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KING AIR 300/350 PILOT TRAINING MANUAL

Record of Revision No. 1

This is a complete reprint of the *King Air 300/350 Pilot Training Manual*.

The portion of the text or figure affected by the current revision is indicated by a solid vertical line in the margin. A vertical line adjacent to blank space means that material has been deleted. In addition, each revised page is marked "Revision 1" in the lower left or right corner.

The changes made in this revision will be further explained at the appropriate time in the training course.



KING AIR 300/350 PILOT TRAINING MANUAL

**VOLUME 2
AIRCRAFT SYSTEMS**

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FOR TRAINING PURPOSES ONLY

NOTICE

The material contained in this training manual is based on information obtained from the aircraft manufacturer's Pilot Manuals and Maintenance Manuals. It is to be used for familiarization and training purposes only.

At the time of printing it contained then-current information. In the event of conflict between data provided herein and that in publications issued by the manufacturer or the FAA, that of the manufacturer or the FAA shall take precedence.

We at FlightSafety want you to have the best training possible. We welcome any suggestions you might have for improving this manual or any other aspect of our training program.

FOR TRAINING PURPOSES ONLY

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CHAPTER 1

AIRCRAFT GENERAL

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CHAPTER 1

AIRCRAFT GENERAL



INTRODUCTION

A good basic understanding of the airplane will help in studying the individual systems and their operation. This section provides the basic and background information helpful in learning the details of airplane operation and its systems and performance in other sections of this training manual.

DESCRIPTION

The General Airplane Section of this manual presents an overall view of the airplane, including external familiarization, cabin arrangements, and cockpit layout. In this section you will find diagrams and data describing the airplane and its systems that may not be included in the *Pilot's Operating Handbook*.

SUPER KING AIR 300 AND 350 (B300) DESCRIPTION

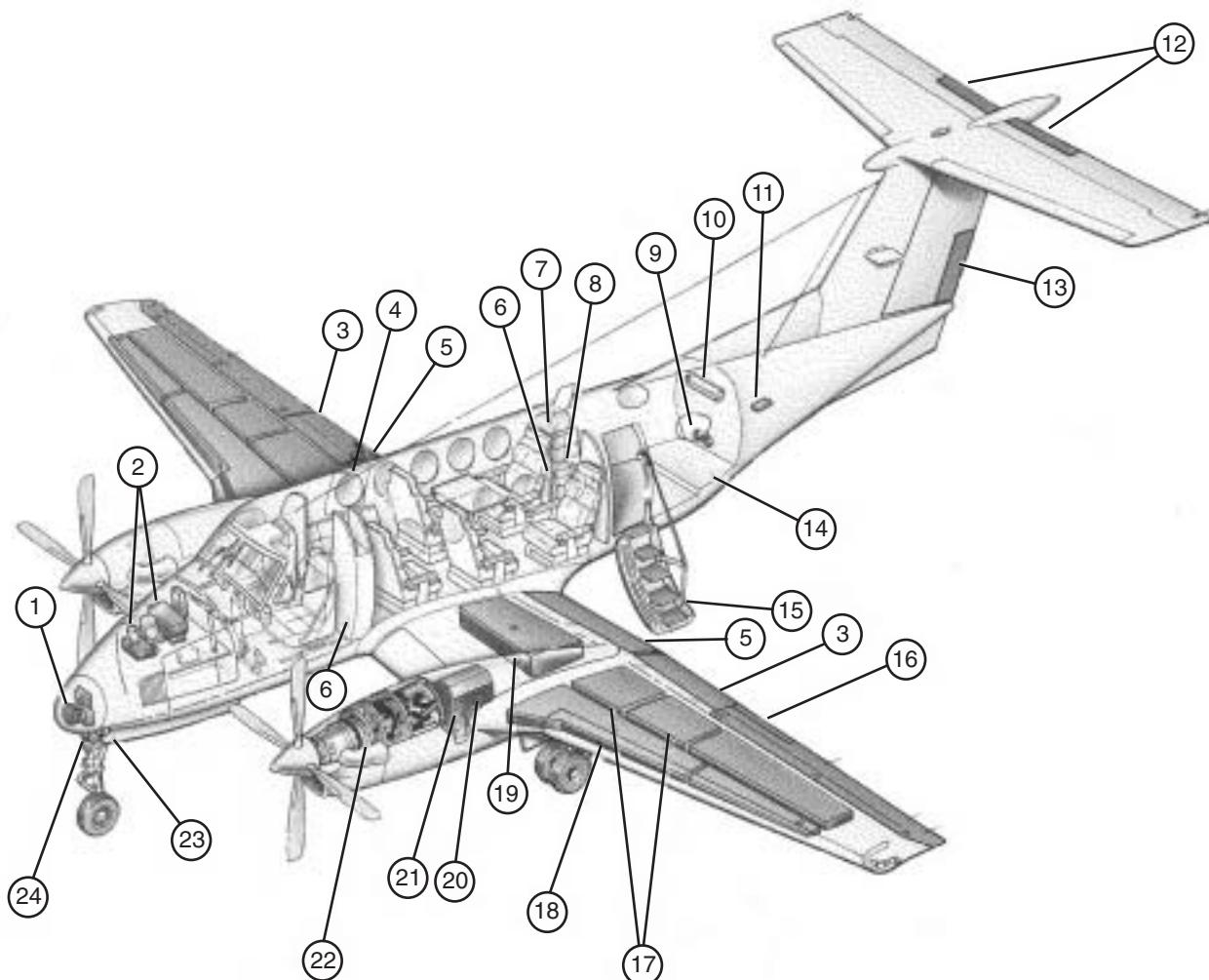
The Beechcraft Super King Air 300 and 350 are high-performance T-tail pressurized twin-engine turboprop airplanes (Figures 1-1, 1-2, and 1-3, and 1-4). They are designed and equipped for flight in IFR conditions, day or night, and into



Figure 1-1 Super King Air 300

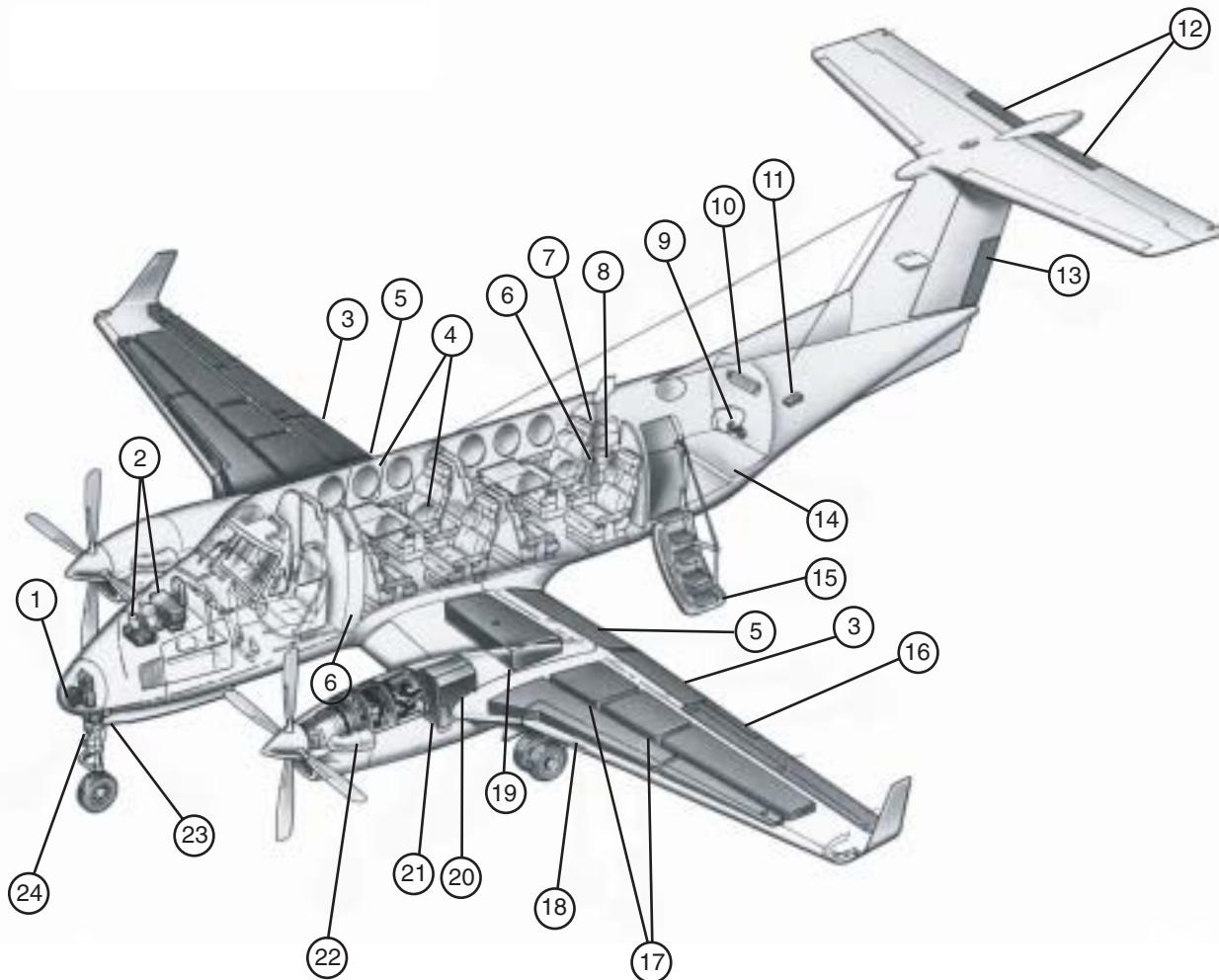


Figure 1-2 Super King Air 350



1. Weather Radar Antenna
2. Communications, Navigation and Radar Equipment
3. Outboard Flap Section
4. Ground Escape Hatch
5. Inboard Flap Section
6. Liquid Storage Cabinet
7. Lavatory Privacy Curtain
8. Belted Lavatory
9. Pressurization Safety and Outflow Valves
10. Oxygen Bottle
11. Emergency Locator Transmitter
12. Elevator Trim Tabs
13. Rudder Trim Tab
14. Baggage Area
15. Airstair Door
16. Aileron Trim Tab
17. Box Section Fuel Tanks
18. Leading Edge Fuel Tanks
19. Auxiliary Fuel Tank
20. Wing Ice Check Light
21. Nacelle Fuel Tank
22. PT6A Turboprop Engine
23. Heated Pitot Mast
24. Landing and Taxi Lights

Figure 1-3 Super King Air 300 General Arrangement



- | | |
|---|-----------------------------|
| 1. Weather Radar Antenna | 13. Rudder Trim Tab |
| 2. Communications, Navigation and Radar Equipment | 14. Baggage Area |
| 3. Outboard Flap Section | 15. Airstair Door |
| 4. Ground Escape Hatch | 16. Aileron Trim Tab |
| 5. Inboard Flap Section | 17. Box Section Fuel Tanks |
| 6. Liquid Storage Cabinet | 18. Leading Edge Fuel Tanks |
| 7. Lavatory Privacy Curtain | 19. Auxiliary Fuel Tank |
| 8. Belted Lavatory | 20. Wing Ice Check Light |
| 9. Pressurization Safety and Outflow Valves | 21. Nacelle Fuel Tank |
| 10. Oxygen Bottle | 22. PT6A Turboprop Engine |
| 11. Emergency Locator Transmitter | 23. Heated Pitot Mast |
| 12. Elevator Trim Tabs | 24. Landing and Taxi Lights |

Figure 1-4 Super King Air 350 General Arrangement



known icing conditions. They are also capable of operating in and out of small airports within the POH operating limits. The Super King Air design is a blend of a highly efficient airframe with proven current technology components providing a reliable, economical, versatile, and cost-productive airplane.

The structure is an all-metal low-wing monoplane. It has fully cantilevered wings and a T-tail empennage. The wings are an efficient, high-aspect ratio design. The airfoil section provides an excellent combination of low drag for cruise conditions and easy handling for low speed terminal conditions or small airport operations. The 350 has NASA designed winglets installed which further improves performance.

A faired oval minimum frontal area nacelle is installed on each side of the wing center section to house both the engine and landing gear. The nacelles are designed and located to maximize propeller-to-ground clearance, minimize cabin noise, and provide a low drag installation of the powerplants on the wing. The "pitot" type intakes (Figure 1-5) and the smaller frontal area of the exhaust stacks boost performance by reducing drag.

The distinctive T-tail on Beechcraft Super King Airs was designed to provide improved aerodynamics, lighter control forces, and a wider CG range. Wind tunnel testing, extensive flight testing, and the experience of thousands of corporate pilots have confirmed that the combination of T-tail and straight wing provides excellent control characteristics in all flight regimes.

The fuselage is a conventional monocoque structure using high strength aluminum alloys. The basic cross-sectional shape of the cabin is a favorable compromise between passenger comfort and efficient cruise performance. The cabin profile is squared-oval, not round. Passengers can sit comfortably without leaning their heads to accommodate sloping walls. The floors are flat from side to side for passenger ease in entering and leaving the cabin.



Figure 1-5 Super King Air 300 and 350 Nacelle Pitot Intake

SUPER KING AIR 300 CONFIGURATION

Beechcraft Super King Air 300's are certificated for up to 15 people, but the normal corporate configuration is from six to ten people plus a crew of 2 (Figures 1-6 and 1-7). Almost any arrangement is possible.

In addition to the standard airplane configurations, Beechcraft offers many optional items which are available at additional cost and weight. The basic configurations, dimensions, weights, and specifications are summarized on pages 1-8 and 1-9. Refer to the respective airplane *POH* for detailed, up-to-date information.

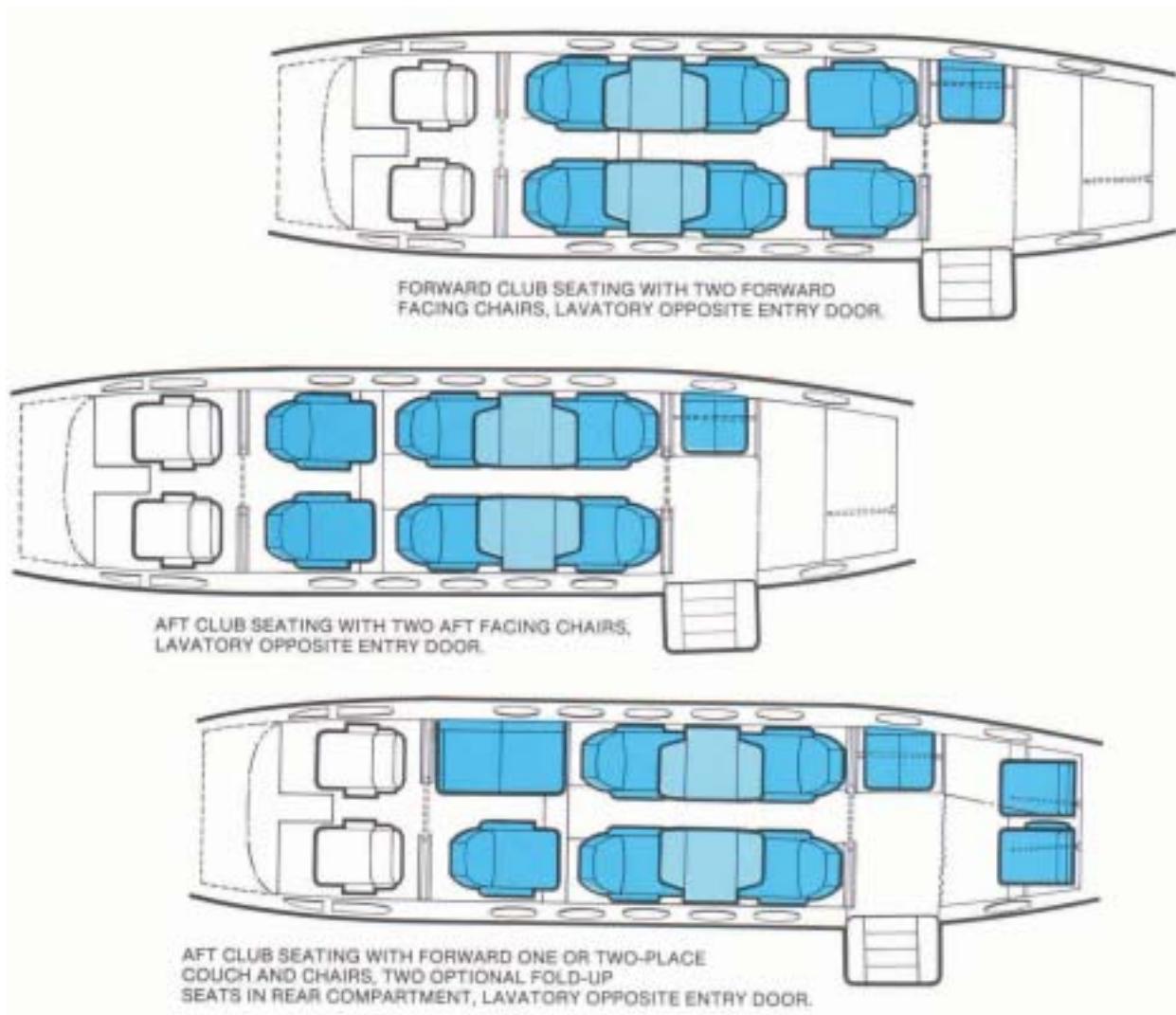


Figure 1-6 Super King Air 300 Cabin Seating Arrangement

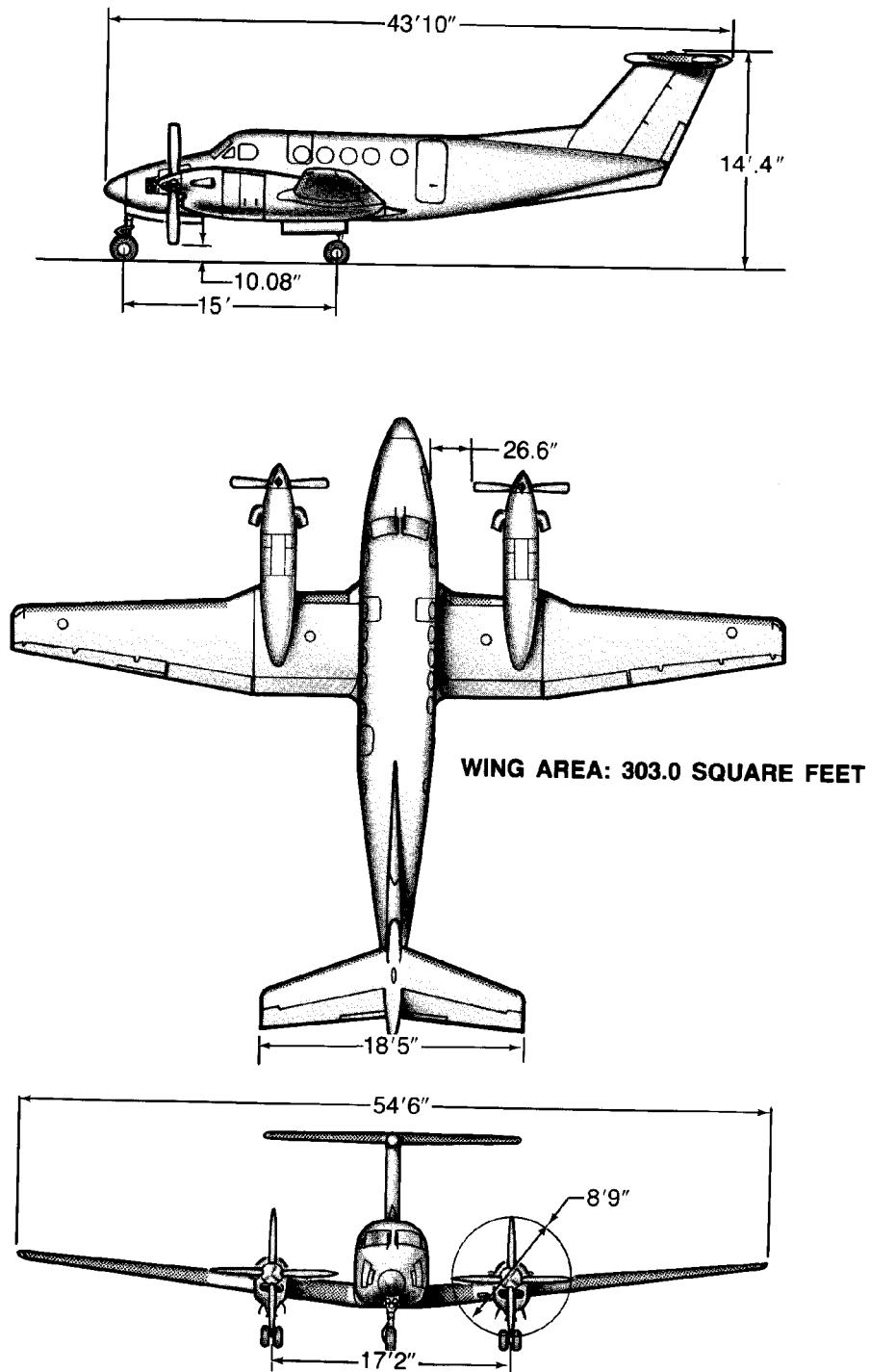


Figure 1-7 Super King Air 300 Dimensions



Specifications—Super King Air 300

Crew—FAA Certificated 1

Except where otherwise prescribed by the appropriate operating regulations, one pilot for all operations with equipment limitations.

Occupancy—Max. FAA Cert. (incl. crew)..... 15

Passengers—Normal Configuration 6 to 10

Engines—P & W Turboprop,
1050 SHP 2 PT6A-60A

Propellers—4 Blade, Reversible 2 Hartzell

Landing Gear - Retractable, Tricycle,
Dual Main Wheels.....Hydraulic

Wing Area.....303 sq. ft.

Maximum Certificated Weights

Maximum Ramp Weight 14,100 pounds

Maximum Takeoff Weight..... 14,000 pounds

Maximum Landing Weight..... 14,000 pounds

Maximum Zero Fuel Weight 11,500 pounds

Maximum Weight in Baggage Compartment:

With Fold-up Seats..... 510 pounds

Without Fold-up Seats..... 550 pounds

Cabin and Entry Dimensions

Cabin Width (Maximum)54 inches

Cabin Length (Maximum
between pressure bulkheads).....22 feet

Cabin Height (Maximum)57 inches

Airstair Entrance Door Width
(Minimum)26.75 inches

Airstair Entrance Door Height
(Minimum)51.5 inches

Pressure Vessel Volume 393 cubic feet

Potential Cargo Area Volume 253 cubic feet

Specific Loadings

Wing Loading: 46.2 pounds per square foot

Power Loading: 6.7 pounds per shaft horsepower

Super King Air 300 Airspeeds

Emergency Airspeeds (14,000 Pounds)

One-Engine-Inoperative Best
Angle-of-Climb (V_{XSE})120 KIAS

One-Engine-Inoperative Best
Rate-of-Climb (V_{YSE})122 KIAS

Air Minimum Control Speeds (V_{MCA}):

Flaps Up92 KIAS

Flaps Approach86 KIAS

Emergency Descent181 KIAS

One-Engine-Inoperative Landing:

Flaps Up131 KIAS

Flaps Down105 KIAS

Intentional One-Engine-
Inoperative (V_{SSE})110 KIAS

Maximum Range Glide134 KIAS



Additional Airspeed Limits

Maneuvering Speed (V_A) 181 KIAS

Maximum Flap Extension and Extended Speeds (V_{FE}):

Approach 200 KIAS

Full Down 157 KIAS

Maximum Landing Gear Extended Speed (V_{LE}) 181 KIAS

Maximum Landing Gear Operating Speeds (V_{LO}):

Extension 181 KIAS

Retraction 163 KIAS

Maximum Operating Speed

V_{MO} 259 KIAS

M_{MO} 58 Mach

Airspeeds for Safe Operation (14,000 Pounds)

Maximum Demonstrated Crosswind Component 25 KIAS

Takeoff Flaps Up:

Rotation (V_1) 112 KIAS

50-ft Speed (V_2) 120 KIAS

Takeoff Flaps Approach:

Rotation (V_1) 101 KIAS

50-ft Speed (V_2) 107 KIAS

Two-Engine Best Angle-of-Climb (V_X) 122 KIAS

Two-Engine Best Rate-of-Climb (V_Y) 137 KIAS

Cruise Climb:

Sea Level to 10,000 feet 160 KIAS

10,000 to 20,000 feet 140 KIAS

20,000 to 25,000 feet 130 KIAS

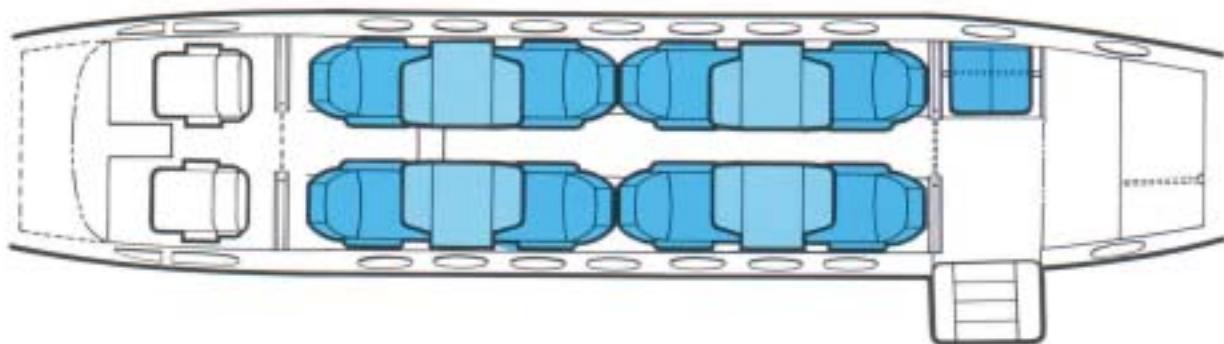
25,000 to 35,000 feet 120 KIAS

Turbulent Air Penetration 170 KIAS

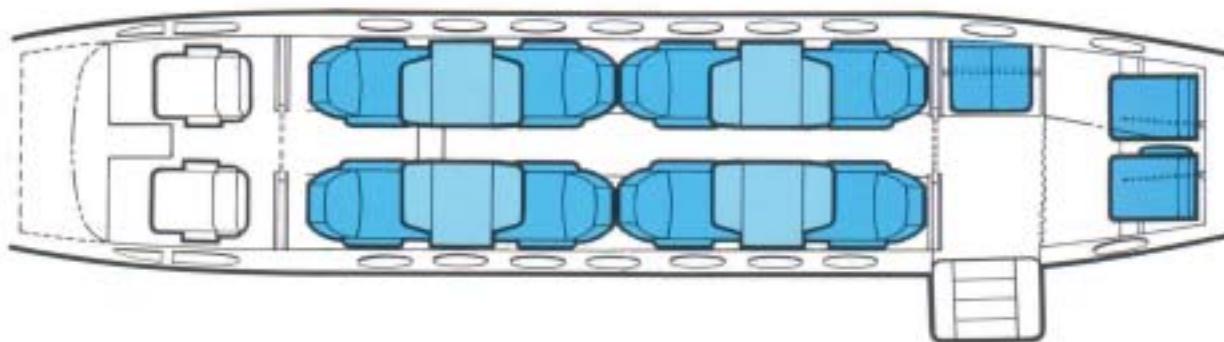
SUPER KING AIR 350 (B300) CONFIGURATION

Beechcraft Super King Air 350's are certificated for up to 17 people (15 passengers and 2 crew), but the normal corporate configuration is from nine to eleven people (Figures 1-8 and 1-9). Almost any arrangement is possible.

In addition to the standard airplane configurations, Beechcraft offers many optional items which are available at additional cost and weight. The basic configurations, dimensions, weights, and specifications are summarized on pages 1-12 and 1-13. Refer to the respective airplane *POH* for detailed, up-to-date information.



NINE PASSENGER SEATING CONFIGURATION



ELEVEN PASSENGER SEATING CONFIGURATION

Figure 1-8 Super King Air 350 Cabin Seating Arrangement

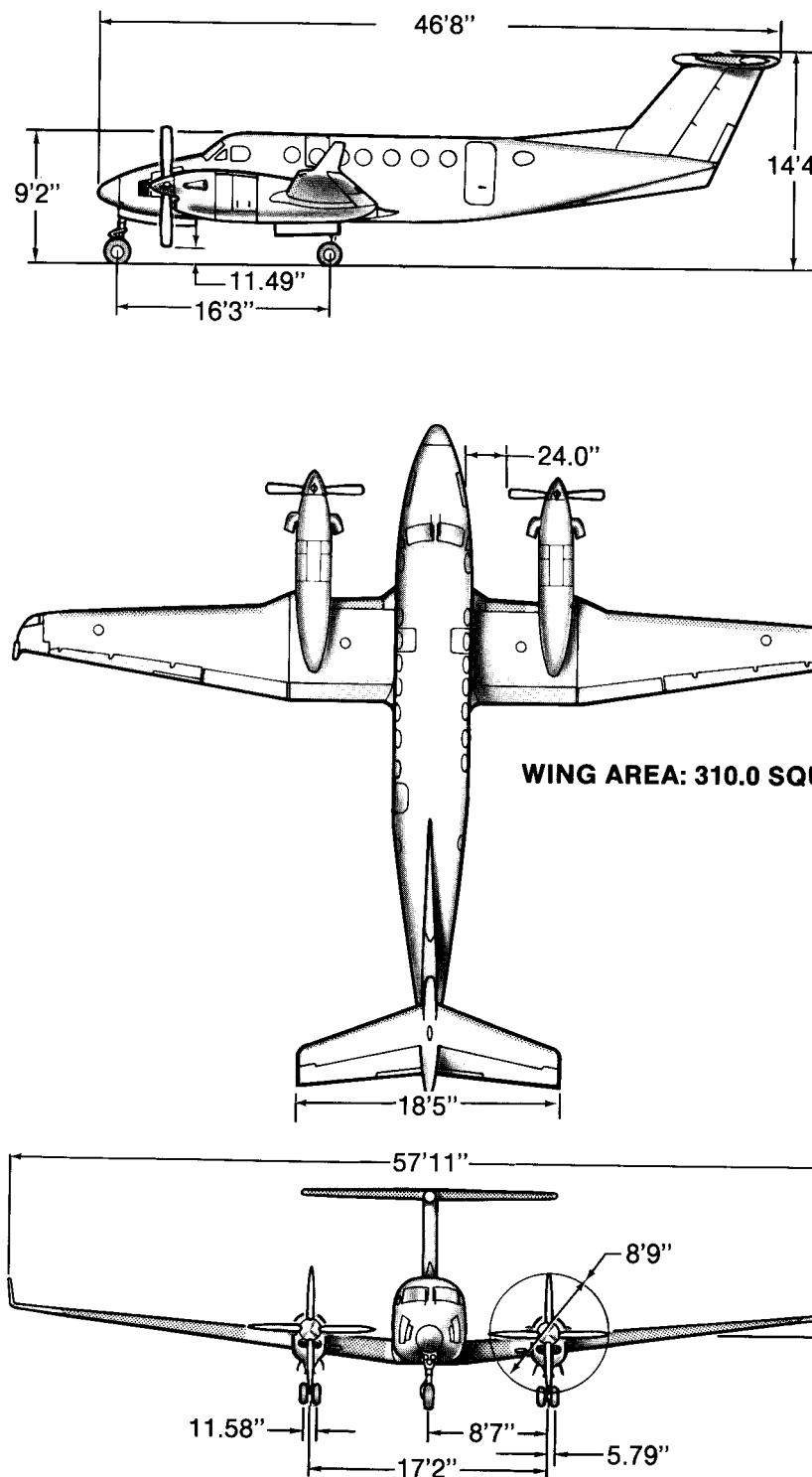


Figure 1-9 Super King Air 350 Dimensions



Specifications—Super King Air 350 (B300)

Crew—FAA Certificated 1

Except where otherwise prescribed by the appropriate operating regulations, one pilot with FAA approved passenger seating configurations of nine or less; or one pilot and one co-pilot for all other approved configurations.

Occupancy—Max. FAA Cert. (incl. crew) 17

Passengers—Normal Configuration 9 to 11

Engines—P & W Turboprop,
1050 SHP 2 PT6A-60A

Propellers—4 Blade, Reversible 2 Hartzell

Landing Gear—Retractable, Tricycle,
Dual Main Wheels Hydraulic

Wing Area 310 sq. ft.

Maximum Certificated Weights (Commuter Category)

Maximum Ramp Weight 15,100 pounds

Maximum Take-off Weight 15,000 pounds

Maximum Landing Weight 15,000 pounds

Maximum Zero Fuel Weight 12,500 pounds

Maximum Weight in Baggage Compartment:

With Foldup Seats 510 pounds

Without Fold-up Seats 550 pounds

Cabin and Entry Dimensions

Cabin Width (Max.) 54 inches

Cabin Length (Max. between pressure bulkheads) 24 feet, 10 inches

Cabin Height (Max.) 57 inches

Airstair Entrance Door Width (Min.) 26.75 inches

Airstair Entrance Door Height (Min.) 51.5 inches

Cargo Door Width 49 inches

Cargo Door Height 52 inches

Pressure Vessel Volume 443 cubic feet

Potential Cargo-area volume 303 cubic feet

Specific Loadings

Wing Loading: 48.4 pounds per square foot

Power Loading: 7.14 pounds per shaft horsepower

Super King Air 350 (B300) Airspeeds

Emergency Airspeeds (15,000 Pounds)

One-Engine-Inoperative Best Angle-of-Climb (V_{XSE}) 125 KIAS

One-Engine-Inoperative Best Rate-of-Climb (V_{YSE}) 125 KIAS

Air Minimum Control Speeds (V_{MCA}):

Flaps Up 94 KIAS

Flaps Approach 93 KIAS

One-Engine-Inoperative

Enroute Climb 125 KIAS

Emergency Descent 184 KIAS

Maximum Range Glide 135 KIAS

Additional Airspeed LimitsManeuvering Speed (V_A) 184 KIASMaximum Flap Extension and Extended Speeds (V_{FE}):

Approach 202 KIAS

Full Down 158 KIAS

Maximum Landing Gear

Extended Speed (V_{LE}) 184 KIAS

Maximum Landing Gear

Operating Speeds (V_{LO})

Extension 184 KIAS

Retraction 166 KIAS

Maximum Operating Speed

 V_{MO} 263 KIAS M_{MO} 58 Mach**Airspeeds for Safe Operation (15,000 Pounds)**

Maximum Demonstrated Crosswind Component 20 KIAS

Two-Engine Best Angle-of-Climb (V_X) 125 KIASTwo-Engine Best Rate-of-Climb (V_Y) 140 KIAS

Cruise Climb:

Sea Level to 10,000 feet 170 KIAS

10,000 to 15,000 feet 160 KIAS

15,000 to 20,000 feet 150 KIAS

20,000 to 25,000 feet 140 KIAS

25,000 to 30,000 feet 130 KIAS

30,000 to 35,000 feet 120 KIAS

Turbulent Air Penetration 170 KIAS

Intentional One-Engine Inoperative Speed (V_{SSE}) 110 KIAS

There is an overspeed warning horn installed on all King Air 350's. It sounds when the airspeed exceeds the barber pole by no more than 6 knots or .01 mach, whichever is less. A test switch installed on the co-pilot's left sub-panel allows the pilot to test the over-speed warning prior to flight.



AIRSTAIR ENTRANCE

The cabin entry airstair door is on the left side of the fuselage, just aft of the wing (Figure 1-10). A swing-down door, hinged at the bottom, provides a convenient stairway for entry and exit. Two of the four steps are movable and automatically fold flat against the door in the closed position. A self-storing platform automatically folds down over the doorsill when the door opens to provide a stepping platform for door seal protection. A plastic encased cable provides support for the door in the open position, a handhold for passengers, and a means of closing the door from inside the airplane. Additional handhold cable is available as an option. A hydraulic damper permits the door to lower gradually during opening. It is important that not more than one person be on the airstair door at a time. Excessive weights could damage the door attach fittings.

Airstair Locking Mechanism

The door locking mechanism is operated by either of the two vertically staggered handles, one inside and the other outside the door. The inside and outside handles are mechanically interconnected. When either handle is rotated per placard instructions, two bayonet pins on each side of the door and two hooks at the top of the door engage the door frame to secure the door. A button next to the door handle must be depressed before the handle can be rotated to open the door. The door can be locked with a key for airplane security on the ground.

When unlocking the door, the release button adjacent to the door handle must be held depressed before the handle can be rotated to unlock the door. As an additional safety measure, a differential pressure-sensitive diaphragm is incorporated into the release button mechanism.

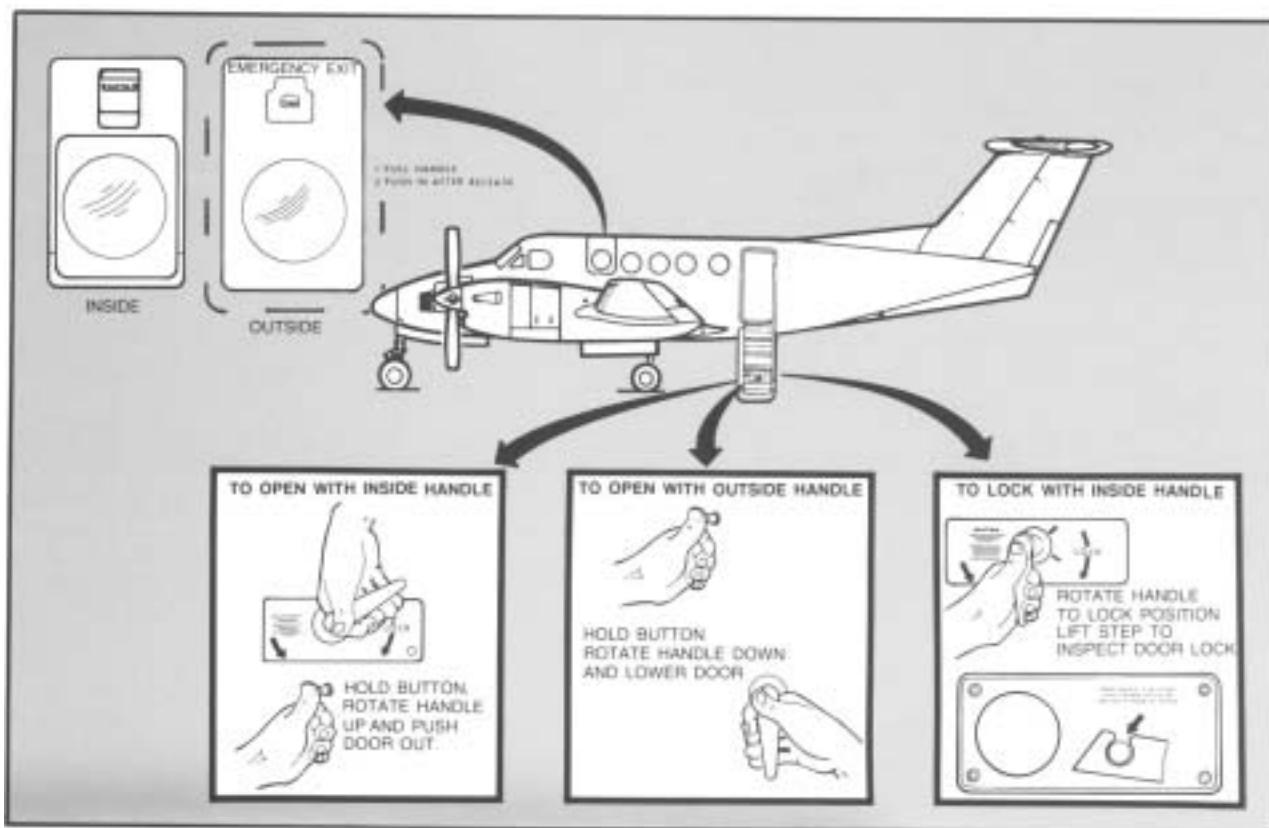


Figure 1-10 Entrance and Exit Provisions



To secure the airstair door inside, rotate the handle clockwise as far as it will go. The release button should pop out, and the handle should be pointing down. Check the security of the airstair door by attempting to rotate the handle counter-clockwise without depressing the release button; the handle should not move. Next lift the folded airstep that is just below the door handle. Ensure that the safety lock is in position around the diaphragm shaft when the handle is in the locked position. To observe this area, depress a red switch near the window that illuminates a lamp inside the door. If the arm is properly positioned around the shaft, proceed to check the indication in each of the visual inspection ports located near each corner of the door (Figure 1-11).



Figure 1-11 Visual Inspection Ports

Ensure the green stripe on the latch bolt is aligned with the black pointer in the visual inspection port. Check the upper door hook engagement by viewing the hooks through two inspection openings in the headliner just above the fore and aft upper corners of the door. The hook engagement areas are illuminated by depressing the CABIN DOOR HOOK, OBSV LT SW button located between the two

inspection openings in the headliner. Perform the “Cabin Door Annunciator Circuitry Check” in the Normal Procedures section prior to the first flight of the day. If any condition specified in this door-locking procedure is not met, DO NOT TAKEOFF.

WARNING

Never attempt to unlock or check the security of the door in flight. If the CABIN DOOR annunciator illuminates in flight, or if the pilot has any reason to suspect that the door may not be securely locked, all occupants should be instructed to remain seated with their seatbelts fastened and the cabin pressure should be reduced to the lowest practical value (considering altitude first). After the airplane has made a full-stop landing, only a crewmember should check the security of the airstair door. Perform the “Cabin Door Annunciator Circuitry Check” in the Normal Procedures section prior to the first flight of the day. If any condition specified in this door-locking procedure is not met, DO NOT TAKEOFF.

EMERGENCY EXITS

The emergency exits are located on the left and right side of the fuselage at the forward ends of the passenger compartment. From the inside, the hatches are released with a pull-down handle, placarded EXIT-PULL. From the outside, the hatches are released with a flush-mounted, pull-out handle. The nonhinged, plug-type hatches can be removed completely from the frame into the cabin when the latches are released.

The hatch can be locked so that it cannot be removed or opened from the outside using the flush-mounted pull-out handle. The hatch is locked when the lock-lever (inside) is in the down or locked position. Locking the hatch is for security when the airplane is parked; the lock-lever should be in the up or unlocked position prior to



flight, to allow removal of the hatch from the outside in the event of an emergency. Removal of the hatch from the inside is possible at all times using the EXIT-PULL handle, since this handle is not locked by the lock-lever. An exit lock placard is placed on the lock-lever so that it can be read when the lever is in the locked position.

350C (B300C) Cargo Door Airstair Entrance

The airstair door is built into the cargo door. It is hinged at the bottom, and swings downward when opened. It has a stairway built onto the inboard side. Two of the staissteps fold flat against the door when the door is closed. When the door is opened, a selfstoring platform automatically folds down over the door sill to protect the rubber door seal. A hydraulic damper ensures that the door will swing down slowly when it opens. While the door is open, it is supported by a plastic-encased cable, which also serves as a handrail. Additionally, this cable is utilized when closing the door from inside the airplane. An inflatable rubber seal is installed around the perimeter of the door, and seats against the door frame as the door is closed. When the cabin is pressurized, air seeps into the rubber seal through small holes in the outboard side of the seal. The higher the cabin differential pressure, the more the seal inflates. This is a passive-seal system with no mechanical connection to a bleed air source. The outside door handle can be locked with a key, for security of the airplane on the ground.

CAUTION

Only one person should be on the airstair door stairway at any one time.

The door locking mechanism is operated by rotating either the outside or the inside door handle, both of which move simultaneously. Three hollow, crescent latches on each side of the door rotate to capture or release latch posts mounted in the cargo door to secure the airstair door. When locked, the airstair door becomes an integral part of the cargo door.

Whether unlocking the door from the outside or the inside, the release button adjacent to the door handle must be held depressed before the handle can be rotated (counterclockwise from inside the airplane, clockwise from outside) to unlock the door. Consequently, unlocking the door is a two-hand operation requiring deliberate action. The release button acts as a safety device to help prevent accidental opening of the door. As an additional safety measure, a differential-pressure-sensitive diaphragm is incorporated into the release-button mechanism. The outboard side of the diaphragm is open to atmospheric pressure, the inboard side to cabin air pressure. As the cabin-to-atmospheric pressure differential increases, it becomes increasingly difficult to depress the release button, because the diaphragm moves inboard when either the outboard or inside release button is depressed.

350C (B300C) CARGO DOOR

A large, swing-up cargo door, hinged at the top, provides access for the loading of large items. The cargo door lock system is operated by two handles: one in the upper aft area of the door, and the other in the lower forward area of the door. Two separate access covers must be opened in order to operate the two handles.

In order to move the upper aft handle out of the locked position, depress the black release button in the handle and rotate the yellow handle upward as far as it will go. This movement is transmitted via cables to two hollow, crescent latches on the forward side and two on the aft side of the cargo door. The latches rotate to release latch posts mounted in the cargo door frame.

In order to move the lower lock handle out of the locked position (forward), lift the orange lock hook from the stud on the yellow lock handle, and rotate the handle aft as far as it will go. This movement is transmitted via linkage to four latch pins on the bottom of the cargo door. The pin-



move aft to disengage latch lugs mounted at the bottom of the cargo door frame.

CAUTION

After unlocking the bottom latch pins, close the forward lock handle access cover. If this cover is left open, it will rotate on its hinge until a portion of it extends below the bottom of the cargo door when the cargo door is opened. Then, when the cargo door is subsequently closed, the access cover will be broken.

To open the cargo door after it is unlocked, push out on the bottom of the door. After the cargo door is manually opened a few feet, gas springs take over and raise the door to the fully open position.

To close the cargo door, pull it down and inboard. The gas springs will resist the closing effort until the door is only open a few feet. Then, as the springs move over center, they begin applying a closing force to the door.

An inflatable rubber seal is installed around the perimeter of the cargo door, and seats against the door frame when closed. When the cabin is pressurized, air seeps into the rubber seal through small holes in the outboard side of the seal. The higher the cabin differential pressure, the more the seal inflates. This is a passive seal system and has no mechanical connection to a bleed air source.

There are no lock handles on the outside of the cargo door, so it can be opened and closed from inside the airplane only.

WARNING

Never attempt to unlock or check the security of the door in flight. If the DOOR UNLOCKED annunciator illuminates in flight, or if the pilot has any reason to suspect that the door

may not be securely locked, all occupants should be instructed to remain seated with their seatbelts fastened and the cabin pressure should be reduced to the lowest practical value (considering altitude first). After the airplane has made a full-stop landing, only a crewmember should check the security of the airstair door. Perform the "Cabin Door Annunciator Circuitry Check" in the Normal Procedures section prior to the first flight of the day. If any condition specified in this door-locking procedure is not met, DO NOT TAKE OFF.

FLIGHT DECK

The pilot and copilot sit side-by-side in individual chairs, separated by the control pedestal. The seats are adjustable fore and aft, vertically, and they also tilt to provide maximum pilot access to the controls.

These seats are provided with adjustable airbag-type lumbar supports for crew comfort. Seat belts and inertia-type shoulder harnesses are provided for each seat. The general layout of the flight deck shows the location of the instruments and controls.

Conventional dual controls are installed so the airplane can be flown by either pilot (Figures 1-12 and 1-13). The controls and instruments are arranged for convenient single-pilot operation, or pilot and copilot crew.

The fuel control panel (Figure 1-14) is located on the left sidewall next to the pilot. Fuel quantity gages and switches and circuit breakers are located on this panel.

The instrument panel (Figure 1-15) contains the flight instruments, engine instruments, and avionics panel. Instrument markings are shown in Figure 1-16. The engine instruments (Figure 1-17) are mounted in a double vertical row next to the avionics panel. The avionics panel in the center contains the nav/comm controls and weather radar unit.



OVERHEAD LIGHT CONTROL PANEL

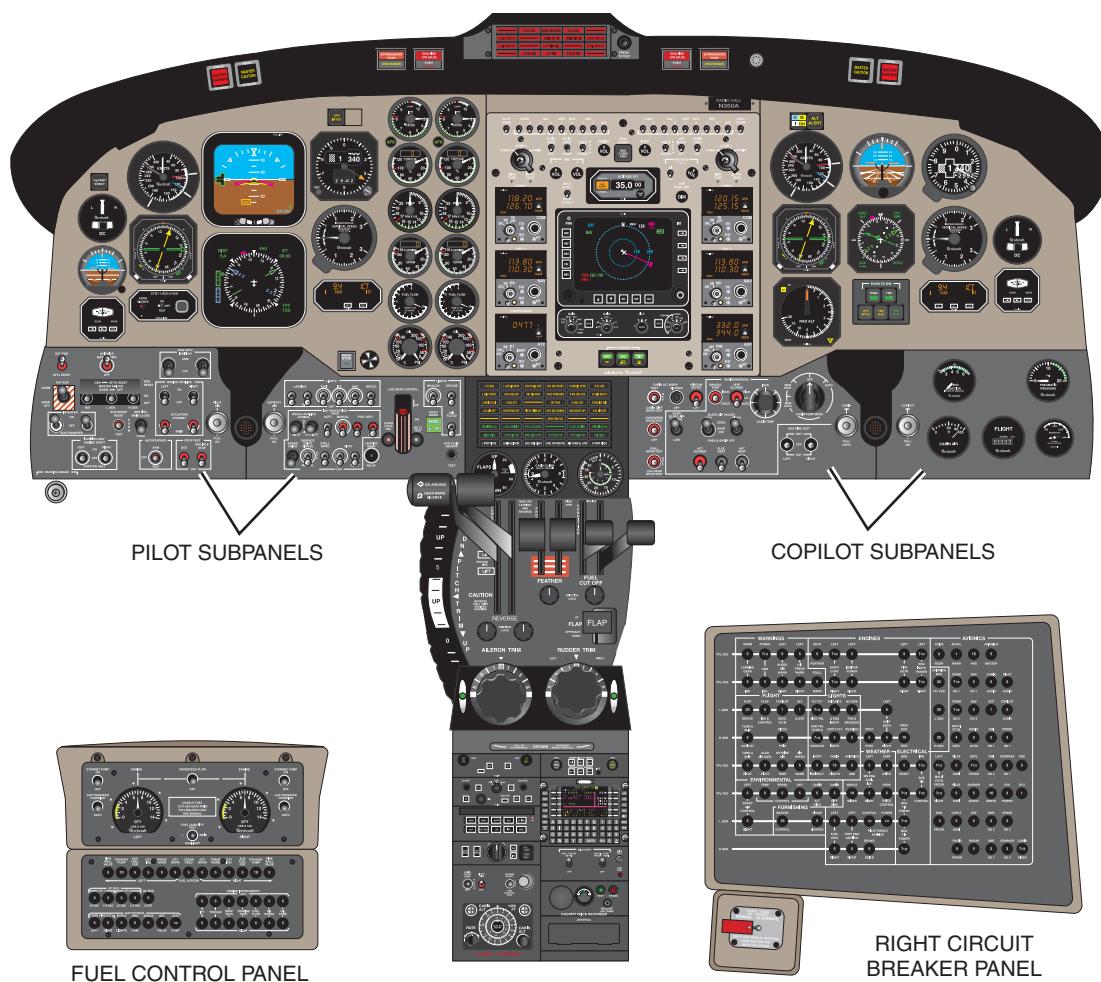


Figure 1-12 Flight Deck Layout

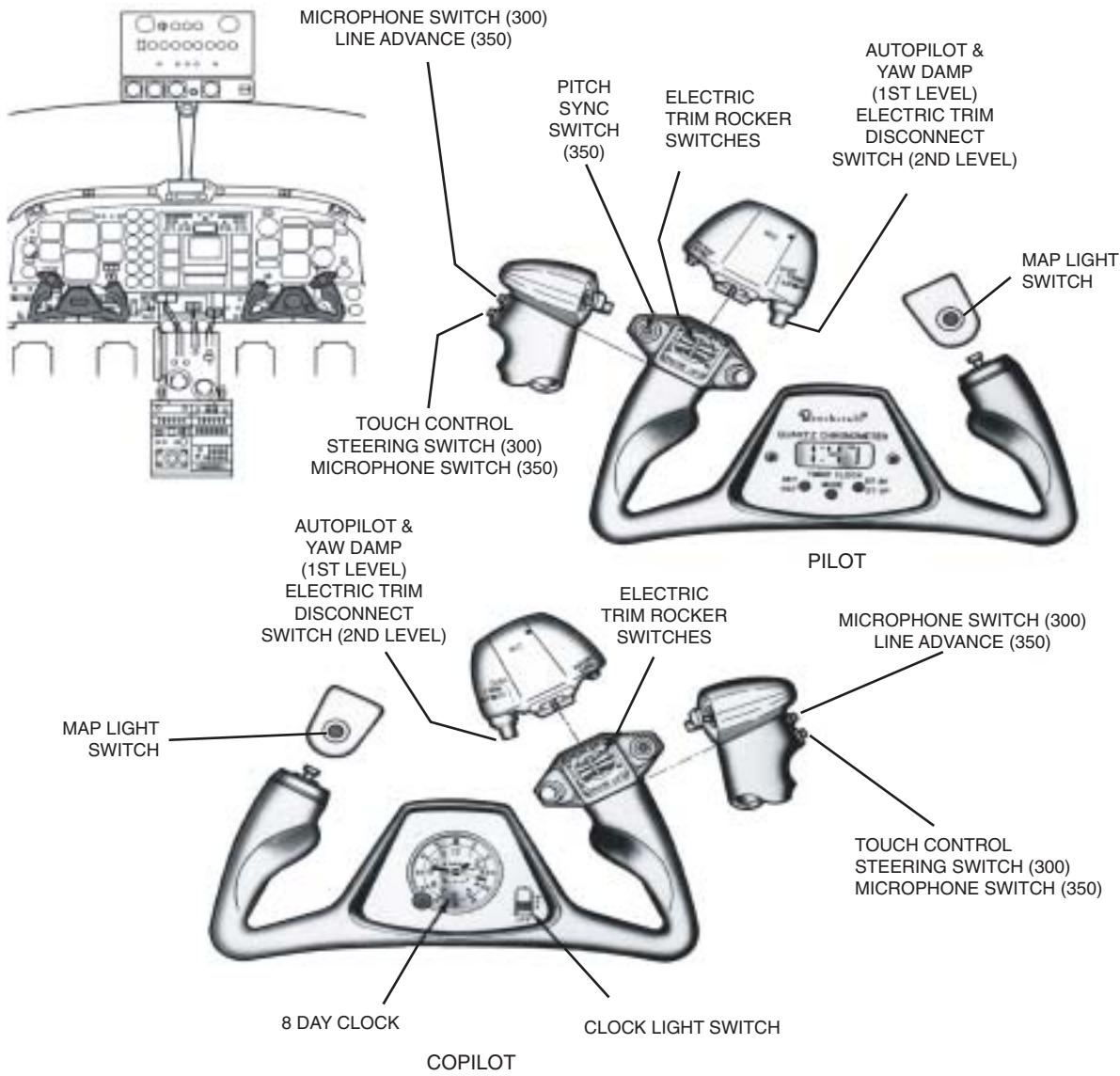
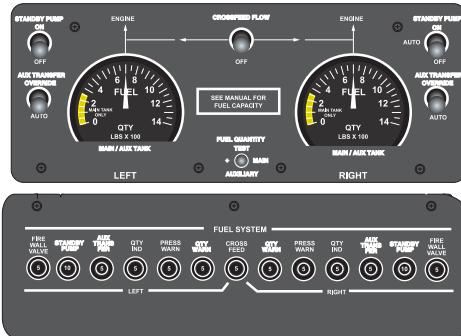
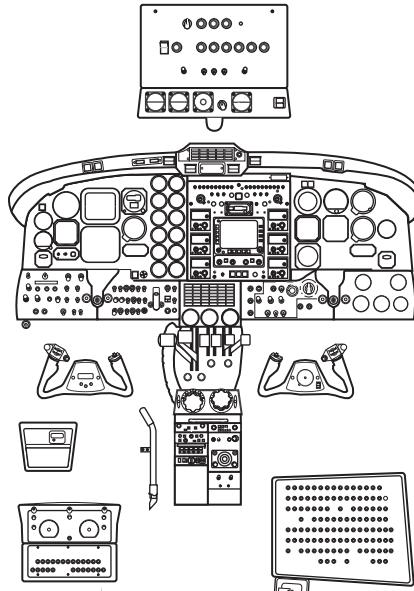


Figure 1-13 Control Wheels



KING AIR 300 FUEL CONTROL PANEL



KING AIR 350 FUEL CONTROL PANEL
PRIOR TO FL120



KING AIR 350 FUEL CONTROL PANEL
FL120 AND AFTER

Figure 1-14 Fuel Panels

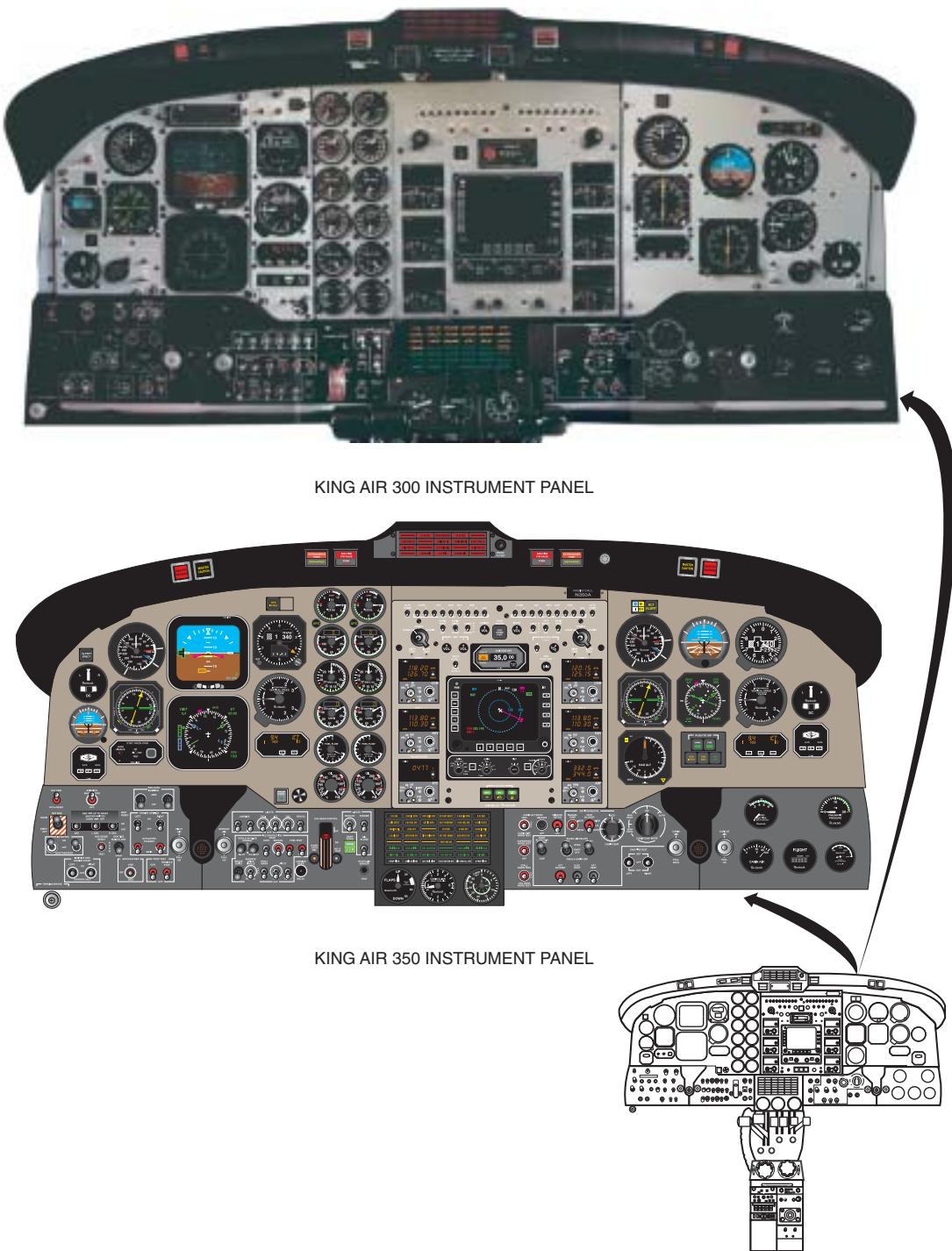


Figure 1-15 King Air 300 and 350 Instrument Panels



NOTE
ALL ILLUSTRATION NEEDLES MAY NOT
REFLECT NORMAL INDICATIONS

300 AIRSPEED INDICATOR

259 KIAS MAXIMUM (V_{MO}) OR .58 MACH

NOTE
MAXIMUM ALLOWABLE AIRSPEED (RED STRIPED)
POINTER IS SELF ADJUSTING WITH ALTITUDE

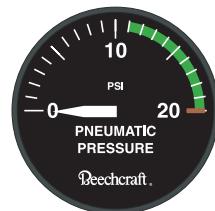
92 KIAS MINIMUM SINGLE-ENGINE CONTROL
SPEED (V_{MCA})

122 KIAS SINGLE-ENGINE BEST RATE OF
CLIMB SPEED (V_{YSE})

81-100 KIAS FULL FLAP OPERATING RANGE

100-157 KIAS (NARROW) FLAPS UP STALL SPEED
(V_s) TO MAXIMUM LIMIT FLAPS MORE THAN 40%

200 KIAS MAXIMUM FLAPS TO/AT APPROACH SPEED



350 AIRSPEED INDICATOR

263 KIAS MAXIMUM (V_{MO}) OR .58 MACH

NOTE
MAXIMUM ALLOWABLE AIRSPEED (RED STRIPED)
POINTER IS SELF ADJUSTING WITH ALTITUDE

94 KIAS MINIMUM SINGLE-ENGINE CONTROL
SPEED (V_{MCA})

125 KIAS SINGLE-ENGINE BEST RATE OF
CLIMB SPEED (V_{YSE})

81-96 KIAS FULL FLAP OPERATING RANGE

96-158 KIAS (NARROW) FLAPS UP STALL SPEED
(V_s) TO MAXIMUM LIMIT FLAPS MORE THAN 40%

202 KIAS MAXIMUM FLAPS TO/AT APPROACH SPEED



FUEL QUANTITY

0-265 LBS NO TAKEOFF RANGE



GYRO SUCTION

HG 2.8-4.3 NORMAL OPERATING RANGE
35,000-15,000 FT

HG 4.3-5.9 NORMAL OPERATING RANGE
15,000 - SEA LEVEL

CABIN ALTIMETER AND PRESSURE DIFFERENTIAL

0-6.6 PSI (APPROVED OPERATING RANGE)

6.6 PSI TO END OF SCALE (UNAPPROVED
OPERATING RANGE)



FLAP POSITION INDICATOR

40% APPROACH



PROPELLER DEICER AMMETER

26-32 AMPERES NORMAL OPERATION

Figure 1-16 Instrument Markings



INTERSTAGE TURBINE TEMPERATURE GAGE

- [Green] 400-820°C NORMAL OPERATING RANGE
- [Red] 820°C MAXIMUM CONTINUOUS LIMIT
- [White] 1000°C MAXIMUM STARTING ONLY LIMIT



TORQUE METER

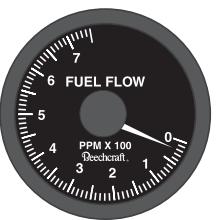
- [Green] 0 TO 100% NORMAL OPERATING RANGE
- [Red] 100% MAXIMUM LIMIT

PROPELLER TACHOMETER (N_p SPEED)

- [Green] 1450-1700 RPM NORMAL OPERATING RANGE
- [Red] 1700 RPM MAXIMUM LIMIT

GAS GENERATOR TACHOMETER (N_1 SPEED)

- [Green] 62-104% NORMAL OPERATING RANGE
- [Red] 104% MAXIMUM LIMIT



NO LIMITATIONS MARKINGS



OIL TEMPERATURE SCALE

- [Green] 0-99°C NORMAL OPERATING RANGE
- [Yellow] 99-110°C CAUTION RANGE
- [Red] 110° MAXIMUM OIL TEMPERATURE LIMIT

OIL PRESSURE SCALE

- [Red] 60 PSI MINIMUM LIMIT
- [Yellow] 60-90 PSI CAUTION RANGE
- [Green] 90-135 PSI NORMAL OPERATING RANGE
- [Red] 200 PSI MAXIMUM LIMIT

DIGITAL GAGE
(KING AIR 350 FL-120
AND SUBSEQUENT)DIGITAL GAGE
(KING AIR 350 FL-120
AND SUBSEQUENT)

Figure 1-17 Engine Instruments



Extending back from the center subpanel is the engine control quadrant and pedestal (Figure 1-18). Engine controls, flap control handle, rudder and aileron trim knobs, and pressurization controls are mounted on this pedestal. The flight director and autopilot systems are usually installed here also.

