOR/GS receiver is tuned to a VOR frequency.

H. The GYRO warning flag monitors the bank and pitch attitude display. It is spring-loaded to become visible when the bank and/or pitch attitude channels are not operating properly, or when power is lost to the attitude reference system, or the vertical gyro has not erected properly. It is energized out of sight when the vertical gyro is operating correctly.



Flight Director Systems Schematic
Figure 2

I. The COMPUTER warning flag monitors power to the computer, plus the various input warning signals from the systems feeding signals to the computer in specific combinations, depending upon the mode of operation of the computer. This is described under Operation. An adjustment screw at the left lower corner of the FDI enables screwdriver adjustment of the vertical trim of the horizon line with respect to the fixed miniature airplane symbol. The slip indicator, at the bottom center of the indicator, is a conventional aid with a weighted ball in a liquid filled tube. It enables the pilot to monitor airplane slip, or skid in turns.

3. <u>Course Deviation Indicator</u>

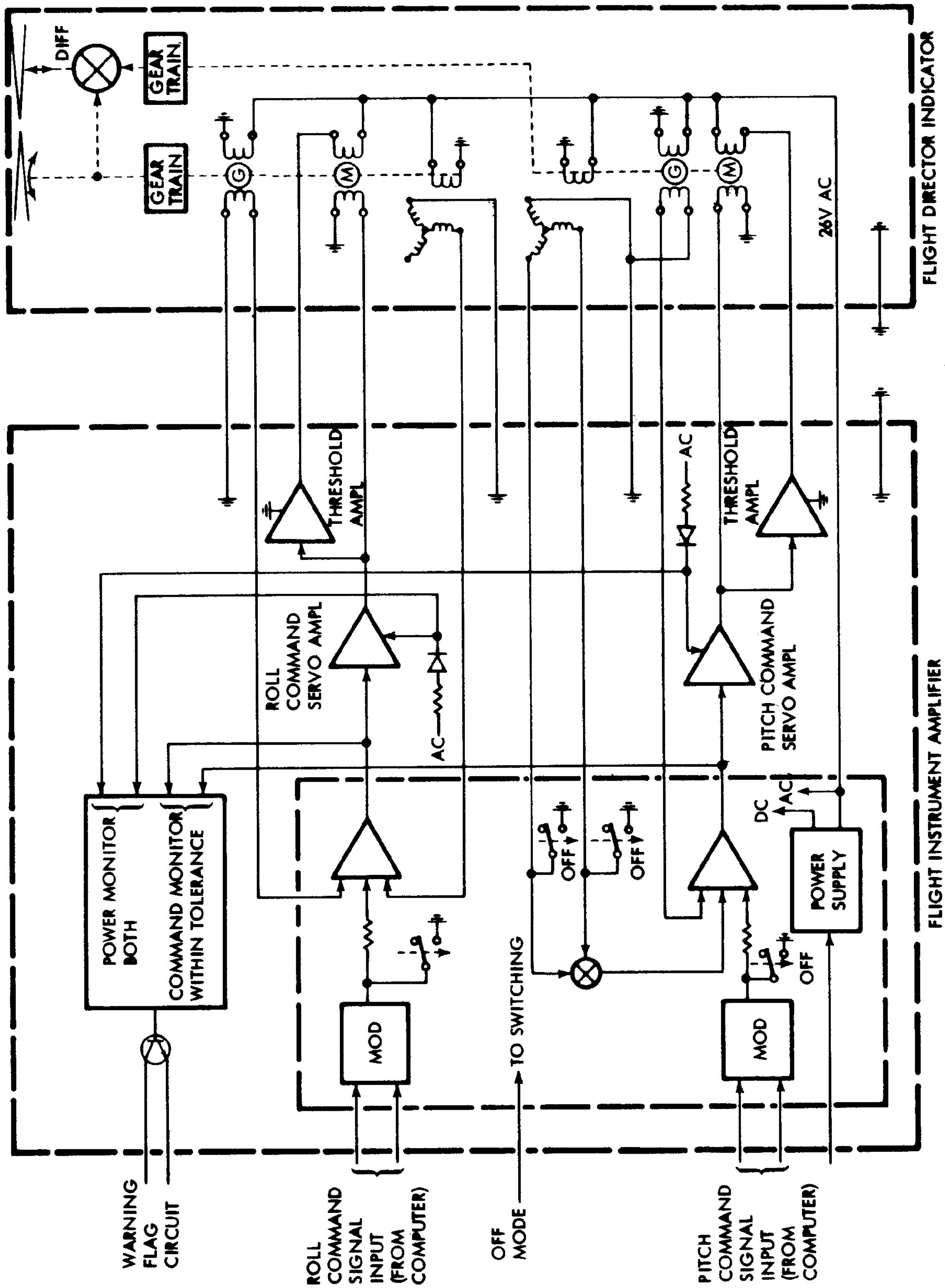
A. The course deviation indicator (Fig. 1) presents a plan view of the navigation situation. The display includes airplane heading with reference to magnetic north, the selected magnetic heading, and the selected radio course.

B. Magnetic heading of the airplane is displayed by the compass card which is read with respect to the lubber line. The card is positioned through the heading servo, and controlled by the heading signal fed from the compass system. This heading signal is also fed to the stators of the heading and course select synchros in the indicator.

C. The heading cursor, which is ganged to the rotor of the heading select synchro, indicates the selected magnetic heading, and is positioned by the HEADING knob. At the time the heading cursor is positioned, the heading error signal, developed in the rotor of the heading select synchro, is fed to the computer. Once set, the heading cursor rotates with the compass card, giving a continuous display of selected heading, and any heading deviation. Heading information is used in computing bank and pitch steering commands when the flight director system is in heading mode.

D. The central portion of the indicator is the deviation section. It contains the course arrow, course deviation bar, course deviation scale and TO-FROM arrows. The entire section rotates with the compass card as the heading of the airplane changes. The various components of the section perform their individual functions as follows:

(1) The course arrow points in the direction of the selected radio course, and is positioned by the COURSE knob. At the time the course arrow is positioned to the selected course, the selected course is also displayed in digital form on the course counter at the upper right corner of the indicator. The course arrow is ganged to the rotor of the course select synchro. The course error signal, developed in the rotor of the course select synchro, is fed to the computer. The course error signal is used in generating bank and roll steering commands when the flight director system is in a radio mode.



Flight Director System Steering Command Servo Loops
Figure 3

(2) The course deviation bar provides an indication of course deviation and moves perpendicular to its length to provide the indication. The amount of course deviation is read off the course deviation scale. When the bar completes the course arrow, the airplane has no lateral displacement from the selected course. The bar is actuated by the VOR or LOC deviation signal fed from the navigation unit.

(3) To-from indication about a VOR station is shown by the broad arrows. They operate only when the associated navigation unit is tuned to a VOR frequency. The arrows appear singly to indicate the direction to the station.

E. The miniature airplane in the center of the indicator simulates the position of the real airplane in flight. It is compared against heading cursor, course arrow and course deviation bar positions, to obtain the pictorial representation of the airplane heading, and any deviation from the desired heading or course.

F. The VOR/LOC flag functions to indicate navigation unit malfunction. It appears whenever power is turned off, or lost to the navigation unit, or whenever the VOR/LOC signal fails, or falls below a certain predetermined value. The GS warning flag functions in a similar manner, but with respect to the GS receiver. When the system is operating in HDG or the VOR submode, the GS deviation pointer and flag are biased out of sight.

G. The GS pointer displays airplane position relative to the glide slope. It is read against the GS scale to indicate position of the airplane with respect to the glide slope. A deflection up indicates that the airplane is below the glide slope, and a deflection down that it is above the glide slope. The pointer is actuated by the GS deviation signal. When the system is operating in HDG or VOR submodes, the pointer and flag are biased out of sight.

H. The MILES counter at the upper left corner of the CDI is the DME indicator (Ref 34-55-0).

4. Flight Instrument Amplifier Rack

A. The flight instrument amplifier rack contains five separate servo-amplifiers, plus a dual (bank and pitch) channel signal converter and a power and command monitor. The servo-amplifiers furnish the servo actuating power required by the bank and pitch servos in the FDI, the heading servo in the CDI, and the bank and pitch steering command servos in the FDI. The signal converter functions to change the dc steering command signals to the ac signals required by the steering command servos. The steering command signals control positioning of the V-pointers in the FDI (Fig. 3). Error signals induced in the steering command synchros are fed to the bank and pitch steering command servo-amplifiers, where after amplification, they are fed back to actuate the bank and pitch steering command servomotors. The motors then operate and drive in such a direction as to null out the induced error signals, at the same time repositioning the pointers to display the new command. The OFF switching in two legs of the position synchro provides the biasing to deflect the V-pointers out of view when the system is OFF. The power and command monitor functions to monitor both power to, and any errors in, the roll and pitch command servo loops. Warning is given through the computer warning flag. When power is available and the servo errors are within tolerance, the transistor is switched on, completing part of the circuit which holds the computer flag out of view.

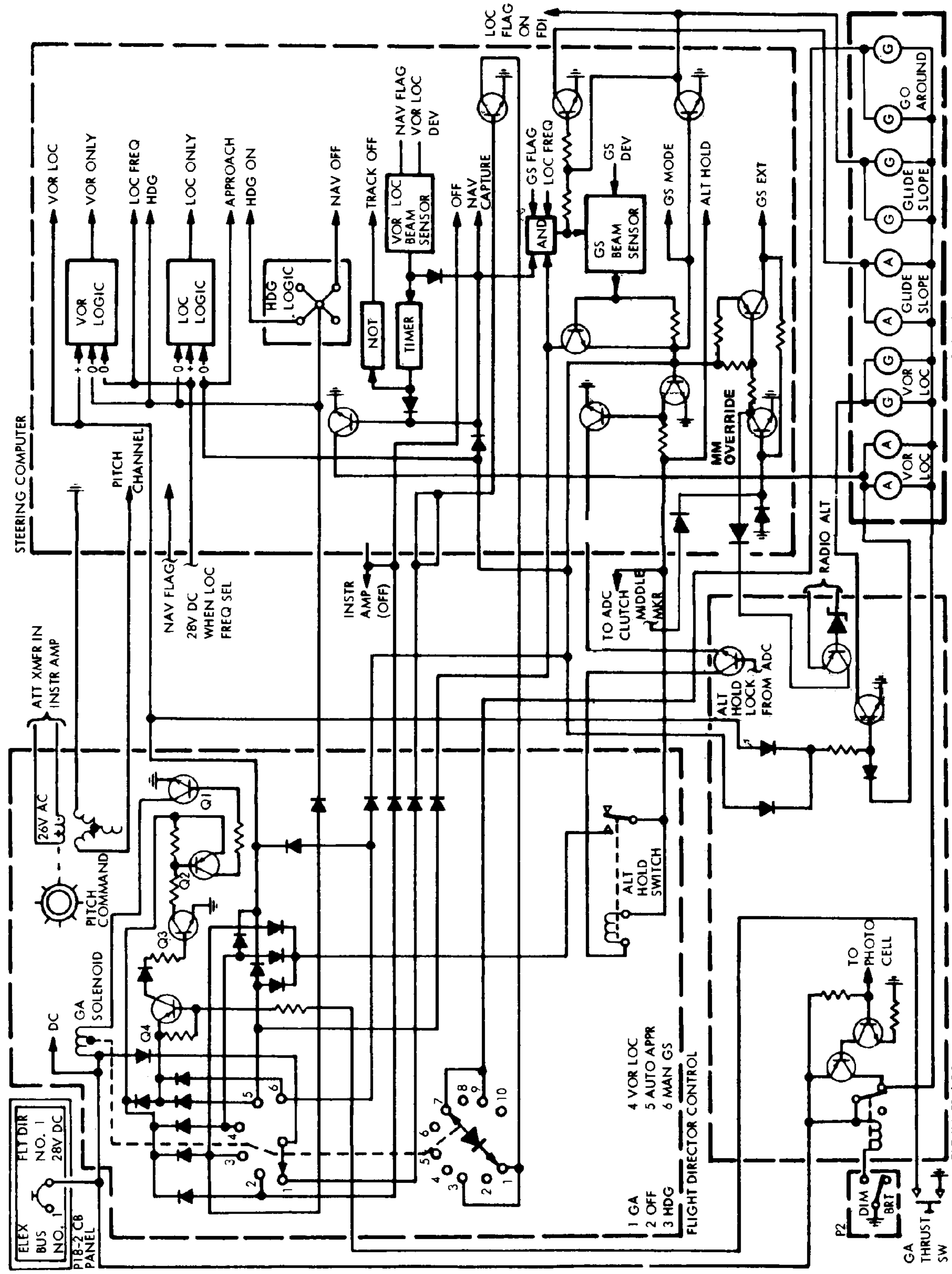
5. Flight Director Control Panel (Mode Selector Panel)

A. The flight director control panel enables switching of the computer to operate in any one of the following modes: GA, HDG, VOR/LOC, AUTO APPR, or MAN GS. It contains the mode selector switch, altitude hold switch and manual pitch command knob.

B. The mode selector switch has the following positions: GA, OFF, HDG, VOR/LOC, AUTO APPR, OR MAN GS. The switch has to be depressed in order to select MAN GS. The switch will automatically return to the GA position from AUTO APPR, or MAN GS position, when go-around switch on thrust level is pressed.

C. The V/L ARM submode is selected by turning the mode selector switch to the HDG mode and keeping it there for 3 seconds, before turning to VOR/LOC mode. When this is done, the VOR/LOC (amber) indicator lights in the display annunciator, and remains in sight until the display is switched to the green VOR/LOC indicator light by mode signals from the computer. This happens when the appropriate beam is sensed. The circuit is shown in figure 4.

D. When GS ARM submode is selected by turning the mode selector switch to AUTO APPR, the amber glide slope light will appear at the annunciator. The mode and annunciation will automatically switch to the green glide slope light when the GS beam capture sensor is operated. (See Operation.) Alternatively, GS mode can be obtained by turning the mode selector switch to MAN GS.



Flight Director System Control Schematic
Figure 4

E. Altitude hold selection is provided by the altitude hold toggle switch, which is solenoid held in the ON position. If the mode selector switch is in either the AUTO APPR or MAN GS mode, and the glide slope has been intercepted, the solenoid circuit will be opened and the altitude hold switch released from the ON position.

F. The manual pitch command synchro, controlled by the manual pitch command knob supplies the manual pitch command signal to the computer.

6. Flight Director (Steering) Computer

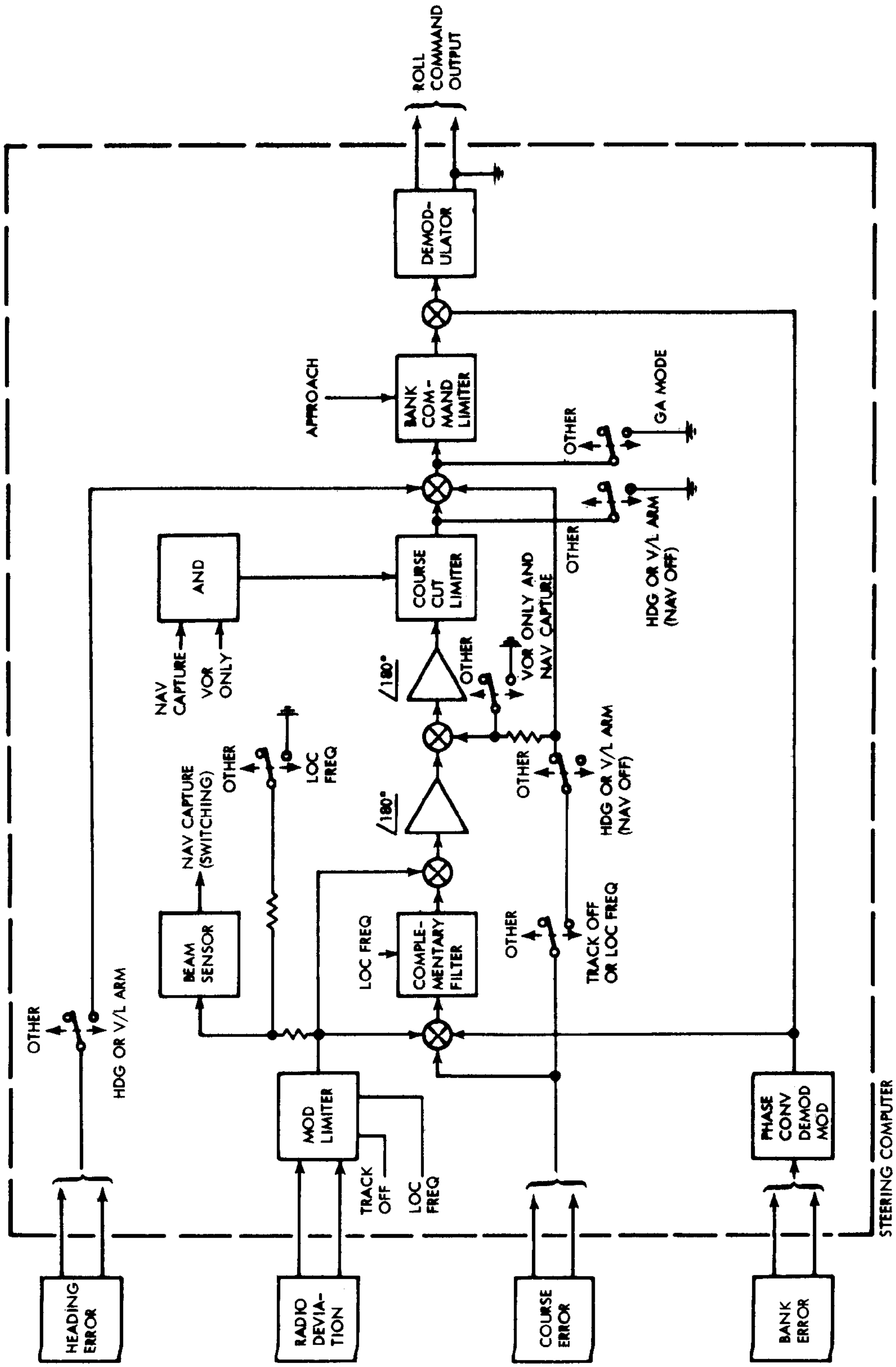
A. The flight director computer provides the bank and pitch steering commands, which are displayed by the V-pointers in the FDI. These signals are dc signals, and are fed first of all to a dual channel signal converter in the flight instrument amplifier, where they are changed to ac signals, before being fed to operate the bank and pitch steering command servos in the FDI.

B. Information concerning radio deviation, selected magnetic heading and bank angle are used in the computation of the bank steering command signal, and information concerning pitch attitude, glide slope deviation, or altitude information are used in the computation of the pitch steering command signal. From this division of the input signals, two signal flow channels result in the computer: bank and pitch. The functioning of these channels is described later under Operation and is shown in figures 5 and 6.

C. The computer also contains two power supply circuits, termed attitude and heading power. The inputs however are connected together, and power to the computer is from a single source. The computer also contains a warning flag circuit plus several logic circuits. The logic circuits are used in switching of the computer to the various modes of operation. Some of these logic circuits are controlled from the flight director control panel, and others are controlled internally in the computer. The COMPUTER warning flag is actuated by a warning flag signal fed from the warning flag circuit. This circuit in addition to monitoring all of the various input warning signals, also monitors the V-pointers and power to the computer.

7. Air Data Computer

A. The air data computer is a device for detecting changes in static pressure altitude, as supplied from the airplane static system. Once altitude hold is selected, any deviation in pressure altitude will result in an error voltage being supplied to the computer pitch channel. The pitch command output positions the V-pointers to visually show the airplane attitude necessary for a smooth return to the desired pressure altitude. The air data computer is described in Chapter 34-12-0, Air Data Pressure Instruments.



Flight Director Systems - Bank Channel
Figure 5

8. Operation

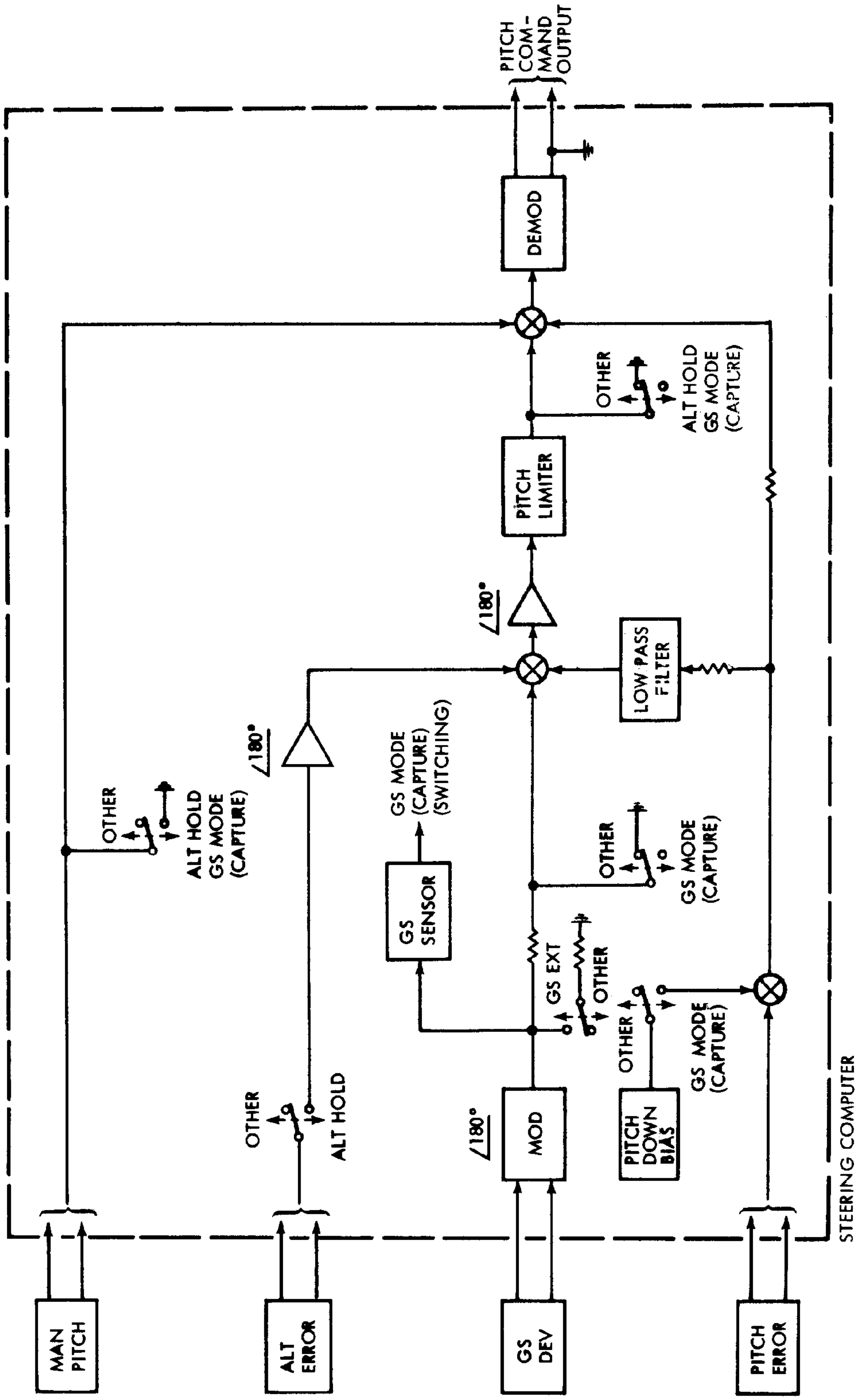
A. Before the flight director systems can be operated, the attitude reference, compass and VOR/GS navigation circuit breakers must be closed and the systems must be operational. Closing the circuit breakers shown in figure 2 provides power to the flight director system.

B. When power is applied to a system, if the mode selector is in OFF position, the V-pointers and the COMPUTER warning flag in the FDI move out of sight. In this mode, as well as in all other modes, the COMPUTER warning flag monitors power to the computer. Turning the mode selector switch to the GA mode, or any one of the other modes causes the V-pointers to come into sight.

C. In the bank channel, the bank error signal (which is developed in the vertical gyro when the airplane is banked) is compared against the computed bank command signal to yield the bank steering command signal. Similarly, in the pitch channel, the pitch error signal, or altitude error signal is compared against the computed pitch command signal to yield the pitch steering command signal. The bank channel is shown in figure 5 and the pitch channel in figure 6. On these schematics, the controlling inputs that alter gain and control switching of the circuits in the various modes are shown. For example: in figure 5, the LOC ONLY input shows that the gain of the complementary filter is altered when functioning in VOR/LOC mode with a LOC frequency selected. Switches are shown, but in the computer, transistors handle the switching.

D. GA Mode

(1) The GA (Go-Around) mode of operation is used for an aborted approach. When initiated, the roll channel commands wings level and the pitch channel commands the climb out angle. The system will automatically switch to GA, from either of the glide slope modes, by depressing the go-around switch on the thrust lever. This energizes the switch release solenoid and permits the mode selector to spring back to the GA position. When the flight director system is in GA mode, the annunciators show GA and the computer produces steering commands in the bank and pitch channels. In the bank channel, the computer receives bank error signals from the vertical gyro which are directed through the flight instrument amplifier as bank steering commands to the FDI. The V-pointers indicate the corrections necessary to keep the airplane level in a lateral plane. In the pitch channel, the computer receives and mixes pitch error signals from the vertical gyro with a preset pitch up bias to produce pitch steering command signals. These steering command signals are supplied through the flight instrument amplifier to the V-pointers in the FDI. The V-pointers then indicate the necessary corrections to keep the airplane in the desired flight path in respect to pitch.



Flight Director Systems - Pitch Channel
Figure 6

E. HDG Mode

(1) In the HDG mode of operation, the selected heading error signal is fed to the computer from the heading select synchro in the CDI. The signal is limited and becomes the bank command signal. The bank error signal is fed from the vertical gyro. This signal is phase detected and amplified before being mixed with the bank command signal to yield a deviation signal, which is further amplified and converted to a dc signal before becoming the bank steering command signal. The steering commands are limited to 25 degrees in roll and 12 degrees in pitch. In the pitch channel, the pitch error signal and the manual pitch command signals are mixed together to provide the pitch steering command signal. The altitude error signal may be used in place of the manual pitch command signal. In this case, the altitude error signal is mixed with the filtered and phase converted pitch error signal, which is amplified and limited, before being mixed with the original pitch error signal to produce the pitch steering command signal. In this mode of operation the COMPUTER warning flag monitors power to the computer, and gyro and compass inputs, and the steering command display system (V-pointers).

F. VOR/LOC Mode

(1) In the VOR/LOC mode, the bank steering command signal provides steering command information to intercept and hold the selected VOR, or LOC course. The steering commands are limited to 25 degrees in roll and 12 degrees in pitch. The action is similar for either course, except that in LOC tighter coupling is provided. This is accomplished through logic circuits that alter the various gains and limits of the associated circuits. In this mode, course rate, VOR, or LOC deviation rate, and bank rate signals are applied to the complementary filter to become a synthetic radio deviation damping signal which opposes the VOR, or LOC deviation signal in order to prevent overshoot in following the command. The course cut limiter combines the output signal from the complementary filter with the VOR, or LOC deviation signal and the course error (phase reversed) signal to produce the bank command signal. The action of the limiter permits interception of the selected course at a fixed heading angle of 45 degrees relative to the selected course.

(2) At approximately 8 to 10 degrees from the VOR beam center, the computer will generate roll commands due to high gain radio rate signals. The V/L beam sensor is activated by V/L mode voltage with NAV FLAG, AND VOR/LOC inputs. When the airplane is within 5 degrees of a selected VOR radial a "nav-capture" signal, generated at the V/L beam sensor, changes the course cut limiter output and reduces the command intercept-angle from 45 to 17 degrees. At the same time however, differentiating circuitry in the computer is producing a damping signal (radio rate) which initiates a roll command. If the airplane path follows the V-pointer roll commands the resultant flight path will be a smooth flare-in to the course. At the same time, the "nav-capture" signal activates a timer which, after a predetermined delay also provides a "nav-capture" output and through an inverter produces a 0-volt level "track-off" signal. The timer output is provided to maintain "nav-capture" switching during temporary loss of a V/L beam sensor input and the "track-off" signal is used for gain switching in the computer.

(3) When flying to a LOC course, "nav-capture" is generated when the airplane intercepts within 2 degrees of the LOC course, but there is no reduction of the course cut limiter output to change the intercept angle from 45 to 17 degrees.

(4) On VOR or LOC the bank command signal is mixed with the bank error signal to produce the bank steering command signal. Pitch channel operation in this mode is identical to that in the HDG mode.

G. V/L Arm Submode

(1) In the V/L arm submode, the pilot is enabled to choose the angle of capture for a VOR or LOC course through selection of a desired heading. This is different from VOR/LOC mode, where an intercept angle of 45° is maintained by the course cut limiter. V/L arm submode is engaged by turning the mode selector switch momentarily (3 seconds or more) to the HDG position and then to the VOR/LOC position. When this is done, the amber VOR/LOC indicator lights at the annunciator, and the system operates initially as in HDG mode. When "nav-capture" is generated at the V/L beam sensor, the computer is automatically switched to either the VOR, or LOC mode of operation. The annunciation changes from an amber VOR/LOC light to a green VOR/LOC light. At this time the COMPUTER warning flag will also begin monitoring VOR information from the navigation unit. The identical COMPUTER warning flag action takes place when capturing a LOC beam, except that the warning flag circuit now monitors LOC inputs instead of VOR information. In this mode, pitch channel operation is the same as that described for HDG mode.

H. AUTO APPR Mode

(1) In the AUTO APPR mode, bank channel operation is as described for V/L arm submode on LOC, or LOC mode. In the initial phase of the GS mode, the GS arm (amber) indicator lights at the annunciator providing a "GS flag" signal is generated by the GS receiver and a "LOC freq" select signal is delivered from the VOR/LOC receiver. When the center of the glide slope beam is reached, a "GS capture" signal is generated in the GS beam sensor and the computer is automatically switched to the final approach phase. The COMPUTER warning flag at this time begins monitoring glide slope information in addition to its other inputs. At this time also, the green glide slope indicator lights in the annunciator. In the final approach phase, the bank channel utilizes the same signals as described for LOC mode and the roll commands are limited to 15°. At glide slope capture, a pitchdown bias causes an approximate 2° pitchdown command. This bias is cancelled after 14 seconds by the low pass filter which passes changing signals but shunts steady state signals. After glide slope capture the localizer deviation pointer comes into view and indicates deviation from the localizer beam at approximately twice the sensitivity of the CDI bar. The complementary filter provides for crosswind correction. Its output is combined with the LOC deviation signal to provide a damping signal. The difference between these two signals becomes the bank command signal. The bank command signal is then mixed with the bank error signal to yield

(2) the bank steering command signal. The pitch channel uses the pitch error signal and the glide slope deviation signal. Glide slope deviation information is amplified, fed to the glide slope capture sensor and is mixed with the pitch error signal to produce the pitch steering command signal. Glide slope gain programming is utilized to provide optimum control on the approach to the runway. The program is initiated after GS capture by the 1500 feet radio altimeter trip or by a radio altimeter invalid signal and is refined when the middle marker signal is received.

I. MAN GS Mode

(1) In this mode of operation, automatic sensing of the glide slope beam, through the GS beam sensor is eliminated. A voltage from the mode selector switch, replaces the GS capture signal, and loss of GS AUTO mode voltage deactivates the sensor. Also, the annunciator will show GS. Other than this, the operation remains identical to that in GS AUTO mode.

J. Altitude Hold Submode

(1) The altitude hold submode can be used in HDG, VOR/LOC and GS AUTO modes. If the air data computer fails, a warning flag voltage switches off a transistor in the accessory box. This opens the altitude hold circuit and the ALTITUDE HOLD switch returns to the OFF position. In GS AUTO mode, as soon as glide slope capture is generated a transistor is turned off in the altitude hold solenoid circuit and the ALTITUDE HOLD switch springs to the OFF position. This enables glide slope deviation signals to be used in the pitch channel instead of pressure altitude deviation signals.

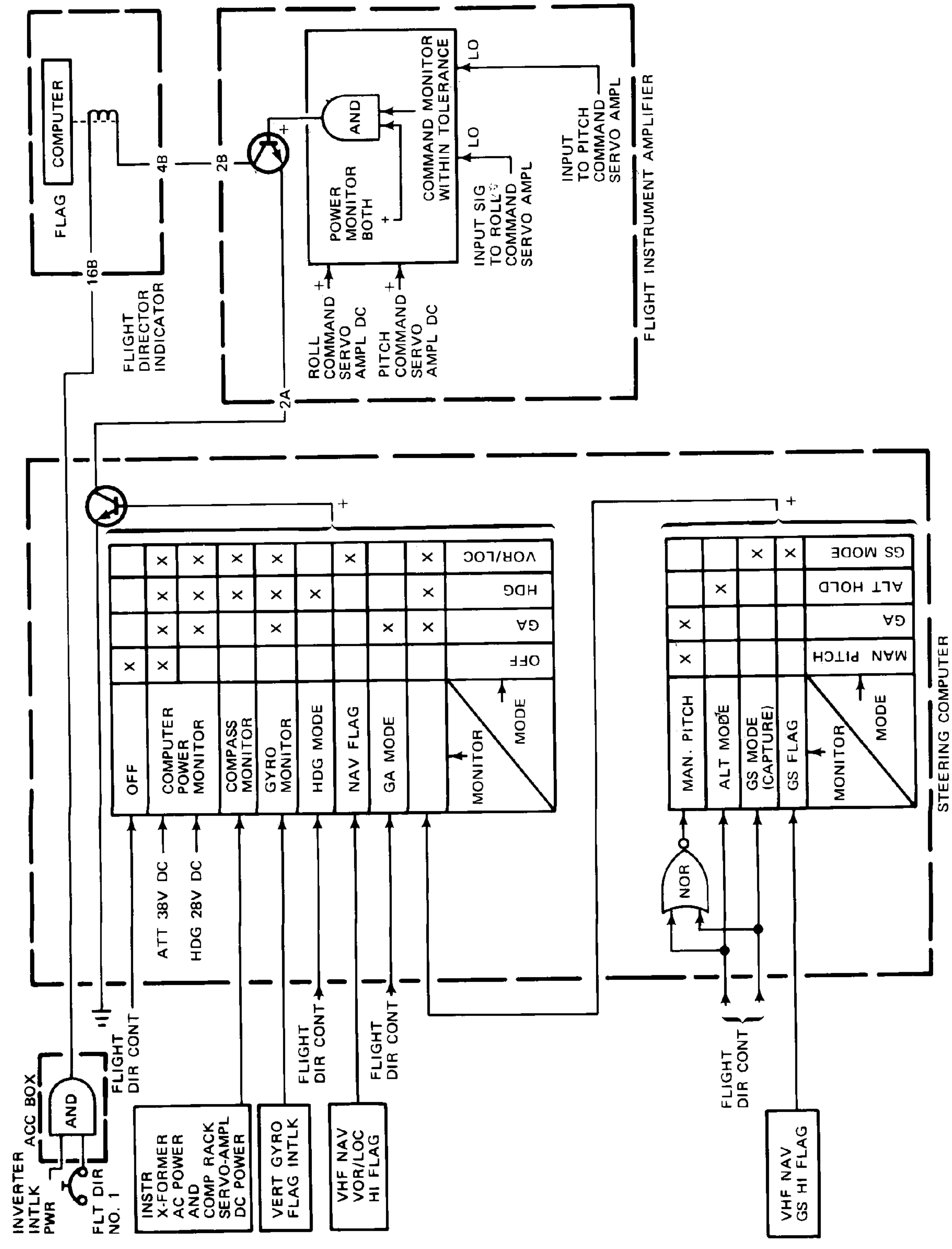
K. Computer Flag Circuit Operation

(1) Figure 7 shows the schematic diagram of the computer flag circuit which provides a ground to operate the computer flag in the FDI. The primary transistors in the circuit are Q3 in the flight instrument amplifier and Q8 in the computer. If the various inputs to the computer are correct and the computer is functioning properly, Q3 and Q8 will be conducting to complete the computer flag solenoid circuit and pull the COMPUTER flag out of view.

(2) The command monitor warning module in the instrument amplifier monitors the B+ supply and the input signal to the roll command servo-amplifier and the pitch command servo-amplifier. If the power supply to either of the amplifiers is defective, Q3 will be biased to cutoff. Similarly if the input signal to either of the amplifiers indicates more than 4.5 degrees of error between the command signal and the V-pointer position signal, Q3 will be biased to cutoff.

(3) The roll monitor and pitch monitor circuits in the computer comprise transistor/diode logic to monitor the various inputs. In the illustration, the input signals are read horizontally and the relevant inputs for a specific mode are read vertically. The "x" indicates that the signal voltage is present. For example, in the roll monitor block in VOR/LOC mode the input signals which are essential to complete the AND gate are PITCH MONITOR, NAV FLAG GYRO MONITOR, COMPASS MONITOR, HEADING POWER (28V DC) and ATTITUDE POWER (38V DC).

(4) The pitch monitor signal is produced in the pitch monitor circuit from various inputs, depending on the operating mode of the pitch channel which has been selected. If MAN GS is selected, the computer will monitor ALT HOLD or MAN PITCH inputs in the pitch monitor until the computer is switched to GS mode by a glide slope capture voltage. In AUTO APPR mode selection, although GS is selected, the computer will be in either MANUAL PITCH or ALT HOLD before GS capture.



Flight Director Computer Flag Schematic
Figure 7

FLIGHT DIRECTOR SYSTEM - TROUBLESHOOTING

1. General
   A. The most effective method of troubleshooting a defective Flight Director System is to perform the tests described in the Flight Director System - Adjustment/Test.
   B. First of all check for power to the defective system, and establish that power is available. Next, substitute units from the operating system where possible. After each change note whether the defective system becomes operative. If interchanging units does not clear up the trouble, system wiring should be checked for continuity and the connectors for security of connection.

FLIGHT DIRECTOR SYSTEM - ADJUSTMENT/TEST

1. Flight Director System Test

A. General

(1) The following systems should be in proper operating condition before performing the flight director adjustment/test: VHF Navigation, Compass, Attitude Reference, Marker Beacon, Radio Altimeter and Air Data System.

(2) Before changing the normal configuration of unfamiliar systems to facilitate the use of test equipment, or using unfamiliar test equipment, refer to the section in the maintenance manual covering the system in question for special instructions.

B. Equipment and Materials

(1) Signal Generator - Collins 479T-2, or equivalent (two required)
(2) Test Oscillator - BC-376, or equivalent
(3) Calibrated Tilt Tables (three required)
(4) Air Pressure Test Set
(5) Radio Altimeter Test Set
(6) Stopwatch

C. Prepare for System Test

(1) Install vertical gyros on calibrated tilt tables and level in roll and pitch (AMM 34-22-0, for precautions on removing gyros).

(2) Set up one 479T-2 signal generator on a localizer frequency with maximum power output.

(3) Energize P6 and P18 circuit breaker panels, by use of external power or APU.

(4) Make sure that captain's and first officer's instruments, No. 1 and 2 navigation and No. 1 and 2 air data circuit breakers are closed.

(5) Synchronize compass system and allow a few minutes for gyros to stabilize. Tune both VHF NAV systems to localizer frequency selected at signal generator.

(6) Place VERTICAL GYRO, VHF NAV, and compass transfer switches in their normal position.

D. Test Flight Director System

(1) Warning Flag Tests

(a) Make sure that computer flags in HDIs (FDIs) are in view when functional requirements are as tabulated below. Restore functional requirements to NORMAL condition after each step and make sure that flags are retracted. Make sure that each side (captain's or first officer's) operates independently of other side.

Functional Requirements for Pilots HDI (FDI) Indicator Computer Flags to be in View			
STEP	MODE SELECTOR POSITION	Captain's Side	First Officer's Side
1	HDG	Flt Dir-1 AC C/B Open	Flt Dir-2 AC C/B Open
2	HDG	Flt Dir-1 DC C/B Open	Flt Dir-2 DC C/B Open
3	HDG	Compass-1 C/B Open	Compass-2 C/B Open
4	HDG	Instr XFMR-1 C/B Open	Instr XFMR-2 C/B Open
5	HDG	Compass Switch to BOTH ON 2 for steps 5, 6, and 7 RMDI-1 (ALT) C/B Open	Compass Switch BOTH ON 1 for steps 5, 6, and 7 RMDI-2 (ALT) C/B Open
6	HDG	Instr XFMR-1 C/B Open	Instr XFMR-2 C/B Open
7	HDG	Instr XFMR-2 C/B Open	Instr XFMR-1 C/B Open
8	HDG	VG Switch to CAPT ON AUX, Aux Vert Gyro (STBY ALT) C/B Open	VG Switch to F/O ON AUX, Aux Vert Gyro (ALT) C/B Open
9	HDG	Vert Gyro-1 C/B Open	Vert Gyro-2 C/B Open
10	HDG	Standby Power Switch in BATT	OMIT
11	VOR/LOC	VHF NAV-1 and VHF NAV-2 Tuned to VOR frequency with VOR signal applied for in steps 11 and 12. VHF NAV-1 VOR/LOC C/B Open	VHF NAV-1 and VHF NAV-2 Tuned to VOR frequency with VOR signal applied for in steps 11 and 12. VHF NAV-2 VOR/LOC C/B Open

12	VOR/LOC	VHF NAV switch in BOTH ON 2. VHF NAV-2 VOR/LOC C/B Open	VHF NAV switch in BOTH ON 1. VHF NAV-1 VOR/LOC C/B Open
13	MAN GS	VHF NAV-1 and VHF NAV-2 Tuned to LOC frequency with LOC signal applied for in steps 13 and 14. VHF NAV-1 GS C/B Open	VHF NAV-1 and VHF NAV-2 Tuned to LOC frequency with LOC signal applied for in steps 13 and 14. VHF NAV-2 GS C/B Open
14	MAN GS	VHF NAV switch in BOTH ON 2. VHF NAV-2 GS C/B Open	VHF NAV switch in BOTH ON 1. VHF NAV-1 GS C/B Open

(2) Command and Annunciation Tests

(a) Place BRIGHT-DIM-TEST switch in BRIGHT and mode selectors in OFF position. Make sure that V-bars and computer flags are retracted from view. Press TEST GREEN switch on pilots' panel mounted approach progress display annunciator panel. Make sure that the green light on all labeled panels comes on. Press TEST AMBER switch and make sure that all labeled panels except G/A come on with an amber light.

(b) Press TEST GREEN switch and place BRIGHT-DIM-TEST switch in TEST position. Make sure that the approach progress lighting does not change. Place BRIGHT-DIM-TEST switch in DIM position. In DIM position only, make sure that the brightness of display varies with intensity of incident light on photocells.

(3) HDG Mode Tests

(a) For captain's instruments place vertical gyro, compass and VHF NAV switches in NORMAL position for all tests unless otherwise noted. Set No. 1 vertical gyro to level and PITCH CMD knob on mode selector to 0.

(b) Set mode selector in HDG position. Set heading cursor on CDI to airplane heading. Tilt vertical gyro No. 1 in pitch axis to align V-bars with miniature airplane symbol. Allow 30 seconds for V-bars to stabilize. Make sure that the tilt table pitch axis reads 0 ±3 degrees.

(c) Set heading cursor to angle 10 degrees greater than airplane heading. Make sure that the V-bar rotates to right and returns to zero command with 15 ±5 degrees right roll of vertical gyro.
(d) Set heading cursor to angle 90 degrees greater than airplane heading. Make sure that the V-bar returns to zero command with 28 ±5 degrees right roll of vertical gyro.
(e) Repeat steps (c) and (d) for left deflection and roll. Return vertical gyro to level and heading cursor to airplane heading.
(f) Repeat step (c) with COMPASS switch in BOTH ON 2 position. Return transfer switch to NORMAL.
(g) Repeat steps (a) thru (f) using first officer's components. Place COMPASS switch in BOTH ON 1 position in step (f).

(4) Auxiliary Gyro Switching Test
(a) Set vertical gyro transfer switch in CAPT ON AUX position. Tilt auxiliary vertical gyro toward right roll and nose down attitude. Check that captain's V-bar indicates fly-left and fly-up.
(b) Repeat step (a) above with vertical gyro transfer witch set in F/O ON AUX position and watching first officer's HDI. Return switch to normal position

(5) Pitch Command and Altitude Hold Modes Tests

CAUTION: DAMAGE MAY RESULT FROM MISUSE OF AIR PRESSURE TEST SET. REFER TO AMM 34-11-0 FOR CAUTIONS AND INSTRUCTIONS, ON OPERATING TEST SET. RATE OF ALTITUDE CHANGE MUST NOT EXCEED 5000 FEET PER MINUTE AND STATIC PRESSURE MUST NOT EXCEED 1000 FEET AFTER ALT HOLD SWITCH IS ON.

(a) For captain's instruments, set PITCH CMD knob CW to 10 degrees (each mark indicates 5 degrees). Check that V-bar indicates pitchup command and returns to zero command with 10 ±5 degrees pitchup tilt of vertical gyro No. 1.
(b) Repeat step (a) for CCW 10 degrees pitch down command. Return vertical gyro to level. Return PITCH CMD knob to zero.
(c) Set pressure altitude to 1000 feet above field elevation and ALT HOLD switch to ON. Check that V-bar indicates zero degree pitch command. Check that ALT HOLD switch remains ON.

(d) Press 60-foot switch on air data computer and then release. Check that V-bar indicates pitchdown command of 1.0 degree or more while switch is pressed.
(e) Rotate PITCH CMD knob throughout its range and return to zero. Check that V-bar remains stationary.
(f) Slowly reduce pressure of reference altitude to simulate 250-foot increase in altitude. Make sure that the V-bar indicates 5.0 degrees or more pitchdown command (pitch command may be read against horizon display in HDI). Return pressure altitude to 1000 feet above field elevation.
(g) Repeat step (e) for 250-foot decrease in altitude (increase in pressure). Check that V-bar indicates pitch up. Return pressure altitude to 1000 feet above field elevation.
(h) Tilt vertical gyro No. 1 to 20 degrees nose up. Check that V-bar deflects downward momentarily. Wait 30 seconds then reduce vertical gyro tilt angle until V-bar indicates zero command. At this point, check that vertical gyro tilt angle is 12 ±3 degrees. Return vertical gyro to level. Disregard position of symbolic airplane with respect to horizon.
(i) Wait 1 minute, then repeat step (h) for opposite direction.
(j) Open air data No. 1 ac circuit breaker. Check that ALT HOLD switch returns to OFF. Close circuit breaker.
(k) Repeat steps (a) thru (j) using first officer's components and vertical gyro No. 2
(l) Return static pressure to ambient.

(6) VOR/LOC Mode Test
(a) Slew captain's CDI to airplane heading of zero degree and keep it there for remainder of this test. Set VOR signal generator to 180 degrees radial (south of VOR station). Tune VHF NAV-1 receiver to signal generator frequency. Set course arrow on captain's CDI to 15 degrees. Set captain's mode selector in VOR/LOC position. Make sure that the captain's approach progress display indicates amber VOR/LOC. Rotate captain's heading select knob. Make sure that the captain's V-bar responds. Return heading cursor to zero degree.
(b) Slowly reduce course error so that deviation bar on CDI is moving towards zero deviation. Make sure that the approach progress display changes to read green VOR/LOC when deviation bar reaches 1.0 ±0.5 dot. Make sure that the V-bar indicates fly-left command. Make sure that the TO-FROM arrow points toward zero degree.

(c) Set course arrow to airplane heading. Wait 2 minutes, then center V-bars by tilting vertical gyro in roll axis. Check that tilt table reads 0 ±4 degrees. Return vertical gyro to level.

(d) Repeat steps (a) thru (c) for opposite course error (345 degrees). Check that V-bar indicates fly-right command.

(e) Set captain's course arrow to 45 degrees. Set mode selector in MAN GS position and then back to VOR/LOC position. Check that captain's approach progress display indicates green VOR/ LOC. Wait 2 minutes, then center V-bars by tilting vertical gyro No. 1 in roll axis. Check that tilt table reads 0 ±10 degrees. Check that V-bar does not respond to movement of heading cursor. Return vertical gyro to level.

(f) Repeat step (e) with course arrow set to 315 degrees.

(g) Set signal generator on localizer frequency. Place captain's mode selector in HDG position and course arrow to 330 degrees. Tune VHF NAV-1 receiver to localizer frequency and apply greater than 2 dots fly-right deviation signal. Return mode selector to VOR/LOC position. Check that captain's approach progress display indicates amber VOR/LOC. Check that V-bar responds to movement of heading cursor.

(h) Slowly reduce localizer deviation signal. Check that approach progress display changes to green VOR/LOC when deviation bar reaches 2.0 ±0.5 dots. Check that V-bar indicates fly-right command. Return course arrow to airplane heading.

(i) Reduce localizer deviation to zero. Wait 2 minutes, then center V-bars by tilting vertical gyro No. 1 in roll axis. Make sure that the tilt table reads 0 ±4 degrees. Return vertical gyro to level.

(j) Repeat steps (g) thru (i) for opposite course error (30 degrees).

(k) Set up second signal generator for glide slope operation. Select GS frequency that pairs localizer frequency of first signal generator and set GS deviation to zero. Place ALT HOLD switch in ON position. Apply fly-up deviation signal in excess of 1 dot. Check that V-bar does not change.

(l) Reduce glide slope signal to zero deviation. When glide slope deviation reaches zero, check that ALT HOLD switch remains on and V-bar does not change. Check that approach progress display indicates green VOR/LOC.

(m) Repeat steps (a) through (l) using first officer's components.

(n) Repeat steps (g) and (h) with compass transfer switches in BOTH ON 2 and VHF NAV transfer switch in BOTH ON NAV-1 position.

(o) Repeat steps (g) and (h) with VHF NAV transfer switch in BOTH ON NAV-2 position.

(7) Approach Automatic AUTO APPR Mode Test

(a) For captain's instruments, tune VHF NAV-1 receiver to signal generator's localizer frequency and apply greater than 2-dot fly-left localizer signal. Set mode selector in HDG and then in AUTO APPR position with ALT HOLD switch in OFF position. Make sure that the approach progress display indicates amber VOR/LOC. Make sure that the V-bar follows heading cursor for bank commands and PITCH CMD knob for pitch commands.

(b) Apply fly-up glide slope signal. Place ALT HOLD switch in ON position. Check that switch remains ON and V-bar no longer follows PITCH CMD knob. Return heading cursor to airplane heading and PITCH CMD knob to zero. If radio altimeter is installed, set radio altitude above 1500 feet.

(c) With fly-up glide slope signal applied, slowly reduce localizer deviation signal. Check that captain's approach progress display changes to indicate green VOR/LOC and amber GLIDE SLOPE when deviation bar reaches 2 ±1/2 dots. Check that V-bar indicates fly-left command. Reduce localizer signal to zero deviation.

(d) Slowly reduce glide slope deviation signal- Check that approach progress display indicates green VOR/LOC and green GLIDE SLOPE when glide slope deviation is 0 ±0.5 dot. Check that V-bar momentarily indicates fly-down command but stabilizes to zero command in 45 seconds. Check that V-bar does not follow PITCH CMD knob and ALT HOLD switch returns to OFF position.

(e) Set marker beacon sensitivity switch to low. Recycle mode selector switch to OFF and back to AUTO APPR. Check that GS annunciator is green, then quickly apply GS deviation for 7 degrees of pitchup command. Check that command reduces to approximately 3 degrees in 120 seconds. W Set Gs deviation for 3 degrees pitchup command. Apply middle marker signal for 3 seconds. Check that the pitch command reduces to 1.5 degrees in 7.5 ±2 seconds.

(f) Repeat steps (a) thru (h) using first officer's components.

(g) Repeat steps (a) thru (d) with VHF NAV transfer switch in BOTH ON 2 position using first officer's instruments.

(8) Manual Glide Slope (MAN GS) mode Test

(a) Repeat steps (a) thru (d) with VHF NAV transfer switch in BOTH ON 1 position.

(b) Place mode selector in HDG and ALT HOLD switch in ON position. Apply greater than 2-dot fly-up and fly-right glide slope and localizer signals. Return mode selector to MAN GS position. Make sure that the approach progress display indicates green VOR/LOC and green GLIDE SLOPE. Make sure that the V-bar indicates fly-up and fly-right and does not follow PITCH CMD knob. Make sure that the ALT HOLD switch returns to OFF position.

(c) Recycle mode selector to OFF and back to AUTO APPR. Set LOC deviation to zero and GS deviation to 7 degrees pitchup command.
(d) Place mode selector in MAN GS position. Check that V-bar command reduces to 3 degrees within 5 seconds.
(e) Set marker beacon sensitivity switch to low and apply a middle marker signal for 3 seconds. Check that V-bar command reduces to 1.5 degrees over a period of 7.5 ±2.0 seconds.
(f) Set course cursor to 0 degree. Apply 1-dot of fly-right deviation. Wait 2 minutes, then make sure that the V-bar returns to zero with 15 degrees right roll of vertical gyro.
(g) Repeat steps (a) thru (e) using the first officer's components.

(9) Go-Around Mode Test
(a) For captain's instruments, select 10 degrees nosedown command on PITCH CMD knob. With mode selector still set in MAN GS position and with 2-dot glide slope and localizer signals applied, momentarily press thrust lever GO-AROUND switch. Make sure that the approach progress display indicates green GO-AROUND. Make sure that the mode selector rotates to GA position. Make sure that the V-bar indicates pitchup and wings level command. Center V-bars by tilting vertical gyro No. 1 in pitch axis. Make sure that the tilt table reads 14 ±5 degrees pitchup. Reduce localizer deviation to zero. Return vertical gyro to zero.
(b) Set mode selector in AUTO APPR position. Make sure that the GO-AROUND annunciator turns off. Momentarily press thrust lever GO-AROUND switch. Make sure that the mode switch rotates to GA position and approach progress display indicates green GO-AROUND only.
(c) Repeat steps (a) and (b) using first officer's components.

E. Restore Airplane to Normal
(1) Turn all systems off and remove test equipment.
(2) Open attitude reference system circuit breakers and install vertical gyros.
(3) Close attitude reference system circuit breakers and verify this system indicates attitude of airplane.
(4) Determine if there is further need for electrical power, if not, remove power.

FLIGHT INSTRUMENT ACCESSORY UNIT - REMOVAL/INSTALLATION

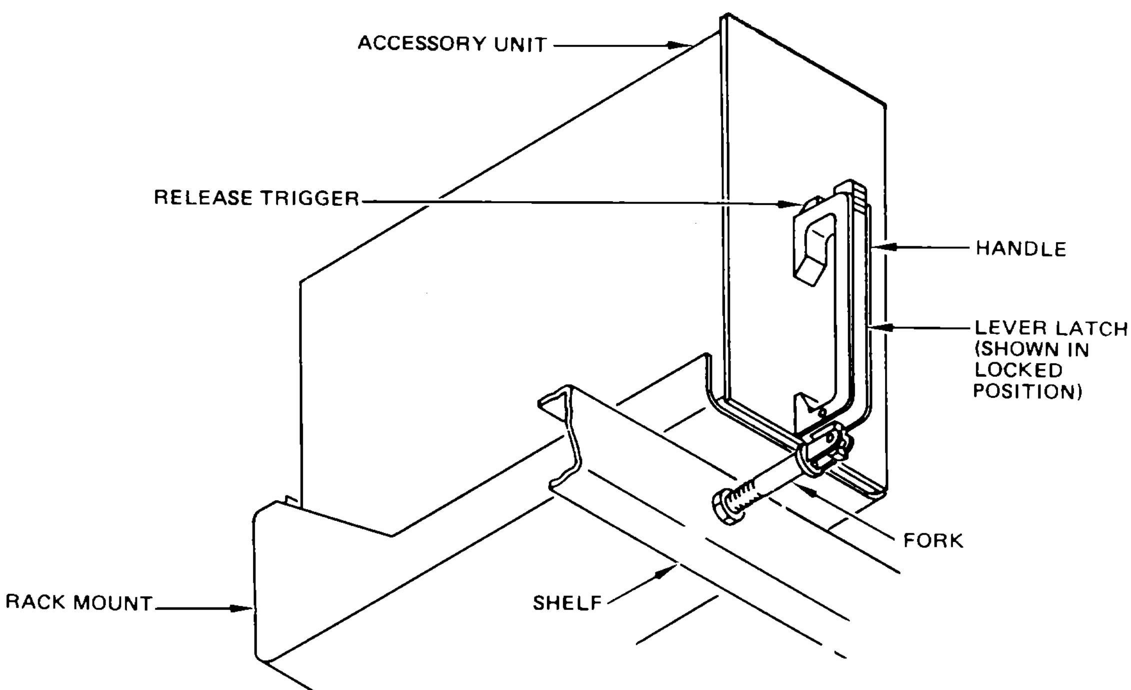
1. General
   A. The flight instrument accessory unit is installed on electronic shelf E1-2. The accessory unit is held in the rack mounting by a lever latch assembly. Correct adjustment of the lever latch assembly is necessary to assure proper electrical connector engagement.
2. Remove Accessory Unit
   A. Open the following circuit breakers on electronic load circuit breaker panel P18:
      (1) For flight instrument accessory unit M160:
         (a) As installed, HORIZON, INSTRUMENT XFMR 1 and 2, AUTOPILOT ROLL DC, and INSTRUMENT TRANSFER circuit breakers.
         (b) All circuit breakers for AIR DATA, FLIGHT DIRECTOR, VERTICAL GYRO and VOR/GS systems No. 1 and No. 2.
   B. Depress lever latch release trigger allowing lever to move away from handle.
   C. Move lever in an opening direction forcing unit away from rack electrical connector.
   D. Remove unit from rack mounting.
3. Install Accessory Unit
   A. Slide accessory unit into rack mounting with lever latch in open position until lever engages rack-mounted fork.

      NOTE: Remove any protective caps or bags from the unit or shelf prior to installation.

   B. Move lever latch to locked position (Fig. 401).
   C. Check electrical connector mating dimensions and lever latch fork adjustment per 20-10-111, Electrical/Electronic Black Box - MP.

      CAUTION: IMPROPER ADJUSTMENT OF ASSEMBLIES ON EQUIPMENT SHELVES MAY CAUSE CIRCUIT MALFUNCTION OR EQUIPMENT DAMAGE

4. Test Accessory Unit
   A. After installation, the following test checks for proper mating of the accessory unit electrical connectors. For a complete operational test of either accessory unit, perform the Flight Director System A/T per 34-26-0/501.
   B. Test Flight Instrument Accessory Unit M160
      (1) Close the circuit breakers opened in par. 2.A.(1) and circuit breakers for the Indicator Lights and Approach Progress Display (APD) Dim and TEST power.
      (2) Depress and hold the bank of captain's and first officer's A/P APD annunciator light caps. Check A/P APD annunciator lights are on bright.



TYPICAL LEVER LATCH TYPE RETENTION

Flight Instrument Accessory Unit Installation
Figure 401

(3) Position the MASTER DIM/TEST switch on the P2 panel to DIM while holding A/P APD light caps down. Check A/P APD annunciator lights dim.

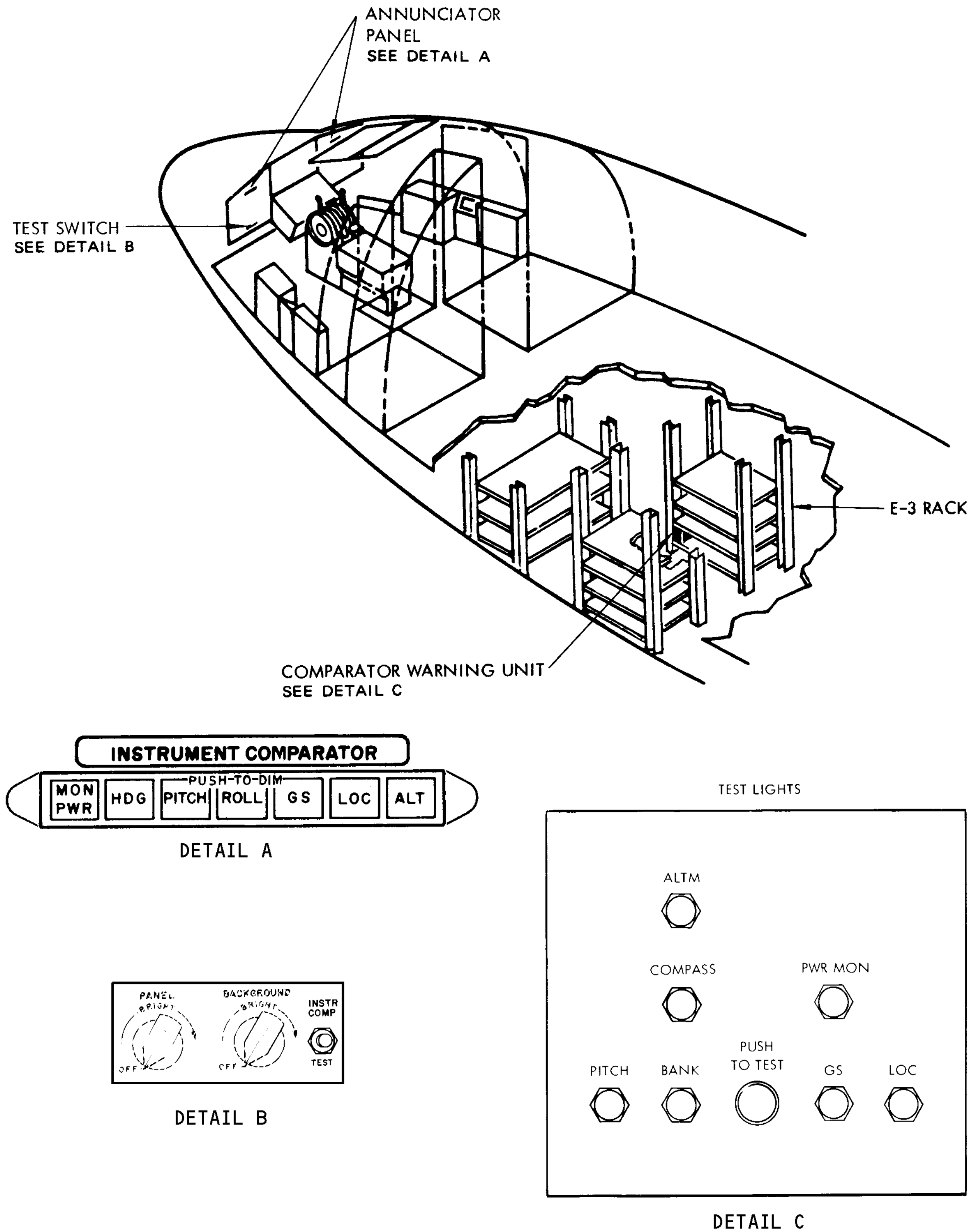
(4) Position the MASTER DIM/TEST switch to BRT. Check A/P APD annunciator lights return to bright.

(5) Release the A/P APD light caps.

(6) Remove the electrical power if it is no longer necessary.

NAVIGATION WARNING SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The navigation warning system provides a warning indication through actuation of warning light after a comparison has been made between output signals as fed from each of the following No. 1 and No. 2 systems: Compass, Attitude Reference, and VOR/GS Navigation. The signals compared are heading (compass), roll and pitch (attitude reference), and localizer deviation and glide slope deviation (VOR/GS navigation).
   B. The system consists of a comparator warning unit, plus two annunciator panels. The annunciator panels contain the warning lights which light to give a warning indication, and also serve to identify the system malfunctioning. AC power to the comparator warning unit is also monitored through warning lights on the annunciator panels. The lights labeled MON PWR light on loss of ac power to the unit. A test button is provided on the captain's instrument panel, and is used to check functioning of the various channels (heading, roll, pitch, localizer deviation, and glide slope deviation) in the system. Provision is made for the indication of disparities in radar altimeter indications. Location of the components is shown in figure 1.
2. Operation
   A. The system receives power when the COMPARATOR circuit breakers at the circuit breaker panel P6 are closed. Refer to figure 2. The heading, and the pitch and roll warning channels operate in a similar manner. Taking the heading warning channel as an example: Two differential resolvers are used for the comparison. The comparison signal is obtained from the resolver in the captain's CDI. The other resolver is in the first officer's CDI. From the transmitter portion of the captain's resolver, heading is transferred to the control transformer portion of the first officer's resolver. Here, a comparison is made between heading as sensed by both compass systems. Any variation in heading produces an output through the transmitter portion of the first officer's resolver to the comparator warning unit. This warning signal is fed to a rectifier and filter time delay circuit before being fed to the threshold detector. The filter and time delay prevent nuisance warnings during short duration errors. The threshold detector senses only signals exceeding a certain predetermined value before providing an output to activate the HDG warning lights. Roll and pitch signals are monitored in the same manner. The roll modifier circuit functions to desensitize the heading warning signal as a function of bank angle in order to prevent unnecessary actuation of the HDG lights. This is necessary due to differences in the dynamic response of the two compass systems when the airplane is banked.



Navigation Warning System Component Location
Figure 1

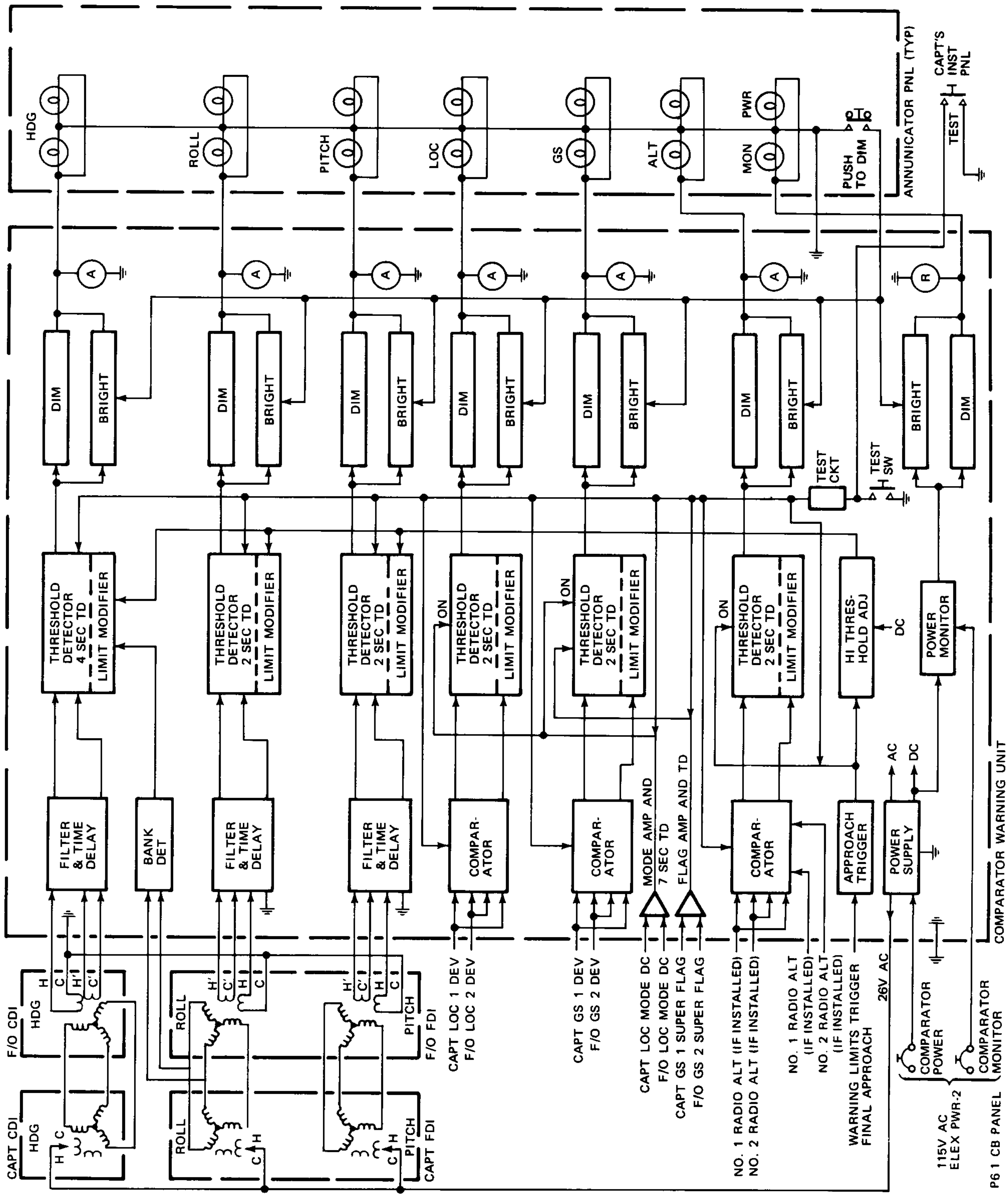
B. When glide slope capture is generated in the computer of the captain's flight director system, a trigger signal is applied which changes the threshold level in the heading, roll and pitch channels. When triggered, the threshold level is reduced in pitch and roll and in heading. The trigger signal, designated APPROACH TRIGGER in the diagram, also activates the altimeter channel when altimeter comparison is made to operate.

C. Localizer and glide slope deviation are compared separately in two comparators. The signals are modified and amplified before being fed to threshold detectors. The operations performed on the localizer and glide slope deviation signals are identical. If the output from either comparator (localizer, or glide slope) exceeds a certain predetermined value, the appropriate threshold detector will provide an output to the associated warning lights. In order to prevent operation of the comparators, other than when the deviation signals are present, they are activated by mode signals. When both navigation units are in the ILS mode, 28 volt dc is applied to both comparators (provided the nav transfer switch is in normal position). After a suitable time delay, the comparators become active. In the case of the glide slope comparator, in addition to the navigation units being in the ILS mode, the glide slope high level (Super) flag signals must be present. With the super flag signals present, the glide slope comparator will be activated. The glide slope super flag signals appear only when glide slope receivers are activated, and glide slope signals of a predetermined value are present to them.

D. The warning lights remain on for as long as a fault condition exists. The lights can be dimmed by pushing on them. The lights will remain dimmed until the fault is cleared, at which time they are automatically reset to their original (bright) condition.

E. The comparator warning unit and the lights may be tested by pressing the test button. When the button is depressed, all the warning lights (except MON PWR) will light, indicating that the comparator warning unit is functioning.

F. A switch and several test lights are provided on the comparator warning unit for system test purposes. (See figure 1.)



Navigation Warning System Schematic
Figure 2

NAVIGATION WARNING SYSTEM - ADJUSTMENT/TEST

1. Navigation Warning System Test
   A. General
      (1) Prior to performing test procedures for the navigation warning system, determine that the VOR/GS navigation, compass, and attitude reference (vertical gyro) systems are operational.
   B. Equipment and Materials
      (1) Ramp and Bench Test Set - Collins Type 972Q-4 or Instrument Flight Research Type NAV 401L (two required)
      (2) Calibrated Tilt Table (two required)
   C. Prepare to Test Navigation Warning System
      (1) Connect external electrical power and energize P6 circuit breaker panel.
      (2) Check that following circuit breakers are closed: captain's and first officer's instruments, VOR-1, VOR-2, GS-1, GS-2, instrument transfer, flight director, and comparator systems.
      (3) Turn on signal generators, allow to warm up and calibrate as indicated in signal generator instruction manual.
   D. Test Navigation Warning System
      (1) Test Power, Dim and Test
         (a) Place vertical gyro, compass, and VHF NAV transfer switches in NORMAL position.
         (b) Synchronize both compass systems and level vertical gyros using calibrated tilt table.
         (c) Check that captain's and first officer's instrument warning panel lights are out.
         (d) Open comparator power circuit breaker. Check that MON PWR light on captain's and first officer's instrument warning panels come on.
         (e) Push either captain's or first officer's instrument warning panel face. Check that instrument warning pane 1 lights dim.
         (f) Close comparator power circuit breaker. Check that MON PWR lights go out.
         (g) Press comparator test button on captain's instrument panel. Check that all lights except MON PWR lights on captain's and first officer's instrument warning panels come on.
      (2) Test Compass Warning
         (a) Place captain's flight director mode selector in VOR/LOC position. Rotate first officer's RMI synchronization knob until captain's CDI compass card reads 6 + 2 degrees higher than first officer's. Check that HDG lights come on in less than 10 seconds.
         (b) Repeat step (a) using captain's RMI synchronization knob and first officer's CDI.
         (c) Resynchronize both compass systems.

(d) Tilt vertical gyros 20 degrees in right roll. Repeat steps (a) through (c) using 10 + 2.5 degrees offsets in captain's and first officer's CDI headings.
(e) Repeat step (d) for 20 degrees left roll.
(f) Level vertical gyros. Place captain's flight director mode selector to GS MAN and repeat steps (a) through (c). Check that difference in captain's and first officer's CDI headings is 4.5 + 1 degrees.
(g) Place captain's flight director mode selector in VOR/LOC position.
(h) Repeat steps (f) and (g) using first officer's flight director mode selector.

(3) Test Pitch Warning
(a) Tilt No. 1 vertical gyro 2 degrees nose up and No. 2 vertical gyro 2.0 + 1.6 degrees nose down. Check that PITCH light on captain's and first officer's instrument warning panels come on.
(b) Tilt No. 1 vertical gyro 10 degrees nose up. Tilt No. 2 vertical gyro 6.0-± 1.6 degrees nose up. Check that PITCH light comes on.
(c) Tilt No. 2 vertical gyro 14 ± 1.6 degrees nose up. Check that PITCH lights comes on.
(d) Repeat steps (b) and (c) for nose down.
(e) Place captains flight director mode selector in GS MAN position. Keep No. 2 vertical gyro level and tilt No. 1 vertical gyro 3 ± 1 degrees nose up. Check that PITCH light comes on.
(f) Repeat step (e) for nose down.
(g) Repeat steps (e) and (f) with No. 1 vertical gyro level and tilting No. 2 vertical gyro.
(h) Place captain's flight director mode selector in VOR/LOC position.
(i) Repeat steps (e) through (h) using first officer's mode selector.

(4) Test Roll Warning
(a) Tilt No. 1 vertical gyro 2 degrees in right roll and No. 2 vertical gyro in 2.0 ± 1.6 degrees left roll. ROLL light on captains and first officer's instrument warning panels should come on.
(b) Tilt No. 1 vertical gyro 20 degrees right roll. Tilt No. 2 vertical gyro 16 ± 1.6 degrees right roll. ROLL- light should come on.
(c) Tilt No. 2 vertical gyro 24 ± 1.6 degrees right roll. ROLL light should come on.
(d) Repeat steps (b) and (c) for left roll.

(e) Place captain's flight director mode selector in GS MAN position. Keep No. 2 vertical gyro level and tilt No. 1 vertical gyro 3 ± 1 degrees right roll. ROLL light on captains and first officers instrument warning panels should come on.
(f) Repeat step (e) for left roll.
(g) Repeat steps (e) and (f) keeping No. 1 vertical gyro level and tilting No. 2 vertical gyro.
(h) Place captain's flight director mode selector on VOR/LOC position.
(i) Repeat steps (e) through (h) using first officer's flight director mode selector.

(5) Test LOC Warning
(a) Prepare the two signal generators by setting generators to different localizer frequencies, one for each NAV receiver.
(b) Set NAV-1 and NAV-2 on localizer frequencies. Captains and first officers CDI deviation bars should be centered. LOC light on captains and first officer' instrument warning panels should be out.
(c) Apply a localizer test signal to NAV-1 receiver such that captains CDI deviation bar indicates that airplane is 1/4 aft right. Apply localizer test signal to NAV-2 receiver such that first officer's CDI deviation bar is between 1/4 and 1/2 aft left. LOC light on captains and first officers instrument warning panels should come on.
(d) Place VHF NAV transfer switch in BOTH ON NAV-1 position. LOC light on captains and first officers instrument warning panels should go out.
(e) Place VHF NAV transfer switch in BOTH ON NAV-2 position. LOC light on captains and first officers instrument warning panels should be out.
(f) Repeat steps (c) through (e) interchanging NAV-1 and NAV-2 test signals and captains and first officers CDI indications.

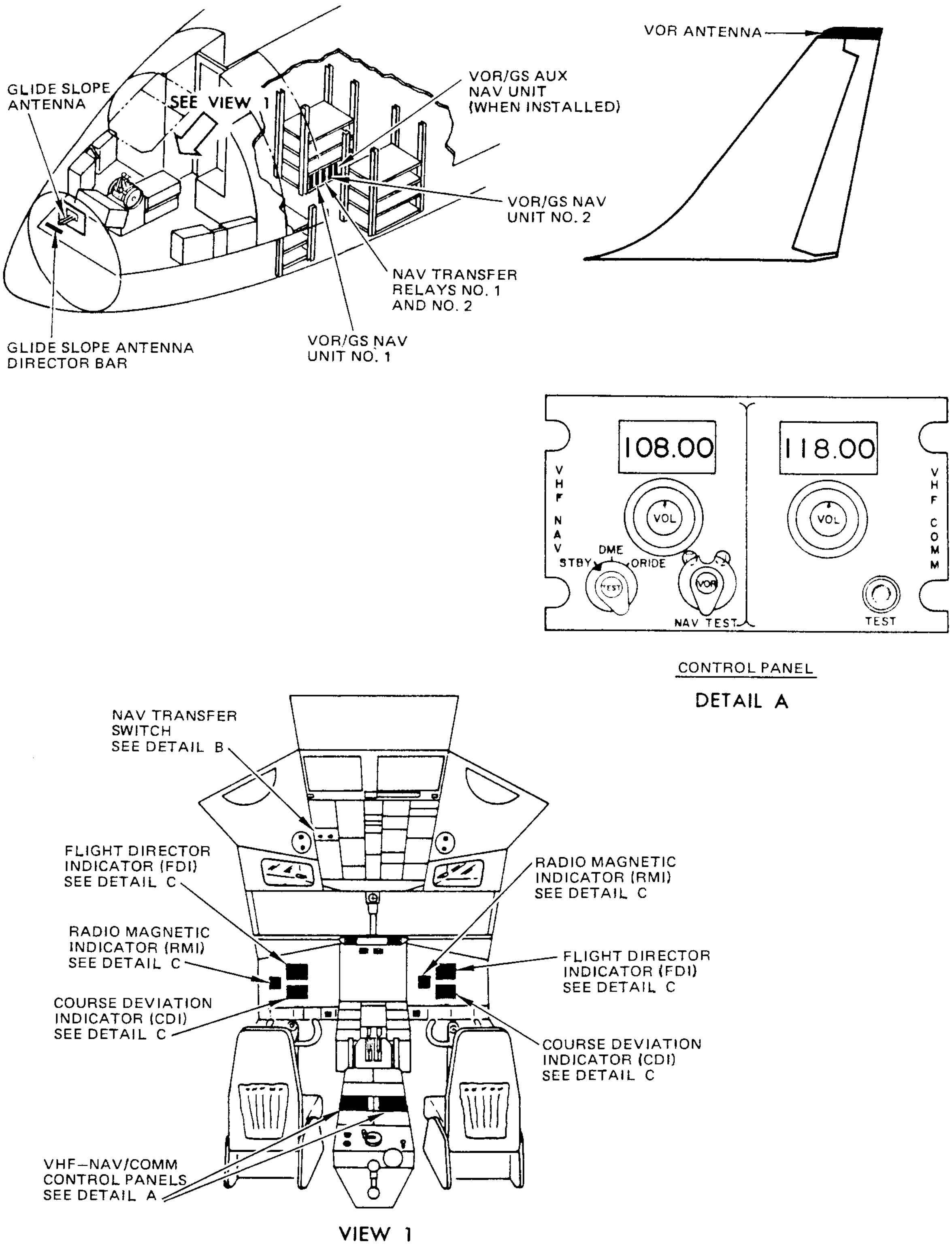
(6) Test Glide Slope (GS) Warning
(a) Place NAV transfer switch in NORMAL position. With NAV-1 and NAV-2 receivers still on localizer frequencies, apply a signal to receivers for a 2-aot fly-up indication on captain's and first officer's CDI'S. GS light on captains and first officers instrument warning panels should be out.
(b) Reduce fly-up signal to NAV-1 receiver to 1-dot. GS light on captain's and first officer's instrument warning panels should come on.
(c) Place VHF NAV transfer switch in BOTH ON NAV-1 position. GS light on captain's and first officer's instrument warning panels should go out.

(d) Place VHF NAV transfer switch in BOTH ON NAV-2 position. GS light on captain's and first officer's instrument warning panels should be out.
(e) Repeat steps (a) through (d) except reduce fly-up signal to l-dot for NAV-2 receiver instead of NAV-1 receiver in step (b).
(f) Repeat steps (a) through (e) for fly-down signal.

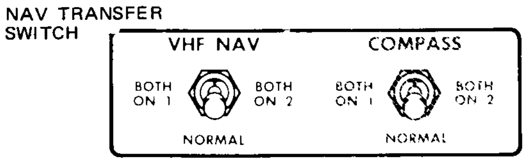
E. Restore Airplane to Normal Configuration
(1) Turn off signal generators and reinstall vertical gyros.
(2) Determine if there is further need for electrical power, if not, remove power.

VOR NAVIGATION SYSTEM - DESCRIPTION AND OPERATION

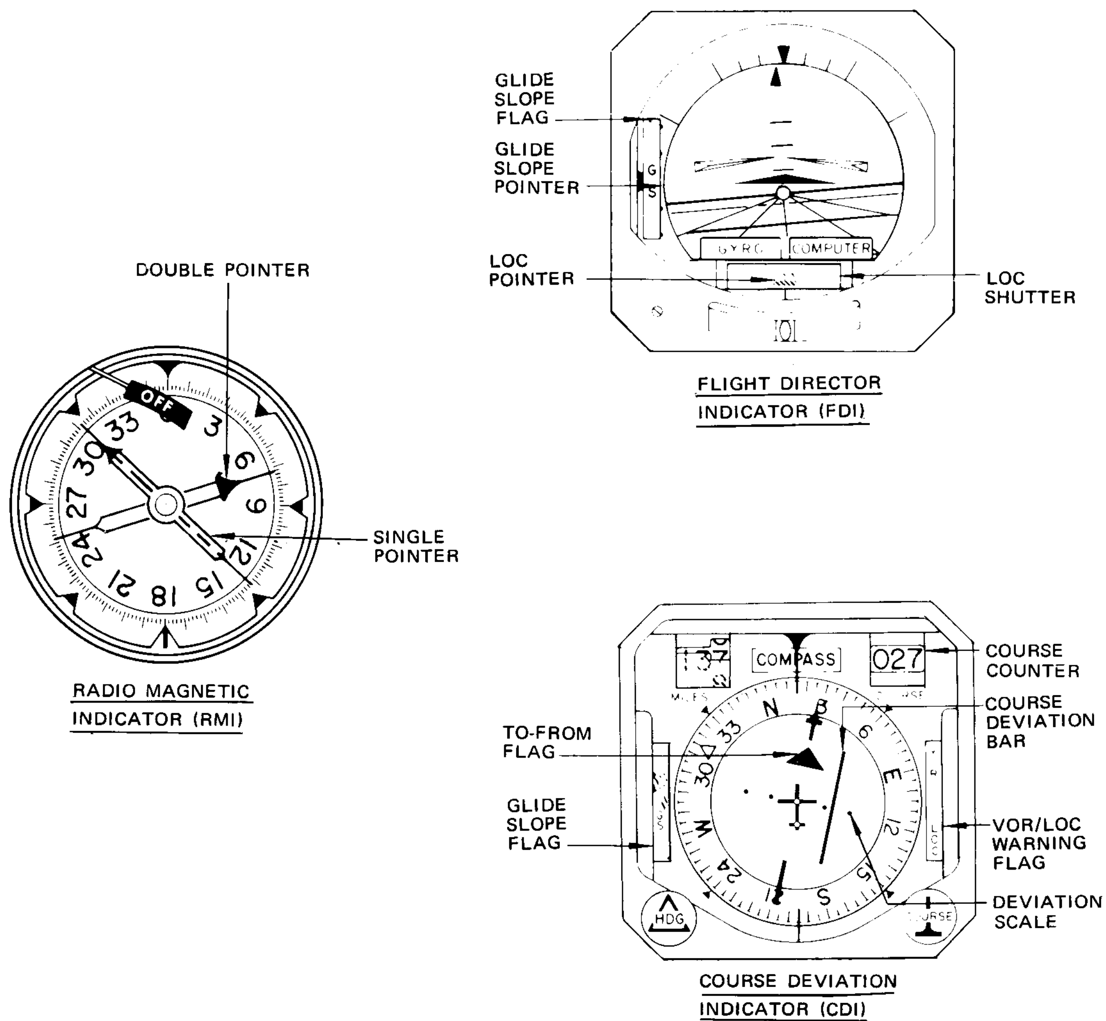
1. General
   A. The VHF omnidirectional range (VOR) and glide slope (GS) navigation systems provide information of airplane position with respect to, and deviation from a selected VOR course, or localizer (LOC) and glide slope approach beams. Information is also provided of airplane magnetic bearing to the VOR ground station under reception.
   B. Two systems, captain's and first officer's, are installed. Each is comprised of the following components: navigation unit (combined VOR/GS receiver and instrumentation unit), VOR antenna, GS antenna and control panel. Location of the components is shown in Fig. 1.
   C. VOR/LOC course deviation information is displayed by the course deviation bar in the course deviation indicator (CDI) and by the localizer pointer in the flight director indicator (FDI) or horizon deviation indicator (HDI). Glide slope (GS) information is displayed by the GS bar in the CDI and the GS pointer on the FDI. VOR station bearing information is displayed by the pointers in the radio magnetic deviation indicator (FMDI or RMI). VOR/LOC and GS information is also supplied to the flight director computers and to the autopilot. The CDI and FDI are described under AMM 34-26-0, Flight Director System, and the RMI under AMM 34-21-0, Compass System.
   D. Switching is provided to enable either system to feed both captain's and first officer's indicators, flight director computers and the autopilot. Switching is accomplished through the NAV transfer relays, and controlled through the VHF NAV transfer switch.
   E. The VOR/LOC and GS warning flags are provided to give an indication of their associated system malfunction. They come into view whenever power is lost to their systems, or whenever the input signals fail, or fall below a certain predetermined value. Warning indicators, to indicate VOR/LOC or GS system malfunctioning, are located on annunciator panels and are described in AMM 34-29-0, Navigation Warning System.
   F. Frequency selection, in a navigation unit, is controlled from the control panel. The GS portion of the unit becomes operative only when a localizer frequency is selected. Localizer frequencies extend from 108.10 to 111.90 MHz, with channels spaced on odd tenths of megahertz. VOR frequencies extend from 108.00 to 118.00 MHz except for above LOC frequencies. Collins 51RV-2B receivers with a 1 in the eighth digit of the part number (xxx-xxxx-1xx) have 40 ILS channels spaced at 50 kHz. All other receivers have 20 ILS channels spaced at 100 kHz.



VOR/GS Navigation System Component Location
Figure 1 (Sheet 1)



DETAIL B



DETAIL C

VOR/GS Navigation System Component Location
Figure 1 (Sheet 2)

G. In VOR operation, two signals (30 Hz REF and 30 Hz VAR) as radiated by a VOR ground station are fed to both the manual and automatic sections of a navigation unit. The 30 Hz REF signal is fed to both automatic and manual resolvers. The signal from each resolver is then phase compared against the 30-Hz VAR signal to produce the station bearing, VOR deviation and TO/FROM signals. In LOC operation, the deviation signal is produced by a comparison of the amplitude difference between two tone modulated signals (90 and 150 Hz) as radiated by a runway localizer transmitter. Portions of these signals are summed to produce the warning flag signals. The GS deviation and warning flag signals are produced in much the same manner as the LOC deviation and warning flag signals.

H. A VOR/ILS (LOC and GS) test circuit is provided in each VOR navigation unit to enable a check to be made of the VOR/ILS circuit function. Test switches are provided for this purpose on the front of each unit. Function of the test circuits is described under Operation.

2. Navigation Unit

A. Each navigation unit consists of a glide slope receiver, a VOR/LOC receiver, manual VOR/LOC and automatic VOR instrumentation units, a VHF/UHF synthesizer, VOR/LOC integrity monitors, and an interconnect unit, with a frame and cover. The navigation unit receives VOR, LOC and GS signals and supplies outputs to activate deviation, flag and to-from indicators. The glide slope receiver processes the VOR /LOC rf carrier and detects voice transmissions on the VOR/LOC carrier. The voice signals are fed to the interphone system and the VOR/LOC af signals are fed to the instrumentation units. The automatic VOR instrumentation unit compares the variable input with the reference signal and provides a synchro output to drive the VOR RMI pointer. The manual VOR/LOC instrumentation unit compares the localizer af signals received from the VOR/LOC receiver and provides an output for the LOC deviation pointers and VOR/LOC flag. During VOR operation, a comparator in the unit detects the phase difference between the variable and reference signals and provides an output for the deviation pointer, VOR/LOC flag and the to-from indicator.

3. VOR/LOC Antenna

A. The VOR/LOC antenna assembly is a fin cap type installed on top of the vertical stabilizer. The antenna consists of two balanced half loops with a hybrid coupler added for increased reliability under fault conditions. The hybrid coupler is composed of two quarter wavelength lines with the dual outputs to the receivers shunted by a 100-ohm resistor. In addition, a 18.5-inch tuning stub is connected to each half loop.

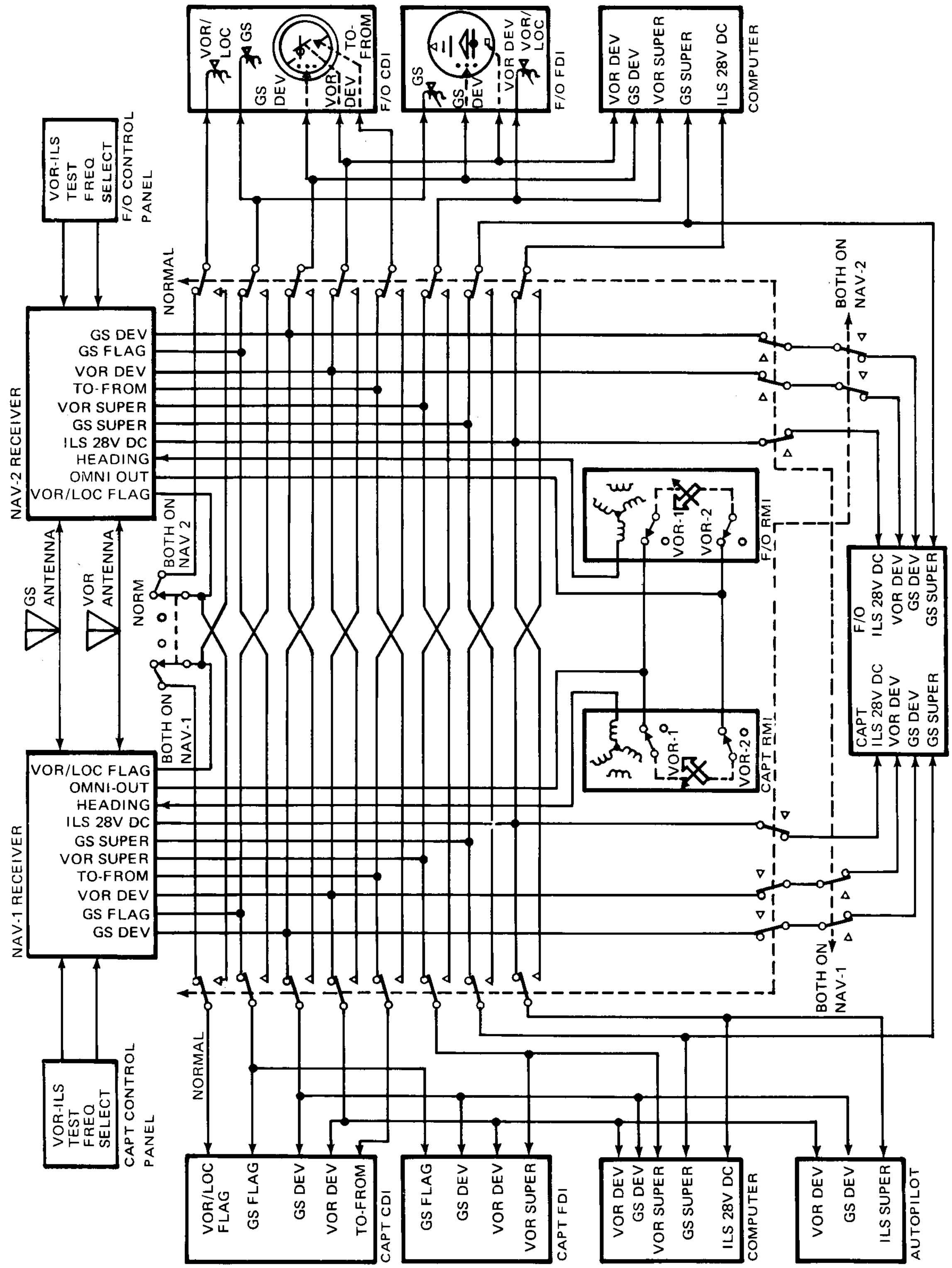
4. GS Antenna

A. The GS antenna is a single horizontally polarized unit, and is installed above the weather radar antenna in the nose radome.