On the right side panel next to the copilot is the main circuit breaker panel (Figure 1-19), where the majority of the system circuit breakers are

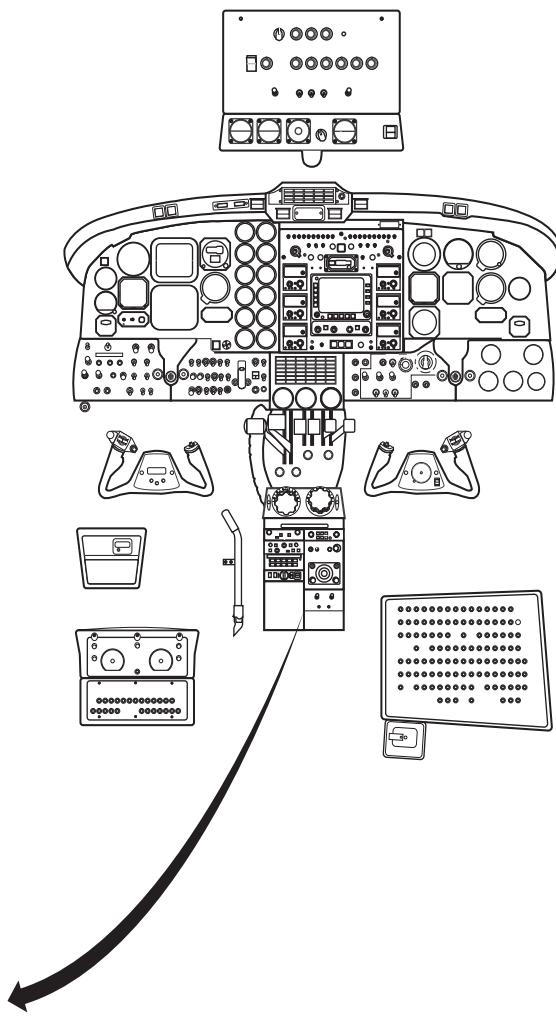
located. The static air selector switch is mounted just below the circuit breaker panel.

Just below the instrument panel are the pilot's subpanel (Figure 1-20), on the left, and the copilot's subpanel, on the right. Aircraft systems controls, engine switches, master switch, and landing gear controls are located on these subpanels.

In the overhead area, between the pilot and copilot, is the lighting control panel (Figure 1-21).



Figure 1-18 Center Pedestal

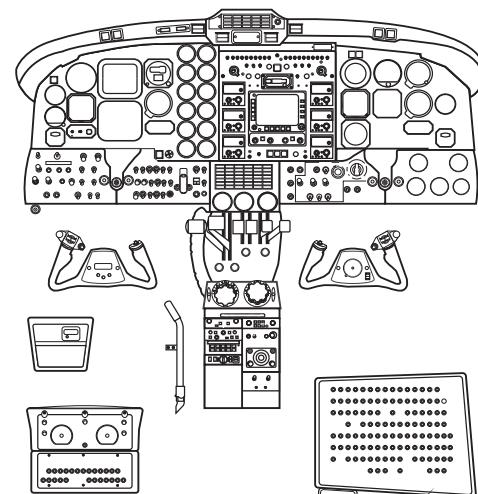
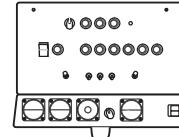




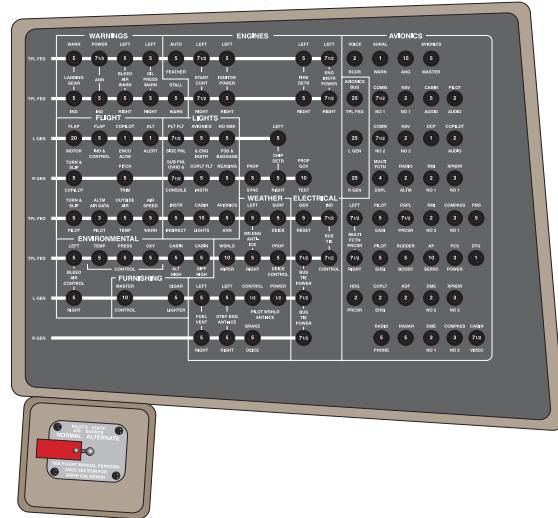
KING AIR 300/350 PILOT TRAINING MANUAL



KING AIR 350 RIGHT CIRCUIT BREAKER PANEL
PRIOR TO FL 120

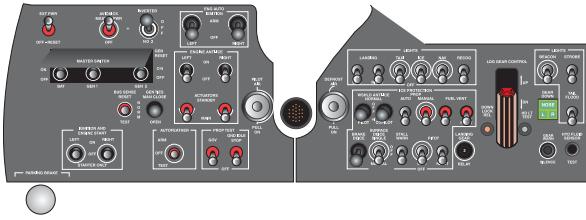


KING AIR 350 RIGHT CIRCUIT BREAKER PANEL
FL 121 THRU FL 288

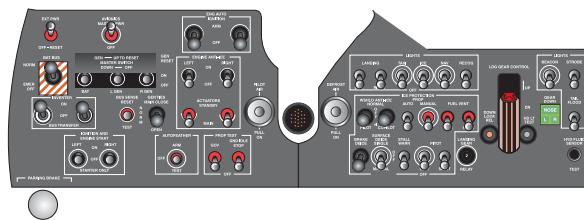


KING AIR 350 RIGHT CIRCUIT BREAKER PANEL
FL 289 AND AFTER

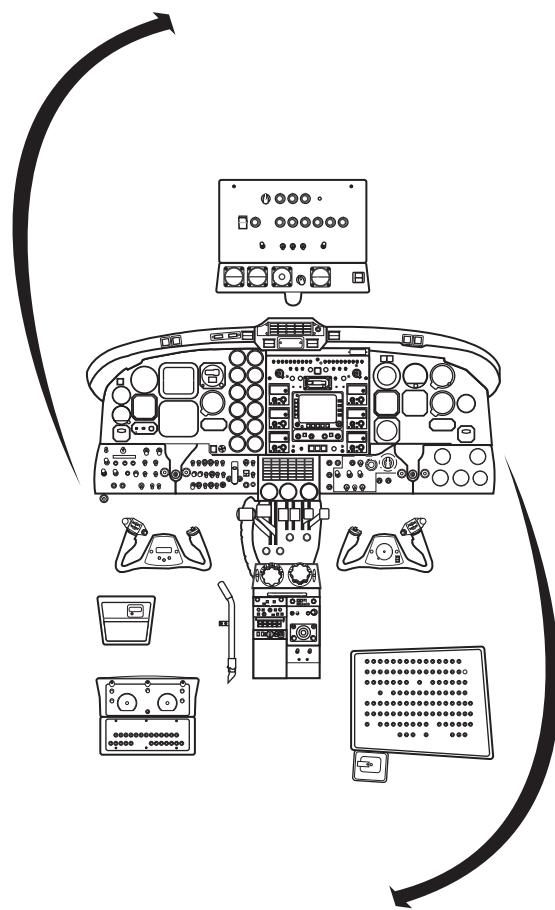
Figure 1-19 Right Side Panel



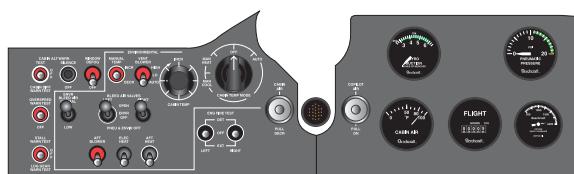
KING AIR 300 PILOT'S SUBPANEL



KING AIR 350 PILOT'S SUBPANEL



KING AIR 300 COPILOT'S SUBPANEL



KING AIR 350 COPILOT'S SUBPANEL

Figure 1-20 Subpanels

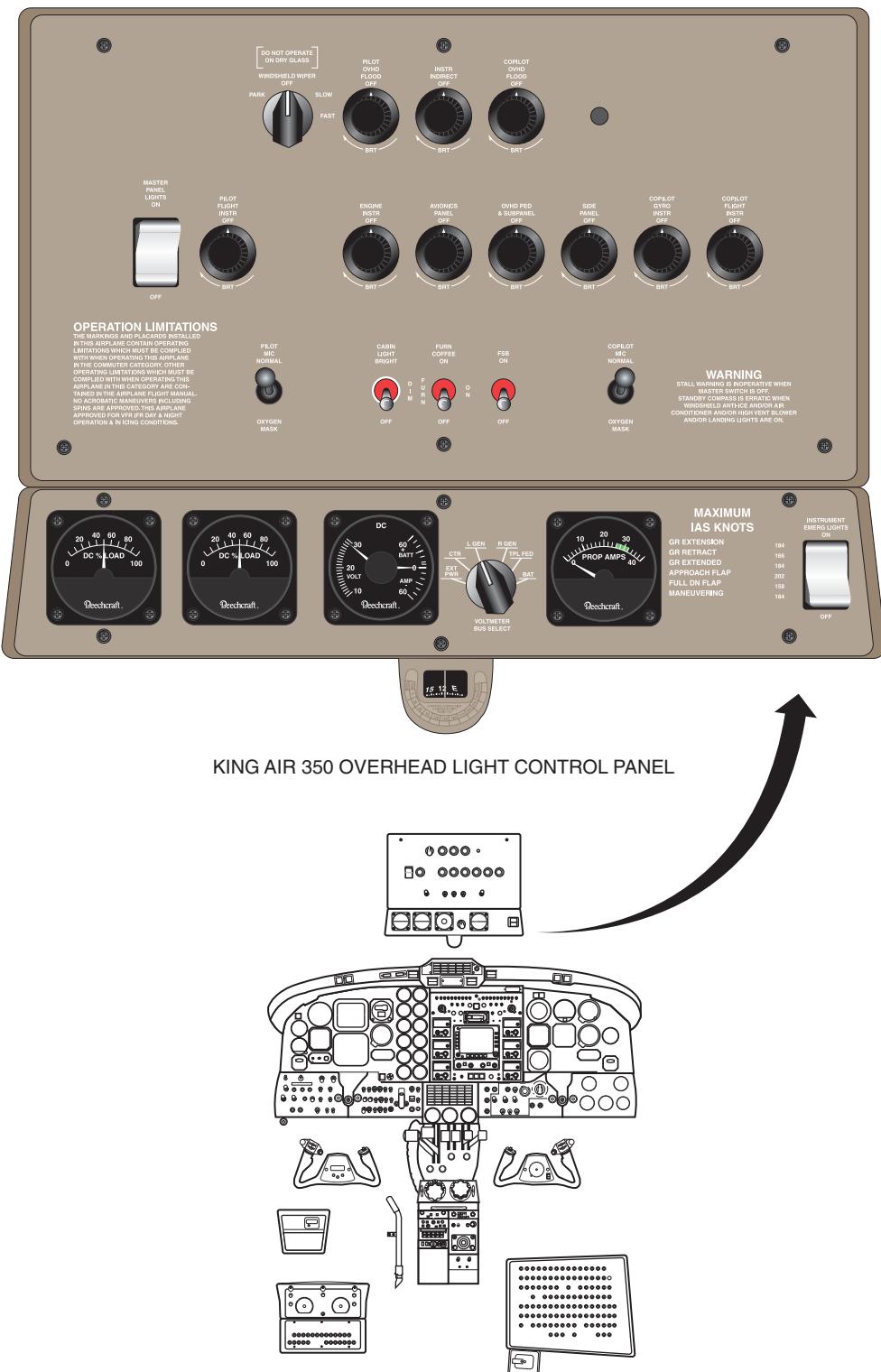


Figure 1-21 Overhead Light Control Panel



The various rheostat controls for the flight deck and instrument lighting are mounted on this panel, convenient to both pilot and copilot.

Also mounted on this panel are the windshield wiper control, generator load and voltage gages, battery ammeter, propeller deice ammeter, and emergency light switch. The pilot and copilot oxygen mask microphone switches, no smoking and fasten seat belt sign switches, and cabin light switches are installed just above the gages on the overhead panel. Operation limitations and air-speed limitations are also placarded on this panel.

CONTROL SURFACES

The Super King Air 300 and 350 are equipped with conventional ailerons and rudder. A T-tail horizontal stabilizer and elevator are mounted at the extreme top of the vertical stabilizer (Figure 1-22). The control surfaces are cable-operated by conventional dual controls in the flight deck.

Any time the airplane is parked overnight or in windy conditions, the rudder gust pin and control locks should be installed to prevent damage to the control surfaces and hinges or the controls (Figure 1-23).



Figure 1-22 T-Tail Configuration



Figure 1-23 Flight Control Locks



The King Air 300 has single push rod actuators installed on all pilot controlled trim tabs while the 350 has dual push rod actuators.

TIEDOWN AND SECURING

When the airplane is parked overnight or during high winds, it should be securely moored with protective covers put in place (Figure 1-24). Place wheel chocks fore and aft of the main gear wheels and nose wheel. Using the airplane mooring points, tie the airplane down with suitable chain or rope. Install the control surface lock and be sure the flaps are up.

Secure the propellers to prevent windmilling. This airplane has free-spinning propellers that

could be hazardous if not restrained. Allowing engine gears and bearings to windmill without lubrication is not a good practice. In addition, when there is blowing dust or rain, install the engine inlet covers.

Two items require particular attention: the parking brake handle mounted just under the left corner of the subpanel, and the rudder gust lockpin mounted in the pilot's floorboard. Before towing the airplane, the parking brake must be released (brake handle pushed in) and the rudder gust lockpin must be removed from the pinhole in the floor. Serious damage to the tires, brakes, and steering linkage can result if these items are not released.



PROPELLER TIEDOWNS

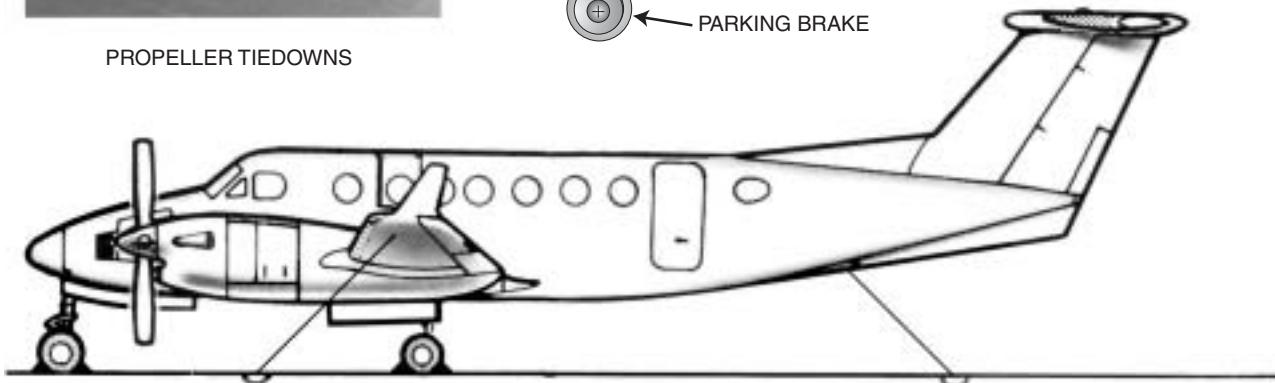
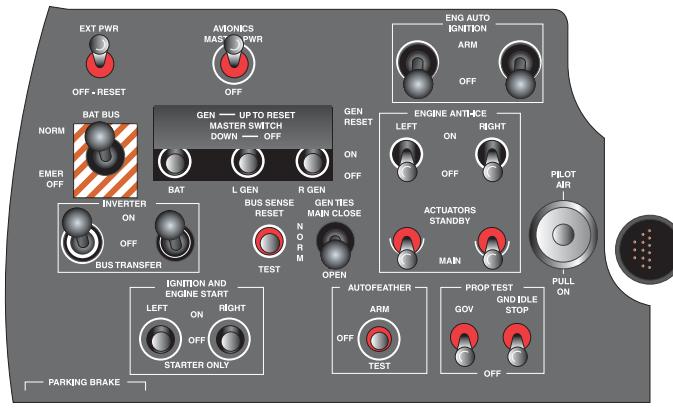


Figure 1-24 Tiedowns



TAXIING

The ground turning radii are predicated on the use of partial braking action and differential power (Figure 1-25). Locking the inside brake can cause tire or strut damage. When turning the airplane, if the wingtip clears obstacles, the tail will also. When taxiing, turning, and starting the engines, there is an area directly to the rear of the engines where the propeller windstream can be hazardous to persons or parked airplanes. While the velocities and temperatures cannot be accurately measured, reasonable care should be taken to prevent incidents within these danger areas.

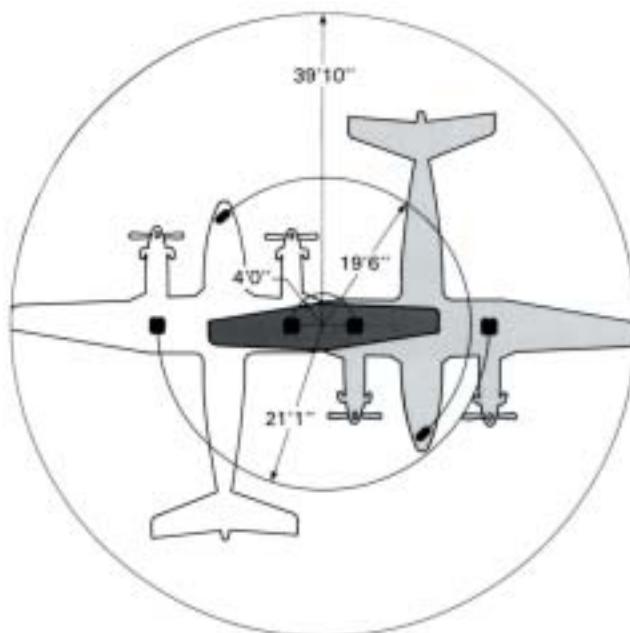
SERVICING DATA

The Handling, Servicing, and Maintenance section of the POH outlines to the owner and operator the requirements for maintaining the Super King Air 300 or 350 in a condition equal to its original manufacture. This information sets time intervals at which the airplane should be taken to a Beechcraft Aviation Center for periodic servicing or preventive maintenance. All limits, procedures, safety practices, time limits, servicing, and maintenance requirements contained in the *POH* are mandatory.

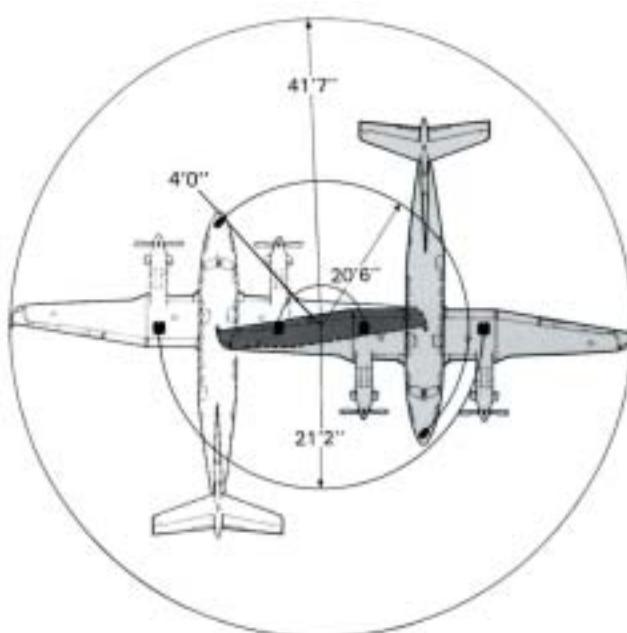
This section of the *POH* includes a Consumable Materials chart which lists approved and recommended materials for servicing the airplane. The Servicing Data diagram (Figure 1-26) lists and illustrates servicing points and materials required. This chart is for reference only and is always superseded by the *POH* information.

PREFLIGHT INSPECTION

The preflight inspection procedure has been divided into five areas as shown in Figure 1-27. The inspection begins in the flight compartment, proceeds aft, then moves clockwise around the aircraft.



**KING AIR 300
GROUND TURNING CLEARANCE**



**KING AIR 350
GROUND TURNING CLEARANCE**

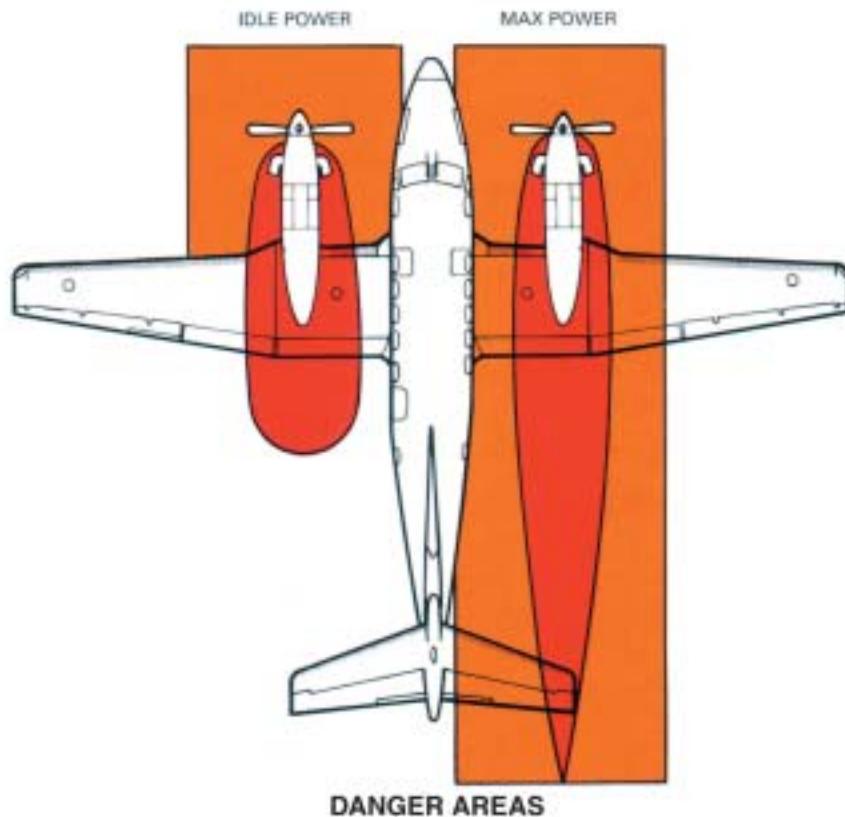
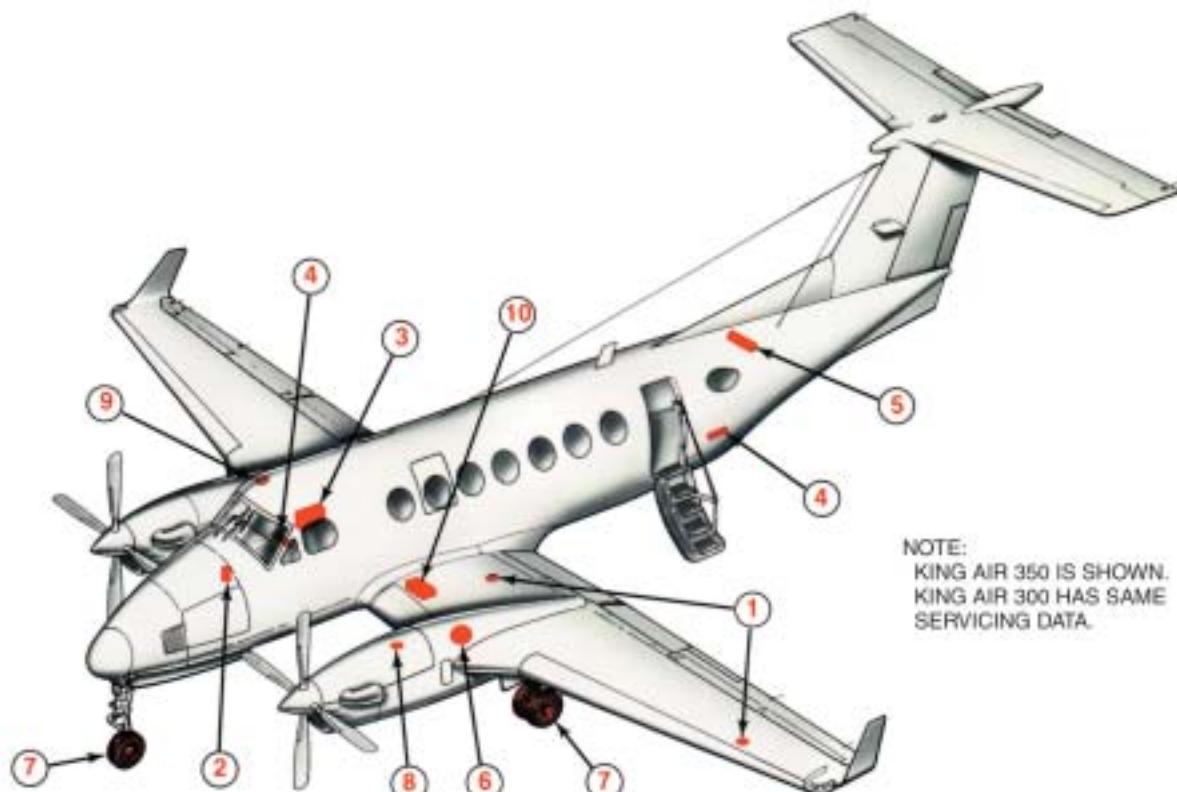


Figure 1-25 Turn Radius and Danger Areas



- ① Fuel Tank Filler Caps (Typical Left and Right)

Approved Fuel Grades and Additives
Recommended Engine Fuels

Commercial Grades:

Jet A
Jet A-1
Jet B

Military Grades:
JP-4
JP-5
JP-8

Emergency Engine Fuels

Commercial Aviation Gasoline Grades:

80 Red (Formerly 80/87)

91/96

100LL Blue*

100 Green (Formerly 100/130)

115/145 (Purple)

Military Aviation Gasoline Grades:

80/87 Red

100/130 Green

115/145 Purple

- ② Hydraulic Fluid Reservoir (Brake)
Specification MIL-H-5606

- ③ Battery
24-Volt 34- or 36-Ampere Hour Nickel-Cadmium

- ④ Fire Extinguishers (Hand-Type)
Halon 1211
Bromochlorodifluoromethane

- ⑤ Oxygen Supply
Oxygen Specification: MIL-O-27210
Aviators Breathing Oxygen: 50, 77, or 115 Cu. Ft.

- ⑥ Engine Fire Extinguisher
Halon 1301
Bromotrifluoromethane

- ⑦ Tire Size:
Main Wheels 19 x 6.75-8 10-ply tubeless
Nose Wheel 22 x 6.75-10 8-ply tubeless
Tire Pressure:
Main Wheel 80-87 psi (unloaded)
Main Wheel 88-92 psi (loaded)
Nose Wheel 55-60 psi

- ⑧ Engine Oil Filler

- ⑨ DC External Power Receptacle (28-Volt)

- ⑩ Hydraulic Fluid Reservoir (Landing Gear)
Specification MIL-H-5606

*In some countries, this fuel is colored green and designated "100L."

Figure 1-26 Service Data



Figure 1-27 Preflight Inspection



KING AIR 300 TO 350 (B300) DIFFERENCES

KING AIR 300 AND 350 (B300)

There are far more similarities between the King Air 300 and 350 (B300) than there are differences. For this reason, among others, the BE-300 BE-1900 type rating allows pilots holding this type rating to operate the King Air 300, King Air 350 (B300), and 1900 Airliner.

MANUAL FORMAT

The format of this training manual discussion is presented with regard to similarities between the King Air 300 and 350. When there are differences between the airplanes (in schematics, drawings, or text), these differences are brought to light with markings, notes, or titles. Look for these distinctions as you use the training manual. When differences are not distinguished, consider the information to be the same for both airplanes.

Since the King Air 350 (B300) evolved from the King Air 300, the following changes from the King Air 300 to the King Air 350 are listed to give an overview of the differences.

General

1. The cabin has been lengthened 34 inches (14.4 forward of the wing and 19.6 aft of the wing) increasing fuselage length from 43 feet 10 inches to 46 feet 8 inches.
2. Dual navigation lights have been installed on the wing tips.
3. A NASA developed winglet has been installed on each wing tip.
4. The wing span has increased approximately 3 feet 5 inches from 54 feet 6 inches to 57 feet 11 inches.

5. The wing area has increased approximately 7 square feet from 303 square feet to 310 square feet.
6. Dual push rods have been installed on all trim tabs.
7. The hinge lines on control surfaces and trim tabs are parallel in relation to each other.
8. There has been an overspeed warning horn installed that sounds when the airspeed exceeds the barber pole by no more than 6 knots or .01 mach.
9. An improved windshield has been installed to minimize distortion.
10. The flaps will always travel to the position selected. The positions between approach and down cannot be selected.
11. The maximum temperature for takeoff operations with the ice vanes extended is +10°C.

Annunciator

General

1. White annunciators, called status annunciators, have been added to the annunciator system. These annunciators have no master flasher.
2. The colors on the Caution-Advisory-Status annunciator panel are arranged from top to bottom: Amber, Green, and White.
3. The Caution-Advisory-Status annunciator panel has been repositioned between the flap, cabin rate of climb/descent, cabin pressure altitude/differential gauges and the power quadrant. It also has been placed at approximately a 30° angle to the instrument panel so that all annunciators can be easily seen.
4. The autofeather annunciators have been moved to the glareshield above the engine gauges.



Red

1. A high cabin pressure differential annunciator has been added that illuminates when the cabin pressure differential exceeds 6.9 psid.
2. The red cabin altitude warning annunciator illuminates at 12,000 feet cabin pressure altitude instead of 12,500. A steady state warning tone will also sound but can be canceled by the warning tone cancel button.
3. A No. 1 A.C. bus annunciator and a No. 2 A.C. bus annunciator replaces the inverter annunciator. An A.C. bus annunciator will illuminate when the voltage on the respective buses (26 volts or 115 volts) is out of tolerance ($>\pm 10\%$) or the frequency is out of tolerance ($>\pm 5\%$). These annunciators will not flash when the inverters are turned on, they will simply extinguish, respectively, if all parameters are within tolerance. In addition, they will not latch on if the voltage or frequency is momentarily out of tolerance.

Amber

1. An oxygen not armed annunciator has been added that illuminates when the oxygen system is not armed.
2. An autofeather off annunciator has been added that illuminates when the autofeather switch is not in the arm position and the landing gear is down and locked.
3. A rudder boost off annunciator has been added that illuminates when the rudder boost is not armed.
4. The bleed-air off annunciators have been changed from green to amber.

Green

1. Wing deice and tail deice annunciators have been added that illuminate when their respective deice boots inflate.

White

1. A cabin altitude annunciator has been added that illuminates at 10,000 feet cabin pressure altitude. A steady state warning tone will also sound but can be canceled by the warning tone cancel button.
2. The prop pitch annunciators have been changed from amber to white.
3. The landing and taxi lights annunciator has been changed from green to white.
4. The passenger oxygen annunciator has been changed from green to white.
5. The N₁ low annunciator has been changed from green to white.

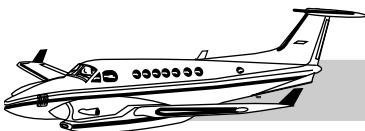
Propeller

1. Electrical power is removed from each ground low pitch stop solenoid by a microswitch on its respective power levers as they are advanced through approximately 68 to 70% N₁. This will help prevent surging as power is added.
2. The ground low pitch stop test switch energizes the ground low pitch stop solenoids instead of deenergizing them.
3. Some blade angles have been changed. They are:

Feathered	+79.3
Flight Low Pitch Stop.....	+12
Ground Low Pitch Stop	+2

Electrical

1. A battery bus switch has been added to the pilot's left subpanel. It is surrounded by red and white hash marks and has a normal and emergency off position. It allows the pilot to turn the hot battery bus off.



2. All fuel indications have been moved to the triple-fed bus. As a result, the generator ties will not be manually closed prior to a battery start.
3. The flashing beacons have been moved to the triple-fed bus and the navigation lights have been moved to the left generator bus.
4. There is a control switch for each inverter. These switches have three positions that are from top to bottom: ON, OFF, and BUS TRANSFER. Both inverters are to be used for flight and they each feed the respective buses. Each A.C. bus has its own A.C. bus annunciator and A.C. bus monitor that monitors the frequency and voltage (400 hertz, 115 volts, and 26 volts). If an A.C. bus annunciator illuminates, determine the cause and if appropriate, place the failed inverter's switch to the BUS TRANSFER position. This will connect all like voltage A.C. buses together and allow one inverter to power all four A.C. buses. The No. 1 inverter is powered only from the center bus and the No. 2 inverter is powered only from the right generator bus. The inverters are powered whenever the respective buses are powered. However, the inverter switch interrupts the control circuit to the oscillator inside the inverter when the respective inverter switch is placed in the OFF or BUS TRANSFER position. Thus, the inverter is not producing any A.C. power until the inverter switch is placed in the ON position. As a result of this change, there are no inverter power relays, inverter power select relays, or an inverter select relay. When both inverters are operating, they are phase locked.

Rudder Boost

1. The rudder boost system senses torque instead of P_3 by way of its own torque transducer on each torque manifold. Rudder boost activation starts at approximately 30% torque differential and is at maximum at approximately 80% differential. Because of this modification, the brake deice does not have to be turned off for takeoff.


Table 1-1 FAR PERFORMANCE CERTIFICATION REQUIREMENTS COMPARISONS

	FAR 23/CAR 3 NORMAL CATEGORY	SFAR 41C DOMESTIC OPERATION	CLIMB SEGMENTS	SFAR 41C ICAO (ANNEX 8)	FAR 25 AND FAR 23 COMMUTER CATEGORY
ONE ENGINE INOPERATIVE TAKEOFF CLIMB	At 5000 P.A., +5°C O.A.T.: Rate of climb $\geq 0.027 \times V_{SO}^2$ (in kts) but not less than 1.2% climb gradient and for a turbine powered aircraft: At 5000 P.A., $+27^\circ C$ O.A.T.: Rate of climb $\geq 0.014 \times V_{SO}^2$ (in kts) but not less than a .06% climb gradient at V_{YSE} .	Positive climb at V_1 when out of ground effect with gear down and flaps T.O. position AND a climb gradient $\geq 2\%$ at V_2 up to 400 ft. above the T.O. surface with gear up and flaps T.O. position AND a climb gradient $\geq 1.2\%$ at 1000 ft. above the T.O. surface at V_{YSE} .	First Segment: Gear down, flaps T.O. position, from 35 ft. to gear up.	Positive climb V_1 when out of ground effect.	Positive climb at V_{LOF} when out of ground effect.
			Second Segment: Gear up, flaps T.O. position, from gear up to 400 ft.	Climb gradient $\geq 2\%$ at V_2 .	Climb gradient $\geq 2.4\%$ FAR 25 or 2.0% FAR 23 C.C. at V_2 .
			Third Segment: Enroute configura- tion 400-1500 ft.	Climb gradient $\geq 1.2\%$ at 1500 ft. above the T.O. surface at V_{YSE} .	Climb gradient $\geq 1.2\%$ at 1500 ft. above the T.O. surface at V_{YSE} .
RUNWAY REQUIRED	All engine T.O. distance over 50 ft.	Accelerate-Stop distance above 12,500 lbs. or all engine T.O. distance over 50 ft. at or below 12,500 lbs. (NOTE 1)		The longest of: All engine T.O. distance over 50 ft. x 1.15, Accelerate-Go, or Accelerate- Stop.	The longest of: All engine T.O. distance over 35 ft. x 1.15, Accelerate-Go, or Accelerate- Stop.
APPROACH CLIMB ONE ENGINE INOPERATIVE	N.A.	N.A.		Climb gradient $\geq 2.1\%$ at V_{REF} .	Climb gradient $\geq 2.1\%$ at V_{REF} .
LANDING CLIMB ALL ENGINES OPERATING	N.A.	Climb gradient $\geq 3.3\%$ at V_{REF} .		Climb gradient $\geq 3.3\%$ at V_{REF} .	Climb gradient $\geq 3.2\%$ FAR 25 or $\geq 3.3\%$ FAR 23 C.C. at V_{REF} .
LANDING DISTANCE	Actual distance over 50 ft. (NOTE 2)	Actual distance over 50 ft. (NOTE 2)		Actual distance over 50 ft. x 1.67 = Field Length (NOTE 2)	Actual distance over 50 ft. (NOTE 2)

NOTE 1: Commercial operators must comply at all weights.

NOTE 2: Multiplied by 1.67 for FAR 121 and 135 operations.

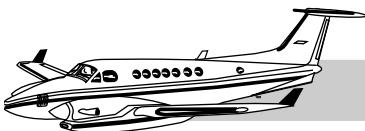


CHAPTER 2

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CHAPTER 2

ELECTRICAL POWER



INTRODUCTION

Familiarity with, and an understanding of, the airplane electrical system will ease pilot workload in normal operations, and in the event of an electrical system or component malfunction. The pilot should be able to locate and identify switches and circuit breakers quickly, and should also be familiar with appropriate corrective actions in abnormal and emergency situations.

DESCRIPTION

The Electrical System section of the training manual presents a description and discussion of the airplane electrical system and components. The electrical system is discussed to the extent necessary for the pilot to cope with normal, abnormal and emergency operations. The location and purpose of switches, indicators, lights, and circuit breakers, along with DC and AC generation and distribution are described. This section also includes some of the limitations and possible faults relating to particular systems or components. Basic electrical symbols used in this chapter are shown in Figure 2-1.

ELECTRICAL SYSTEM

The airplane electrical system is a 28-volt DC (nominal) system with the negative lead of each power source grounded to the main airplane structure (Figures 2-2 and 2-3). Direct Current (DC) electrical power is provided by one 24-volt, 34- or 36-ampere-hour, nickel-cadmium battery, or 42 amp-hour sealed lead acid battery and two 28-volt, 300-ampere starter/generators connected in parallel (Figures 2-27 and 2-42). This system is capable of supplying power to all subsystems necessary for normal operation of the airplane. The battery and generator switches are used to control power from the battery and generators

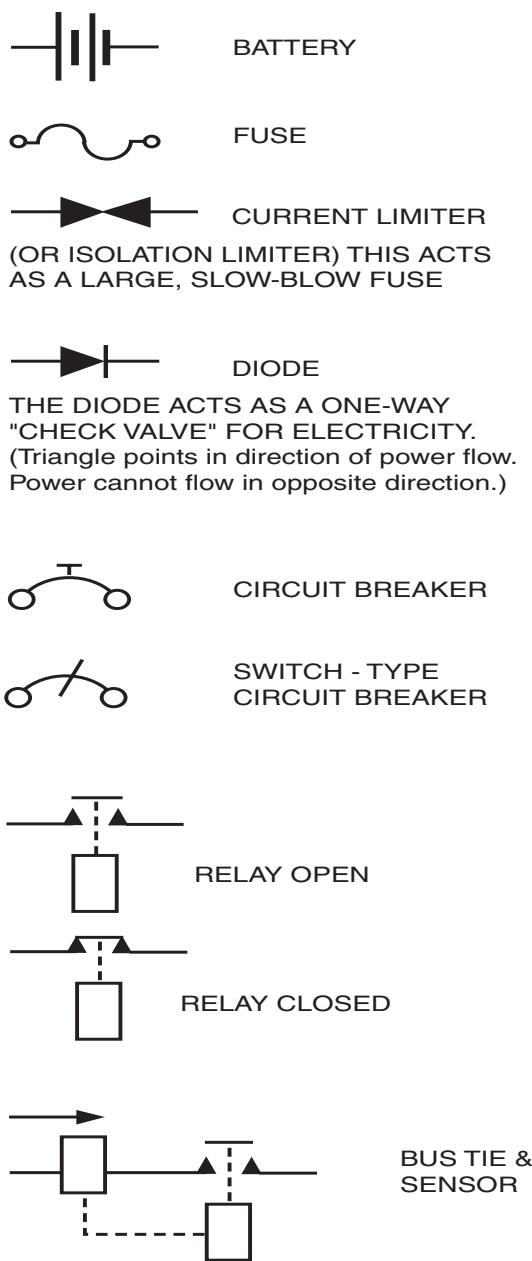


Figure 2-1 Basic Electrical Symbols

into the airplane electrical system. They are located on the pilot's left subpanel (Figure 2-4), and are labeled BAT, GEN 1 and GEN 2 on the King Air 300 or BAT, L GEN, and R GEN on the King Air 350.

The King Air 300 and 350 utilize a multi-bus system. The buses which supply power to most of the electrically controlled systems are the left and right generator buses and the triple-fed bus. There is also a center bus that is used primarily to tie the electrical system together. There are some components which get power from the center bus. On the 300, there is a hot battery bus. The hot battery bus is connected directly to, and is located next to, the battery. Electrical power flow between the battery and the hot battery bus cannot be controlled from the cockpit. On the 350, there is a battery bus and a dual fed bus which are connected to the battery through a remote control circuit breaker. The remote control circuit breaker is located next to the battery, but the battery bus and the dual fed bus are located on the left side circuit breaker panel in the cockpit. This remote control circuit breaker can be controlled from the cockpit by way of the battery bus switch. All switches in the cockpit which receive power from the center or triple-fed buses are identified by a white ring on the panel around the switch.

Electrical loads are divided among the buses as noted on the Electrical System Buses and Subbus Feeders Chart (Figure 2-5). Equipment on the buses is arranged so that all items with duplicate functions (such as right and left landing lights) are connected to different buses. Each bus is also divided into feeder buses or subbuses. Many of these subbuses can be identified in the cockpit on the copilot's circuit breaker panel. Circuit breakers on the same feeder are connected by white lines on the panel faceboard (Figure 2-6).

On the King Air 300, the battery is always connected to the hot battery bus (Figure 2-28). Both are located in the right wing center section. Operation of equipment on the hot battery bus does not depend on the position of the battery switch.

On the King Air 350, the battery is connected to the battery bus and the dual fed bus through a remote control circuit breaker. The battery bus and dual fed bus are located on the fuel circuit breaker panel and is labeled battery bus. The battery and the remote control circuit breaker is located in the right wing center section. The control switch for the remote control circuit breaker



LEGEND ABBREVIATIONS USED

L = LEFT

EPR = EXTERNAL POWER RELAY

R = RIGHT

STR/GEN = STARTER GENERATOR

B = BATTERY

GEN CONT = GENERATOR CONTROL

BT = BUS TIE

EXT PWR = EXTERNAL POWER

LC = LINE CONTACTOR

CTR BUS = CENTER BUS

SB = SUB BUS

RG = RIGHT GENERATOR

SR = STARTER RELAY

LG = LEFT GENERATOR

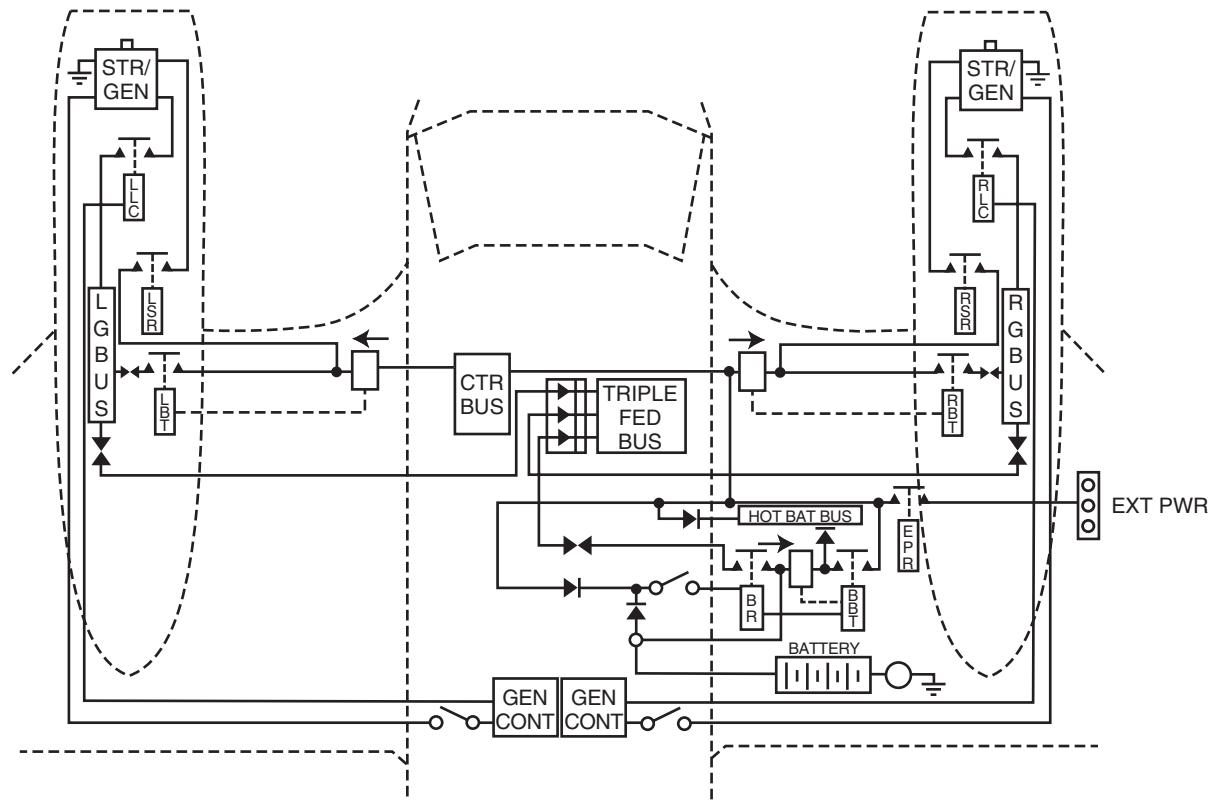


Figure 2-2 King Air 300 Electrical System Component Locations



LEGEND ABBREVIATIONS USED

L = LEFT	EPR = EXTERNAL POWER RELAY
R = RIGHT	STR/GEN = STARTER GENERATOR
B = BATTERY	GEN CONT = GENERATOR CONTROL
BT = BUS TIE	EXT PWR = EXTERNAL POWER
LC = LINE CONTACTOR	CTR BUS = CENTER BUS
SB = SUB BUS	RG = RIGHT GENERATOR
SR = STARTER RELAY	LG = LEFT GENERATOR
BB = BATTERY BUS	RCCB = REMOTE CONTROL CIRCUIT BREAKER
DFB = DUAL FED BUS	

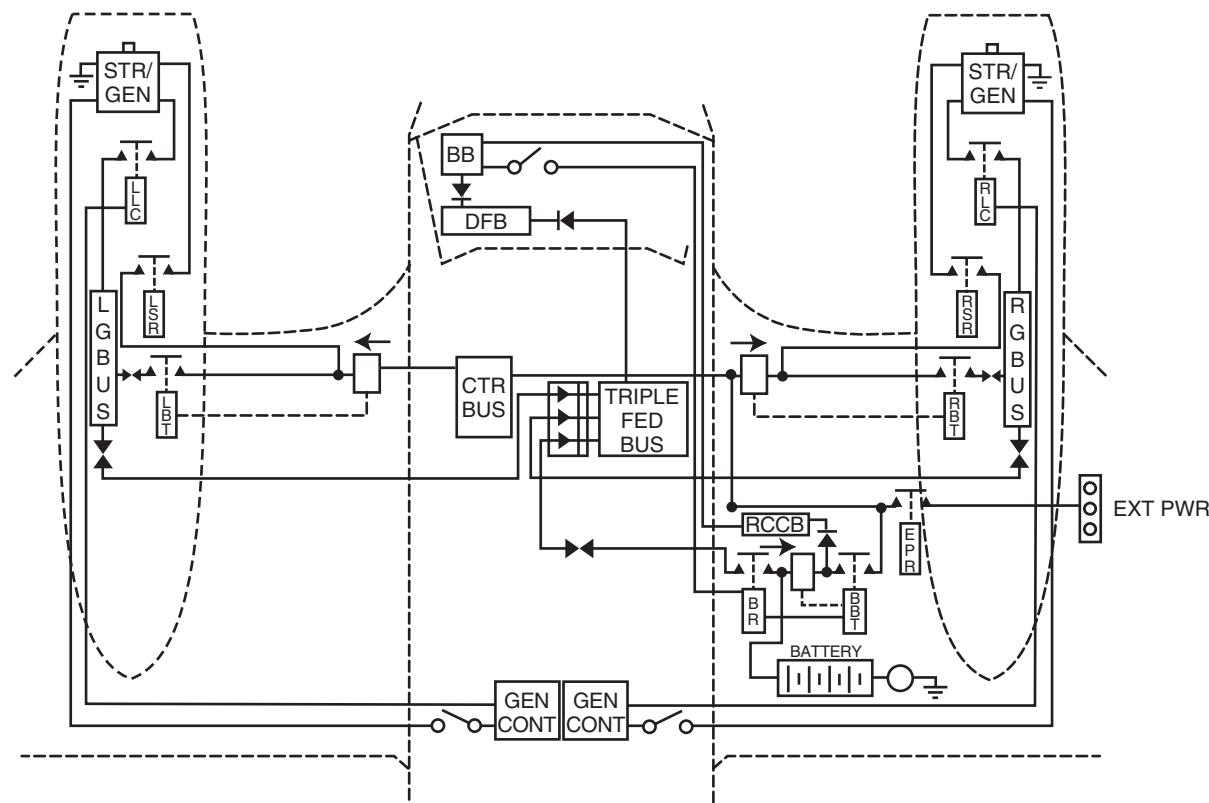


Figure 2-3 King Air 350 Electrical System Component Locations

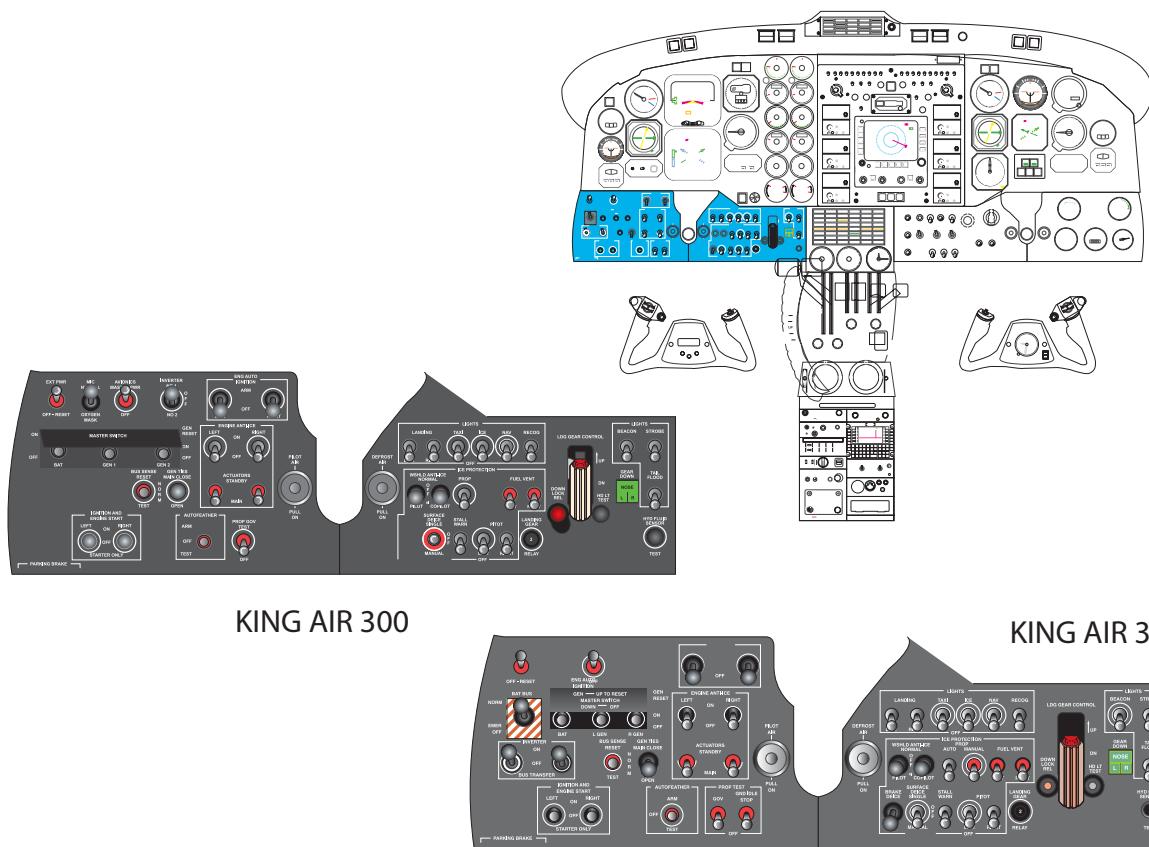


Figure 2-4 Pilot's Left Subpanel

is on the left pilot's subpanel and is labeled BATTERY BUS. This switch has two positions: NORMAL and EMERGENCY OFF. The EMERGENCY OFF position disconnects the battery from the hot battery bus and is used only when the hot battery bus needs to be turned off.