5. Glide Slope Antenna Director Bar
   A. The glide slope (G/S) antenna director bar is a passive element of the glide slope antenna used to alter the G/S radiation patterns such that the navigation units have maximum glide slope sensitivity.
   B. The bar consists of a 13-inch strip of aluminum foil, pressure sensitive tape, installed horizontally inside the nose radome approximately 22 inches forward of the aft edge of the nose radome top centerline.
6. Control Panel
   A. The controls for each navigation unit are on separate control panels The controls on each panel consist of a volume control, two frequency selectors, and a NAV-TEST switch.
7. Operation
   A. Frequencies in the VOR/LOC bands are selected by rotating the frequency selectors on the control panel. The audio output of the signal is available through the interphone system. Volume is controlled through the volume control on the control panel.
   B. Taking one system as an example: position of the course deviation bar with respect to the miniature airplane, shows real airplane position with respect to the selected course. The course deviation dots are reference points for showing airplane angular displacement from the selected course. TO-FROM indication is shown by the broad arrow (flag). As the airplane passes over the selected VOR ground stations, the arrow reverses 180 degrees.
   C. Glide slope (GS) information is displayed by the GS pointers in the CDI and FDI. The pointers remain horizontal and move vertically across the GS scale to show airplane deviation about the glide path.
   D. Rotation of the course knob selects the desired VOR/LOC course, positions the course arrow and displays the selected course on the course counter. The manual resolver and course select synchro are positioned in accordance with the selected course. A signal from the resolver is fed to the manual VOR portion of the navigation unit, and compared in the phase detector. This signal deflects the course deviation bar.
   E. The high-level (super) and GS VOR/LOC warning flags signals are monitored by the autopilot and flight director computers.

F. The 51RV-2B navigation unit provides a full flag indication when over the VOR station. It also presents a positive indication that the VHF NAV signal is not usable. The later model 51RV-2B navigation unit is designed with flag delay circuitry to prevent a flickering flag when in the fringe reception area of the VOR/ILS station. Once a usable signal has been detected it must remain good for approximately 2 seconds while in VHF NAV mode, and 4 seconds for ILS mode before the flag is removed from view. When another enroute VOR station is selected, the monitoring system of the navigation unit is such that a flag will come into view until both the RMI bearing and course deviation bar are within ± 2 degrees of the correct bearing. The VHF/NAV course deviation is not usable until the RMI indication is correct and the VHF/NAV flag is biased out-of-view. The maximum time for flag in-view when switching stations is approximately 12 seconds. The navigation unit also incorporates a feature that "parks" the RMI needle to 90 degrees from zero when a localizer frequency is selected. The VHF NAV flag will also appear until the RMI needle has rotated from the 90-degree (needle parking) position to the bearing of the VOR station. When the navigation unit receiver is channeled from VHF NAV to an ILS frequency, the VOR/LOC and glide slope flags will appear and the RMI needle will again rotate to (needle parking) position. The flags will remain in-view for approximately 5 seconds after changing to ILS mode. This time delay is incorporated to eliminate nuisance flags, and allows the monitoring feature time to check that a valid signal is being received.

G. A system self-test feature is incorporated. A ILS-NAV TEST switch located on the VHF NAV control panel controls the self-test. When the switch is in center position, it can be pressed for VOR test. It controls a relay, solid-state switches and two test oscillators in the navigation unit. For VOR test, a VOR station is tuned in and a course of zero degrees is selected on the CDI. When the VOR test button is pressed, the course deviation bar in the CDI centers, the FROM flag becomes visible and the associated RMI pointers indicate a heading of 180 degrees. In the ILS test, the ILS-NAV TEST switch is turned to one of the two positions marked on the control panel, (up left and down right), in turn. Depending upon the position selected, the course deviation bar and the runway LOC symbol will deflect left or right, and the GS pointers will deflect up or down.



VOR/GS Navigation System Schematic
Figure 2

H. Operation of the navigation systems is shown in Fig. 2. RF signals from the VOR or GS antenna are fed to their respective nav receivers. Heading generated in the compass system is also sent from the RMIs to the nav receiver. Using these signal inputs, the receiver processes the signals used by the navigation system. With the navigation transfer switch in NORMAL position (as shown in Fig. 2), the signal output of nav receiver No. 1 controls the warning flags and GS and course deviation indicators on the captain's CDI and FDI, and the single pointer on the RMI. Signals from nav receiver No. 1 also go to the flight director computer, autopilot, and comparator. NAV receiver No. 2 and the first officer's CDI, FDI, and RMI operate in a similar manner. When the navigation transfer switch is in BOTH ON (NAV) 2 position, the captain's and the first officer's CDI, FDI, and RMI receive their inputs from nav receiver No. 2. When the nav transfer switch is in BOTH ON (NAV) 1 position all instruments indicate nav receiver No. 1 outputs.

VOR/ILS NAVIGATION SYSTEM - TROUBLESHOOTING

1. General
   A. The most effective method of troubleshooting a defective VOR/GS navigation system is to perform the tests described in the VOR/GS Navigation Systems - Adjustment/Test.
   B. First check for power to the defective system, and establish that power is available. Next, substitute units from the operating system where possible. After each change note whether the defective system becomes operative. If interchanging units does not clear up the trouble, system wiring should be checked for continuity and the connectors for security of connection.

VOR/GS NAVIGATION SYSTEMS - ADJUSTMENT/TEST

1. VOR/GS Navigation Systems Test
   A. Equipment and Materials
      (1) Signal Generator - Collins Type 479T-2, or equivalent
   B. Prepare to Test VOR/GS System
      (1) Provide electrical power.
      (2) Make sure that the circuit breakers for the following functions are closed:
         (a) VHF NAV 1 and 2
         (b) Instrument Transformers 1 and 2
         (c) Compass 1 and 2
         (d) Interphone System 1 and 2
         (e) Expanded Localizer 1 and 2
         (f) Flight Director 1 and 2
      (3) Turn on signal generator, allow to warm up, and calibrate as indicated in test set instructions: Signal Generator 479T-2.
   C. Test VOR/GS System
      (1) Test VOR/GS system frequency controls and audio output.
         (a) Place signal generator approximately 75 feet from VOR antenna. Set up signal generator on 108.0 MHz VOR frequency for maximum output signal modulated at 1000 Hz.
         (b) Place NAV transfer switch in NORMAL position. Turn both VOR/GS receivers on and tune captain's VHF NAV receiver to signal generator frequency. Make sure that 1000-Hz tone is available at all audio select panels when VHF NAV-1 selector switches are on.
         (c) Using captain's VHF NAV control, select VOR frequency that does not correspond to signal generator frequency. Make sure that 1000-Hz tone is not audible in interphone system.
         (d) Repeat steps (a) thru (c) using first officer's VOR/GS receiver.
      (2) Test VOR operation.
         (a) With NAV transfer switch in NORMAL position, adjust signal generator for maximum VOR output and 0 degree omniradial.

            NOTE: Compass system need not be operating, however, it must indicate a heading of 0 degree.

         (b) Make sure that narrow needle on both captain's and first officer's VOR RMI instruments indicate magnetic bearing of 180 ±3 degrees. Make sure that localizer and glide slope deviation indicators on captain's FDI is biased out of view. Make sure that glide slope pointer on captain's CDI is biased out of view.
         (c) Rotate course knob on captain's CDI until course deviation bar is centered. Make sure that course counter indicates 0.0 ±2 degrees. Make sure that rotating card (on which course deviation bar is mounted) is in position where course deviation bar is parallel to VOR RMI narrow needles.

(d) With compass card continuing to indicate airplane heading of 0 degree, make sure that course deviation bar is centered, course select cursor aligned with lubber line and FROM flag is visible on CDI.
(e) Rotate course selector knob on CDI until course deviation bar moves to second left dot. Make sure that course window reads 350 ±4 degrees.
(f) Rotate course selector knob until course deviation bar moves to second right dot. Make sure that course window indicates 10 ±4 degrees. Return course selector knob until course window reads 0 degree.
(g) With compass card indicating heading of 0 degree and course selector indicating 0 degree, adjust signal generator for 315-degree omniradial indication. Make sure that narrow needles in captain's and first officer's VOR-RMI instruments indicate 135 ±3 degrees and course deviation bar in captain's CDI deflects full-scale to right. Make sure that FROM flag is visible.
(h) Repeat step (g) using omnisignal of 45 degrees. Make sure that VOR needles indicate 225 ±3 degrees and course deviation bar deflects full-scale left. Make sure that FROM flag is visible.
(i) Adjust signal generator for a 0-degree omniradial. Make sure that needles in captain's and first officer's VOR-RMI instruments indicate 180 ±3 degrees out of phase with signal generator. Repeat for signal generator readings of 90, 135, 180, 225, and 270 degrees.
(j) Repeat steps (c) thru (i) using VOR system No. 2, first officer's CDI and FDI, and both captain's and first officer's VOR-RMI instruments.

NOTE: Utilize wide pointers in the VOR-RMI instruments while testing VOR system No. 2.

(k) Repeat step (k) with NAV transfer switch in BOTH ON 2 position.
(l) Repeat steps (a) thru (e) with NAV transfer switch in BOTH ON 1 position. Make sure that indications on captain's and first officer's CDI, FDI, and RMI read the same.

(3) Test Localizer Operation
(a) Set signal generator for maximum output to localizer frequency 108.1 MHz. Set LOC-GS switch for zero deviation (on course) signal.
(b) Set NAV transfer switch in NORMAL position. Tune captain's VHF NAV receiver to signal generator frequency.
(c) Rotate captain's CDI course selector knob to bring course select cursor in line with stationary lubber line at top. Place flight director mode selector in MAN GS position.

(d) Check that captain's CDI and FDI localizer pointers are centered. Check that captain's and first officer's RMI single needles position horizontally to right (relative east).
(e) Adjust localizer signal generator control for LEFT indication. Check that captain's CDI course deviation bar and FDI LOCALIZER pointer move left. Repeat for RIGHT indication. Check that course deviation bar and localizer pointer move right.
(f) Turn signal generator off. Check that warning flags come into view on captain's CDI and FDI.
(g) Repeat steps (a) thru (f) using VOR No. 2 system and first officer's CDI, FDI, and VOR No. 2 needles (wide needles) in the RMIs.
(h) Repeat steps (a) thru (f) with NAV transfer switch in BOTH ON 1 position.
(i) Repeat step (g) with NAV transfer switch in BOTH ON 2 position.

(4) Test Glide Slope
(a) Connect signal generator antenna to GS output terminals and place signal generator about 50 feet from airplane GS antenna. Set up generator on 334.7 MHz frequency and adjust for maximum ON COURSE output.
(b) Tune captain's VHF NAV receiver to 108.1-MHz localizer frequency. Place NAV transfer switch in NORMAL position. Check that GS scale on captain's CDI and FDI are in view.
(c) Set signal generator output control to GS FLY DOWN. Make sure that GS deviation indicators on captain's CDI and FDI are deflected downward.
(d) Set signal generator output control to GS FLY UP. Check that GS deviation indicators on captain's CDI and FDI are deflected upward.
(e) Turn off signal generator. Check that GS warning flags on captain's CDI and FDI come into view.
(f) Repeat steps (a) thru (e) using first officer's receiver, FDI, and CDI.
(g) Repeat steps (a) thru (e) with NAV transfer switch in BOTH ON 1 position.
(h) Repeat step (f) with NAV transfer switch in BOTH ON 2 position.
(i) Place NAV transfer in NORMAL position and turn signal generator off.

(5) VOR/ILS Self-Test

(a) With NAV transfer switch in NORMAL position, tune captain's VHF receiver to local VOR station. Set CDI course knob for 0 degree. Push VOR TEST switch on captain's VHF NAV control. Make sure that captain's CDI course deviation bar is centered. Make sure that TO-FROM indicator indicates FROM and RMIs indicate 180 degrees. Make sure that VOR LOC flag is in view for 9 seconds, disappears, and returns to view 20 seconds after initiating test.

(b) Select localizer frequency at captain's VHF NAV control panel. Push UP/LT TEST switch. Check that course deviation needle on captain's CDI and localizer pointer on FDI are deflected left. Check that GS needle on CDI and FDI are deflected upward. Check that GS and VOR/LOC flags are in view for 3 seconds, disappears, and returns to view 20 seconds after test initiation.

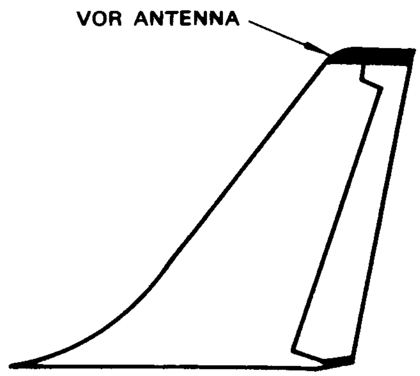
(c) On captain's VHF NAV control panel, push DN /RT TEST switch. Check that course deviation bar on captain's CDI and localizer pointer on FDI are deflected right. Check that GS pointers on CDI and FDI are deflected downward.

(d) Repeat steps (a) thru (c) using first officer's VHF NAV-2 system, CDI, and FDI.

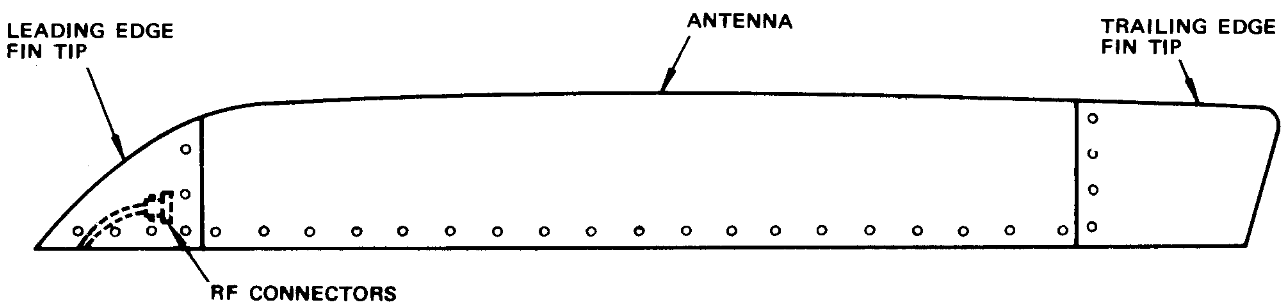
D. If no longer required, remove electrical power from airplane.

VOR ANTENNA - REMOVAL/INSTALLATION

1. Equipment and Materials
   A. Bonding Meter (Ref 20-22-01)
2. Remove Antenna
   A. Remove 12 leading edge fin tip mounting screws (Fig. 401).
   B. Remove leading edge fin tip.
   C. Remove two rf connectors from antenna.
   D. Remove 40 antenna mounting screws along base of antenna. Do not remove eight screws securing trailing edge fin tip to antenna.
   E. Remove antenna and trailing edge fin tip.
   F. Bag and stow rf connectors and mounting screws.
3. Install Antenna
   A. Clean faying surfaces of antenna and airplane structure to provide an electrical bond with a maximum resistance of 0.01 ohm.
   B. Apply a protective coating of corrosion preventive compound LPS-3 to mating surfaces of antenna and airplane structure per 51-21-91, Cleaning/Painting.
   C. Install trailing edge fin tip to antenna, if not already installed. Do not install leading edge fin tip.
   D. Place antenna and trailing edge fin tip in position on top of vertical fin.
   E. Connect rf cables to antenna.
   F. Install leading edge fin tip to antenna assembly using 12 mounting screws.
   G. Install base mounting screws and tighten.
   H. Check electrical bond between antenna and airplane structure per 20-22-01. Resistance should not exceed 0.01 ohm.
   I. Test VOR antenna for satisfactory operation (Ref 34-31-0 A/T).



VERTICAL FIN



VOR ANTENNA ASSEMBLY

VOR Antenna Installation
Figure 401

GLIDE SLOPE ANTENNA - REMOVAL/INSTALLATION

1. Equipment and Materials
   A. Bonding Meter (AMM 20-22-01)
2. Remove Glide Slope Antenna
   A. Open radome.
      (1) Remove screws from edge of radome.
      (2) Insert fingers at bottom aft end of radome and rotate bottom edge forward and up.

      CAUTION: DO NOT PRY RADOME OPEN.

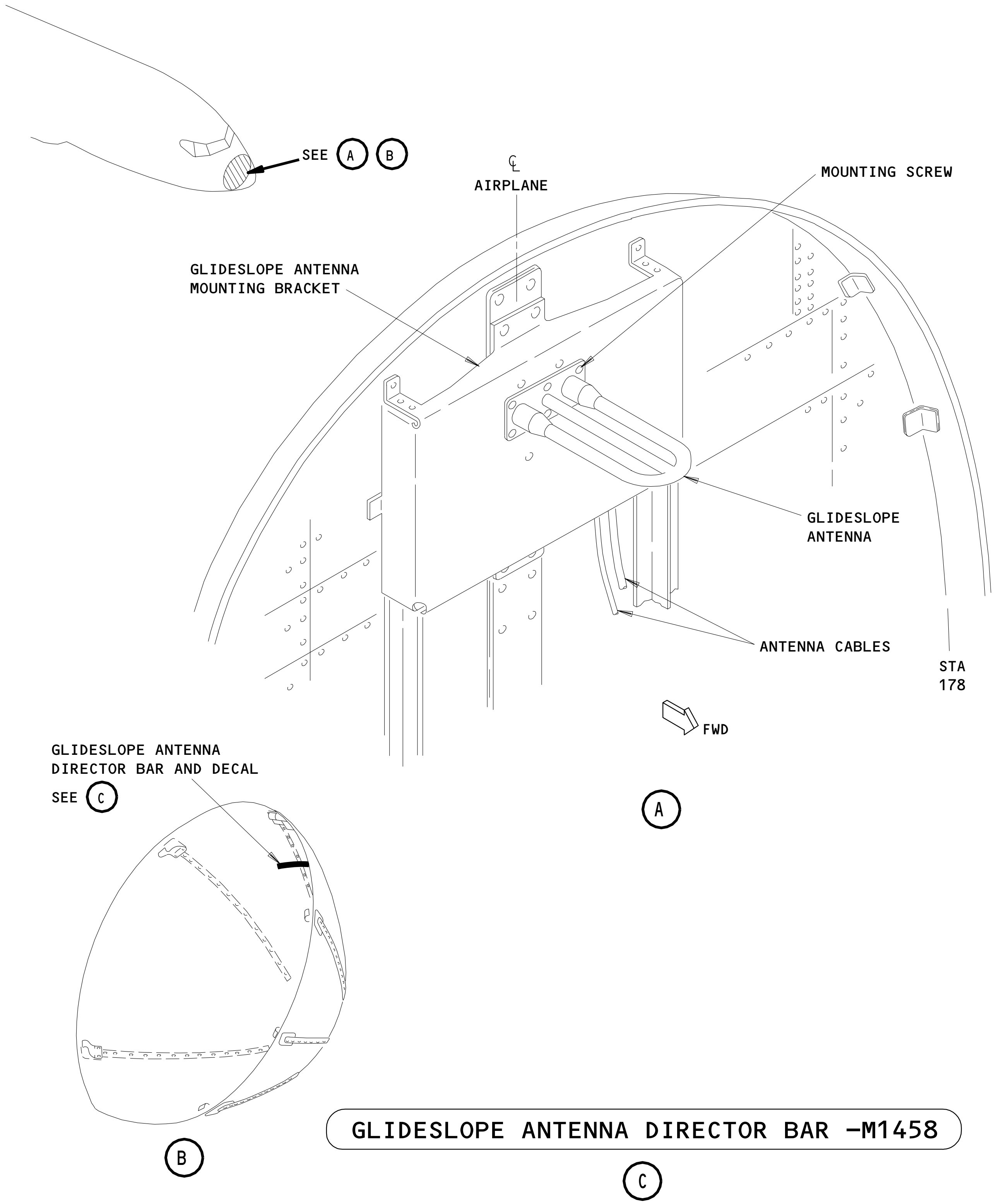
      (3) Install support when radome is open.

      NOTE: Before removing glide slope antenna, inspect the glide slope antenna director bar. It is a 13-inch continuous strip of aluminum foil tape, positioned horizontally across the centerline on the inside surface of the nose radome, approximately 22 inches from the radome upper aft edge.

      If director bar is undamaged and still in the 13-inch continuous configuration, proceed with glide slope antenna removal. Replace the tape if is has been cut, damaged or losing adhesion per instruction in AMM 53-52-0.

   B. Remove screws holding glide slope antenna to support plate (Fig. 401).
   C. Pull antenna assembly forward far enough to gain access to antenna connectors.
   D. Disconnect antenna connectors and remove antenna assembly.
3. Install Glide Slope Antenna
   A. Clean faying surfaces of antenna and airplane structure to provide an electrical bond with a maximum resistance of 0.01 ohm.
   B. Hold antenna adjacent to mounting plate and connect antenna cables.
   C. Set antenna in place on mounting plate and install mounting screws (Fig. 401).

D. Make sure that antenna cables are not kinked.
E. Make sure that electrical bond between antenna and airplane structure per AMM 20-22-01. Resistance should not exceed 0.01 ohm.
F. Close radome and reinsert mounting screws.
G. Test glide slope antenna for satisfactory operation (AMM 34-31-0/501).



Glide Slope Antenna Installation
Figure 401

VHF NAV CONTROL PANEL - REMOVAL/INSTALLATION

1. General
   A. The VOR/ILS control panels are located on aft electronics panel P8. Each panel is held in place by four fasteners.
2. Remove Control Panel
   A. Open circuit breakers on electronic load control center circuit breaker panels P6 and P18 as follows:

Control Panel	Circuit Breaker/Panel	
	P6 Panel	P18 Panel
NO. 1	ELECTRONICS PANEL LIGHTS	G/S-1 VOR/LOC-1
NO. 2	ELECTRONICS PANEL LIGHTS G/S-2 VOR/LOC-2	

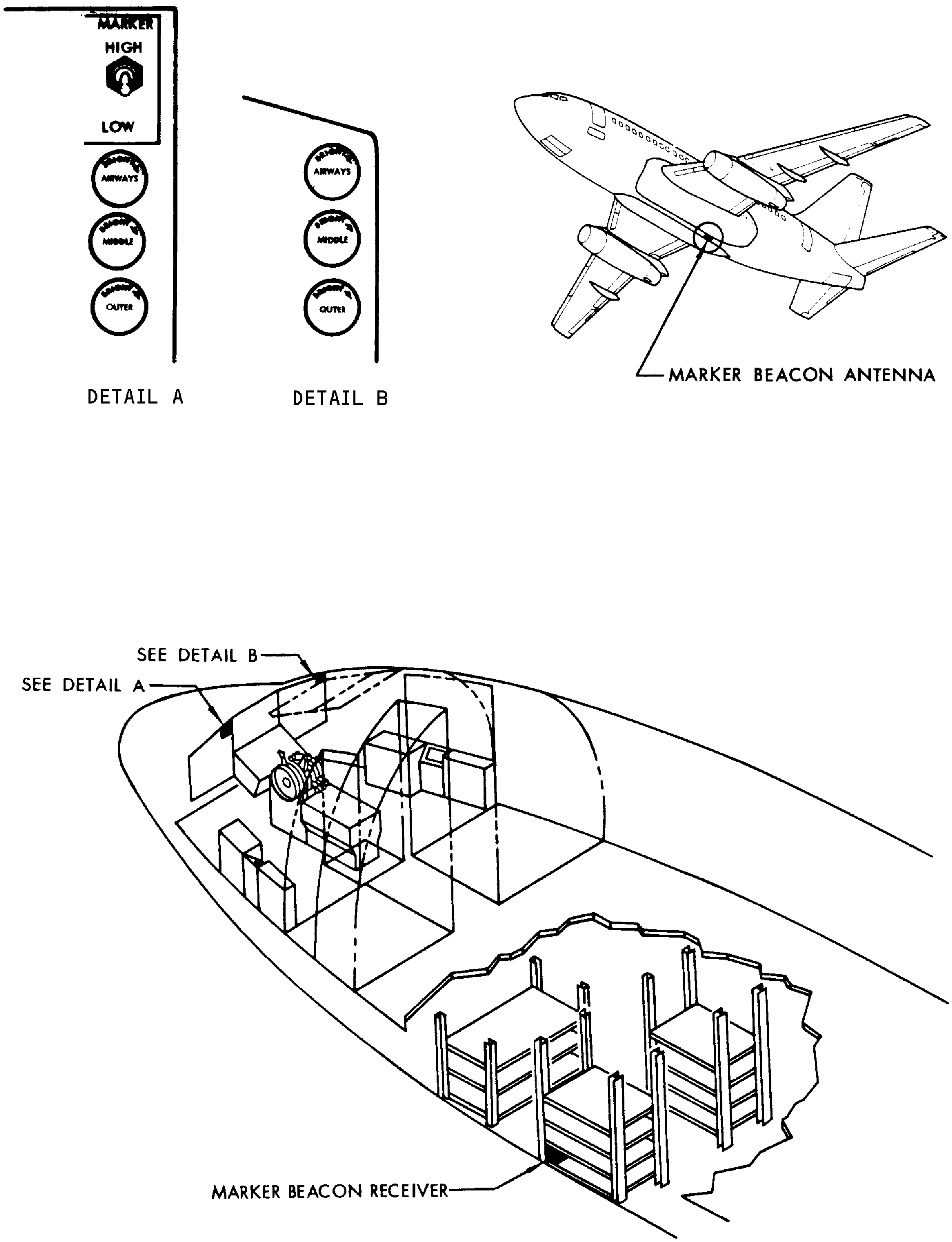
   B. Loosen fasteners securing panel to mounting.
   C. Withdraw panel from mounting and disconnect electrical connector(s).
3. Install Control Panel
   A. Connect electrical connector(s) to control panel.
   B. Secure control panel to mounting.
   C. Provide electrical power.
   D. Close circuit breaker opened in par. 2.A. above and make sure that panel lights are on.

      NOTE: The flight deck panel lights control must be on bright.

   E. Remove electrical power if no longer required.

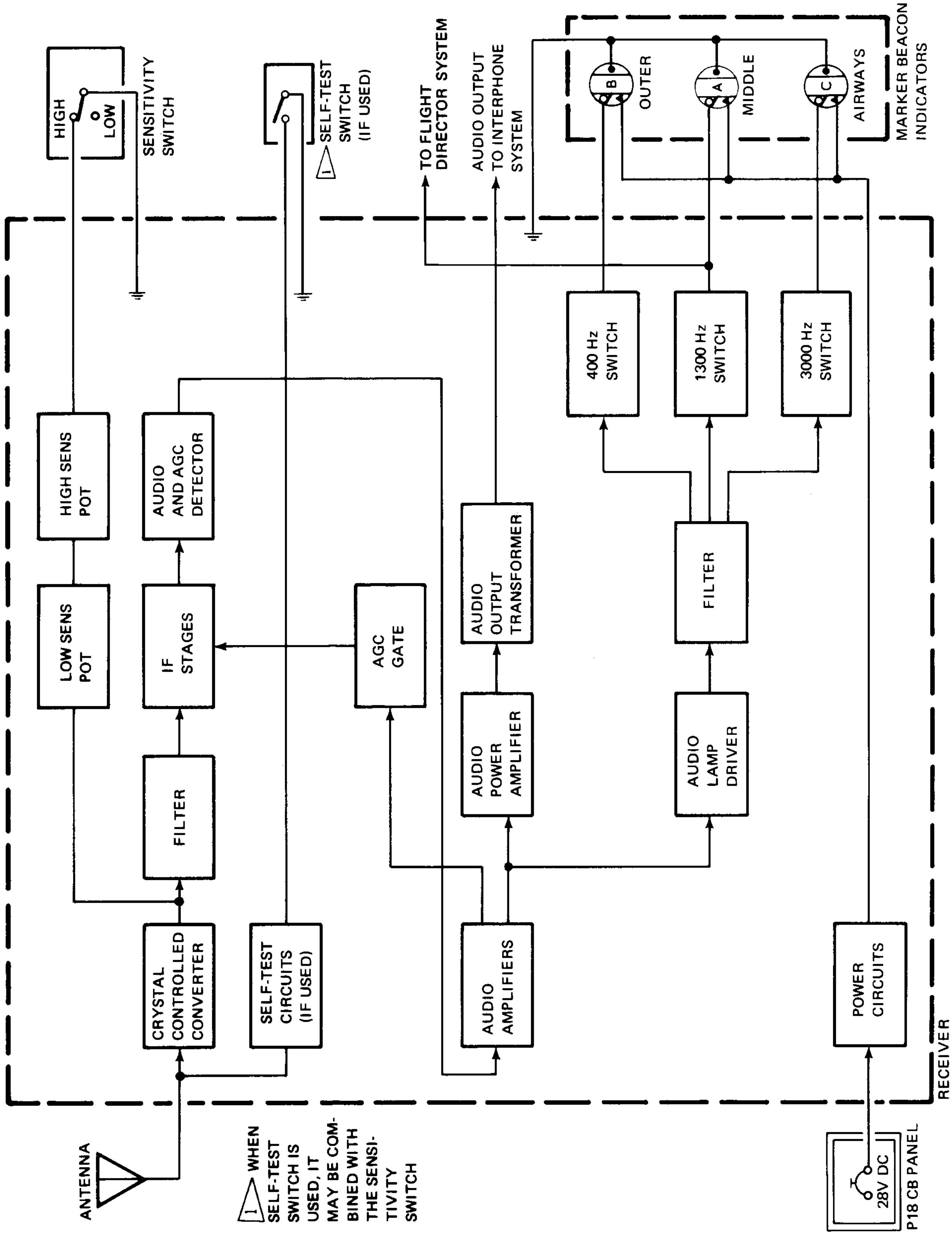
MARKER BEACON SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The marker beacon system provides the pilots with both visual and aural indication of airplane passage over airways fen markers, station location Z markers and instrument landing system (ILS) middle and outer markers.
   B. The marker beacon system consists of a receiver, an antenna, two sets of three indicator lights and a sensitivity switch. (See figure 1.)
   C. The ground marker beacon transmitters all operate on a fixed frequency of 75 MHz and are modulated by one of four audio signals, depending on the type of marker beacon. The signals, received by the airborne receiver as the airplane passes over the transmitter, are made to light indicator lights. The audio signal is also made available to the crew by actuation of the MKR control on the audio selector panels. Following are the four types of marker beacons with their tone frequency and indicator light color.
      (1) ILS Outer Marker - An intermittent 400-Hz tone with the blue light flashing.
      (2) ILS Middle Marker - An intermittent 1300-Hz tone with the amber light flashing.
      (3) Airways Fan Marker - An intermittent 3000-Hz tone with the clear light flashing.
      (4) Station Location Z Marker - A steady 3000-Hz tone with the clear light steadily lighted.
   D. When an ILS middle marker signal is received, this information is fed to the flight director system for the low approach mode. (Refer to 34-26-0, Flight Director System.)
2. Receiver
   A. The receiver is a single conversion crystal-controlled superheterodyne operating at a fixed frequency of 75 MHz Sensitivity of the receiver is variable and controlled by a switch located an the captain's instrument panel. Audio filters tuned to 400, 1300, and 3000 Hz are used for separation of the three types of tone modulation used by the beacons. The receiver, which contains an integral power supply, is installed on the electronic equipment rack (E-2).
3. Antenna
   A. The antenna is a blade-type unit located on the bottom centerline of the airplane at body station 620. The antenna is connected to the receiver by a coaxial cable.



Marker Beacon System Component Location
Figure 1

4. Indicator Lights
   A. There are two sets of three indicator lights, one set is on the captain's instrument panel, and the other is on the first officer's instrument panel. The lights are covered by white, blue and amber lens and are the push-to-test type
5. Sensitivity Control
   A. The sensitivity control is installed on the captain's instrument panel. The LOW and HIGH positions of the control indicate the degree of receiver sensitivity and are normally used when the airplane is flying at low and high altitudes, respectively.
6. Operation (Fig. 2)
   A. The marker beacon receiver is fixed-tuned to receive 75-MHz amplitude modulated signals. The 75-NHz incoming signal from the antenna is applied through a selective LC filter to a crystal controlled converter where it is mixed with a 70.4-MHz local oscillator signal to generate a 4.6-
      MHz intermediate frequency (IF) signal. This difference or IF frequency is selected by a 4.6-NHz IF filter and applied through three stages of IF amplification.
   B. The audio component (400, 1300, or 3000 Hz) of the 4.6-MHz IF signal is detected in the audio and AGC detector and amplified by the first audio amplifier.
   C. Two audio outputs are provided by the first audio amplifier. The AGC audio output is rectified through the AGC gate and applied to all three IF amplifier stages where it provides automatic gain control of the incoming signal. The second audio output drives the second audio amplifier.
   D. The second audio amplifier output divides and drives the audio power amplifier and the audio light drives. The output signal from the audio ,power amplifier is fed through an output transformer to the flight interphone system for aural monitoring of the marker beacon system. Audio output from the audio light driver is fed through a three-stage frequency selective filter which separates the audio into its 400, 1300 or 3000 Hz component and drives the respective 400, 1300, or 3000 Hz light switches which turn on the corresponding marker beacon lights.
   E. The self-test circuit supplies a 75-MHz amplitude modulated test signal to the input of the crystal controlled converter stage when the push-to-test (PTT) button on the front of the receiver is pressed. The test signal is generated by a switched three-frequency (400, 1300, and 3000 Hz) multivibrator which sequentially modulates a 75-MHz test signal oscillator.



Marker Beacon System Block Diagram
Figure 2

MARKER BEACON SYSTEM - TROUBLESHOOTING

1. General
   A. The following troubleshooting procedures are based on performance of the system operational test and are presented in a tree-type format to aid in rapid fault isolation.
   B. When a test step does not check out, find the box containing the trouble symptom and perform the stated action. Continue to follow a single line by analyzing the results of each test step until the required corrective action is determined. Perform the specified corrective action, then repeat the step at which the failure was encountered and complete the test to check out the system.
   C. All troubleshooting procedures are based on the assumption that wiring is OK and that electrical power is available. If the corrective action in the procedure does not correct the problem, check wiring using the wiring diagram.
2. Prepare for Troubleshooting
   A. Make sure that marker beacon and flight interphone circuit breaker on P18 circuit breaker panel are closed.
   B. Set sensitivity control to HIGH (or HI) position.
   C. Activate the marker beacon control on the audio selecting panel.
   D. Apply electrical power.

Press each of captain's and first officer's marker beacon indicator lights. Check that all indicator lights come on when pressed. IF -

MARKER BEACON INDICATOR LIGHT(S) DOES NOT COME ON - Replace defective light bulb(s) and repeat test.

ALL MARKER BEACON INDICATORS DO NOT COME ON - Replace marker beacon receiver

MARKER BEACON LIGHTS COME ON WHEN PRESSED - Press and hold TEST switch on front panel of marker beacon receiver for 5 seconds. Check that blue, amber, clear indicator lights come on in sequence (approximately 2 seconds) and that 400Hz, 1300Hz, 3000Hz tones are audible over interphone, respectively. IF -

MARKER BEACON INDICATOR LIGHTS COME ON AND TONES AUDIBLE BUT FAILED DURING FLIGHT - Remove marker beacon antenna, check antenna cable for opened or shorted center conductor, cabling crimping and improper connector installation. IF -

MARKER BEACON LIGHT(S) DOES NOT COME ON AND/OR TONE(S) NOT AUDIBLE - Replace marker beacon receiver

ANTENNA CABLE FAULTY - Repair or replace cabling.

ANTENNA CABLE SATISFACTORY - Replace marker beacon antenna

Marker Beacon System
Figure 101

MARKER BEACON SYSTEM - ADJUSTMENT/TEST

1. Marker Beacon System Test
   A. Equipment and Materials
      (1) Test Oscillator BC-376 or equivalent.
   B. Test Marker Beacon System
      (1) Supply external power to the airplane and energize load control centers P18 and P6.
      (2) With the marker beacon and interphone circuit breakers on load control centers P18 and P6 closed, allow 10 minutes for receiver to warm up. Energize interphone system.
      (3) Push-to-test marker beacon indicator lights.
      (4) Place test oscillator on ground, approximately 15 feet away from marker beacon antenna. Extend antenna on test oscillator to its fullest extent, and place parallel to fore and aft axis of the airplane. Turn test oscillator on and allow to warm up.
      (5) Set marker sensitivity control switch to HIGH. With carrier frequency set at 75.00 Mc, turn oscillator ON and set modulation to 400 cps. Blue lights on captains and first officers instrument panels should light.
      (6) Connect headphone to interphone system. The 400 cps tone should be heard.
      (7) Repeat steps (5) and (6) with modulation set at 1300 cps and 3000 cps, amber and white lights should come on respectively, and tones should be audible.
      (8) Repeat (5), (6) and (7) with marker sensitivity switch set to LOW. (Move test oscillator closer to marker antenna, if necessary.)
      (9) Ascertain whether there is any further need for external power, if not, remove external power.

MARKER BEACON ANTENNA - REMOVAL/INSTALLATION

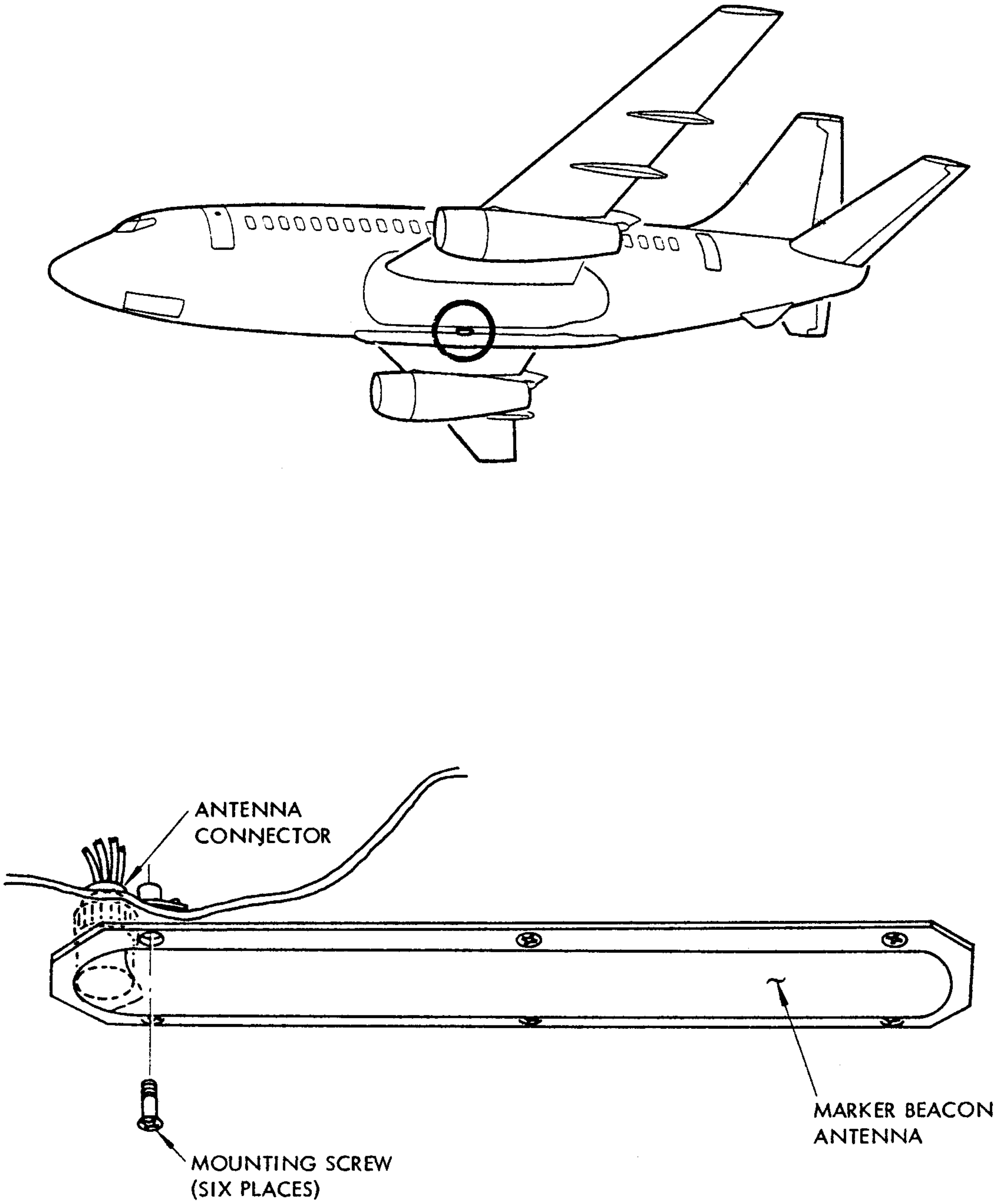
1. Equipment and Materials
   A. Bonding Meter (Ref 20-22-01)
2. Remove Marker Beacon Antenna
   A. Remove screws around outer edge of antenna (Fig. 401).
   B. Lower antenna from mounting recess until antenna cable can be disconnected.

      CAUTION: DO NOT PULL ANTENNA CABLE.

   C. Disconnect antenna cable from antenna connector.
   D. Remove antenna.
3. Install Marker Beacon Antenna (Fig. 401)
   A. Check antenna, mounting surface and airplane faying surfaces for corrosion or other foreign material and clean if necessary.

      CAUTION: BE CERTAIN FAYING SURFACES ARE CLEAN. INADEQUATE RF GROUNDING WILL CAUSE IMPROPER SYSTEM OPERATION.

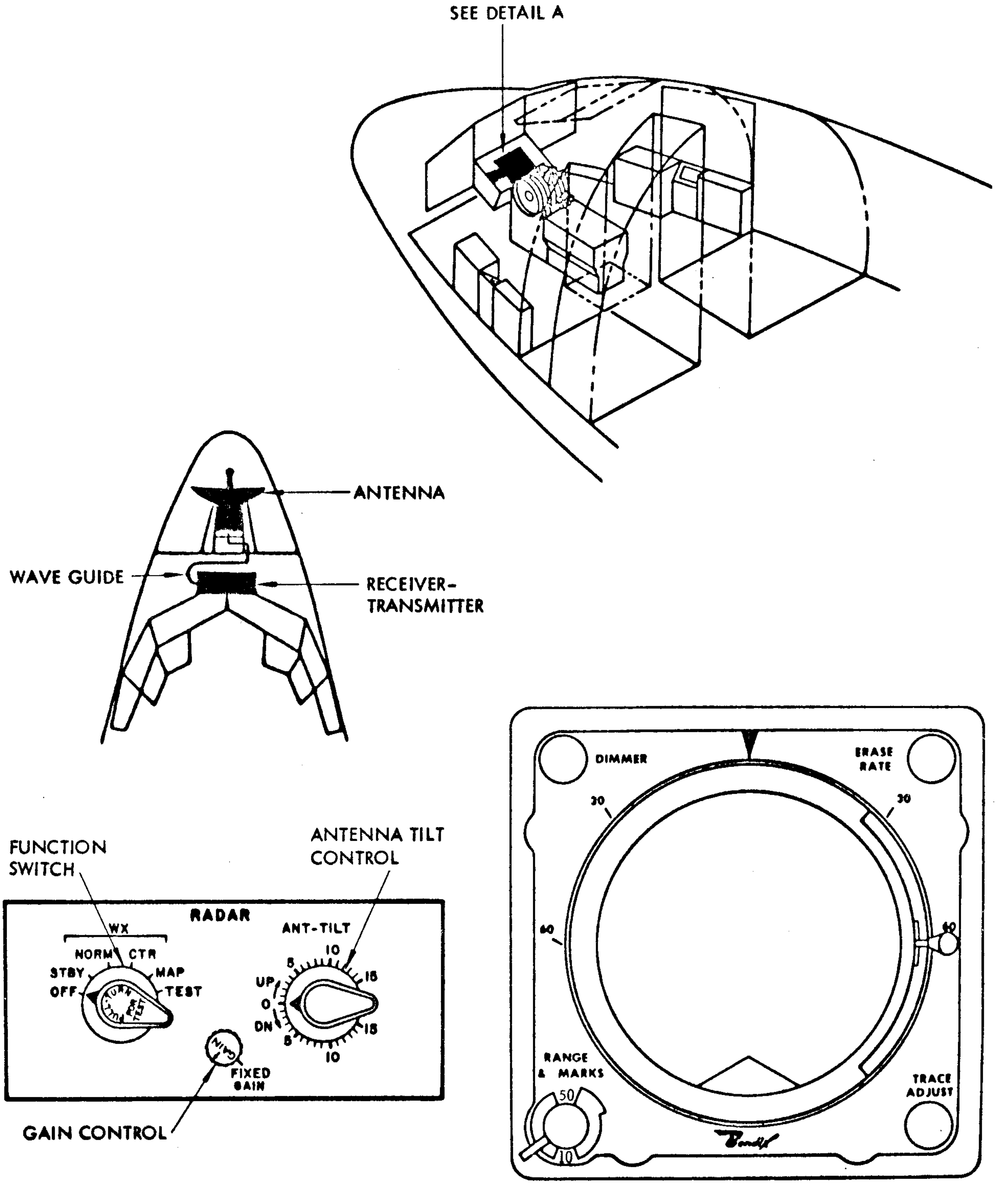
   B. Apply corrosion preventative material to antenna, mounting plate and structure faying surfaces (Ref 51-21-91 CP).
   C. Connect antenna cable to antenna connector.
   D. Position antenna in recess and install mounting screws.
   E. Do a check of the electrical bond:
      (1) Remove one antenna mounting screw.
      (2) Position the bonding meter between the mounting screw hole and the airplane skin.
      (3) Check the bond (Ref 20-22-01).
      (4) Make sure the resistance does not exceed 0.1 ohm.
   F. Install the antenna mounting screw.
   G. Perform System Adjustment/Test (Ref 34-35-0).