The battery switch may close the battery bus tie, which will connect the battery to the center bus, as well as the battery relay, which will connect the battery to the triple-fed bus. The battery bus tie and the battery relay may be opened by the battery switch as well.

The generators are controlled by individual generator control units which allow constant voltage to be presented to the buses during variations in engine speed and electrical load requirements. The load on each generator is indicated by left and right loadmeters located on the overhead

meter panel (Figure 2-7). A normal system voltage of 28.25 (± 0.25) volts maintains the battery at full charge.

In normal operation, all buses are automatically tied into a singleloop system where all sources supply power through individual protective devices. The triple-fed bus is powered from the battery and both generator buses. The left and right generators supply power to their respective left and right generator buses. The center bus is fed by the two generator buses and the battery, which are automatically connected to the center bus whenever the bus ties are closed. The Power Distribution Schematics (Figures 2-27 thru 2-56) show how the buses are interconnected.

Voltage on each bus may be monitored on the voltmeter (located in the overhead panel) by selecting the desired bus using the VOLTME-

**LEFT GENERATOR BUS**

- + Subbus Feeder
 - + Auto Prop Deice
 - Auto Prop Deice CB Switch
 - + Fuel Panel Feeder
 - L Aux Fuel Warn & Xfer
 - * L Firewall Valve
 - L Fuel Press Warn
 - L Fuel Quantity
 - L Fuel Quantity Warn
 - * L Stby Pump
 - + L Refreshment Bar
 - ** + No. 1 Inverter Power
 - ** + No. 1 Inverter Power Control
 - + No. 1 Inverter Bus Select
 - No. 1 Inverter Power Select
 - + No. 2 Avionics Bus Feeder
 - No. 2 Avionics
 - + Vent Blower
- + Subpanel Feeder
 - L Landing Light
 - Beacon Lights
 - Tail Flood Lights (Opt)
- + Circuit Breaker Panel Feeder
 - Altitude Alert (Opt)
 - AM-FM Stereo (Opt)
 - Avionics & Eng Inst Lts
 - Bus Tie Power, GEN 1
 - Bus Tie Control
 - Bus Tie Indication
 - Cigarette Lighter
 - CP Encoding Altimeter (Opt)
 - Flap Ind & Control
 - Flap Motor
 - Flight Inst Lights
 - Furnishings Master Control
 - L Chip Detect
 - L Fuel Vent Heat
 - L Stby Engine Anti-Ice
 - No Smk, FSB & Reading Lts
 - Prop Gov Test (FA-2 thru 110)
 - R Bleed Air Control
- + Triple-Fed Bus Feeder
 - Pilot's W/S Anti-Ice Power RCCB
 - Pilot's W/S Anti-Ice Control RCCB

- * Primary Power Source, Alternate on Triple-Fed Bus
- ** Primary Power Source, Alternate on Center Bus
- + Circuit Protection Device Not Directly Accessible In Flight

CENTER BUS

- + Battery Switch/Relay (Alt Pwr)
- + Condenser Blower
- + Electric Heat (Aft)
- + Electric Heat (Fwd)
- + Landing Gear Motor
- + L Prop Deice Power (Manual)
- + R Prop Deice Power (Manual)
- + No. 1 Inverter Power (Alt)
- + No. 2 Inverter Power (Alt)
- + Subpanel Feeder:
 - Ice Lights
 - Nav Lights
 - Taxi Light

RIGHT GENERATOR BUS

- + Subbus Feeder
 - + Aft Evaporator Blower
 - + Air Conditioner Clutch
 - + Copilot W/S Anti-Ice Power
 - + Copilot W/S Anti-Ice Control
 - + Fuel Panel Feeder
 - Crossfeed
 - R Aux Fuel Warn & Xfr
 - * R Firewall Valve
 - R Fuel Press Warn
 - R Fuel Quantity
 - R Fuel Quantity Warn
 - * R Stby Pump
- ** + No. 2 Inverter Power
 - ** + No. 2 Inverter Power Control
 - + No. 2 Inverter Bus Select
 - No. 2 Inverter Power Select
- + No. 3 Avionics Bus Feeder
 - No. 3 Avionics
- + R Refreshment Bar
- + Toilet
- + Subpanel Feeder
 - Recognition Lights (Opt)
 - R Landing Light
 - R Pitot Heat
 - Stall Warning Heat
 - Strobe Lights (Opt)
- + Circuit Breaker Panel Feeder
 - Brake Deice (Opt)
 - Bus Tie Power, GEN 2
 - Bus Tie Control
 - Bus Tie Indication
 - Pitch Trim (Opt)
 - Prop Gov Test (FA-111 & after)
 - Prop Sync (Opt)
 - R Chip Detect
 - R Fuel Vent Heat
 - R Stby Engine Anti-Ice
 - Side Panel & Ovhd Flood Lights
 - Subpanel, Ovhd & Console
 - Lights
- + Triple-Fed Bus Feeder

Figure 2-5 Typical King Air 300 Electrical Power Distribution (1 of 2)

**TRIPLE-FED BUS**

- + No. 1 Circuit Breaker Panel Feeder
 - Annunciator Power
 - Autofeather
 - Avionics Master
 - Ldg Gear Warning
 - L Bleed Air Warning
 - L Fire Detect
 - L Fuel Flow
 - L Ignitor Power
 - L Oil Press Warning
 - L Oil Temp & Press (FA-2 thru 110)
 - L Oil Press (FA-111 & after)
 - L Oil Temp (FA-111 & after)
 - L Start Control
- + No. 2 Circuit Breaker Panel Feeder
 - Annunciator Indication
 - Lndg Gear Indication
 - R Bleed Air Warning
 - R Fire Detect
 - R Fuel Flow
 - R Ignitor Power
 - R Oil Press Warn
 - R Oil Temp & Press (FA-2 thru 110)
 - R Oil Press (FA-111 & after)
 - R Oil Temp (FA-111 & after)
 - R Start Control
 - Rudder Boost (FA-39 & after)
 - Stall Warning
- + No. 3 Circuit Breaker Panel Feeder
 - Cabin Lights
 - Gen Reset
 - Inst Indirect Lights
 - L Main Engine Anti-Ice
 - Outside Air Temp
 - Pilot Alt & Air Data (Opt)
 - Pilot Turn & Slip Indicator
 - Surface Deice
 - Windshield Wiper
- + Circuit Protection Device Not Directly Accessible In Flight

HOT BATTERY BUS

- + Cabin Entry Lights (Note 1)
- + Ground Comm (Opt)
- + L Engine Fire Ext
- + R Engine Fire Ext
- + RNAV Memory (Opt)

BATTERY

- + Battery Switch/Relay

Note 1:

The Cabin Entry Lights Circuit Breaker also feeds the following:
Aft Baggage Compartment Lights (Eyeball Lights)
Chart Holder
Control Wheel Clock (LCD Only)
Door Handle Position Observation Lights
Door J-Hook Observation Lights
Door Step Lights
Emergency Instrument Indirect Lights
Exterior Entry Light (Under Wing)
Refreshment Bar Clock (Opt)

Figure 2-5 Typical King Air 300 Electrical Power Distribution (2 of 2)



LEFT GENERATOR BUS	CENTER BUS	RIGHT GENERATOR BUS
+ Subbus Feeder	+ Condenser Blower	+ Subbus Feeder
+ Auto Prop Deice	+ Electric Heat (Aft)	+ Aft Evaporator Blower
Auto Prop Deice CB Switch	+ Electric Heat (Fwd)	+ Air Conditioner Clutch
+ No. 2 Avionics Bus Feeder	+ Landing Gear Motor	+ Copilot W/S Anti-Ice Power
L GEN Avionics	+ No. 1 Inverter	+ Copilot W/S Anti-Ice Control
+ Refreshment Bar	+ L Prop Deice Power (Manual)	+ No. 2 Inverter
+ Vent Blower Power	+ R Prop Deice Power (Manual)	+ No. 3 Avionics Bus Feeder
	+ Subpanel Feeder	R GEN Avionics
+ Subpanel Feeder	Beacon Lights	+ Toilet
L Landing Light	Ice Lights	+ Subpanel Feeder
Nav Lights	Taxi Light	Recognition Lights
Tail Flood Lights		R Landing Light
+ Circuit Breaker Panel Feeder		R Pitot Heat
+ Altitude Alert (Opt)		Stall Warning Heat
AM-FM Stereo (Opt)		Strobe Lights
Avionics & Eng Inst Lts		+ Circuit Breaker Panel Feeder
Bus Tie Power, L GEN		Brake Deice (Opt)
Bus Tie Control		Bus Tie Power, R GEN
Bus Tie Indication		Bus Tie Control
Cigarette Lighter		Bus Tie Indication
CP Encoding Altimeter (Opt)		Pitch Trim (Opt)
Flap Ind & Control		Prop Gov Test
Flap Motor		Prop Sync
Flight Inst Lights		Reading Lights
Furnishings Master Control		R Chip Detect
L Chip Detect		R Fuel Vent Heat
L Fuel Vent Heat		R Stby Engine Anti-Ice
L Stby Engine Anti-Ice		Side Panel Lights
No Smk, FSB, & Baggage Lts		Subpanel, Ovhd, & Console Lts
R Bleed Air Control		
+ Triple-Fed Bus Feeder		+ Triple-Fed Bus Feeder
Pilot's W/S Anti-Ice Power RCCB		
Pilot's W/S Anti-Ice Control		
RCCB		
+ Circuit Protection Device Not Directly Accessible In Flight		

Figure 2-6 Typical King Air 350 (B300) Electrical Power Distribution (1 of 2)


TRIPLE-FED BUS

+ No. 1 Circuit Breaker Panel Feeder	+ No. 4 Circuit Breaker Panel Feeder
Annunciator Power	Bus Tie Power, TPL FED
Autofeather	Bus Tie Control
Avionics Master	Bus Tie Indication
Lndg Gear Warning	Cabin Alt High
L Bleed Air Warning	Cabin Diff High
L Fire Detect	Cabin Pressure Control
L Fuel Flow	Cabin Temp Control
L Ignitor Power	L Bleed Air Control
L Oil Pressure	Manual Prop Deice Control
L Oil Press Warning	Oxygen Control
L Oil Temp	R Main Engine Anti-Ice
L Start Control	Windshield Wiper
+ No. 2 Circuit Breaker Panel Feeder	+ No. 1 Fuel Panel Feeder
Annunciator Indication	L Aux Fuel Warn & Xfer
Lndg Gear Indication	L Firewall Valve
R Bleed Air Warning	L Fuel Press Warn
R Fire Detect	L Fuel Quantity
R Fuel Flow	L Fuel Quantity Warn
R Ignitor Power	L Stby Pump
R Oil Press	+ No. 2 Fuel Panel Feeder
R Oil Press Warning	AC Bus Control
R Oil Temp	Crossfeed
R Start Control	* Dual Fed Bus
Rudder Boost	R Aux Fuel Warn & Xfer
Stall Warning	R Firewall Valve
+ No. 3 Circuit Breaker Panel Feeder	R Fuel Press Warn
Airspeed Warn	R Fuel Quantity
Avionics Annunciator (Opt)	R Fuel Quantity Warn
Cabin Lights	R Stby Pump
Gen Reset	+ Subpanel Feeder
Inst Indirect Lights	Landing Gear Control
L Main Engine Anti-Ice	L Pitot Heat
Outside Air Temp	+ No. 1 Avionics Bus Feeder
Pilot Alt & Air Data (Opt)	TPL FED Avionics
Pilot Turn & Slip	
Surface Deice	

+ Circuit Protection Device Not Directly Accessible In Flight

* The Dual Fed Bus is fed from the Triple-Fed Bus No. 2 Fuel Panel Feeder and the Battery Bus

BATTERY BUS

Avionics
Battery Switch/Relay
* Dual Fed Bus
Ground Comm

DUAL FED BUS

Cabin Entry Lights (Note 1)
L Engine Fire Ext
R Engine Fire Ext

BATTERY

Battery Bus Control

Note 1:

The Cabin Entry Lights Circuit
Breaker also feeds the following:
Aft Baggage Compartment Lights
(Eyeball Lights)
Chart Holder
Control Wheel Clock (LCD Only)
Door Handle Position
Observation Lights
Door J-Hook Observation Lights
Door Step Lights
Emergency Instrument Indirect
Lights
Exterior Entry Light (Under Wing)
Refreshment Bar Clock (Opt)
Left Seat Riser Threshold Light
Spar Cover Light

Figure 2-6 Typical King Air 350 (B300) Electrical Power Distribution (2 of 2)



Figure 2-7 Overhead Light Control and Meter Panel

TER BUS SELECT switch adjacent to the voltmeter (Figure 2-7).

The electrical system is designed to provide maximum protection against loss of electrical power should a ground fault (or short) occur. High current (Hall Effect) sensors, bus tie relays and current limiters are provided to isolate a fault from its power source. The electrical system bus arrangement is designed to afford multiple power sources for certain systems.

DC POWER DISTRIBUTION

The DC power distribution system is commonly called a “triplefed” system since most buses receive power from three sources (Figures 2-27 and 2-42). Many systems receive power from the triplefed bus. The triple-fed bus is powered from the three sources of electrical power onboard the aircraft and it only receives power. It is not used to transfer electricity from one part of the electrical system to another, such as the center bus. In this manner, a backup power source is available to most of the aircraft electrical systems.

In normal operation, all buses are automatically tied together so that all sources collectively supply power through individual protective devices.

Three in-flight DC power sources are available:

- One 24-volt, 20-cell, 34- or 36-ampere-hour Nickel-Cadmium Battery.
- Two 28-volt, 300-Ampere Start/Generators

When the battery switch is turned ON, the battery relay and the battery bus tie relays close (Figures 2-29 and 2-45). Battery power is routed through the battery relay to the triple-fed bus, and through the battery bus tie relay to the center bus and to both starter relays to permit starting either engine. However, without generator or external power neither generator bus is powered, since the generator bus ties are normally open when only battery power is available.

Normal before-starting procedures on the King Air 300 require the generator ties to be closed manually. The King Air 350 does not.

After either engine has been started and its generator switch has been moved to RESET, the generator control unit (GCU) will bring the generator up to normal system voltage. Releasing the spring-loaded switch to the center ON position closes the generator line contactor, thereby powering the generator bus and closing both generator ties automatically (the MAN TIES CLOSED advisory annunciation will extinguish if the generator ties have been manually closed). This action distributes power through the right 250-amp current limiter and generator bus tie relay to the center bus. From the center bus, electricity will flow to the battery, through the battery bus tie, and to the left generator bus, through the left generator bus tie and 250-amp current limiter. Power is also fed to the triple-fed bus from the right generator bus. When both generators are operating, each generator directly feeds its own generator bus which in turn feed the center bus, triple-fed bus, hot battery bus, and the battery, if it is discharged.



The generator buses and the battery are tied together by the center bus. The triple-fed bus is powered (or fed) from the battery and each generator bus through 60-amp limiters and through diodes providing fault isolation protection between the power sources.

BUS TIE SYSTEM

The electrical system is protected from excessively high current flow by the bus tie system (Figures 2-27 and 2-42). Three current sensors, consisting of Hall Effect devices and solid-state circuitry, are used to sense current flow through the portion of the circuit being monitored. Two bus tie sensors and their relays are located between the generator buses and the center bus, and a third is between the battery and the center bus.

With no power applied to the aircraft electrical system, all three bus tie relays are open. When the BAT switch is turned ON, battery voltage energizes the control circuit of the battery bus tie relay and the battery relay, thereby closing them. This action has no effect on the generator bus ties.

A similar action occurs when a generator or external power is brought on-line. When either generator is brought on-line, voltage from the generator control unit energizes the control circuit of both generator bus tie relays. This causes the L and R GEN TIE OPEN annunciators to extinguish and the generator bus tie relays to close. When external power is brought on-line to the aircraft electrical system, voltage from the external power switch energizes the control circuit of both generator bus tie relays. Neither generator nor external power affect the battery bus tie circuitry.

The switch on the pilot's left subpanel, (Figure 2-4) placarded GEN TIES - MAN - CLOSE - OPEN, is used to manually close both generator bus tie relays prior to energizing a generator. It is also used to manually open both generator bus tie relays at any time. This switch must be lifted (lever-lock) to move it from center (NORM) to either OPEN or MAN CLOSE. It will lock

OPEN but will spring back to NORM from MAN CLOSE. Only the generator bus tie relays may be manually opened or closed with this switch. Manually closing the generator bus tie relays will connect the generator buses to the center bus and power the entire system (Figures 2-30 and 2-46). Momentarily placing the switch to MAN CLOSE applies bus voltage to the coil of the generator bus tie relays, completes a latching circuit, activates the MAN TIES CLOSED annunciator and closes the generator bus tie relays. The latching circuit is completed through the contacts of the control relays for the generator line contactor. A generator bus tie relay cannot be manually closed if a fault opened the tie; the BUS SENSE switch must be momentarily activated to RESET, which resets the tie.

When a generator is brought on-line to the aircraft electrical system, the power source for the generator bus tie control is transferred from the manual close circuitry to the generator control units, which continue to hold the generator bus ties closed. In addition, the MAN TIES CLOSED annunciator extinguishes only when a generator is brought in-line.

When the generator ties are already closed, the GEN TIES switch can open the generator bus ties as procedure may dictate. When the GEN TIES switch is positioned to OPEN, the grounding portion of the control circuit is removed from both generator bus tie control circuits, allowing the generator bus ties to spring open.

HALL EFFECT SENSORS

Hall Effect sensors are solid state devices that sense current flow in one direction. They do this by actually sensing the intensity of the magnetic field around the conductor, since current flow and magnetic field intensity have a proportionate relationship.

Activation of an internal solid-state switch within the sensor by a current of at least 275 (± 5) amperes will open the control circuit of the associated bus tie relay, causing it to open. The control circuit of the bus tie relay is latched open to prevent the bus tie relay from closing. Opening



a bus tie relay will illuminate the appropriate TIE OPEN annunciator. When a bus tie relay has been opened by excessively high current flow through the Hall Effect sensor (i.e. a bus fault), it can only be reset by momentarily activating the BUS SENSE switch on the pilot's left subpanel to RESET (Figure 2-4). All Hall Effect sensors are unidirectional. They only sense overcurrent in the direction of the arrow on the symbol.

The bus sense switch located on the pilot's left subpanel, controls the over current sensing portion of the bus tie system. This switch, placarded BUS SENSE - TEST - RESET, is spring-loaded to the center (NORM) position. Momentarily activating it to TEST connects bus voltage to all three current sensor test circuits (Figures 2-36 and 2-52). This voltage simulates the condition resulting from a high current through each bus tie relay. The solid-state switches of each sensor are thus activated to de-energize their respective relays, thereby opening the bus tie relays and activating the annunciators. Once activated, the test circuitry latches the bus ties open, preventing their automatic closing.

Current sensor reaction time is approximately 0.010 seconds for the generator current sensors and 0.120 seconds for the battery current sensor. Once activated, the relays latch open, and reaction time for the system is limited to the reaction time of the relays. Therefore, only momentary activation of the TEST switch is required. Prolonged activation of this switch will damage or destroy the sensor modules and should be avoided.

Momentary activation of the switch to RESET powers the coil of the bus tie relays, unlatching the test circuits and permitting the bus ties to energize (close). Since high-current sensing is latched out when the switch is in RESET, only momentary activation is desirable. This prevents accidental welding of the bus tie relay contacts and/or opening a 250-amp current limiter by a bus grounding fault.

BUS ISOLATION

Bus isolation is one noteworthy feature of the multi-bus electrical system. The two generator buses and the center bus are protected in part by high-current sensing (Hall Effect) devices. In case of excessive current draw on one bus, the associated sensor will partly isolate the affected bus by opening its bus tie, allowing the other buses to continue operating as a system. There may be other current limiters or devices that may open to complete the bus isolation. All current sensors are desensitized during engine starts, to allow the required high-current flow to pass from the power sources to the starter motor without causing the bus ties to open. The battery bus tie current sensor is desensitized and the battery bus tie is held closed during landing gear operation.

A 250-amp current limiter (a large slow-blow fuse) is also located in the circuitry between the center bus and each of the generator buses. Since the Hall Effect devices sense high current in only one direction, the current limiters provide protection in the opposite direction. If an overcurrent situation causes a current limiter to open, it will also, in part, cause bus isolation.

The current protection for the triple-fed bus is provided exclusively by 60-amp current limiters. Triple-fed bus isolation will occur only if all three of these limiters open.

For typical examples of bus isolation, refer to Figures 2-38 and 2-54 (generator bus), 2-39 and 2-55 (center bus), and 2-40 and 2-56 (triple-fed bus).

LOAD SHEDDING

Load shedding is another highly beneficial feature of the multibus electrical system. The electrical system will automatically remove excess loads (generator buses), when the power source is reduced to battery only. When both generators are secured (or fail), the generator bus ties open and the generator bus loads are "shed" (Figures 2-37 and 2-53). The battery will continue to power the center, triple-fed, and hot battery buses. If necessary, power to the generator buses can be restored by closing the generator ties man-



ually with the GEN TIES switch (Figures 2-30 and 2-46). When load shedding occurs in flight, land as soon as practical, unless the situation can be remedied and at least one generator brought back on-line. Refer to the Abnormal Indications discussion in this section and to the emergency procedures section of the *POH* for more details.

KING AIR 300 LOAD MANAGEMENT

Use of the equipment in Table 2-1 will reduce battery duration by the approximate times listed. Multiple usage of the following equipment is additive.

Table 2-1 KING AIR 300 LOAD MANAGEMENT

EQUIPMENT	OPERATING TIME (MIN)	REDUCTION IN BATTERY DURATION (MIN)
Single Stby Fuel Pump	5	2
Windshield Wipers	5	1
L Pitot Heat	Continuous	8
Manual Prop Deice	5	8
Surface Deice	1	0.5
Ice Lights	5	0.5
Indirect Instr Lights	5	0.5
Cabin Lighting	5	2
Taxi Lights	5	2
Nav Lights	Continuous	8
CONN 1 Xmit	5	1

NOTE

Equipment which remains operable is designated with a white circle around the control switch. Attitude reference will depend upon the specific instrument panel equipment. The landing gear should be manually extended to maximize battery duration. Taxi lights should be used for landing if required. With the equipment listed under LOAD MANAGEMENT selected OFF, the battery duration will be approximately 60 minutes (based on 25-amp load and a 75% battery capacity).

WARNING

Do not place the GEN TIES switch in the MAN CLOSE position. This action reconnects the L and R generator bus loads and severely limits the battery duration.

KING AIR 350 LOAD MANAGEMENT

The equipment listed in Table 2-2 remains operable after a dual generator failure. With only the equipment operating listed as "continuous" in the "OPERATING TIME" column, the battery duration will be approximately 30 minutes (based upon a 50-amp load and a 75% battery capacity).

Use of the equipment with prescribed operating times will reduce the battery duration by the approximate times listed. Multiple usage of this equipment is additive.

NOTE

Equipment which remains operable is designated with a WHITE CIRCLE around the control switch. Attitude reference will depend upon the specific instrument panel equipment. Refer to the LOAD MANAGEMENT table to



determine which attitude instruments will remain operable with a dual generator failure.

WARNING

Do not place the GEN TIES switch in the MAN CLOSE position. This action reconnects the left and right generator bus loads and severely limits the battery duration.

Table 2-2 KING AIR 350 LOAD MANAGEMENT

EQUIPMENT	OPERATING TIME (MIN)	REDUCTION IN MAIN BATTERY DURATION (MIN)
Air-driven Attitude Gyro	Continuous	----
Standby Attitude Gyro	Continuous	None*
Inverter 1	Continuous	----
Comm 1 Xmit	3	0.5
Pilot Audio	Continuous	----
Nav 1	Continuous	----
RMI 2	Continuous	----
Pilot Altimeter	Continuous	----
Pilot ADI (Electro-mechanical)	Continuous	----
Pilot EADI (EFIS)	Not Operational	----
Pilot HSI (Electro-mechanical)	Continuous	----
Pilot EHSI (EFIS)	Not Operational	----
Turn & Slip Indicator	Continuous	----

* Optional equipment. Powered by Auxiliary battery.

Table 2-2 KING AIR 350 LOAD MANAGEMENT (CONT.)

EQUIPMENT	OPERATING TIME (MIN)	REDUCTION IN MAIN BATTERY DURATION (MIN)
Annunciator Panel	As Required	----
Instrument Indirect/Emergency Lights	Continuous	----
Cabin Lights	5	2
Ice Light	5	0.5
Beacon Lights	Continuous	----
Taxi Lights	1	0.5
Digital OAT	Continuous	----
Fuel Quantity Indicator	Continuous	----
Single Standby Fuel Pump	5	1
Left Bleed Air Valve	Continuous	----
Pressurization Control	Continuous	----
Cabin Temperature Control	Continuous	----
Engine Ignition	0.5	0.1
Surface Deice	6 cycles	0.1
Left and Right Main Engine Anti-ice	Single Operation	0.1
Manual Prop Deice	3	3
Windshield Wiper	1	0.1
Left Pitot Heat	Continuous	----
Landing Gear	Single Operation	0.5

* Optional equipment. Powered by Auxiliary battery.



BATTERY

The nickel-cadmium (Ni-Cad) or sealed lead-acid battery is located in the right wing center section in an air-cooled box (Figure 2-8). The battery relay, battery bus tie relay, external power relay, battery bus tie relay hall effect sensor, right generator bus tie relay hall effect sensor, battery to triple-fed bus 60 amp current limiter, and the charge monitor shunt are mounted in the battery compartment immediately forward of the battery.

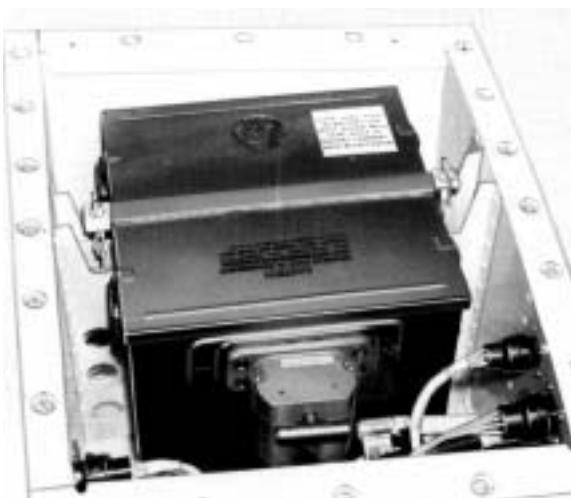


Figure 2-8 Battery Installation

Power to the triple-fed and center buses is routed from the battery via the battery relay and battery bus tie respectively, which are controlled by the battery switch on the pilot's left subpanel.

On the King Air 300, there is a hot battery bus for powering a few systems without having to turn the battery switch on. The hot battery bus is located in the battery compartment.

On the King Air 350, the battery bus and dual-fed bus, which serve the same purpose as the hot battery bus on the 300, are located on the fuel circuit breaker panel in the cockpit. A battery bus switch, on the pilot's left subpanel, controls a remote control circuit breaker located in the battery compartment. When this switch is in the NORMAL position, the remote control circuit

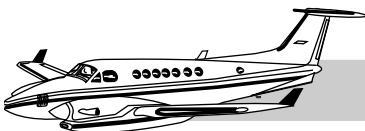
breaker is closed, which will connect the battery directly to the battery bus and dual-fed bus. When it is in the EMERGENCY OFF position, the remote control circuit breaker is open, which will disconnect the battery from the battery bus and dual-fed bus.

The hot battery bus on the King Air 300 and the battery bus and dual-fed bus on the 350 provide power directly to a few aircraft systems (Figures 2-27 and 2-42). These systems may be operated without turning the battery switch on. Care should be taken, however, to insure that utilization of these systems is minimal when the generators are inoperative and/or the aircraft is secured to prevent excessive discharge of the battery.

A battery charge monitor system advises the pilot of battery charge/discharge status. On King Air 300 serials FA-10 and later and all 350s, a battery ammeter located in the overhead meter panel provides a continuous readout of the current charge into, or drawn from, the battery. On all aircraft, a charge into the battery of more than 7 amperes for 6 or more seconds, will trigger the yellow BATTERY CHARGE annunciator.

The BATTERY CHARGE annunciator may occasionally be illuminated for short intervals when heavy electrical draw items are cycled. For example, following a battery-powered engine start, the battery recharge current is very high and causes illumination of the BATTERY CHARGE annunciator, thus providing an automatic self-test of the battery monitor system. As the battery approaches a full charge and the charge current decreases to a satisfactory level, the annunciator will be extinguished. This normally occurs within a few minutes after an engine start, but may require a longer time if the battery has a low state of charge.

These systems are designed for continuous monitoring of battery condition. Refer to the Abnormal Indications paragraph of training manual or the Abnormal and Normal Procedures sections of the *POH* for battery check procedures.



STARTER/GENERATORS

The starter/generators are dual-purpose, engine-driven units (Figure 2-9). The same unit is used as a starter to drive the engine during engine start and as a generator to provide electrical power when driven by the engine. A series starter winding is used during starter operation and a shunt field winding is used during generator operation. The regulated output voltage of the generator is 28.25 (± 0.25) volts with a maximum continuous load rating of 300 amperes.

In addition to the starter/generators, the generator system consists of control switches, generator control units (GCU), line contactors and loadmeters.

Starter power to each individual starter/generator is provided from the center bus through a starter relay (Figures 2-31 and 2-47). The start cycle is controlled by a three-position switch,

one for each engine, placarded: IGNITION AND ENGINE START - LEFT - RIGHT - ON - OFF -STARTER ONLY, located on the pilot's left subpanel (Figure 2-4). Actuating a switch to either the STARTER ONLY or ON position supplies a signal to the start relay to energize the starter, and to the generator field sense relay to disable the shunt field and prevent generator operation during the start cycle. The starter drives the compressor section of the engine through accessory gearing.

The starter relay connects the battery to the starter/generator during engine starts. With one engine running and its generator on the line, the opposite engine can be started using power from the battery and the operating generator channeled through the starter relay. This is called a cross-generator start. Normally one engine is started on battery power alone, and the second engine uses a cross-generator start.

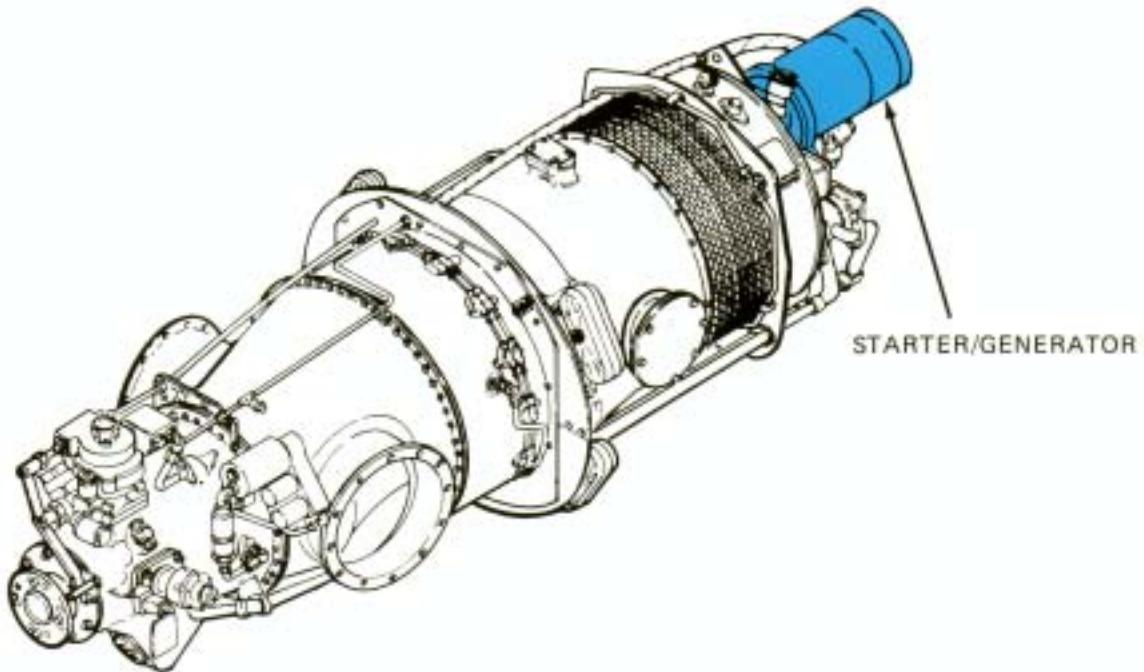


Figure 2-9 Starter/Generator Installation



The operating generator control unit will limit its generator's output to no more than 400 amps during a cross-generator start (Figure 2-33). This assures that the 250-amp current limiter on the operating generator side will not open due to transient surges, since the generator would normally provide most of the current required for the start. In addition, while a starter is selected, the bus tie sensors are disabled to prevent them from opening their respective bus tie relays.

CAUTION

Do not exceed the starter motor operating time limits of 30 seconds ON, five minutes off, 30 seconds ON, five minutes off, 30 seconds ON, then 30 minutes off.

NOTE

The above limitation is only applicable when the starter is driving the engine, not when the engine is driving the starter.

DC Generation

The generator phase of operation is controlled by the generator switches, located in the pilot's left subpanel, next to the BAT switch under the MASTER SWITCH gang bar (Figure 2-4). The switches provide OFF, ON and RESET capabilities.

The generating system is self-exciting and does not require electrical power from the aircraft electrical system for operation. It utilizes generator residual voltage for initial generator buildup.

Generator operation is controlled through two generator control units (GCU), mounted below the center aisle floor, that make constant voltage available to the buses during variations in engine speed and electrical load requirements. The load on each generator is indicated by the respective left and right loadmeters located in the overhead meter panel (Figure 2-7).

The GCUs provide the following functions:

- Voltage Regulation and Line Contactor Control
- Overvoltage and Overexcitation Protection
- Paralleling/Load Sharing
- Reverse-Current Protection
- Cross-Generator-Start Current Limiting

Voltage Regulation and Line Contactor Control

The generators are normally regulated to 28.25 ($\pm .25$) VDC.

When the generator switch is held in RESET, generator residual voltage is applied through the GCU to the generator shunt field causing the generator output voltage to rise. When the switch is released to ON, the 28-volt regulator circuit takes over and begins controlling the generator shunt field in order to maintain a constant output voltage. The voltage regulator circuit varies shunt field excitation as required to maintain a constant 28-volt output from the generator for all rated conditions of generator speed, load and temperature.

When the generator switch is released to ON generator voltage is applied to the GCU to enable the line contactor control circuit. The GCU compares generator output voltage with aircraft bus voltage. If the generator output voltage is within 0.5 volts of aircraft bus voltage, the GCU sends a signal to the line contactor, which closes and connects the generator to the aircraft bus (Figures 2-32 and 2-48), and closes both generator ties to connect the center bus and the generator buses, allowing the generator to recharge the aircraft battery and power all aircraft electrical loads.

When a generator fails or is turned off, the GCU opens the line contactor and isolates the inoperative generator from its bus.



Overvoltage and Overexcitation Protection

The GCU provides overvoltage protection to prevent excessive generator voltage from being applied to aircraft electrical equipment. If a generator output reaches the maximum allowable 32-volts, the overexcitation circuits of the GCU will detect which generator is producing excessive voltage output and attempting to absorb all the aircraft electrical loads. The GCU overexcitation circuit will then disconnect the generator from the electrical system.

Overexcitation protection is provided by the generator control panel. This portion of the control panel will activate in the event the generator load and speed conditions are such, or the nature of the fault is such, that the generator voltage increases without control, but does not reach an overvoltage condition. Should the generator field reach the designed limitation value, the circuitry providing overexcitation protection will be activated to remove the affected generator from the bus.

Paralleling/Load Sharing

The paralleling circuit averages the output of both generators to equalize load levels. The paralleling circuits of both GCUs become operative when both generators are online. The paralleling circuits sense the interpole winding voltages of both generators to provide an indication of the load on each generator. The voltage regulator circuits are then biased up or down as required to increase or decrease generator loads until both generators share the load equally. The GCU's are designed to balance loads to within 10 percent.

Reverse-Current Protection

Reverse-current protection is provided by the GCU. When a generator becomes underexcited or cannot maintain bus voltage for some reason (i.e., low generator speed during engine shutdown), it will begin to draw current (reverse current) from the aircraft electrical system. The GCU senses reverse current by comparing generator output voltage to generator bus voltage. When bus volt-

age exceeds output voltage, the GCU opens the line contactor to protect the generator.

Cross-Generator-Start Current Limiting

During a cross-generator start, when the IGNITION AND ENGINE START switch on the second engine is activated to ON, a signal from the switch is applied to the GCU of the operating generator. This activates the cross-start current limiting circuit and limits output of the operating generator to no more than 400 amps which protects the 250-amp current limiter on the operating generator side. While a starter is selected the bus tie sensors are disabled to prevent them from opening their respective bus tie relays. When using STARTER ONLY to motor the engine, the same functions occur.

EXTERNAL POWER

The external power receptacle, located under the right wing outboard of the nacelle, is provided to connect an external power unit to the electrical system when the airplane is parked. The power receptacle is designed for a standard AN plug.

When external power is connected, the external power relay can close only if the polarity of the voltage being supplied to the external power receptacle is correct (Figures 2-41 and 2-57).

Whenever an external power plug is connected to the receptacle and the BAT switch is ON, the yellow EXT PWR annunciator will either be illuminated continuously or flash dependent upon certain other parameters. If the external power unit is OFF, or the EXT PWR switch is OFF, the annunciator will flash. With the EXT PWR switch and external power unit ON, the annunciator will be on steady, provided external power voltage is sufficient to power the aircraft electrical system.

External power voltage can be monitored anytime—even before the EXT PWR switch on the pilot's left subpanel is switched ON - by turning the VOLTMETER BUS SELECT switch in the



overhead panel (Figure 2-7) to the EXT PWR position and reading the voltage on the voltmeter.

A high-voltage sensor locks out and opens the external power relay if external power exceeds 31 (± 0.5)volts DC. A low-voltage sensor causes the EXT POWER annunciator to flash when external power output voltage is below 24 (± 1.5) volts DC.

When the EXT PWR switch is ON, the external power relay closes. As external power enters the aircraft, the left and right generator bus tie relays close, permitting power to reach all buses. Consequently, the entire electrical system can be operated, including the starters (Figure 2-41 and 2-57).

The ground power source should be capable of delivering up to 1000 amps momentarily, 300 amps continuous, during engine starts. Use of an inadequate ground power unit can cause a voltage drop below that required by the starter relay, resulting in possible relay chatter and welded contacts. Aircraft electrical loads should be reduced to the minimum practical prior to attempting an external power start.

Observe the following precautions when using an external power source:

CAUTION

THE RECOMMENDED MINIMUM INDICATED BATTERY VOLTAGE PRIOR TO CONNECTING EXTERNAL POWER IS 23 VOLTS. HOWEVER, NEVER CONNECT AN EXTERNAL POWER SOURCE TO THE AIRPLANE UNLESS A BATTERY INDICATING A CHARGE OF AT LEAST 20 VOLTS IS IN THE AIRCRAFT.

If the battery voltage is less than 20 volts, the battery must be recharged, or replaced with a battery indicating at least 20 volts, before connecting exter-

nal power. Use only an external power source fitted with an AN-type plug.

Voltage is required to energize the avionics master power relays to remove the power from the avionics equipment. Therefore, never apply external power to the airplane without first applying battery voltage.

The battery may be damaged if exposed to voltages higher than 30 volts for extended periods of time.

Refer to the Normal Procedures section of the *POH* for procedural details of using external power.

AVIONICS MASTER POWER

The avionics systems installed on each airplane usually consist of individual nav/com units, each having its own on-off switch. Avionics packages will vary on different airplane installations. Due to the large number of individual receivers and transmitters, a Beech avionics master switch placarded AVIONICS MASTER POWER is installed on the pilot's left subpanel. An Avionics Master Power Schematic diagram is shown in Figure 2-10. Refer to the Avionics section of this training manual for details of the avionics system.

AC POWER

KING AIR 300 AC POWER DISTRIBUTION

AC power for the avionics equipment and the AC-powered engine instruments is supplied by two inverters (Figure 2-11). Either one may be used at the pilot's discretion through the inverter selector switch. Each inverter provides two levels of power: 115 volts, 400 Hz for avionics equipment and 26 volts, 400 Hz for the torque meters and applicable avionics. Output of the torque meter's inverter is rated at 250 volt-amps.

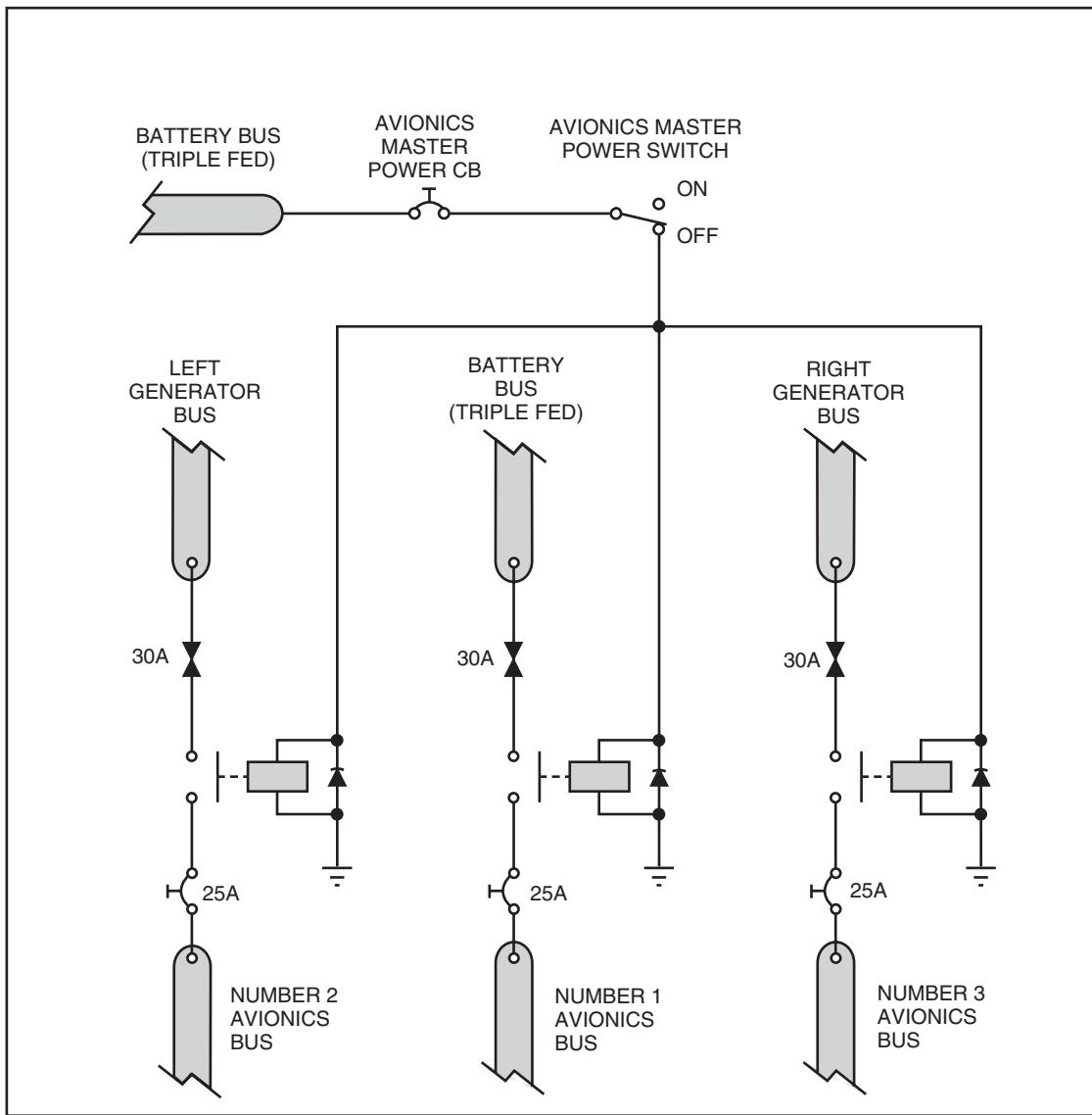


Figure 2-10 Avionics Master Power Schematic

Optional inverters are available, rated as necessary to meet aircraft AC power requirements. The inverters are installed in the wings, immediately outboard of each nacelle.

Inverter operation is controlled by the INVERTER NO. 1 - OFF - NO. 2 select switch on the pilot's left subpanel (Figure 2-4). Selection of either inverter actuates the corresponding inverter power relay to supply it with DC power. An inverter-select relay provides the necessary switching to permit the operating inverter to sup-

ply 26 VAC avionics and instrument power, and 115 VAC avionics and test-jack power. The inverter select relay is energized when the number one inverter is selected (Figure 2-12). It is deenergized when the inverter switch selects either OFF or NO. 2.

Dual sources of DC power are provided for each inverter (Figures 2-13 and 2-14). The power-select relay for each inverter is automatically selected to provide inverter power from the adjacent generator bus, or from the center bus if the

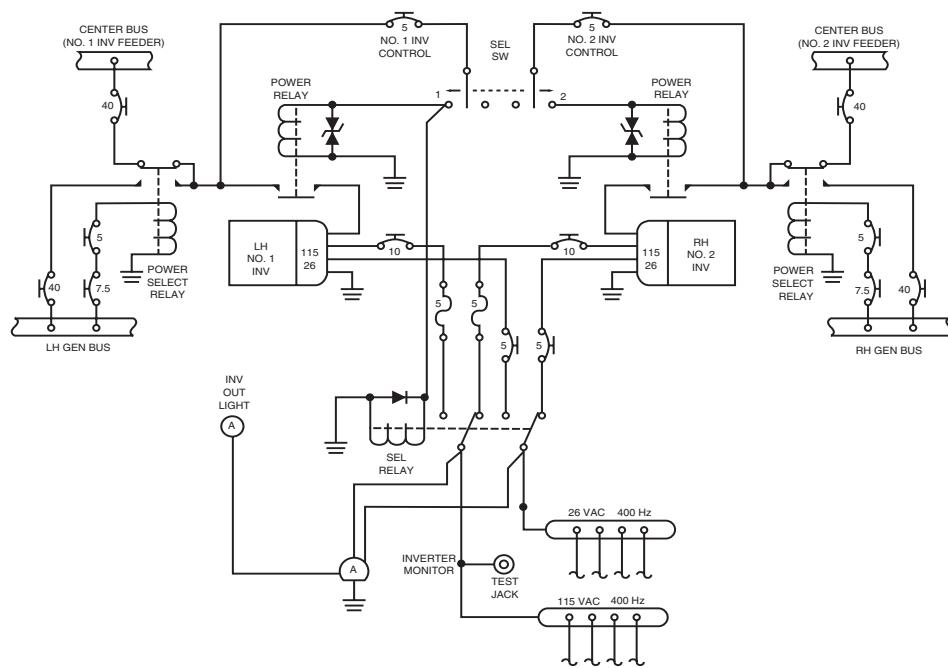


Figure 2-11 King Air 300 Simplified Inverter Schematic

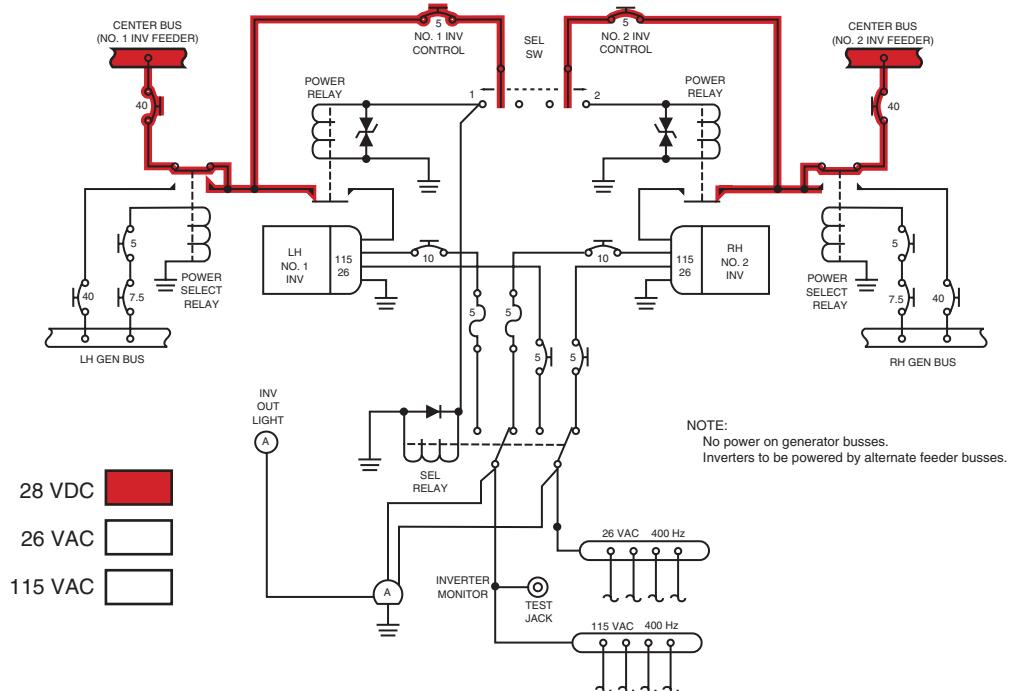


Figure 2-12 King Air 300 Inverter Schematic—Condition 1

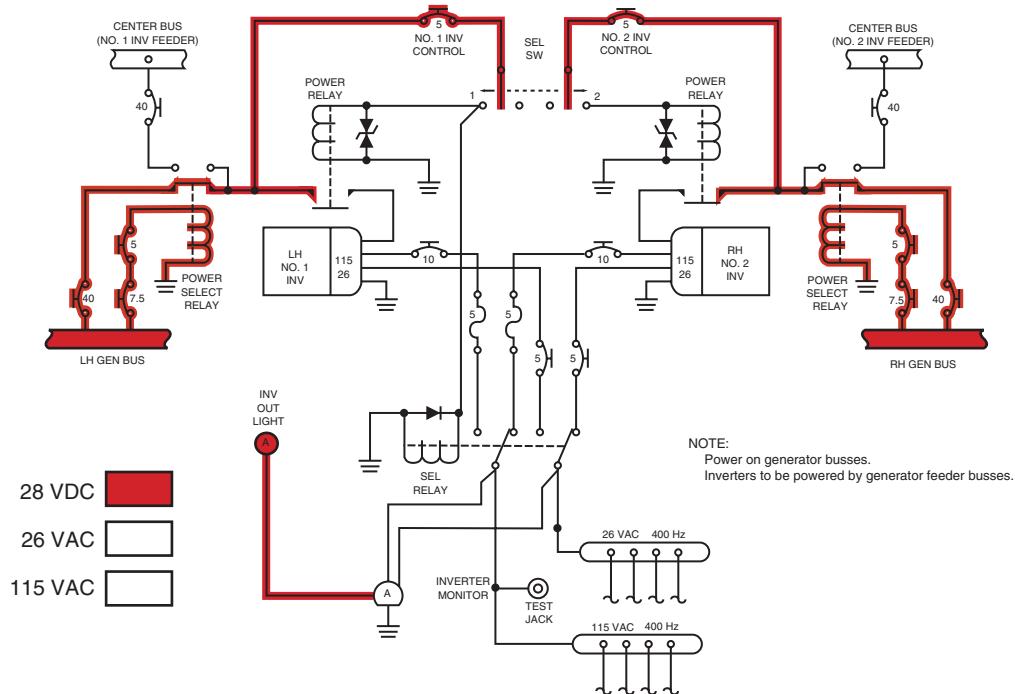


Figure 2-13 King Air 300 Inverter Schematic—Condition 2

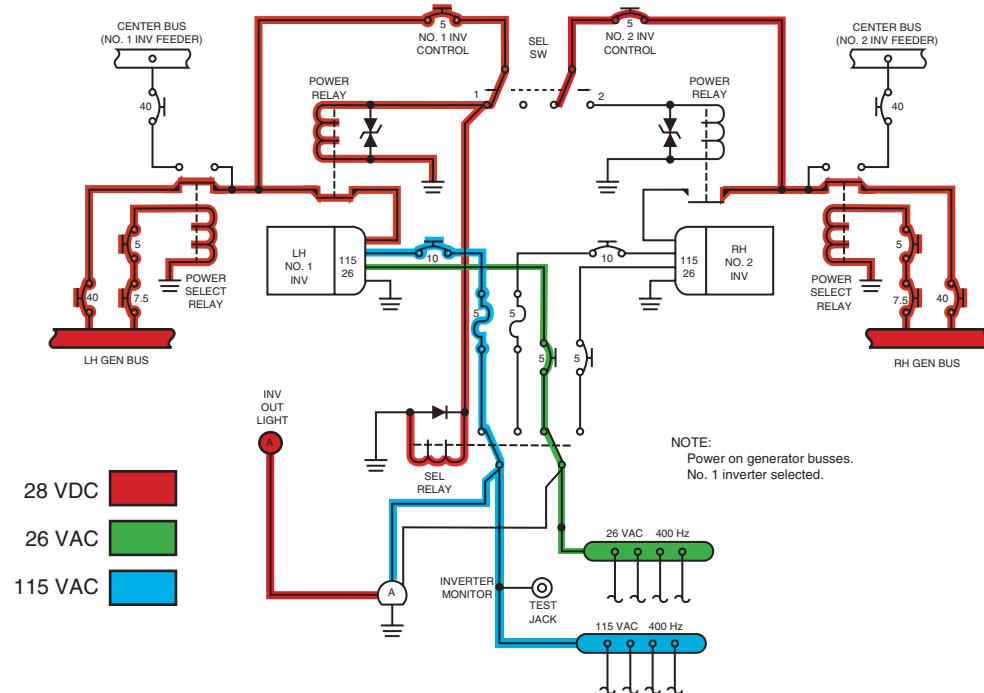


Figure 2-14 King Air 300 Inverter Schematic—Condition 3



generator bus is deenergized. When battery power is applied to the center bus prior to engine start (Figure 2-29), inverter power is routed through a circuit breaker and the normally-closed contacts of an inverter power-select relay to the power relay of each inverter. When the generator buses are powered (Figure 2-34), voltage is also routed through a circuit breaker on the copilot's circuit breaker panel to the coil of each inverter power-select relay, causing the inverter to be powered by its generator bus.

During normal operation, the inverter power select relays are energized and power is supplied from the respective generator bus. Should a fault occur that would interrupt power to that bus, the power select relay would de-energize and inverter input power would be taken from the center bus of the airplane, precluding loss of an inverter due to failure of the generator bus.

When either inverter is initially selected, the INVERTER annunciator will flash three times (Figure 2-15). The inverter monitor checks for proper inverter output of 115 VAC (± 10 percent) and 26 VAC (± 10 percent), both at 400 Hz (± 5 percent). If the inverter annunciator remains extinguished after the third flash, it indicates proper operation of the inverter.

Inadequate inverter output power is indicated by the illumination of the INVERTER annunciator. If the INVERTER annunciator becomes illuminated and remains illuminated, select the other inverter. Refer to the Emergency Procedures section of the *POH*.

The two 115/26-VAC 400 Hz inverters are controlled by the INVERTER switch. Inadequate inverter power is indicated by the illumination of the INVERTER annunciator. When either inverter is first selected, the INVERTER annunciator will flash three times in three seconds indicating proper operation of the inverter monitor. Any intermittent condition will cause the INVERTER annunciator to lock ON. If the inverter annunciator illuminates, select the other inverter.

KING AIR 350 AC POWER DISTRIBUTION

AC power for certain avionics and the torque meters is supplied by 2 solid-state inverters (Figures 2-16, 2-17, 2-18, 2-19, and 2-20). Both inverters are to be used for flight. The 2 AC output voltages are: 115 volts and 26 volts. The frequency of the AC power is 400 Hz. There is a 115 volt AC bus as well as a 26 volt AC bus for each inverter. The AC equipment is distributed so that approximately one half of the equipment is powered by each inverter. The output of a typical inverter installed on a King Air 350 is 750 volt-amps. The inverters are installed in the wings, immediately outboard of each nacelle.

The DC power for the number 1 inverter comes from the center bus and for the number 2 inverter from the right generator bus. There is a control switch for each inverter. These switches have 3 positions that are from top to bottom: ON, OFF and BUS TRANSFER.