Marker Beacon Antenna
Figure 401

WEATHER RADAR SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The weather radar (Bendix-X Band) system presents the pilots with an accurate and continuous picture of weather conditions ahead of the aircraft. This presentation of weather conditions in terms of range and azimuth enables course changes to be made to avoid turbulent areas. In addition to weather mapping, the system provides for ground mapping, terrain avoidance and self-test.
   B. Weather mapping is based on the fact that water particles present in the air as rainfall will reflect a radar beam in direct relation to the density or concentration of the moisture. During the normal mode of operation, the receiver translates echo returns from these moisture concentrations into a picture on the radar indicator where they appear as bright and lighted areas. During iso-echo contour mode of operation, the areas of heaviest rainfall usually associated with storm centers are shown as black spots in the lighted areas on the indicator.
   C. The weather radar system consists of a receiver-transmitter (R/T), an indicator, a control panel, waveguide, and an antenna (Fig. 1). The system has an operating range of 150 or 180 nautical miles depending on the indicator installed. Power for the system is supplied through circuit breakers on circuit breaker panel P18.
2. Controls
   A. Most of the operating controls for the system are on a control panel and the radar indicator. These units are on the forward electronic control panel (Fig. 1). The control panel has a six-position function switch OFF, STBY, NORM, CTR, MAP AND TEST, a gain control, stabilization switch (if installed) and an antenna tilt control. Controls on the indicator include a range select switch, dimmer control, erase rate control, and trace adjust control.
   B. Other system controls include elevation and azimuth power switches on the antenna, and a meter selector switch and REL-LOCK switch on the R/T.
3. Receiver-Transmitter
   A. The receiver-transmitter supplies the pulses of microwave energy that are transmitted by the antenna. The power output pulses from the magnetron oscillator arc of 2.5 microseconds duration and have a pulse repetition rate of 400 ±20 Hz. Peak output power of the transmitter is 50 kilowatts at a frequency of 9375 ±30 MHz in X-band. Certain self-test functions are also generated in the receiver-transmitter.
   B. The unit consists of synchronizing, video, iso-contour, IF, sensitivity time control, stabilization, and test circuitry for the radar system. In addition, it provides filament and plate power for the indicator as well as the preamplifier.
   C. The receiver-transmitter is mounted in a shockmount in the pressurized area just aft of the nose radome at body station 195. It is connected to the antenna by a waveguide.



DETAIL A

Weather Radar System Component Location
Figure 1

D. Standby and normal operation is described as follows (Fig. 2 and 3). When the function switch is placed in STBY position, the 28 volts dc from the function switch energizes the standby relay. This applies 115 volts ac to the 3-minute time delay and the indicator.

(1) For R/T without the filament relay, 115 volts ac is applied to the filament transformer, -250-volt dc power supply and blower motor. The blower motor operates at a lower than normal speed to allow faster heating of the magnetron. After 3 minutes, a ground is applied to one side of the power supply relay. When the function switch is placed in NORM, CTR, or MAP position, the power supply relay will be energized. The 3-minute time delay cannot be bypassed by initially placing the function switch in NORM position. After 3 minutes, 28 volts dc is applied to energize the filament relay (if installed).

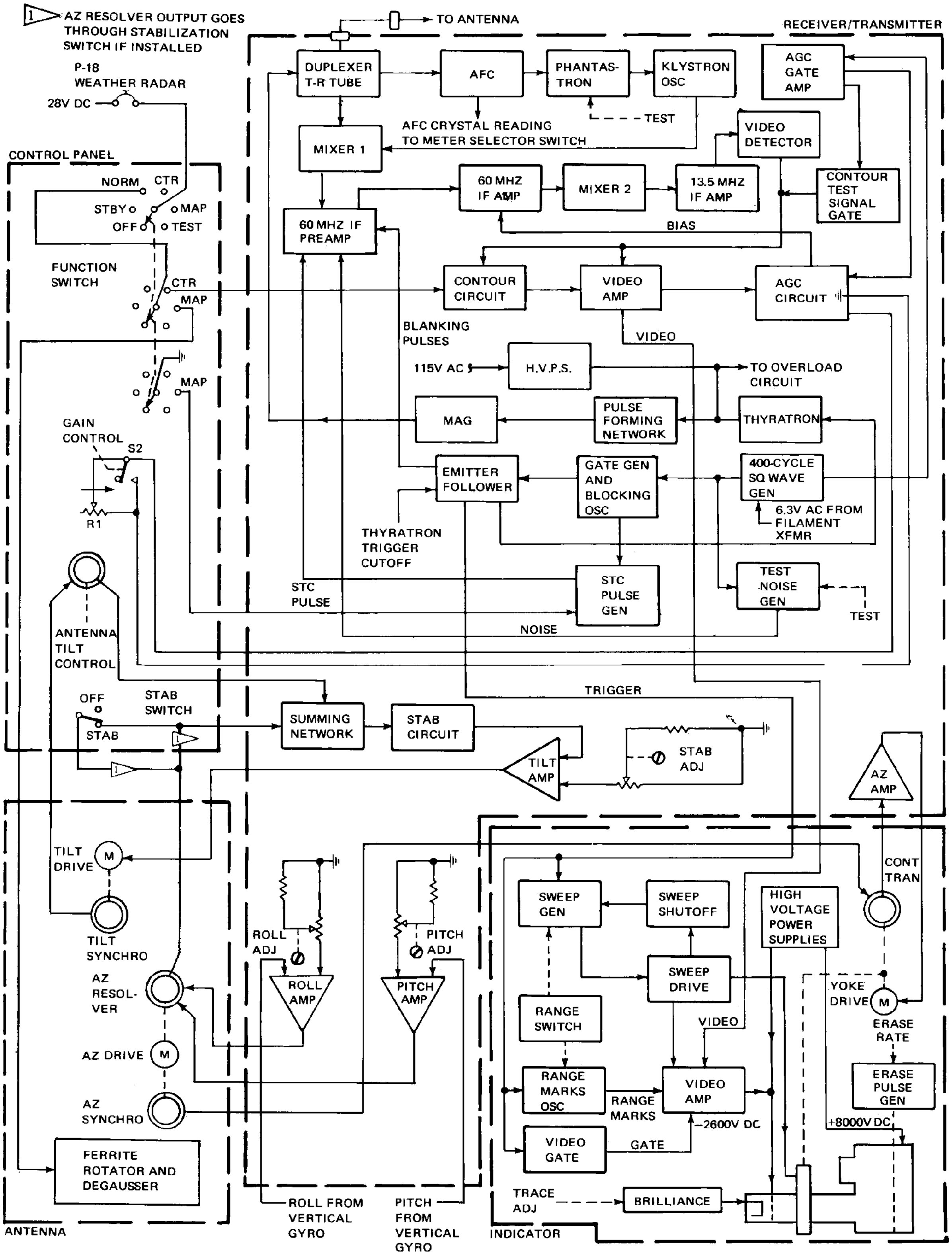
(2) For R/T with the filament relay, 115 volts ac is applied to the filament transformer, -250-volt dc power supply and blower motor. Through contacts of the power supply relay, 115 volts ac is applied to the R/T power supplies, reference transformer and the antenna and indicator. It also energizes the transmit relay and applies 26 volts ac to the antenna. The filament relay when energized, also applies 115 volts ac to a different winding in the filament transformer to reduce the magnetron filament voltage. With the transmit relay energized, the R/T high voltage power supply receives power and the system begins to transmit and the blower motor operates at normal speed.

E. The high voltage power supply (Fig. 2) with the pulse forming network produces a negative 2.5-microsecond pulse each time the thyratron is keyed by a pulse from the emitter follower. The energy pulse is conveyed through the duplexer and waveguide to the antenna. The AFC circuit of the receiver maintains a constant 60-MHz intermediate frequency at the output of the duplexer-mixer. If the transmitter frequency varies from 9375 MHz, the frequency output of the klystron oscillator must continuously follow 60 MHz above the transmitted signal frequency. When the klystron oscillator produces a signal that, when mixed in the duplexer-mixer with the received signal is at 60 MHz, the phantastron stops sweeping and the AFC locks on to the proper signal. The T-R tube fires during transmission forming a short to protect the receiver diodes from damage and reopens during the receive period. The resultant 60 MHz IF is processed through the 60-MHz IF preamplifier. Synchronized blanking pulses turn off the preamplifier during transmission time to protect the receiver circuits.

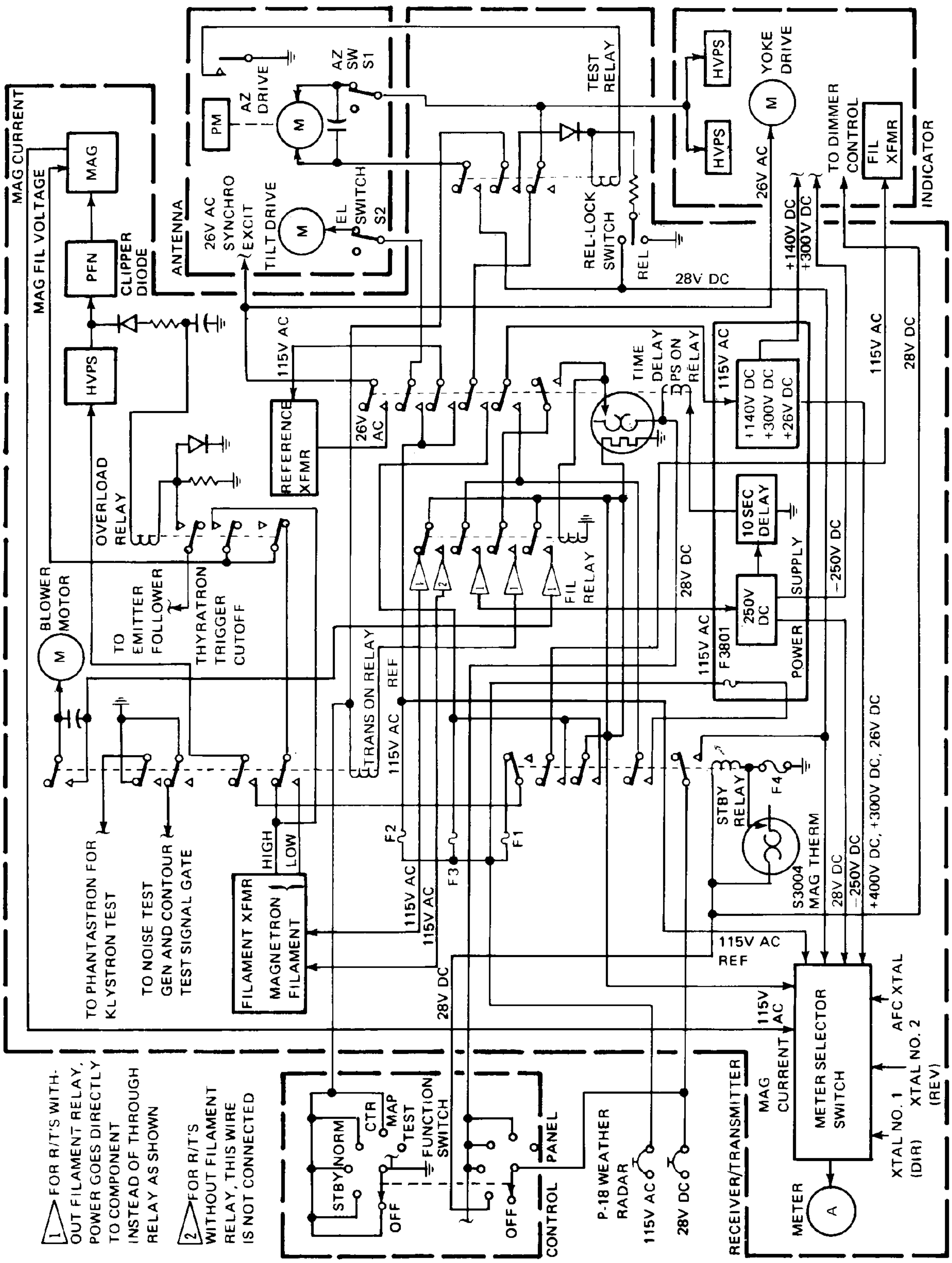
F. The signal from the preamplifier is sent to and amplified by the 60 MHz IF amplifier and sent to mixer 2. Mixer 2 combines this signal with a 46.5 MHz signal and the resultant 13.5 MHz signal is sent to the 13.5 MHz IF amplifier. Video information is extracted by the video detector and amplified by the video amplifier. The video amplifier applies the video signal to the indicator video amplifier. A portion of the detected video signal is fed to the AGC circuit. The filament transformer 6.3 volt ac input to the 400 Hz square wave generator produces 400 Hz square wave pulses which go to the agc gate amplifier, gate generator and test noise generator. After 100 miles of return signal, the AGC circuit samples the average noise level. The video signal is amplified and compared against the setting of the gain control on the control panel. When the gain control is in AUTO position, the R/T AGC circuits control the video gain of the 60 MHz IF amplifier.

G. Iso-echo contour (contour) operation is used to determine the density of rainfall areas. Contour operation is selected by placing the function switch on the control panel in CTR position. The contour circuit cuts off any video in excess of a set level from the video detector to produce dark areas on the indicator to identify the clouds with heavy precipitation. The blocking oscillator triggers the STC pulse generator. The STC circuit controls the gain of the IF preamplifier as a function of time so that near-in targets, up to 30 miles, appear equal in intensity on the indicator. The blanking oscillator also triggers the emitter follower that produces the sweep trigger for the indicator. It also turns on the thyratron and sends blanking pulses to the 60 MHz IF preamplifier. The STC pulse generator is disabled when the function switch is placed in MAP position.

H. Self-testing of the system may be run with or without the magnetron operating (Fig. 2 and 3). When the function switch is placed in TEST position, the transmit relay is de-energized and the test noise generator and contour test signal generator develops a video display on the indicator. To obtain a test pattern with the magnetron operating, the LOCK-REL switch is placed in LOCK position. This applies 28 volts dc to the test relay. When the antenna rotates to the 180-degree position, a ground from the antenna PM switch energized the test relay. This removes 115-volt ac power to the antenna and applies 28 volts dc to stop the azimuth drive motor at 180 degrees. Transmit relay, test relay, and the antenna PM switch are now in series. Both relay coils remain energized when the LOCK-REL switch is placed in the neutral position. Note that when the transmiter relay is de-energized a higher magnetron filament voltages is applied to maintain proper temperature. The R/T has a meter selector switch to check various voltages, crystal currents, and the magnetron current for proper operation.



Weather Radar System Schematic
Figure 2



Power Control and Testing Simplified Schematic
Figure 3

I. The magnetron is protected from overheating by a magnetron thermostat. When the magnetron thermostat closes, 28 volts dc blows fuse F4 and the system is turned off. If a high voltage overload occurs, the overload relay is energized removing the low magnetron filament voltage and applies the high voltage. A ground is applied to the emitter follower to cut off the thyratron trigger. If the overload is momentary, the system will return to normal operation. The overload relay will cycle as long as the cause of magnetron malfunction exists or until fuse F4 blows.

4. Indicator

A. The indicator consists of a light weight housing which contains a 5-inch cathode-ray tube on which target information is displayed. It also contains the final video amplifier, sweep generating, sweep shutoff, range mark, and erase circuits, yoke-rotating mechanism, high voltage power supply for the cathode-ray tube and an adjustable polaroid filter. Controls are provided for range selection, erase rate, trace adjustment, and panel light dimming. The indicator is mounted on the forward electronic control panel and is accessible to both the captain and first officer.

B. The sweep circuits are controlled by the trigger from the R/T. In the sweep generator sawtooth voltages and superimposed range marks are combined and sent to the sweep drive. The sweep drive develops deflection yoke current. A sweep shutoff stage stops the multivibrator and thereby determines the end of each sweep. The deflection yoke is rotated around the neck of the indicator tube in synchronism with the antenna by a servo system. This servo system is composed of the yoke drive control transformer, yoke drive gear train in the indicator, an azimuth synchro in the antenna, and azimuth amplifier in the R/T.

C. The erase rate control adjusts the depth and speed of the erase pulse generator. The trace control adjusts the storage tube bias and the dimmer control adjusts the panel lamp brilliance. The adjustable filter changes the color and intensity of the display for day and night operations.

5. Antenna

A. The antenna consists of a reflector-spoiler assembly, rf feed section, azimuth and elevation drive systems and associated synchros and resolvers. The antenna also has an EL switch and an AZ switch that can be used to disable the drive motors and a ferrite rotator and degausser. The antenna is located at the nose of the airplane in nose radome. It is connected to the R/T by sections by rigid and flexible waveguide.

B. Antenna stabilization circuits are contained in the control panel, antenna, and R/T, and receive inputs from the vertical gyro as shown in Fig. 2. Vertical gyro pitch and roll inputs may be adjusted with the ROLL, PITCH and GAIN ADJS potentiometers on the front of the R/T. Amplified values of pitch and roll go to the azimuth resolver to be resolved with the azimuth position of the antenna. Tilt error corrections from the azimuth resolver go through the stabilization switch (if installed) to the summation network. Values of tilt error correction caused by pitch and roll motions of the airplane and signals from the manual tilt control are processed in the summing network. The stabilization circuit amplifies the tilt signal to drive the antenna tilt drive motor. The tilt amplifier gain may be changed by adjusting the STAB ADJS potentiometer.

C. Ground mapping with the system provides a plan view of prominent land marks and terrain features such as cities, shore lines, mountains, etc. The terrain features are represented on the indicator in slant range, and in azimuth bearing of the airplane heading. This mode is initiated by placing the function switch in MAP position. In MAP position, the STC circuits are disabled and the ferrite rotator in the antenna receives 28 volts dc. The ferrite rotator changes the normal pencil beam to an equal energy beam which provides equal intensity returns from 5 to 50, or 50 to 100 nautical miles depending on airplane altitude and setting of the tilt control.

6. <u>Weather Radar Waveguide</u>

A. The waveguide is a standard rectangular section X-band waveguide and consists of removable, rigid, and flexible sections which connect the R/T to the antenna. The waveguide and antenna waveguide is pressurized from a small hole in the waveguide inside the R/T. The R/T is in a pressurized area of the airplane.

7. <u>Operation</u>

A. Prior to operation, set the controls in the following positions. On the control panel, place the gain control in mid position, the antenna tilt to 0, stabilization switch (if installed) to STAB and function selector switch to STBY. On the indicator, turn the trace adjust control fully counterclockwise and the erase rate control fully clockwise.

B. After warmup, turn the function selector switch to NORM position. The range selector switch on the indicator can be turned to either the 20, 50 or 150 position. On the 20-mile range, four range marks will appear with 5 miles between each mark. On the 50-mile range, five range marks will appear with 10 miles between each mark. On the 150-mile range, six range marks will appear with 25 miles between each mark.

C. The gain control on the control panel and the erase rate control on the indicator may now be varied from their initial positions for the optimum presentation of targets. The antenna tilt control may be used to tilt the antenna either up or down to search for weather or to get ground targets. The function switch, when placed in CONTOUR position, causes the areas of greatest rainfall density on the indicator to be shown as black spots.

WEATHER RADAR SYSTEM - TROUBLESHOOTING

1. General
   A. A prerequisite for accurately troubleshooting any system is a good knowledge of normal system operation and system component location. Refer to weather radar description and operation for review when required.
   B. To maintain the integrity of the weather radar system and other associated systems that can be affected whenever a unit is changed, a complete functional test of the weather radar system as specified in the weather radar adjustment/test procedure of the maintenance manual is required.
   C. Position airplane or tilt reflector so that large close by objects are not within 180-degree area scanned by antenna.

WARNING: DO NOT OPERATE THE WEATHER RADAR IN A HANGAR OR WITHIN 160 FEET OF ANY PERSONNEL. THESE CONDITIONS CAN CAUSE INJURY TO PERSONNEL.

WARNING: DO NOT OPERATE THE WEATHER RADAR WITHIN 50 FEET OF A FUEL SPILL OR OPEN FUEL CELLS. OPERATION OF THE WEATHER RADAR WITHIN THE 50 FT LIMIT CAN CAUSE A FIRE OR EXPLOSION. A FIRE OR EXPLOSION CAN CAUSE SERIOUS INJURY OR DEATH TO PERSONS AND CAUSE DAMAGE TO EQUIPMENT.

CAUTION: THE RECEIVER CRYSTALS MAY BE DAMAGED AS A RESULT OF STRONG RETURNS FROM NEARBY METALLIC SURFACES.

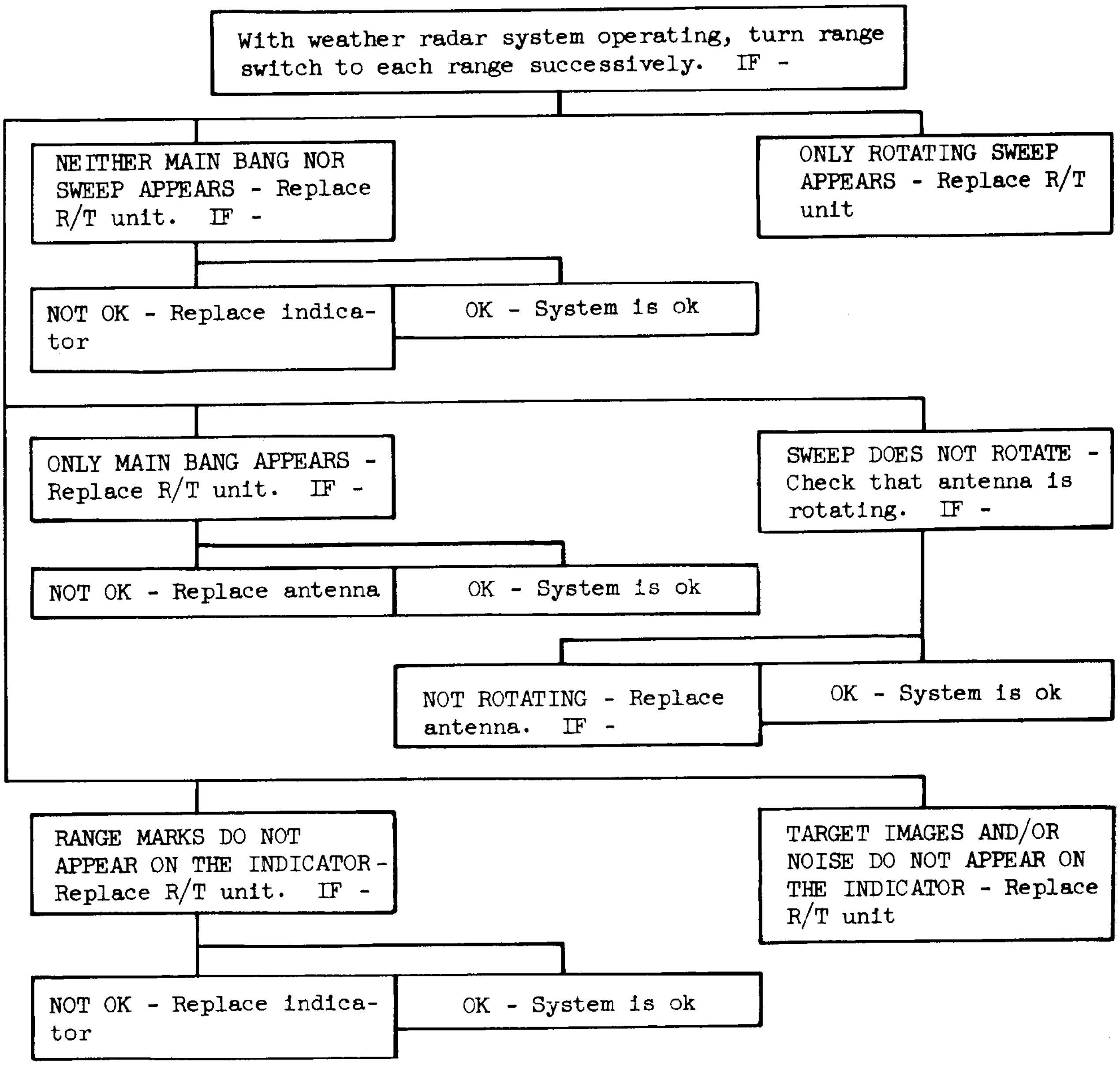
NOTE: Stabilization problems may occur as the result of an improperly mounted weather radar antenna. For correct operation, the antenna must be oriented to allow tracking of the vertical gyro (INS) pitch and roll signals within 1 degree. Alignment of the antenna support assembly is not necessary when simply replacing the antenna because it is accomplished at the time of manufacturing by the use of shims on the antenna support assembly.

2. Troubleshooting Chart

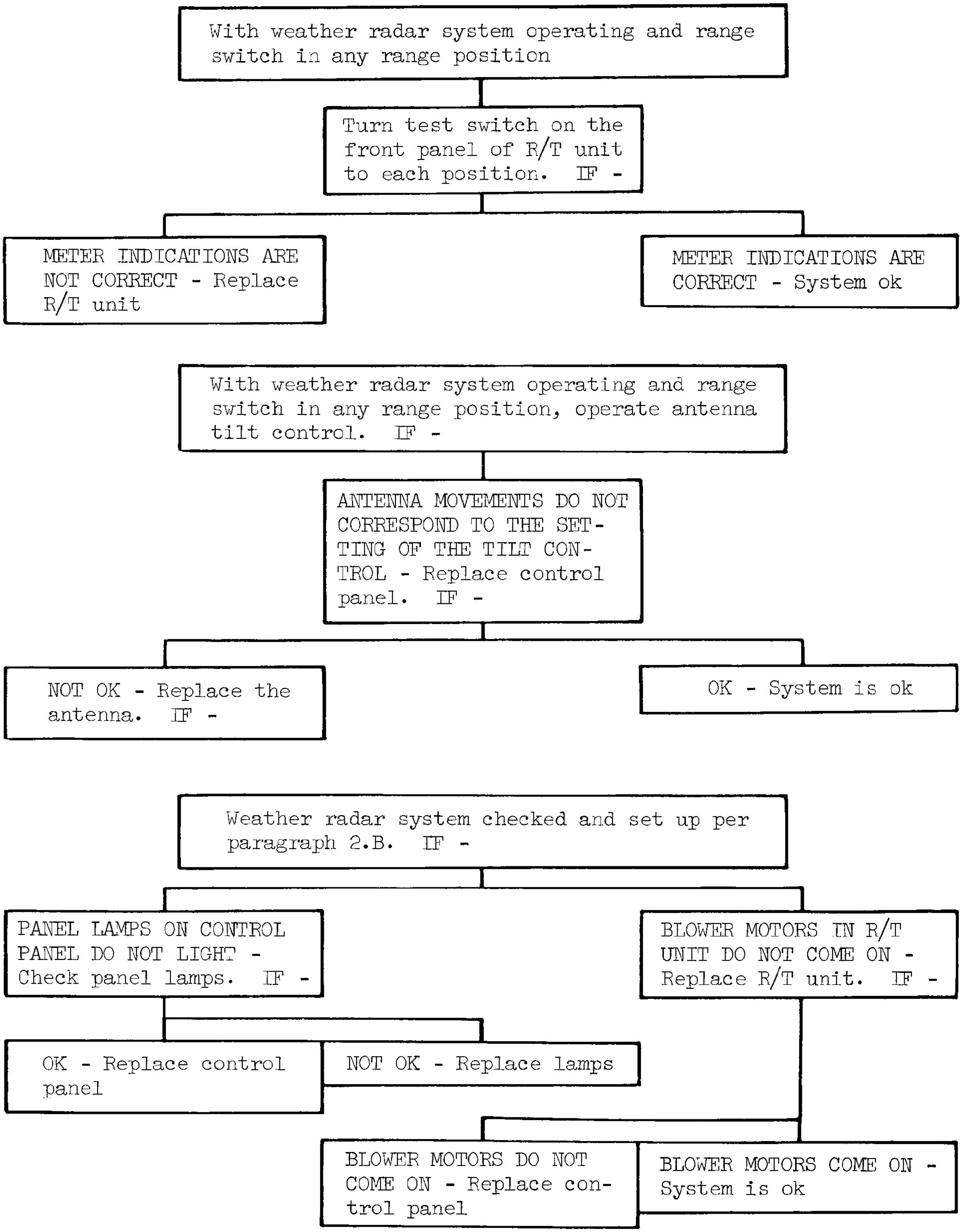
A. Substitution of components can be a convenient method of isolating trouble in the system. The troubleshooting chart should be used to determine the order in which to make tests and/or substitute components.

CAUTION: IF INDICATOR IS REMOVED FROM AIRPLANE, DO NOT TURN INDICATOR FACE DOWN OR STORAGE MESH MAY BE DISPLACED CAUSING A BURNOUT.

B. Before trying to troubleshoot the system, check to see if operating power is available. If power is available, turn the function switch to STBY and allow 5 minutes for equipment warmup. Adjust indicator for proper viewing.



Weather Radar Troubleshooting
Figure 101 (Sheet 1)



Weather Radar Troubleshooting
Figure 101 (Sheet 2)

WEATHER RADAR SYSTEM - ADJUSTMENT/TEST

1. Weather Radar System Test
   A. General
      (1) The vertical gyro system must be operational before proceeding with this test.

      WARNING: DO NOT OPERATE THE WEATHER RADAR IN A HANGAR OR WITHIN 160 FEET OF ANY PERSONNEL. THESE CONDITIONS CAN CAUSE INJURY TO PERSONNEL.

      WARNING: DO NOT OPERATE THE WEATHER RADAR WITHIN 50 FEET OF A FUEL SPILL OR OPEN FUEL CELLS. OPERATION OF THE WEATHER RADAR WITHIN THE 50 FT LIMIT CAN CAUSE A FIRE OR EXPLOSION. A FIRE OR EXPLOSION CAN CAUSE SERIOUS INJURY OR DEATH TO PERSONS AND CAUSE DAMAGE TO EQUIPMENT.

      CAUTION: RECEIVER CRYSTALS MAY BE DAMAGED AS A RESULT OF STRONG RETURNS FROM NEARBY METALLIC SURFACES.

   B. Equipment and Materials
      (1) Radar Stabilization Test Set - Bendix RST-1A, or equivalent
      (2) Tilt table
   C. Prepare to Test System
      (1) Set switch controls on control panel as follows:
         (a) Function switch in OFF.
         (b) Gain control fully counterclockwise.
         (c) Antenna tilt control to zero.
         (d) Stabilization switch in STAB.
      (2) Set indicator controls as follows (if installed):
         (a) RANGE and MARKS to any position.
         (b) ERASE RATE fully clockwise.
         (c) DIMMER fully counterclockwise.
         (d) TRACE ADJUST fully counterclockwise.
      (3) Open nose radome (AMM Chapter 53, Nose Radome). Make sure that switches on antenna are as follows:
         (a) Azimuth drive switch in AZ.
         (b) Elevation drive switch in EL.
         (c) ANT-1S only, roll switch in ROLL.
      (4) Set receiver-transmitter (R/T) meter selector switch in OFF position and REL-LOCK switch in center position.
      (5) Apply external power to airplane and energize circuit breaker panel P18. Close ac and dc weather radar circuit breakers.
   D. Test Weather Radar System

      NOTE: Steps (1) thru (6) check the R/T operation.

(1) Place function switch in STBY position. Make sure that blower in receiver-transmitter is operating.

NOTE: Blower will not operate in R/T with P/N 2067157-( ) on nameplate.

(2) Place meter selector switch in -250 position. Test meter should read approximately half-scale.

NOTE: This step not applicable to R/T with P/N 2067157-( ) on nameplate.

(3) After allowing minimum of 3 minutes for warmup, place function switch in TEST position. Blower motor should speed up.

(4) Place REL-LOCK switch in LOCK position for 5 seconds. Antenna should lock at 180 degrees azimuth and R/T should transmit.

NOTE: This step is not applicable to ANT-1S antenna.

(5) Place meter selector switch in following positions and check voltages on test meter with chart below.

POSITION	METER READING	POSITION	METER READING	POSITION	METER READING
AFC	0.3 to 0.7	-250	0.45 to 0.55	115	0.4 to 0.5
DIR	0.3 to 0.7	+140	0.45 to 0.55	115 REF	0.4 to 0.5
REV	0.3 to 0.7	+26	0.4 to 0.6	MAG	0.4 to 0.7
+300	0.45 to 0.55	+28	0.45 to 0.55		

A steady needle indicates afc lock-in.
Readings should be within 0.2 of each other.

(6) Place REL-LOCK switch in REL position, antenna should rotate. R/T should stop transmitting as indicated by zero reading on test meter. Blower motor should be operating at slower than normal speed.

NOTE: Steps (7) thru (14) check antenna and indicator operation.

(7) Adjust ERASE RATE and TRACE ADJUST controls to obtain clear display on indicator.

NOTE: Avoid setting TRACE ADJUST too high when antenna is stopped.

(8) Place function switch in NORM position. Operate azimuth drive (AZ-OFF) switch to stop antenna in dead-ahead position as determined by index pointer. Sweep on indicator should be within 5 degrees of zero mark on indicator.
(9) Place elevation switch in EL position.
(10) Make sure that the vertical gyro which supplies attitude reference information to weather radar system is operational.
(11) Operate antenna tilt control up and down. Antenna movements should follow antenna tilt control.
(12) Return azimuth drive switch to AZ and check that sweep starting point circumscribes a small circle centered on apex at bottom of indicator mask. Small open center should not exceed 1/8 inch in diameter. Sweep and range marks should appear sharp and well defined. Sweep should rotate smoothly and there should be no evidence of "spoking" on any range.
(13) Adjust range select switch on indicator to various positions and check for following quantity of range marks with corresponding switch position.

Range Switch Position	Number of Range Marks
Indicator PPI-1G-( ) 30	3
80	4
180	6
Indicator PPI-1E-( ) 20	4
50	5
150	6

(a) Last range mark on every range should be 1/16 to 3/4 inch from top of presentation. Range marks should not be jittery or vary in intensity.

NOTE: On PPI-1G-( ), indicator outer range mark intensities may be slightly lower than inner range marks.

(14) Verify smooth and continuous control of indicator panel lamps by operating DIMMER control.

E. Self-Test

(1) Adjust antenna tilt control to 10 degrees noseup. Place function switch in TEST position and RANGE and MARKS switch in maximum range position.

(2) Adjust ERASE RATE and TRACE ADJUST controls for continuous display and check for following display (Fig. 501).

(a) 0 to 45 miles

1) Absence of noise indicates proper STC function.

(b) 45 to 60 miles

1) Threshold test signal appears as intense noise (system sensitivity check).

(c) 60 to 110 miles

1) System noise (as in normal operation with no targets in range).

(d) 110 to 135 miles

1) Contour signal appearing as two bright bands, separated by a dark bank of approximately equal width (8 miles).

(e) 135 to 180 miles

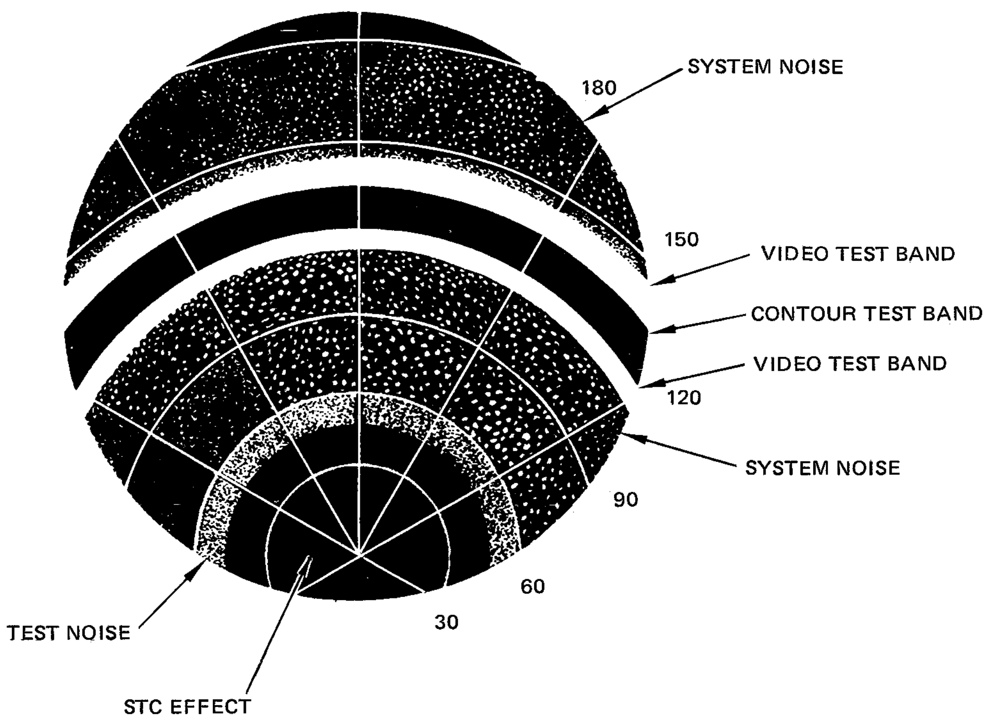
1) System noise.

F. Antenna Stabilization Check (Optional)

(1) Remove vertical gyro from mount, which supplies weather radar system with pitch and roll signals (AMM 34-22-0/001).

CAUTION: DO NOT EXCEED 20 DEGREES OF GYRO DISPLACEMENT DURING THESE TESTS.

NOTE: Do not adjust antenna stabilization with auxiliary vertical gyro (if installed).



Indicator During Self Test
Figure 501

(2) Tilt gyro vertical centerline towards left side of airplane and check that radar antenna is directed above horizontal when approaching relative bearing of 270 degrees, and below horizontal when approaching relative bearing of 90 degrees.
(3) Tilt gyro vertical centerline toward nose of airplane and check that radar antenna is directed above horizontal when approaching dead-ahead.

G. Antenna Stabilization Check (Using RST-1A Test Set)

(1) Place vertical gyro on tilt table by performing step F.(1) above.

NOTE: Observe any notes or cautions following step F.(1) above.

(2) Place test set antenna toggle switch to ANT-1( ) position. Connect test set to mating connector on front of rcvr/xmtr unit.
(3) Place antenna tilt control 10 degrees up and function switch in TEST position. Check that antenna is rotating.

NOTE: Test set meter sensitivity is 5 microamperes full-scale deflection for STAB ERR, OSC TEST, and AZ ERR test positions. For all other positions sensitivity is 25 microamperes except when 5°FS button is depressed. 5°FS button increases sensitivity to 5 microamperes full-scale deflection.

Perform steps (4) thru (11) below for system with ANT-1N antenna without MOD 6.

(4) Place test set selector switch in STAB ERR position. Make sure that meter needle oscillates left and right with antenna 0.7 to 0.9 degree in direction of maximum deflection. Make sure that deflection in other direction is less than 0.8 degree (note direction of maximum deflection).
(5) If results of step (4) are not in tolerance, adjust STAB GAIN potentiometer on front of rcvr/xmtr for a deflection of 0.7 degree in direction of maximum deflection noted above.
(6) Place test set selector switch in OSC TEST position. Check that meter needle oscillates in synchronization with antenna rotation. Check that meter needle is not jittery, but may hesitate slightly. Adjust antenna tilt control 10 degrees down. Check that meter needle is not jittery but may hesitate slightly.

(7) Place test set selector switch in ANT POS position. Adjust antenna tilt control as follows and observe deflection of meter needle.

Antenna Tilt	Meter Deflection
10 degrees up	10 ±2 degrees right
10 degrees down	10 ±2 degrees left
zero	0 ±1 degree

(8) Adjust antenna tilt control to 0 degree and tilt table to 20 degrees nosedown. Meter needle should oscillate smoothly between 20 ±2 degrees left and right in synchronization with antenna rotation.
(9) If results of step (8) are out of tolerance, adjust PITCH GAIN potentiometer on rcvr/xmtr as required for minimum error. Maximum allowable error is ± 2 degrees. Stop antenna at 0-degree azimuth position and verify 20 ±2-degree pitch.
(10) Adjust tilt table for 20 degrees right roll. Make sure that meter needle oscillates smoothly between 20 ±2 degrees left and right in synchronization with antenna rotation.
(11) If results of step (10) are out of tolerance, adjust ROLL GAIN potentiometer on rcvr/xmtr as required for minimum error. Maximum allowable error is ± 2 degrees. Stop antenna at 90-degree azimuth position and verify 20 ±2-degree roll.

NOTE: Perform steps (12) thru (20) below for system ANT-1N antenna with MOD 6.

(12) Make sure that RDR-1E/SYN-1B switch, if installed (located on back of antenna), is in RDR-1E position.
(13) Place test set selector switch in 10° PITCH position. Make sure that meter needle oscillates with antenna between 0 ±1 and 20 ±2 degrees right.

(14) If results of step (13) are not in tolerance, adjust PITCH GAIN ADJ potentiometer on front of rcvr/xmtr for test set meter deflection within above tolerances. Lock potentiometer and record exact meter deflection.
(15) Place test set selector switch in STAB ERR position. Make sure that meter needle oscillates left and right with antenna 0.4 to 0.5 degree in direction of maximum deflection. Make sure that deflection in other direction is less than 0.5 degree (note direction of maximum deflection).
(16) If results of step (15) are not in tolerance, adjust STAB GAIN potentiometer on front of rcvr/xmtr for deflection of 0.4 degree in direction of maximum deflection noted above.
(17) Place test set selector switch in OSC TEST position. Make sure that meter needle oscillates with antenna rotation. Make sure that meter needle is not jittery, but may hesitate slightly. Place antenna tilt control 10 degrees down and make sure that meter needle continues to oscillate as before.
(18) Place test set selector switch in 10° ROLL position. Adjust ROLL GAIN ADJ potentiometer on front of rcvr/xmtr for same meter deflection as recorded in step (13). Allowable difference between pitch deflection of step (14) and roll deflection in this step is 0.5 degree. Lock potentiometer.
(19) Place test set selector switch in AZ ERR position. Make sure that meter indicates less than 1 degree.
(20) Place test set selector switch in ANT POS position. Adjust antenna tilt control as follows and observe deflection of meter needle.

Antenna Tilt	Meter Deflection
10 degrees up	10 ±2 degrees right
10 degrees down	10 ±2 degrees left
zero	0 ±1 degree

H. Restore Airplane to Normal

(1) Open attitude reference system circuit breakers on circuit breaker panels P18 and/or P6. Disconnect and remove all test equipment.

(2) Return vertical gyro to mount and secure. For Vertical Gyro Removal/Installation procedures refer to AMM 34-22-0, Attitude Reference System. Close attitude reference system circuit breakers and make sure that system indicators indicate attitude of airplane.

(3) Close and secure radome.

(4) If no longer required, remove electrical power from airplane.

WEATHER RADAR ANTENNA - REMOVAL/INSTALLATION

1. Remove Weather Radar Antenna

WARNING: BE SURE POWER IS OFF AND WEATHER RADAR CIRCUIT BREAKERS ARE PULLED OPEN ON CIRCUIT BREAKER PANEL P18. OPERATION OF RADAR EQUIPMENT INVOLVES THE USE OF HIGH VOLTAGES THAT ARE DANGEROUS TO LIFE. MAINTENANCE PERSONNEL MUST AT ALL TIMES, OBSERVE ALL SAFETY REGULATIONS. DO NOT WORK ON EQUIPMENT WITH VOLTAGE SUPPLY ON. UNDER CERTAIN CONDITIONS DANGEROUS POTENTIALS MAY EXIST IN CIRCUITS WITH POWER SWITCH OFF AND CIRCUIT BREAKERS OPEN, DUE TO CHARGES RETAINED BY CAPACITORS, FAULTY SWITCHES, AND SO FORTH.