Anytime the center bus is powered, the number 1 inverter has DC power to it. Also, when the right generator bus is powered, the number 2 inverter has DC power to it. However, the inverters will not operate until the respective inverter switch is placed to the ON position. The inverter switch will complete the internal inverter control circuit when the switch is in the ON position. When the switch is in the OFF or BUS TRANSFER position, the respec-

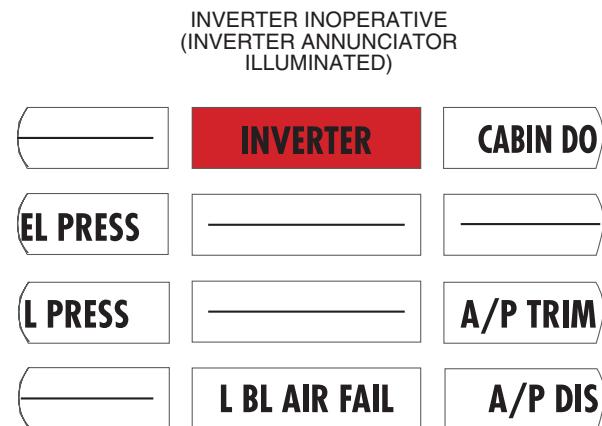


Figure 2-15 INVERTER Annunciator

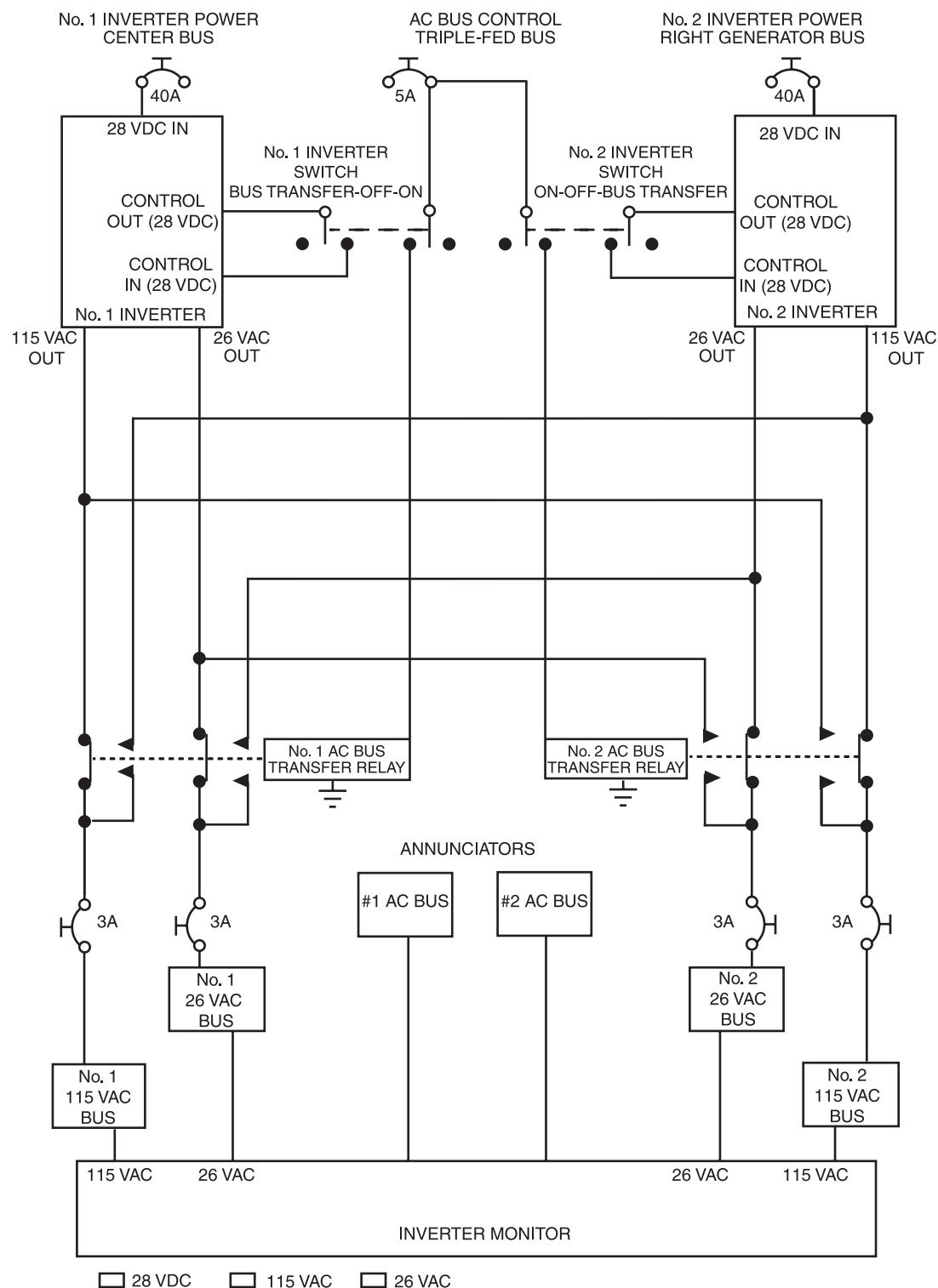


Figure 2-16 King Air 350 Simplified Inverter Schematic

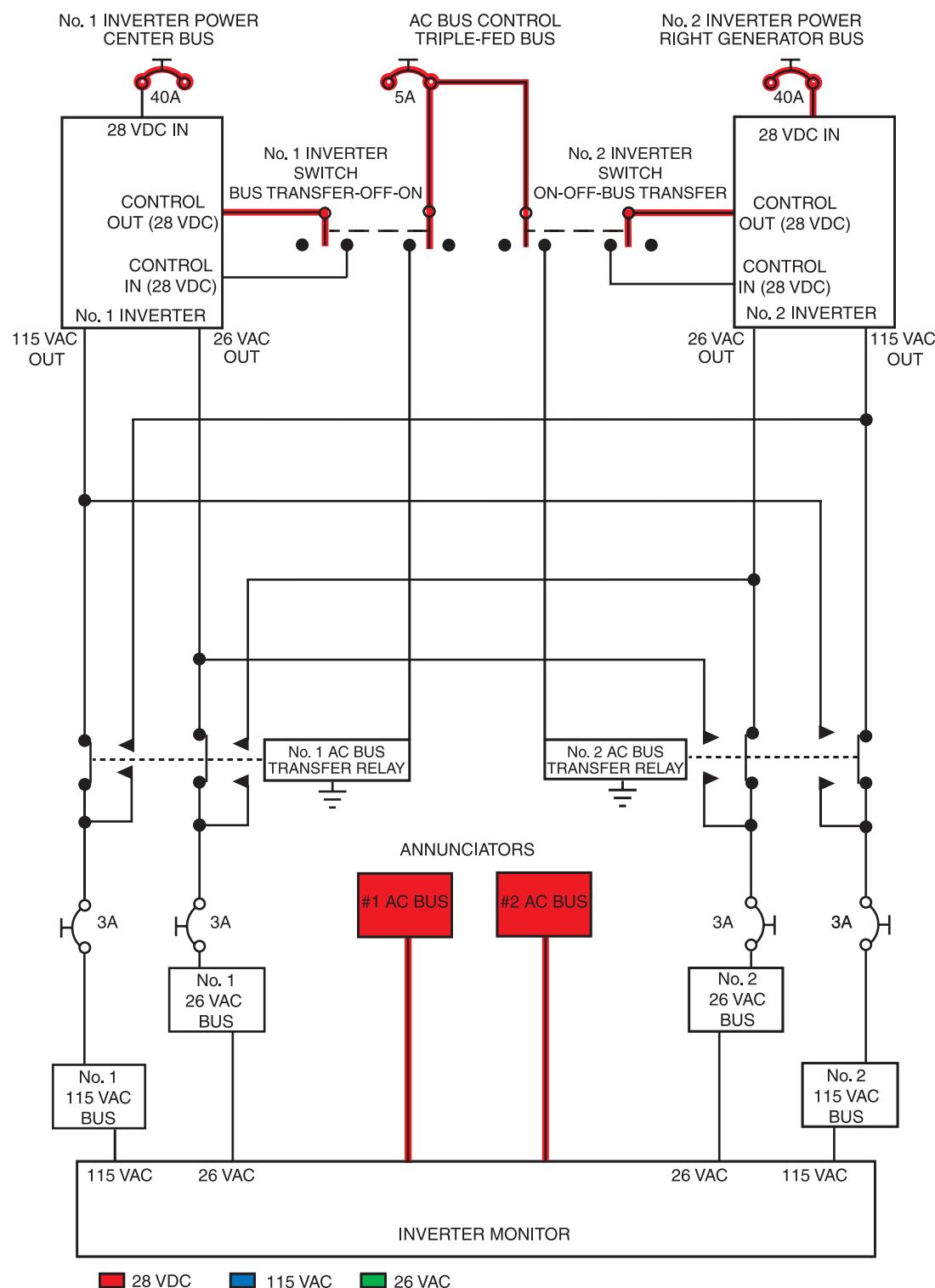


Figure 2-17 King Air 350 AC Power—All DC Buses Powered, Both Inverters OFF

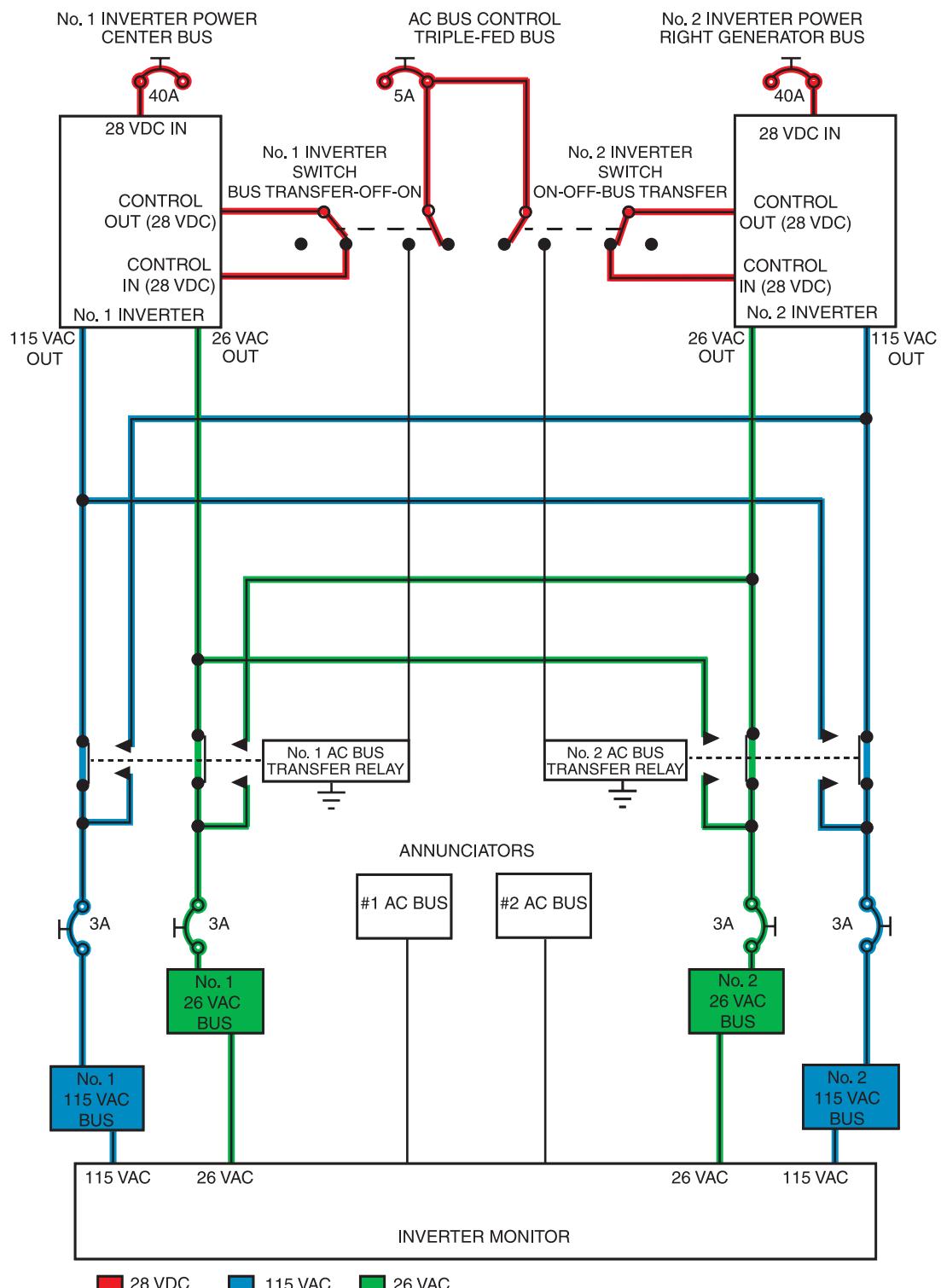


Figure 2-18 King Air 350 AC Power—All DC Buses Powered, Both Inverters ON

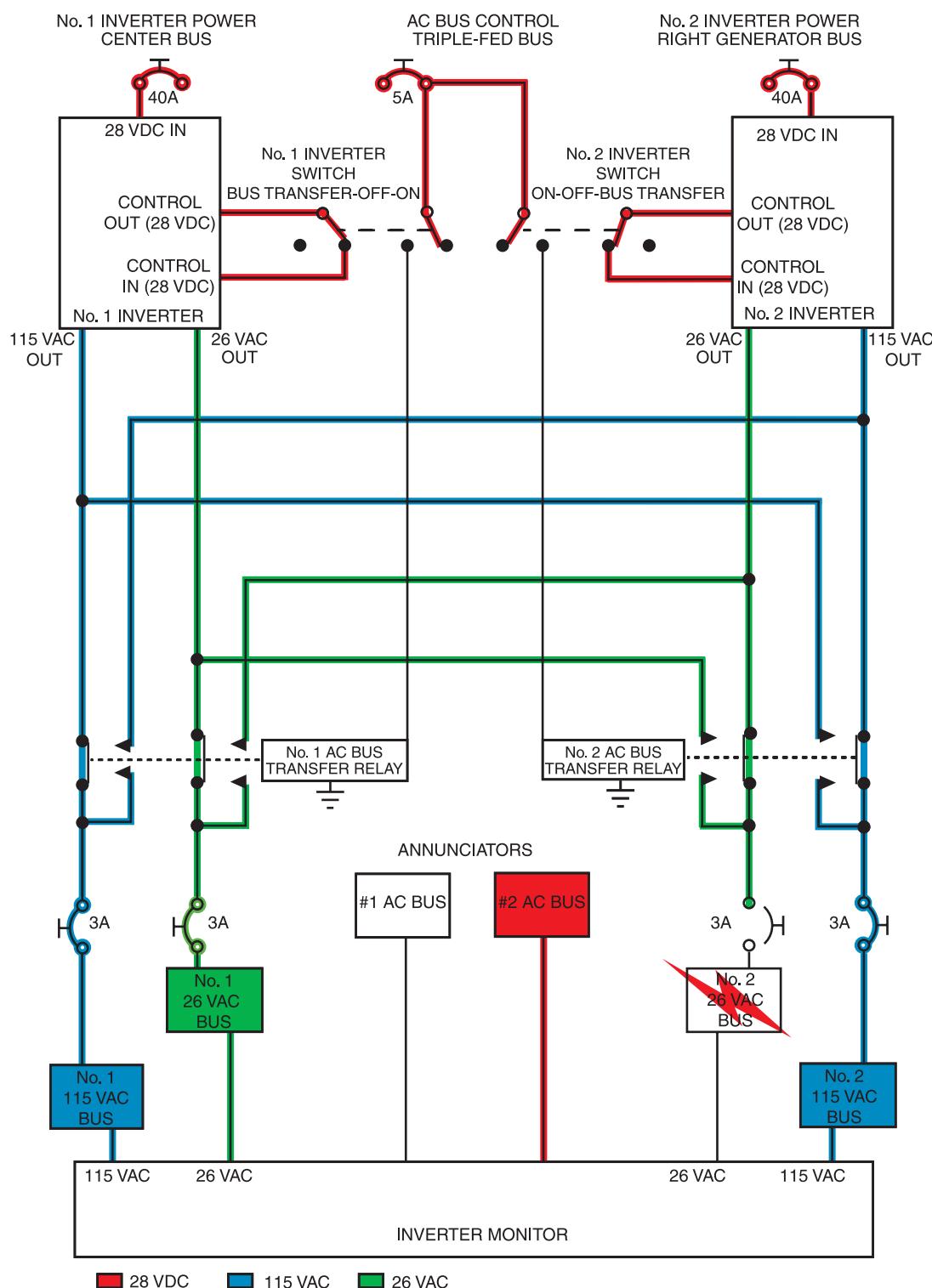


Figure 2-19 King Air 350 AC Power—All DC Buses Powered, Both Inverters ON, No. 2 (26 VAC) Bus Fault

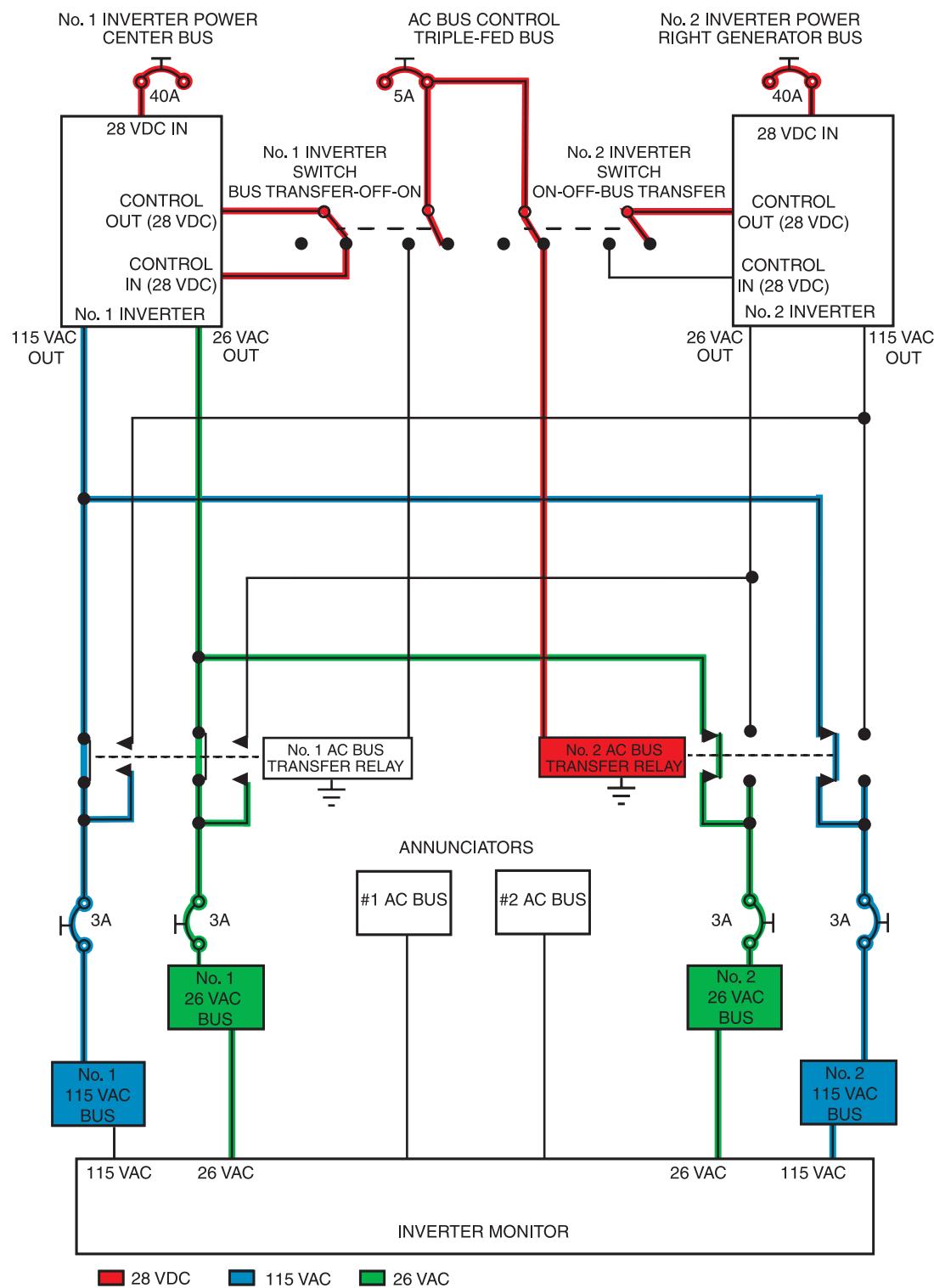
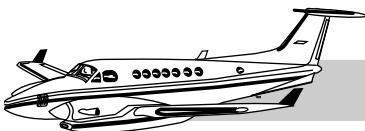


Figure 2-20 King Air 350 AC Power—All DC Buses Powered, No. 1 Inverter ON, No. 2 Inverter BUS TRANSFER



tive internal inverter control circuit will be interrupted and the inverter will not operate. Both inverters are to be used for flight and they each feed their respective AC buses. When both inverters are on, a frequency sync circuit from each inverter to the inverter monitor ensures that both inverters are phase locked.

Each set of AC buses (the 26 and 115 volt buses which are fed from the same inverter) has its own red AC BUS annunciator labeled #1 AC BUS or #2 AC BUS. An inverter monitor takes care of monitoring all 4 AC buses, and if voltage or frequency goes out of tolerance on an AC bus, for whatever reason, the inverter monitor will illuminate the proper AC BUS annunciator.

If an inverter malfunctions in some way (the inverter stops operating, the 115 volt AC or 26 volt AC outputs are out of tolerance, or the 400 hertz frequency is out of tolerance) the malfunctioning inverter's AC buses may be disconnected from that inverter and reconnected to the remaining inverter by placing the malfunctioning inverter's control switch in the **BUS TRANSFER** position.

When an inverter control switch is placed in the **BUS TRANSFER** position, a circuit is completed from the 5 amp AC bus control circuit breaker, through the inverter control switch to the respective AC bus transfer relay. This energizes the control circuit of this AC bus transfer relay, which then disconnects the inverter from the AC buses normally fed by that inverter, and reconnects them to the opposite inverter. All like voltage AC buses will then be connected together and to the same inverter.

If a fault exists with a particular AC bus, that AC bus circuit breaker will trip. The AC bus circuit breakers are located on the fuel panel circuit breaker panel.

Refer to the *POH* for the AC BUS annunciator emergency procedure.

CIRCUIT BREAKERS

Both AC and DC power are distributed to the various aircraft systems via circuit breaker panels which protect most of the components in the airplane. Two of these panels are located in the cockpit. Each of the circuit breakers has its amperage rating printed on it.

The smaller breaker panel is located to the left of the pilot below the fuel management panel (Figure 2-21). On the 300, this panel contains the circuit breakers for the fuel system only. On the 350, this panel contains the circuit breakers for the fuel system, AC electrical system, and the battery bus and dual-fed bus.

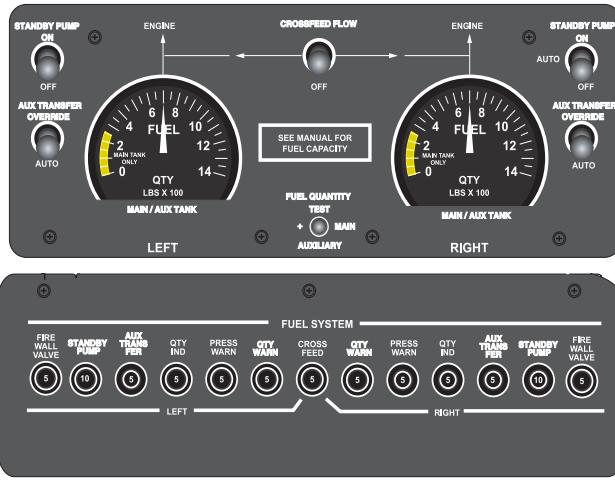
The larger circuit breaker panel is located on the copilot's sidewall (Figure 2-22). This panel contains the breakers for many electrical systems, including major engine-related systems, avionics components, environmental systems, lights, annunciator warning systems, and other small systems. Equipment powered by the same feeder is connected by white lines on the panel faceboard. The circuit breakers for the bus tie systems and avionics bus feeders are also located on this panel.

Some components, such as exterior lighting and ice protection equipment, are protected by circuit breaker switches on the pilot's right subpanel. Such switches have the amperage rating stamped on the end of the switch.

Procedures for handling tripped circuit breakers and other related electrical system warnings can be found in the Emergency and Abnormal Procedures section of the *Pilot's Operating Handbook*. As a general rule, if a non-essential circuit breaker trips in flight, do not reset it. Resetting a tripped breaker could cause further damage to the component or system. If an essential system circuit breaker such as an avionics breaker trips, let it cool and then reset it. If it fails to reset, DO NOT attempt to reset it again. Take corrective action according to the procedures in the appropriate section of the *POH*.



King Air 350



King Air 300



King Air 350 FL121 and After

Figure 2-21 Fuel Management Panel



If, for some reason, all the avionics equipment drops off-line but does not trip the circuit breaker, the trouble may be in the AVIONICS MASTER switch. The switch can be bypassed, and your radios returned to service, by pulling the AVIONICS MASTER circuit breaker on the copilot's circuit breaker panel (Figure 2-22). Figures 2-28 through 2-36 and 2-45 through 2-52 are presented in the order that is appropriate when going through the Before Engine Starting (Battery), Engine Starting and Before Taxi checks.

EMERGENCY AND ABNORMAL INDICATIONS

Electrical fires are covered in the Emergency Procedures section of the *Pilot's Operating Handbook (POH)*.

Emergency and abnormal in-flight situations are described in the Emergency and Abnormal Procedures section of the *POH*. Generator and battery irregularities are described there under Electrical System Failures.

Generator bus and center bus faults/malfunctions are normally associated with illumination of the corresponding bus tie annunciators. A triple-fed bus fault/malfunction does not have an associated bus tie indication. In all cases, a bus problem should be investigated by referencing other bus indications, such as loss of related equipment, and by checking bus voltages with the voltmeter selector switch and/or loadmeter indications.

Complete bus loss is a highly unlikely situation. The multi-bus electrical system has protection devices that will normally isolate a fault with minimum equipment loss. In addition to the main bus structure, each bus is subdivided into several subbuses. A loss of only a few pieces of equipment on any particular bus may be indicative of a partial bus fault/malfunction or a subbus fault. Figure 2-6 provides a listing of many items that are normally grouped together on buses and subbuses.

Battery malfunctions are extremely rare due to the battery installation and design. There have been a few cases in aircraft with similar installations where malfunctions have occurred; however, the battery monitoring system has provided sufficient warning to the pilot for timely corrective action to be completed before the situ-

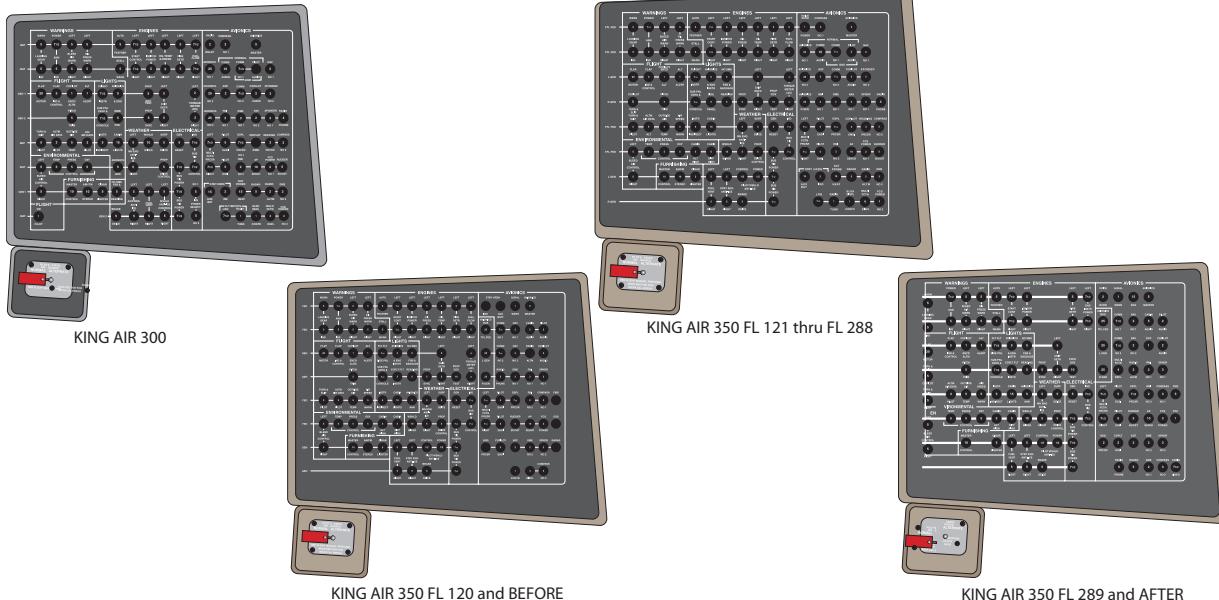


Figure 2-22 Right Side Panel (Typical)



ation could deteriorate to a more serious condition. There are two reasons for this safety margin: firstly battery malfunctions are historically slow to develop and can be identified early with the charge monitoring system; and secondly, sufficient cooling is provided to the battery in flight which diminishes the likelihood of serious damage to the airplane or its electrical system. Therefore, when identified early, a battery malfunction will not deteriorate into a serious condition as long as the pilot complies with the proper procedures as outlined in the *POH*.

L OR R DC GEN ANNUNCIATOR ILLUMINATED (GENERATOR INOPERATIVE)

If the left or right DC GEN annunciator (Figure 2-23) on the caution/advisory panel is illuminated while in flight, verify with the associated loadmeter, then push the corresponding generator to switch to RESET and after one second to ON. If the generator will not reset, turn it off and rely on the other generator. Monitor the loadmeter so the load on the remaining generator does not exceed 100 percent. Turn off all non-essential electrical equipment as necessary.



Figure 2-23 L/R DC GEN Annunciators

BATTERY CHARGE ANNUNCIATOR ILLUMINATED

Normal battery charge rates (one to four amps with a fully charged battery) are below the threshold for

the BATTERY CHARGE annunciator (Figure 2-24) circuit monitor; therefore, in-flight illumination of the BATTERY CHARGE annunciator indicates a charge current above normal and a possible battery malfunction. If this occurs, the battery charge annunciator illuminated procedure per the *POH* should be completed.



Figure 2-24 Battery Charge Annunciator

On King Air 300s with serial numbers FA-2 through FA-9, place the battery switch to OFF if the battery charge annunciator illuminates in flight.

On King Air 300s with serial numbers of FA-10 and higher and all King Air 350s, a battery ammeter is installed which always indicates the charge/discharge status of the battery. Negative readings mean the battery is discharging. Battery usage should therefore be minimized due to the limited power capabilities of the battery. Positive readings indicate that the battery is accepting a charge. With the BATTERY CHARGE annunciator illuminated and a positive but decreasing indication on the battery ammeter, continue monitoring the ammeter until the BATTERY CHARGE annunciator extinguishes. However, when the annunciator is illuminated and the charge rate on the ammeter is steady or increasing, the battery must be isolated from the electrical system by placing the BAT switch to OFF.

NOTE

On King Air 300s with serials FA-2 through FA-9, if the BATTERY CHARGE annunciator is not extinguished with the BAT switch OFF, land as soon as practical. If this situation occurs on King Air 300s with serials FA-10 and later and all King Air 350s, monitor the battery ammeter. If it indicates a charge current, land as soon as practical; however,



safe flight procedures and techniques should not be abandoned to expedite this landing. On all aircraft, the GEN TIE switch may be placed in the open position to stop current flow from the generator buses to the center bus and ultimately to the battery, thereby preventing further damage to the battery if the battery bus tie doesn't open. In this condition the landing gear should be manually extended and the air conditioning system should be turned off prior to landing gear extension so that the condenser blower does not operate. On King Air 350s, the pilot may want to place the number 1 inverter switch to the transfer position so that it does not further deplete or damage the battery.

Refer to the Abnormal Procedures section of the *POH* for procedural details.

CIRCUIT BREAKER TRIPPED

Should one of the circuit breakers located in the cockpit trip: if it is a non-essential circuit-DO NOT RESET IN FLIGHT; if it is an essential circuit-PUSH TO RESET. If the circuit breaker trips again-DO NOT RESET. In this situation, a connected item may be inoperative; this can be determined from the Electrical System and Bus Feeders chart (Figure 2-5 and 2-6).

L OR R GEN TIE OPEN ANNUNCIATOR ILLUMINATED

Illumination of a L or R GEN TIE OPEN annunciator indicates a left or right generator tie is open (Figure 2-25). This signals a possible generator bus fault/malfunction. Monitor the corresponding loadmeter. If it is less than 100 percent and a normal indication, move the BUS SENSE switch to RESET. If it is greater than 100 percent or an abnormal indication, turn the appropriate generator OFF and monitor the opposite loadmeter not to exceed 100 percent (Figures 2-38 and 2-54). If the generator bus tie relay will not reset, monitor the loadmeters.

In this situation, the generator paralleling circuit is open; therefore, the generators will not share the aircraft electrical loads equally and the loadmeters need not be within 10% of each other. The loadmeters must be monitored to ensure that neither exceeds 100% total load. Refer to the *POH* for procedural details.

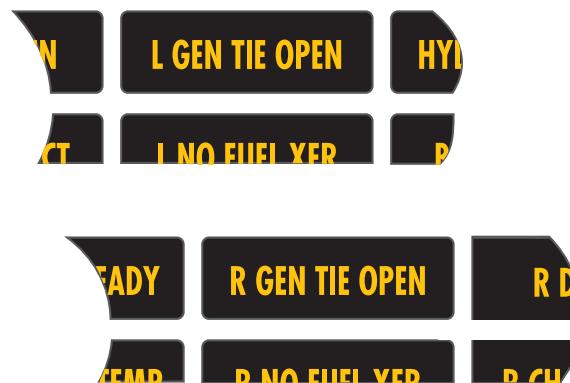


Figure 2-25 L/R GEN TIE OPEN

BAT TIE OPEN ANNUNCIATOR ILLUMINATED

If the BAT TIE OPEN annunciator is illuminated, the battery bus tie relay is open (Figure 2-26). This indicates a possible center bus fault/malfunction. Check the center bus for proper voltage indication.

If it is within normal limits (24 to 28 volts), attempt to reset the bus tie by momentarily actuating the BUS SENSE switch to RESET. If this is successful, a transient spike in the electrical system tripped the Hall Effect sensor and opened the battery tie relay. If this procedure was unsuccessful, there is a probable malfunction within the battery bus tie circuitry which cannot be reset at this time.

If center bus voltage is zero, there is a possible center bus fault (Figures 2-39 and 2-55). Open the generator bus tie relays by moving the GEN TIES switch to OPEN. Check to ensure that both



GEN TIE OPEN annunciators illuminate. Pull the LANDING GEAR RELAY circuit breaker on the pilot's right subpanel.

WARNING

If electrical power is applied to the landing gear hydraulic pump motor relay with the center bus shorted, damage may occur to the electrical system.

NOTE

It will not be possible to charge the battery and the landing gear will have to be manually extended.

With the center bus unpowered, the air conditioning system should be turned off prior to landing.

On King Air 350s, the number 1 inverter switch should be placed to the bus transfer position. This will assure that the number 1 AC buses will remain powered.

Refer to the *POH* for procedural details.



Figure 2-26 BAT TIE OPEN

NI-CAD BATTERY CONDITION CHECK (FOR GROUND OPERATION ONLY)

Following an engine start, the battery recharge current is very high and causes illumination of the BATTERY CHARGE annunciator. It should normally extinguish within five minutes. If it does

not, or if it reappears, the battery charge current should be monitored until it decreases to a level which will extinguish the BATTERY CHARGE annunciator. Use the battery ammeter on King Air 300s after FA-9 and all 350s or the following loadmeter procedure on 300s FA-2 through FA-10 to determine the battery charge current.

1. One Generator-OFF
2. Voltmeter Bus Select Switch-BAT-Read 28 volts
3. Momentarily turn the battery switch OFF, noting the change in loadmeter indication.

NOTE

The change in loadmeter indication is the battery charge current and should be no more than 2.5 percent (barely perceptible needle movement). If the results are unsatisfactory, repeat the check until the charge current decreases to less than 2.5 percent.

Before engine shutdown on the ground, the battery condition can also be checked. If the charge value is not less than 2.5 percent within five minutes, have it checked by a qualified nickel-cadmium battery shop.

On King Air 300s with serial numbers FA-10 and after and all 350s, observe the battery ammeter to see if the aircraft is accepting a charge of 7 amps or greater. Normally the charge rate on the battery ammeter should decrease below 7 amps within five minutes after engine start. If the ammeter is not reading 7 amps, continue to monitor making sure that the reading is decreasing to below 7 amps. If the reading holds steady or increases, turn the battery OFF and have it checked by a qualified nickel-cadmium battery shop.

The battery ammeter can also simplify the Battery Condition Check during shutdown. If the charge rate is not below 7 amps after five minutes, have the battery checked by a qualified nickel-cadmium battery shop.

Refer to the *POH* for procedure details.



POWER DISTRIBUTION SCHEMATICS

KING AIR 300 POWER DISTRIBUTION SCHEMATICS

The following 15 figures are the King Air 300 Power Distribution Schematics for various conditions:

- Figure 2-27—King Air 300 Power Distribution Schematic
- Figure 2-28—King Air 300 Power Distribution Battery OFF
- Figure 2-29—King Air 300 Power Distribution—Battery ON
- Figure 2-30—King Air 300 Power Distribution—Battery ON (Generator Ties Manually Closed)
- Figure 2-31—King Air 300 Power Distribution—Right Engine Start (Generator Ties Manually Closed)
- Figure 2-32—King Air 300 Power Distribution—Right Generator ON
- Figure 2-33—King Air 300 Power Distribution—Cross-Generator-Start (Left Engine)
- Figure 2-34—King Air 300 Power Distribution—Both Generators ON
- Figure 2-35—King Air 300 Power Distribution—Both Generators ON (Generator Ties Open)
- Figure 2-36—King Air 300 Power Distribution—Both Generators ON (Bus Sense Test)
- Figure 2-37—King Air 300 Power Distribution—Both Generators Failed (Load-Shedding)
- Figure 2-38—King Air 300 Power Distribution—Bus Isolation (Left Generator Bus Short)
- Figure 2-39—King Air 300 Power Distribution—Bus Isolation (Center Bus Short)
- Figure 2-40—King Air 300 Power Distribution—Bus Isolation (Triple-Fed Bus Short)
- Figure 2-41—King Air 300 Power Distribution—External Power

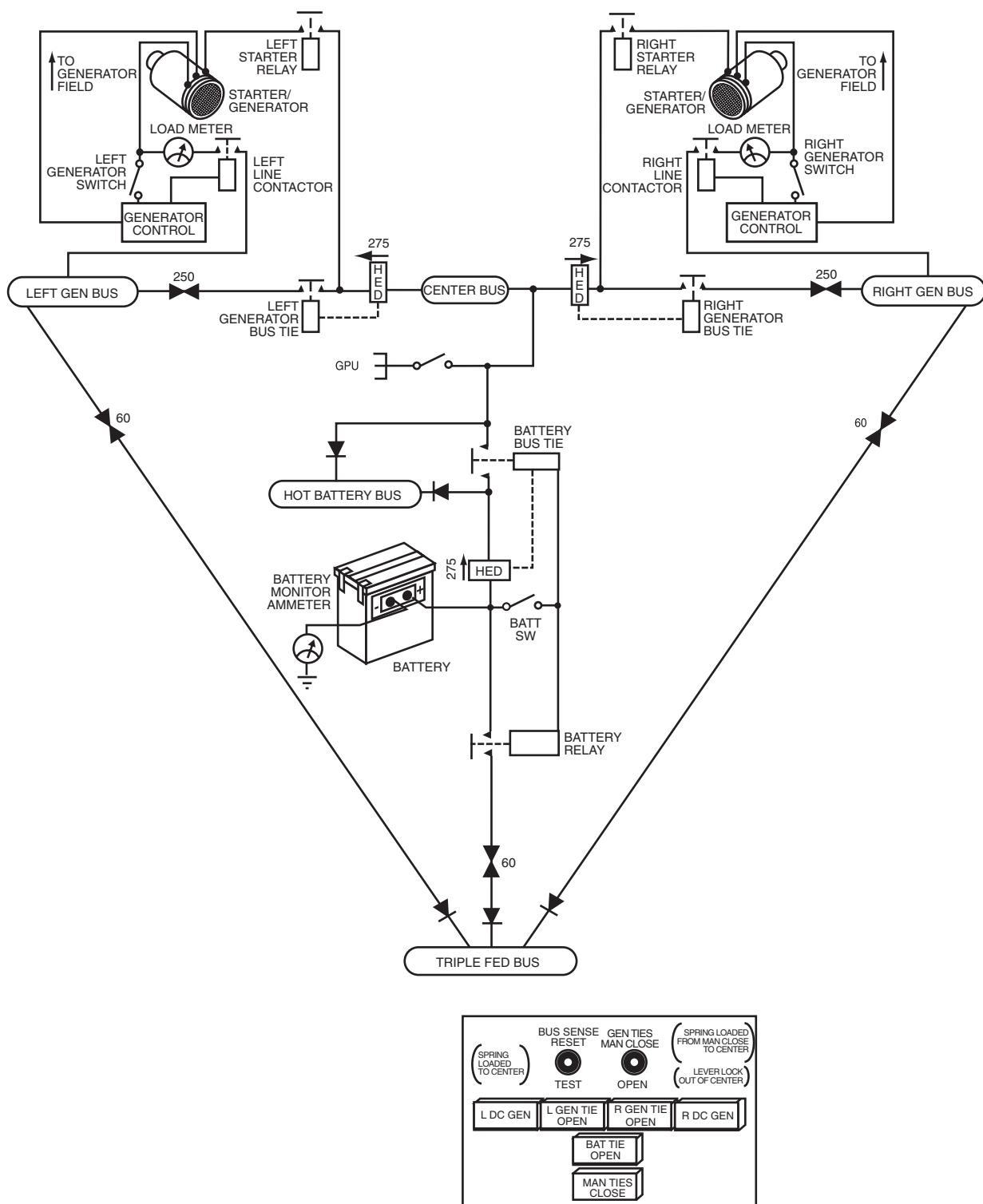
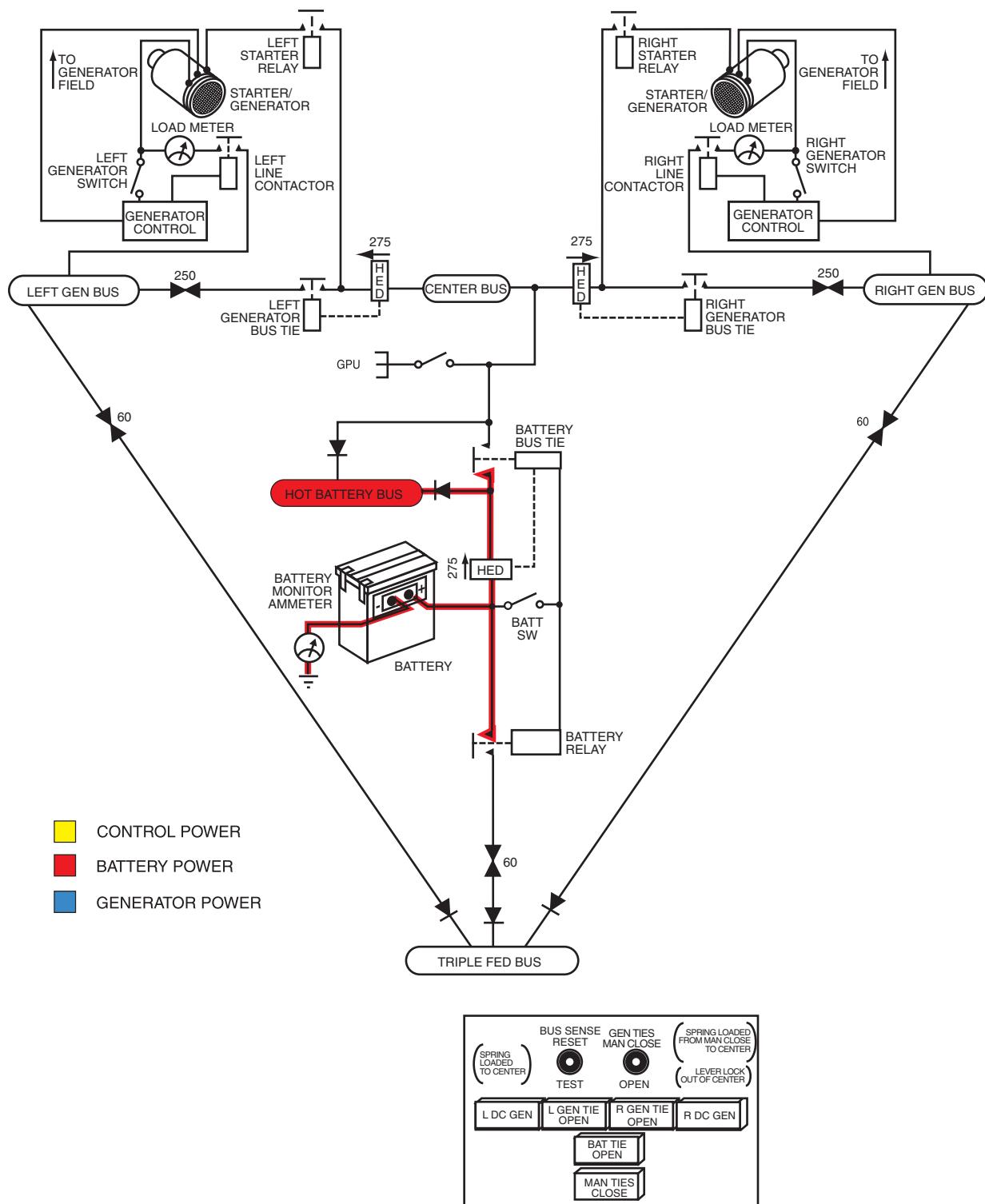


Figure 2-27 King Air 300 Power Distribution Schematic


Figure 2-28 King Air 300 Power Distribution—Battery OFF

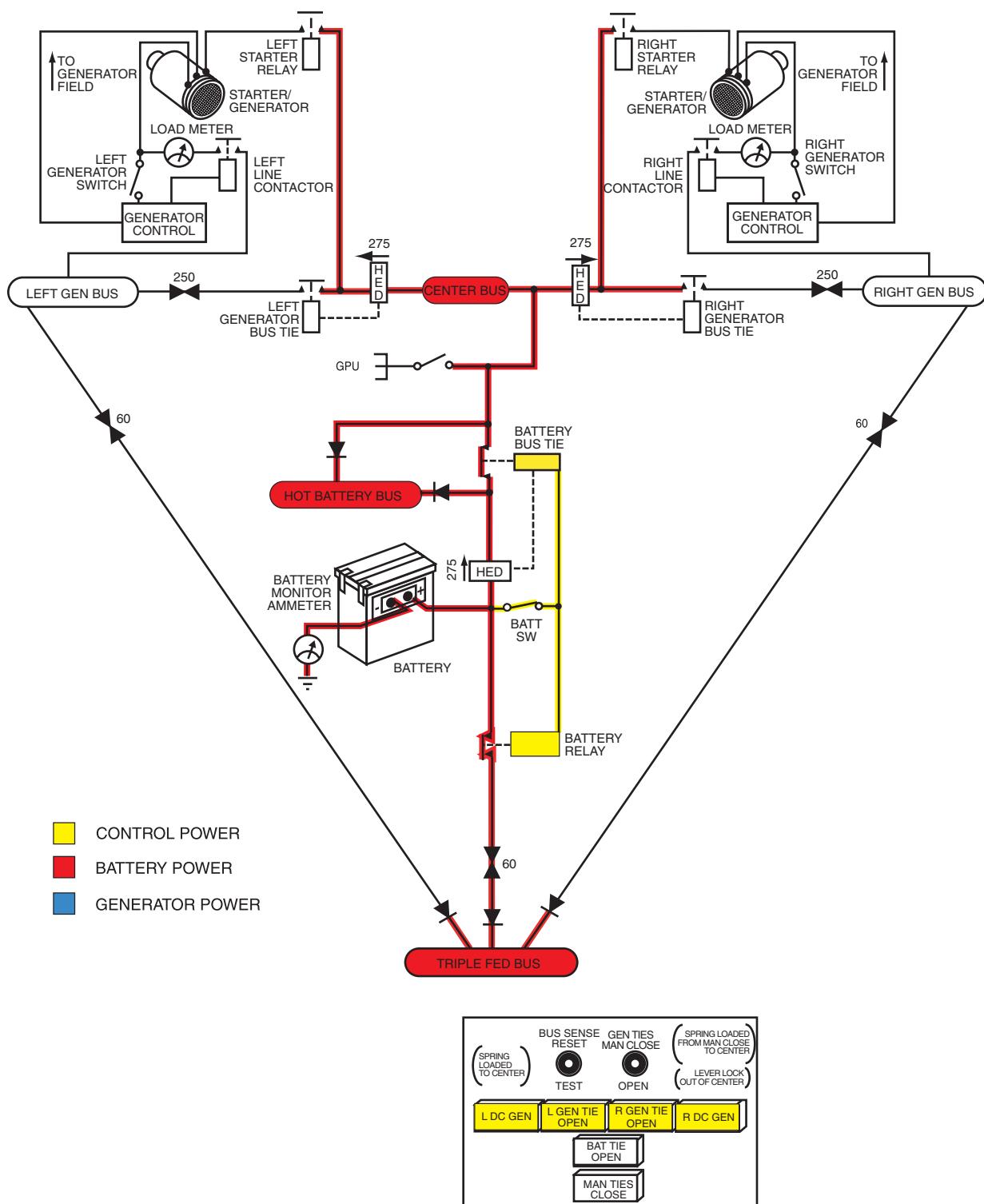


Figure 2-29 King Air 300 Power Distribution—Battery ON

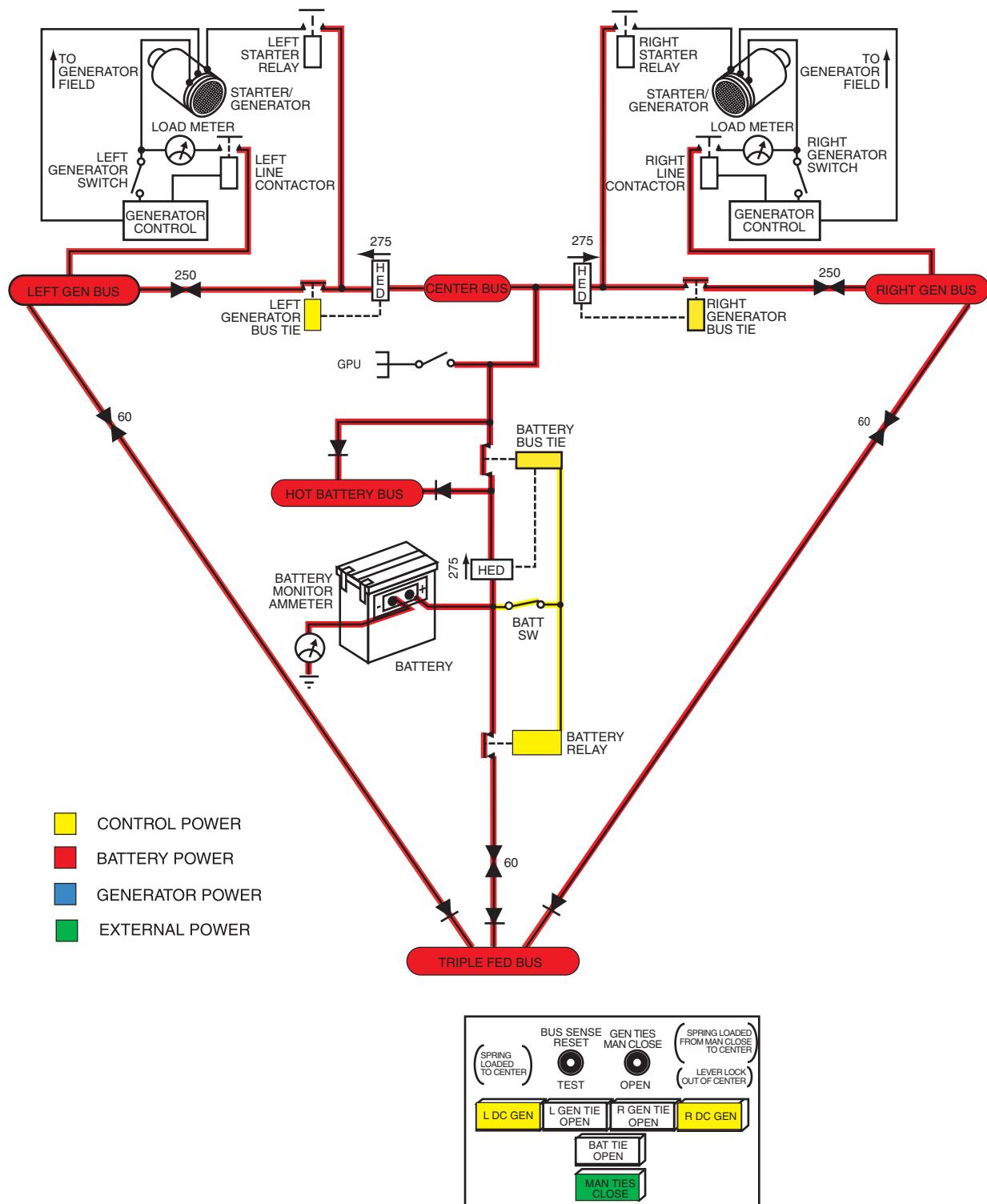


Figure 2-30 King Air 300 Power Distribution—Battery ON (Generator Ties Manually Closed)

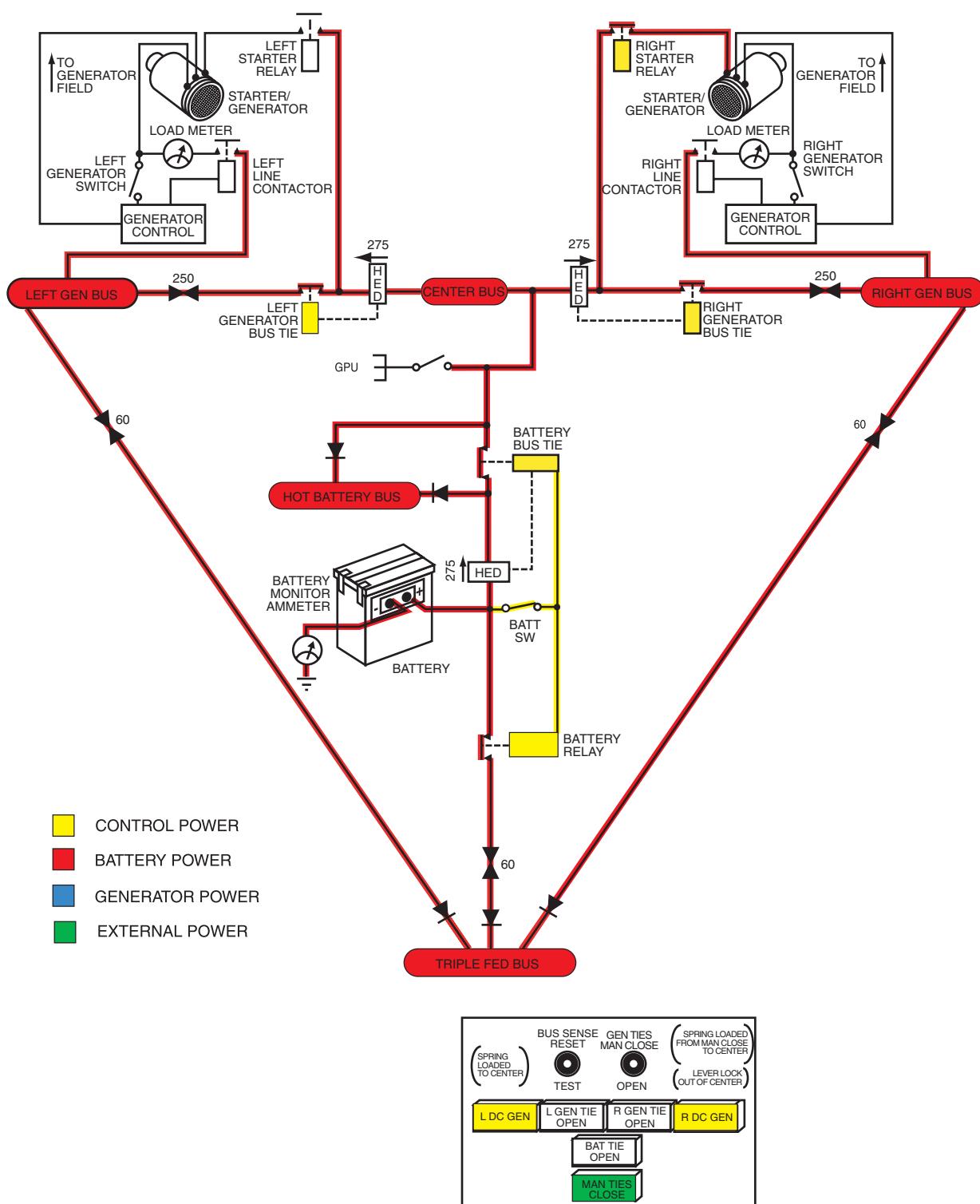
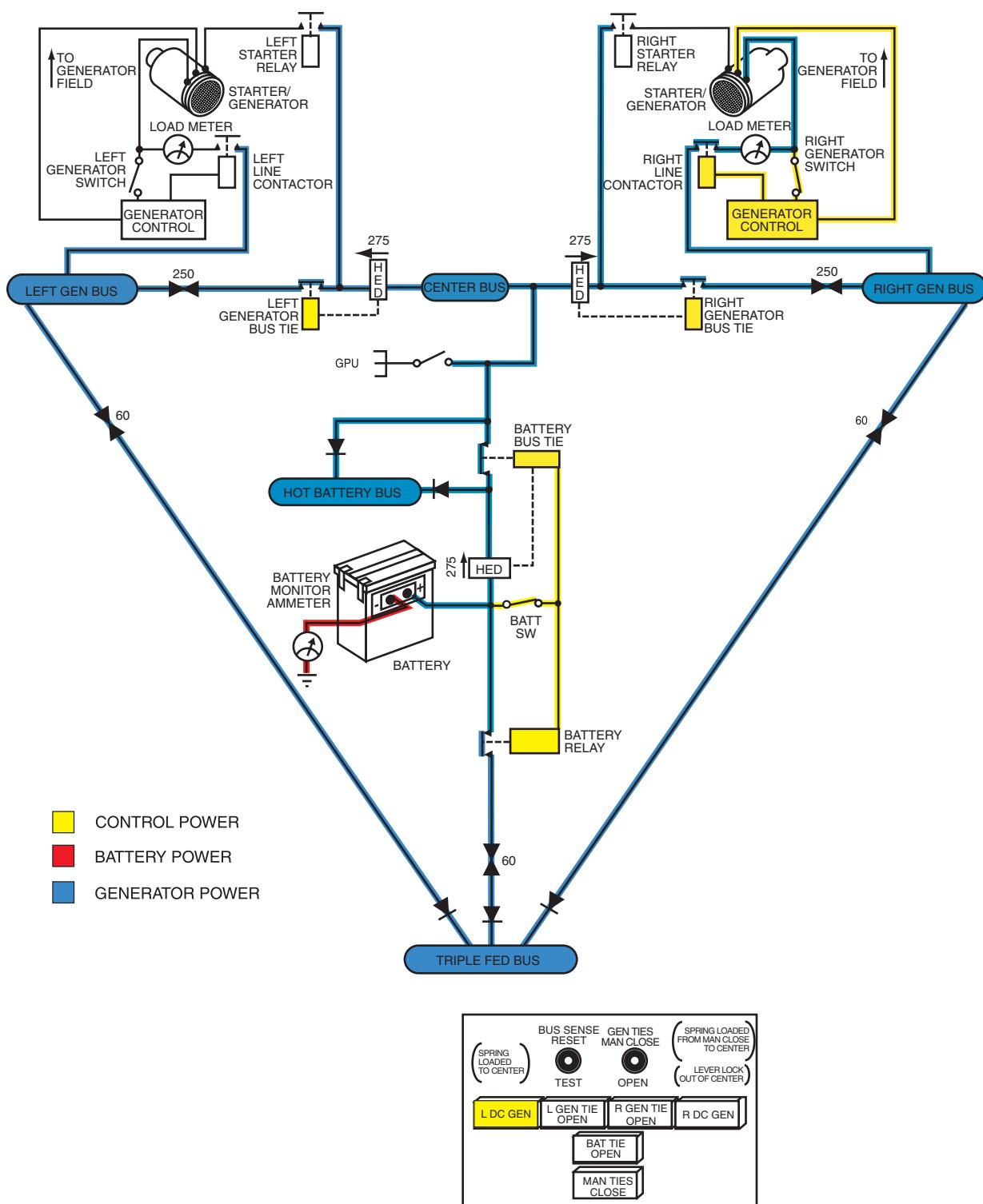


Figure 2-31 King Air 300 Power Distribution—Right Engine Start (Generator Ties Manually Closed)


Figure 2-32 King Air 300 Power Distribution—Right Generator ON

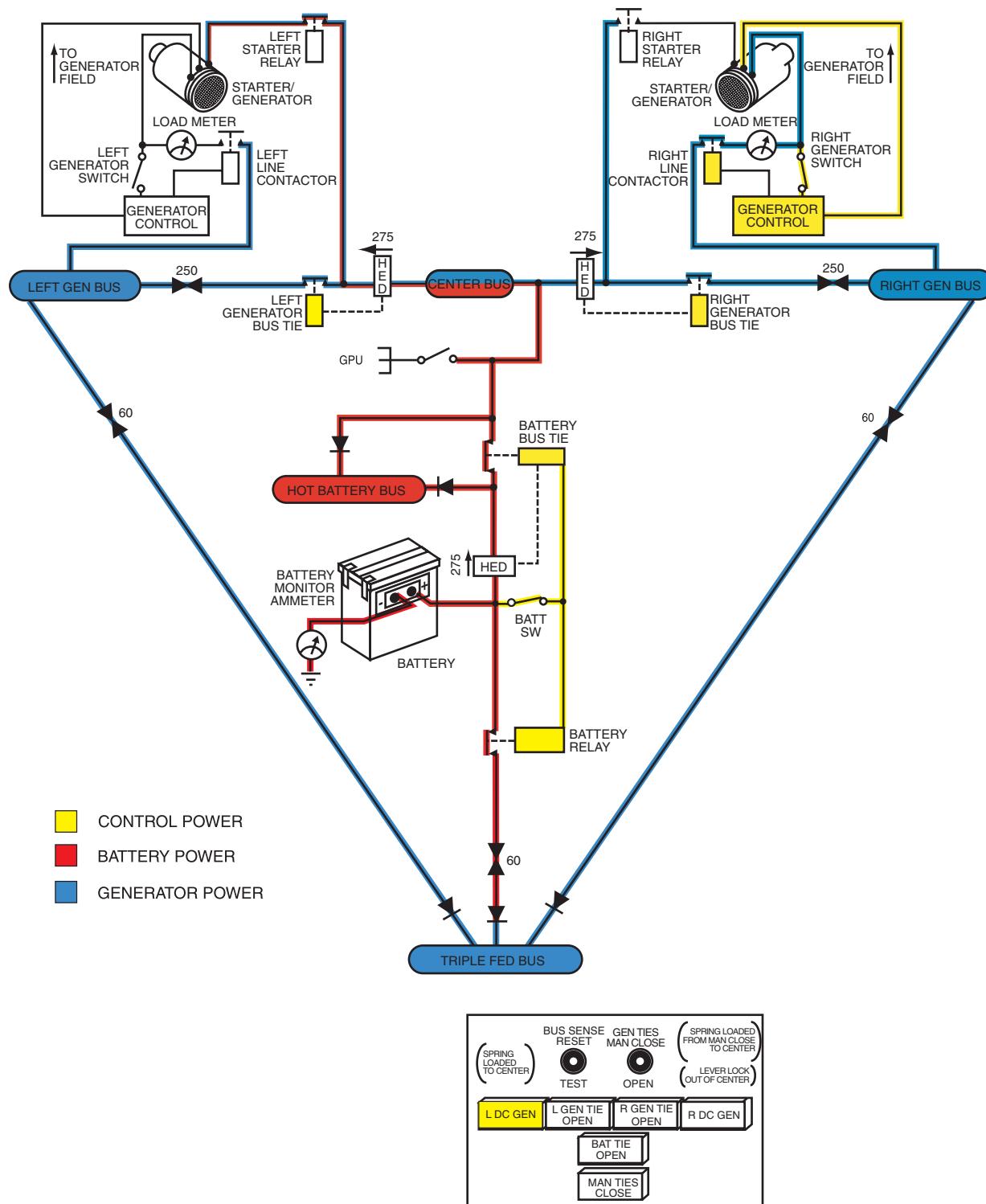
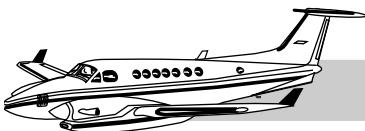


Figure 2-33 King Air 300 Power Distribution—Cross-Generator-Start (Left Engine)

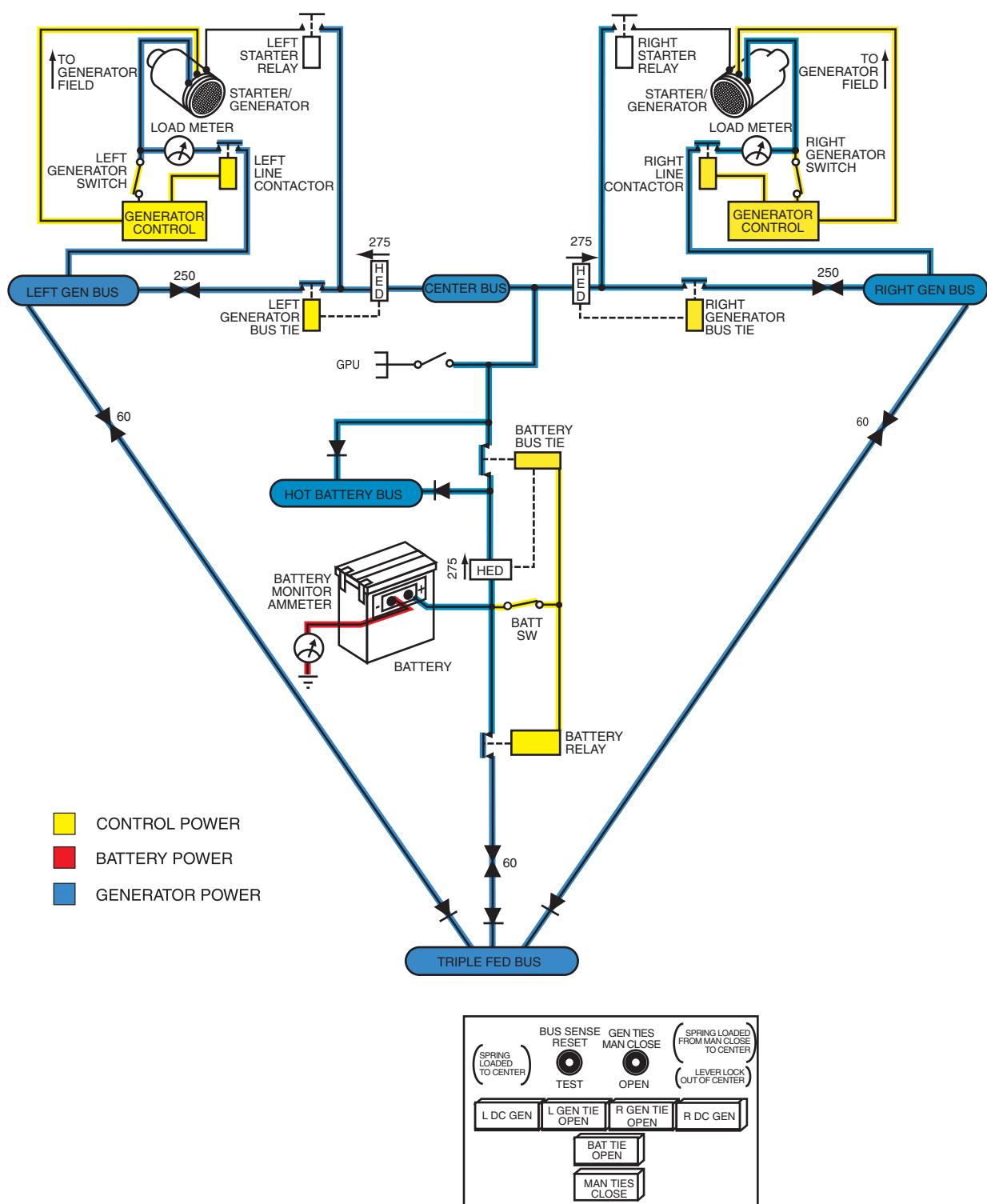


Figure 2-34 King Air 300 Power Distribution—Both Generators ON

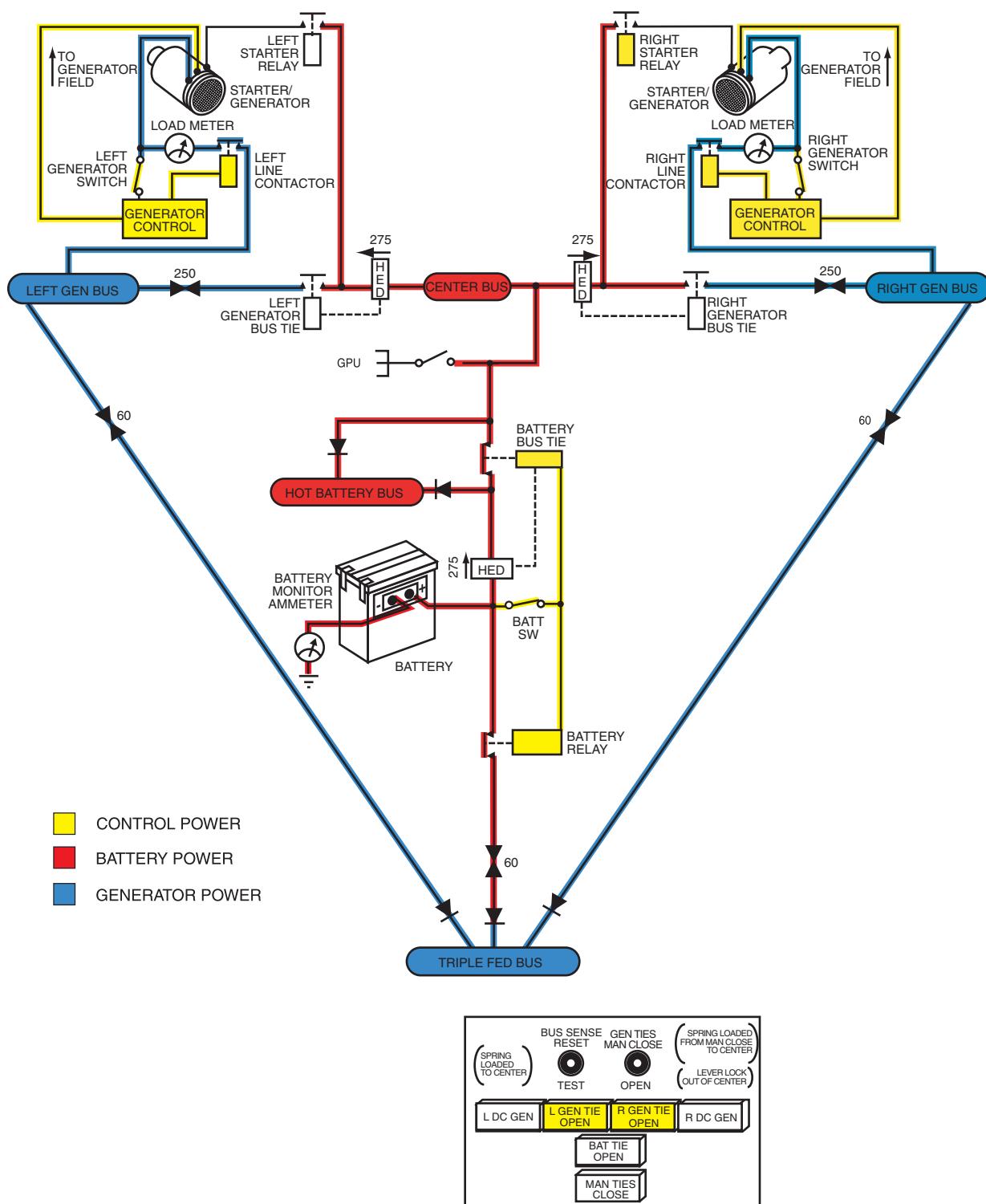


Figure 2-35 King Air 300 Power Distribution—Both Generators ON (Generator Ties Open)

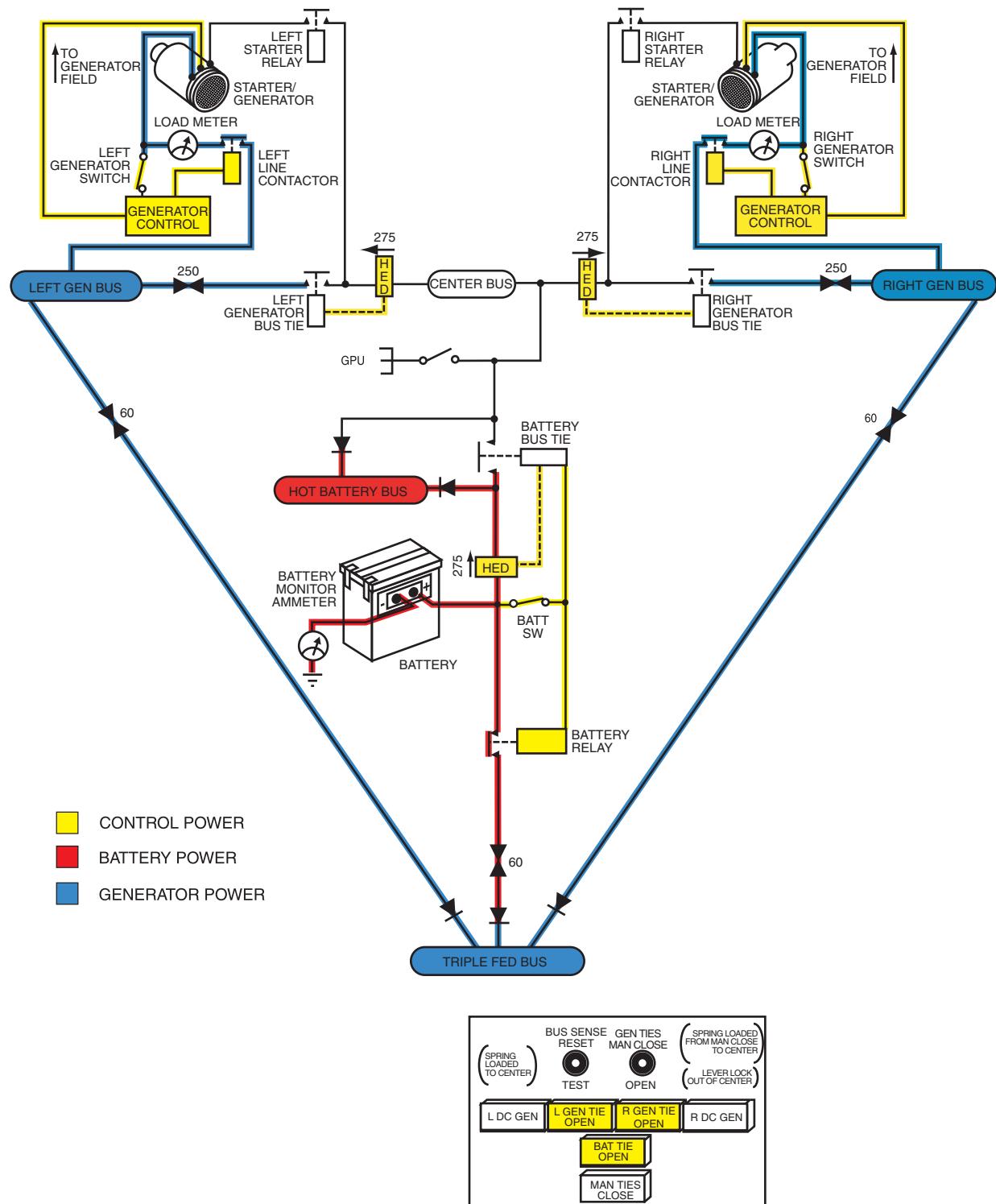


Figure 2-36 King Air 300 Power Distribution—Both Generators ON (Bus Sense Test)

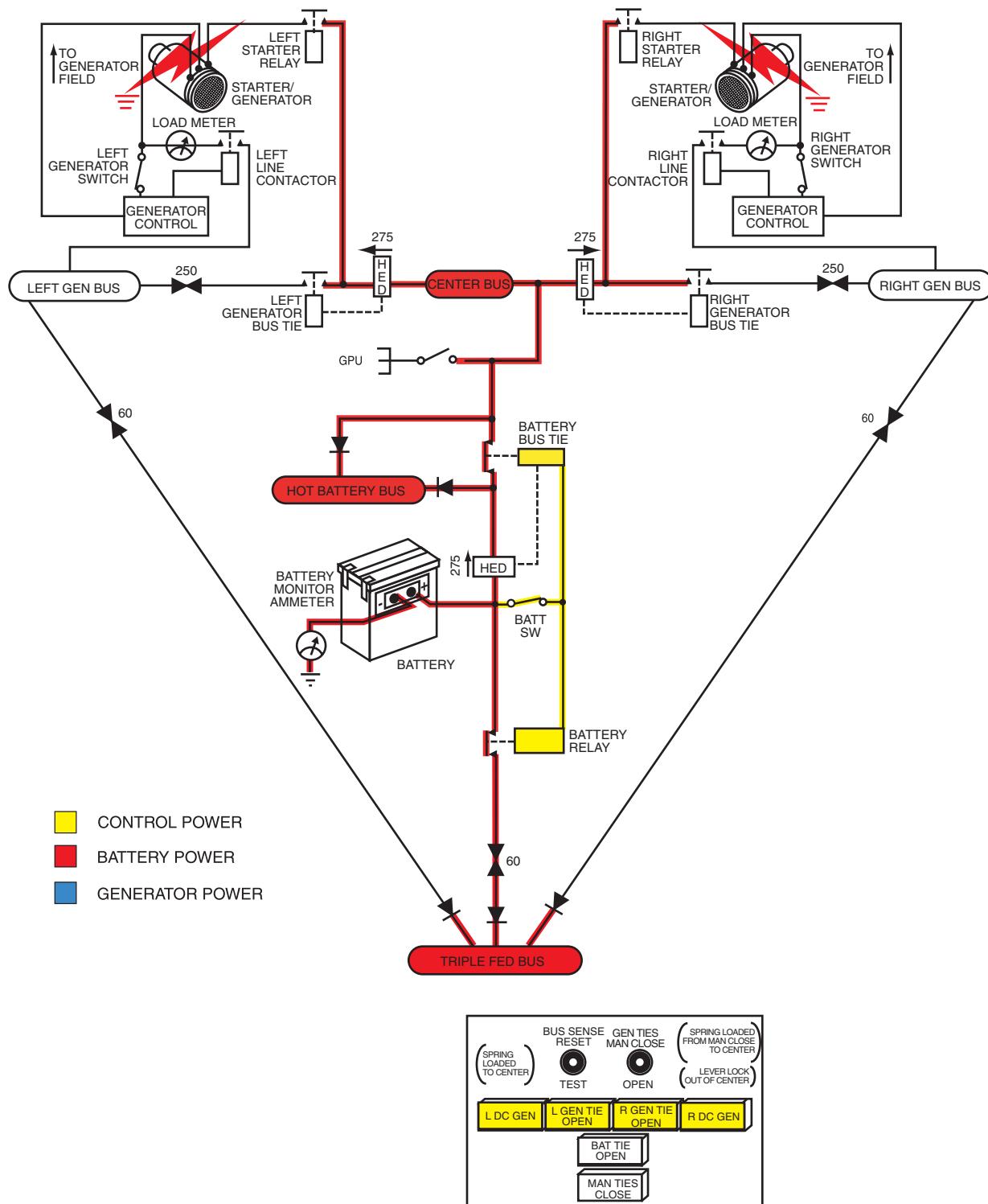
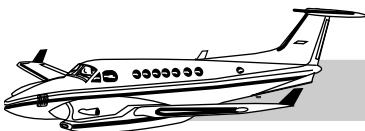
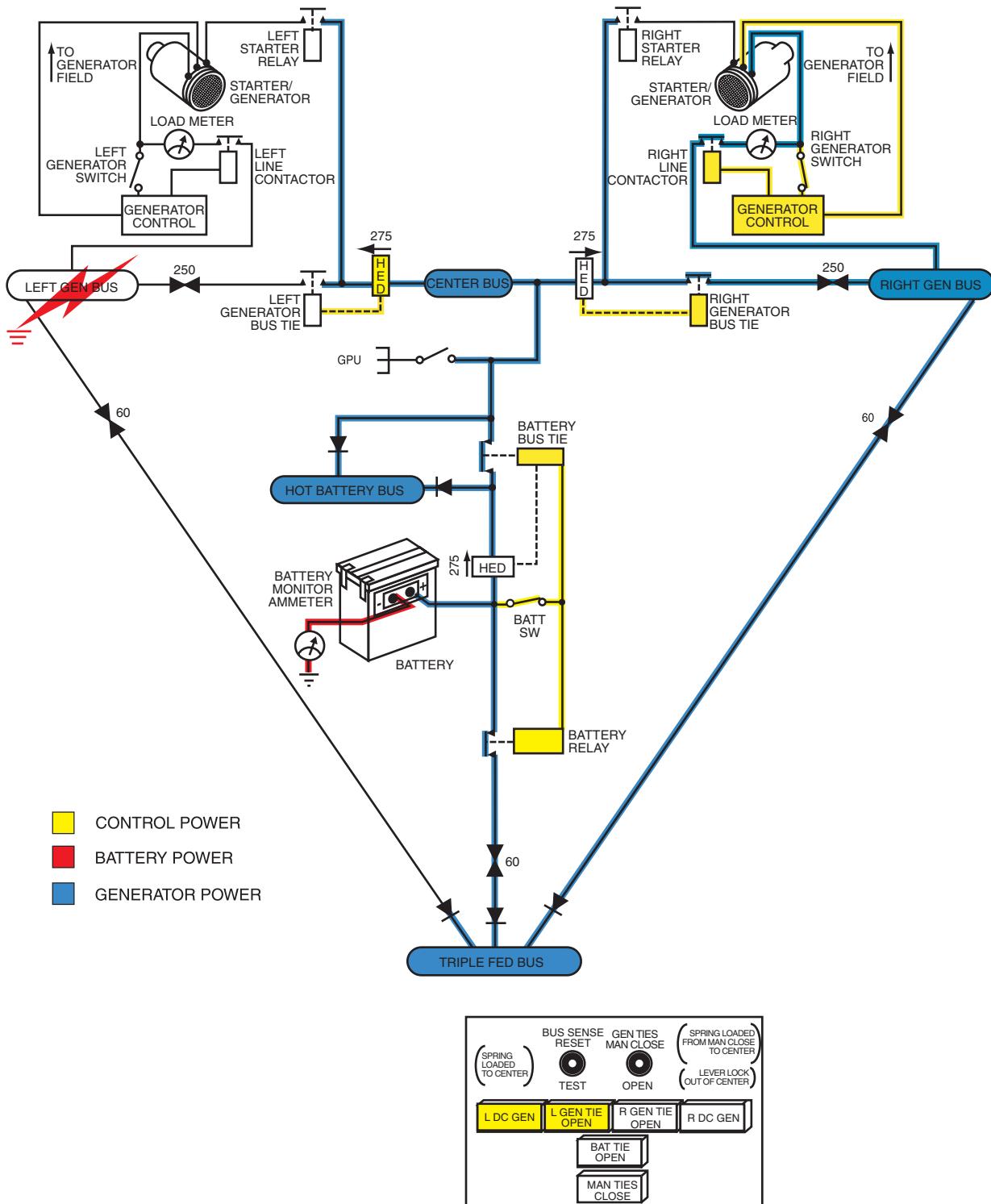


Figure 2-37 King Air 300 Power Distribution—Both Generators Failed (Load-Shedding)


Figure 2-38 King Air 300 Power Distribution—Bus Isolation (Left Generator Bus Short)

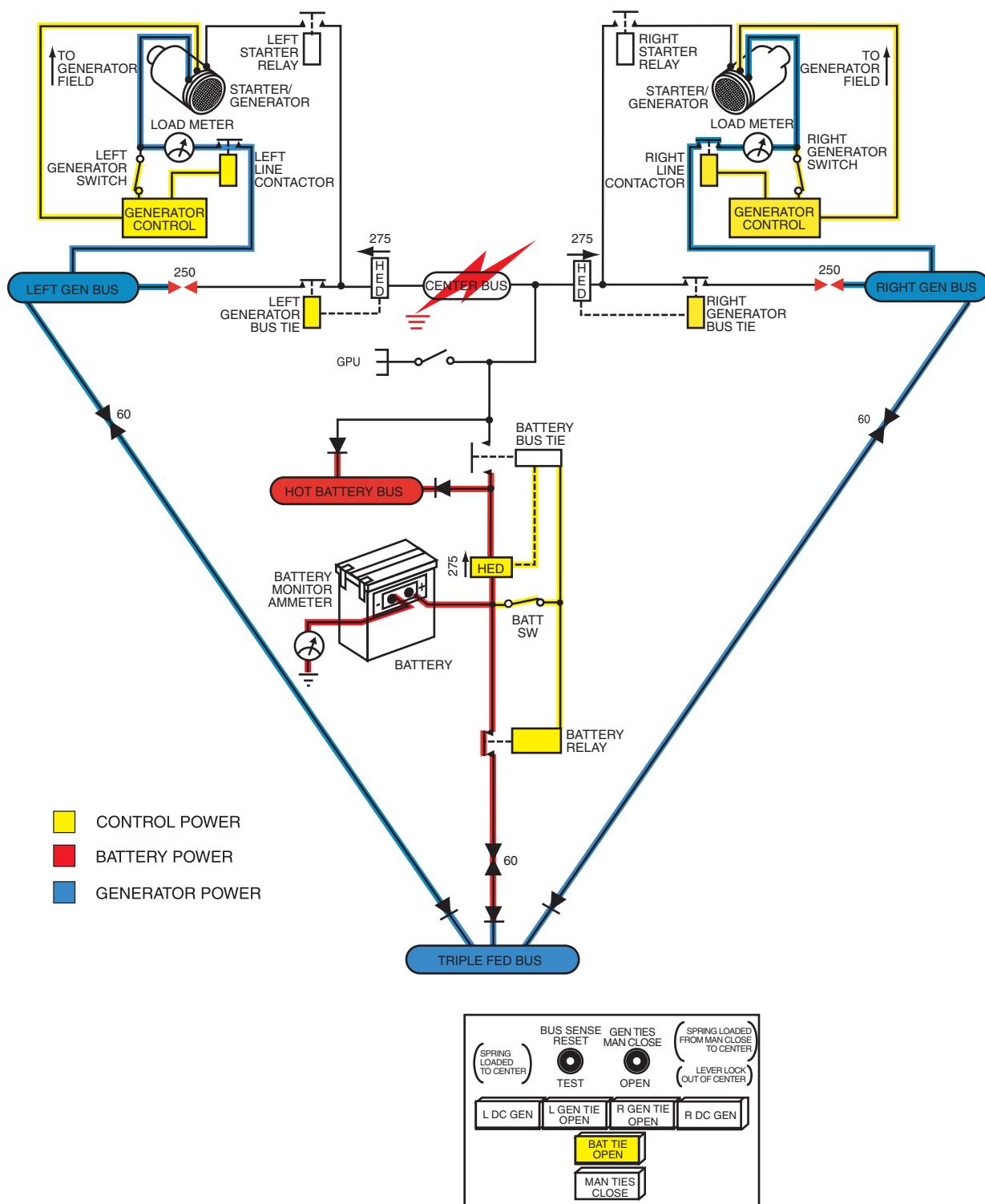


Figure 2-39 King Air 300 Power Distribution—Bus Isolation (Center Bus Short)

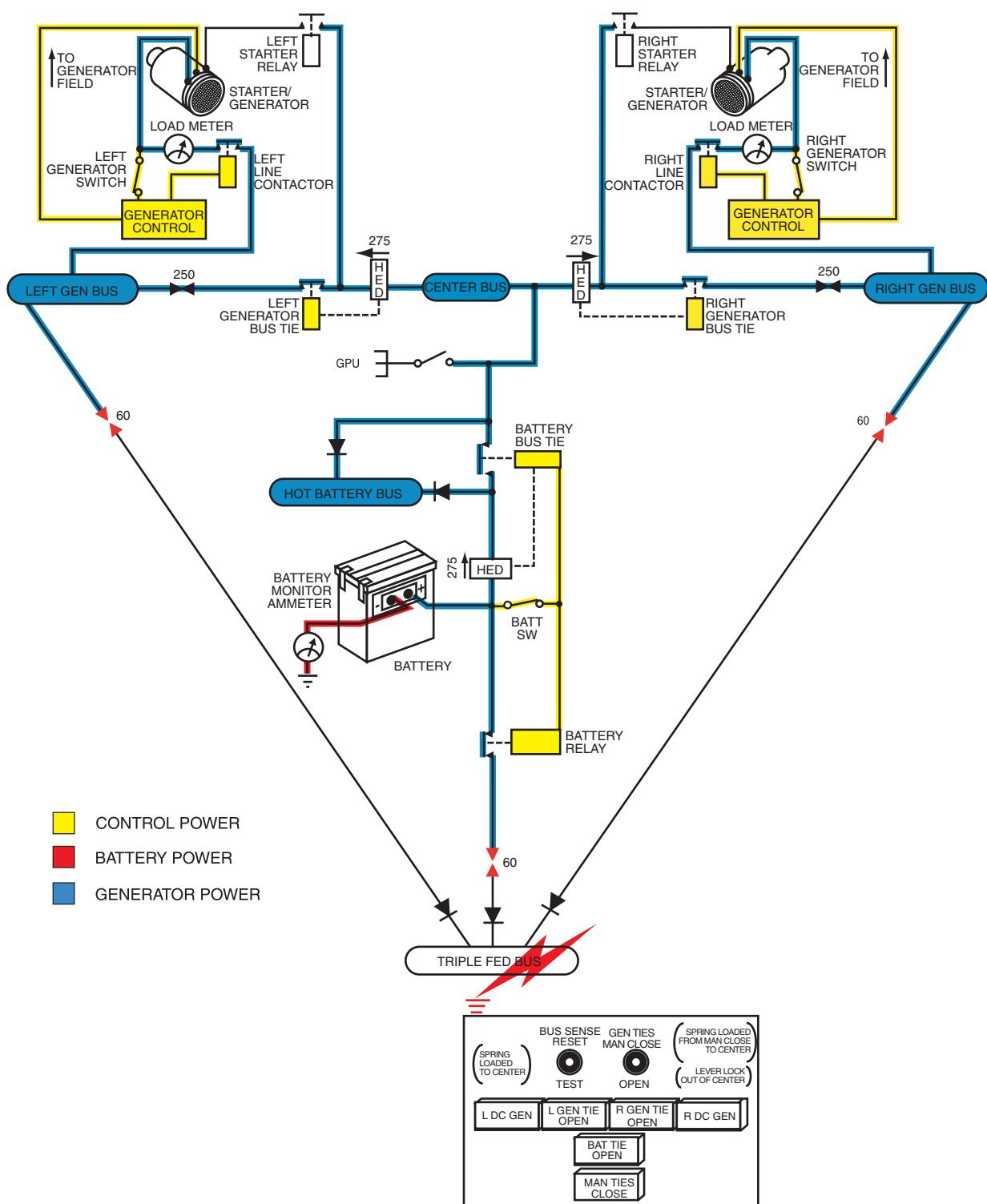


Figure 2-40 King Air 300 Power Distribution—Bus Isolation (Triple-Fed Bus Short)

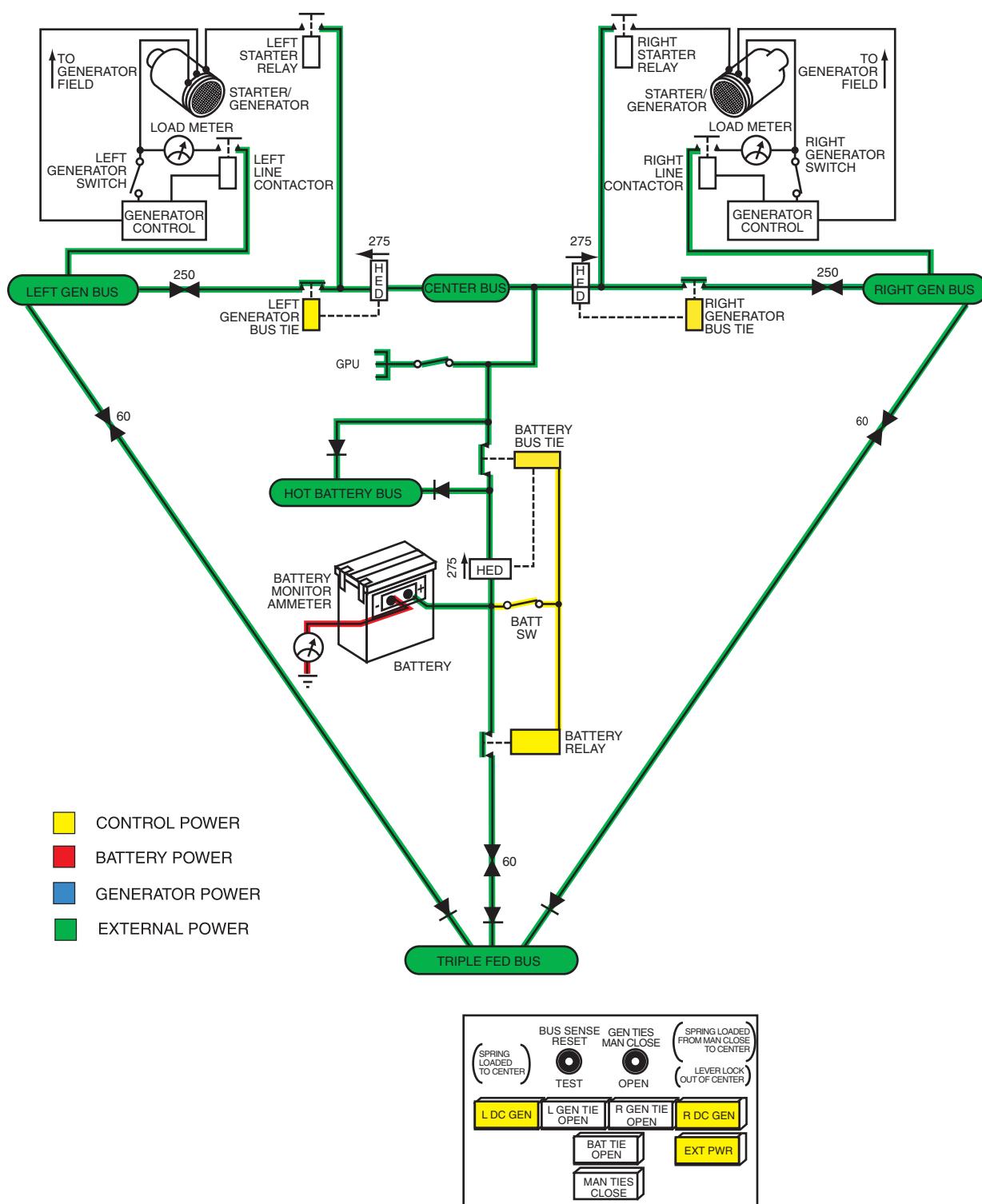
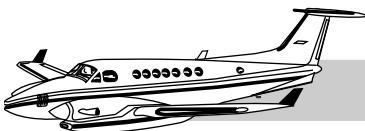


Figure 2-41 King Air 300 Power Distribution—External Power



KING AIR 350 POWER DISTRIBUTION SCHEMATICS

The following 16 figures are the King Air 350 Power Distribution Schematics for various conditions:

- Figure 2-42—King Air 350 Power Distribution Schematic
- Figure 2-43—King Air 350 Power Distribution—Battery Switch OFF, Battery Bus Switch Emergency OFF
- Figure 2-44—King Air 350 Power Distribution—Battery Switch OFF, Battery Bus Switch Normal
- Figure 2-45—King Air 350 Power Distribution—Battery Switch ON
- Figure 2-46—King Air 350 Power Distribution—Battery Switch ON (Generator Ties Manually Closed)
- Figure 2-47—King Air 350 Power Distribution—Right Engine Start
- Figure 2-48—King Air 350 Power Distribution—Right Generator ON
- Figure 2-49—King Air 350 Power Distribution—Cross-Generator Start (Left Engine)
- Figure 2-50—King Air 350 Power Distribution—Both Generators ON
- Figure 2-51—King Air 350 Power Distribution—Both Generators ON (Generator Ties Open)
- Figure 2-52—King Air 350 Power Distribution—Both Generators On (Bus Sense Test)
- Figure 2-53—King Air 350 Power Distribution—Both Generators Failed (Load-Shedding)
- Figure 2-54—King Air 350 Power Distribution—Bus Isolation (Left Generator Bus Short)
- Figure 2-55—King Air 350 Power Distribution—Bus Isolation (Center Bus Short)
- Figure 2-56—King Air 350 Power Distribution—Bus Isolation (Triple-Fed Bus Short)
- Figure 2-57—King Air 350 Power Distribution—External Power

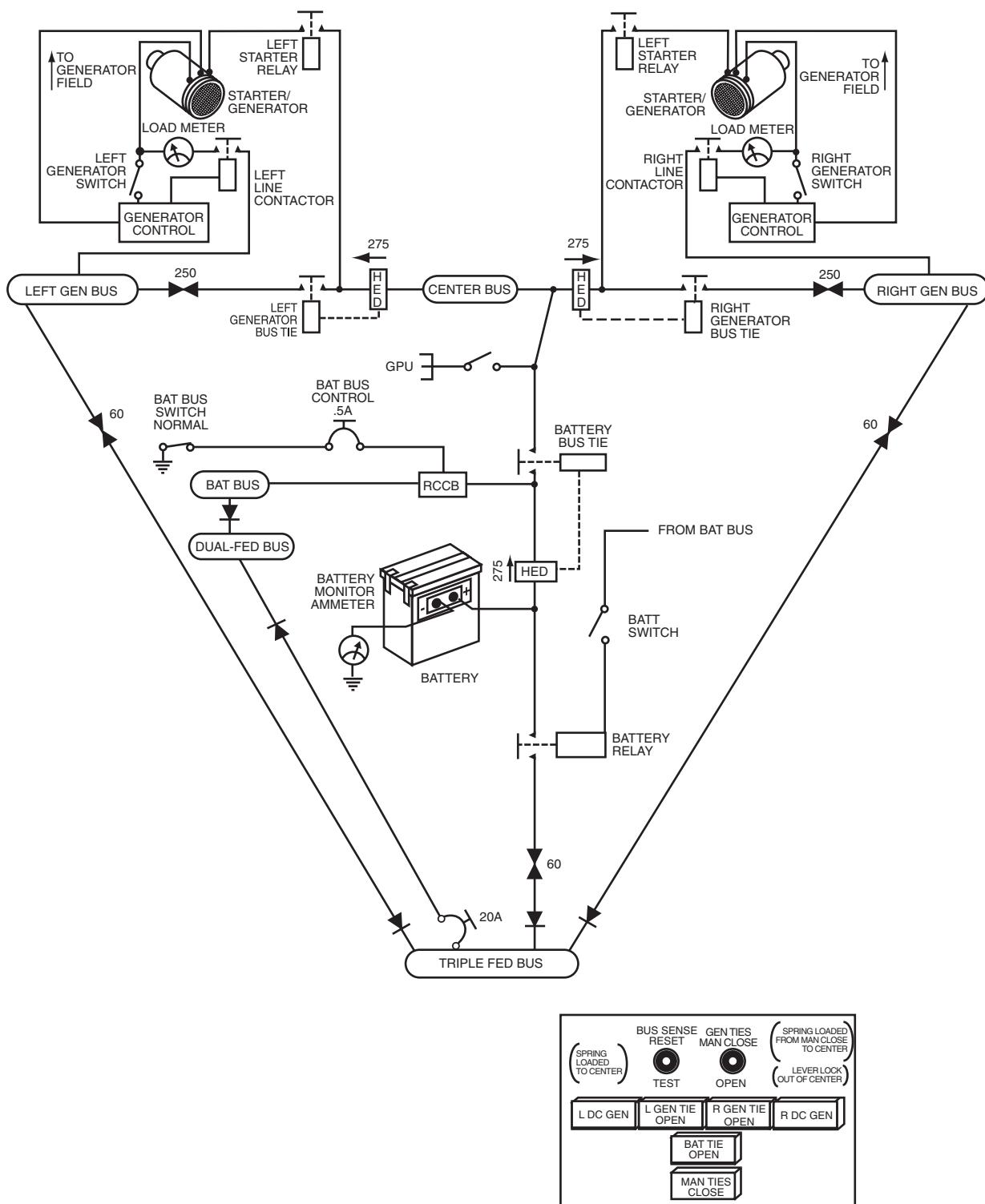


Figure 2-42 King Air 350 Power Distribution Schematic

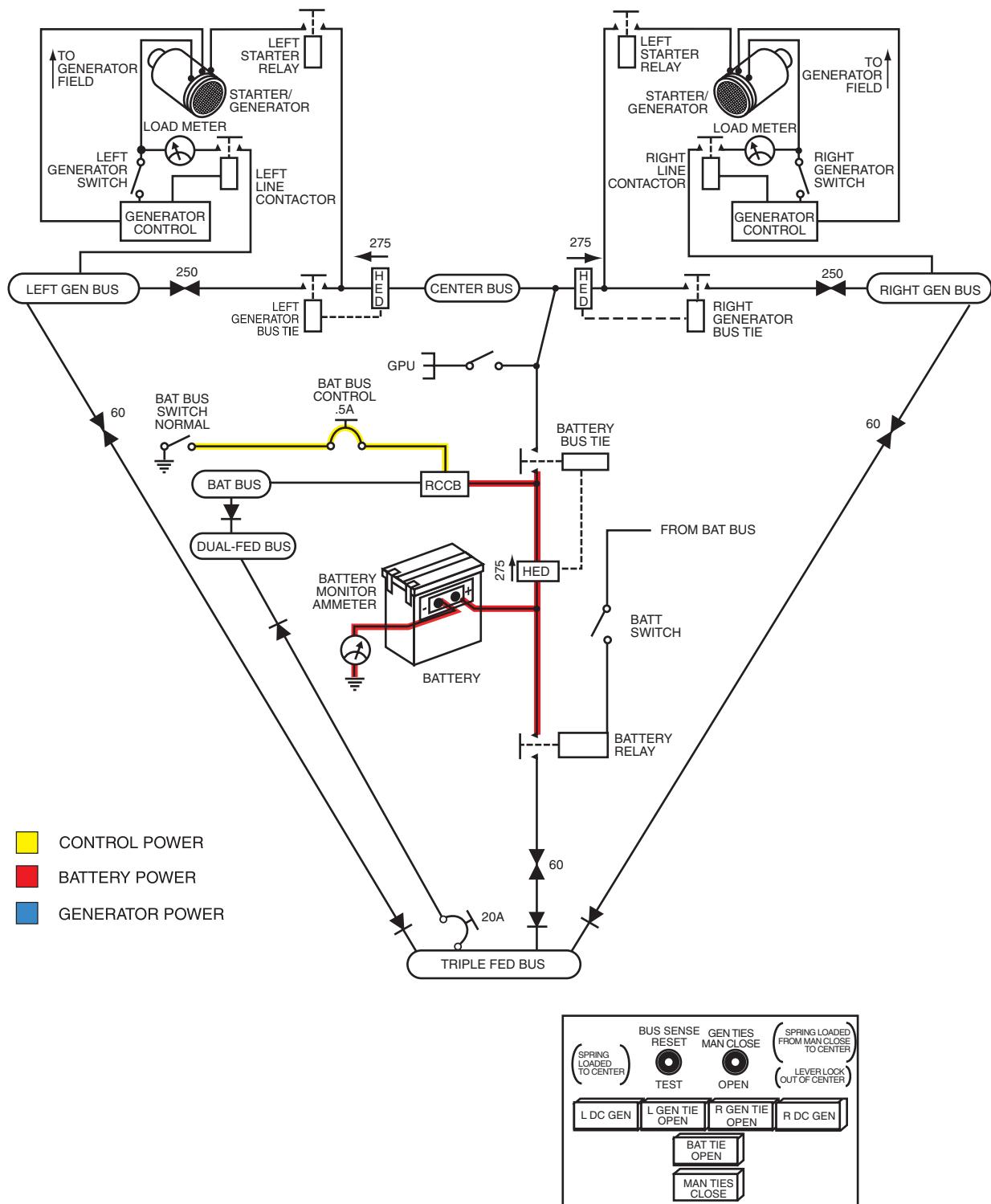


Figure 2-43 King Air 350 Power Distribution—Battery Switch OFF, Battery Bus Switch Emergency OFF

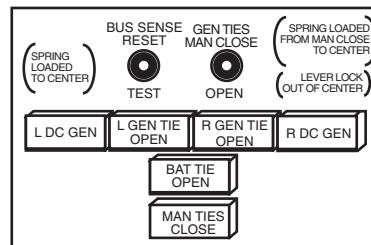
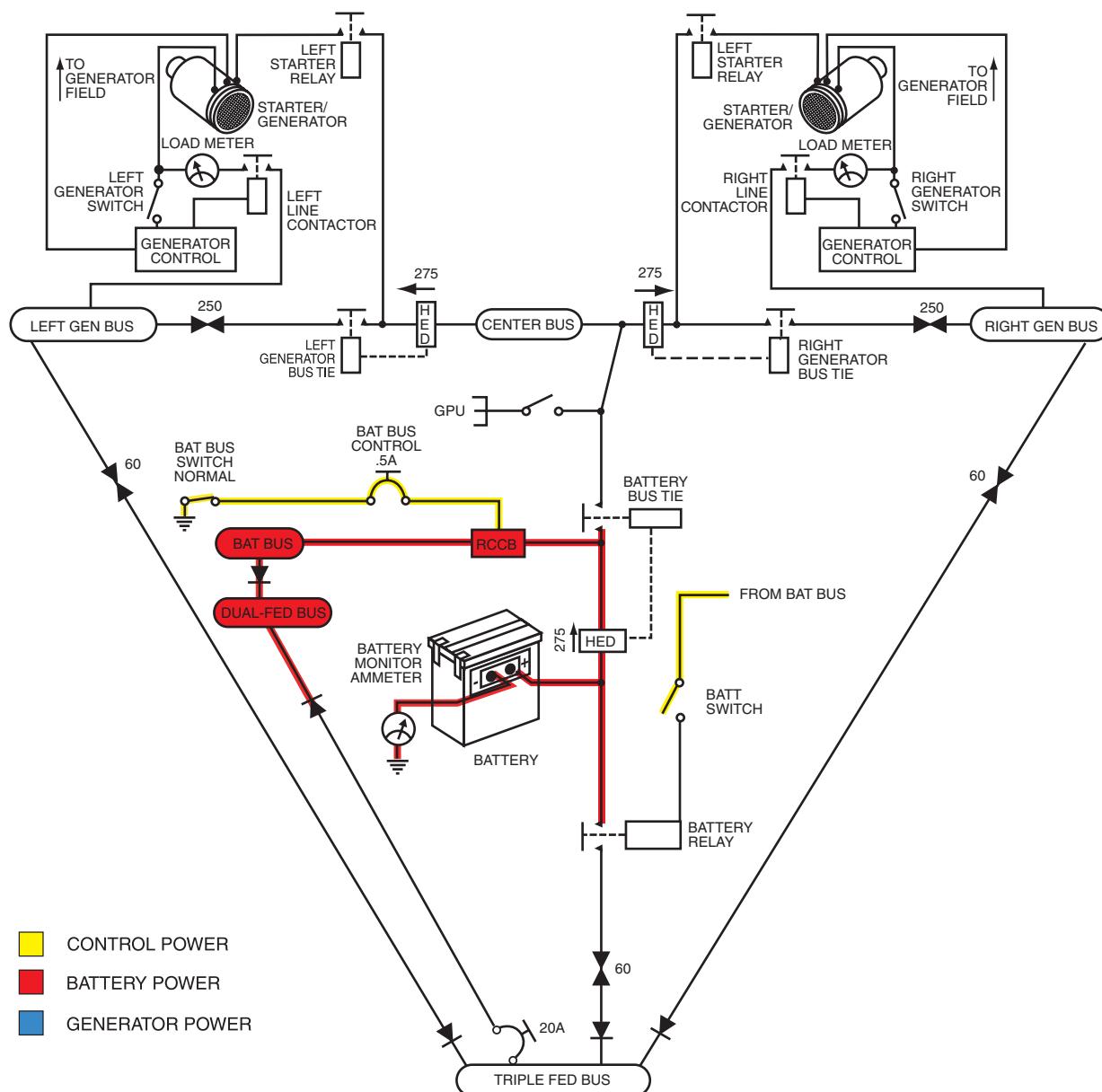


Figure 2-44 King Air 350 Power Distribution—Battery Switch OFF, Battery Bus Switch Normal

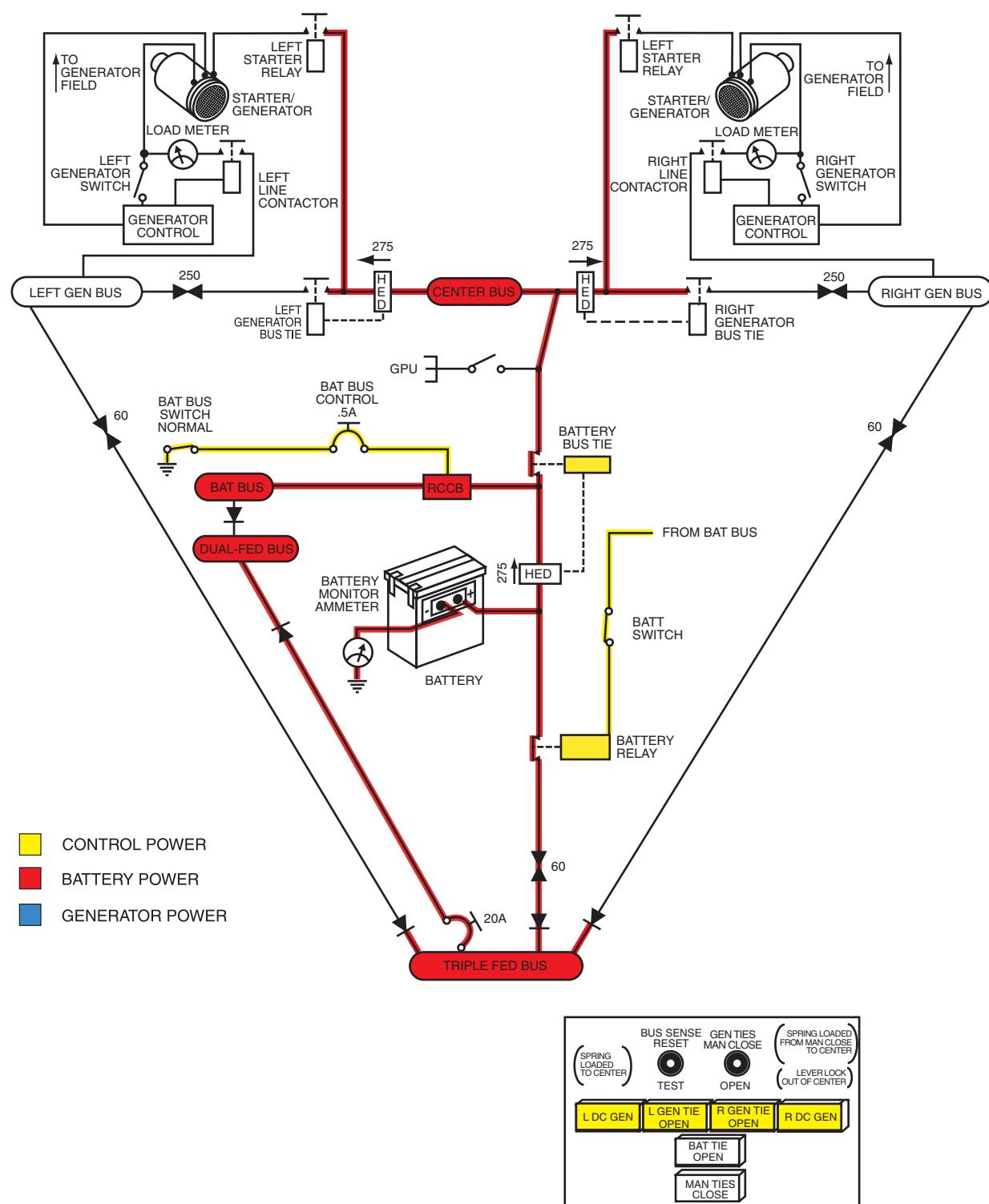


Figure 2-45 King Air 350 Power Distribution—Battery Switch ON

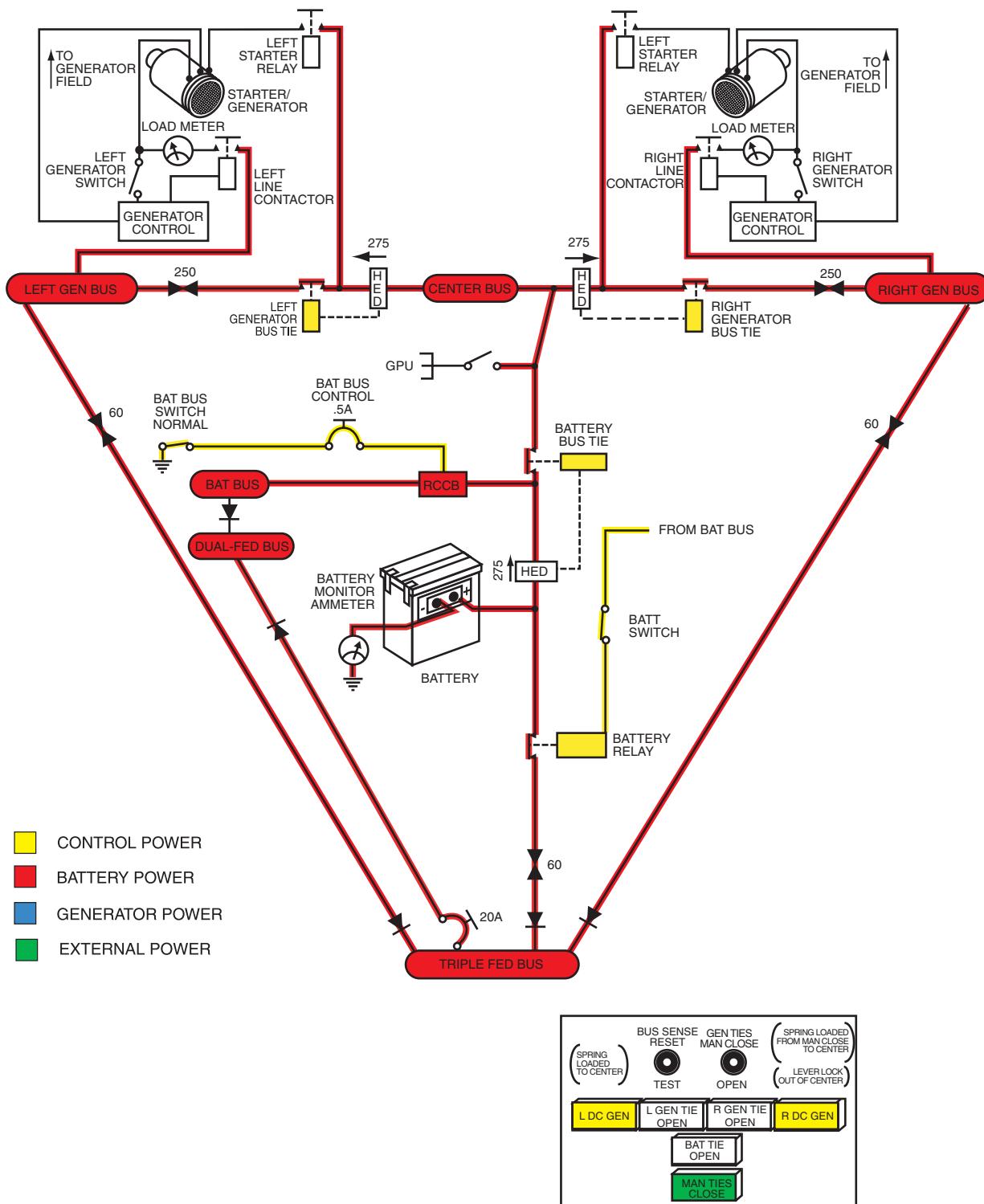
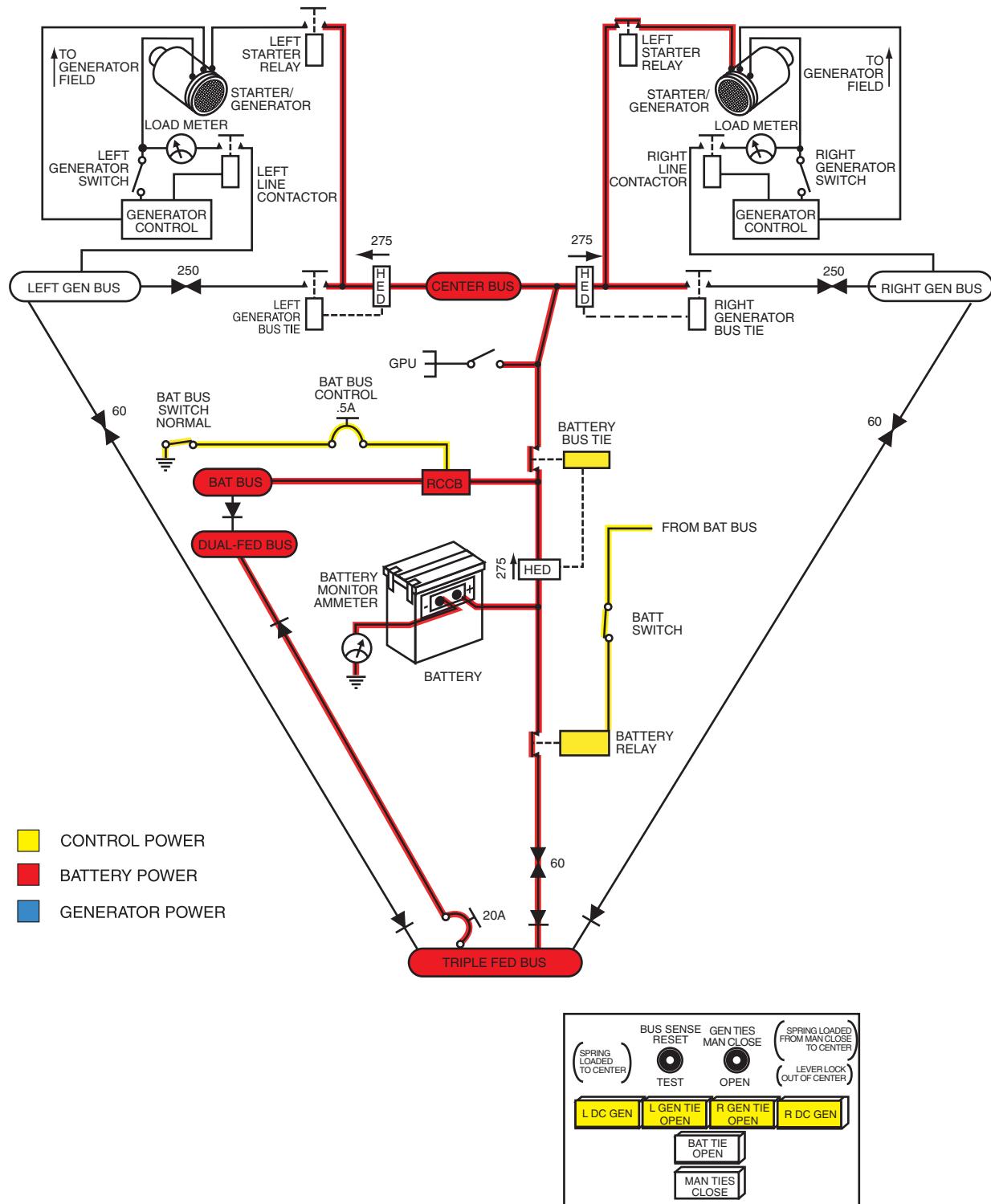