A. Remove screws securing antenna radome, swing radome up, and lock in position.

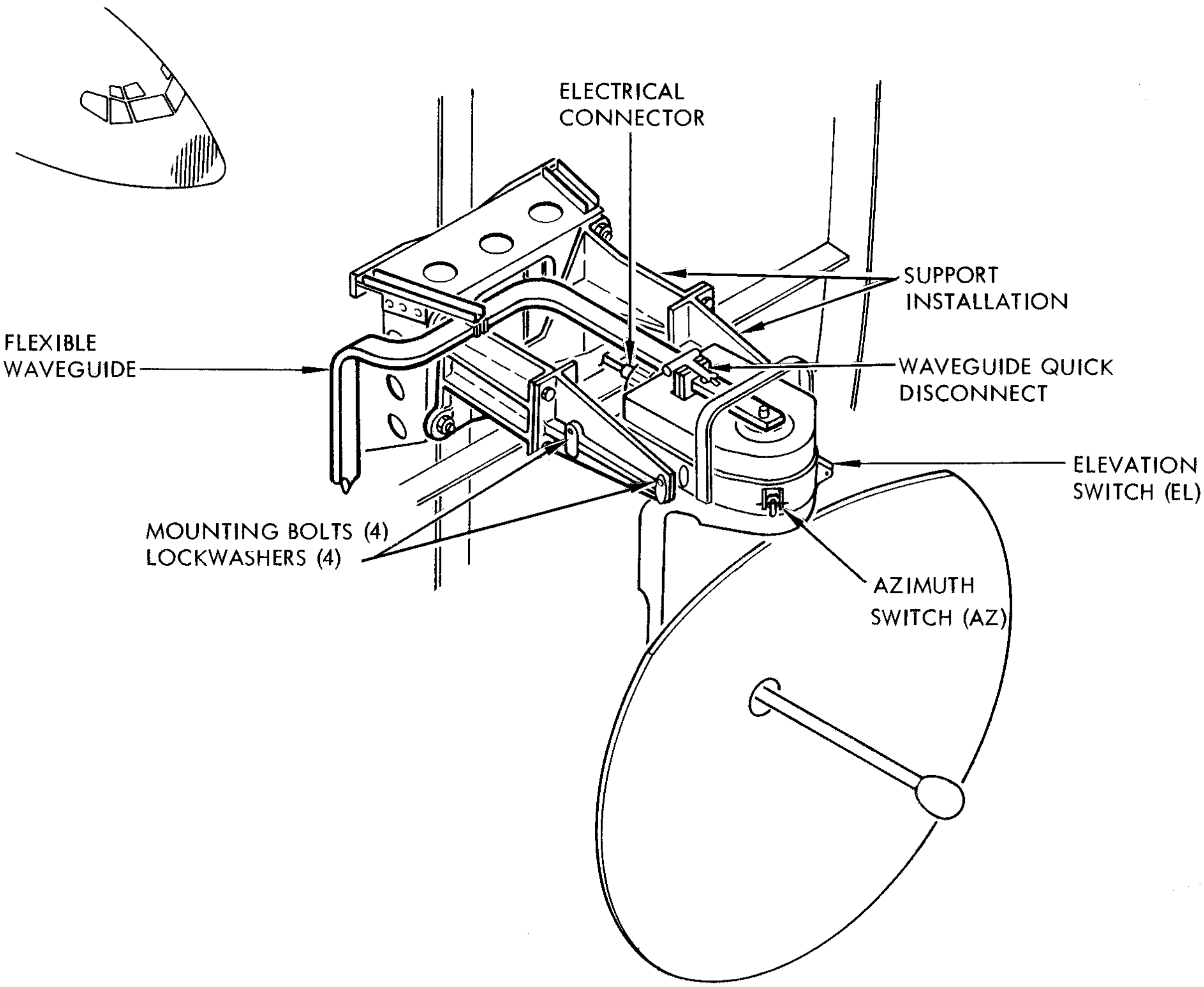
CAUTION: DO NOT PRY RADOME OR USE ANY LEVER IN SCREW HOLES. A LARGE SUCTION CUP MAY BE PRESSED ON RADOME TO OBTAIN HANDHOLD FOR OPENING. IN ORDER TO PREVENT WIND DAMAGE TO THE RADOME IT IS NECESSARY THAT BOTH RODS, DESIGNED TO SUPPORT THE RADOME WHEN OPEN, BE ATTACHED PROPERLY TO STRUCTURE WHENEVER THE RADOME IS OPEN.

B. Remove weather radar antenna connector plug. (See figure 401.)
C. Disconnect waveguide section from antenna by disengaging waveguide quick disconnect and removing gasket.
D. Remove four antenna mounting bolts and lower the antenna unit.

NOTE: Antenna reflector must be handled with care so that no undue pressure is exerted, thus, bending the dish and distorting the beam. If the antenna is not to be replaced or reinstalled for any period of time, the waveguide and connector must be protected from foreign matter.

2. Install Weather Radar Antenna

A. Remove protective covers from waveguide and electrical connector.
B. Place antenna unit into place and install four mounting bolts and lockwashers. (Fig. 401)
C. Lockwire mounting bolts.
D. Install gasket between waveguide flanges.
E. Couple two mating flange surfaces together and connect with waveguide quick-disconnect.
F. Install antenna electrical connector.
G. Close radome and install mounting screws.
H. Perform Weather Radar System - Adjustment/Test. Refer to 34-41-0.



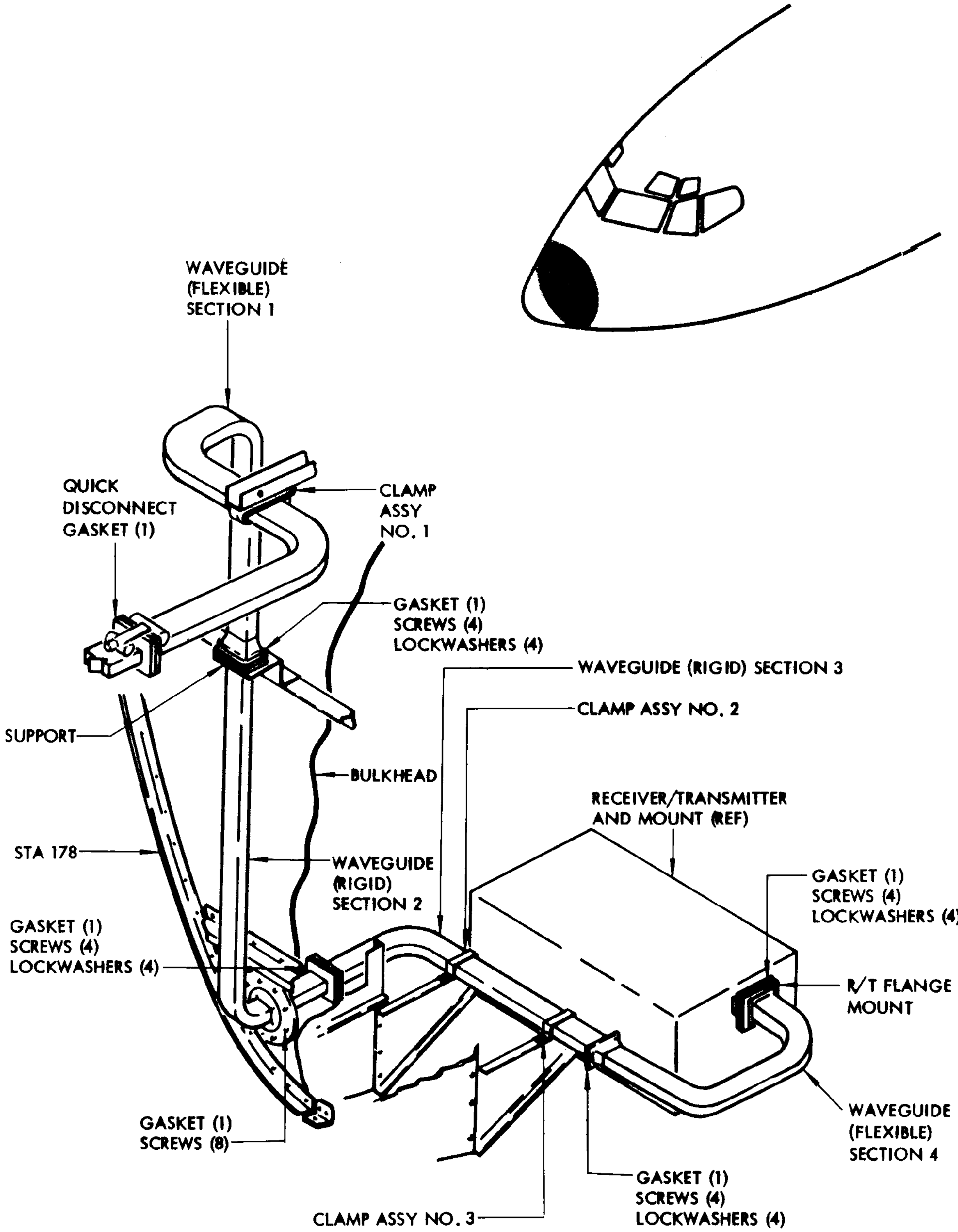
Weather Radar Antenna Installation
Figure 401

WEATHER RADAR WAVEGUIDE - REMOVAL/INSTALLATION

1. General
   A. The waveguide transmission line consists of rigid and flexible waveguide sections (Ref Fig. 401 for section identification).
2. Prepare to Remove Waveguide
   A. Open radome compartment (Ref Nose Radome - Chapter 53).
   B. Open forward access door (1103) to gain access to waveguide sections forward of receiver-
      transmitter (RT) unit.
   C. Remove weather radar RT unit.
3. Remove Waveguide
   A. Disconnect section 1 by disengaging waveguide quick disconnect at antenna. Remove the gasket.
   B. Remove four screws and four lockwashers between sections 1 and 2.

      NOTE: Screws securing sections 1 and 2 are 1/8" longer than screws for others sections.

   C. Remove clamp assembly No. 1. Remove two support assembly washers, section 1 and gasket.
   D. Remove four screws and lockwashers on section 4 flange at the RT.
   E. Support section 4 and remove four screws and lockwashers between section 3 and section 4. Remove section 4 and gaskets.
   F. Remove four screws, gasket and lockwashers between sections 2 and 3.
   G. Remove clamp assemblies No. 2 and No. 3 on section 3. Remove section No. 3.
   H. From radome compartment remove eight screws on section 2 flange on bulkhead. Carefully remove section 2 through hole in bulkhead.
4. Prepare to Install Waveguide
   A. Remove protective covers from waveguide sections, if installed.
   B. Check all waveguide sections for any damage or foreign matter.
   C. Check all removed gaskets for any damage and replace as necessary.
5. Install Waveguide
   A. Slide section 2 through hole in radome bulkhead and place in final position.
   B. Fasten eight screws on section 2 flange to bulkhead.
   C. Align flanges of waveguide sections 1 and 2. Fasten together support plate, gasket, flanges and two support assembly washers with four screws and four lockwashers. Use proper screw size to connect sections 1 and 2.
   D. Connect section 1 to antenna with gasket and waveguide quick-disconnect.
   E. Place clamp assembly No. 1 on section 1.
   F. Fasten clamp assembly No. 1.
   G. Place section 3 in position and align with section 2. Adjust clamp assemblies No. 2 and No. 3 to minimize strain between sections 2 and 3. Fasten sections 2 and 3 with gasket, four screws and four lockwashers.



Weather Radar Waveguide Installation
Figure 401

H. Align section 4 with R/T flange mount and connect with gasket, four screws and lockwashers.
I. Align sections 3 and 4 and fasten with gasket, four screws, and lockwashers.
J. Install R/T unit in mount.
K. Close radome.
L. Perform operational check of weather radar system (Ref Weather Radar System - A/T).

WEATHER RADAR WAVEGUIDE - ADJUSTMENT/TEST

1. Weather Radar System Test
   A. General
      (1) A VSWR waveguide test is required only when maintenance has been performed on waveguide installation.
   B. Equipment and Materials
      (1) Radar Test Set AN/UPM 12/12A or equivalent (X-Band VSWR Test Set)
   C. Prepare to Test Waveguide
      (1) Remove receiver-transmitter from mount. Determine if flexguide is required to connect test set to waveguide.
      (2) Calibrate test set to frequency of 9375 MHz. (See test set instructions.)

CAUTION: DO NOT PLACE TEST SET ON-OFF SWITCH TO ON POSITION UNLESS INTERCONNECTING CABLE ASSEMBLY IS PROPERLY CONNECTED.

NOTE: If flexguide is to be used, make calibration per applicable test set instructions and calibrate VSWR of waveguide accordingly.
Allow approximately 5 minutes for test set to reach operating temperature.

      (3) Connect test set to waveguide. Disconnect waveguide at antenna and connect dummy load to waveguide.
   D. Test Waveguide
      (1) Measure VSWR of waveguide run. VSWR should not exceed 1.25:1.

NOTE: If flexible waveguide is used maintain same curvature as it had during calibration in step C (2).

   E. Restore Airplane to Normal.
      (1) Disconnect dummy load and reconnect waveguide to antenna.
      (2) Disconnect test set from waveguide. Reconnect waveguide to receiver-transmitter shockmount.
      (3) Place function switch in OFF position. Install receiver-transmitter in shockmount.
      (4) Perform operational check of weather radar system (Ref Weather Radar System - Adjustment/Test).
      (5) If no longer required, remove electrical power from airplane.

GPWS CONTROL MODULE - SERVICING (BULB REPLACEMENT)

1. General
   A. This subject has one task. The task is for replacement of the bulb. It is also necessary to do a test of the bulb operation.
2. GPWS Control Module Switch/Indicator Bulb Replacement
   A. Standard Tools and Equipment
      (1) Lamp - 5 volts, 60ma - P/N MS24515-685
   B. References
      (1) 24-22-0, Manual Control (Apply Power)
   C. Procedure
      (1) Open these circuit breakers and attach DO-NOT-CLOSE tags:
         (a) P18
            1) GND PROX WARN
         (b) P6
            1) DIM & TEST and INDICATOR LIGHTS

CAUTION: DO NOT USE PLIERS OR ANY OTHER TOOLS TO REMOVE THE CAP ASSEMBLIES. THIS CAN CAUSE DAMAGE TO THE SWITCH CAP.

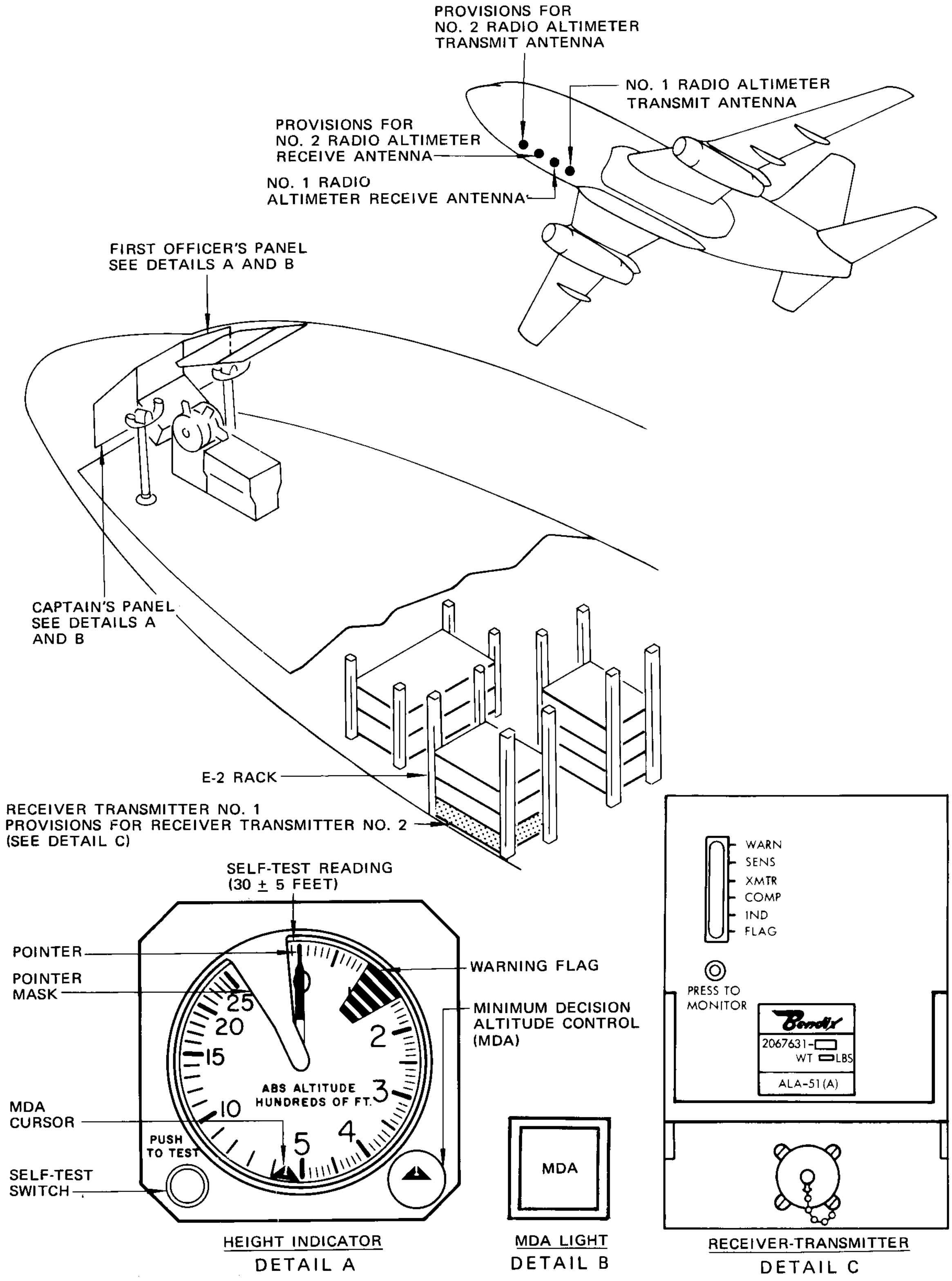
      (2) Replace lamps:
         (a) Hold the cap assembly tightly on the two sides and pull the assembly out of the switch.
         (b) Turn the cap assembly to get access to the lamps.
         (c) Remove lamps from the cap assembly.
         (d) Install the replacement lamps.
         (e) Turn the cap and put the cap assembly into the switch.
      (3) Close these circuit breakers:
         (a) P18
            1) GND PROX WARN
         (b) P6
            1) DIM & TEST and INDICATOR LIGHTS
      (4) Supply electrical power (Ref 24-22-0).
      (5) Set the LIGHTS switch on the Pilots' center panel to the TEST position. Make sure the lamps come on.
      (6) Remove electrical power if it is not necessary (Ref 24-22-0).

GPWS CONTROL MODULE - REMOVAL/INSTALLATION

1. General
   A. This subject has two tasks. One is for the removal and one is for the installation of the GPWS control module. These tasks have steps for the removal of the control module, the installation of the replacement control module, and a test of the control module panel lights.
   B. The GPWS control module is installed on the first officer's instrument panel, P3.
2. GPWS Control Module Removal
   A. Procedure
      (1) Open these circuit breakers and attach DO-NOT-CLOSE tags:
         (a) P6 Main Power Distribution Panel
            1) DIM & TEST INDICATOR LIGHTS
         (b) P18 Overhead Circuit Breaker Panel
            1) GND PROX WARN
      (2) Remove the GPWS control module.
         (a) Loosen the fasteners on the control module.
         (b) Pull the control module out from the instrument panel until you can get to the electrical connector.
         (c) Disconnect the electrical connector from the control module.
3. GPWS Control Module Installation
   A. References
      (1) 24-22-0, Manual Control (Apply Power)
   B. Procedure
      (1) Open these circuit breakers and attach DO-NOT-CLOSE tags:
         (a) P6 Main Power Distribution Panel
            1) DIM & TEST INDICATOR LIGHTS
         (b) P18 Overhead Circuit Breaker Panel
            1) GND PROX WARN
      (2) Install the GPWS control module:
         (a) Connect the electrical connector to the control module.
         (b) Lightly push the control module into the instrument panel.
         (c) Tighten the fasteners that attach the control module to the instrument panel.
      (3) Remove the DO-NOT-CLOSE tags and close these circuit breakers:
         (a) P6 Main Power Distribution Panel
            1) DIM & TEST INDICATOR LIGHTS
         (b) P18 Overhead Circuit Breaker Panel
            1) GND PROX WARN
      (4) Supply electrical power (Ref 24-22-0).
      (5) Make sure the GPWS control module panel lights are on.
      (6) Remove electrical power if it is not necessary (Ref 24-22-0).

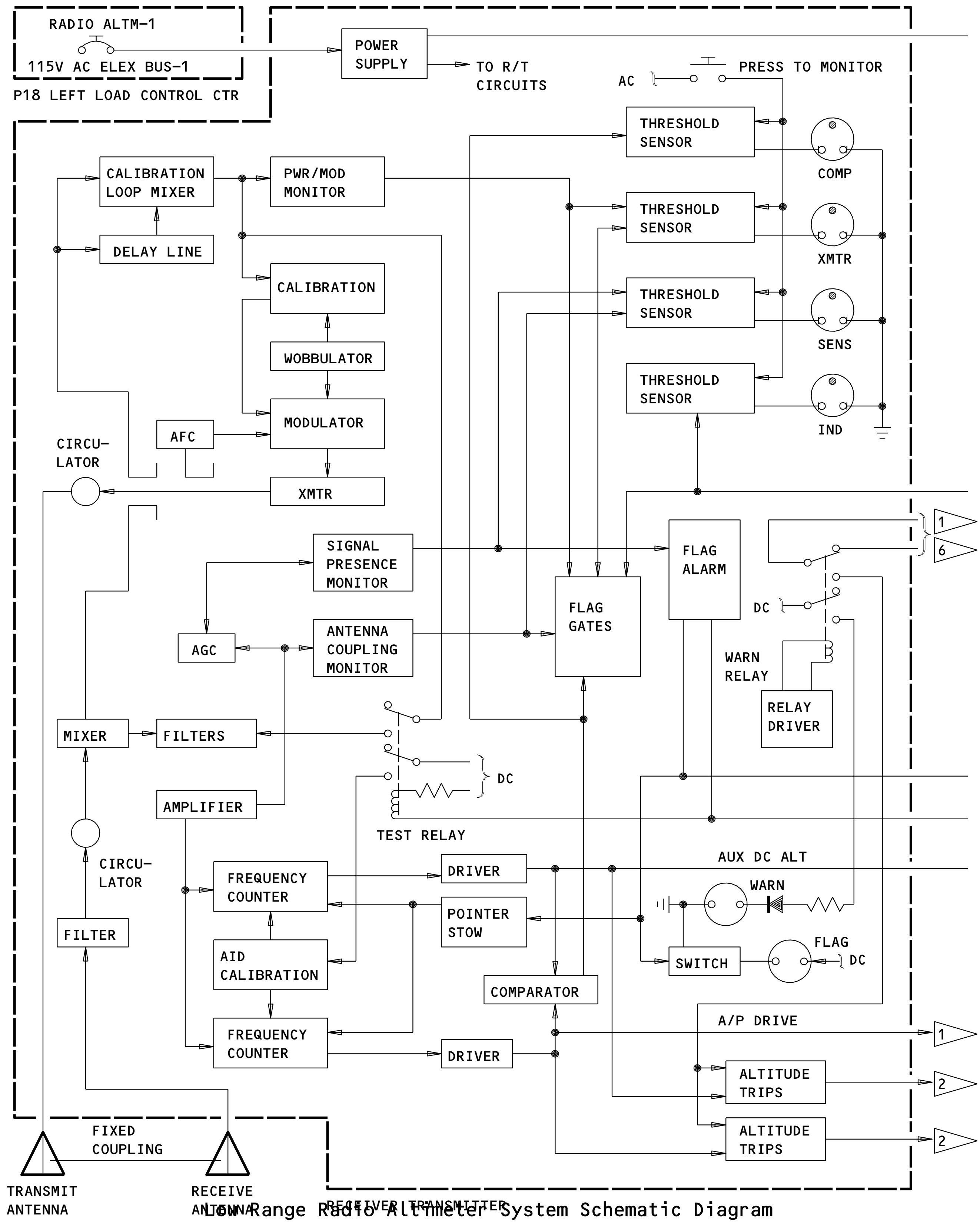
LOW RANGE RADIO ALTIMETER SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The low range radio altimeter (LRRA) system supplies vertical position information for use by the pilots during approach and landing phases of flight operation. The LRRA system provides an accurate measurement of absolute altitude (height above terrain) from 2500 feet to touchdown. Flag alarm, A/P failure warning, dc altitude, and vertical terrain clearance (altitude trip) outputs are available for use by other systems.
   B. The system consists of a receiver-transmitter, two height indicators, two instrument panel mounted MDA lights, transmit antenna, and receive antenna. The location of components is shown on Fig. 1.
   C. The system receives 115-volt ac power through circuit breaker RADIO ALTM-1 located on electronic circuit breaker panel P18-1 at the left load control center.
   D. The LRRA system supplies failure warning and dc altitude signals to the autopilot system, flag alarm, and 200- and 1500-foot altitude trip signals to the flight director system.
2. Receiver-Transmitter (Fig. 1)
   A. The receiver-transmitter is located on shelf four of E2 electronic rack. All electrical connections to the receiver-transmitters are made through a connector located on the rear panel. An auxiliary connector, located on the front panel, is provided for connecting to test equipment.
   B. Controls and indicators located on the front panel are WARN, SENS, XMTR, COMP, IND, and FLAG monitor indicators and a PRESS TO MONITOR button.
3. Height Indicator (Fig. 1)
   A. Airplane altitude is read from a circular dial, which is calibrated from -10 to 2500 feet. Dial calibration is linear from -10 to 480 feet and logarithmic from 480 to 2500 feet. Below 500 feet, the dial is divided into 10-foot increments. Above 500 feet, it is divided into 100-foot increments.
   B. The MDA control knob is used to adjust the position of the MDA cursor to any altitude shown on the dial. The MDA cursor setting designates the altitude at which the instrument panel mounted MDA light will be energized during airplane descent or de-energized during airplane ascent.
   C. The warning flag is solenoid operated and will come into view whenever the solenoid is de-energized.
   D. The PUSH TO TEST switch is used to initiate a functional check of the LRRA system.

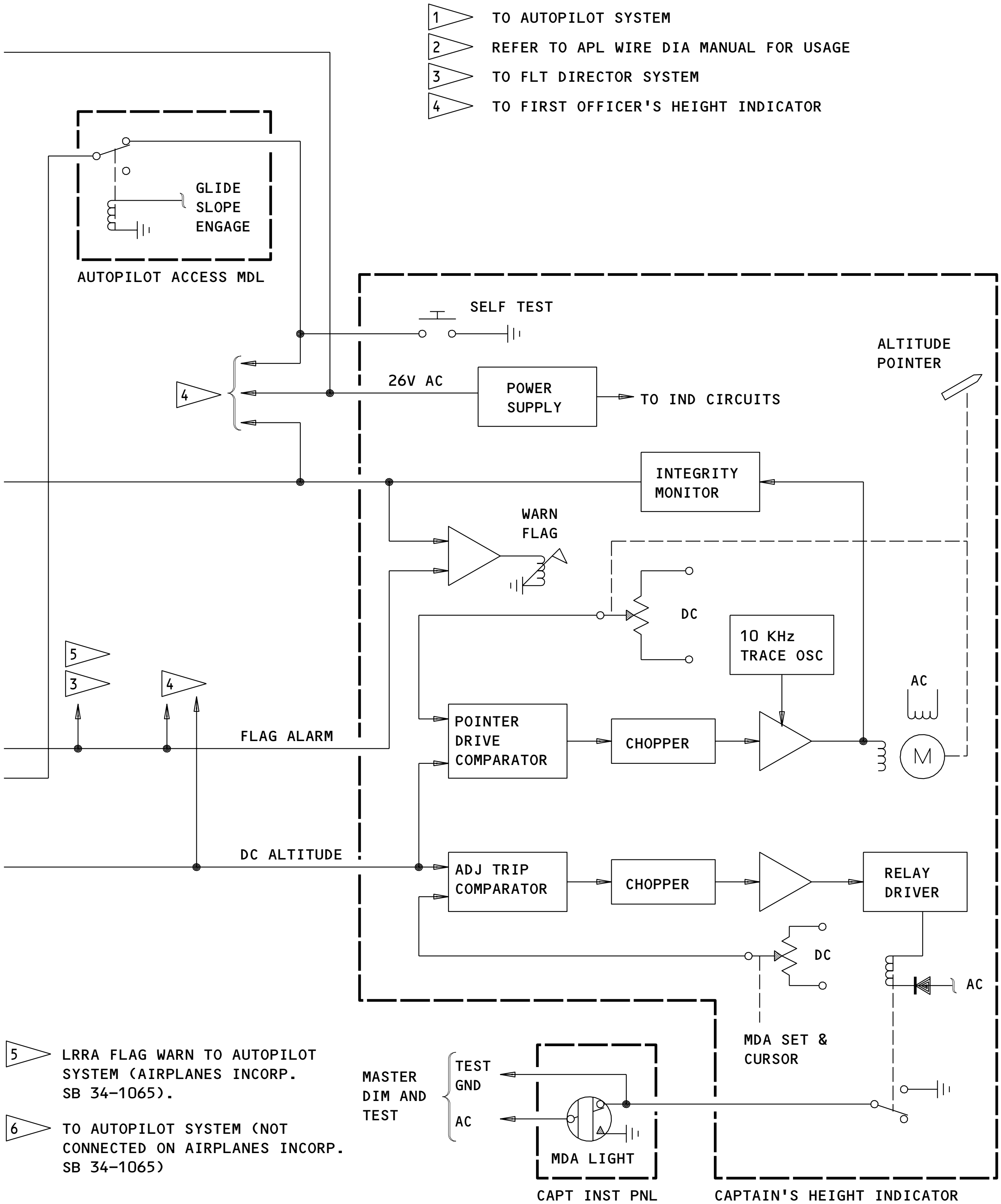


Low Range Radio Altimeter System Component Location
Figure 1

4. Antenna
   A. The antennas are linearly polarized horns. Two identical antennas are used in each system, one for transmitting and one for receiving. The antennas are interconnected with a coaxial cable, which provides a controlled leakage input signal to the receiver transmitter integrity monitor circuits. The antennas are connected to the receiver-transmitter with coaxial cables and are flush-mounted on the lower fuselage centerline of the airplane as follows:
      (1) TRANSMIT ANT STA 450
      (2) RECEIVE ANT STA 430
5. MDA Light (Fig. 1)
   A. A MDA light is installed on each pilot's instrument panel and is controlled by the height indicator located on the same panel. The light has a press-to-test feature and brightness is controlled by the master dim control.
6. Operation (Fig. 2)
   A. Altitude Measurement Operation
      (1) An oscillator in the modulator generates a 155-Hz modulation signal. This signal is wobbulated by a 31-Hz input signal from the wobbulation generator and then applied to a generator which produces the triangular wave modulating signal for the transmitter.
      (2) The LRRA system provides altitude data when power is applied to the receiver-transmitter. Low voltage power is then supplied to all circuits in the receiver-transmitter and to the height indicator power supply. The LRRA system is turned on by closing its circuit breaker.
      (3) The signal from the modulator frequency modulates an oscillator signal in the transmitter. The resultant signal is amplified and frequency multiplied to produce the nominal 4.3-GHz transmitter output signal.
      (4) A portion of the transmit signal is coupled to the AFC circuits. The AFC loop acts to hold the center frequency of the transmitted signal to 4.3 GHz. Another sample of the transmitted signal is coupled to the calibration loop mixer and delay line. The resultant difference frequency output from the calibration loop mixer is applied to the calibration circuits where it is converted to a dc voltage (altitude). This voltage is compared to a standard reference voltage and any difference is sensed and applied to the modulator for altitude self-calibration. The calibration is accomplished by varying the slope of the triangular wave to reduce the calibration circuit difference (error) signal to zero.



Low Range Radio Altimeter System Schematic Diagram
Figure 2 (Sheet 1)



Low Range Radio Altimeter System Schematic Diagram
Figure 2 (Sheet 2)

(5) The output signal from the transmitter is applied through the transmit circulator to the transmit antenna. The signal from the antenna is radiated to the earth and reflected back to the receive antenna. The received signal is applied to a filter to eliminate interference frequencies and the output is routed through the receive circulator to the mixer. The transmit and receive circulators transfer any reverse direction power to high loss loads to provide protection (isolation) for receiver-transmitter circuits.

(6) In the mixer, the received signal is beat against a sample of the transmit signal and the difference frequency output is applied to the filters. The mixer output is directly proportional to the time required for the transmit signal to travel to the ground and back, and is proportional to the distance from the ground (altitude).

(7) The AGC circuit determines the relative signal strength in the receiver section by monitoring the amplifier output. An AGC signal is applied to the filters to vary the filter cutoff frequency and maintain the receiver signal level near a preset threshold. Signal strength normally increases with decreasing airplane altitude. The output of the filters is applied to the amplifier where it is amplified to a level suitable to drive the two frequency counters, AGC circuit, and antenna coupling monitor.

(8) The operation of both frequency counter and driver circuits is identical. The amplifier input to a frequency counter is converted to a dc analog voltage which is proportional to the input frequency (altitude). This voltage is amplified by a driver and the driver output is the dc altitude voltage supplied to external using equipment. The output from one driver is supplied to the height indicator and the output from the other driver is available for use by other airplane systems. In addition, the output from the two drivers is applied to the receiver-transmitter high and low altitude trip circuits.

(9) The trip comparator also controls the 800-Hz audio warning output supplied by the audio amplifier. The audio warning is initiated by comparator when the airplane altitude is 50 to 100 feet above (internally adjustable initiation point) the MDA altitude setting. As the airplane descends in altitude, the tone increases in level until the MDA altitude is reached. When the airplane reaches the MDA altitude, the tone is reduced to a low level and remains at the low level as long as the airplane is below the MDA altitude. If the airplane now ascends to an altitude which causes the MDA light to be turned off, the audio warning is initiated at a maximum level and decreases as the airplane altitude increases. At the initiation point, the audio tone is reduced to a low level.

(10) The operation of all six altitude trip circuits is identical. The dc altitude input voltage from the driver is compared to a reference voltage (altitude trip setting). If the input voltage exceeds the reference voltage, the trip relay remains de-energized. When the input voltage is less than the reference voltage, the trip relay is energized. The output from each altitude trip circuit is a ground signal to using equipment.

(11) The dc altitude voltage from the receiver-transmitter is applied to the height indicator pointer drive comparator. The comparator compares this voltage with the voltage from the altitude pointer potentiometer. If there is a voltage difference, the comparator supplies an error signal to the servo-amplifier. The amplifier output signal drives the servomotor, which repositions the altitude pointer potentiometer, until the comparator output error signal is nulled. The potentiometer and altitude pointer are both geared to the servomotor. This causes the altitude pointer to be positioned to the proper altitude.

(12) The receiver-transmitter dc altitude voltage is also supplied to the height indicator adjustable trip comparator. The comparator compares this voltage with the voltage from the adjustable trip potentiometer. As long as the airplane altitude exceeds the altitude of the adjustable trip setting (MDA cursor), the trip relay and MDA light(s) will remain de-energized. When the airplane altitude is slightly less than the altitude of the adjustable trip setting, the trip relay will be energized causing the MDA light(s) to illuminate.

B. Self-Test Operation

(1) Self-test of the LRRA system may be initiated at either the captain's or first officer's height indicator. Relays and mode logic in the automatic flight control system (AFCS) accessory box control the LRRA self-test circuits. When the test switch is depressed, a ground signal is applied through a relay in the autopilot accessory box to the test circuits in the receiver-transmitter. The accessory box relay inhibits the self-test command when the autopilot is in the glide slope engage submode.

(2) Depressing the test switch energizes the test relay. One set of relay contacts routes a test signal from the calibration loop mixer to the receiver circuits where it is processed in the same manner as a normal altitude signal. The other set of contacts applies a dc voltage to test circuits in the AID calibration unit to bring the dc altitude to the predetermined test altitude. During self-test, each height indicator should display the test altitude indicated on Fig. 1.

C. Monitor, Fault Indicator, and Flag Operation

(1) The two dc altitude output voltages are applied to the comparator and if the two voltages do not agree within a specified tolerance, the comparator provides an output to the flag gates and COMP threshold sensor signifying improper operation.

(2) At altitudes above the operational range of the receiver-transmitter, the low level signal coupled between antennas through the external interconnecting coaxial cable is processed through the receiver circuits and detected in the antenna coupling monitor to check the operation of the antenna system. At altitudes within the operational range of the receiver-transmitter, the normal received signal is processed through the receiver circuits and detected in the AGC circuit. The AGC circuit functions to maintain receiver gain and provides a signal to the signal presence monitor. The output from the antenna-coupling monitor is applied to the SENS indicator threshold sensor and flag gate circuit. The output from the signal presence monitor is applied to the SENS indicator threshold sensor and flag alarm circuit.

(3) An output from the calibration loop mixer is applied to the power modulation monitor and altitude calibration circuits. If the monitor detects a loss in signal that exceeds tolerances, an output is provided to the flag gates and XMTR threshold sensor signifying improper operation. The calibration circuits will do the same if the amount of altitude correction required is beyond the limits of the self-calibration circuits.

(4) In the height indicator, the 10-kHz tracer signal to the servo-amplifier is processed to the integrity monitor when the servo is in a nulled state. The detected tracer signal maintains the flag amplifier in the conducting state allowing the flag solenoid to be energized (flag out of view). The tracer signal output from the integrity monitor is also supplied to the receiver-transmitter IND threshold sensor and flag gates. The absence of the tracer signal signifies improper height indicator operation, therefore, both height indicators must fail before the receiver-transmitter indicates a height indicator malfunction.

(5) The inputs to the flag gates must be of sufficient magnitude to inhibit a flag alarm condition. If any one of the inputs drops below a specific value, an output is supplied to the flag alarm circuits which turns off the flag driver causing the height indicator warning flag to come into view and the receiver-transmitter FLAG indicator to illuminate. In addition, the A/P warn relay is de-energized to provide warning signal to the autopilot system, deactivation of the altitude trip circuits, and illumination of the receiver-transmitter WARN indicator.

(6) An output from the flag alarm circuit is also provided to the pointer stow circuit. An alarm condition activates this circuit causing the height indicator altitude pointer to freeze in position for approximately 1 second and then to be driven up behind the pointer mask.

(7) When the airplane is at an altitude of over 2500 feet or for any condition when received signal strength is inadequate for normal system operation, an input is provided to the flag alarm circuit to de-energize the A/P warn relay. This results in immediate warning to the autopilot system, deactivation of all altitude trip circuits, altitude pointer stow, and illumination of the receiver-transmitter WARN indicator.

(8) The receiver-transmitter lights marked WARN (yellow) and FLAG (red) are controlled directly by the autopilot disconnect and flag alarm functions, respectively. The WARN lamp will light when there is a loss of signal and both the WARN and FLAG lamps will light if a malfunction occurs in the LRRA system. The four neon lights marked SENS, XMTR, COMP, and IND are controlled by the integrity gates. Power is applied to the neon lights only when the PRESS TO MONITOR button is pressed. Pressing this button will cause one or more of the neon lamps to light identifying the area in which a fault has occurred.

LOW RANGE RADIO ALTIMETER SYSTEM - TROUBLESHOOTING

1. General
   A. If a malfunction occurs in the radio altimeter system the defective component should be replaced with a component known to be in good operating condition. Excluding the malfunctions listed below in paragraph B., substitutions should always be in the following order.
      (1) If system is malfunctioning replace receiver transmitter with one known to be in good operating condition.
      (2) Perform operational check system. If the system is still malfunctioning replace height indicator. However both indicators are individually connected in parallel and malfunctioning can be determined by comparison.
      (3) If system still malfunctions, inspect antennas for damage, airplane wiring and all connectors for improper connections or damaged items.
   B. The following malfunctions are directly traceable to the height indicator. If the system operation is normal and one or more of the following malfunctions occur do not replace the receiver-transmitter. Instead, replace the height indicator and check its wiring.
      (1) Dial lighting (all malfunctions).
      (2) MDA trip light (all malfunctions).
      (3) MDA control knob or bug (all malfunctions).
      (4) Pointer visible when circuit breaker is pulled.
      (5) Flag not visible when circuit breaker is pulled.

LOW RANGE RADIO ALTIMETER SYSTEM - ADJUSTMENT/TEST

1. Low Range Radio Altimeter System Test
   A. Prepare to Test Low Range Radio Altimeter System
      (1) Provide electrical power.
      (2) Check that RADIO ALTM-1 circuit breaker on panel P18 is closed.
   B. Test Low Range Radio Altimeter System
      (1) At both height indicators, set MDA cursor between -15 and -20 feet and check that:
         (a) Both MDA lights are extinguished.
         (b) Altitude pointer on both height indicators indicates between 0 and -10 feet.
         (c) Warning flag on both height indicators is out of view.
      (2) At captain's height indicator, depress TEST button and check that:
         (a) Both MDA lights are extinguished.
         (b) Altitude pointer on both height indicators indicates 30 ±5 feet.
         (c) Warning flag on both height indicators is in view.
      (3) Release TEST button on the captain's height indicator and check that:
         (a) Both MDA lights are extinguished.
         (b) Altitude pointer on both height indicators returns to 0 to -10 feet indication.
         (c) Warning flag on both height indicators retracts from view.
      (4) Repeat steps 1.B.(2) and (3) using first officer's height indicator in place of captain's height indicator.
      (5) At captain's height indicator, depress TEST button, then slowly adjust MDA cursor to increasing altitude setting and check that:
         (a) Captain's MDA light(s) illuminates when MDA cursor setting is within ± 5 feet of displayed test altitude.

            NOTE: The panel-mounted MDA light will decrease in intensity approximately 1/2 second after it comes on.

      (6) Release TEST button and check that captain's MDA light(s) remains illuminated.
      (7) Adjust master dim control and check that intensity of panel-mounted MDA light is controlled.
      (8) At captain's height indicator, set MDA cursor to between -15 and -20 feet and check that captain's MDA light(s) goes out.
      (9) Repeat steps 1.B.(5) thru (8) using first officer's MDA light(s) and height indicator in place of captain's.
      (10) Pull RADIO ALTM-1 circuit breaker and check that:
         (a) Altitude pointer on both height indicators remains fixed in position.
         (b) Warning flag on both height indicators comes into view.

(11) Remove electrical power if no longer required.

LOW RANGE RADIO ALTIMETER ANTENNA - REMOVAL/INSTALLATION

1. General
   A. The following procedure applies to both transmit and receive antennas.
2. Remove Antenna (Fig. 401)
   A. Remove screws securing antenna to airplane structure.
   B. Lower antenna until coaxial connector(s) are accessible.

      NOTE: When lowering antenna, note direction of antenna connector(s) with respect to airplane centerline. This will identify orientation of antenna for installation.

   C. Disconnect coaxial cable(s) from antenna and remove antenna gasket.
3. Install Antenna (Fig. 401)
   A. Check condition of antenna gasket and replace if defective.

      NOTE: Gasket serves the dual function of providing an air pressure seal and conductive path for rf energy.

   B. Check antenna and retaining ring (airplane structure) faying surfaces for corrosion or other foreign material and clean if necessary.

      CAUTION: BE CERTAIN GASKET IS IN GOOD CONDITION AND FAYING SURFACES ARE CLEAN. INADEQUATE RF GROUNDING WILL CAUSE IMPROPER SYSTEM OPERATION.

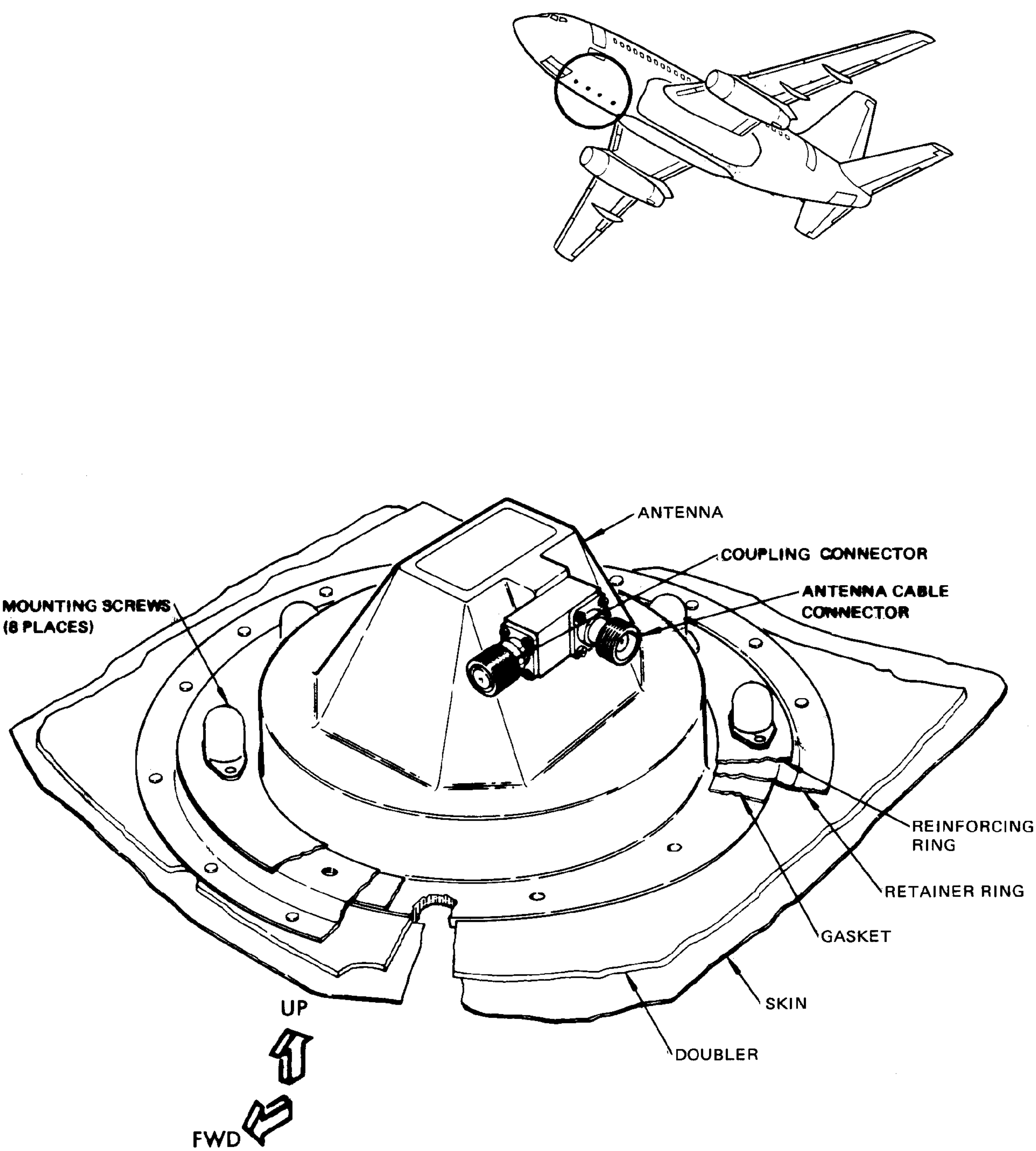
   C. Apply corrosion preventive material to antenna and retaining ring faying surfaces (Ref 51-21-91, Cleaning/Painting).
   D. Place gasket on antenna and align over keying pin (if installed).
   E. Connect coaxial cable(s) to antenna.
   F. Place antenna in mounting recess and align antenna so keying pin (if installed) mates with its positioning hole in retaining ring. If keying pin is not installed, align antenna so connector orientation is in same direction as connector(s) on other LRRA antenna(s).

      CAUTION: BE CERTAIN ANTENNAS ARE PROPERLY ALIGNED. SIGNAL COUPLING BETWEEN IMPROPERLY ALIGNED ANTENNAS WILL CAUSE SYSTEM OPERATIONAL PROBLEMS.

   G. Install screws securing antenna to airplane structure.

      NOTE: Small voids between antenna and airplane structure may be filled with aerodynamic sealer.

   H. Check dc resistance between antenna base and airplane skin. Resistance should not exceed 0.10 ohm.

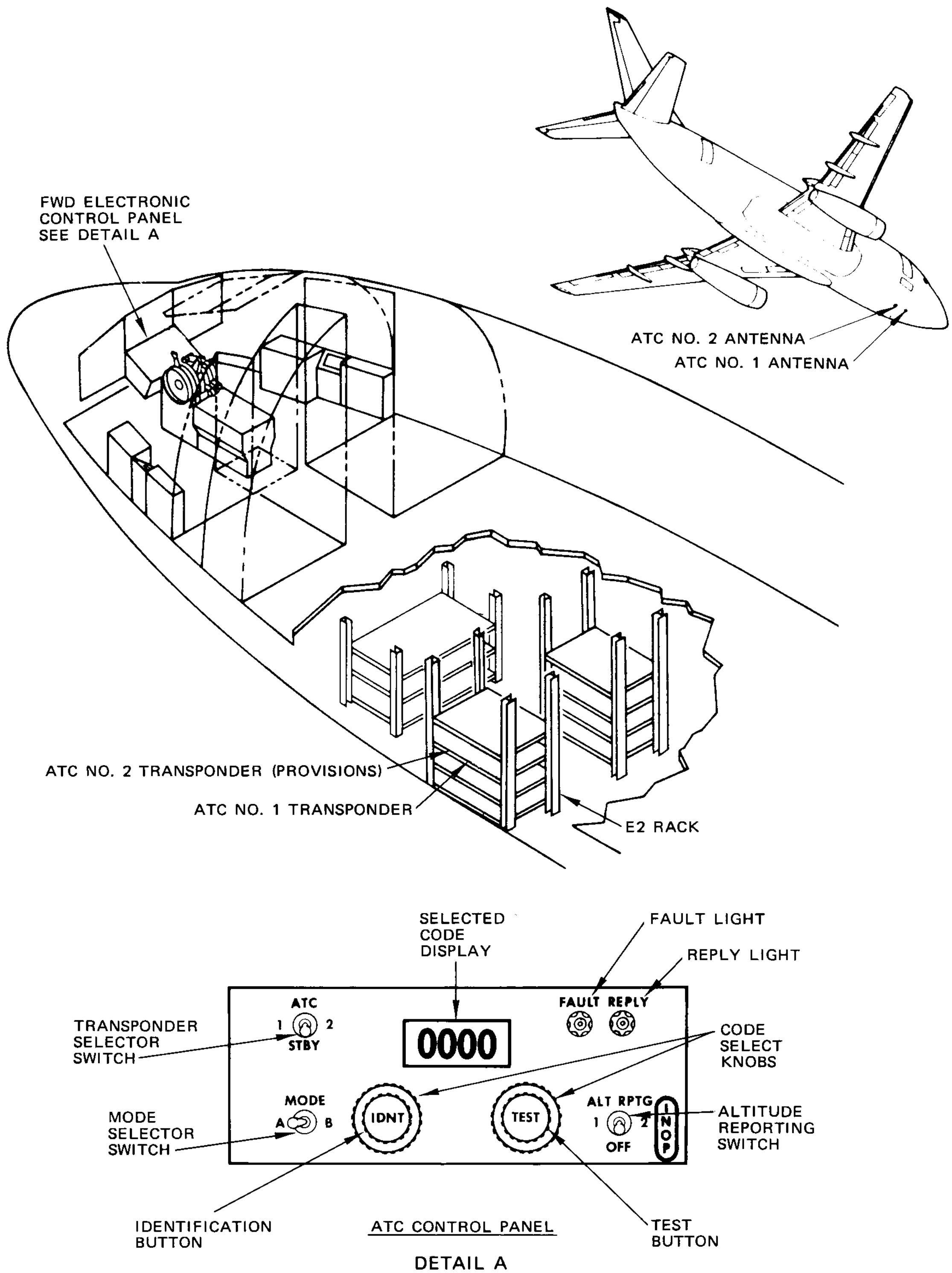


Low Range Radio Altimeter Antenna Installation
Figure 401

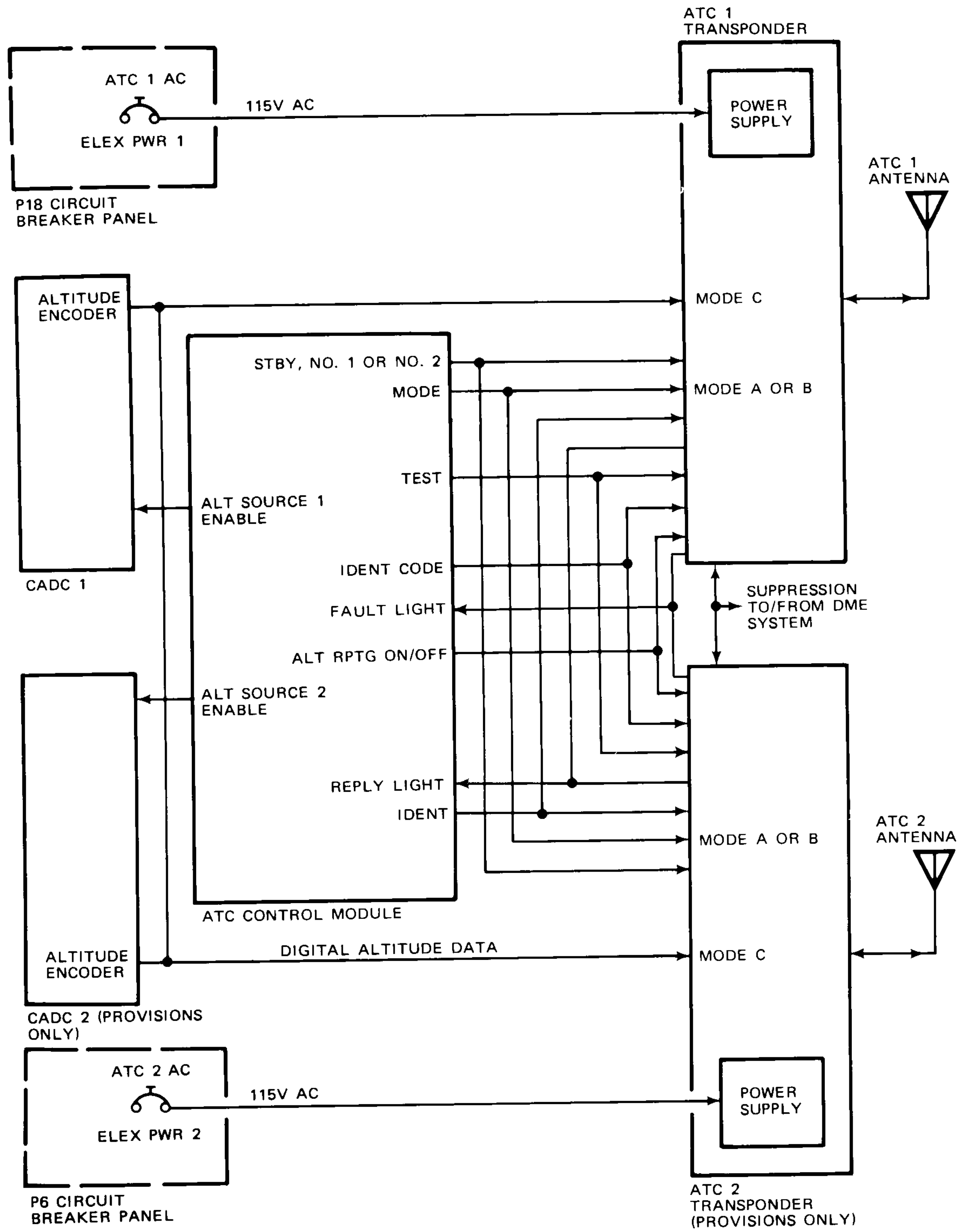
I. Perform operational check of system (Ref 34-48-0, Adjustment/Test).

ATC SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. AR LV-JMW thru LV-LEB;
      Provisions only are installed for two ATC (Air Traffic control) systems.
   B. ALL EXCEPT AR LV-JMW thru LV-LEB;
      One complete system is installed with complete provisions for the installation of a second ATC system (see Fig. 1 for equipment locations).
   C. The ATC system, when interrogated by a signal from a ground station, automatically transmits a coded signal to the ground station. This coded signal is then displayed on the ground station radar indicator and shows the identity of the aircraft as well as its range and bearing.
   D. The transponder receives digital altitude data from the air data computer (AMM 34-12-0). Complete wiring provisions are installed to allow digitized altitude data to be supplied from two air data computers (if installed), equipped with altitude digitizers.
   E. Interface between the ATC system and other airplane systems is shown on Fig. 2. To prevent interference between the ATC and DME systems, suppression signals are used to suppress one system when the other is transmitting.
   F. ATC No. 1 transponder receives 115-volt ac power through ATC-1 circuit breaker on P18 left load control center. ATC No. 2 transponder receives 115-volt ac power through ATC-2 circuit breakers on P6 right load control center.
2. ATC Controls
   A. Controls for the transponders are on a control panel located on the pilots' forward electronic control panel. The ATC control panel is common to both ATC No. 1 and 2 (if installed) transponders.
   B. The position of the function selector (if installed) switch determines the operational status: both transponders at standby; transponder No. 1 on, with No. 2 at standby, or transponder No. 2 on, with No. 1 at standby. A mode control switch permits the selection of mode A or The position of the ALT RPTG switch determines whether the altitude encoder is on or off and the ALT SOURCE switch selects CADC 1 or CADC 2 (if installed).
   C. Two pair of concentric rotary knobs are used to select the desired transponder reply code. The selected numerical code is displayed in a window directly above the knobs. Depressing the IDENT button causes the special position identification (SPI) pulse to be added to the normal mode A or B reply.
   D. The FAULT light will illuminate when a malfunction exists in the system. The REPLY light will illuminate when the system transmits a valid reply to an interrogation. Self-test of the system may be initiated by depressing the TEST button. If the transponder is working properly, the REPLY light will illuminate.

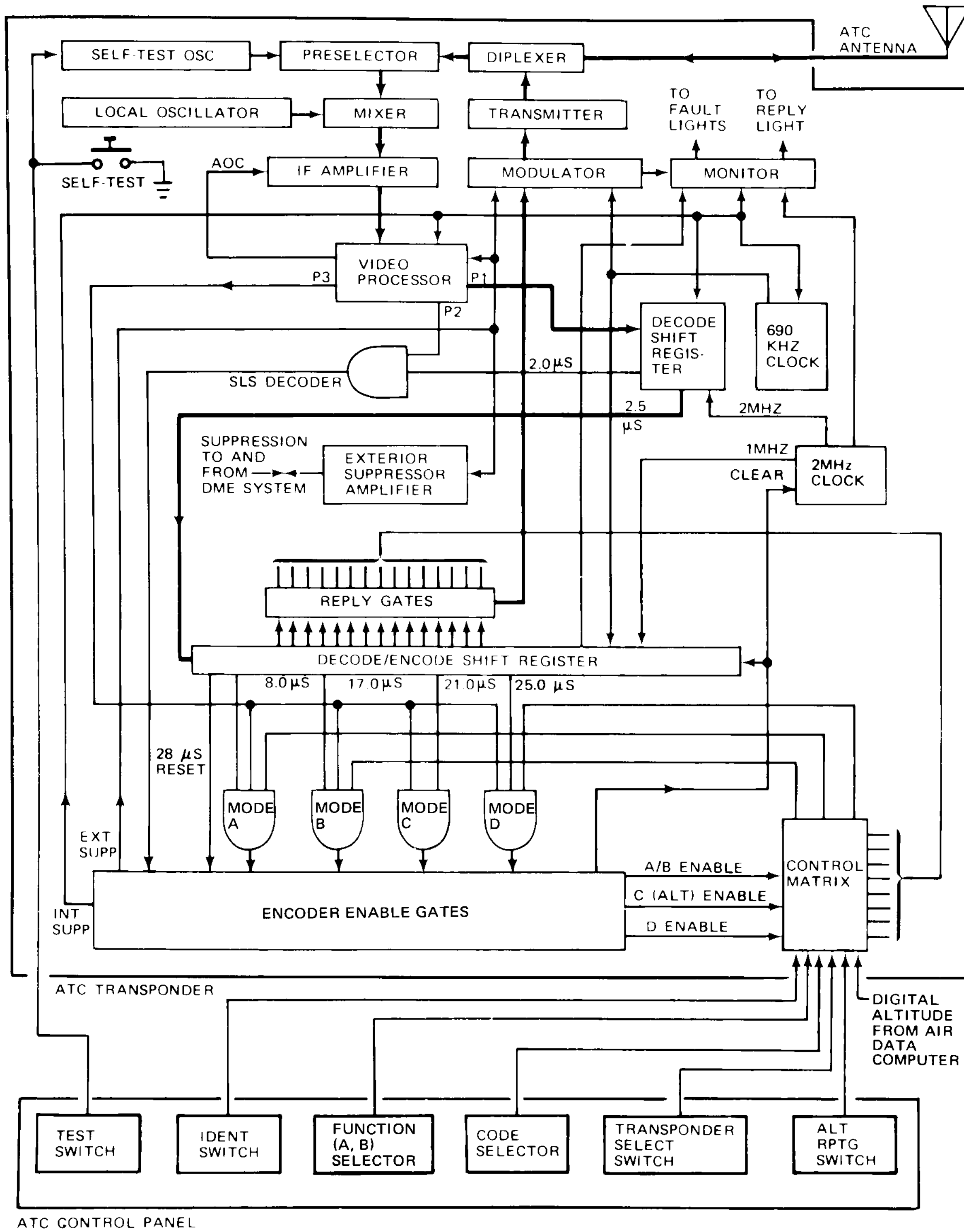


ATC System Component Location
Figure 1



ATC System Control and Interface
Figure 2

3. ATC Transponder
   A. All electrical connections to the transponder are made through a connector located on the lower portion of the rear panel. The connector located on the upper portion of the rear panel is used to connect automatic test equipment to the system.
   B. Controls and indicators mounted on the front panel of the transponder are a SELF-TEST button and reply light (combined), R/T and ANT fault indicators, and RESET button. The SELF-TEST button and reply light are functionally paralleled to the TEST button and REPLY light on the ATC control panel. The R/T and ANT fault indicators are ball-type indicators, which rotate to display the yellow side to indicate a malfunction. The black display indicates a no-fault condition. The RESET button is used to reset the ANT fault indicator to a no-fault condition.
4. Antenna
   A. The two ATC antennas are conventional blade-type units connected to the transponders by coaxial cable. Each antenna performs the transmitting and receiving functions for its respective transponder (if installed).
5. Operation (Fig. 3)
   A. Functional Description
      (1) The 1030 MHz interrogation signal received at the antenna is applied to the input of the diplexer. The diplexer routes the received signals to the preselector, the transmitted signals to the antenna, and provides receiver-transmitter isolation. This allows one antenna to be used for both receiving and transmitting.
      (2) The preselector contains three resonant cavities that provide rejection of images and other spurious responses. The output of the preselector is coupled-to the mixer where the signal is combined with the local oscillator signal to produce the IF signal. The 60-MHz IF signal is coupled through six stages of amplification. The output of the last stage is video detected, amplified and coupled to the video processor.
      (3) The video processor analyzes the video pulses to determine pulse amplitude, pulse width, and amplitude relationship between the P1 and P2 pulses of the interrogation signal. Decoder action is determined by the type of interrogation signal received. The P1 output of the video processor is applied to the shift register input timer. The input timer applies P1 to the decoder shift register in proper phase relationship with the 2-MHz clock signal. This pulse is shifted along the decoder shift register by the clock. At the 2-usec output, the pulse is applied to the side lobe suppression (sls) decoder.



ATC System Block Diagram
Figure 3

(4) For a side-lobe (invalid) interrogation, the video processor develops a P2 pulse output. The P2 pulse (following P1 by 2 usec) is applied to the other input of the sls decoder. The 2-usec delay generated by the decoder shift register results in the P1 and P2 pulses being applied to the sls decoder simultaneously. The pulses are applied to an AND gate and the resulting pulse is applied to the encoder enable gate as the sls start pulse. The encoder enable gate then generates an internal suppression signal. This signal inhibits the decoder for a period of approximately 30 usec to prevent further decoder action. At the conclusion of the internal suppression signal, the decoder is again enabled.

(5) For a valid mode interrogation, the P2 pulse is not present at the sls decoder and the P1 pulse is shifted on to the 2.5-usec output of the decoder shift register. The 2.5-usec output is applied to the shift register transition gate. This gate applies the output of the decoder shift register to the decode/encoding shift register and ensures that the pulse is in proper phase relationship with the 1-MHz clock signal. The 1-MHz clock signal shifts the P1 pulse along the decode/encode shift register to the 8-usec output, The pulse from the 8-usec output is applied to the mode A decoder. The P3 pulse in mode A interrogations follows P1 by 8 usec. This pulse is applied to another input of the mode decoder. The 8-usec delay generated by the decoder shift register and the decode/encode shift register results in the (delayed) P1 and P3 pulses being applied to the mode A decoder simultaneously.

(6) When the function selector on the control panel is set to the mode A position, an enable signal is applied to the third input of the mode A decoder. The P1, P3, and mode A enable signals are applied to the mode A decoder AND gate and the resulting output pulse (mode A start) is applied to the encoder enable gate and the control matrix. This pulse initiates the encode mode of operation. The mode A start pulse applied to the control matrix enables the replay gates that correspond to the code selected at the control unit.

(7) When a valid mode B, C, or D interrogation signal is received, decoder operation is similar to the mode A decoder operation. Each mode of interrogation uses its own decoder gate and decode/encode shift resister output. The shift register delay period for modes B, C, and D are 17, 21, and 25 usec respectively.

(8) At the completion of the decode mode of operation, a mode start pulse is applied to the encoder enable gate from one of the decoders. Encoder operation is determined by the type of mode of interrogation signal received.

(9) When a side-lobe interrogation signal is received and encoded, an sls start pulse is applied to the encoder enable gate. The sls start pulse causes the sls enable output signal to be developed. The sls enable signal, applied to the suppressor gate, generates an internal suppression signal. This output is applied to the decoder and clear generator circuits. The internal suppression signal applied to the decoder prevents further decoder action and enables the 690-kHz clock output. The suppression signal applied to the clear generator produces a signal pulse. This pulse is applied to the decode/encode shift register as a clear pulse. The pulse clears the register of any remaining data and inputs a single data bit into the first register element. The 690-kHz clock signal is applied to the clock terminals of the register elements to shift the data bit through the register. The shift register output pulse (delayed 28 usec in the register) is applied to the encoder enable gate as a reset pulse. This pulse removes the sls signal from the encoder gate, causing the internal suppression signal to be inhibited. With the internal suppression signal inhibited, the transponder returns to the decode mode of operation.

(10) When a valid mode A interrogation signal is received and decoded, a mode A start pulse is applied to the encoder enable gate. The resulting A/B enable signal from the encoder enable gate is applied to the reply matrix and the suppressor gate. The enable signal and the code control signals from the control panel are applied to an AND gate in the reply matrix. The resulting signals are applied to the reply gates as code enable signals. The code enable signals are applied only to the code reply gates corresponding to the reply code selected at the control panel. No enable signal is applied to the remaining reply gates, thus unused code reply gates are inhibited.

(11) The A/B enable signal applied to the suppression gate generates the external suppression and internal suppression signals. The external suppression signal enables the modulator and inhibits the DME system to prevent interference. The internal suppression signal is applied to the decoder and the clear generator. The signal applied to the decoder prevents further decoder action and enables the 690-KHz clock signal. The internal suppression signal applied to the clear generator produces a single clear pulse that is applied to the decode/encode shift register. The pulse clears any information remaining in the register and inputs a single data bit into element one.

(12) The data bit is shifted along the register by the 690-KHz clock signal. As it is being shifted through the register, the bit is applied to the input of each of the reply gates. The shift register outputs and the reply matrix outputs (code enable signals) are applied to a NAND gate in the enabled relay gates. The resulting code pulses, corresponding to the code selected at the control unit, are applied to the modulator. When the data bit is shifted to the 28-usec output, a pulse is applied to the encoder enable gate. This pulse resets the A/B encoder gate, inhibiting the A/B enable signal. With this signal inhibited, no suppression signals are generated and the transponder returns to the decode mode of operation.

(13) Encoder action when a mode-B interrogation signal is received is similar to encoder action when a mode A interrogation is received. The difference is the start pulse and enable signals. These are determined by the mode of interrogation signal received and decoded.

(14) As the single data bit is applied to the reply gates, a pulse is applied to the modulator send monitor for each reply gate enabled by the control matrix. The modulator converts the pulses from the reply gates to high voltage pulses to drive the transmitter tube. At the same time, the automatic overload control (aoc) counts the number of pulses applied. If either the number of reply gate pulses or ext suppr pulses exceed a preset level, an aoc signal is generated and applied to the IF amplifier. This signal reduces the overall sensitivity of the IF amplifier. Thus, weaker signals then do not initiate a reply.

(15) The ATC control panel contains a reply light and fault light; also, the front panel of the transponder contains a reply light included in the self-test switch and two fault indicators (R/T and ANT). The monitor circuit supplies signals to these indicators to provide an indication of the operational status of the ATC system. When three consecutive replies are completed, the reply lights are energized. When a fault occurs, a voltage output from the monitor energizes one of the two fault indicators on the transponder and the fault light on the ATC control panel. The fault light will remain on for 15 seconds longer than the fault condition exists. The ANT fault indicator must be reset manually by depressing the RESET switch on the transponder. The R/T indicator must be reset by grounding a terminal inside the transponder. If the fault condition clears and the system completes three consecutive replies, the reply lights will come on as in normal operation but the fault indicators will continue to display the fault until manually reset.

(16) The R/T monitor circuits check the transponder for proper output power, frequency, mode decoding, clock frequency, and shift-register fill-up. The ANT monitor circuits check the dc resistance of the antenna and coaxial cable. If the circuits detect a dc resistance greater than 200 ohms, the ANT fault indicator will be activated.

(17) Depressing the SELF-TEST switch on the transponder or the TEST button on the ATC control panel causes a signal to be generated by the self-test oscillator. The self-test signal is injected into the preselector. From this point it is processed as a valid interrogation signal. If no fault exists, a reply is generated and the reply lights are energized as if a valid interrogation signal had been received and processed.

(18) The transponder may be self-tested in mode A or B by selecting that mode on the ATC control panel. Mode C will be automatically self-tested every other self-test interrogation cycle if the altitude-reporting switch is in the ALT RPTG position.

(19) The special position identification (SPI) pulse is initiated by depressing the IDENT button on the control panel. This causes the SPI pulse to be transmitted in every reply from the time the IDENT button is depressed until approximately 20 seconds after the button is released.

(20) The transponder will automatically operate in identity mode A unless an external ground is supplied from a control panel to select mode B or D. Mode A or B may be selected on the ATC control panel. Mode D is presently unassigned.

(21) The transponder will automatically reply to a valid mode C (altitude-reporting) interrogation if the altitude-reporting switch is in the ALT RPTG position. Altitude reporting is independent of mode A or B selection. The altitude-reporting function may be disabled by setting the altitude-reporting switch to the OFF position. The transponder select switch, in addition to selecting transponder No. 1 or 2, enables the altitude digitizer in the air data computer allowing the digital altitude data to be applied to the control matrix in the transponder.

ATC TRANSPONDER SYSTEM - TROUBLESHOOTING

1. General

A. Troubleshooting the ATC system is best accomplished by performing the ATC System - Adjustment/Test. Satisfactory completion of the adjustment/test procedures indicates that the system is operating properly. When satisfactory operation is not obtained, the tabulated data below should be consulted. If using the tabulated data fails to correct the trouble, airplane wiring should be suspect and repaired or replaced as necessary.

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
System completely inoperative	Circuit breaker	Check that system circuit breakers on P18 (and P6 if both systems installed) circuit breaker panel is closed	Close circuit breaker
	No power available to transponder	Check that 28 volts dc is available at pin 31 and 115 volts ac is available at pin 29 of plug connected to transponder	Refer to AMM Chapter 24, Electrical Power
Monitor light does not come on	Transponder or control panel	With system turned on and appropriate transponder selected, place test switch (if provided) in TEST position. If monitor light comes on, transponder faulty. If monitor light does not come on, control panel faulty	Replace transponder. Replace control panel

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
System does not operate properly in any mode	Transponder		Replace transponder
Unable to select code on control panel	Control panel		Replace control panel

ATC SYSTEM - ADJUSTMENT/TEST

1. General
   A. The ATC system may be tested using ATC Transponder Ramp Test Set, Collins 476X-3, Instrument Flight Research ATC-600A or Tel-Instruments T-33B. If Collins 476X-3 Test Set is used, proceed to par. 2. If T-33B Instrument Flight Research ATC-600A is used, proceed to par. 3. If Tel-Instruments T-33B is used, proceed to par. 4.
2. ATC System Test Using Collins 476X3
   A. Equipment and Materials
      (1) ATC Transponder Ramp Test Set - Collins 476X-3
      (2) Portable pressure equipment to simulate 0 to 35,000 feet altitude
   B. Prepare to Test ATC System

      NOTE: The test set is a source of interference for radio and L-band radar equipment operating on airplanes located in the vicinity of the test set. Turn test set off as soon as the test is completed or when other radio checks are being performed on the airplane.

      (1) Place test set mode selector switch in ABC position. Check that mode flags indicate A, B, and C in sequence.
      (2) Place test set mode selector switch in SELF TEST and T/R ATTEN switch in the 25-dB position (make sure that battery is charged).
      (3) Place CALIBRATE OUTPUT/BATTERY VOLTAGE switch in CALIBRATE OUTPUT position. Adjust OUTPUT control for red line indication on meter.
      (4) Perform tests in following table.

CODE SELECTOR POSITION	FLAG INDICATION
0000	CODE and IDENT Steady
7777	CODE and IDENT Steady
7773	CODE Blinking; IDENT Steady
0004	CODE Blinking; IDENT Steady

      (5) Place 476X-3 ramp test set at a distance of between 40 and 50 feet from the ATC antenna in a position easily visible from the cockpit, and with an unobstructed radiation path between the tester and the antennas.

(6) Orient tester such that inside of opened cover (antenna element) faces airplane antennas.
(7) Make sure that the system and altitude source (Altimeter or CAC) circuit breakers on panels P6 and P18 are closed.

C. Test ATC System
(1) Position controls on test set as follows:
(a) Function Selector - OFF
(b) R/T ATTN dB - 0
(c) Code Switches - Any desired code
(2) Position controls on ATC control panel as follows:
(a) ALT RPTG - OFF
(b) Code - Same as on test set
(3) Set up and operate system in mode A. Verify REPLY/MONITOR remains off.
(4) After system warm up, activate ATC system test. Verify REPLY/MONITOR light comes on.
(5) On test set, position function selector to A-B-C. Observe that code flags appear each time mode A flag appears.
(6) During time interval when mode A flag is in view, activate IDENT on ATC control panel. Verify a blinking indication of both IDENT and code flags.
(7) Test altitude reporting as follows:
(a) On ATC control panel, select ALT RPT and ALT source.
(b) Connect air data test set to PITOT/STATIC system.
(c) Adjust altimeter to 29.92 inches of mercury.
(d) Using air data test set, apply a static pressure to obtain altitude reading on PITOT/STATIC test set as specified in following table. For each altitude in table, set corresponding code on 476x3 test set. Verify mode C code flag appears each time mode C flag appears and altimeter indicates ± 100 feet of test altitude.

ALTITUDE REPORTING TEST ALTITUDES	
ALT Feet	CODE
1,000	0320
4,100	4730
15,700	3440
31,000	1024

(8) Restore airplane and test equipment to original configuration.
(9) If no longer required, remove electrical power from airplane.

3. ATC System Test Using Instrument Flight Research ATC-600A

A. Equipment and Materials
(1) Transponder/DME Test Set - IFR Model ATC-600A
(2) Portable pressure equipment to simulate 0 to 35,000 feet altitude and 0 to 400 knots airspeed

B. Prepare to Test ATC System

NOTE: The test set is a source of interference for radio and L-band radar equipment operating on airplanes located in the vicinity of the test set. Turn test set off as soon as the test is completed or when other radio checks are being performed on the airplane.

(1) Position remote test set antenna approximately 21 inches from ATC antenna for system under test. Test set antenna should be approximately same height as the airplane antenna.

CAUTION: NEVER PLACE REMOTE TEST SET ANTENNA CLOSER THAN 15 INCHES TO THE AIRCRAFT ANTENNA WITH TEST SET "ON". DAMAGE TO THE TEST SET WILL RESULT.

(2) Route and connect coax from remote test set antenna to Test set.
(3) Position controls on test set as follows:
(a) PWR Switch - OFF
(b) Mode Switch - A/C, Code
(c) SLS Switch - Center position
(d) Framing Control - 0
(e) INTERR Control - 0
(f) XPDR SIG Control - FULLY CCW
(g) XMTR FREQ Control - 0

C. Test ATC System

CAUTION: IN THE FOLLOWING TESTS DO NOT SELECT CODES 3100, 7500, 7600, OR 7700. THESE ARE EMERGENCY CODES.

NOTE: In the following tests the reply light on the ATC control panel shall illuminate whenever the ATC transponder transmits a reply to the test set, unless otherwise stated.

Also, the % reply meter on the test set shall indicate 100% when the transponder is replying (unless specified otherwise).

(1) Close these circuit breakers:
   (a) ATC-1 P18 Panel
   (b) ATC-2 P6 Panel (If installed)
   (c) Altitude Source (ALTIMETER or CADC)
(2) Activate ATC system No. 1 in Mode A with ALT RPTG off and code 0000 selected.
(3) After system warmup, hold TEST/MONITOR switch on ATC control panel in TEST. Make sure that the REPLY light comes on.
(4) Place PWR switch on test set to AC or BAT as appropriate. Test set display should indicate 0000.

   NOTE: For information regarding battery testing, timing and recharging refer to operation section of ATC-600A test set operators manual.

(5) Test receiver sensitivity as follows:
   (a) On test set, verify % REPLY meter indicates 100.
   (b) Slowly rotate XPDR SIG control clockwise until % REPLY meter indicates 90. The XPDR SIG control should be between 69 and 77 (note control reading, this is transponder system minimum triggering level).
   (c) Select ALT RPTG on ATC control panel and A/C ALT on test set.
   (d) Repeat step (b). XPDR SIG control reading shall not differ from reading in step (b) by more than 1.
   (e) Return XPDR SIG control fully counterclockwise. Position test set mode switch to A/C CODE.
(6) Test Side Lobe Suppression as follows:
   (a) Adjust the XPDR SIG control on test set to 3 dB above minimum triggering level (the control reading in step 5.(b) minus 3).

      NOTE: If the transponder sensitivity is near the insensitive limit, it may be necessary to move antennas closer together in order to get 3 dB above minimum triggering level. At closer than normal antenna spacing, re-establish the dial setting for MTL and adjust the dial 3 dB above MTL and make the SLS test.

(b) Position SLS switch on test set to 0 dB. Transponder shall stop all replies.
(c) Position SLS switch on test panel to -9 dB. Transponder shall reply to at least 90% (minimum) to 100% as indicated on test set % REPLY meter.
(d) Turn test set XPDR SIG control fully counterclockwise.

(7) Test transmitter frequency as follows:
(a) Set test set FREQ/POWER meter switch to FREQ.
(b) On ATC control panel, select code 0000 and position ALT RPTG switch to OFF.
(c) Adjust gain control on test set for a mid-scale indication of FREQ/POWER meter. Rotate frequency control for a peak indication of meter. Frequency control dial shall read 0 ±3.

(8) On ATC control panel, rotate code selector knobs to any code desired. Test set numerical display should readout selected code.

(9) Change code selection on ATC control panel to test code compliment. Compliment of test code is 7777 minus test code being used. This checks full code capability of transponder under test.
(a) EXAMPLE:
1) If test code being used is 0340, compliment is 7777.
2) If test code being used is -0340, compliment is 7437.

(10) On ATC control panel, activate IDENT. IDENT lamp on test set should come on.

(11) Test altitude reporting as follows:
(a) On ATC control panel, select ALT RPT and ALT source 1.
(b) Connect air data test set to captain's (No. 1) PITOT/STATIC system.
(c) Adjust captain's altimeter to 29.92 inches of mercury.
(d) Position test set mode switch to A/C ALT.

NOTE: In A/C ALT position, numerical display of test set will indicate airplane altitude in thousand feet, being reported by transponder.

NOTE: In following tests ALT INVALID light shall remain off. When on, this light indicates an unassigned altitude code is being transmitted.

(e) Using air data test set, apply a static pressure to obtain altitude reading on test set as specified in following table. Make sure that the captain's altimeter indicates ± 100 feet of test altitude.

TABLE - ALTITUDE REPORTING TEST ALTITUDES	
ALT Feet	CODE
1,000	0320
4,100	4730
15,700	3440
31,000	1024

(12) Do the Test ATC System steps for the No. 2 ATC transponder.

4. ATC System Test using Tel-Instruments TIC T-33B Test Set

A. Equipment and Materials

(1) ATC Transponder Ramp Test Set - Tel-Instruments T-33B

(2) Pitot-Static Test Kit

B. Prepare to Test ATC System

NOTE: Steps B.(1) thru C.(8) are to be performed for receiver sensitivity checks. Otherwise, perform steps C.(9) thru C.(28).

(1) Remove No. 1 ATC antenna (AMM 34-53-11).

(2) Using a T-33B tester ONLY, connect antenna end of transmission line to tester as follows:

(a) Connect 6-foot test cable to BNC jack on tester.

CAUTION: NEVER CONNECT TEST CABLE TO TNC CONNECTOR ON TESTER AS TESTER WILL BE DAMAGED. ALSO, ANTENNA (IN COVER) MUST BE DISCONNECTED FROM TNC JACK AND PROTECTOR CAP MUST BE INSTALLED ON THIS TNC CONNECTOR TO PREVENT RADIATION OF SIGNALS.

(b) Using proper between-series adapter, connect other end of test cable to ATC antenna transmission line.

(3) Position test set controls as follows:
(a) RCVR SENS - Mid Scale
(b) 1090 MHz - Zero
(c) REPLY/SLS - 100%
(d) MODE - A
(e) READOUT - PILOT

(4) Turn on test set. Ascertain BATTERY CHECK meter is in white portion of scale. If not, recharge battery from a 115V AC 50-400 Hz source.

NOTE: It is permissible to operate test set while battery is being charged.

C. Test ATC System
(1) Close following circuit breakers:
(a) ATC-1 P18 panel
(b) ATC-2 P6 panel
(c) Altitude Source (ALTIMETER or CADC)

(2) Activate ATC system No. 1 in Mode A with ALT RPTG off and code 0000 selected.

(3) On test set, adjust RCVR SENS control for a meter reading of 90%. Sensitivity as read on dial must be -73 ±4 DBM.

(4) Select ALT RPTG on cockpit ATC control unit and MODE C on test set.

(5) Repeat step (3). Sensitivity must be within 1 dB of MODE A sensitivity.

(6) Disconnect test set.

(7) Connect and re-install No. 1 ATC antenna.

(8) Repeat steps B.(1) thru C.(7) for the No. 2 ATC system (if installed).

NOTE: Following tests will suffice for all but receiver sensitivity tests (refer to NOTE preceding step B.(1)).

(9) Perform steps C.(1) and C.(2).

(10) After system warmup, hold TEST switch on ATC control panel in TEST. Make sur

(11) Position controls on test set as follows:
- (a) RCVR SENS - 83
- (b) REPLY/SLS - LEVEL
- (c) 1090 MHz - 0
- (d) MODE - 3/A
- (e) Power - ON

(12) Check that test set meter deflects full scale and readouts display FR1 and FR2 indicating a reply from the transponder.

(13) On ATC control panel, rotate code selector knobs to any code desired. Make sure that the test set numerical display indicates selected code.

(14) On ATC control panel, activate IDENT. A decimal point should appear in test set display.

NOTE: If correct indications do not appear, it may be necessary to change position of test set to achieve sufficient signal strength. To assure that the test set antenna is coupling to the transponder antenna, the 1090 MHz control can be adjusted + or - from zero to adjust test set frequency to transponder frequency. Limit of adjustment is ± 6 MHz.

(15) On ATC control panel, change code selection to code compliment. Compliment of test code is 7777 minus code being used. This checks full code capability of transponder.

EXAMPLE:	
	If test code being used is 0340, compliment is 7777
	If test code being used is -0340, compliment is 7437

(16) On test set, adjust RCVR/SENS control so meter deflection is about 1/4 scale. Adjust 1090 MHz control above and below 0 for MAXIMUM meter deflection. Maximum error permissible is ± 3 MHz.

(17) On test set, position REPLY/SLS switch to 100% position. Test set meter now indicates reply rate as a percentage of interrogation rate, with full scale of 100%.

NOTE: A properly operating transponder must reply with a rate of 90% or greater.

(18) To check SLS circuits of transponder, position test set REPLY/SLS switch to 10%/0 dB. The meter, with full scale equal to 10%, should read 1.3% or less for a properly operating transponder. Note test set display extinguishes, indicating transponder is not replying to interrogations.

CAUTION: SOME TRANSPONDERS, WHEN RECEIVING A SIGNAL NEAR MTL (MINIMUM TRIGGERING LEVEL), MAY TRANSMIT A REPLY EVEN THOUGH SLS PULSE IS PRESENT IN INTERROGATION. IF A REPLY IS RECEIVED WITH SLS PULSE AT 0 DB, REPEAT SLS TEST WITH TEST SET PLACED CLOSER TO TRANSPONDER ANTENNA TO INCREASE SIGNAL STRENGTH.

(19) Position test set SLS switch to -9 dB and READOUT switch to PILOT. Note presence of ATC code and framing pulses in display.

(20) On ATC control panel, select ALT RPTG and altitude source No. 1.

(21) Connect pitot-static test kit to captain's (No. 1) PITOT/STATIC systems.

(22) Adjust captain's altimeter to 29.92 inches of mercury.

(23) On test set, position MODE switch to mode C, and READOUT switch to ALTITUDE. Hold POWER switch ON, readout windows should indicate field elevation as indicated by altimeters ± 75 feet.

NOTE: READOUT switch may be changed to PILOT position and test set will indicate altitude transmission code for indicated altitude. READOUT selector may be placed in BINARY position and will indicate actual number of A-B-C-D pulses transmitted.

NOTE: The tester will add 400,000 feet to altitude if any invalid C pulses are present (C0 - C5 - C7), 200,000 feet if any invalid D pulses are present (D1 - D3 - D7), and 600,000 feet if both erroneous C and D bits are present.

NOTE: When there is no Mode C information, such as would occur with a disconnected digitizer, test set altitude readout will be 4007. This can be verified by setting readout switch to PILOT. A display of 0000 will be indicated.

(24) Using air data test set, apply a static pressure to obtain altitude reading on test set as specified in following table. Make sure that the captain's altimeter indicates ± 100 ft of test altitude.

ALTITUDE REPORTING TEST ALTITUDES	
ALT Feet	CODE
1,000	0320
4,100	4730
15,700	3440
31,000	1024

(25) Repeat step (19) thru (23) with ALT RPTG source switch on ATC control panel placed in No. 2 position (if provided).
(26) Do the Test ATC System steps for the Test No. 2 ATC system.
(27) On ATC control panel position function selector to OFF or STBY as appropriate.
(28) Secure ramp test equipment.
(29) Determine whether there is further need for external power on airplane and if not, remove external power.

5. System Test - ATC System, Using Instrument Flight Research ATC-601

A. General
(1) The system test is a more complete check of the ATC system. The system test first does the ATC Operational Test. Then it uses a ramp test set to examine the left and the right ATC system.
(2) This system test satisfies the requirements in FAR 91.413 appendix F of part 43, which calls for the periodical check of the ATC transponder, without removing the transponder from the airplane.

B. Equipment and Materials
(1) Transponder Test Set - IFR Model ATC-601
(2) Air Data Test set - Castleberry AT700-2, Castleberry Instr & Avionics, Austin, TX, or equivalent

C. Reference
(1) AMM 34-11-0/501, Pitot-Static System

D. Prepare for the System Test

CAUTION: DO NOT PLACE THE REMOTE TEST SET ANTENNA CLOSER THAN 15 INCHES (.40 METERS) TO THE AIRCRAFT ANTENNA WITH THE TEST SET ON. THIS WILL CAUSE DAMAGE TO THE TEST SET.

NOTE: The test set is a source of interference for radio and L-band radar equipment. Turn the test set off as soon as the test is completed or when other nearby airplanes do radio checks.

(1) Put the remote test set antenna between 6 and 300 feet (2 and 90 meters) from the ATC antenna for the related system.

(2) Put the test set antenna in position towards the aircraft antenna.
(3) Connect the coax cable from the remote test set antenna to the test set.
(4) Set the PWR switch on the test set to the ON position.

NOTE: The test set is a source of interference for radio and L-band radar equipment operating on the airplane and located near the test set. Turn the test set off as soon as the test is completed or when you must perform other radio checks on the airplane.

(5) Push the SETUP key to enter the SET UP menu.

NOTE: For information about the battery test, timing and recharging refer to the operation section of the test set operators manual.

(6) Enter the HEIGHT and RANGE (in feet) of the ATC antenna.
(a) Use the slew keys to change the values.
(b) Use the select keys to change the items.
(7) Enter the gain listed on the test set antenna in to the GAIN_1030 and GAIN_1090 field.
(8) Enter the cable loss listed on the cable in the LOSS field.

E. Test ATC system

CAUTION: DO NOT USE CODES 7500,7600-7677, OR 7700-7777. THESE ARE EMERGENCY CODES.

(1) Set the switches on ATC control panel as follows:
(a) Transponder Select - Applicable system
(b) Altitude Reporting - OFF
(c) Code Select - 0000
(2) Push the AUTO TEST key on the test set.
(a) Make sure the results of the last auto test are shown.

F. Reply Frequency Test
(1) Do a check of the Reply Frequency.
(a) Make sure the reply frequency of the transponder is 1090 ±3 MHz.

(2) Do a check of the Reply Delay.
  (a) Make sure the reply delay is 3.00 ±0.50 us for ATCRBS mode.
(3) Do a check of the Reply Jitter.
  (a) Make sure the reply jitter is < 0.06 us for all call.
  (b) Make sure the reply jitter is < 0.1 us for ATCRBS modes.
(4) Do a check of the ATCRBS Reply.
  (a) Make sure the spacing of the F1 to F2 pulse is 20.3 ±0.10 us.
  (b) Make sure the duration of the F1, F2 pulse is 0.45 ±0.10 us.

G. Suppression Test
(1) Do a check of the Side Lobe Suppression (SLS).
  (a) Make sure the reply is received when the SLS pulse is -9 dB and no reply is received when the SLS pulse is 0 dB.

NOTE: Run the SLS level test in less than 95 feet (28.96 meters) of the UUT antenna.

H. Sensitivity Test
(1) Do a check of the Minimum Triggering Level (MTL) Difference.
  (a) Make sure the Minimum Triggering Level (MTL) difference between mode A and mode C is < 1.0 dBm
(2) Do a check of the Minimum Triggering Level (MTL).
  (a) Make sure the Minimum Triggering Level (MTL) is -73 ±4 dBm.

I. Peak Output Power Test
(1) Do a check of the Peak Output Power.
  (a) Make sure the peak power output of the transponder is between 125W (ERP = 51.0 dBm) and 500W (ERP = 57.0 dBm).

J. Transponder Code Capability Test

CAUTION: DO NOT USE CODES 7500,7600-7677,7700-7777. THESE ARE EMERGENCY CODES.

(1) Set the code select switch on the ATC control panel to an arbitrary code.
  (a) Make sure the ATC test set shows PASSED and the selected ATC code is shown on the code field.

(2) Push and release the IDENT switch on the ATC control panel.
(a) Make sure the ATC test shows ID in the code field for 15 to 30 seconds.