Figure 2-46 King Air 350 Power Distribution—Battery Switch ON (Generator Ties Manually Closed)


Figure 2-47 King Air 350 Power Distribution—Right Engine Start

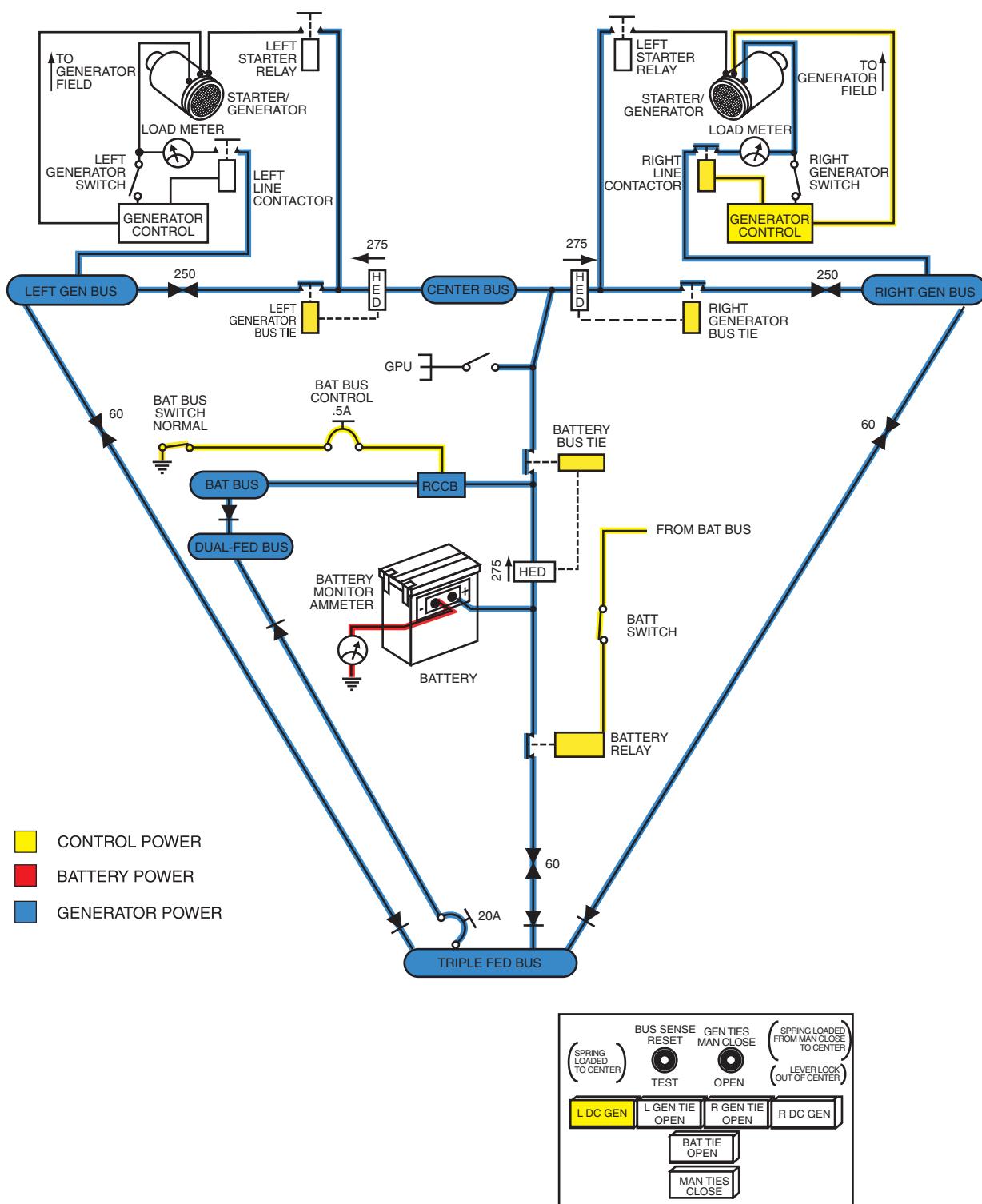
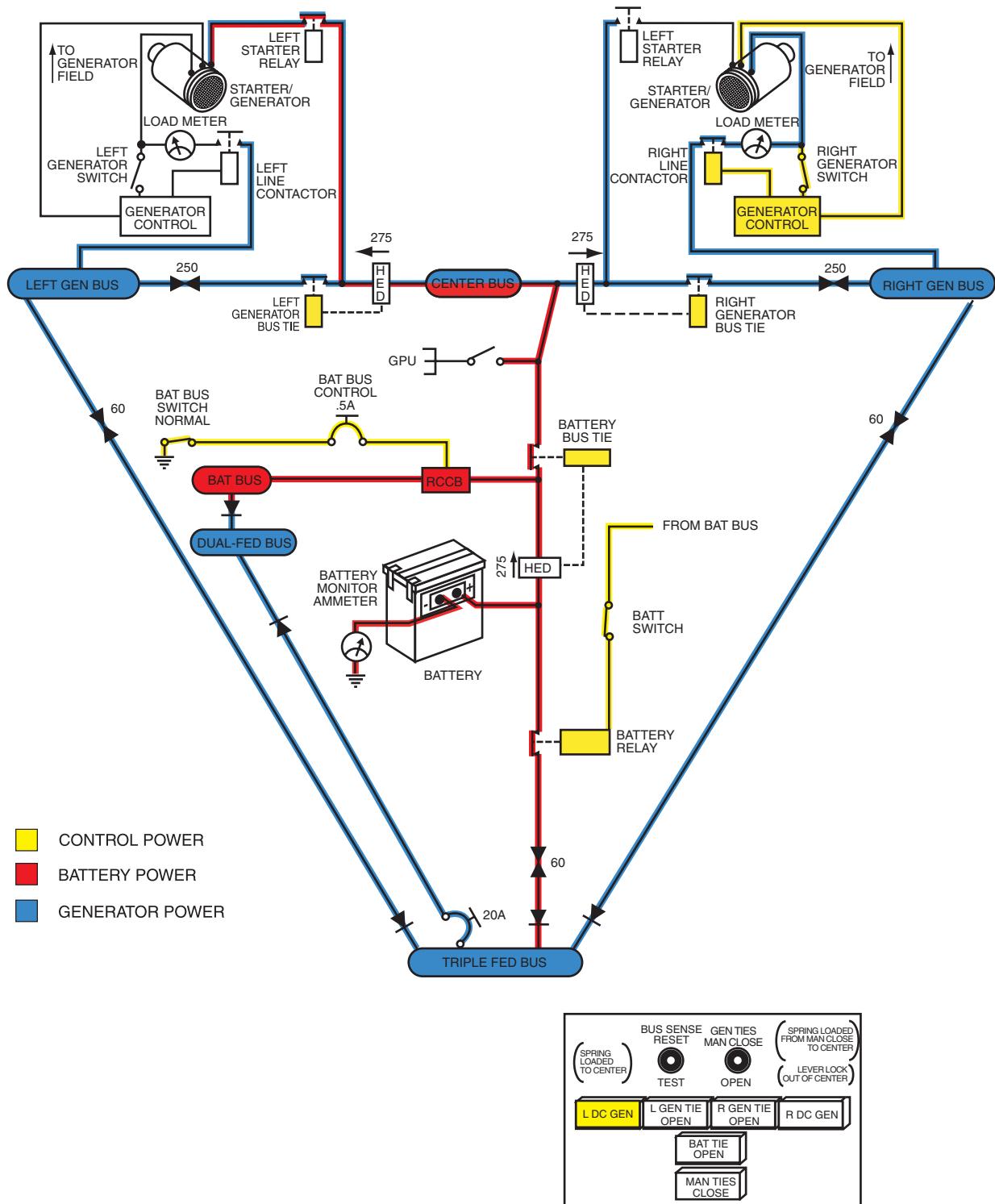


Figure 2-48 King Air 350 Power Distribution—Right Generator ON


Figure 2-49 King Air 350 Power Distribution—Cross-Generator Start (Left Engine)

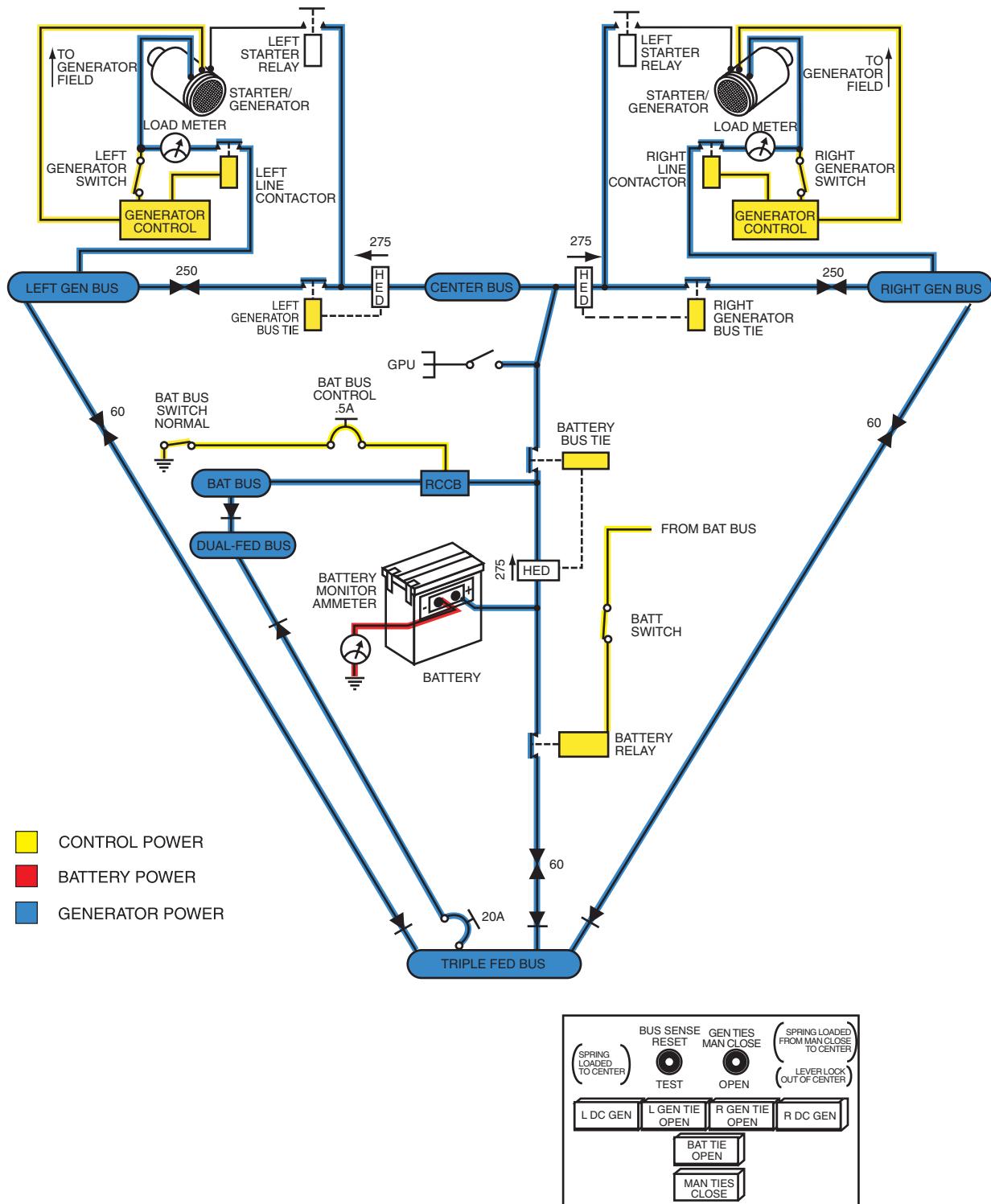


Figure 2-50 King Air 350 Power Distribution—Both Generators ON

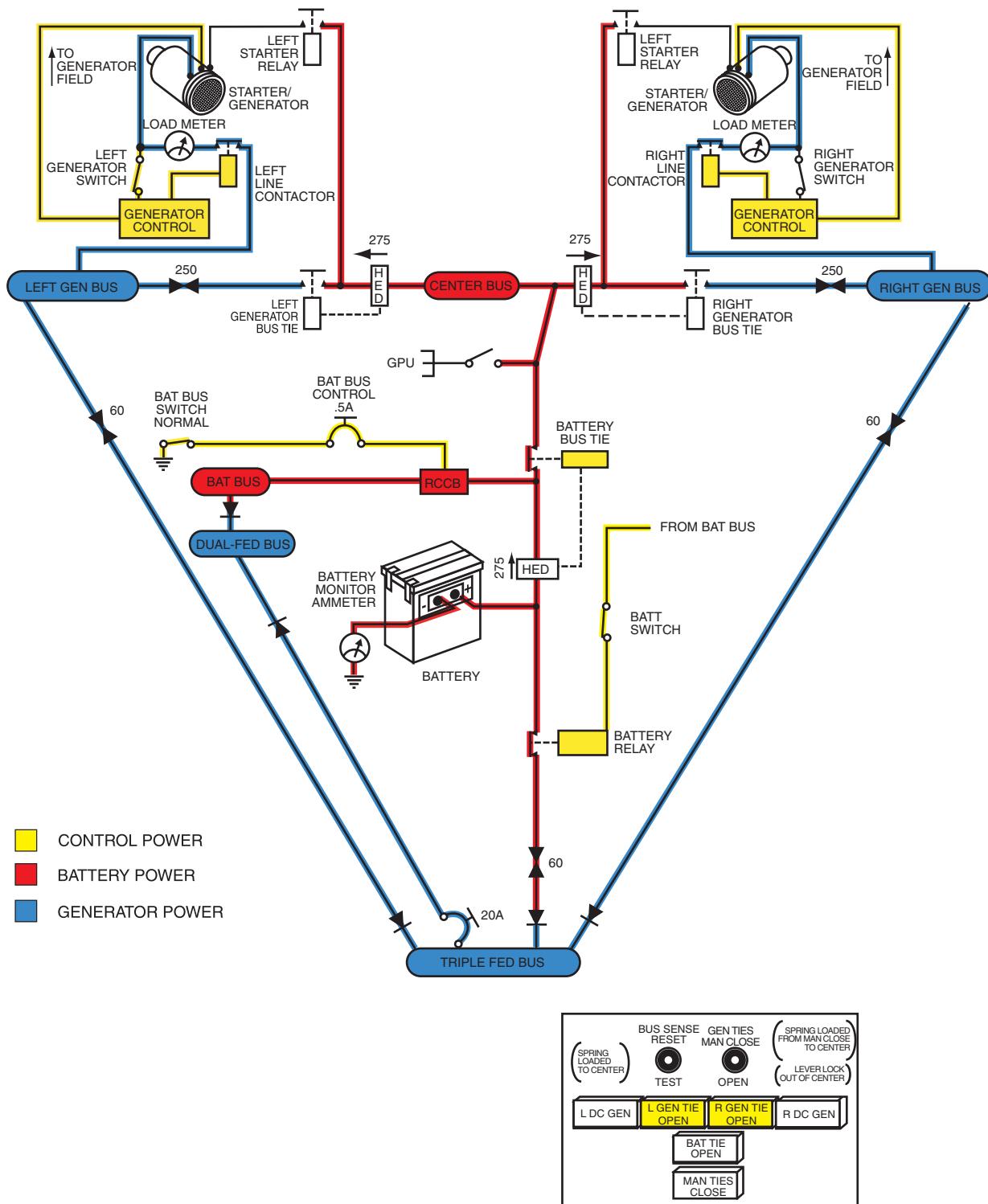


Figure 2-51 King Air 350 Power Distribution—Both Generators ON (Generator Ties Open)

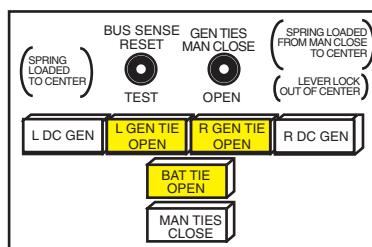
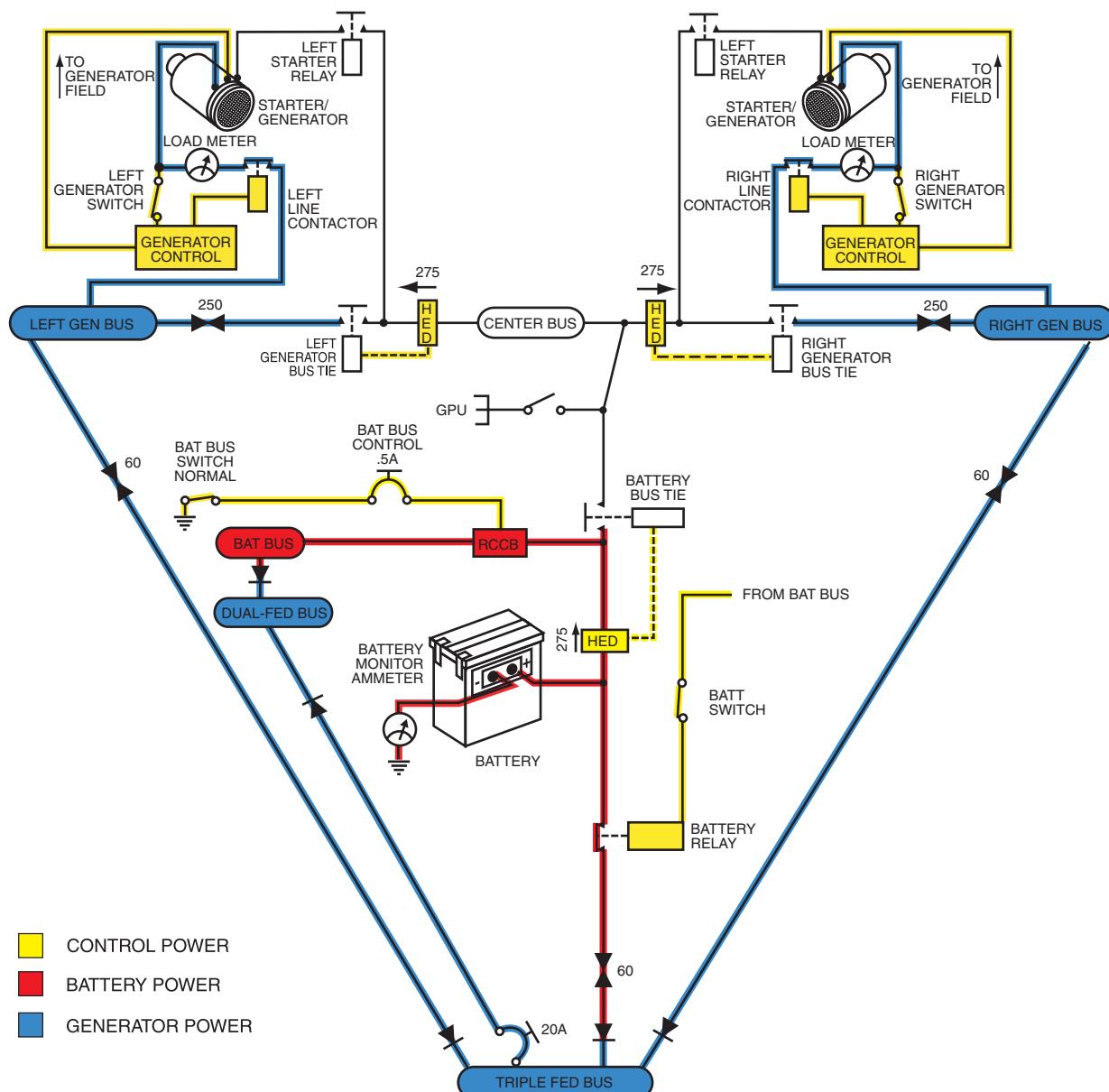
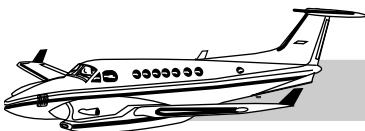


Figure 2-52 King Air 350 Power Distribution—Both Generators On (Bus Sense Test)

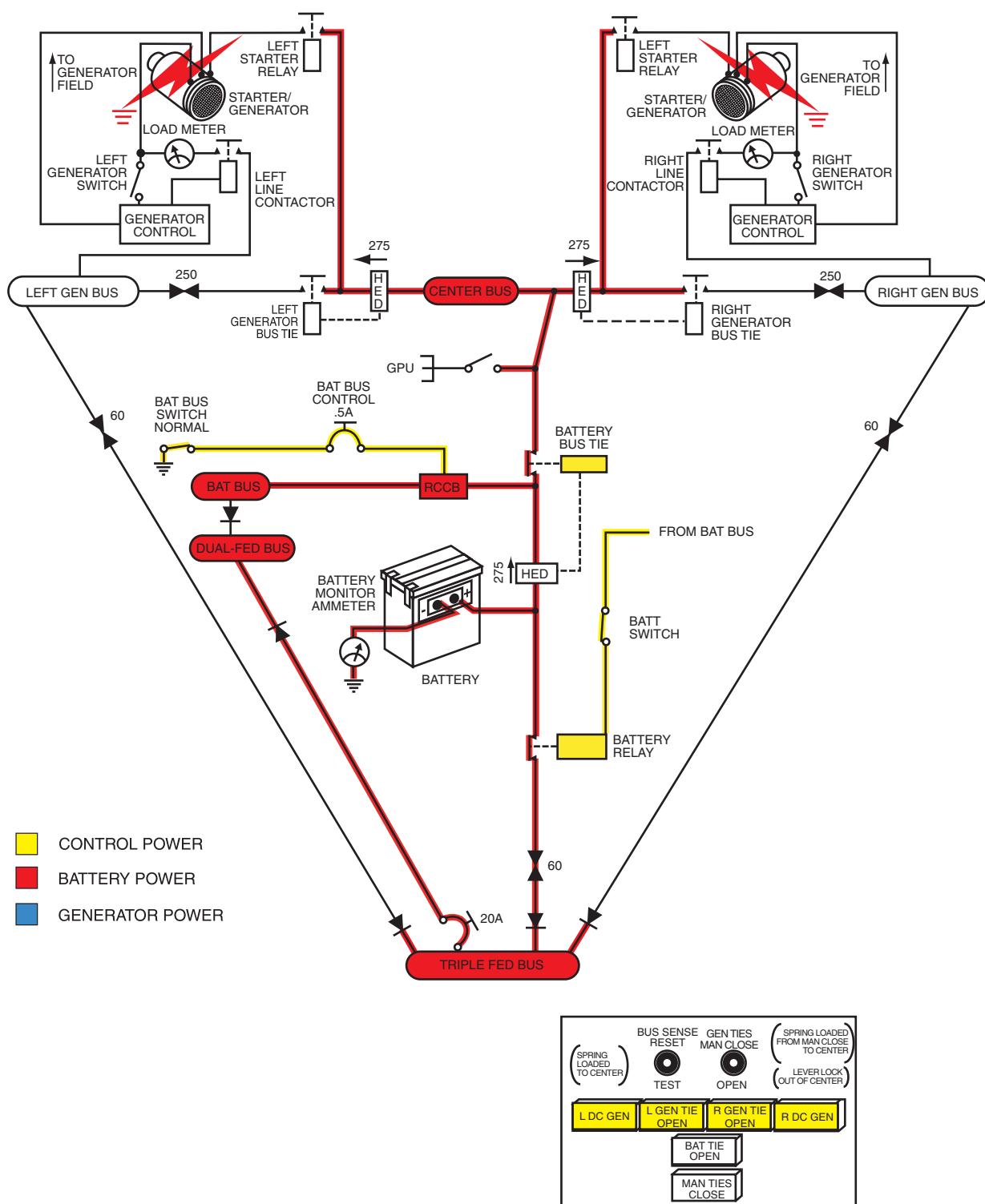


Figure 2-53 King Air 350 Power Distribution—Both Generators Failed (Load-Shedding)

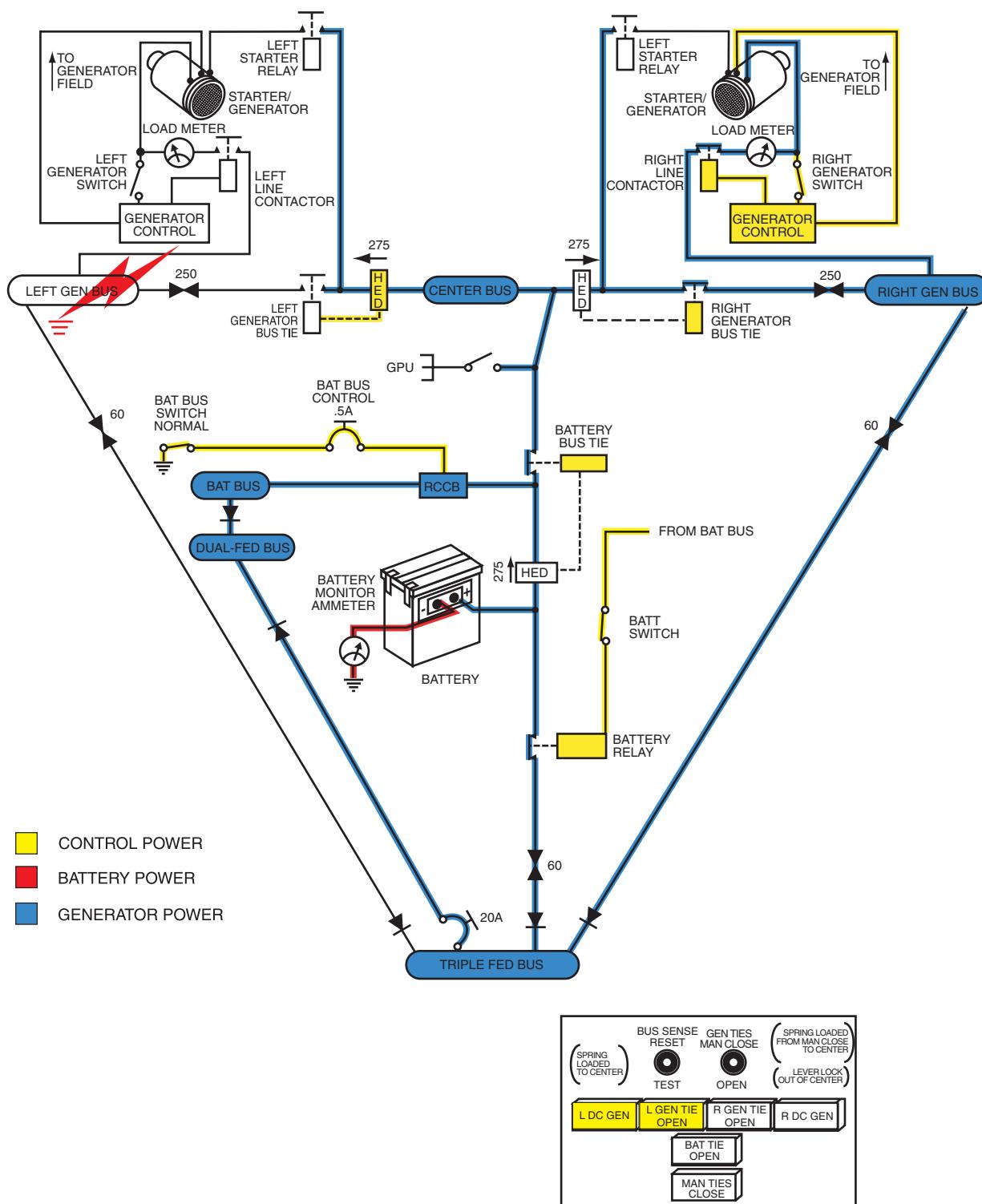


Figure 2-54 King Air 350 Power Distribution—Bus Isolation (Left Generator Bus Short)

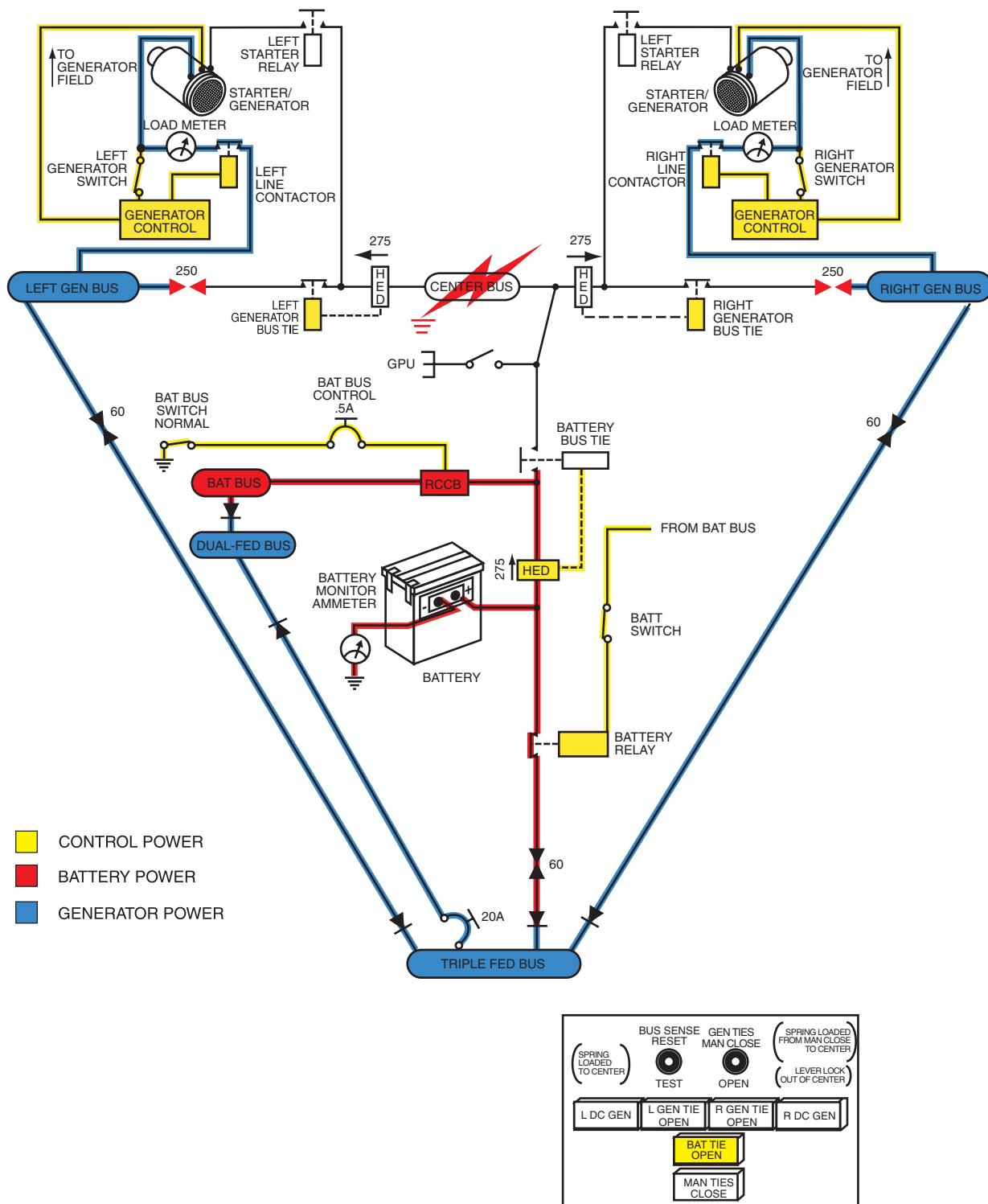


Figure 2-55 King Air 350 Power Distribution—Bus Isolation (Center Bus Short)

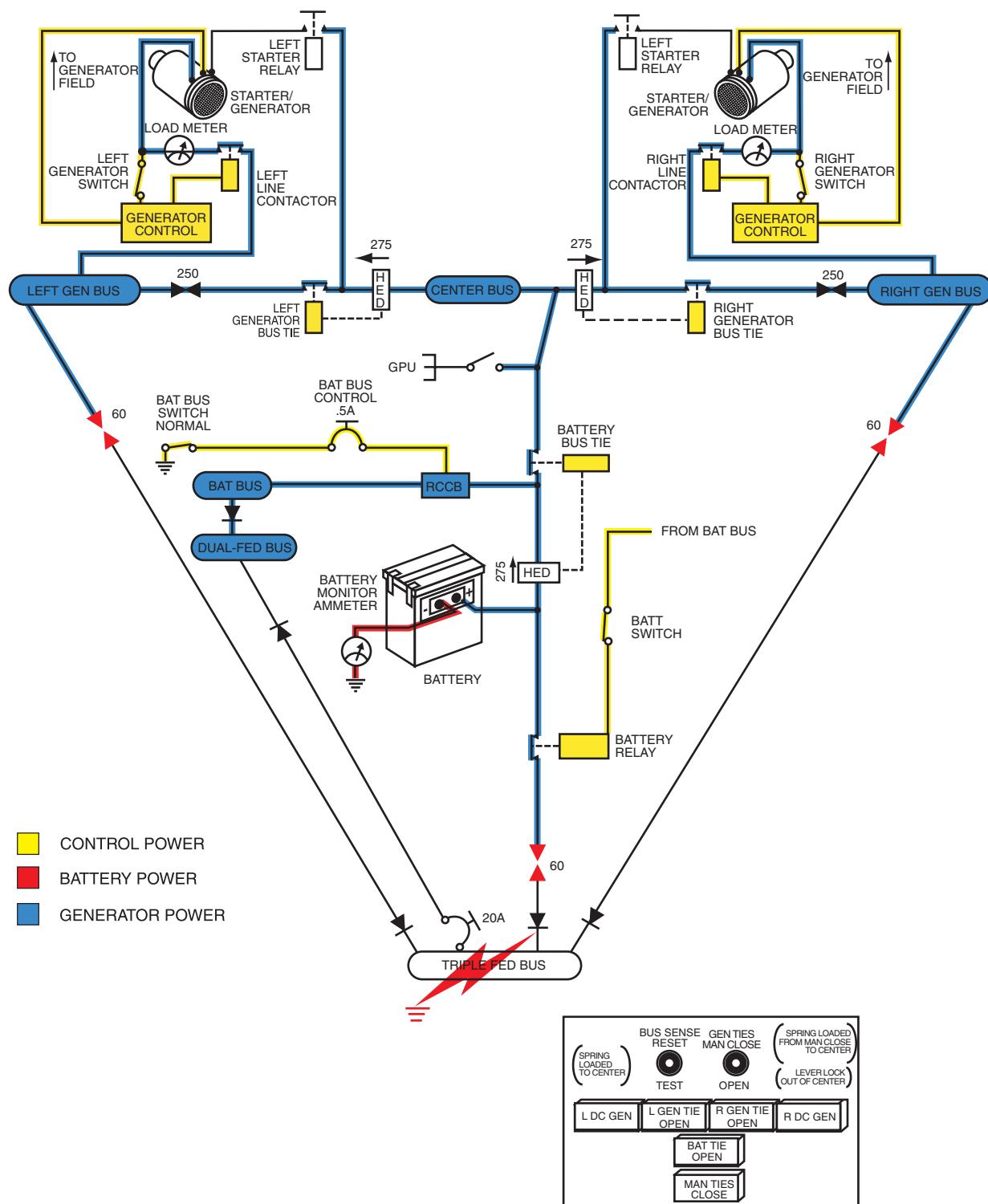


Figure 2-56 King Air 350 Power Distribution—Bus Isolation (Triple-Fed Bus Short)

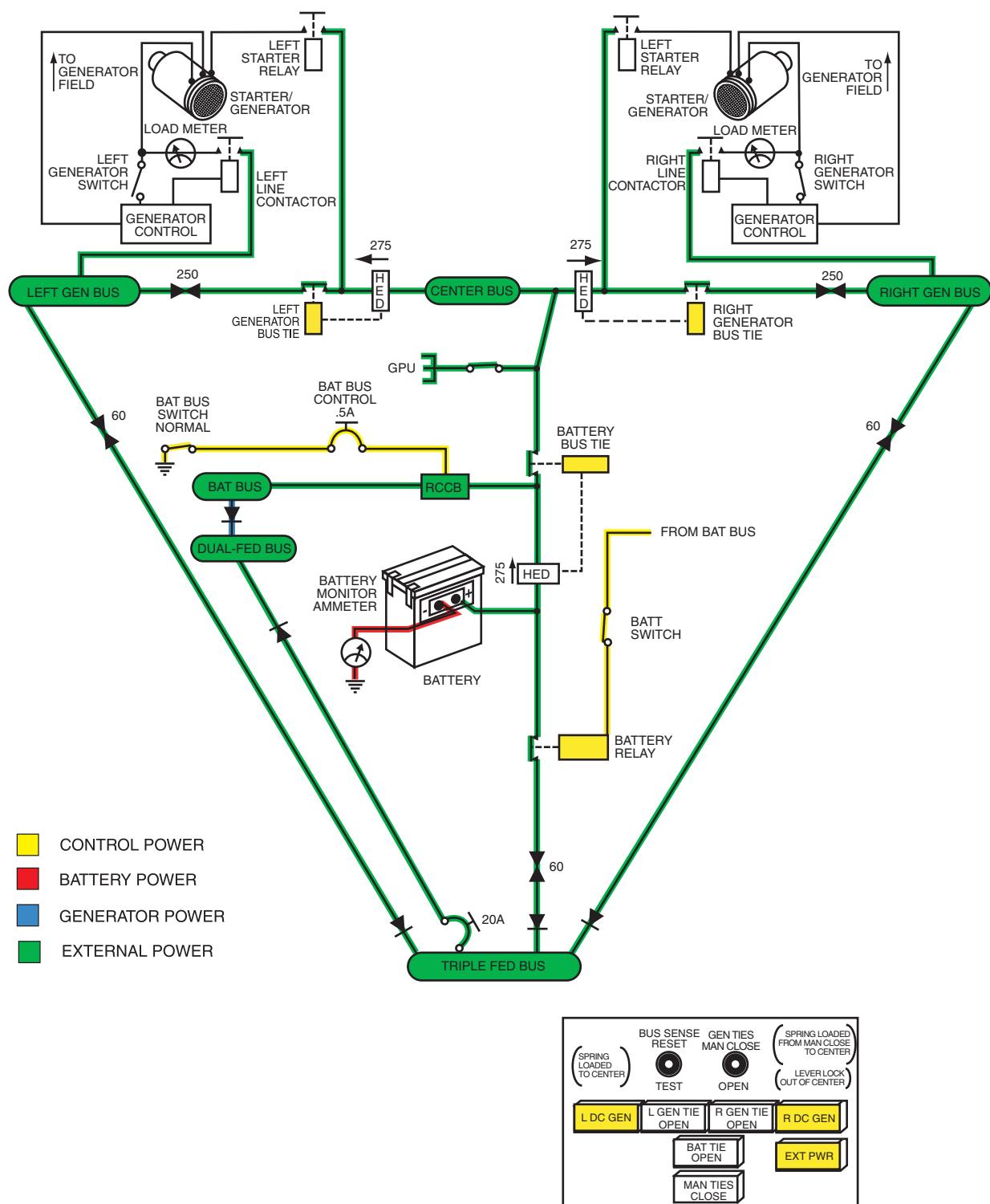


Figure 2-57 King Air 350 Power Distribution—External Power



CHAPTER 3

LIGHTING

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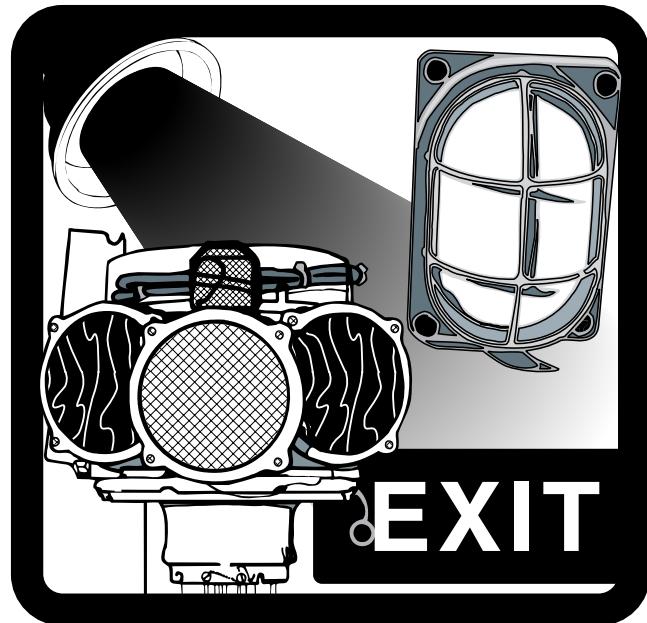
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CHAPTER 3

LIGHTING



INTRODUCTION

The King Air 300/350 lighting system consists of cockpit-controlled interior and exterior lights.

Interior lights are in the cockpit and passenger cabin and also include entry and exit threshold lights and baggage area lights. Exterior lights consist of landing, taxi, wing ice, navigation, recognition, rotating beacons, wingtip and tail strobe lights, and the tail and threshold flood-lights. The airplane is also equipped with emergency lights.

DESCRIPTION

COCKPIT LIGHTING

An overhead light control panel , easily accessible to both pilot and copilot, incorporates a functional arrangement of all lighting systems in the cockpit. Each light group has its own rheostat switch placarded "BRT-OFF."

The MASTER PANEL LIGHTS ON-OFF switch is the master switch for: PILOT FLIGHT INSTR, COPILOT FLIGHT INSTR, COPILOT GYRO INSTR, ENGINE INSTR, AVIONICS PANEL, OVHD, PED & SUB-PANEL, and SIDE PANEL. PILOT OVHD FLOOD, INSTR INDIRECT, and COPILOT

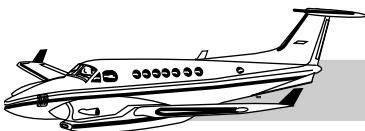


Figure 3-1 Cabin Light Control Panel

OVHD FLOOD are individually controlled by separate rheostat switches.

An instrument emergency light switch, located to the right of the electrical gages on the overhead panel, controls lights under the glareshield. Those lights may be used any time the normal cockpit lighting is not working. This light system is connected to the hot battery bus on the 300 or to the dual-fed bus on the 350.

CABIN LIGHTING

A three-position switch on the overhead light control panel, placarded "CABIN LIGHTS BRIGHT-DIM-OFF," controls the indirect fluorescent cabin lights. A switch to the right of the interior light switch activates the cabin no-smok-

ing/fasten-seat-belt signs and accompanying chimes. This three-position switch is placarded "NO SMK & FSB-OFF-FSB."

The baggage area lights are controlled by a three-position switch placarded "BAGGAGE-OFF-INDIRECT" just inside the airstair door, aft of the doorframe (Figure 3-2). The INDIRECT position of this switch is operable only when the triple-fed bus is powered, and the BAGGAGE position is connected directly to the hot battery bus on the 300 or the dual-fed bus on the 350.

A threshold light is located forward of the airstair door at floor level, and two aisle lights are located at floor level on both sides of the spar cover.

A switch adjacent to the threshold light turns these lights on and off (Figure 3-3). This switch

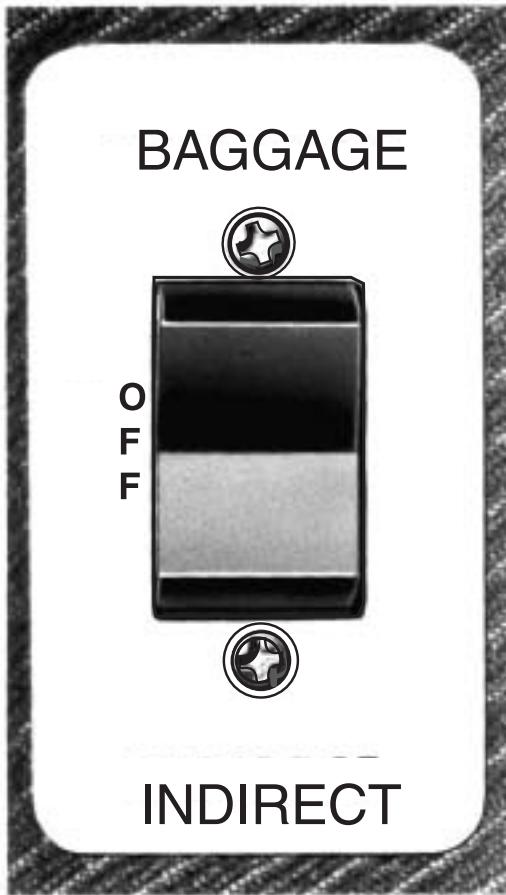


Figure 3-2 Baggage Compartment Light Switch

also activates the exterior entry light (under the left wing center section) and the lights under each step on the airstair door. When the airstair door is closed and latched, all the lights controlled by the threshold light switch are extinguished. These lights are powered by the hot battery bus on the 300 or the dual-fed bus on the 350.

When the aircraft master switch is on, the individual reading lights along the top of the cabin may be turned on or off by the passengers using the pushbutton switch adjacent to each light.

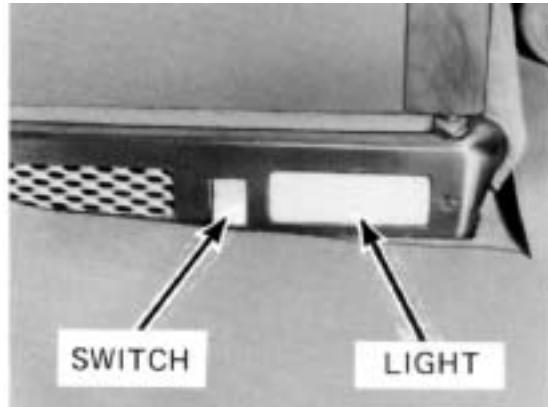


Figure 3-3 Threshold Light Switch

EXTERIOR LIGHTING

Switches for the landing lights, taxi lights, wing ice lights, navigation lights, recognition lights, rotating beacons, and wingtip and tail strobe lights are located on the pilot's right subpanel (Figure 3-4). They are appropriately placarded as to their function.

Tail floodlights, if installed, are incorporated into the horizontal stabilizers and are designed to illuminate both sides of the vertical stabilizer. A switch for these lights, placarded "LIGHTS TAIL FLOOD-OFF," is located on the pilot's right subpanel (Figure 3-4).

A flush-mounted floodlight is located forward of the flaps, in the bottom of the left wing. This light provides illumination of the area around the airstair door, to provide passenger convenience at night.

It is controlled by the threshold light switch, just inside the door on the forward doorframe, and is extinguished automatically whenever the cabin door is closed and latched (Figure 3-3).

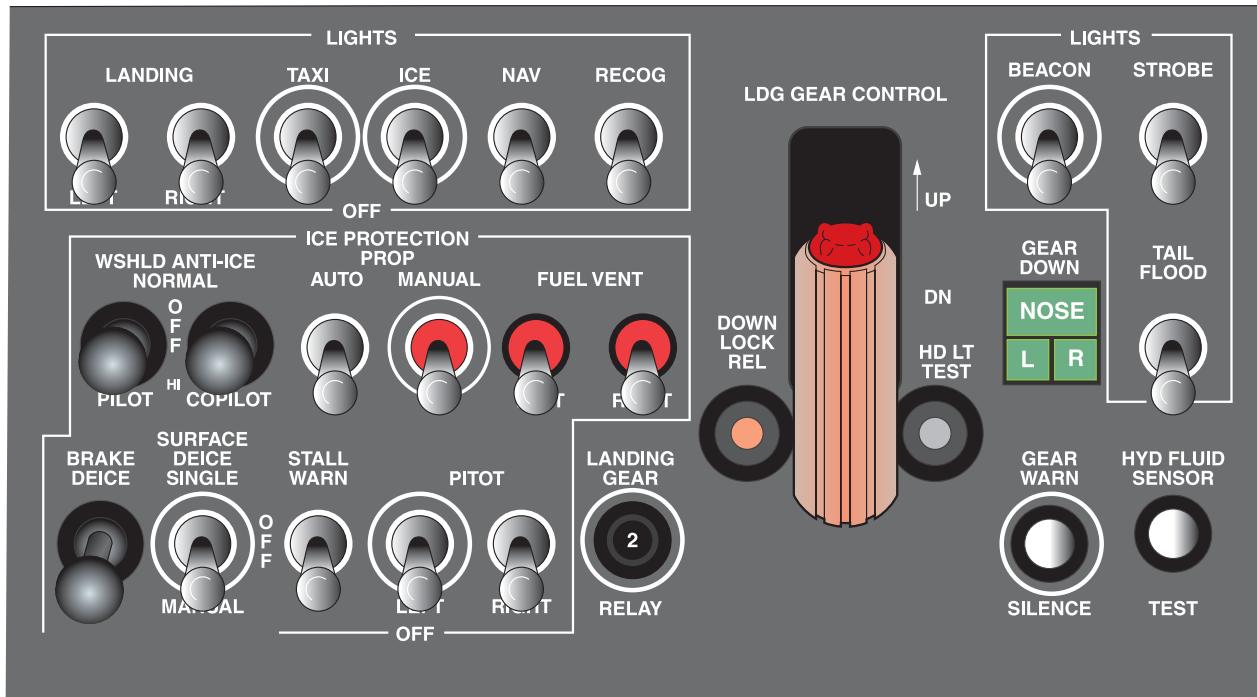
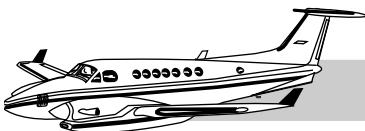


Figure 3-4 Exterior Lights Control

WING ICE LIGHTS

Wing ice lights, located on the outboard side of each nacelle (Figure 3-5), are provided to illuminate the wing leading edges to determine ice buildup.

They are controlled by the ICE lights circuit-breaker switch, located on the pilot's right sub-panel in the LIGHTS group (Figure 3-6). The wing ice lights should be used as required during night flight to check for wing ice accumulation. Since the wing ice lights operate at a high temperature, they should not be used for prolonged periods while the airplane is on the ground.



Figure 3-5 Wing Ice Light



Figure 3-6 Wing Ice Lights Switch



CHAPTER 4

MASTER WARNING SYSTEM

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CHAPTER 4

MASTER WARNING SYSTEM



INTRODUCTION

Warning and caution annunciators are the first indication of trouble or malfunction in a system or component of the airplane. Crewmembers should have complete familiarity with these annunciators and the action required to correct the problem or cope with the situation until a safe landing can be made or, in the case of an on-ground indication, the problem can be corrected.

GENERAL

This chapter presents a description and discussion of the warning, caution, advisory, and status annunciators. The annunciator panels are

described in detail, including each annunciator and the associated cause for illumination.



ANNUNCIATOR SYSTEM

The annunciator system (Figure 4-1) consists of a warning annunciator panel (red) centrally located in the glareshield (Figure 4-2) and a caution/adv-

sory annunciator panel on the King Air 300 or a caution/advisory/status annunciator panel on the King Air 350 (yellow for caution, green for advisory, white for status). All are located on the center subpanel (Figure 4-3).

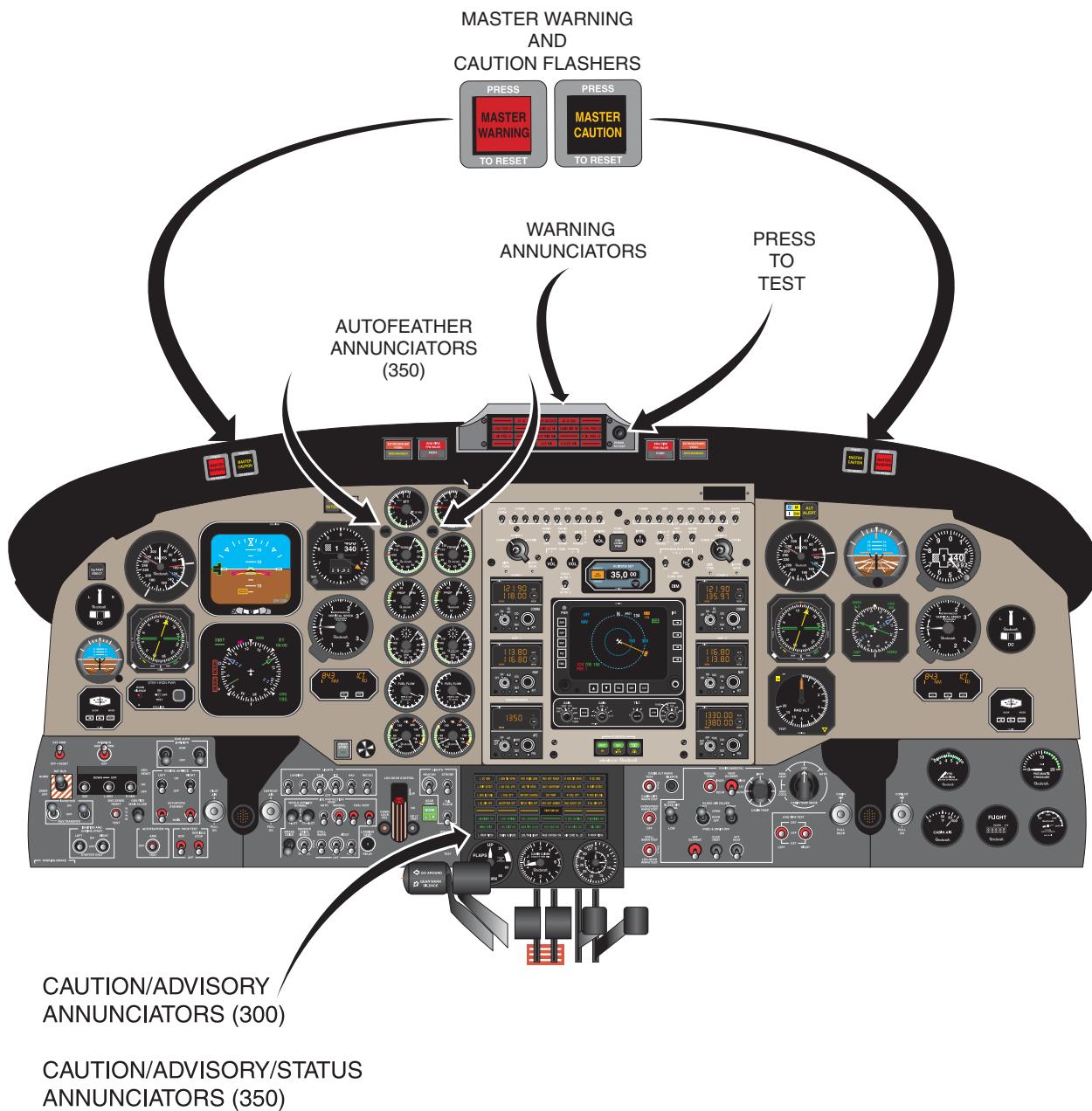


Figure 4-1 Master Warning System



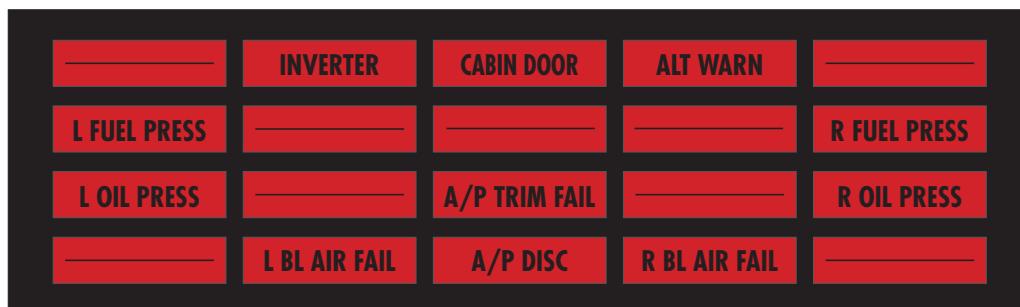
Two red MASTER WARNING flashers (Figure 4-4) located in the glareshield (one in front of the pilot and one in front of the copilot) are a part of the system, as are two yellow MASTER CAUTION flashers (located just inboard of the MASTER WARNING flashers), and a PRESS TO TEST switch located immediately to the right of the warning annunciator panel.