K. Altitude Reporting Test
(1) On ATC control panel, select ALT RPT and ALT source.
(2) Connect air data test set to captain's (No. 1) PITOT/STATIC system (AMM 34-11-0/501).
(3) Adjust captain's altimeter to 29.92 inches of mercury.
(4) Position test set mode switch to A/C ALT.

NOTE: In A/C ALT position, numerical display of test set will indicate airplane altitude in thousand feet, being reported by transponder.

NOTE: In following tests ALT INVALID light shall remain off. When on, this light indicates an unassigned altitude code is being transmitted.

(5) Using air data test set, apply a static pressure to obtain altitude reading on test set as specified in following table. Make sure that the captain's altimeter indicates ± 125 feet of test altitude.

TABLE - ALTITUDE REPORTING TEST ALTITUDES

ALT Feet	CODE
1,000	0320
4,100	4730
15,700	3440
31,000	1024

(6) Repeat steps K.(1) thru K.(5) with ALT RPTG No. 2 (if installed).

L. Repeat steps D. thru K. for No. 2 ATC transponder (if installed).

AIR TRAFFIC CONTROL (ATC) ANTENNA - REMOVAL/INSTALLATION

1. General
   A. The procedure has a removal, installation, and test of the antenna.
2. Equipment and Materials
   A. Equipment
      (1) Sealing gun - 6-inch length cartridge, Senco Research, or equivalent (not required for material furnished in tubes)
      (2) Bonding Meter (AMM 20-22-01)
      (3) Varnish brush - 1- or 2-inch
      (4) Spatula
   B. Materials
      (1) Aerodynamic smoother - class B-1/2
      (2) Accelerator - as indicated on aerodynamic smoother container
      (3) Aliphatic Naphtha - TT-N-95
      (4) Anticorrosion Compound - BMS 3-23
3. Remove Antenna
   A. Remove four mounting screws (Fig. 401).
   B. Using care, lower antenna from mounting until electrical cable can be disconnected.

      CAUTION: DO NOT PULL ANTENNA.

   C. Disconnect antenna cable from antenna connector.
   D. Remove antenna and O-ring.
4. Install Antenna
   A. Remove aerodynamic smoother from antenna base and mounting area (AMM 51-31-0/201).

      CAUTION: OBEY THE INSTRUCTIONS IN THE PROCEDURE TO REMOVE THE AERODYNAMIC SMOOTHER. IF YOU DO NOT OBEY THE INSTRUCTIONS, DAMAGE TO THE AIRPLANE SURFACE CAN OCCUR.

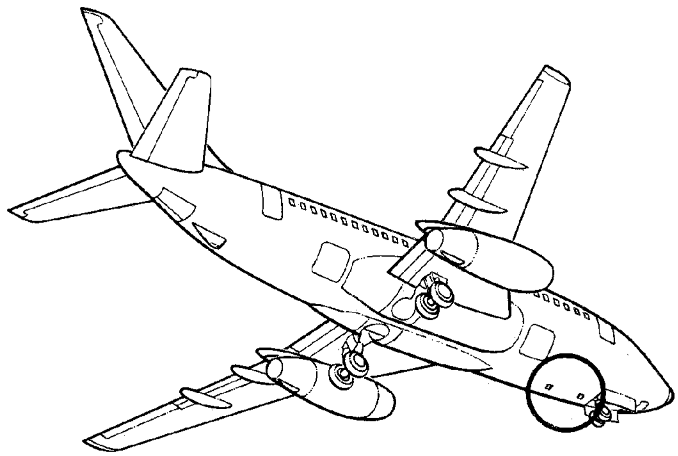
   B. Remove grease, oil, dirt, and chips from antenna base and mounting area. Use small varnish brush to apply fresh aliphatic naphtha for cleaning. Wipe cleaner off with clean cloth.
   C. Apply a protective coat of corrosion preventive compound to faying surfaces of antenna and airplane structure (AMM 51-21-91/701).
   D. Place O-ring in position on antenna base.
   E. Connect antenna cable to antenna connector (Fig. 401).
   F. Position antenna in place and install four mounting screws.
   G. Check electrical bond between antenna base and airplane skin (AMM 20-22-01).
      (1) Make sure that the resistance is not more than 0.1 ohm.
   H. Apply fillet of aerodynamic smoother around antenna base using sealing gun or tube (AMM 51-31-0/201).

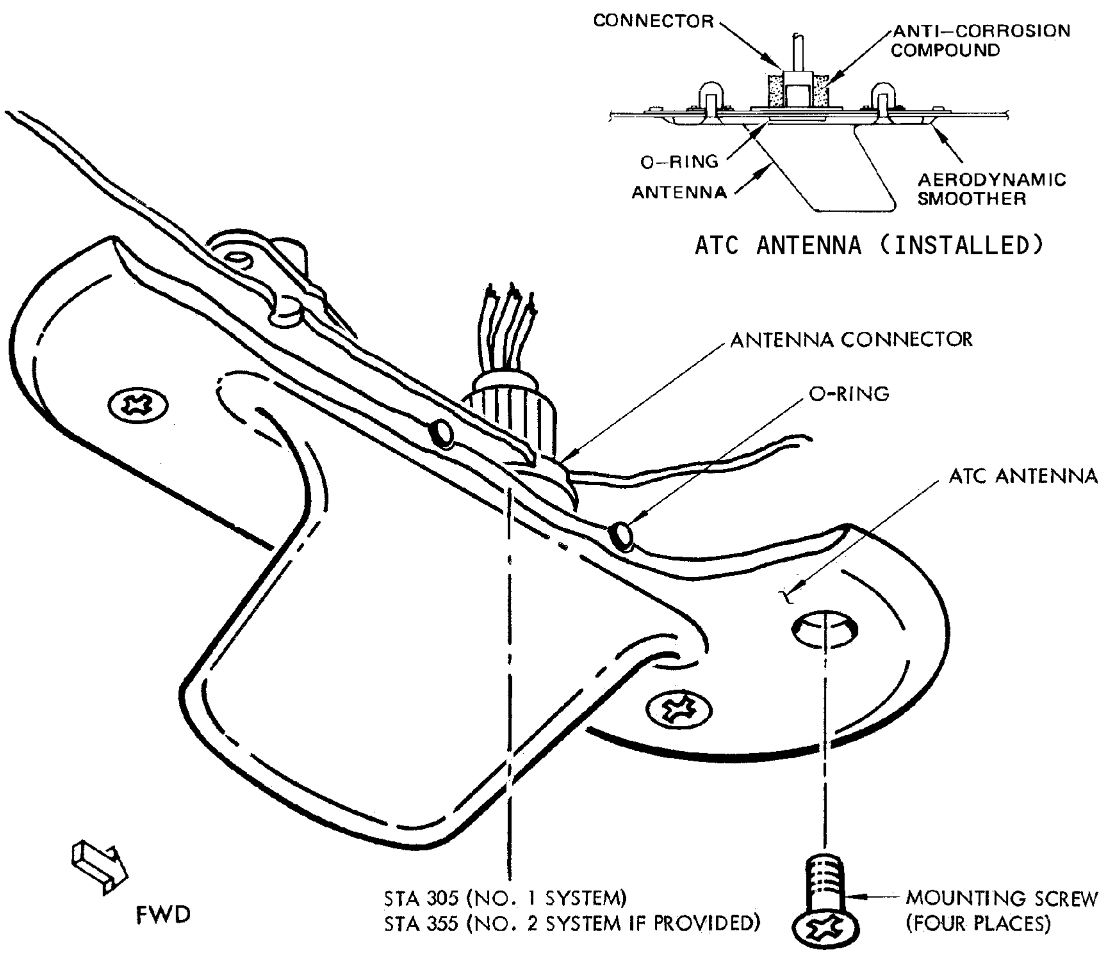
      NOTE: Make sure that no air is trapped during filling. Overfill to allow for smoothing and leveling.

I. Use spatula to smooth compound to an even 45-degree fillet. Remove excess compound (AMM 51-31-0/201).

CAUTION: OBEY THE INSTRUCTIONS IN THE PROCEDURE TO REMOVE THE AERODYNAMIC SMOOTHER. IF YOU DO NOT OBEY THE INSTRUCTIONS, DAMAGE TO THE AIRPLANE SURFACE CAN OCCUR.

J. Perform System Adjustment/Test (AMM 34-53-0)





ATC Antenna Installation (Typical)
Figure 401

ATC TRANSPONDER - REMOVAL/INSTALLATION

1. General
   A. The ATC transponder is installed in the main equipment center on shelf E2-1. The unit is shock mounted and held in place by two hold down hooks. Some transponders require an adapter to fit into the existing mount. The transponder and adapter should be removed or installed as one unit.
2. Remove ATC Transponder
   A. Open ATC 1 and 2 circuit breakers on panel P18.
   B. Disconnect cable connectors on front of transponder (if required).
   C. Loosen hold-down fasteners at front of unit, and pull out transponder.
   D. On airplanes with ATC transponder adapters installed,
      (1) Remove antenna and suppressor cables from adapter.
      (2) Remove shock mount hold-down clamps.
      (3) Carefully pull unit from shock mount.
3. Install ATC Transponder
   A. Slide transponder into rack and secure.
   B. On airplanes requiring ATC transponder adapters.
      (1) Ensure transponder is securely fastened into adapter.
      (2) Slide unit into shock mount.
      (3) Fasten shock mount hold-down clamps.
      (4) Connect antenna and suppressor cables.
   C. Close circuit breakers opened in step 2.A. above.
   D. Test transponder
      (1) Provide electrical power.
      (2) Select STBY on ATC control module. Allow 1 minute warm up.
      (3) On ATC control module, select installed transponder. Set up and operate appropriate ATC system in mode A.
      (4) On installed transponder, depress TEST button. Verify REPLY light comes on.
      (5) Release TEST button.
      (6) Perform altitude-reporting test (if required) in par. 3, 4, or 5 of ATC System Adjustment/Test (Ref 34-53-0).
      (7) On ATC control module position function selector to STBY or OFF as appropriate.
      (8) Remove electrical power if no longer required.

DISTANCE MEASURING EQUIPMENT - DESCRIPTION AND OPERATION

1. General
   A. Distance Measuring Equipment (DME) is used for determining the line-of-sight distance between an airplane and a VOR/DME ground station. The interrogator transmits shaped and coded interrogation pulses which triggers corresponding ground station reply pulses. This reply is received by the interrogator receiver section, detected and applied to the computing circuits. The time interval, between the interrogating signal sent from the airplane and the received reply from the ground station is measured and converted into distance information on direct reading indicators.
   B. One or two complete DME systems are installed in the airplane. Each system consists of an interrogator (transmitter-receiver), an antenna, one or more distance indicators, and a remote control panel. DME system component location is shown in Fig. 1, which specifies number of components installed.
2. Control Panel
   A. Each VOR/DME ground installation is assigned a specific VOR frequency and an associated DME frequency. Since the DME system works in conjunction with the VOR system, controls for both are located on the same control panel and consist of a VOR/DME frequency selector and a DME control switch. Normally DME No. 1 is controlled from the captain's VOR/DME control panel and DME No. 2 from the first officer's VOR/DME control panel.
3. Antennas
   A. The DME antennas are conventional omnidirectional flush-mounted or blade-type units connected to the interrogators by coaxial cable. The No. 1 system antenna is on the bottom of the fuselage at approximately body station 468. The No. 2 system antenna is located at approximately body station 580.
4. Interrogators
   A. The interrogators are installed in the electrical/electronic compartment (Fig. 1).
   B. When a VOR station frequency is selected, the associated DME interrogator is automatically tuned to the DME station at the same installation. The interrogator transmits a pair of high power VHF pulses which trigger a similar ground station reply to the airplane.

C. Each DME interrogator sends a suppression pulse to the ATC radar transponders during the transmission and reception of signals. Since both the ATC and DME systems are pulse-coded and operate in the same frequency range, the suppression pulse acts to prevent system interference. Similarly, the ATC transponders, when transmitting, suppress the DME interrogators.

D. The interrogators operate in the frequency range of 962 to 1213 MHz. There are 126 transmitting channels between 1025 and 1150 MHz and two 63-channel receiving bands from 962 to 1024 and 1151 to 1213 MHz.

5. Indicators

A. The DME indicators are electromechanical, digital counting types and indicate nautical miles on three individual counter dials measuring in units, tens, and hundreds.

B. Normally the No. 1 DME, system drives the captain's CDI and first officer's DME No. 1 indicators while system No. 2 feeds the first officer's CDI and captain's DME No. 2 indicators. On some airplanes the DME information is presented on the CDI indicators only (Fig. 1).

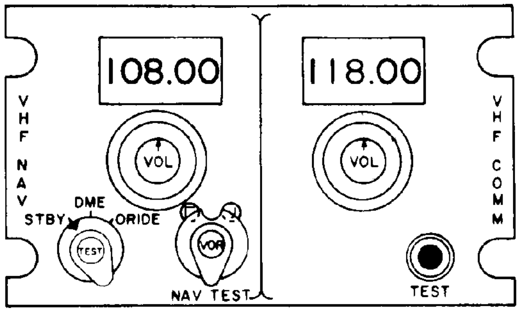
C. Warning flags come into view when the system is inoperative, is in the search or the memory mode, or when the DME completes the prescribed memory cycle without achieving lock-on.

6. Operation

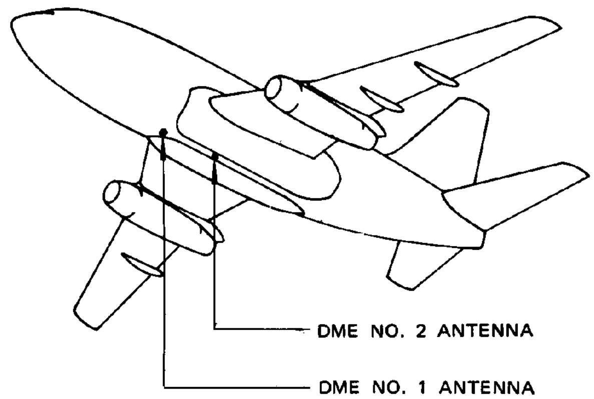
A. The DME system is placed into operation through the DME control switch on the VOR/DME panel.

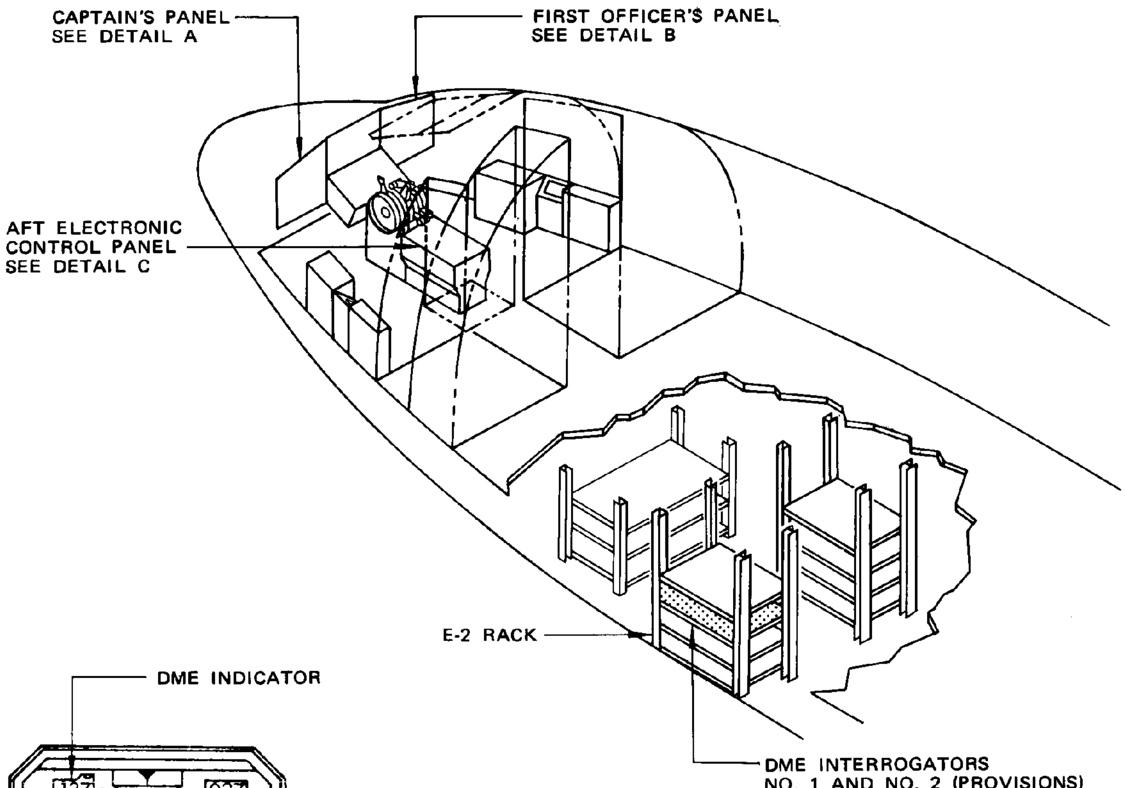
(1) Placing the switch to the on position (if installed) causes primary power to be applied to the system for warmup only. The interrogator incorporates a time-delay circuit which inhibits the transmitter for approximately 1 minute after application of power. In this position warning flags appear across the DME indicators.

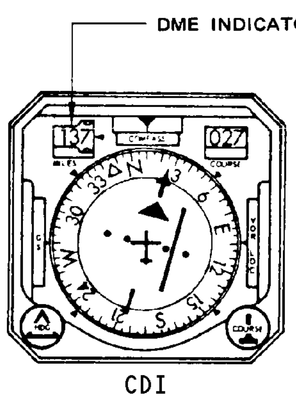
(2) The system incorporates a signal controlled search (SCS) feature, to prevent unnecessary search when tuned to a VOR only station, or when out of range. When the ground station signal is too weak to be usable, the signal controlled search circuits place the interrogator in the standby mode. When this occurs, the search operation stops, the transmitter is turned off, and the warning flags appear. When a usable ground station signal reappears, the system automatically starts operating.



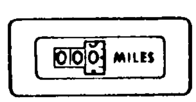
CONTROL PANEL
DETAIL C







CDI
DETAIL A



DME INDICATOR
DETAIL B

DME System Component Location
Figure 1

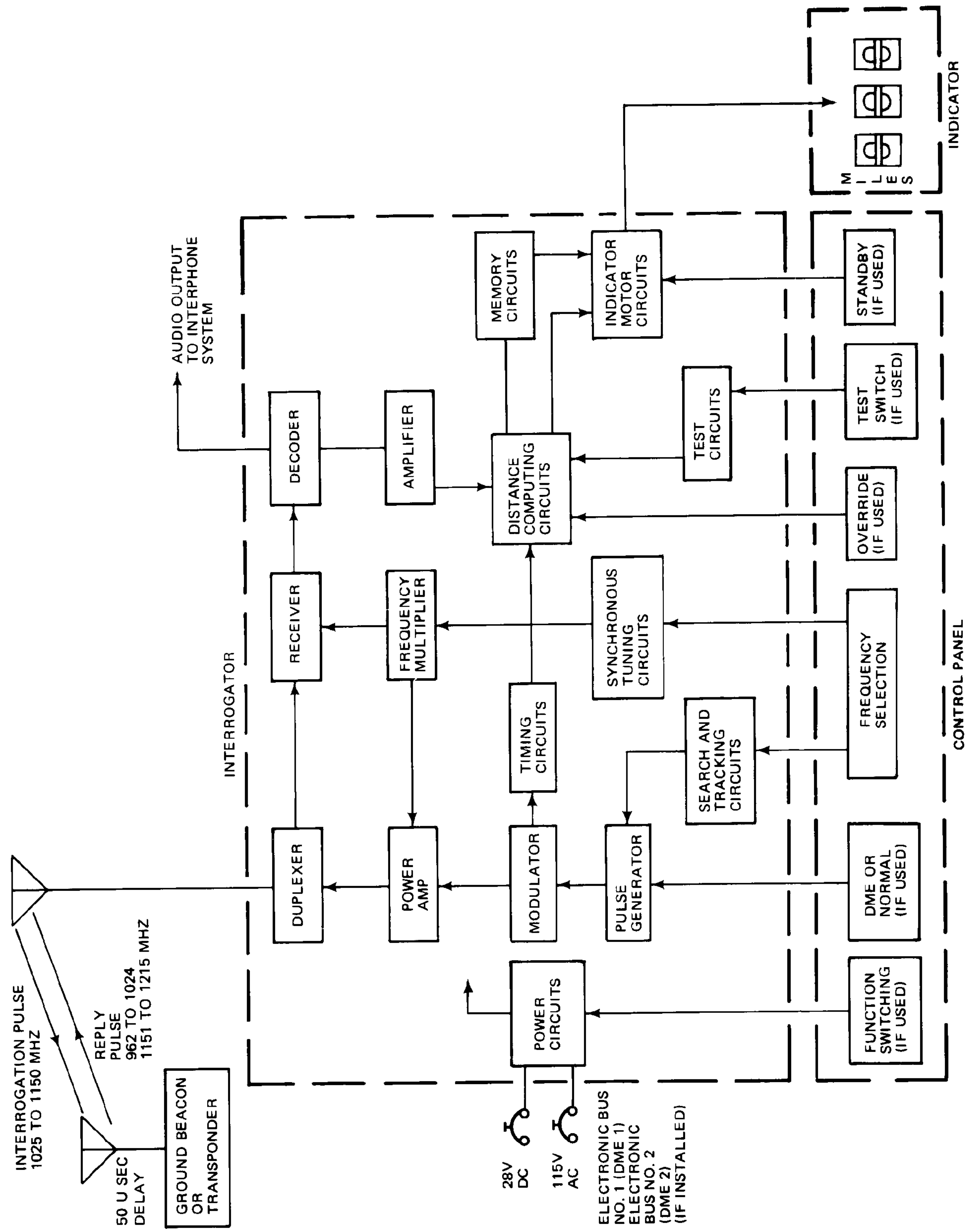
(3) The echo protector circuitry (EPC) prevents the interrogator from remaining locked to an echo reply at any distance up to the range limit setting for more than 10 seconds. When an echo reply is detected by the EPC, it causes the interrogator to break lock, disable the transmitter section momentarily and slew the DME indicator(s) to zero miles. The interrogator will automatically start the search cycle again from zero miles to assure that the correct reply pulse is encountered first. The EPC is disabled when the interrogator is in the search or standby mode.

(4) In the DME or ON position of switch, the system performs normal search and measurement functions within the 200 or 300-mile range capability except for DME ground stations which are located with ILS (instrument landing system) ground stations. When a localizer channel between 108.0 and 111.9 MHz (channels 17-56) is selected on the VOR control panel, DME operation is intentionally limited to a 50-mile range. In this instance the DME will search from zero miles, slew back down to zero, and repeat the cycle until it locks to a signal.

(5) Operation of the system in the override position (where installed) is similar to normal operation except that on localizer channels between 108.0 and 111.9 MHz, the 50-mile range limitation is overridden and all channels can be utilized for full range DME operation.

(6) Placing the DME control switch to the test position (where installed) overrides any signal to the indicator and causes the distance dial to rotate backwards slightly below zero and then align at zero. The flag should be out of view. This test checks the accuracy of the distance measuring portion of the system.

B. The DME interrogators always search outbound, that is going up in mileage to ensure that the system locks on a fundamental signal. However, occasionally the indicators will run backwards, decreasing in mileage reading. Any time the system is rechanneled or loses lock at an indication less than 50 miles, the distance indicators will slew to slightly below zero miles before the outbound search begins. This feature (reciprocating search) saves the time formerly required to drive the indicators through maximum range limits to zero miles.



DME System - Block Diagram
Figure 2

C. When the DME system is locked on a station, the indicators decrease in reading as the airplane flies inbound and increase in mileage as the airplane flies outbound from the station. A memory circuit prevents the system from recycling to the search mode, during momentary signal interruption.

D. Audio monitoring of the DME station identification is accomplished through the interphone system, and controlled through VOR/DME controls on the audio selector panels. The DME signal is distinguished by the higher tone station identification (morse code) which follows every fourth VOR identification signal.

E. Operating power for the DME system is supplied through 115-volt ac and 28-volt dc circuit breakers located on the P18 (DME-1) or, P6 (DME-2, if installed).

DME SYSTEM - TROUBLESHOOTING

1. General
   A. A prerequisite for accurately troubleshooting any system is a good knowledge of normal system operation and systems component locations. Refer to DME navigation systems description and operation for review when required.
   B. Troubleshooting the DME System is best accomplished by performing the tests described in DME System - Adjustment/Test. Satisfactory completion of the Adjustment/Test procedures indicates that the system is operating properly. When satisfactory operation is not obtained, the tabulated data below should be consulted. If using the tabulated data fails to correct the trouble, airplane wiring should be suspect and repaired or replaced as necessary.
   C. The tabulated data is written for one DME system. When two systems are installed, troubleshooting may be accomplished by interchanging components.

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
System completely inoperative	Circuit breaker	Check that system circuit breakers on P18 (DME-1) or P6 (DME-2) circuit breaker panel is closed	Close circuit breaker
	No power available to interrogator	Check that 28 volts dc is available at pin 29 and 115 volts ac at pin 25 of plug connected to interrogator. If power is available, interrogator faulty. If power is not available, airplane wiring faulty	Replace interrogator. Refer to AMM Chapter 24, Electrical Power
Distance indicator reading incorrect	Indicator or interrogator	Check second indicator being fed by same interrogator. If second indicator reading is correct, first indicator faulty. If second indicator reading is incorrect, interrogator faulty	Replace indicator. Replace interrogator

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Unable to select channel	Control panel		Replace control panel
Improper channel selection	Interrogator		Replace interrogator
Indicator inoperative	Indicator or interrogator	Perform self-test (if provided). If self-test operates correctly, indicator faulty. If self-test does not operate correctly, interrogator faulty	Replace indicator Replace interrogator

DISTANCE MEASURING EQUIPMENT - ADJUSTMENT/TEST

1. DME System Test
   A. Equipment and Materials
      (1) DME Ramp Test Set - Collins 475D-1 modified for AUTO 5-180 feature
   B. Prepare to Test DME System
      (1) Connect test set antenna cable to J503 (top right corner of test set).
      (2) Place test set function selector in BATT position. Check that meter indicates in OK portion of scale. If it does not, recharge self-contained battery.
      (3) Turn test set T/R ATTN DB selector to 25. Place function selector in 17 OUT position. Adjust RF output control for red line indication on meter.
      (4) Place function selector in 5-MI position. Press and hold self-test button. Make sure that the meter needle indicates 5 miles.
      (5) With self-test button still depressed, place function selector in 180-MI position. Make sure that meter needle indicates 180 miles.
      (6) With self-test button still depressed, place function selector in AUTO 5-180 position. Make sure that meter needle indicates in 5-MI portion of scale.
      (7) Release self-test button and then press again. After approximately 2 seconds, make sure that meter needle indicates in 180-MI portion of scale.
      (8) Release self-test button and then press again. After approximately 2 seconds, make sure that meter needle indicates in 5-MI portion of scale.
   C. Test DME Interrogator Using Self-Test Function (if provided)
      (1) Provide electrical power and energize circuit breaker panels P18 and P6. Close DME circuit breakers.
      (2) Position the DME control to the DME or ON position (if provided) and allow approximately 2 minutes for proper equipment warmup.
      (3) Press and hold the test switch ON (located on captain's DME control panel or on front of interrogator). After approximately 5 seconds, make sure that DME indicator reads 0.0 ±0.5 miles. For airplanes equipped with AVQ-70 DME interrogator, make sure that DME indicator reads 94.5 ±0.5 miles.
      (4) Repeat self-test steps (2) and (3) for first officer's DME or No. 2 DME (if provided).
   D. Test of the DME System with OVERRIDE Capability

      NOTE: Override capability is provided by override switch on DME control panel or by pin 19 being permanently grounded (see appropriate wiring diagram) on the DME interrogator.

(1) Place the tester at a distance of approximately 30 to 40 feet from the DME antennas.
(2) Set the TR ATTEN DB control on the tester to the 15-dB position. Set the Function Selector to 5 MI position.
(3) On the captain's DME control panel, select 108.00 MHz and position the DME to the OVERRIDE position (if provided).
(4) With DME-1, and interphone circuit breakers closed; position DME-1 audio listen switches (when provided) on the audio selector panel to the ON position and verify that an audio tone is present.

NOTE: Some airplanes have DME audio connected in parallel with NAV audio and some require the voice range selector be in BOTH or RANGE position.

(5) Check that the flag on the captain's DME indicator (CDI in some airplanes) and on the first officer's DME-1 indicator (if provided) is pulled completely out of view when the DME-1 interrogator in locked on and in view when DME-1 interrogator is searching.
(6) Check that the captain's DME locks on at 5 miles as displayed on the distance indicator. Position Function Selector to 180 MI. After approximately 18 seconds, check that the captain's DME goes to search mode and then tracks inbound and outbound in the region of 171 to 189 miles.
(7) Position Function Selector to 5 MI. After approximately 18 seconds, check that the captain's DME goes into search mode and then locks on at 5.0 ±0.5 miles.
(8) While the captain's DNE is locked on in the 5-mile position, set the OVERRIDE switch to normal position. If DME is permanently in override position (pin 19 on DME interrogator is grounded) omit this step. Position Function Selector to 180 MI. Verify that the captain's DME goes into search mode but does not lock on.
(9) Return Function Selector to 5 MI. Make sure that the captain's DME locks on at 5.0 ±0.5 miles.
(10) Switch the captain's DME to a different channel, then back to 108.00 MHz. Check for lock-on and the presence of tone.
(11) Repeat steps (3) thru (10) for the No. 2 DME system 1 substituting first officer for captain and DME-2 for DME-1 (if second DME interrogator is not installed, and there are provisions for second system, it is necessary to place the No. 1 DME interrogator in the installation provided for No. 2 unit).

NOTE: If No. 2 interrogator was installed in the No. 1 interrogator position, it is necessary to re-install the No. 1 interrogator in the No. 1 interrogator position.

E. Test of the DME System Without OVERRIDE Capability

(1) Repeat steps D.(1) and (2).

(2) Select 108.00 MHz on the captain's DME control panel.

(3) Repeat step D.(4).

(4) Make sure that DNE No. 1 read-out on the captain's indicator (CDI on some airplanes) and on the first officer's DME-1 indicator (if provided) reads 5.0 ±0.5 miles and the warning flags are out of view.

(5) With the DME-2 or first officer's DME set to the STBY position (if provided) make sure that the warning flag is visible on the No. 2 system indicator(s).

(6) Switch the captain's DME to a different channel, then back to 108.00 MHz. Check for lock-on and the presence of tone.

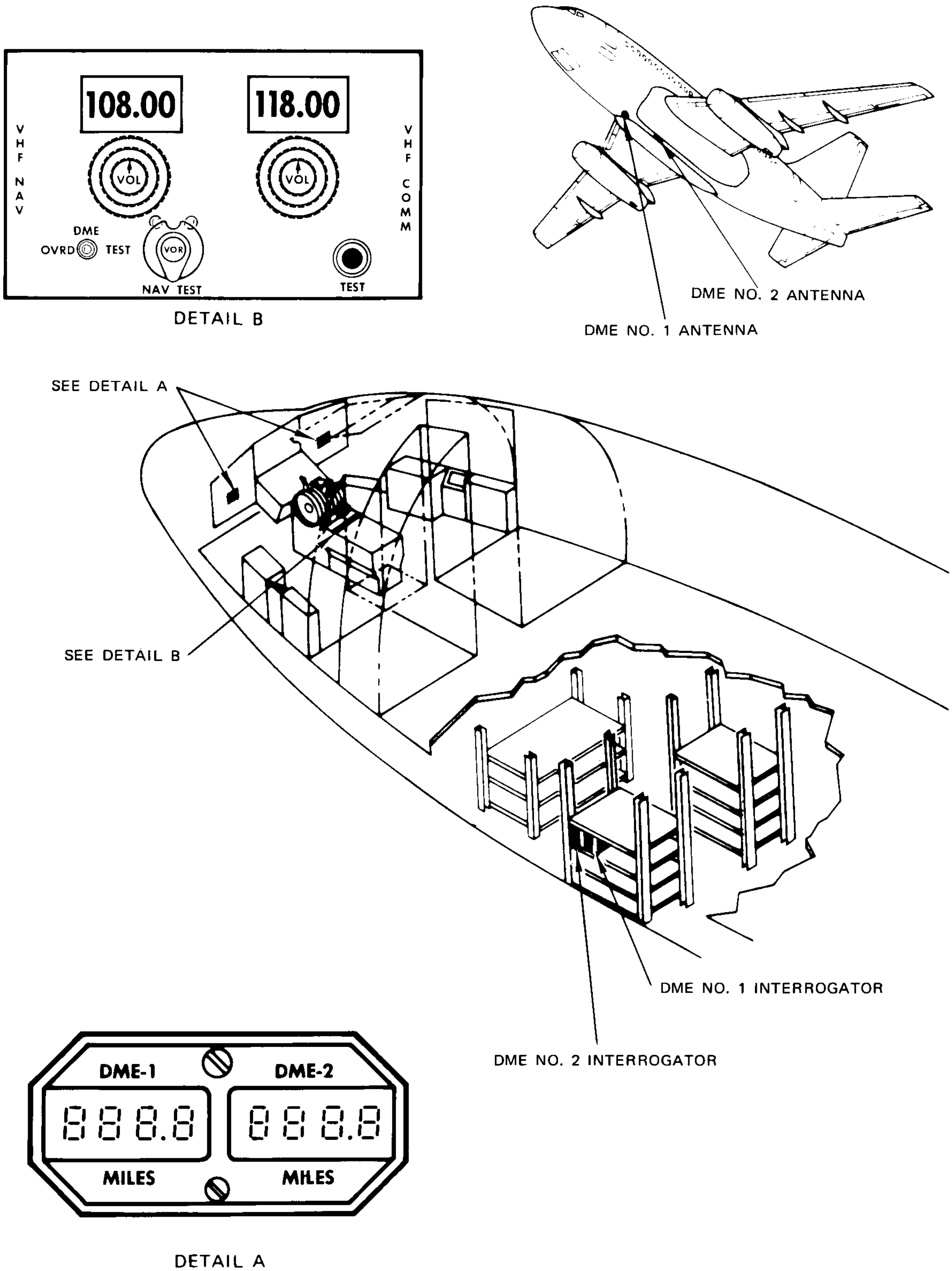
(7) Repeat steps (2) thru (6) for DME system No. 2 substituting first officer for captain and DME-2 for DME-1. If second DME interrogator is not installed, and there are provisions for second system, it is necessary to place the No. 1 DME interrogator in the installation provided for No. 2 unit.

NOTE: If No. 2 interrogator was installed in the No. 1 interrogator position, it is necessary to reinstall the No. 2 interrogator in the No. 1 interrogator position.

(8) If no longer required, remove electrical power from airplane.

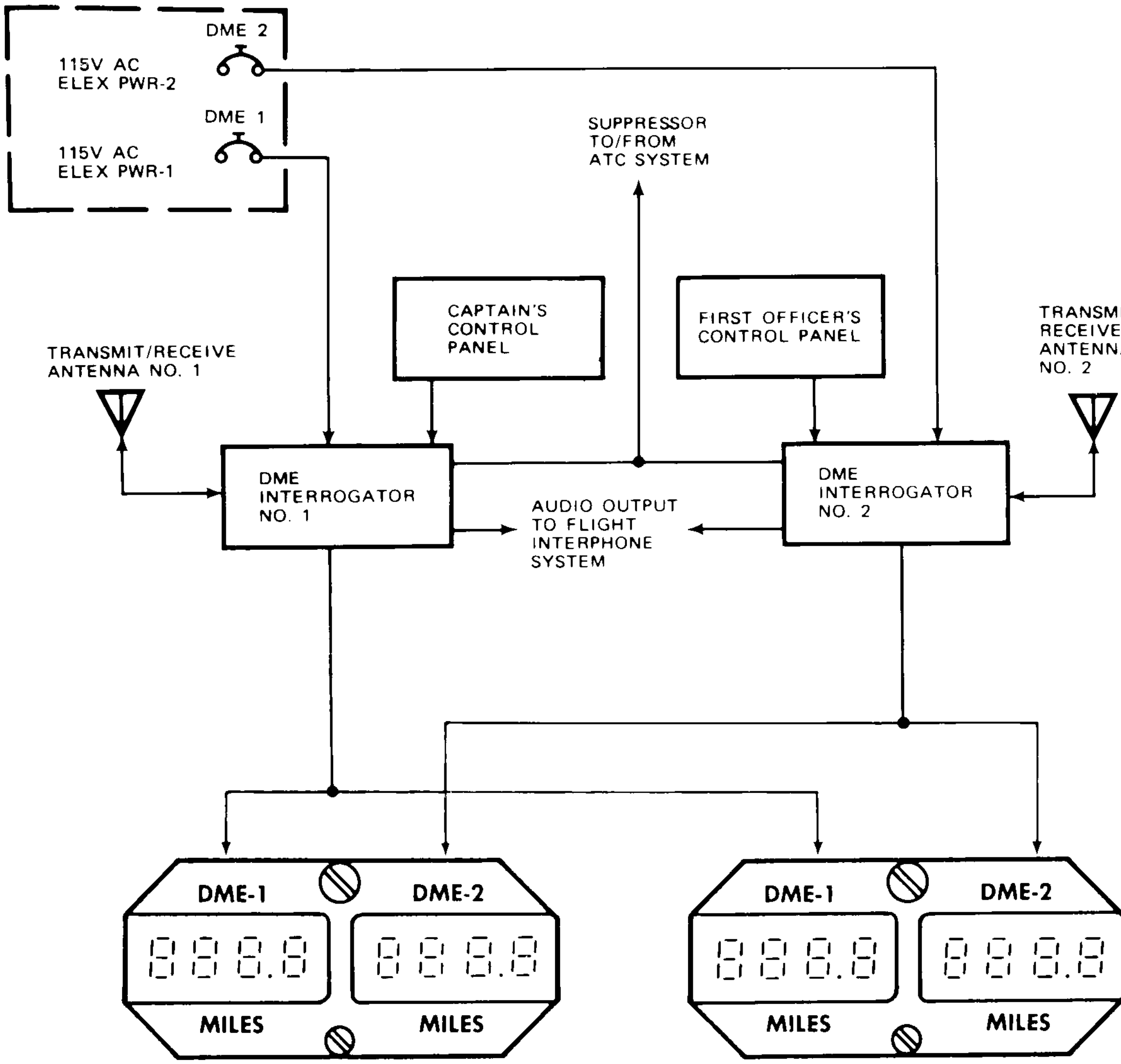
DME SYSTEM - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The DME (distance measuring equipment) system automatically measures the distance (in nautical miles) between the airplane and a selected ground station. The system measures the airplane-to-station slant-range distance (line-of-sight) and the information is displayed on a digital readout indicator as miles-to-go. The method of measuring this distance is primarily a function of an interrogator (receiver-transmitter) unit, operating in the 960- to 1215-MHz range. The DME frequency is automatically selected when a VOR frequency is selected.
   B. The interrogator unit transmits a pulse-coded interrogation signal which actuates a corresponding ground station reply. This reply signal is received and processed by the interrogator. The time interval between the interrogation signal transmitted from the airplane and the ground station reply signal received by the airplane is measured, converted into distance, and displayed on the indicator.
   C. Two complete DME systems are installed in the airplane. Each system consists of an interrogator (transmitter-receiver) unit, an antenna, two distance indicators and a remote tuning control panel (Fig. 1). The system is designed to indicate a range of approximately 200 nautical miles in normal operation and 400 nautical miles in override.
   D. The 115-volt ac, 400-Hz power is applied to DME system No. 1 from the essential radio bus. An internally mounted power supply converts the 115 volts ac to required ac and dc operating power.
   E. Power to DME system No. 2 is the same as that supplied to DME No. 1 except that it is supplied from radio bus No. 2.
   F. The DME system interface is shown in Fig. 2. The output of the interrogator is applied to the digital readout of the DME indicator. A sample pulse is taken from each interrogator and coupled to the ATC system, which operates in the same general frequency range. This pulse acts as a suppression pulse. An audio signal from the DME interrogator is applied to the flight interphone system.
2. Control Panel (Fig. 1 and 2)
   A. Each ground station installation is assigned a specific VOR frequency and associated DME frequency. Since the DME system operates in conjunction with the VOR/ILS navigation system, controls for both are located on the same control panel and consist of a NAV frequency selector and a DME function selector switch. DME system No. 1 is controlled from the captain's control panel and DME system No. 2 from the first officer's control panel.
   B. The DME function selector control is a 3-position toggle switch with positions: OVRD, DME and TEST.



DME System Component Location
Figure 1

3. Antennas
   A. The DME antennas are conventional flush-mounted or blade-type units which are connected to the interrogator units by coaxial cable. The No. 1 system antenna is installed on the bottom of the fuselage at station 468 . The No. 2 system antenna is installed at station 580. Each antenna performs both transmitting and receiving functions for its respective DME system in the frequency range of 960 to 1215 MHz.
4. Interrogator Units
   A. Each interrogator unit provides the capability to compute the time interval between transmitted and received pulse-pair signals, convert this time difference to distance, and apply this information to the DME indicator.
   B. The interrogator units are standard 1/2 ATR short assemblies which mount (side-by-side) on a shelf in the electrical/electronics equipment center (Fig. 1).
   C. The front panel of each interrogator contains three fault indicators and a reset switch. The fault indicators are placarded R/T, IND, and ANT. The fault indicators turn yellow to indicate a malfunction occurring in the interrogator (R/T) or the antenna unit (ANT) and are used as an aid for trouble shooting the system. The IND fault indicator is not used. The RESET button allows the operator the option of resetting the IND and ANT fault indicators to normal (black) indication. To reset the R/T fault indicator it is necessary to ground pin A1J4 through a hole in the right side of the unit.
5. Indicators
   A. Two dual BCD (binary-coded-decimal), seven-segment indicators are installed in the airplane as shown in Fig. 1.
   B. Each indicator provides a dual display of distance information from DME systems No. 1 and 2, with each half of the dual indicator operating independently of the other (Fig. 2).
   C. The indicator receives a BCD pulse signal from either or both DME systems. Logic in the indicator decodes this signal and converts it into a digital display showing miles-to-go to the interrogated ground station.
   D. The display numerals are formed by energizing discrete segments of an incandescent, seven-segment, lamp matrix in response to the pulse-coded information. Any numeral (from 0 to 9) can be formed using a maximum of seven segments. For example: the numeral 8 uses all seven segments (Fig. 1).



DME System Interface Diagram
Figure 2

6. Operation

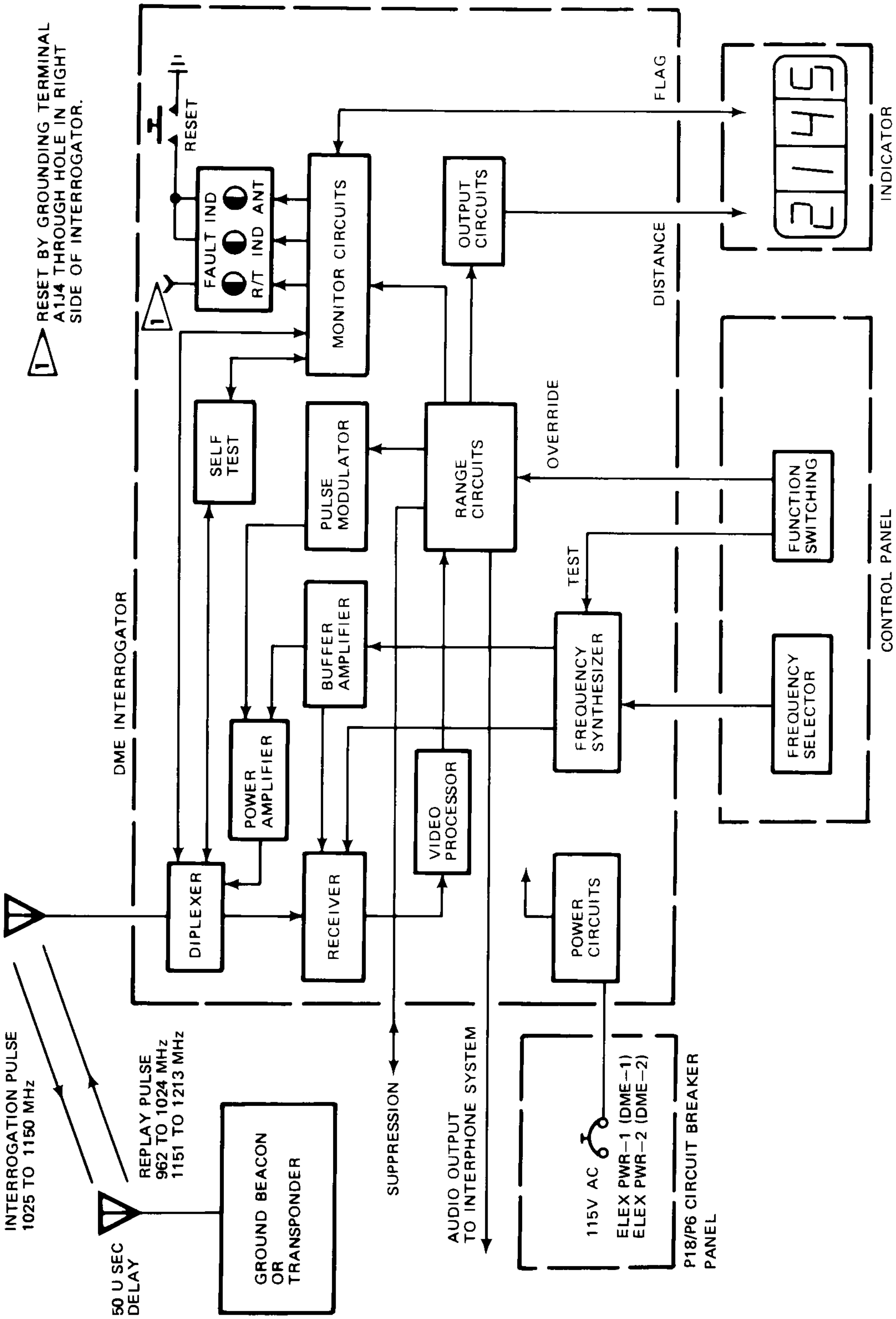
A. Functional Description (Fig. 3)

(1) The DME interrogation signal from the airborne equipment consists of pulses of rf energy transmitted in pairs. The pulses in a pair are spaced 12 microseconds apart for the assigned 126 DME channels. A reply signal is transmitted from the ground station for each interrogation pulse-pair from the airplane equipment. The reply signal consists of pulse pairs with the same characteristics as the interrogation pulse pairs, except for rf frequency. The reply pulse rf frequency is 63 MHz above or below the interrogation pulse frequency. The exact relationship of the frequencies depends upon the particular DME channel selected. The DME system searches for synchronous pulses and provides valid distance data when locked on and tracking.