The annunciators are of the word-readout type. Whenever a fault condition covered by the annunciator system occurs, a signal is generated, and the appropriate annunciator is illuminated.

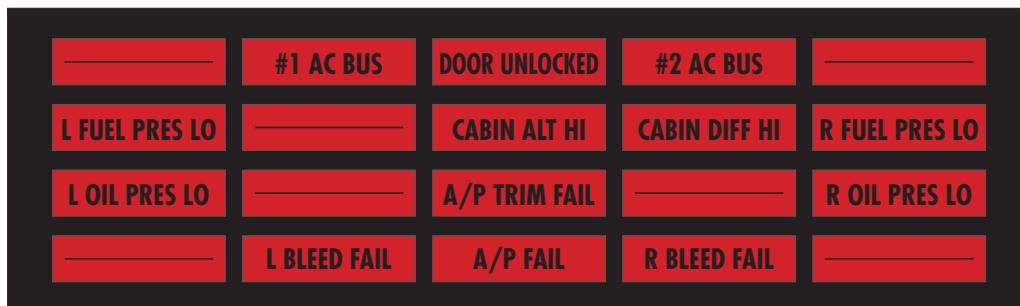
MASTER WARNING FLASHERS AND WARNING ANNUNCIATORS

If the fault requires the immediate attention and reaction of the pilot, the appropriate red warning annunciator in the warning annunciator panel illuminates, and both MASTER WARNING flashers begin flashing.

Any illuminated lens in the warning annunciator panel will remain on until the fault is corrected. However, the MASTER WARNING flashers can be extinguished by depressing the face of either MASTER WARNING flasher, even if the fault is not corrected. In such a case the MASTER WARNING flashers will again be activated if an additional warning annunciator illuminates.



KING AIR 300



KING AIR 350

Figure 4-2 Warning Annunciator Panel



When a warning fault is corrected, the affected warning annunciator will extinguish, but the MASTER WARNING flashers will

continue flashing until one of them is depressed (Figure 4-4).

L DC GEN	L GEN TIE OPEN	HYD FLUID LOW	RVS NOT READY	R GEN TIE OPEN	R DC GEN
L CHIP DETECT	L NO FUEL XFR	BAT TIE OPEN	DUCT OVERTEMP	R NO FUEL XFR	R CHIP DETECT
L ENG ICE FAIL	L FUEL QTY	BATTERY CHARGE	EXT PWR	R FUEL QTY	R ENG ICE FAIL
L PROP PITCH	—	LDG/TAXI LIGHT	PASS OXY ON	PROP GND SOL	R PROP PITCH
L AUTOFEAUTHER	L ENG ANTI-ICE	ELEC TRIM OFF	AIR COND N ₁ LOW	R ENG ANTI-ICE	R AUTOFEAUTHER
—	L BK DEICE ON	MAN TIES CLOSE	ELEC HEAT ON	R BK DEICE ON	—
L IGNITION ON	L BL AIR OFF	—	FUEL CROSSFEED	R BL AIR OFF	R IGNITION ON

KING AIR 300

L DC GEN	L GEN TIE OPEN	HYD FLUID LOW	RVS NOT READY	R GEN TIE OPEN	R DC GEN
L CHIP DETECT	L NO FUEL XFR	BAT TIE OPEN	DUCT OVERTEMP	R NO FUEL XFR	R CHIP DETECT
L ENG ICE FAIL	L FUEL QTY	BATTERY CHARGE	EXT PWR	R FUEL QTY	R ENG ICE FAIL
L BL AIR OFF	AUTOFEAUTHER OFF	PITCH TRIM OFF	OXY NOT ARMED	RUD BOOST OFF	R BL AIR OFF
L IGNITION ON	L ENG ANTI-ICE	FUEL CROSSFEED	PROP GND SOL	R ENG ANTI-ICE	R IGNITION ON
WING DEICE	L BK DEICE ON	MAN TIES CLOSE	ELEC HEAT ON	R BK DEICE ON	TAIL DEICE
L PROP PITCH	CABIN ALTITUDE	LDG/TAXI LIGHT	PASS OXY ON	AIR COND N ₁ LOW	R PROP PITCH

KING AIR 350 Prior to FL289

L DC GEN	L GEN TIE OPEN	HYD FLUID LOW	RVS NOT READY	R GEN TIE OPEN	R DC GEN
L CHIP DETECT	L NO FUEL XFR	BAT TIE OPEN	DUCT OVERTEMP	R NO FUEL XFR	R CHIP DETECT
L ENG ICE FAIL	L FUEL QTY	—	EXT PWR	R FUEL QTY	R ENG ICE FAIL
L BL AIR OFF	AUTOFEAUTHER OFF	PITCH TRIM OFF	OXY NOT ARMED	RUD BOOST OFF	R BL AIR OFF
—	—	—	PROP GND SOL	—	—
L IGNITION ON	L ENG ANTI-ICE	FUEL CROSSFEED	—	R ENG ANTI-ICE	R IGNITION ON
WING DEICE	L BK DEICE ON	MAN TIES CLOSE	ELEC HEAT ON	R BK DEICE ON	TAIL DEICE
L PROP PITCH	CABIN ALTITUDE	LDG/TAXI LIGHT	PASS OXY ON	AIR COND N ₁ LOW	R PROP PITCH

KING AIR 350 FL289 and After

Figure 4-3 Caution/Advisory Annunciator Panel

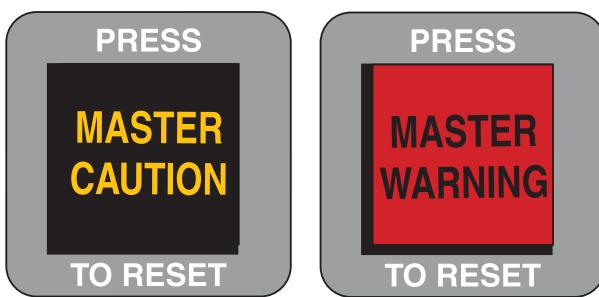


Figure 4-4 Master Warning and Master Caution Flashers

MASTER CAUTION FLASHERS AND CAUTION ANNUNCIATORS

Whenever an annunciator-covered fault occurs that requires the pilot's attention but not his immediate reaction, the appropriate yellow caution annunciator illuminates, in the caution/advisory panel on the King Air 300 or the caution/advisory/status panel on the King Air 350, and both MASTER CAUTION flashers begin flashing.

The flashing MASTER CAUTION lights can be extinguished by pressing the face of either of the flashing lights to reset the circuit. Subsequently, when any caution annunciator illuminates, the MASTER CAUTION flashers will be activated again. An illuminated caution annunciator will remain on until the fault condition is corrected, at which time it will extinguish, but the MASTER CAUTION flashers will continue flashing until one of them is depressed (Figure 4-4).

ADVISORY AND STATUS ANNUNCIATORS

The caution/advisory annunciator panel on the King Air 300 and the caution/advisory/status panel on the King Air 350 contain the green advisory annunciators. Only the King Air 350 annunciator panel has white status annunciators in addition to the green annunciators. There are no master flashers associated with the green or white annunciators, since they indicate functional situa-

tions which do not demand the immediate attention or reaction of the pilot. These annunciators can be extinguished only by changing the condition indicated by the illuminated lens.

On the King Air 300, the green advisory annunciators indicate a condition which can be normal or abnormal and which may or may not have been initiated by the pilot.

On the King Air 350, the green advisory annunciators confirm that an operation initiated by the pilot did occur as required. The white status annunciators indicate a condition which can be normal or abnormal and which may or may not have been initiated by the pilot.

Also on the King Air 350, on aircraft prior to FL-120, two green AUTOFEATHER annunciators are located on the pilot's glareshield, between the left fire extinguisher switch and the pilot's MASTER WARNING and CAUTION flashers (prior to FL120). On airplanes FL-120 and subsequent, the two green AUTOFEATHER lights are adjacent to the torquemeters on the instrument panel.

KING AIR 300 DIMMING

The warning annunciators, caution annunciators, advisory annunciators, annunciators in the firewall fuel valve pushbuttons, yellow MASTER CAUTION flashers, landing gear handle lights, and gear position lights feature both a BRIGHT and a DIM mode. Both modes are at a preset illumination intensity.

The DIM mode will be selected automatically whenever all of the following conditions are met:

- A generator is on line.
- Pilot's or copilot's overhead floodlights are OFF.
- MASTER PANEL LIGHTS switch is ON.
- Pilot's flight lights are ON.
- Ambient light level in the cockpit (as sensed by a photoelectric cell located in



the overhead light control panel) is below a preset value.

Unless all these conditions are met, the BRIGHT mode will be selected automatically. The MASTER WARNING flashers and the annunciators in the fire extinguisher pushbuttons do not have a DIM mode.

KING AIR 350 DIMMING

The red MASTER WARNING flashers, yellow MASTER CAUTION flashers, warning annunciators, caution annunciators, advisory annunciators, status annunciators, annunciators in the firewall fuel valve pushbuttons, landing gear handle lights, and gear position lights have a BRIGHT and a DIM mode of variable intensity as sensed by a photoelectric cell.

The DIM mode will be selected automatically whenever all of the following conditions are met:

- A generator is on line.
- Pilot's or copilot's overhead floodlights are OFF.
- MASTER PANEL LIGHTS switch is ON.
- Pilot's flight lights are ON.
- Ambient light level in the cockpit (as sensed by a photoelectric cell located in the overhead light control panel) is below a preset value.

Unless all these conditions are met, the BRIGHT mode will be selected automatically. The annunciators in the fire extinguisher pushbuttons do not have a DIM mode. The AUTOFEATHER annunciators on the glareshield are dimmed by a rheostat located on the instrument panel.

TESTING AND LAMP REPLACEMENT

The lamps in the annunciator system should be tested before every flight and any time the integ-

rity of a lamp is in question. Depressing the PRESS TO TEST button, located to the right of the warning annunciator panel in the glareshield, illuminates all the annunciator lights, MASTER WARNING flashers, and MASTER CAUTION flashers. Any lamp that fails to illuminate when tested should be replaced (Figure 4-5).

To replace an annunciator lamp, first depress the center of the annunciator with your finger. Release your finger, and the annunciator lens will pop out slightly. Pull the annunciator from the panel, and remove the lamp from the rear of the annunciator. Replace the failed bulb with a spare lamp contained in an unused annunciator. Depress the annunciator until it locks in place.

WARNING AND CAUTION/ADVISORY PANEL DESCRIPTIONS

Tables 4-1 through 4-7 list all the warning, caution, advisory, and status annunciators on the King Air 300 and 350. The cause for illumination is included beside each annunciator.

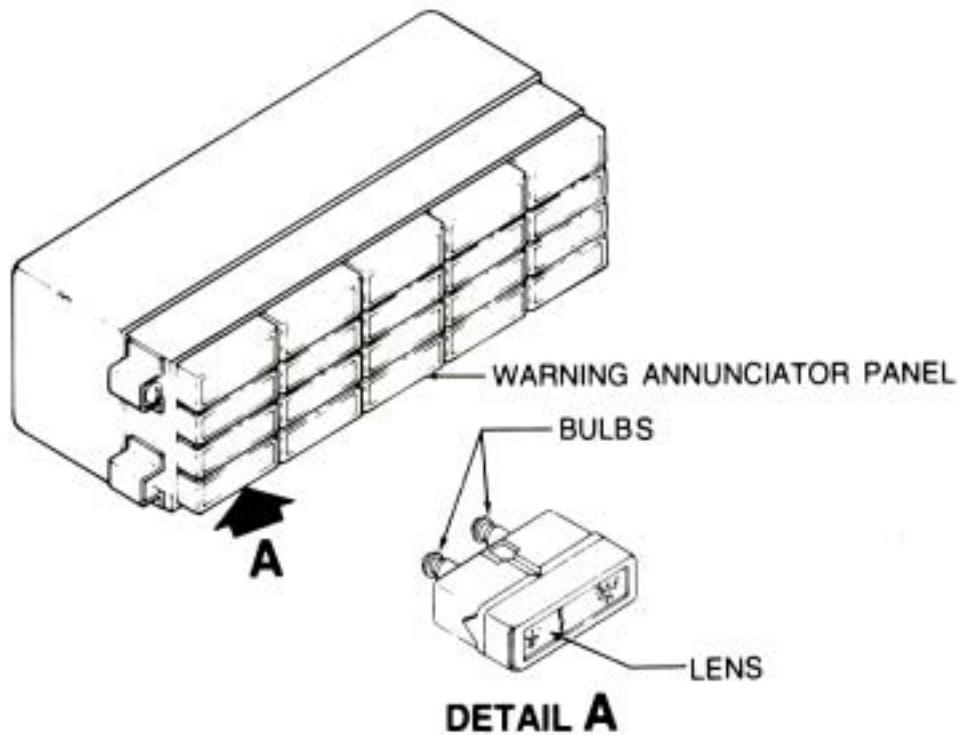


Figure 4-5 Annunciator Lamp Replacement

Table 4-1 KING AIR 300 WARNING ANNUNCIATORS

ANNUNCIATOR	CAUSE FOR ILLUMINATION
INVERTER	No inverter is selected, or the inverter selected is inoperative (flashes three times during self-test).
CABIN DOOR	Cabin door is open or not secure.
ALT WARN	Cabin pressure altitude exceeds 12,500 feet.
L FUEL PRESS	Low fuel boost pressure on left side (below approximately 10 psi).
R FUEL PRESS	Low fuel boost pressure on right side (below approximately 10 psi).
L OIL PRESS	Low oil pressure left engine (below approximately 60 psi).

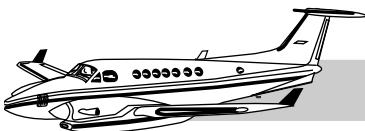


Table 4-1 KING AIR 300 WARNING ANNUNCIATORS (Cont)

ANNUNCIATOR	CAUSE FOR ILLUMINATION
A/P TRIM FAIL	Improper trim or no trim from autopilot trim command.*
R OIL PRESS	Low oil pressure right engine (below approximately 60 psi).
L BL AIR FAIL	Melted or failed plastic left bleed-air failure warning line.
A/P DISC	Autopilot is disconnected by a failed component.*
R BL AIR FAIL	Melted or failed plastic right bleed-air failure warning line.

* Optional equipment

Table 4-2 KING AIR 300 CAUTION ANNUNCIATORS

ANNUNCIATORS	CAUSE FOR ILLUMINATION
L DC GEN	Left generator is off line.
L GEN TIE OPEN	Left generator bus is isolated from the center bus.
HYD FLUID LOW	Hydraulic fluid level in the reservoir is low.
RVS NOT READY	Propeller levers are not positioned fully forward with the landing gear extended.
R GEN TIE OPEN	Right generator bus is isolated from the center bus.
R DC GEN	Right generator is off line.
L CHIP DETECT	Metal contamination in left engine oil is detected.
L NO FUEL XFR	No fuel transfer from the left auxiliary tank.
BATTIE OPEN	Battery is isolated from the center and generator buses.



Table 4-2 KING AIR 300 CAUTION ANNUNCIATORS (Cont)

ANNUNCIATORS	CAUSE FOR ILLUMINATION
DUCT OVERTEMP	Cabin environmental duct air is too hot (above approximately 300° F).
R NO FUEL XFR	No fuel transfer from the right auxiliary tank.
R CHIP DETECT	Metal contamination in right engine oil is detected.
L ENG ICE FAIL	Left engine anti-ice system is not functioning properly.
L FUEL QTY	Left main fuel quantity—Less than 30 minutes remaining at maximum continuous power.
BATTERY CHARGE	Battery charge rate is too high (Nicad).
EXT PWR	External power connector is plugged in (flashes with low external power voltage).
R FUEL QTY	Right main fuel quantity—Less than 30 minutes remaining at maximum continuous power.
R ENG ICE FAIL	Right engine anti-ice system is not functioning properly.
L PROP PITCH	Left propeller is below the flight idle stop.
PROP GND SOL	One or both ground idle low-pitch-stop solenoids are powered by 28 volts.*
R PROP PITCH	Right propeller is below the flight idle stop.

* FA226 and subsequent



Table 4-3 KING AIR 300 ADVISORY ANNUNCIATORS

ANNUNCIATOR	CAUSE FOR ILLUMINATION
LDG/TAXI LIGHT	Landing lights or the taxi light is on with the landing gear up.
PASS OXY ON	Oxygen system is on and available to the passenger masks.
L AUTOFEATHER	Left autofeather system is armed, with the power levers advanced above approximately 88% N ₁ .
L ENG ANTI-ICE	Left engine anti-ice vanes are in position for icing conditions.
ELEC TRIM OFF	Electric trim is deenergized by trim disconnect switch on the control wheel, with ELEV TRIM switch on the pedestal turned on.*
AIR COND N₁ LOW	Right engine N ₁ is too low for the air-conditioning load.
R ENG ANTI-ICE	Right engine anti-ice vanes are in position for icing conditions.
R AUTOFEATHER	Right autofeather system is armed, with the power levers advanced above 88% N ₁ .
L BK DEICE ON	Left brake deice system is in operation.*
MAN TIES CLOSE	Manually closed generator bus ties.
ELEC HEAT ON	Electric heat is on.
R BK DEICE ON	Right brake deice system is in operation.*
L IGNITION ON	Left IGNITION AND ENGINE START switch is on, or left auto-ignition system is armed with left engine torque below approximately 17%.
L BL AIR OFF	Left bleed-air valve switch is not open.
FUEL CROSSFEED	Fuel crossfeed is selected.



Table 4-3 KING AIR 300 ADVISORY ANNUNCIATORS (Cont)

ANNUNCIATOR	CAUSE FOR ILLUMINATION
R BL AIR OFF	Right bleed-air valve switch is not open.
R IGNITION ON	Right IGNITION AND ENGINE START switch is on, or right auto-ignition system is armed with right engine torque below approximately 17%.

* Optional equipment

Table 4-4 KING AIR 350 WARNING ANNUNCIATORS

ANNUNCIATOR	CAUSE FOR ILLUMINATION
#1 AC BUS	No. 1 AC bus failure.
DOOR UNLOCKED	Airstair door or cargo door is open or not secure.
#2 AC BUS	No. 2 AC bus failure.
L FUEL PRES LO	Fuel pressure failure on the left side.
CABIN ALT HI	Cabin pressure altitude exceeds 12,000 feet.
CABIN DIFF HI	Cabin differential pressure exceeds 6.9 psi.
R FUEL PRES LO	Fuel pressure failure on the right side.
L OIL PRES LO	Low oil pressure left engine.
A/P TRIM FAIL	Improper trim or no trim from autopilot trim command.*



Table 4-4 KING AIR 350 WARNING ANNUNCIATORS (Cont)

ANNUNCIATOR	CAUSE FOR ILLUMINATION
R OIL PRES LO	Low oil pressure right engine.
L BLEED FAIL	Melted or failed plastic left bleed-air failure warning line.
A/P FAIL	Autopilot monitor has sensed a failure.*
R BLEED FAIL	Melted or failed plastic right bleed-air failure warning line.

* Optional equipment

Table 4-5 KING AIR 350 CAUTION ANNUNCIATORS

ANNUNCIATORS	CAUSE FOR ILLUMINATION
L DC GEN	Left generator is off line.
L GEN TIE OPEN	Left generator bus is isolated from the center bus.
HYD FLUID LOW	Hydraulic fluid in the power pack is low.
RVS NOT READY	Propeller levers are not in the high-rpm, low-pitch position with the landing gear extended.
R GEN TIE OPEN	Right generator bus is isolated from the center bus.
R DC GEN	Right generator is off line.
L CHIP DETECT	Metal contamination in left engine oil is detected.
L NO FUEL XFR	No left auxiliary fuel transfer.



Table 4-5 KING AIR 350 CAUTION ANNUNCIATORS (Cont)

ANNUNCIATORS	CAUSE FOR ILLUMINATION
BATTIE OPEN	Battery is isolated from the generator buses.
DUCT OVERTEMP	Duct air is too hot.
R NO FUEL XFR	No right auxiliary fuel transfer.
R CHIP DETECT	Metal contamination in right engine oil is detected.
L ENG ICE FAIL	Left engine selected anti-ice system is inoperative.
L FUEL QTY	Left fuel quantity—Less than 30 minutes remaining at maximum continuous power.
BATTERY CHARGE	Battery charge rate is too high.*
EXT PWR	External power connector is plugged in.
R FUEL QTY	Right fuel quantity—Less than 30 minutes remaining at maximum continuous power.
R ENG ICE FAIL	Right engine selected anti-ice system is inoperative.
L BL AIR OFF	Left bleed-air valve switch is not open.
AUTOFATHER OFF	Autofeather switch is not armed, and landing gear is extended.
PITCH TRIM OFF	Electric trim is deenergized by trim disconnect switch on the control wheel, with PITCH TRIM switch on the pedestal turned on.
OXY NOT ARMED	Oxygen arming handle has not been pulled, or system failed to charge.
RUD BOOST OFF	Rudder boost switch is off.

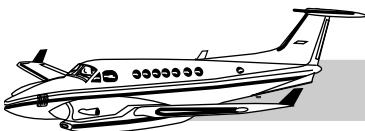


Table 4-5 KING AIR 350 CAUTION ANNUNCIATORS (Cont)

ANNUNCIATORS	CAUSE FOR ILLUMINATION
R BL AIR OFF	Right bleed-air valve switch is not open.
PROP GND SOL	One or both ground idle low-pitch-stop solenoids are powered by 28 volts.**

* Aircraft prior to FL215 and FM10

** FL115 and subsequent, FM10 and subsequent, FN2 and subsequent

Table 4-6 KING AIR 350 ADVISORY ANNUNCIATORS

ANNUNCIATOR	CAUSE FOR ILLUMINATION
L IGNITION ON	Left IGNITION AND ENGINE START switch is on, or left auto-ignition system is armed with left engine torque below 17%.
L ENG ANTI-ICE	Left engine anti-ice vanes are in position for icing conditions.
FUEL CROSSFEED	Fuel crossfeed is selected.
R ENG ANTI-ICE	Right engine anti-ice vanes are in position for icing conditions.
R IGNITION ON	Right IGNITION AND ENGINE START switch is on, or right auto-ignition system is armed with right engine torque below 17%.
WING DEICE	Wing surface deice system is in operation.
L BK DEICE ON	Left brake deice system is in operation.*
MAN TIES CLOSE	Manually closed generator bus ties.
ELEC HEAT ON	Electric heat is on.

**Table 4-6 KING AIR 350 ADVISORY ANNUNCIATORS (Cont)**

ANNUNCIATOR	CAUSE FOR ILLUMINATION
R BK DEICE ON	Right brake deice system is in operation.*
TAIL DEICE	Horizontal stabilizer surface deice system is in operation.

* Optional equipment

Table 4-7 KING AIR 350 STATUS ANNUNCIATORS

ANNUNCIATOR	CAUSE FOR ILLUMINATION
L PROP PITCH	Left propeller is below the flight idle stop.
CABIN ALTITUDE	Cabin altitude exceeds 10,000 feet.
LDG/TAXI LIGHT	Landing lights or the taxi light is on with the landing gear up.
PASS OXY ON	Passenger oxygen system is charged.
AIR COND N₁ LOW	Right engine N ₁ is too low for the air-conditioning load.
R PROP PITCH	Right propeller is below the flight idle stop.



CHAPTER 5

FUEL

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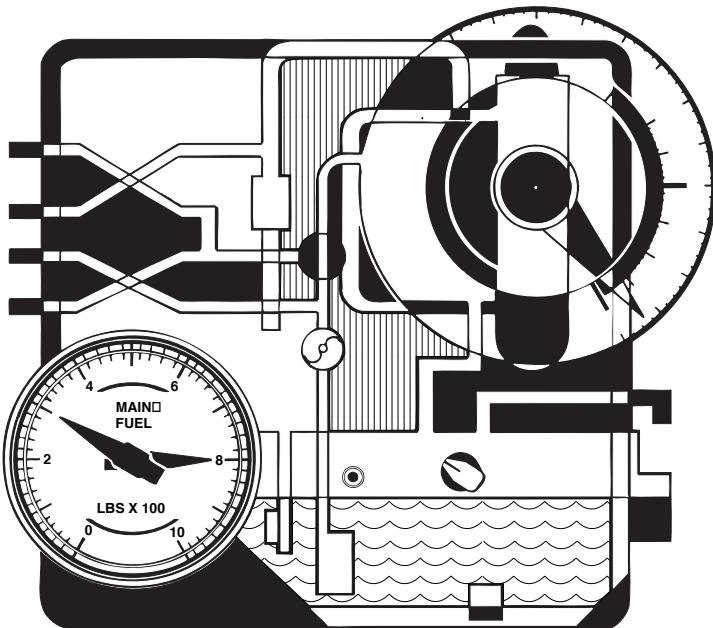
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CHAPTER 5

FUEL



INTRODUCTION

A complete understanding of the fuel system is essential to competent and confident operation of the aircraft. Management of fuel and fuel system components is a major everyday concern of the pilot. This section gives the pilot needed information for safe, efficient fuel management.

DESCRIPTION

The Fuel System section of the training manual presents a description and discussion of the fuel system. The physical layout of the fuel cells and fuel system are described in this section. Correct use of boost pumps, transfer pumps, crossfeed system, and firewall fuel valves are discussed. Fuel vents and drains with their location and type are also described along with correct procedure for taking and inspecting samples of fuel. Approved fuels, additives, and tank filling sequence are included.

FUEL SYSTEM

The Beechcraft Super King Air 300 and 350 fuel system is designed to simplify cockpit flight procedures and provide easy access for ground servicing (Figure 5-1). There are two separate wing fuel systems, one for each engine, connected by a valve-controlled crossfeed system. Each fuel system consists of separate main and auxiliary fuel tanks. Total usable fuel capacity is 539 gallons.



KING AIR 300/350

PILOT TRAINING MANUAL

FlightSafety
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FOR TRAINING PURPOSES ONLY

5-2

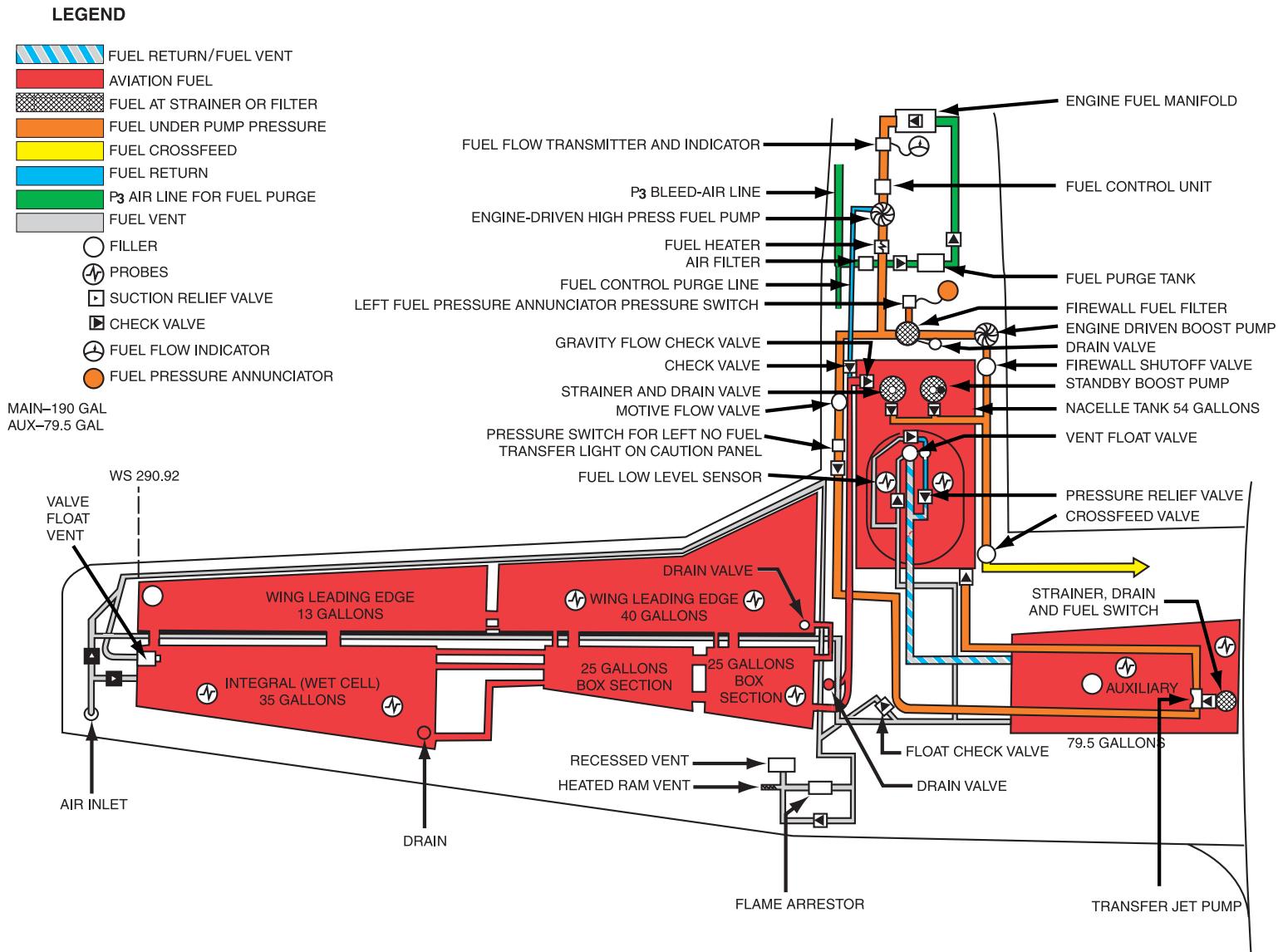


Figure 5-1 Fuel System Schematic Diagram



MAIN FUEL TANK SYSTEM

The main fuel system (Figure 5-2) in each wing consists of two wing leading edge bladder-type cells, two box-section bladder-type cells, one wet wing integral-type cell, and the nacelle tank. All of these cells are interconnected by gravity feed lines and flow into the nacelle tank. Fuel for each engine is pumped directly from its nacelle fuel tank.

The filler cap for the main tank system is located near the wingtip by the leading edge. An anti-

siphon valve is installed at each filler port to prevent the loss of fuel or collapse of fuel tank bladder in the event the filler cap is improperly installed.

Each nacelle tank is connected to the engine on the opposite side by a crossfeed line for single engine operation. Crossfeed requires electric boost pump operation in the feeding nacelle tank. This system makes it possible for fuel in either wing system to be available to either engine and is to be used during single-engine operation only.

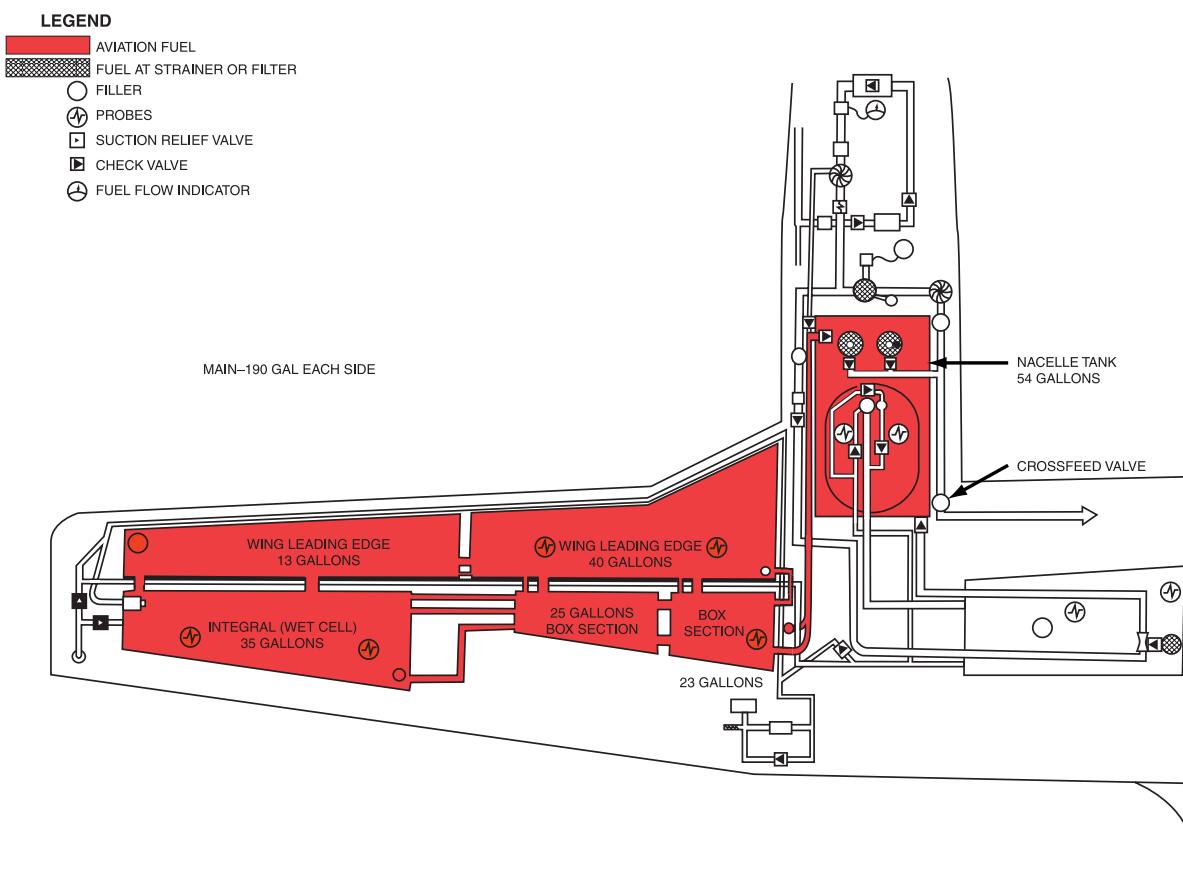
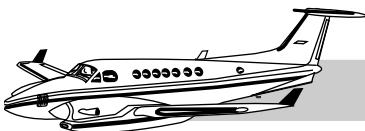


Figure 5-2 Main Fuel Tank System



AUXILIARY FUEL TANK SYSTEM

The auxiliary fuel system consists of a fuel tank on each side of the aircraft in the wing center section (Figure 5-3). The auxiliary tanks cannot gravity feed into the nacelle tanks due to their location lower than the nacelle tanks. Each auxiliary tank is equipped with its own filler opening, which has an antisiphon valve installed.

The auxiliary transfer system is automatic. If the auxiliary tank contains any usable fuel, the system will automatically transfer it to the nacelle tank. Since the fuel transfer system will insure that the nacelle tank stays full, auxiliary tank fuel is used first during normal operations. When the auxiliary fuel system is being used, fuel is trans-

ferred from it to the main fuel system by a jet pump mounted adjacent to the outlet strainer and drain in the auxiliary fuel cell.

If the tank has fuel but does not transfer due to some system discrepancy, an amber caution panel light advises the pilot of L or R NO FUEL XFER. The pilot has an override switch to back up the automatic system.

FUEL CAPACITY

The fuel quantity system is a capacitance gaging system with one quantity indicator per wing (Figure 5-4). A spring-loaded selector allows the pilot to individually check main and auxiliary tank quantity. The main system has a total capacity of

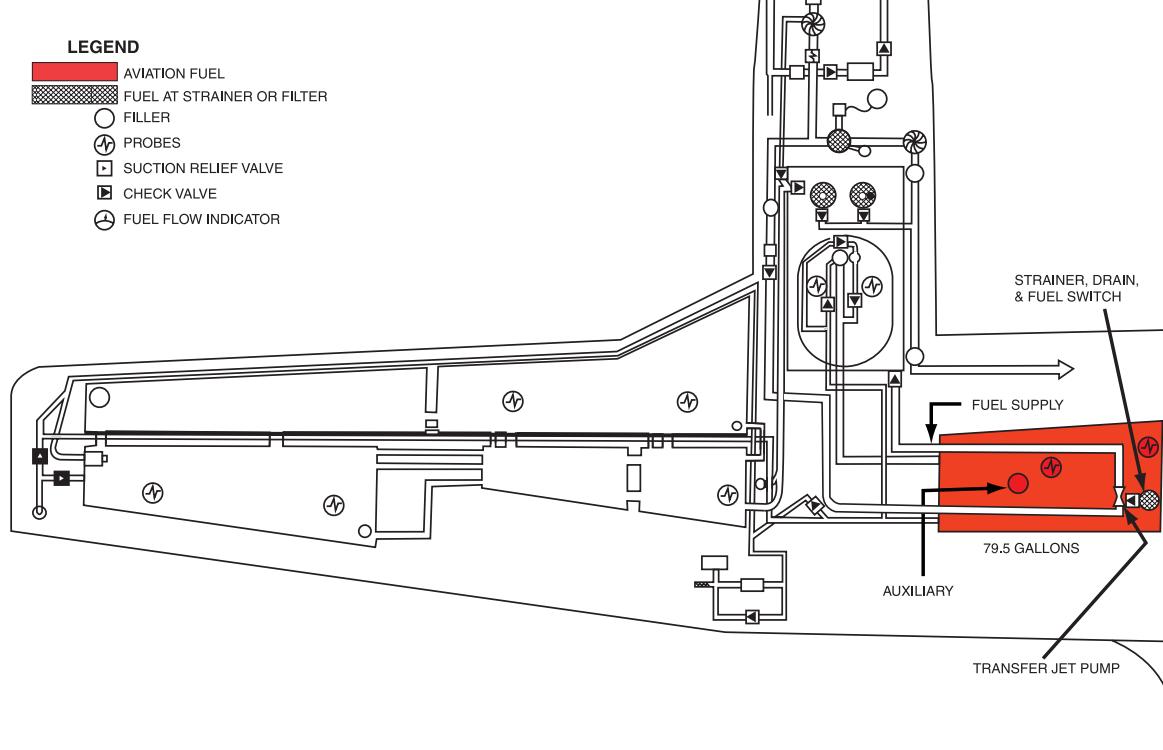


Figure 5-3 Auxiliary Fuel Tank System



380 gallons of usable fuel, 190 gallons on each side. The auxiliary system has a total of 159 gallons, 79.5 gallons on each side. The fuel quantity gages and the engine fuel flow indicators read in pounds times 100. At 6.7 pounds per gallon, approximately 2,546 pounds are available in the

main system, with approximately 1,273 on each side. Approximately 1,065 pounds are available in the auxiliary system with approximately 533 on each side. Total usable fuel is 539 gallons, or approximately 3,611 pounds.



KING AIR 350 PRIOR TO FL120



KING AIR 300



KING AIR 350 FL121 AND AFTER

Figure 5-4 Fuel Control Panels



FUEL SYSTEM OPERATION

Fuel flow from each wing outer main cell and auxiliary tank system is automatic without pilot action (Figure 5-1). The outer wing cells gravity-feed into the nacelle tank through a line extending from the aft inboard wing cell, forward along the outboard side of the nacelle tank, aft of the firewall immediately under the motive flow valve. A gravity flow check valve in the end of the gravity feed line prevents any backflow of fuel into the outer wing cells.

The supply line from the nacelle tank is routed from the inboard side of the nacelle tank, through a motorized firewall fuel valve installed in the fuel line immediately behind the engine firewall, forward to the engine-driven boost pump.

The firewall fuel valve for each engine fuel system is actuated by its respective alternate action push button switch located on the glareshield (Figure 5-5). When this switch is closed, its respective firewall fuel valve closes to shut off fuel flow to the engine. The bottom half of the switch illuminates L or R FW VALVE CLOSED to indicate valve position. A flashing light indicates the valve is in transit. From the firewall fuel

valve, fuel is routed to the engine-driven boost pump and then to the main fuel filter on the lower center of the engine firewall. The filter incorporates a bypass valve to permit fuel flow in case of plugging, and a drain valve to drain the filter prior to each flight. A pressure switch mounted directly above the filter senses boost pump fuel pressure at the filter. From the main filter, fuel is routed through the fuel heater that utilizes heat from the engine oil to warm the fuel. The fuel is then routed to the high pressure pump and fuel control unit (FCU) that regulates fuel flow to the fuel nozzles. A fuel flow transmitter located adjacent to the FCU sends a signal to an electric DC powered fuel flow gage in the cockpit. It indicates fuel flow in pounds per hour.

FUEL PRESSURE INDICATION

The fuel pressure switch is located directly above the firewall mounted fuel filter and indicates fuel boost pressure. At 9 to 11 psig of decreasing pressure, the switch closes and actuates the red L or R FUEL PRESS light in the warning annunciation panel. With low boost pressure indicated, the electric standby boost pump should be switched on, unless a fuel leak is indicated.

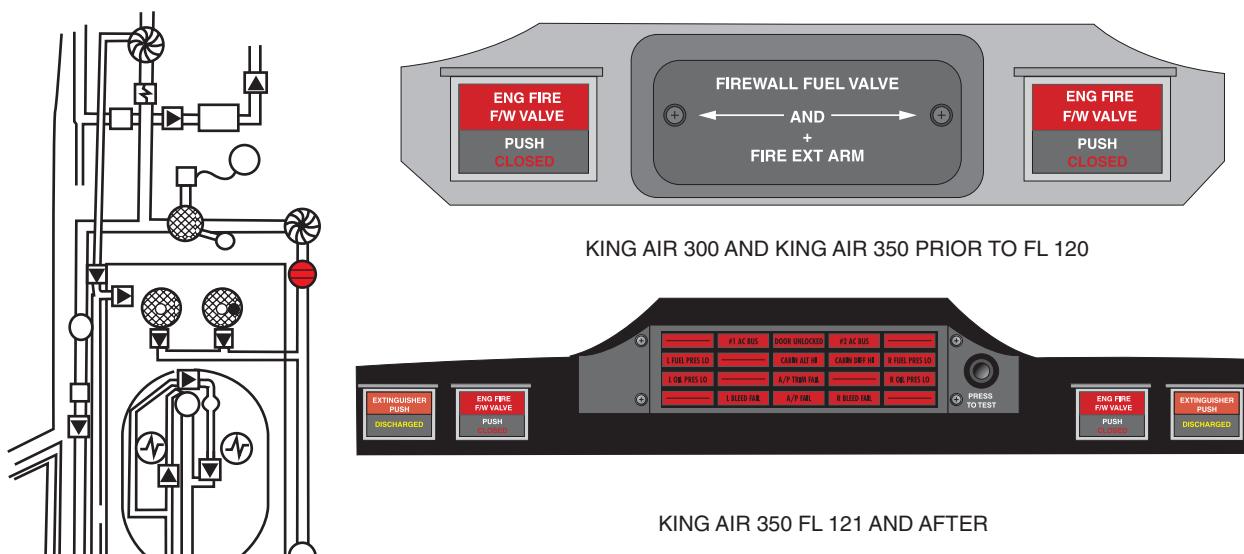


Figure 5-5 Firewall Fuel Valves

**CAUTION**

Operation with the FUEL PRESS light on is limited to 10 hours between overhaul or replacement of the engine-driven high pressure fuel pump. Should both boost pumps fail, suction lift operation may be employed; however, suction lift operation is restricted to 10 hours total time between high pressure pump overhaul periods. If the pump is operated on suction lift beyond the 10-hour limit, overhaul or replacement of the high-pressure pump is necessary. Windmilling time is not equivalent to operation of the engine at high power with respect to the effects of cavitation on fuel pump components. Consequently, windmilling time is not to be included in the 10-hour limit on engine operation without a boost pump.

The red FUEL PRESS light will go out with 9 to 11 psig of increasing fuel pressure.

FUEL PUMPS

Fuel pressure required to operate the engine is provided by an engine-driven high-pressure fuel pump mounted in conjunction with the fuel control unit on the accessory case. This pump is protected against fuel contamination by an internal strainer. The high-pressure fuel pump has an output pressure of up to a maximum of 1050 psi which varies with N_1 rpm and FCU operation.

Fuel is pumped to the high-pressure pump by an engine-driven boost pump also mounted on a drive pad at the aft accessory section of the engine. Any time the gas generator (N_1) is turning, this pump operates to provide sufficient fuel to the high-pressure fuel pump for all conditions except operation with crossfeed or operation with aviation gasoline. This boost pump has an operating capacity of 1250 pounds per hour at a pressure of 30 psi. The engine-driven boost pump is backed up by an electrically-driven standby boost pump located in the bottom of each nacelle tank.

Either the engine-driven boost pump or its backup unit, the electric standby boost pump, is capable of supplying fuel to the engine-driven high pressure fuel pump at the minimum pressure requirements of the engine manufacturer. The engine can operate with the failure of one or both boost pumps; however, failure of the engine-driven high-pressure fuel pump will cause the engine to flame out.

In addition to serving as a backup unit in the event of a malfunction in the engine-driven boost pump, the standby boost pump provides the pressure required for the crossfeed of fuel from one side of the aircraft to the other.

The electrical power with which the standby pumps are operated is controlled by lever-lock toggle switches on the fuel control panel. One source of power to the standby boost pumps is supplied from the respective generator bus which supplies the fuel subpanel circuit breakers. The circuit is protected by two 10-ampere circuit breakers located below the fuel control panel. The other source of power to the standby pumps is the triple-fed bus. A diode network prevents interaction between the two power sources.

FIREWALL FUEL VALVES

The firewall fuel valves (Figure 5-5) are controlled by two alternate action, push button annunciator switches located in the glareshield below each end of the warning annunciator panel. The valves are closed by depressing the switches and opened by depressing the switches again. The L or R FW VALVE CLOSED annunciator in the switch illuminates to indicate that the valve is in the closed position, extinguishes to indicate that the valve is in the open position, and flashes to indicate that the valve is not in the selected position. When either switch is actuated to the closed position, the EXTINGUISHER PUSH (red) annunciator in the corresponding fire extinguisher switch illuminates to indicate that the fire extinguisher is armed regardless of battery switch position.



FUEL TRANSFER MOTIVE FLOW SYSTEM

Fuel pressure from the engine-driven boost pump or the electrical standby boost pump provides the motive flow for the jet transfer pump (Figures 5-6, 5-7, 5-8, and 5-9). The fuel line that supplies the motive flow to the jet transfer pump is routed along the outboard side of the nacelle, through the motive flow valve just aft of the firewall, to the jet transfer pump installed in the sump of the auxiliary tank. A check valve is installed in the motive flow line immediately aft of the motive flow control valve to prevent the engine from taking in air when the boost pump is not operating.

The jet pump transfers fuel from the sump of the auxiliary tank to the nacelle tanks. The transfer

jet pumps are actuated by lever-lock toggle switches on the fuel control panel. This switch may be placed in either the automatic (AUTO) or (OVERRIDE) position.

When the switch is in the AUTO position, power may be applied to the normally closed motive flow valve by the automatic fuel transfer module. When fuel pressure is sensed by the boost pump pressure switch and fuel is sensed in the auxiliary tank by the float switch, power is supplied by the fuel transfer module to the motive flow valve and the motive flow valve should open. Once the motive flow valve has opened, the jet transfer pump will pump fuel from the sump of the auxiliary fuel tank into the nacelle fuel cell for as long as either the engine-driven boost pump or the electrical standby boost pump are operating and there is fuel in the auxiliary tank.

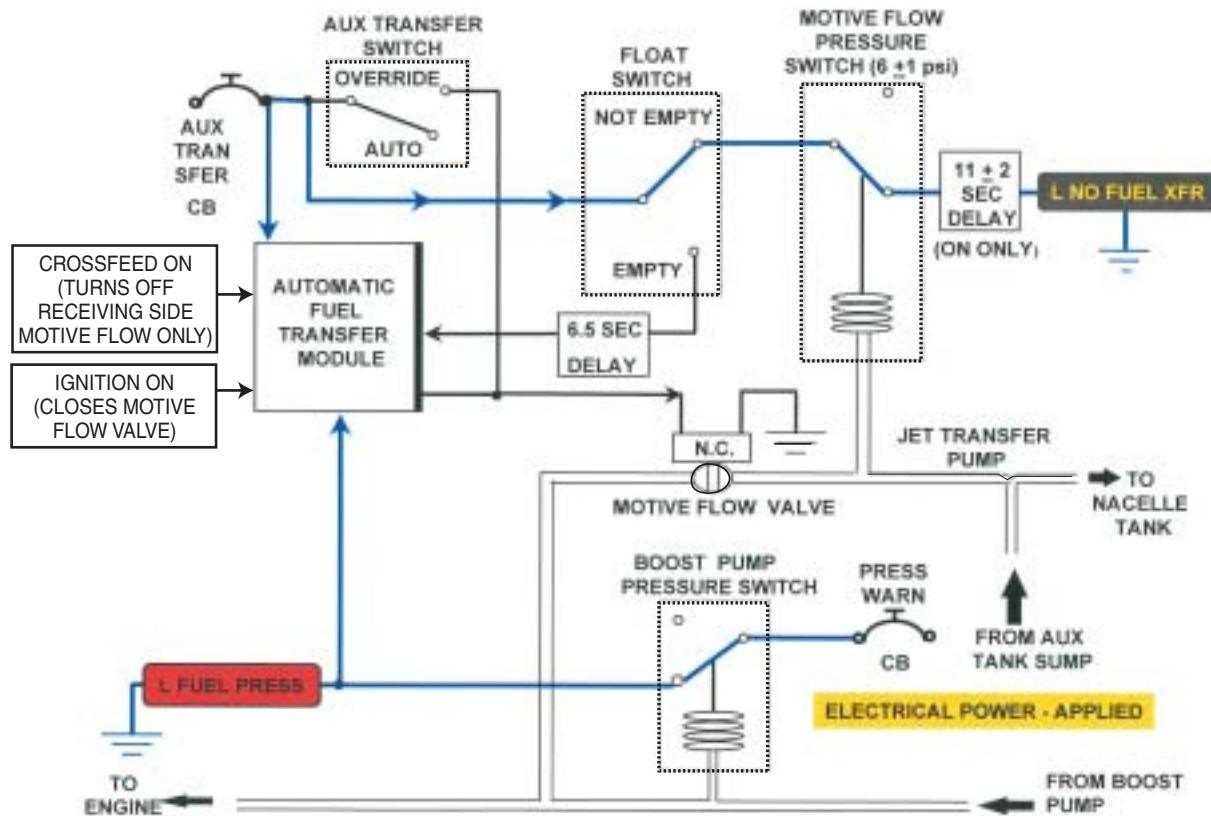


Figure 5-6 Auxiliary Fuel Transfer System

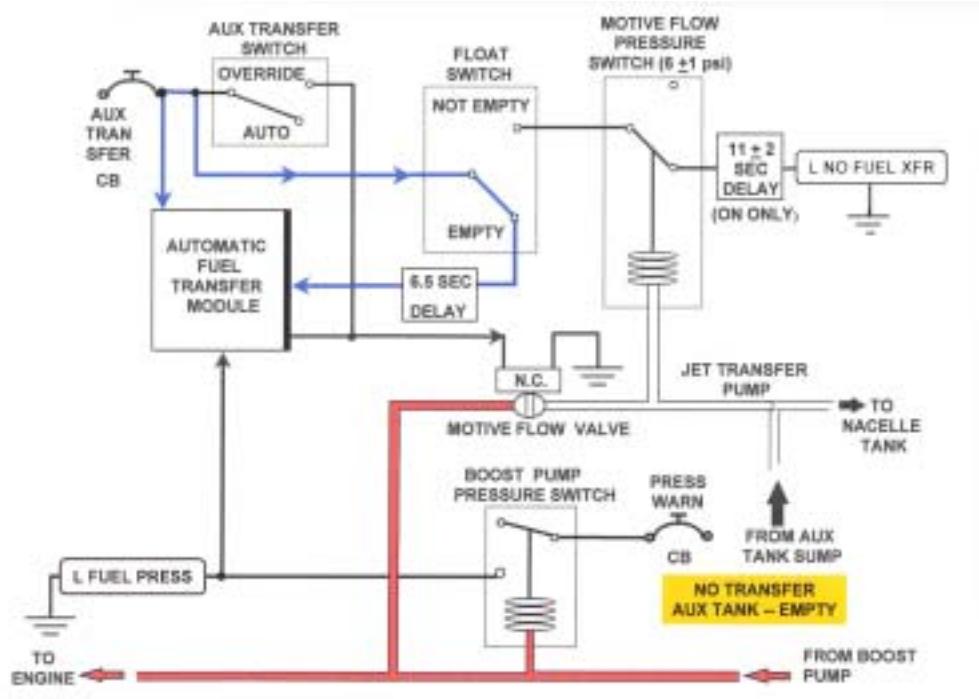


Figure 5-7 Auxiliary Fuel Transfer System—Empty

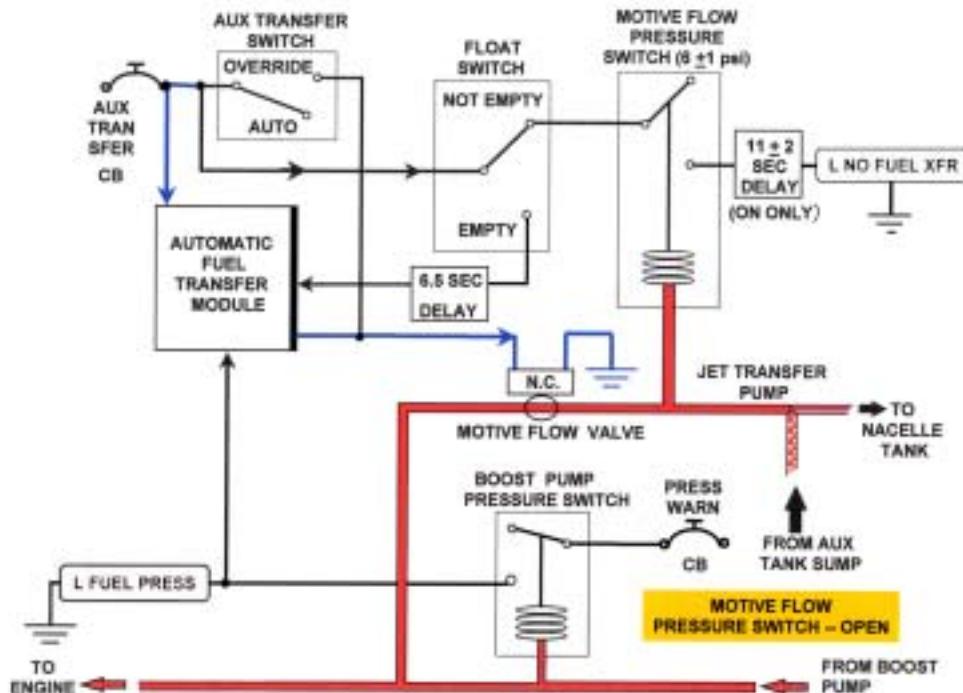


Figure 5-8 Auxiliary Fuel Transfer System—Operating



A motive flow pressure switch set to actuate at 6 (± 1) psi is installed in the fuel line between the motive flow valve and check valve in order to confirm motive flow fuel pressure. A yellow L or R NO FUEL XFR caution annunciator light will illuminate when there is less than 6 (± 1) psi of pressure on the pressure switch adjacent to the motive flow valve, provided the float switch in the auxiliary tank is not sensing an empty tank.

The L or R NO FUEL XFR light has an 11 second delay when it is to be illuminated. This prevents transients from triggering both it and the MASTER CAUTION flasher. With fuel in the auxiliary tank, should this pressure switch not be actuated, the L or R NO FUEL XFR light on the caution annunciator panel will illuminate or remain illuminated to indicate that the motive flow valve is still closed. The auxiliary fuel transfer switch can then be placed in OVERRIDE position to bypass the fuel transfer module and apply power directly to the motive flow valve.