(2) DME channel selection is enabled by tuning the VOR/ILS system to a VOR frequency on the VHF NAV No. 1 or 2 control panel (Fig. 3). This operation automatically tunes the DME through a 2 of 5 bit coding system which is converted to BCD in the interrogator.

(3) The BCD output is applied to the frequency synthesizer, the digital to analog converter and the control circuit. The control circuit decodes the BCD information and the resultant represents the DME frequency band of operation. The BCD frequency information and the frequency band information control the output from the frequency synthesizer. The signal is applied to the DME driver where it is multiplied, amplified, and pulse-modulated from signals received from the distance circuits. The DME driver output is applied to the transmitter section (Fig. 3). The modulator receives low-level pulse-pair trigger signals from the distance circuits and synchronously generates high-level shaped pulses for application to the transmitter. When the modulation pulses are present, the rf is applied to the diplexer where appropriate isolation of the transmitted and received signals are enabled. The signals are finally applied to the transmit-receive antenna and radiated to the ground station as interrogation signals.

(4) Reply signals from the ground station are received at the antenna and applied to the diplexer where the signal is then directed to the receiver section of the interrogator. The receiver rf circuits also receive an input from the digital-to-analog (D/A) converter. This input is an analog tuning voltage which the digital-to-analog converter develops from the BCD frequency information at the BCD frequency converter output, and the frequency band information at the control circuit output. The tuning voltage tunes the receiver rf circuits to the DME receive frequency and also tunes the frequency synthesizer output.



DME System - Block Diagram
Figure 3

(5) The IF receiver output consists of video pulses which are applied to the decoder. The decoder distinguishes between valid and invalid pulse-pairs, accepting only valid pulse pairs. It also generates an agc voltage which is applied back to the receiver rf and IF circuits.

(6) The decoded video pulses are applied from the decoder to the distance circuits. The distance circuits measure the time between the leading edge of the first pulse of the interrogation pulse-pair and the leading edge of the first pulse in the reply pulse-pair. The time measurement is converted to an output consisting of BCD distance word bits. This output is combined with the video signal from the receiver circuits and applied to the ident/blanking circuits where the station identification signal is developed. This signal is then applied to the airplane flight interphone system for reproduction in the captain's and first officer's headphones.

(7) The BCD distance word bits are clock-pulse synchronized and amplified into a 32-bit serial BCD word that is applied to the DME indicator. The data clock signal and word sync signal are also applied to the DME indicator. The 32-bit digital word is loaded into the indicator and decoded to provide the decimal-number display.

(8) The monitor circuits provide fault monitoring of the interrogator unit and DME antenna. A binary coded fault indicator located on the monitor card provides indications that enable precise isolation of malfunctioning circuits within the interrogator unit. The monitor card also contains amplifier circuits that drive the fault indicators on the interrogator front panel (R/T, IND and ANT). These indicators are magnetic latching indicators that when initially set must be reset to remove the indication. These fault indicators enable isolation of a faulty receiver/transmitter (interrogator unit) or antenna. The IND fault indicator is not used.

(9) The monitor also provides a flag output signal. The flag signal is a stimulus for the flag alarm circuit which results in the DME indicator display showing blank when a malfunction occurs in the system.

(10) Power is applied to the system when the circuit breakers are closed. The interrogator incorporates a time delay circuit which inhibits the transmitter for approximately 1 minute after application of power. During this period, dashes appear on the DME indicators.

(11) With the function select switch in the DME position, the system performs normal search and measurement functions to a range of 200 nautical miles.

(12) With the DME function select switch in OVRD or TEST, a ground is applied to the BCD frequency converter which in turn appropriately programs the distance circuits. The system also has an automatic standby feature which places it in the standby mode when a ground station is not available or when the ground station signal is too weak to be usable. When this occurs, the search operation stops, the transmitter is turned off, the dashed lines appear, and the system is in all respects on standby. When a usable ground station signal reappears, the system automatically starts operating.

(13) Operation of the system in the OVRD position is essentially identical to normal operation. However, the 200-mile range limitation is overridden and all channels can be utilized for full-range DME operation (390 nautical miles).

(14) With the DME function selector switch set to TEST, the distance circuits develop a synthetic reply pulse spaced to represent the functional test distance of zero nautical miles. The distance circuits measure the time between the interrogation and self-test reply pulse and apply appropriate distance information to the control circuits. The control circuits then develop the serial BCD word which is applied to the DME indicator. When the test is activated, the DME indicator displays the shutter for 1 second, followed by dashed lines for 1 second, then followed by 000.0-nautical-mile readout.

7. <u>Control</u>

A. The DME system is turned on by closing the DME 1 or DME 2 circuit breakers on the P18 or P6 circuit breaker panel and tuning the system (using the VHF NAV frequency tuning controls) to a DME channel with the DME function select switch to DME position.

DME SYSTEM - TROUBLESHOOTING

1. General
   A. This fault isolation procedure quickly isolates problems without the use of additional test equipment.
      (1) The tree-type format makes use of the self-test features of the interrogator unit.
   B. The procedure is written for one DME system. When two systems are installed, troubleshooting may be accomplished by interchanging components.
   C. If this fault isolation procedure does not identify the problem, do a check of the wiring. Use the wiring diagrams.

DME SYSTEM - ADJUSTMENT/TEST

1. General
   A. This section contains an operational test and system test of the DME system. The operational test is intended to be used without tools or test equipment for a relatively quick check of the operational status of the system. Special tools and equipment are required for the system test.
2. Operational Test - DME System
   A. Prepare for Test
      (1) Ensure that following circuit breakers are closed:
          DME-1 AC (P18 panel)
          DME-2 AC (P6 panel)
      (2) Provide electrical power.
   B. Test DME System Operation (No. 1 or 2 system)
      (1) Adjust frequency tuning controls for 108.00 MHz.
      (2) Select TEST on DME function selector switch of appropriate NAV control panel, and check that within a few seconds, the numerals 000.0 appear on captain's and first officer's DME DISTANCE indicators.
3. System Test - DME System
   A. Equipment and Materials
      (1) DME Ramp Test Set - Collins 475D-1 (modified for AUTO 5-180/270 feature) or equivalent
   B. Prepare for Test
      (1) Calibrate ramp test set as indicated in test set instruction book.
      (2) Ensure that following circuit breakers are closed:-
          DME-1 AC (P18 panel)
          DME-2 AC (P6 panel)
          INTERPHONE
      (3) Provide electrical power.
   C. Test DME System
      (1) Place DME test set approximately 40 feet from DME antenna. Ensure there are no obstructions between the test set and DME antennas. (Run test on line-of-sight basis.)
      (2) Turn test set on and set TR ATTEN DB control to 15-db position.
      (3) Set FUNCTION SELECTOR on test set to AUTO 5-180/270 position.
      (4) Set RANGE switch on test set to 270.
      (5) Select 108.00 MHz on the captain's NAV/DME panel and 109.00 MHz on the first officer's NAV/DME panel.
      (6) Select O'RIDE/OVRD on DME function select switch of both captain's and first officer's NAV/DME panels.
      (7) Verify that DME-2 readouts display dashes.

(8) Verify that the DME-1 responds to the ramp test set's programmed test cycle as follows:
   (a) The DME-l readouts on captain's and first officer's DME indicators will display 5 ±0.1 miles.
   (b) After approximately 18 seconds the DME-1 readouts will track inbound to 263 ±5 miles and outbound to 277 ±5 miles.
   (c) After approximately 18 seconds, the DME-1 readouts will display 5 ±0.1 miles.

   NOTE: The above test cycle will repeat as long as the function selector switch is in the O'RIDE/OVRD position.

(9) Verify that when the DME-1 is locked on, the DME-1 audio output is available at each audio selector panel.
(10) While DME-1 is locked on in the 5-mile position, select ON/NORM/DME on the captain's control panel. Verify that DME-1 now responds to the following test cycle:
   (a) DME-1 readouts will display 5 ±0.1 miles.
   (b) After approximately 18 seconds, the DME-1 readouts will display dashes.
   (c) After approximately 18 seconds, the DME-1 readouts will lock on at 5 ±0.1 miles again.

   NOTE: The above test cycle will repeat as long as the function selector switch is in the ON/NORM/DME position.

(11) Set RANGE switch on test set to 180.
(12) Verify that DME-1 now responds to the following test cycle:
   (a) DME-1 readouts will display 5 ±0.1 miles.
   (b) After approximately 18 seconds, DME-1 readouts will track inbound to 173 ±2 miles and then outbound to 187 ±2 miles.
   (c) After approximately 18 seconds, the DME-1 readouts will display 5 ±0.1 miles.

   NOTE: The above test cycle will repeat as long as the function selector switch is in the ON/NORM/DME or O'RIDE/OVRD position.

(13) Tune captain's control panel to a different channel, then back to 108.00 MHz. Check for lock on and presence of tone at each audio selector panel.
(14) Repeat steps C.(1) thru C.(13) substituting DME-1 for DME-2. DME-2 for DME-1, captain's for first officer's and first officer's for captain's.
(15) Remove electrical power.

DME ANTENNA - REMOVAL/INSTALLATION

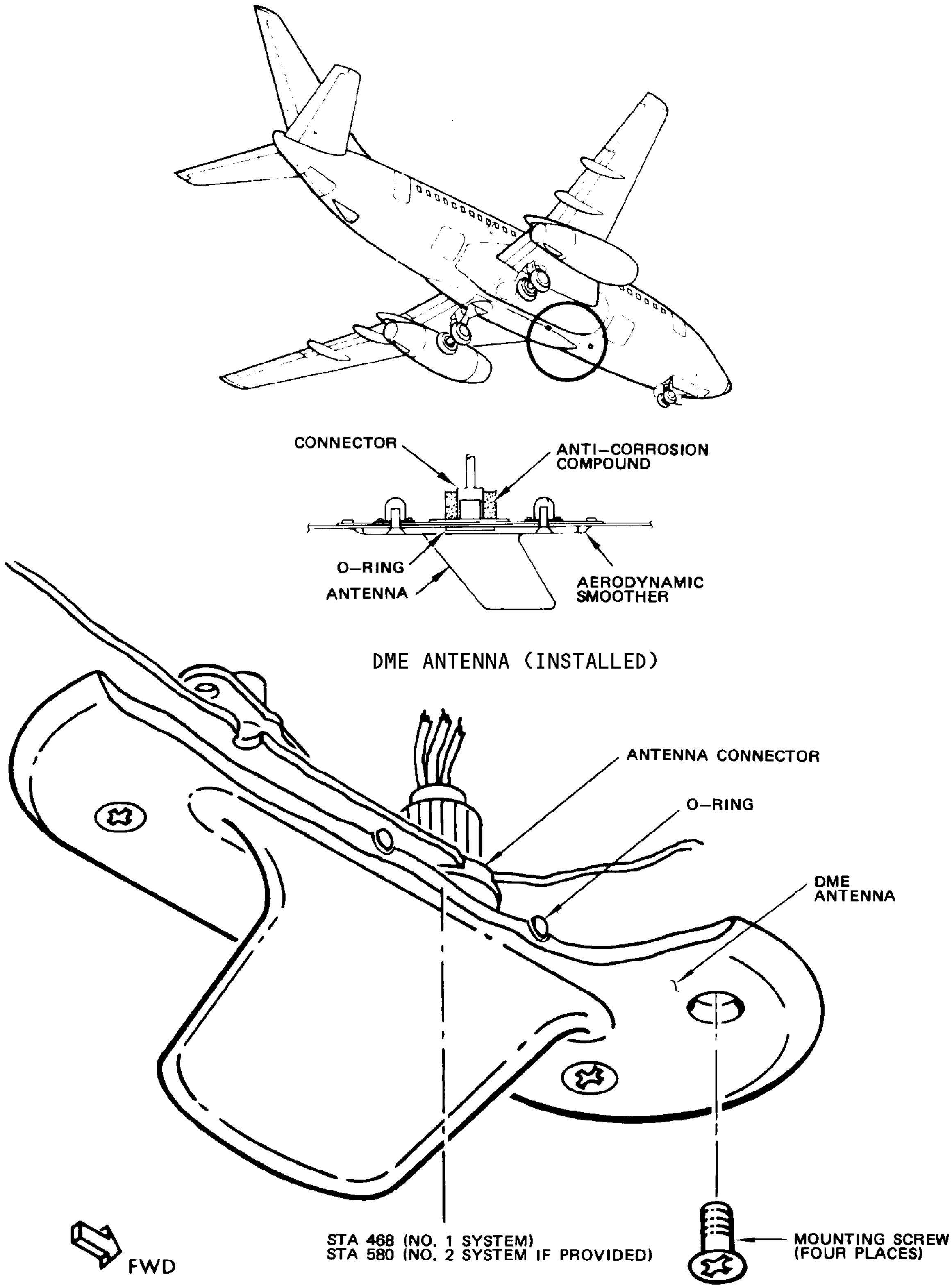
1. Equipment and Materials
   A. Equipment
      (1) Sealing gun - 6-inch Length cartridge, Senco Research, or equivalent (not required for material furnished in tubes)
      (2) Bonding Meter (Ref 20-22-01)
      (3) Varnish brush - 1- or 2-inch
      (4) Spatula
   B. Materials
      (1) Aerodynamic smoother - BMS 5-79, class B-1/2 (Ref 20-30-11)
      (2) Accelerator - as indicated on aerodynamic smoother container (Ref 20-30-11)
      (3) Aliphatic Naphtha - TT-N-95 (Ref 20-30-31)
      (4) Anticorrosion compound - BMS 3-23
2. Remove Antenna
   A. Remove four mounting screws (Fig. 401).
   B. Using care, lower antenna from mounting until cable can be disconnected.

      CAUTION: DO NOT PULL ANTENNA.

   C. Disconnect antenna from antenna connector.
   D. Remove antenna and O-ring.
3. Install Antenna
   A. Remove aerodynamic smoother from antenna base and mounting area (AMM 51-31-0/201).

      CAUTION: OBEY THE INSTRUCTIONS IN THE PROCEDURE TO REMOVE THE AERODYNAMIC SMOOTHER. IF YOU DO NOT OBEY THE INSTRUCTIONS, DAMAGE TO THE AIRPLANE SURFACE CAN OCCUR.

   B. Remove grease, oil, dirt, and chips from antenna base and mounting area. Use small varnish brush to apply fresh aliphatic naphtha for cleaning. Wipe cleaner off with clean cloth.
   C. Clean mating surfaces of antenna and aircraft structure.
   D. Apply a protective coat of corrosion preventive compound (Ref 51-21-91 CP).
   E. Place O-ring in position on antenna base.
   F. Connect antenna cable to antenna connector (Fig. 401).
   G. Position antenna in place and install three of the four mounting screws.
   H. Check electrical bond between antenna base and airplane skin per 20-22-01.
      (1) Connect the ohmmeter or the resistance measuring bridge between the empty antenna mounting screw hole (antenna base plate) and the airplane skin.
      (2) Make sure the resistance is not more than 0.1 ohm.
      (3) Install and tighten the last screw.



DME Antenna - Removal/Installation
Figure 401

I. Apply fillet of aerodynamic smoother around antenna base using sealing gun or tube (AMM 51-31-0/201).

<u>NOTE</u>: Make certain no air is trapped during filling. Overfill to allow for smoothing and leveling.

J. Use spatula to smooth compound to an even 45-degree fillet. Remove excess compound (AMM 51-31-0/201).

<u>CAUTION</u>: OBEY THE INSTRUCTIONS IN THE PROCEDURE TO REMOVE THE EXCESS COMPOUND. IF YOU DO NOT OBEY THE INSTRUCTIONS, DAMAGE TO THE AIRPLANE SURFACE CAN OCCUR.

K. Perform system Adjustment/Test (Ref 34-55-0).

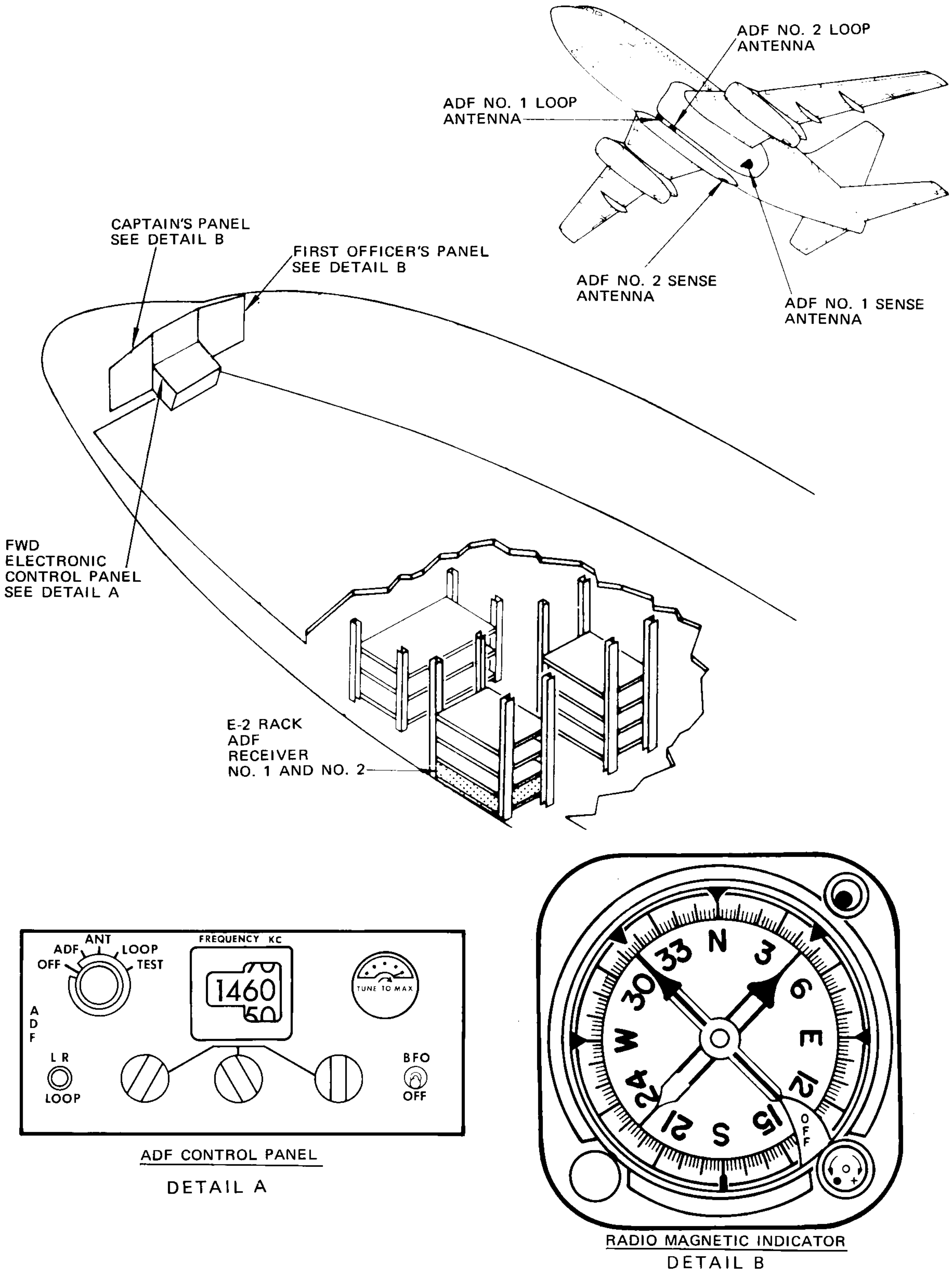
DME INTERROGATOR - REMOVAL/INSTALLATION

1. General
   A. The DME Interrogator is installed in the main equipment center on shelf E2. The unit is shock mounted and held in place by two hold down hooks.
2. Remove DME Interrogator
   A. Open DME 1 and DME 2 (when installed) circuit breakers on P6 and P18 panels.
   B. Disconnect cable connectors on front of interrogator (if required).
   C. Loosen hold-down fasteners at front of unit, and pull out interrogator.
3. Install DME Interrogator
   A. Slide interrogator into rack and secure.
   B. Connect cable connectors on front of interrogator (if required).
   C. Close circuit breakers opened in step 2.A. above.
   D. Test interrogator for satisfactory operation as follows:
      (1) Provide electrical power.
      (2) On appropriate VHF NAV control panel, tune frequency display to local VOR/ILS station. Check captain's and first officer's DME indicator for appropriate readout. If no local VOR/ILS station is available, perform DME system test (Ref 34-55-0 Adjustment/Test).
      (3) Remove electrical power if no longer required.

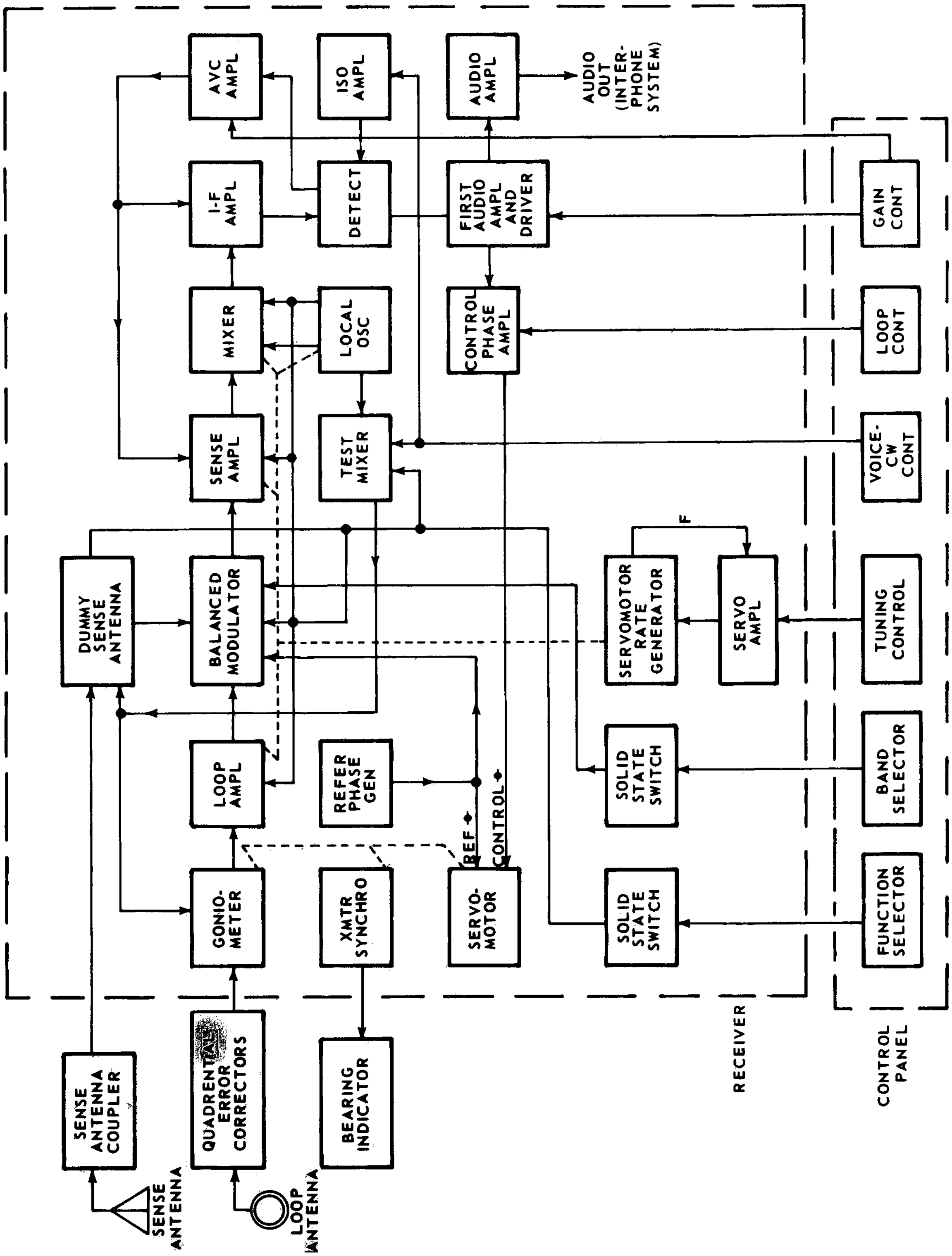
AUTOMATIC DIRECTION FINDER SYSTEMS - DESCRIPTION AND OPERATION

1. General
   A. Two separate and completely independent Automatic Direction Finder (ADF) systems are installed in the airplane. Each system consists of a receiver, sense antenna, sense antenna coupler, fixed loop antenna, radio-magnetic indicator and control panel. (See figure 1.)
   B. ADF systems are used as navigation aids using radio signals from sources in the frequency range of 190 to 1750 kilocycles per second. Sources operating in this frequency range include standard broadcast stations, low frequency radio ranges, etc. The ADF systems may be used for automatic determination of bearing to the station being received, manual determination of bearing, flying radio ranges, or reception of weather and other broadcast programs.
   C. The ADF receivers receive signals from the sense and fixed loop antennas. Circuits in the receiver determine the bearing of radio stations electrically and transmit bearing information as a synchro signal to the radio magnetic indicators. Audio signals from the receivers are monitored through the interphone system.
2. Automatic Direction Finder Controls
   A. Each ADF system is remotely controlled from individual control panels located on the forward electronic control panel. Each control panel contains a function selector switch, loop control switch, frequency selector knobs, volume control, beat frequency oscillator control switch, tuning meter, and a frequency indicating window.
   B. The frequency of the source being received is indicated by a dial so arranged that the frequency may be read directly.
3. Automatic Direction Finder Sense Antennas
   A. The sense antennas located on the wing body fairings provide a non-directional reception pattern and high strength signal to the receivers. (See figure 1.)
4. Automatic Direction Finder Fixed Loop Antennas
   A. Fixed loop antennas are sealed units, flush-mounted from outside in individual antenna cavities on the bottom center of the fuselage at stations 560 and 600 and have no moving parts.
   B. The antenna consists of two pairs of ferrite cored coils, one pair oriented parallel to the fore and aft axis of airplane and a second pair perpendicular to it. Coupling of the rf signals to the receiver is through quadrantal error correctors.
   C. Each quadrantal error corrector has either one or two fixed attenuators and after removing the quadrantal error correction from the signals impresses the signals on the resolver in the receiver. The correctors are installed in their respective antenna cavities and mate with antennas when installed. The correctors can be removed as individual units.

5. Automatic Direction Finder Sense Antenna Input Coupler
   A. An input coupler is used in each of the ADF systems to match the capacitance of the sense antenna to the capacitance of the transmission line between the antenna and receiver. The coupler contains the necessary circuits to provide optimum performance. The total capacitance of the sense antenna, coupler, and line, as seen by receiver input, is approximately 3000 mmfd.
6. Automatic Direction Finder Receiver
   A. Each receiver is completely transistorized and employs a single conversion superheterodyne circuit in the receiver portion. The receiver operates in the range of 190 kilocycles to 1750 kilocycles in three bands. Tuning is by servomotor and the band and function switching is solid- state.
   B. The receiver has inputs from the loop antenna through the quadrantal error corrector and from the sense antenna through its coupler. (See figure 2.) The receiver, through its system of loop and servo-amplifier circuits, converts the received radio signal to a synchro output to operate the radio-magnetic indicators. The receiver sends out audio monitoring signal to the airplane interphone system. (See Chapter 23.)
7. Operation
   A. Operating power for the automatic direction finder systems is taken from circuit breaker panel P18 for ADF No. 1 system and P6 for ADF No. 2 system. (See figure 2.) The systems are energized by closing the ADF and instrument transformers circuit breakers.
   B. Function Selection
      (1) Function Selection is accomplished through a five-position rotary switch having OFF--ADF--ANT--LOOP--TEST positions. When the switch is in the ADF position, circuits are selected to determine automatically the bearing to the station being received. During ADF operation, both the sense and loop antenna circuits operate, and bearing information is displayed on the radio -magnetic indicators.
      (2) When the ANT function is selected, only the sense antenna circuits are utilized, and the receiver is used for reception of audio signals. The ANT function is used for reception of weather broadcasts and radio range signals.
      (3) The LOOP function is used for manual determination of bearing to the station being received. The loop function is electronically controlled by the loop control switch through a servo action and a null in reception is determined by monitoring the tuning meter. The LOOP function may also be used for audio reception under conditions of severe precipitation static, since the loop is shielded electrostatically, and may reduce interference. The LOOP function should not be used on radio ranges since it may give confusing and unreliable reception of range signals.



ADF System Component Location
Figure 1



ADF System Block Diagram
Figure 2

(4) The TEST function is used during inflight test procedures to check operation and calibration of the ADF system. During the calibration test, a zero beat will be heard at a given frequency on each of the three bands and this given frequency will line up with the frequency indicator if the set is calibrated. The operation test checks that the bearing will read 180 degrees throughout the full frequency travel of all three bands.

C. Loop Control

(1) Loop control is provided through a loop control switch and is used only when the LOOP function is selected by the function selector switch. Placing the loop switch to L or R position will transmit a synchro signal to the loop amplifier and radio magnetic indicator. The correct position for slow rotation of loop function is at the midpoint of the switch position. When the switch is in any position but the center position, two nulls can be determined, as the radio magnetic indicator goes through 360 degrees.

(2) Manual loop control is used to determine aural nulls. Determination of the null point is achieved by observing the tuning meter during loop control. The point of minimum deflection of the tuning meter needle is the null point.

D. Frequency Selection

(1) Selection of stations operating in the frequency range of 190 to 1750 kc is accomplished by operating the frequency selector knobs on the control panel. As the tuning knobs are rotated, the frequency appears in indicator window.

(2) Tuning is best accomplished with the function selector in the ANT position. While tuning to the desired frequency, the tuning meter should be observed to determine the point of maximum signal strength.

E. Gain Control

(1) Audio output from the ADF receivers is directed to the interphone system. Audio output level may be controlled from minimum to maximum through the gain control on the remote control panel.

F. Voice - CW Control

(1) Continuous wave signals are made audible by mixing a beat frequency oscillator output with the received signals to produce a tone. The beat frequency oscillator is operative when the BFO-OFF control switch is placed in the BFO position.

(2) Aural determination of bearings may be made more easily when receiving amplitude modulated signal by turning the BFO on. The BFO produces a tone even though the carrier being received is not modulated, and therefore enables accurate determination of the true null.

G. Tuning Meter

(1) A meter on the control panel provides a relative indication of received signal strength. During tuning, the frequency tuning knob is adjusted to produce maximum meter deflection. During manual ADF, the meter provides the most accurate determination of the null, or minimum signal point and enables accurate bearing determination.

AUTOMATIC DIRECTION FINDER SYSTEM - TROUBLESHOOTING

1. General
   A. Troubleshooting the ADF system is best accomplished by performing the ADF System - Adjustment/Test. Satisfactory completion of the adjustment/test procedures indicates that the system is operating properly. When satisfactory operation is not obtained, the tabulated data below should be consulted. If using the tabulated data fails to correct the trouble, airplane wiring should be suspect and repaired or replaced as necessary.
   B. The tabulated data is written for one ADF system. Since two systems are installed, troubleshooting may be accomplished by interchanging components.

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
System completely inoperative	Circuit breaker	Check that system circuit breaker on P18 (ADF-1) or P6 (ADF-2) circuit breaker panel is closed.	Close circuit breaker
	No power available to receiver	Check that 28V dc is available at pin 26 of plug connected to receiver. If power is available, receiver faulty. If power is not available, see following procedure.	Replace receiver
		Check for 28V dc at pin 26 of plug connected to control panel. If available, control panel faulty. If not available, airplane wiring at fault.	Replace control panel
			Refer to electronic wiring diagrams
Incorrect reading on one RMI	RMI		Replace RMI
Incorrect reading on both RMIs	RMI	Monitor audio channel to check that receiver is operating and tuned to known station. If receiver is operative, operate loop antenna. If loop operates, RMI faulty.	Replace RMI
	Control panel or loop antenna	If loop does not operate properly, control panel or antenna faulty.	Replace control panel
			Replace lopp antenna
	Control panel or receiver	Receiver is inoperative or is not tuned to station.	Replace control panel
			Replace receiver

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
No audio	Audio station being monitored inoperative	Check that audio is available at another station.	Refer to AMM Chapter 23, Communications
	Control panel		Replace control panel
	Receiver		Replace receiver

AUTOMATIC DIRECTION FINDER SYSTEM - ADJUSTMENT/TEST

1. Automatic Direction Finder (ADF) Systems Test
   A. General
      (1) Make sure that the interphone system is operational before this test is performed (AMM 23-52-0, Interphone System). The interphone system is used for monitoring audio output of the ADF receivers.
   B. Prepare for Test
      (1) Provide electrical power (AMM 24-22-0/201).
      (2) Make sure that the following circuit breakers on panel P18 are closed:
         (a) ADF No. 1
         (b) Compass System
      (3) Make sure that the following circuit breakers on panel P6 are closed:
         (a) ADF No. 2
         (b) Compass System
         (c) Interphone System
      (4) Turn on radio master power switch.
   C. Test ADF System
      (1) Place ADF-1 function selector switch in ANT and BFO switch in BFO (CW) position. Tune in known radio range station in 200 to 400 kHz frequency range. Begin by tuning well below desired frequency and tune for first peak on tuning meter and BFO tone in interphone. Make sure that indicated tuned frequency is within ± 3.5 kHz of frequency selected. Make sure that audio tone is loud and clear.

         NOTE: If incoming signal is from a questionable station, verify by tuning to an identification signal 1020 Hz above tuned frequency with BFO turned off.

      (2) Place ADF-1 function selector switch in ADF and BFO switch in OFF (VOICE) position. Tune in known station in 1000 to 1400 kHz range. Make sure that indicated turned frequency is within ± 7 kHz of frequency selected. Make sure that audio tone is loud and clear. Make sure that RMI needle indicates bearing to station selected.
      (3) Place ADF-1 function selector switch in LOOP position. Rotate loop switch counterclockwise and then clockwise. Make sure that RMI needle rotates counterclockwise and then clockwise.

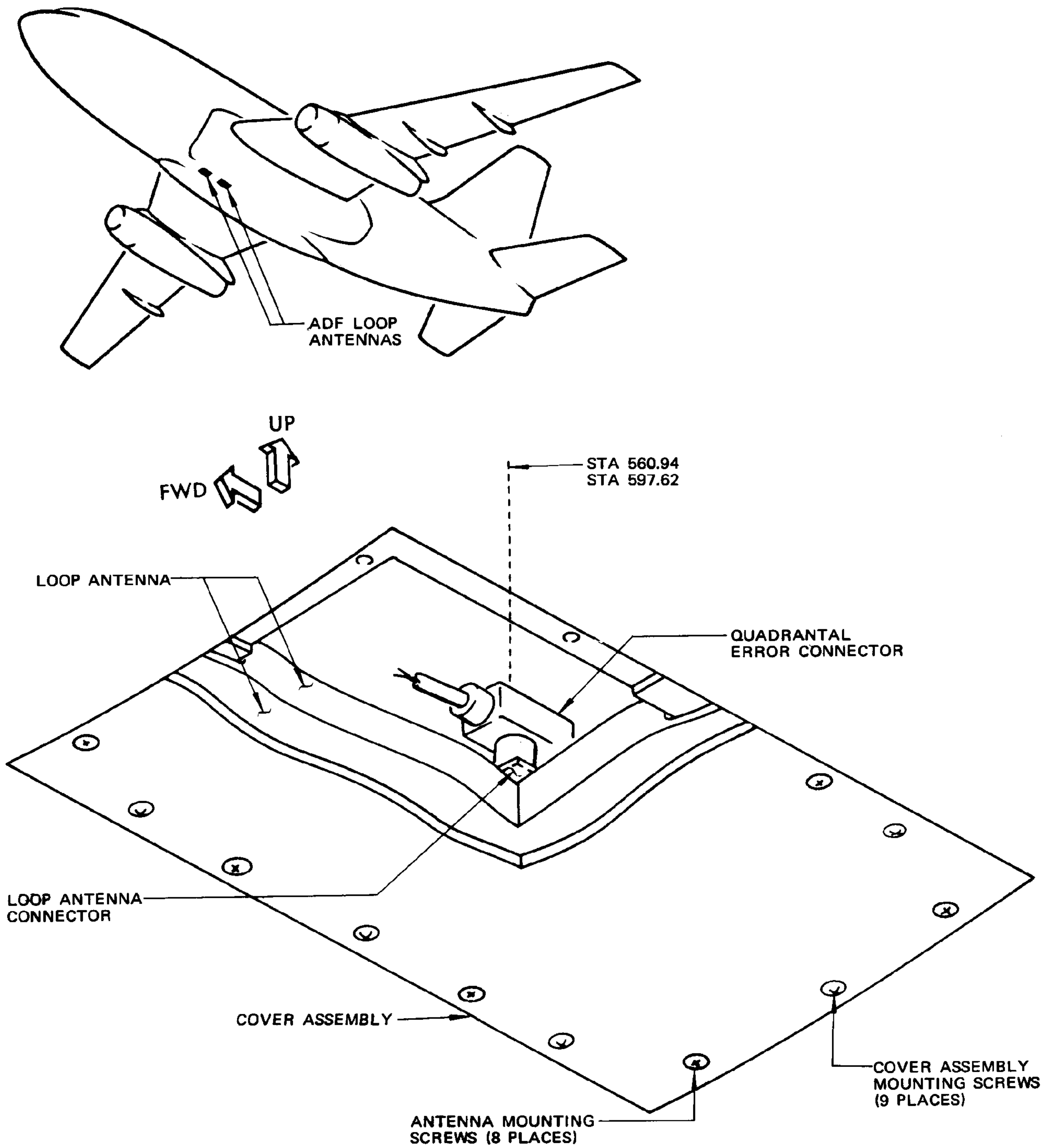
         NOTE: ADF-1 operates narrow needle and ADF-2 operates broad needle. Make sure that fast rotation of RMI needle occurs when loop switch is rotated fully counterclockwise or clockwise.

      (4) Place BFO switch in BFO (CW) position. Make sure that audio tone is minimum when indicated station bearing is 0 or 180 degrees from station bearing.

(5) Rotate loop switch for RMI indication 170 degrees from station bearing. Make sure that RMI needle rotates without jitter.
(6) Place ADF-1 function selector switch in ADF position. Make sure that RMI needle returns smoothly to station bearing.
(7) Operate volume (gain) control in ADF, LOOP, and ANT modes. Make sure that audio tone level changes smoothly.
(8) Repeat steps (1) thru (7) for ADF-2.
(9) While tuned to station, make sure that audio tone is available at each audio selector panel.
(10) If no longer required, remove electrical power from airplane.

AUTOMATIC DIRECTION FINDER LOOP ANTENNAS - REMOVAL/INSTALLATION

1. Remove Antenna
   A. Remove loop antenna cover by removing mounting screws (Fig. 401).
   B. Support loop antenna assembly. Remove mounting screws.
   C. Lower loop antenna assembly.
   D. Disconnect antenna connector from quadrantal compensator assembly.
   E. Remove antenna assembly.
2. Install Antenna
   A. Clean mating surfaces of antenna and airplane structure to provide an electrical bond.
   B. Apply a protective coating of corrosion preventive compound to mating surfaces of antenna and airplane structure (Ref 51-21-91 CP).
   C. Connect antenna connector to quadrantal compensator assembly.
   D. Position loop antenna and secure all but one of the mounting screws.
   E. Do an electrical bonding check between antenna and airplane structure per 20-22-01.
      (1) Connect the ohmmeter or the resistance measuring bridge between the empty antenna mounting screw hole (antenna base plate) and the airplane skin.
      (2) Make sure the resistance is not more than 0.1 ohm.
      (3) Install and tighten the last mounting screw.
   F. Hold antenna cover in place and install the screws.
   G. Perform operational check of system (Ref 34-57-0 A/T).



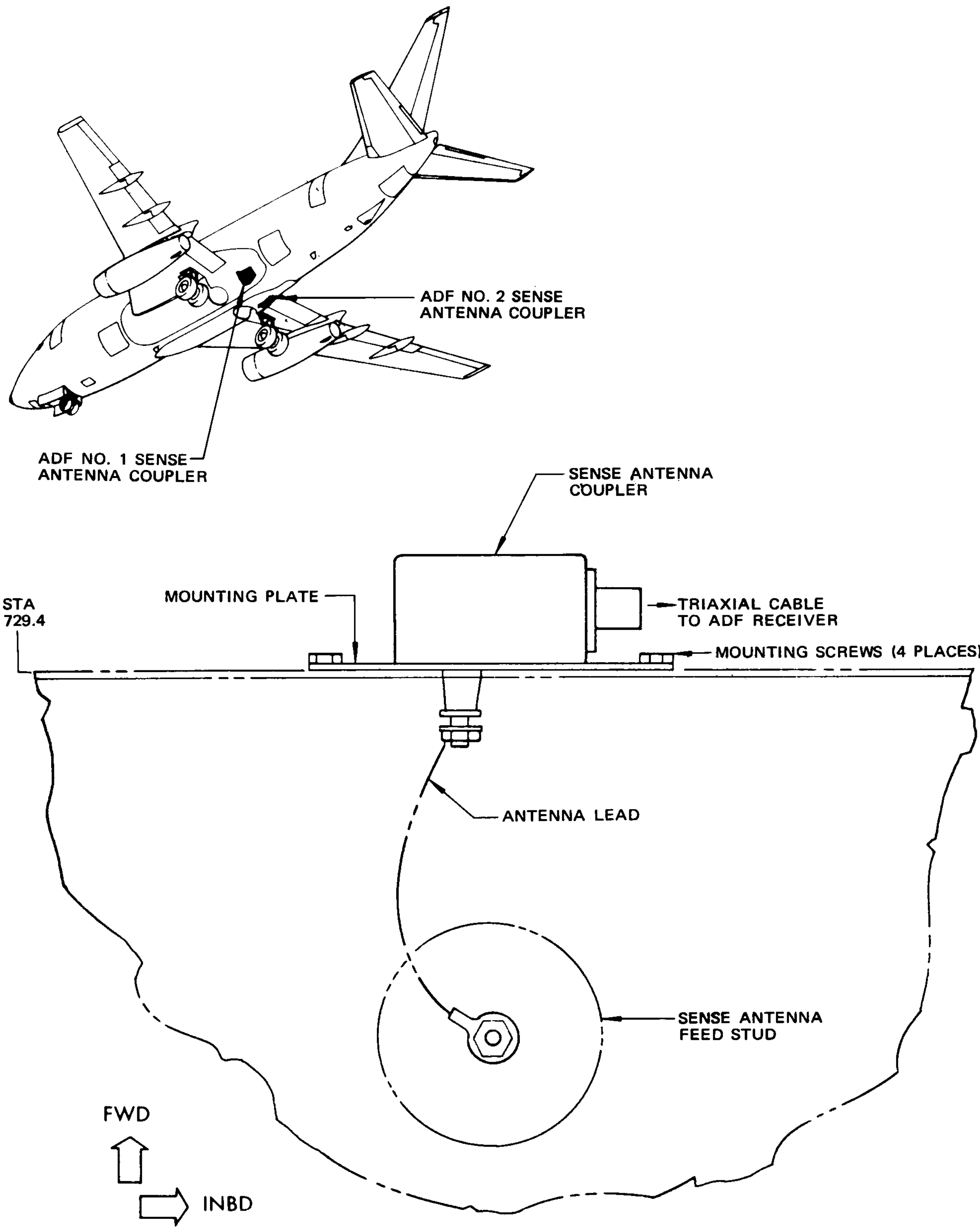
ADF Loop Antenna Installation
Figure 401

ADF SENSE ANTENNA COUPLERS - REMOVAL/INSTALLATION

1. General
   A. The sense antenna couplers may be removed from the airplane as individual units (Fig 401).
2. Remove Coupler
   A. Gain access to the couplers located at the aft outboard corner of the main wheel well.
   B. Disconnect electrical connector from inboard end of coupler.
   C. Support coupler and remove four mounting screws.

      CAUTION: DO NOT PULL COUPLER. WIRE TERMINATIONS MAY BE DAMAGED.

   D. Disconnect antenna wire from aft face of coupler.
   E. Remove coupler.
3. Install Coupler
   A. Connect antenna wire to coupler.
   B. Locate coupler and install four mounting screws.
   C. Connect electrical connector to coupler.
   D. Perform operational check of system (Refer 34-57-0, Automatic Direction Finder System - Adjustment/Test).



ADF Sense Antenna Coupler Installation
Figure 401

ADF SENSE ANTENNA - REMOVAL/INSTALLATION

1. General
   A. The ADF sense antenna is flame-sprayed on the reinforced wing body fairing. The metal coating of the fairing is used as the antenna element, and is not removable from the fairing.
   B. To remove or replace the wing body fairing, refer to 53-51-21, Aft Wing-to-Body Fairing - Removal/Installation.