The fuel transfer rate is greater than normal engine fuel consumption. As a result, an overflow

return line is required. The overflow line is plumbed from the nacelle tank back to the auxiliary tank to provide a return for excess fuel delivered by the jet transfer pump. The overflow of fuel from the nacelle tank comes out of an overflow tube at the top of the nacelle tank. It then continues past a 1-1/2 psi pressure relief valve and into a fuel return line to the auxiliary tank.

When the auxiliary fuel is depleted, the float switch sends a signal to the automatic fuel transfer module after a six- to seven-second time delay, the fuel transfer module deenergizes the motive flow valve and it closes. The time delay prevents cycling of the motive flow valve due to sloshing fuel.

The auxiliary fuel system will not feed into the main fuel system if there is a failure of both boost pumps or a failure of the motive flow valve. This condition will be noted on the auxiliary tank fuel gage and by the illumination of the L or R NO FUEL XFER on the caution annunciator panel.

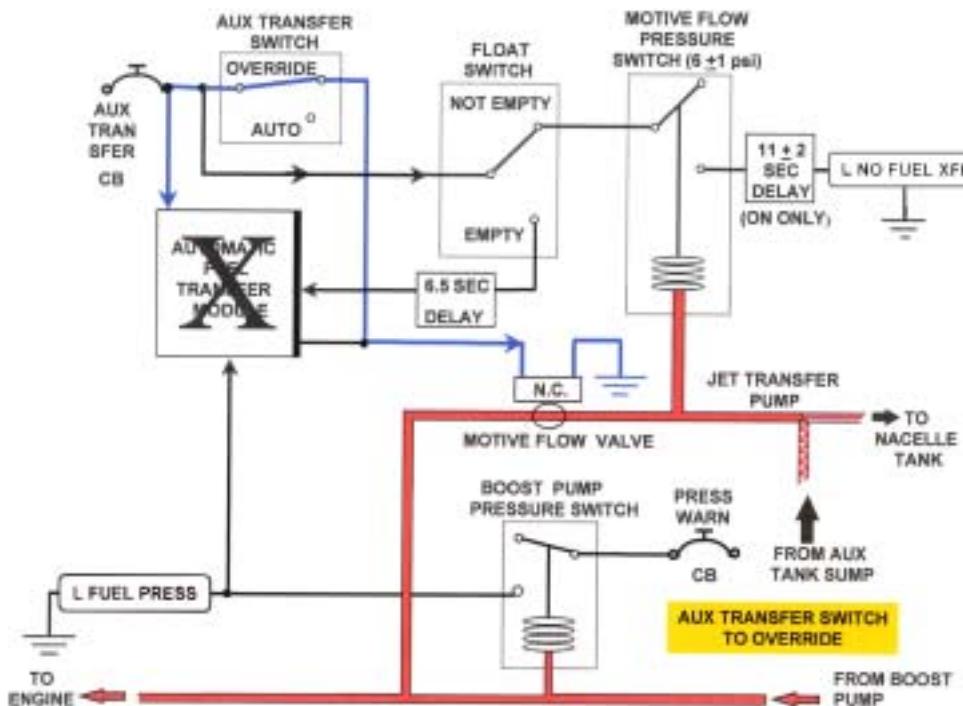


Figure 5-9 Auxiliary Fuel Transfer System—Override



If fuel boost pressure drops below 10 psi (FUEL PRESS light on), the automatic fuel transfer module removes power to close the motive flow valve, thus preventing continued operation of the jet transfer pump. The jet transfer pump will not be damaged by operation after the tank is dry, but extended operation with an empty auxiliary tank may tend to draw unnecessary moist air into the main fuel system from the empty, vented auxiliary tanks.

Any time the engine ignition circuit is powered (either through AUTO IGNITION or the START & IGNITION switch), power will be removed from the motive flow valve by the automatic fuel transfer module. If the system was transferring fuel, the valve should close, resulting in a L or R NO FUEL XFR light.

Selecting crossfeed will cause the fuel transfer module to interrupt electricity to and close the motive flow valve, interrupting fuel transfer on

the side of the operating engine. The appropriate NO FUEL XFER light will also illuminate if there is fuel in the auxiliary tank. Crossfeed has no effect on fuel transfer from the dead engine side.

CROSSFEED OPERATION

Each nacelle tank is connected to the engine on the opposite wing by a crossfeed line routed from the inboard side of the nacelle aft to the center wing section and across to the inboard side of the opposite nacelle. The crossfeed line is controlled by a valve connected into the line at the aft inboard corner of the left nacelle (Figure 5-10).

Crossfeed requires operation of the standby boost pump on the side from which crossfeed is desired to ensure an adequate flow of fuel to the receiving engine and to maintain motive flow for the jet transfer pump on the supply side.

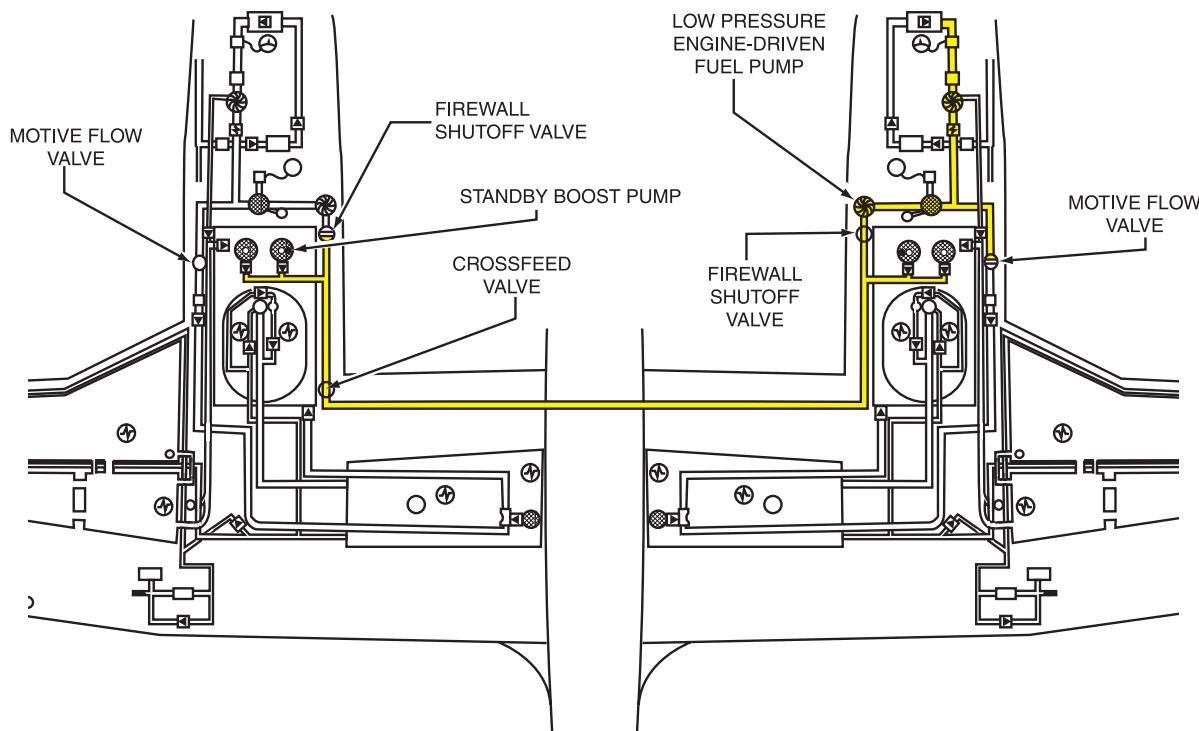
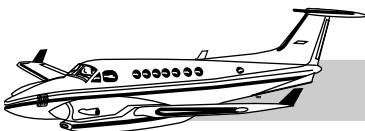


Figure 5-10 Crossfeed Schematic



When the CROSSFEED switch on the fuel control panel is actuated, power is drawn from a five-ampere circuit breaker on the fuel control panel to the solenoid that opens the crossfeed valve.

The automatic fuel transfer module simultaneously energizes the standby pump on the side from which crossfeed is desired and deenergizes (closes) the motive flow valve on the side being crossfed. When crossfeed is selected, the green FUEL CROSSFEED light on the annunciator panel will illuminate. On the 300, the FUEL CROSSFEED annunciator indicates that crossfeed has been selected. On the 350, the FUEL CROSSFEED annunciator indicates that the crossfeed valve has opened. Crossfeed will not transfer fuel from one cell to another; its primary function is to supply fuel from one side to the opposite engine during an engine-out condition. If the standby boost pumps on both sides are operating and the crossfeed valve is open, fuel will be supplied to the engines in the normal manner because pressure on each side of the crossfeed valve will be equal.

When performing crossfeed, be aware of some precautions:

- The auxiliary transfer switch must be in the AUTO position on the side receiving fuel. If the switch were in the OVERRIDE position, the motive flow valve on the receiving engine would remain open. In addition, the incoming fuel would also start filling the tanks on the receiving engine through the auxiliary transfer line which could result in fuel being dumped overboard.
- Both standby pump switches should be in the OFF position. The crossfeed system will automatically turn on the pump it needs to establish crossfeed.
- If the firewall fuel valve was closed on the inoperative engine during shutdown, the fuel pressure annunciator for the engine would remain illuminated and any auxiliary fuel on that side would be unusable due to lack of motive flow pressure.

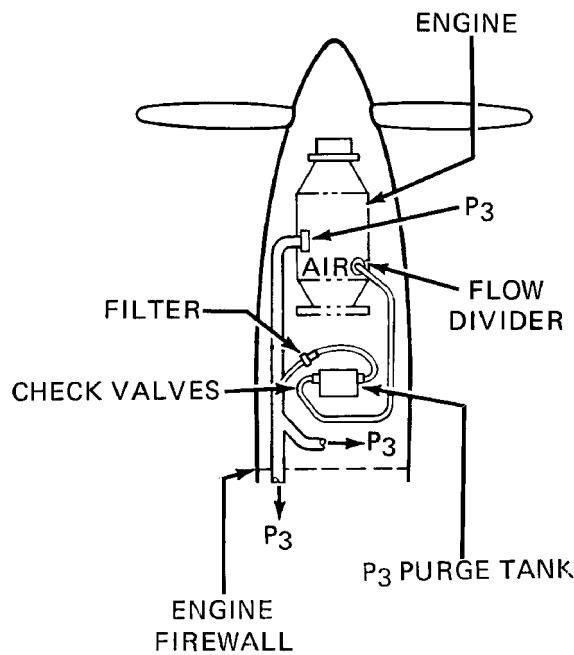


Figure 5-11 Fuel Manifold Purge System Schematic



FUEL MANIFOLD PURGE SYSTEM

This airplane is equipped with a fuel manifold purge system (Figure 5-11), which is designed to assure that any residual fuel in the fuel manifold is consumed during engine shutdown. See the Fuel Manifold Purge System in the Engine section of this workbook for more information.

FUEL GAGING SYSTEM

The airplane is equipped with a capacitance-type fuel quantity indication system (Figure 5-12). A Density Variation of Aviation Fuel graph is provided in the Weight and Balance Equipment List section of the *POH* to allow more accurate calculations of weights for all approved fuels.

The left fuel quantity indicator on the fuel control panel indicates the amount of fuel remaining in the left side main fuel system tanks when the FUEL QUANTITY select switch is in MAIN (center) position, and the amount of fuel remaining in the left side auxiliary fuel tank when the fuel quantity select switch is in AUXILIARY (lower) position. The right fuel quantity indicator indicates the same information for the right side fuel systems, depending upon the position of the fuel quantity switch. The gages are marked in pounds. The switch is spring-loaded to the center position. The TEST (upper) position of the switch provides a test function of the L and R FUEL QTY fiber optic sensing circuitry and caution annunciators.

NACELLE TANK INSTALLATION

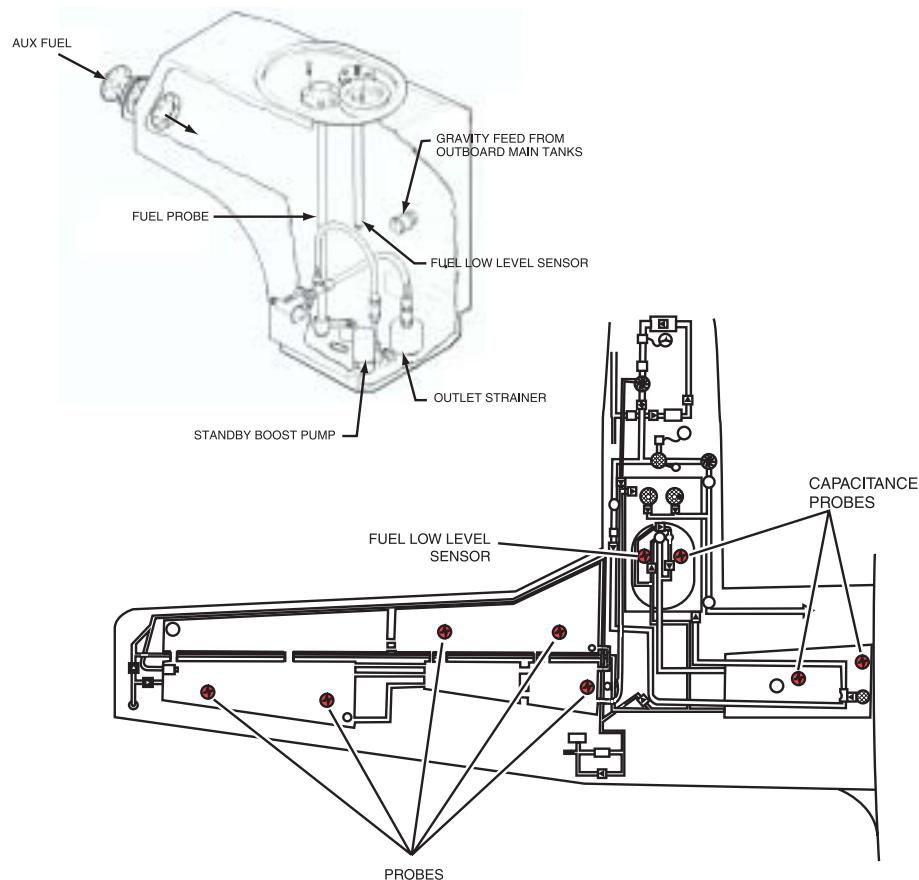


Figure 5-12 Fuel Quantity Indication System



The fuel quantity indicating system is a capacitance type that is compensated for specific gravity and reads in pounds on a linear scale. An electronic circuit in the system processes the signals from the fuel quantity (capacitance) probes in the various fuel cells for an accurate readout by the fuel quantity indicators.

Each side of the airplane has an independent gaging system (Figure 5-12) consisting of a fuel quantity (capacitance) probe in the nacelle fuel cell, one in the aft inboard fuel cell, two in the integral (wet wing) fuel cell, two in the inboard leading edge fuel cell, and two in the center section fuel tank. When the fuel selector switch is in MAIN position, power is supplied from the five-ampere QTY IND circuit breaker through the fuel quantity indicator to the capacitance probes in the main fuel system tanks.

When the fuel selector switch is placed in the AUXILIARY position, power is drawn from the five-ampere circuit breaker to ground through the coil of the indicator switching relay. Power is then supplied through the fuel quantity indicator to the capacitance probes in each auxiliary fuel tank.

Fuel density and electrical dielectric constant vary with respect to temperature, fuel type, and fuel batch. The capacitance gaging system is designed to sense and compensate for these variables. The fuel quantity probe is simply a variable capacitor comprised of two concentric tubes. The tubes serve as fixed electrodes and the fuel of the tank in the space between the tubes acts as the dielectric of the fuel quantity probe.

The capacitance of the fuel quantity probe varies with respect to the change in the dielectric that results from the ratio of fuel to air in the fuel cell. As the fuel level between the inner and outer tubes rises, air with a dielectric constant of one is replaced by fuel with a dielectric constant of approximately two, thus increasing the capacitance of the fuel quantity probe. This variation in the volume of fuel contained in the fuel cell produces a capacitance variation that is a linear function of that volume. This function is converted to linear current that actuates the fuel quantity indicator.

Low Fuel Quantity Indication

In order to alert the pilot of a low fuel situation, fiber optic sensors are installed in both nacelle fuel tanks. When fuel quantity remaining in the main system is below approximately 300 pounds (45 gallons) (30 minutes of fuel remaining at maximum continuous power) on either side, the corresponding L or R FUEL QTY annunciator on the caution annunciator panel will illuminate, which will also trigger the master caution flashers. A five- to seven-second delay is built into the annunciator circuit to reduce the likelihood of fuel sloshing causing transient indications. A holding circuit will keep the annunciators illuminated for three to five seconds once they actually illuminate.

This system may be tested by holding the FUEL QUANTITY switch on the fuel panel in TEST. A simulated low quantity signal is sent to the fiber optic sensor in the nacelle tank, which then illuminates the L and R FUEL QTY annunciators after approximately a five second delay. When the switch is released, it will spring-load back to MAIN (center) and the annunciators will extinguish after approximately five seconds.

FUEL TANK VENTS

The main and auxiliary fuel systems are vented through a recessed vent coupled to a heated ram vent on the underside of the wing adjacent to the nacelle (Figure 5-13). One vent is recessed to prevent icing. The other vent is heated to prevent icing.

Air is vented into or out of the auxiliary fuel cell through a line that extends from the recessed and heated ram vents, through the float check valve to the auxiliary fuel cell.

The wing cells are cross-vented with one another through a float operated vent valve, otherwise known as a vent float valve, on the integral fuel cell. There are four ways for air to get into the wing cells. The first two listed are primary. The last two are applicable in flight only if the first two passages are plugged.



1. A line extends from the recessed and heated ram vents through the leading edge of the wing to the vent float valve on the integral fuel cell.
2. A second passage extends from the recessed and heated ram vents, through the float check valve, through the center of the wing, to the vent float valve on the integral fuel cell.
3. A third passage extends from the air inlet on the underside of the wing near the tip, through a check valve to the vent float valve on the integral fuel cell. This passage is used primarily as a siphon break. It is designed to prevent siphoning of fuel from the auxiliary tank through the wing tip to the recessed and heated ram vents while the aircraft is shut down on the ground.
4. A fourth passage bypasses the vent float valve altogether. It extends from the air inlet on the underside of the wing near the tip, then through a tee and a check valve to the integral fuel cell.

Air is vented into the nacelle tank through a vent float valve installed in the tip of the tank and/or through a tube installed next to the vent float valve. Both the vent float valve and the tube installed next to it, have a check valve downstream to prevent air or fuel from expanding out of the nacelle tank through these passages. Air flows to these passages and into the nacelle tank from the recessed and heated ram vents through the float check valve to a tee, that is just prior to the auxiliary tank, and then through a vent line that leads to the top of the nacelle tank. There is

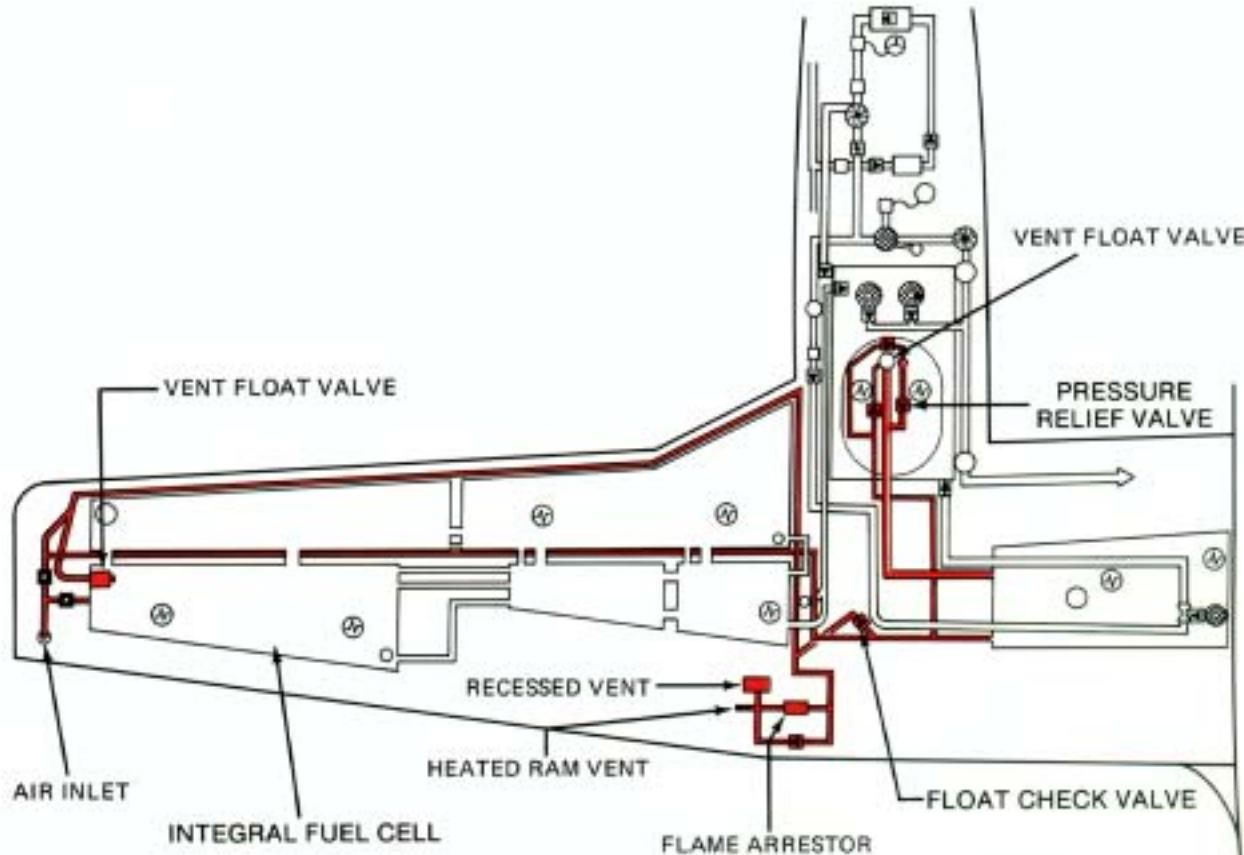


Figure 5-13 Fuel Vents



another tee on top of the nacelle tank that divides this line into the passages that lead to the vent float valve or to the tube installed next to the vent float valve. Air can escape from the nacelle tank through the vent float valve and then through the fuel return line leading to the auxiliary tank. From there the air is vented overboard.

When the aircraft is shut down on the ground and all tanks are full of fuel that is colder than the ambient air temperature, the fuel must expand overboard. Fuel will expand from the wing cells through the gravity feed line to the nacelle tank. Fuel will then expand out the top of the nacelle tank through the fuel return tube, pressure relief valve, and fuel return line to the auxiliary tank.

When fuel expands out of the auxiliary fuel tank to the float check valve, the float closes the check valve, thus preventing some of the excess fuel from being discharged through the vents. When the check valve closes, the auxiliary fuel expands through a line routed outboard from the check valve through the center of the wing to the wing tip. It then continues down the wing leading edge vent line to the recessed and heated ram vents and onto the ground.

When the fuel has expanded fully, the siphon break will prevent continued siphoning of fuel from the auxiliary tank through the wing tip to the recessed and heated ram vents while the aircraft is shut down on the ground.

The vent float valve installed in the integral fuel cell and the vent float valve installed in the top of the nacelle tank are designed to allow air to flow either direction. However, when the fuel level rises up to the level of the vent float valve, the float will rise with the fuel and seal the vent system at that point. This prevents fuel from flowing into the vent system by way of the vent float valve.

In principle, the float check valve accomplishes the same job as the vent float valve, although the design is different. It, like the vent float valve, allows air to pass either direction but does not allow fuel to pass through it. The float check valve looks like an ordinary check valve. Unlike the vent float valve, it is not installed in any tank or cell. It is installed in a short vent line between

the recessed and heated ram vents and the auxiliary tank. When fuel is not present at the float check valve, the float is down and allows air to pass either direction. When fuel is present, the float rises with the fuel and seals the check valve to prevent fuel flow through it.

FUEL DRAINS

During each preflight, the fuel drains on the tanks, lines, and filters should be drained to check for fuel contamination. There are four tank drains, one line drain, and one filter drain in each wing (Figure 5-14). See Table 5-1.

Table 5-1 FUEL DRAIN LOCATIONS

DRAINS	LOCATION
Flush fuel drain	Underside of wing forward of aileron
Gravity line drain	Outboard of nacelle underside of wing
Fuel drain	Outboard of nacelle underside of wing
Strainer drain	Bottom of nacelle
Filter drain	Forward of wheel well
Inboard of fuel tank drain	Underside of wing by wing root

The main and auxiliary fuel systems are equipped with five sump drains, a standby pump drain manifold, and a firewall filter drain in each wing. The drain valve for the firewall fuel filter is located to the right of the filter at the firewall on the underside of the nacelle. The nacelle tank has two drains located on the bottom of the nacelle forward of the wheel well. The inboard drain is for the standby boost pump and the outboard drain is for the nacelle fuel sump and strainer.

The standby pump drain is not to be drained on preflight. The leading edge tank has a drain on the underside of the wing just outboard of the nacelle. The integral (wet wing) fuel tank has a



sump drain located approximately midway on the underside of the wing aft of the main spar. The drain for the auxiliary tank is at the wing root midway between the main and aft spars.

The gravity feed line from the wing tanks to the nacelle tank also has a drain line that extends aft along the outboard side of the main gear wheel well to a drain valve just aft of the wheel well.

Since jet fuel and water are of similar densities, water does not settle out of jet fuel as easily as from aviation gasoline. For this reason, the airplane should sit perfectly still, with no fuel being added, for approximately four hours prior

to draining the sumps for maximum water and fuel separation. However, water and fuel separation will occur soon after fueling or moving the aircraft if there is a substantial amount of water in the fuel. Although turbine engines are not as critical as reciprocating engines regarding water ingestion, water should be removed periodically to prevent formations of fungus and contamination-induced inaccuracies in the fuel gaging system.

When draining the flush-mounted drains, do not turn the draining tool. Turning or twisting of the draining tool will unseat the O-ring seal and cause a leak.

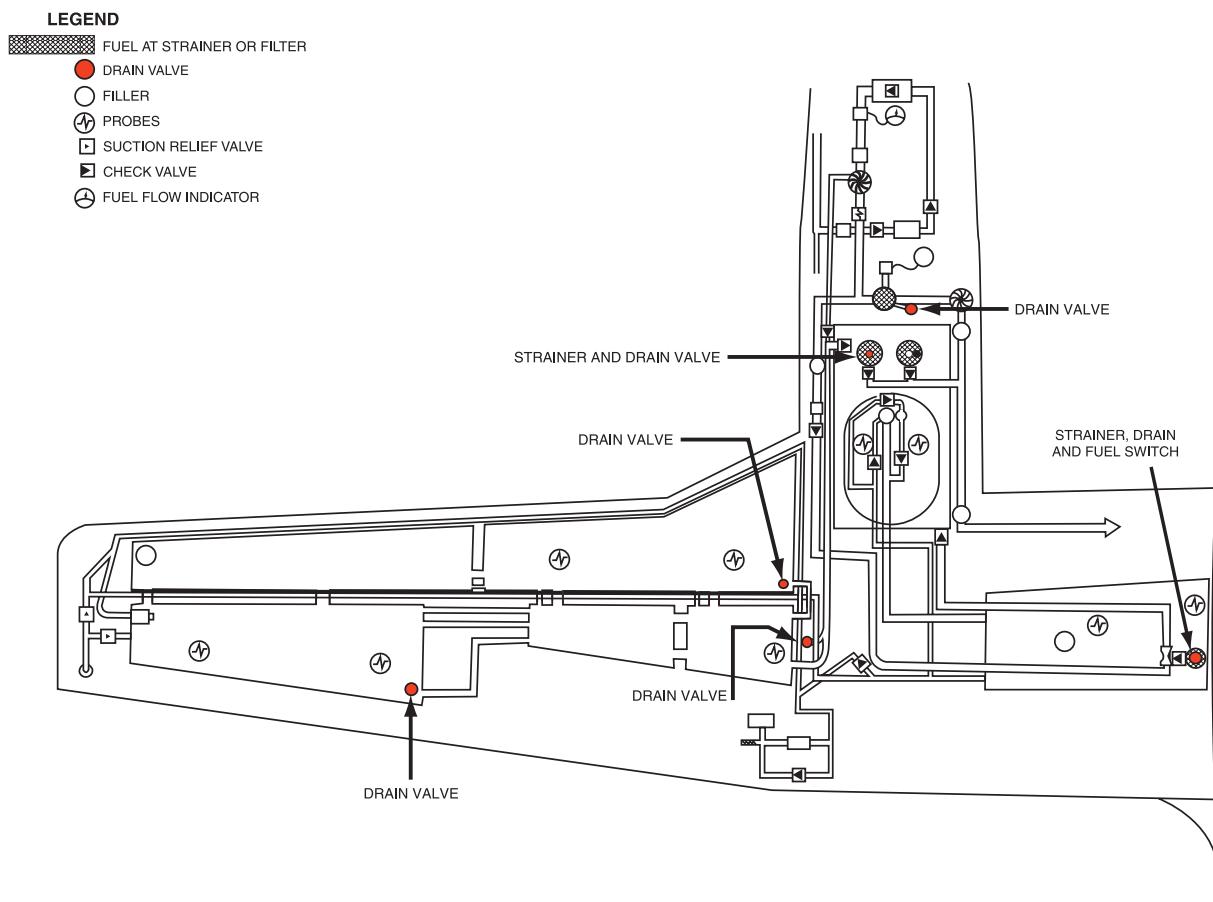


Figure 5-14 Fuel Drain Locations



FUEL SYSTEM LIMITATIONS

The limitations that pertain to the fuel system are briefly summarized below. Refer to the *Pilot's Operating Handbook, Maintenance Manual*, and other specific topics in this section for more details.

1. Operation with a fuel pressure light illuminated is limited to ten hours before overhaul or replacement of the engine-driven high-pressure fuel pump.
2. The King Air 300 maximum zero fuel weight is 11,500. The King Air 350 maximum zero fuel weight is 12,500.
3. Maximum operation with aviation gasoline is limited to 150 hours between engine overhauls. Use of aviation gasoline is limited to 150 hours due to lead deposits which form in the turbine section during aviation gas consumption and cause power degradation. Since the aviation gas will probably be mixed with jet fuel already in the tanks, it is important to record the number of gallons of aviation gas taken aboard. As a rough approximation, it is expected that the PT6A-60A will have an average fuel consumption of 55 gallons per hour per engine, therefore each time 55 gallons of aviation gasoline are added, one hour of the 150-hour limitation is being used for that engine. Consult the *Maintenance Manual* for more details.
4. If the tanks have been serviced with aviation gasoline, operation is prohibited if either standby boost pump is inoperative. The chart found in the Weight and Balance section of the *Pilot's Operating Handbook* shows that the density of aviation gasoline is considerably less than that of jet fuel. Because it is less dense, aviation gasoline delivery is much more critical than jet fuel delivery. Aviation gasoline feeds well under pressure feed but does not feed well on suction feed, particularly at high altitudes. For this reason, two alternate means of pressure feed must be

available for aviation gasoline at high altitude. These two means are the standby boost pump and crossfeed from the opposite side. Thus, a crossfeed capability is required for climbs above 20,000 feet pressure altitude.

5. When fueling the Super King Air 300 or 350, the main fuel tanks should be full before any fuel is put in the auxiliary tanks to reduce the structural bending moment in flight.
6. The Super King Air 300 and 350 have a maximum fuel imbalance of 300 pounds between wing fuel systems.
7. Takeoff is prohibited when the fuel quantity indicator needles are in the yellow arc or when there is less than 265 pounds of fuel in each main system.
8. Crossfeeding of fuel is permitted only when one engine is inoperative.
9. Minimum known or forecast air temperature for operation without fuel anti-icing additive is -45°C .

WARNING

One operative standby fuel pump is required for takeoff when using recommended engine fuels, but in such a case, crossfeed of fuel will not be available from the side of the inoperative standby fuel pump.

FUEL HANDLING PRACTICES

All hydrocarbon fuels contain some dissolved and some suspended water. The quantity of water contained in the fuel depends on temperature and the type of fuel. Kerosene, with its higher specific gravity, tends to absorb and suspend more water than aviation gasoline. Along with the water, it will suspend rust, lint and other foreign materials longer. Given sufficient time, these suspended contaminants will settle to the bottom of the tank.



The settling time for kerosene is five times that of aviation gasoline; therefore, jet fuels require good fuel handling practices to assure that the airplane is serviced with clean fuel. If recommended ground procedures are carefully followed, solid contaminants will settle and free water can be reduced to 30 parts per million (ppm), a value that is currently accepted by the major airlines.

Since most suspended matter can be removed from the fuel by sufficient settling time and proper filtration, it is not a major problem. Dissolved water has been found to be the major fuel contamination problem. Its effects are multiplied in aircraft operating primarily in humid regions and warm climates.

Dissolved water cannot be filtered from the fuel by micronic-type filters, but can be released by lowering the fuel temperature, which will occur in flight. For example, a kerosene fuel may contain 65 ppm (8 ounces per 1,000 gallons) of dissolved water at 80°F. When the fuel temperature is lowered to 15°F, only about 25 ppm will remain in solution. The difference of 40 ppm will have been released as supercooled water droplets which need only a piece of solid contaminant or an impact shock to convert them to ice crystals.

Tests indicate that these water droplets will not settle during flight and are pumped freely through the system. If they become ice crystals in the tank, they will not settle since the specific gravity of ice is approximately equal to that of kerosene. Forty ppm of suspended water seems like a very small quantity, but when added to suspended water in the fuel at the time of delivery, it is sufficient to ice a filter. While the critical fuel temperature range is from 0°F to -2°F, which produces severe system icing, water droplets can freeze at any temperature below 32°F.

Water in jet fuel also creates an environment favorable to the growth of a microbiological "sludge" in the settlement areas of the fuel cells. This sludge, in addition to other contaminants in the fuel, can cause corrosion of metal parts in the fuel system as well as clogging the fuel filters. Although this airplane uses bladder-type fuel cells in addition to an integral (wet wing) fuel

cell in each wing, and all metal parts (except the standby boost pumps and jet transfer pumps) are mounted above the settlement areas, the possibility of filter clogging and corrosive attacks on fuel pumps exists if contaminated fuels are consistently used.

The primary means of fuel contamination control by the owner/operator is "good housekeeping." This applies not only to fuel supply, but to keeping the aircraft system clean. The following is a list of steps that may be taken to recognize and prevent contamination problems:

1. Know your supplier. It is impractical to assume that fuel free from contaminants will always be available, but it is feasible to exercise caution and be watchful for signs of fuel contamination.
2. Assure, as much as possible, that the fuel obtained has been properly stored, that it is filtered as it is pumped to the truck, and again as it is pumped from the truck to the aircraft.
3. Perform filter inspections to determine if sludge is present.
4. Maintain good housekeeping by periodically flushing the fuel tank system. The frequency of flushing will be determined by the climate and the presence of sludge.
5. Aviation gas is an emergency fuel. The 150 hours maximum operation on aviation gasoline per a "Time Between Overhaul" should be observed.
6. Use only clean fuel-servicing equipment.
7. After refueling, allow a settle period of at least four hours whenever possible, then drain a small amount of fuel from each drain.

CAUTION

Remove spilled fuel from the ramp area immediately to prevent the contaminated surface from causing tire damage.



Even if the fuel does not contain water, or you have drained the water out, there is still a possibility of fuel icing at very low temperatures. The oil-to-fuel heat exchanger is used to heat the fuel prior to entering the fuel control unit. Since no fuel temperature measurement is available for fuel prior to the heat exchanger, the temperature must be assumed to be the same as the outside air temperature.

Fuel Grades and Types

Jet A, Jet A-1, Jet B, JP-4,-5, and -8 fuels may be mixed in any ratio. Aviation Gasoline Grades 80/87, 91/96, 100 LL, 100/130 and 115/145 are emergency fuels and may be mixed in any ratio with the normal fuels when necessary. Use of the lowest octane rating available is suggested due to its lower lead content. The use of aviation gasoline shall be limited to 150 hours operation during each Time Between Overhaul (TBO) period.

The Fuel Brands and Type Designations chart in the Handling, Service, and Maintenance section of the *POH* gives fuel refiner's brand name, along with the corresponding designations established by the American Petroleum Institute (API) and the American Society of Testing Material (ASTM). The brand names are listed for ready reference and are not specifically recommended by Beech Aircraft Corporation. Any product conforming to the recommended specification may be used.

Anti-icing Fuel Additive

Engine oil is used to heat the fuel before it enters the FCU. Since no temperature measurement is available for the fuel at the nacelle tank, it must be assumed to be the same as the outside air temperature (OAT).

If the OAT is below -45°C, ice formation could occur during takeoff or in flight. An anti-icing additive per MIL-I-27686 should be mixed with the fuel at refueling to ensure safe operation. Refer to the *POH* and *Maintenance Manual* for procedures to follow when blending anti-icing additive with the airplane fuel.

Fuel Biocide Additive

Fuel biocide-fungicide "BIOBOR JF" in concentrations of 135 ppm or 270 ppm may be used in the fuel. BIOBOR JF may be used as the only fuel additive, or it may be used with the anti-icing additive conforming to MIL-I-27686 specification. Used together, the additives have no detrimental effect on the fuel system components.

See the *Maintenance Manual* for concentrations to use and for procedures for adding BIOBOR JF to the airplane fuel.

Filling the Tanks

When filling the aircraft fuel tanks, always observe the following:

1. Make sure the aircraft is statically grounded to the servicing unit and to the ramp.
2. Service the main tanks on each side first. The main filler caps are located in the outboard fuel cell on the leading edge of each wing near the wingtip. The auxiliary filler caps are located on top of the center section, inboard of each nacelle.
3. Allow a four-hour settling period whenever possible, then drain a sufficient amount of fuel from each drain point to remove water and contaminants.

Defueling the Aircraft

As an integral part of the nacelle fuel tank, a defueling adapter, located aft of the standby pump, contains a check valve to prevent fuel drainage when the plug is removed. Each wing fuel system may be drained as follows:

1. Remove the cover on the bottom of the nacelle to gain access to the adapter plug.
2. Remove the plug and screw the long end of an AN832-12 union into the adapter. The fuel will begin draining as the check valve is unseated by the union.



3. The fuel may be gravity-drained or, to facilitate defueling, pumped out with the aid of a fuel truck.

Refer to the *Maintenance Manual* for more details.



The material normally covered in this chapter is not applicable to this airplane.

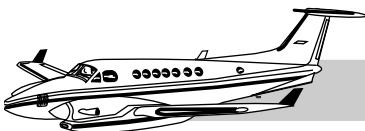


CHAPTER 7

POWERPLANT

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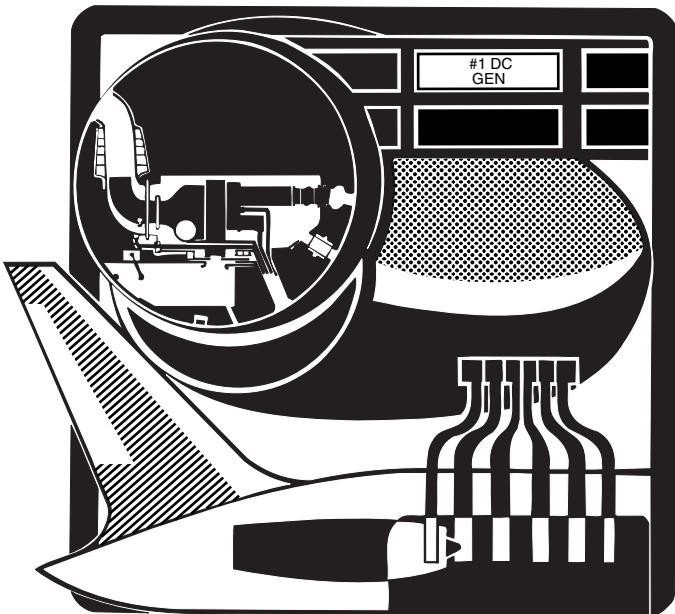


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CHAPTER 7

POWERPLANT



INTRODUCTION

In-depth knowledge of the powerplants and propeller systems is essential to good power management by the pilot. Operating within the design parameters of the powerplant and propeller systems extends engine life and ensures safety. This section describes the basic components of the engines and their limits and checks.

POWERPLANT DESCRIPTION

The powerplant section of the training manual presents a description and discussion of the Pratt & Whitney Canada PT6A-60A turboprop engines. These engines will be described in sufficient detail for flight crewmembers to understand normal operational practices and limitations.

The purpose of this section is to provide the flight crew with sufficient details of the engine operation to familiarize themselves with normal, abnormal, and emergency procedures.



SUPER KING AIR 300 AND 350 POWERPLANT

The powerplant for the Super King Air 300 and 350 is the Pratt and Whitney of Canada PT6A-60A free-turbine-turboprop engine driving a four-bladed propeller (Figures 7-1 and 7-2). It is flat-rated to 1,050 shaft horsepower.

The powerplants are equipped with conventional, four-blade, full-feathering, reversing, constant-speed propellers, mounted on the output shaft of the engine reduction gearbox. The propeller pitch and speed are controlled by engine oil pressure through single-action, engine-driven propeller governors. The propellers will feather automatically when the engines are shut down and will unfeather when the engines are started.

When reference is made to the right or left side of the airplane or engine, it is always looking from the rear to the front (Figure 7-3).

TURBOPROP ENGINE RATINGS

In turboprop engines, power is measured in equivalent shaft horsepower (ESHP) and shaft horsepower (SHP). SHP is determined by propeller rpm and torque applied to turn the propeller shaft. The hot exhaust gases also develop some kinetic energy as they leave the engine, similar to a turbojet engine. This jet thrust amounts to about 10% of the total engine horsepower. ESHP is the term applied to horsepower total delivered, including the jet thrust. Turboprop engine specifications usually show both ESHP and SHP, along with limiting ambient temperatures. Figure 7-4 shows the engine rating and temperatures.

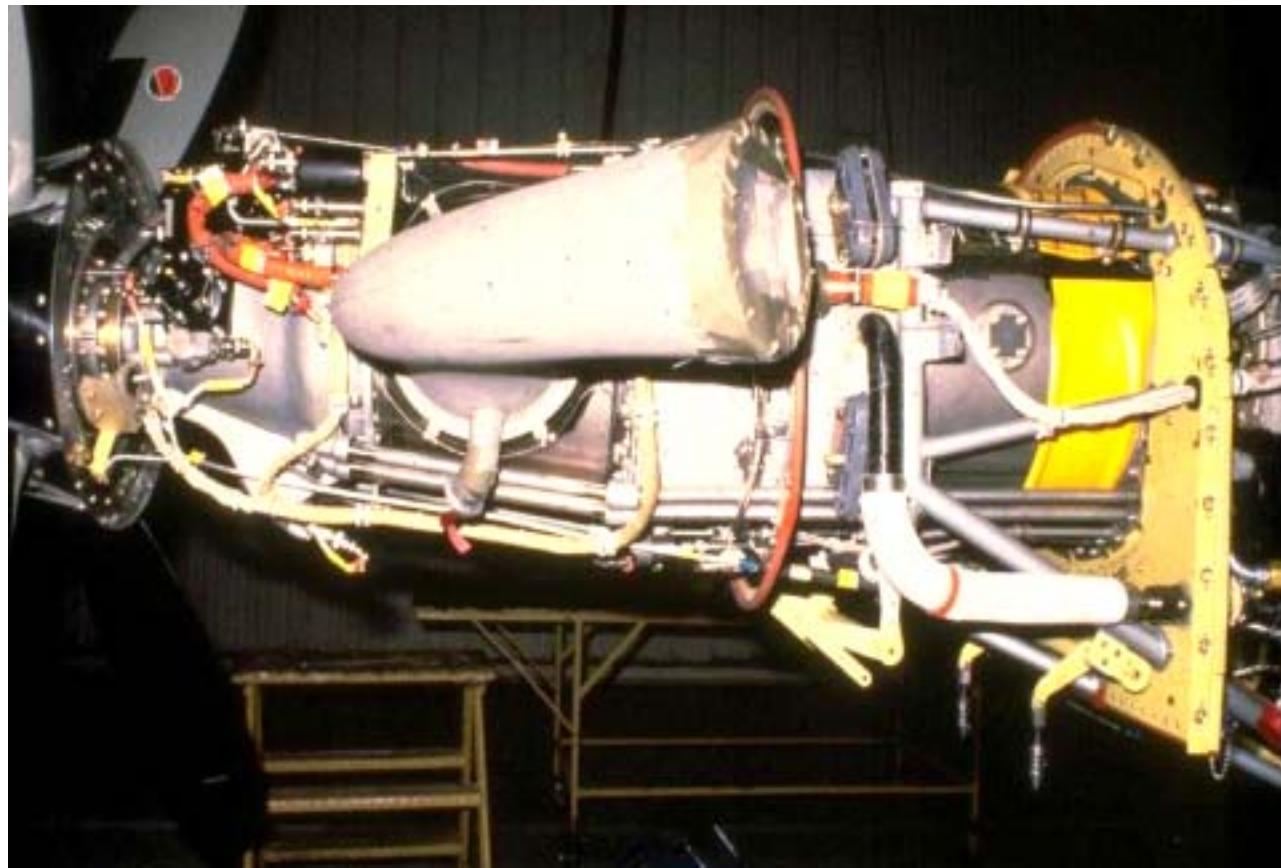
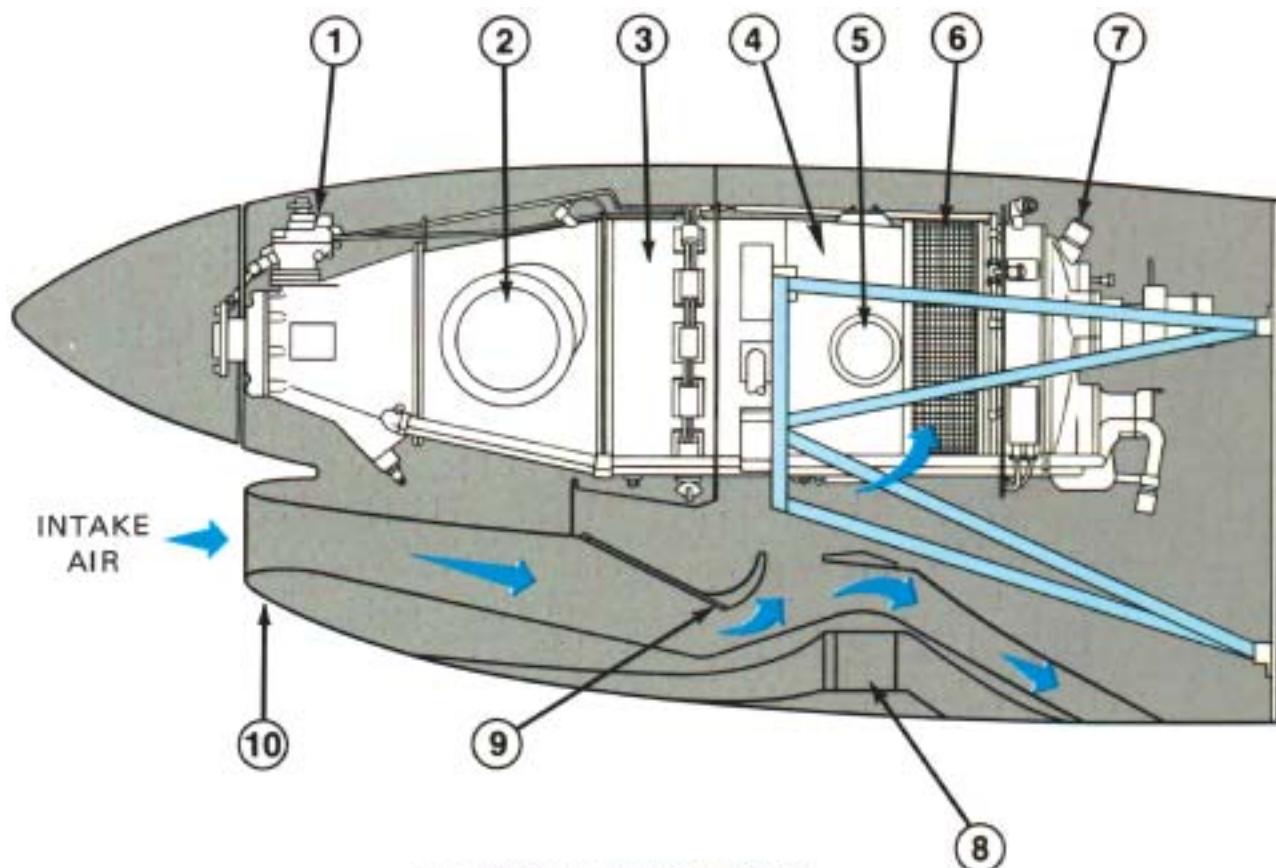


Figure 7-1 PT6A-60A Powerplant Installation



1. PROPELLER GOVERNOR
2. EXHAUST
3. COMBUSTION CHAMBER
4. COMPRESSOR SECTION
5. BLEED VALVE
6. ENGINE AIR INLET
7. OIL FILLER AND DIPSTICK
8. ENGINE OIL COOLER
9. INERTIAL SEPARATOR VANE
10. INLET LIP HEAT (EXHAUST)

Figure 7-2 PT6A-60A Specifications

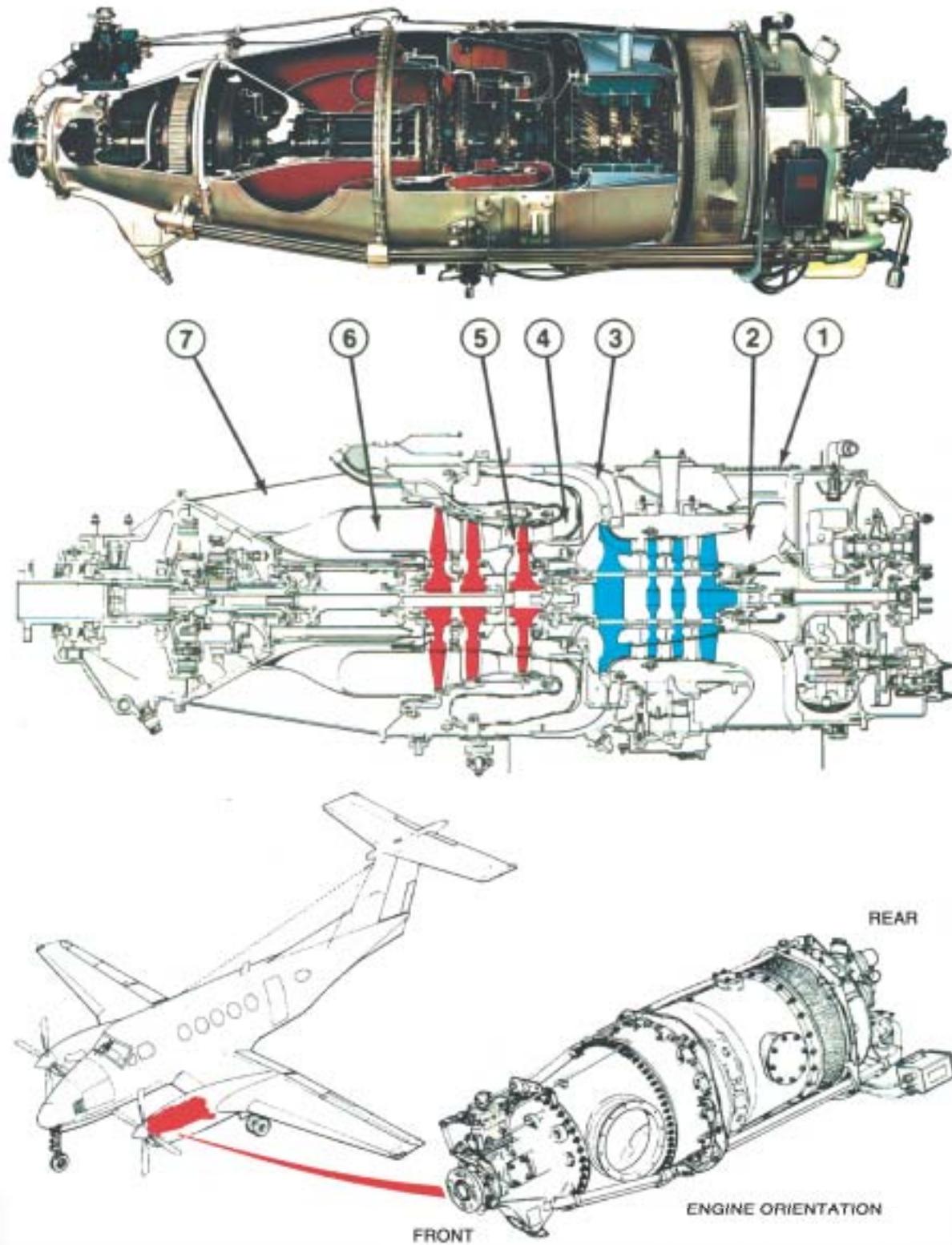


Figure 7-3 Engine Cutaway



(Sea Level Static ICAO Standard Atmosphere Conditions)

OPERATING CONDITION	ESHP	SHP	PROPELLER RPM (a)	JET THRUST (lbs.)	JP-4 FUEL CONSUMPTION (lb./eshp/hr.) at 15°C (59°F)
Takeoff	1113*	1050	1700	158	0.548
Max. Continuous	1113*	1050	1700	158	0.548
Max. Climb +	1060***	1000	1700	150	0.557
Max. Cruise +	1060**	1000	1700	150	0.557
Normal Cruise +	1060***	1000	1700	150	0.557

* Available to 25.0°C (77°F)

** Available to 28.3°C (83°F)

*** Available to 23.3°C (74°F)

+ For information only; not certification ratings

(a) Corresponding Rotor Speed: Power Turbine - 29,920 RPM

Engine Type	Free Turbine
Type of Combustion Chamber	Annular
Compression Ratio	9:1
Compressor Shaft Rotation (looking forward)	Counterclockwise
Propeller Shaft Rotation (looking forward)	Clockwise
Propeller Shaft Gear Ratio	17.6:1
Oil Consumption, Maximum Average	0.2 lb./hr. (0.0907 kg/hr.)
Gas Generator Speed 100 percent N ₁	37,500 RPM
Max. Continuous Gas Generator Speed 104 percent N ₁	39,000 RPM

Figure 7-4 PT6A-60A Specifications

ENGINE STATIONS

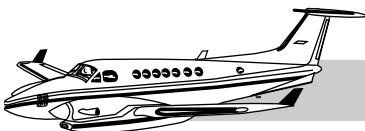
To identify points in the engine, it is common practice to establish engine-station numbers at various points. To refer to pressure or temperature at a specific point in the engine airflow path, the appropriate station number is used, such as P₃ for the Station 3 pressure or T₅ for the gas temperature at Station 5. For instance, temperature of the airflow is measured between the compressor and first stage power turbine at engine station number 5. This is called inter-stage turbine temperature (ITT) or T₅. Bleed air is taken off the engine after the centrifugal com-

pressor stage and prior to entering the combustion chamber. This air, commonly referred to as P₃ air, is used for cabin heat, pressurization, and the pneumatic system.

ENGINE TERMS

To properly understand the operation of the PT6A series engines, there are several basic terms you should know:

- N₁ or N_g—Gas generator rpm in percent of turbine speed.



- N_p —Propeller rpm.
- N_2 or N_f —Power turbine rpm (not indicated on engine instruments).
- $P_{2.5}$ —Air pressure between engine stations 2 and 3. Also referred to as axial stage air or compressor interstage air.
- P_3 —Air pressure at engine station 3, the source of bleed air used for some aircraft systems.
- ITT or T_5 —Interstage turbine temperature in degrees centigrade at engine station 5.

Review and remember these terms. They will be used often to describe PT6A engines.

FREE-TURBINE REVERSE-FLOW PRINCIPLES

The PT6A-60A engine consists basically of a free-turbine, reverse-flow engine, driving a propeller through planetary gearing (Figure 7-5). The term “free-turbine” refers to the design of the turbine sections of the engine. There are two turbine sections: one, called the compressor turbine, drives the engine compressor and accessories; and the other, consisting of dual power turbines, drives the power section and propeller. The power turbine section has no physical connection to the compressor turbine.

The compressor and power turbines are mounted on separate shafts and are driven in opposite directions by the gas flow across them (Figure 7-6). The term “reverse flow” refers to airflow through the engine. Inlet air enters the compressor at the aft end of the engine, moves forward through the combustion section and the turbines, and is exhausted at the front of the engine.

ENGINE MODULAR FREE-TURBINE DESIGN

An important feature of the PT6A-60A engine is its modular concept. With the modular free-tur-

bine design, the engine is basically divided into two modules, a gas generator section and a power section (Figure 7-7). The gas generator section includes the compressor and the combustion section. Its function is to draw air into the engine, add energy to it in the form of burning fuel, and produce the gases necessary to drive the compressor and power turbines.

The power section’s function is to convert the gas flow from the gas generator section into mechanical action to drive the propeller. This is done through an integral planetary gearbox, which converts the high speed and low torque of the power turbine to the low speed and high torque required at the propeller. The reduction ratio from power turbine shaft rpm (N_f) to propeller rpm (N_p) is approximately 17.6:1.

The engine requires a minimum of maintenance. A hot section inspection (HSI) is usually carried out at mid-TBO. This involves splitting the engine between the compressor and power turbines. Since it is not necessary to remove the engine from the airplane to carry out the HSI, the inspection is both simple and fast. Due to the modular design, the gas generator section or the combustion section can be completely replaced independently of the other section. This design permits easy maintenance, modular overhaul, and on-wing HSI.

ENGINE AIRFLOW

Another important feature of the PT6A-60A engine is the reverse flow design. Inlet air enters the rear of the engine through an annular plenum chamber, formed by the compressor inlet case, where it is directed forward to the compressor (Figure 7-5). The compressor consists of three axial stages combined with a single centrifugal stage, assembled as an integral unit on a common shaft.

A row of stator vanes, located between each stage of compression, diffuses the air, raises its static pressure, and directs it to the next stage of compression. The compressed air passes through diffuser tubes, which turn the air through 90° in direction and convert velocity to static pressure. The diffused

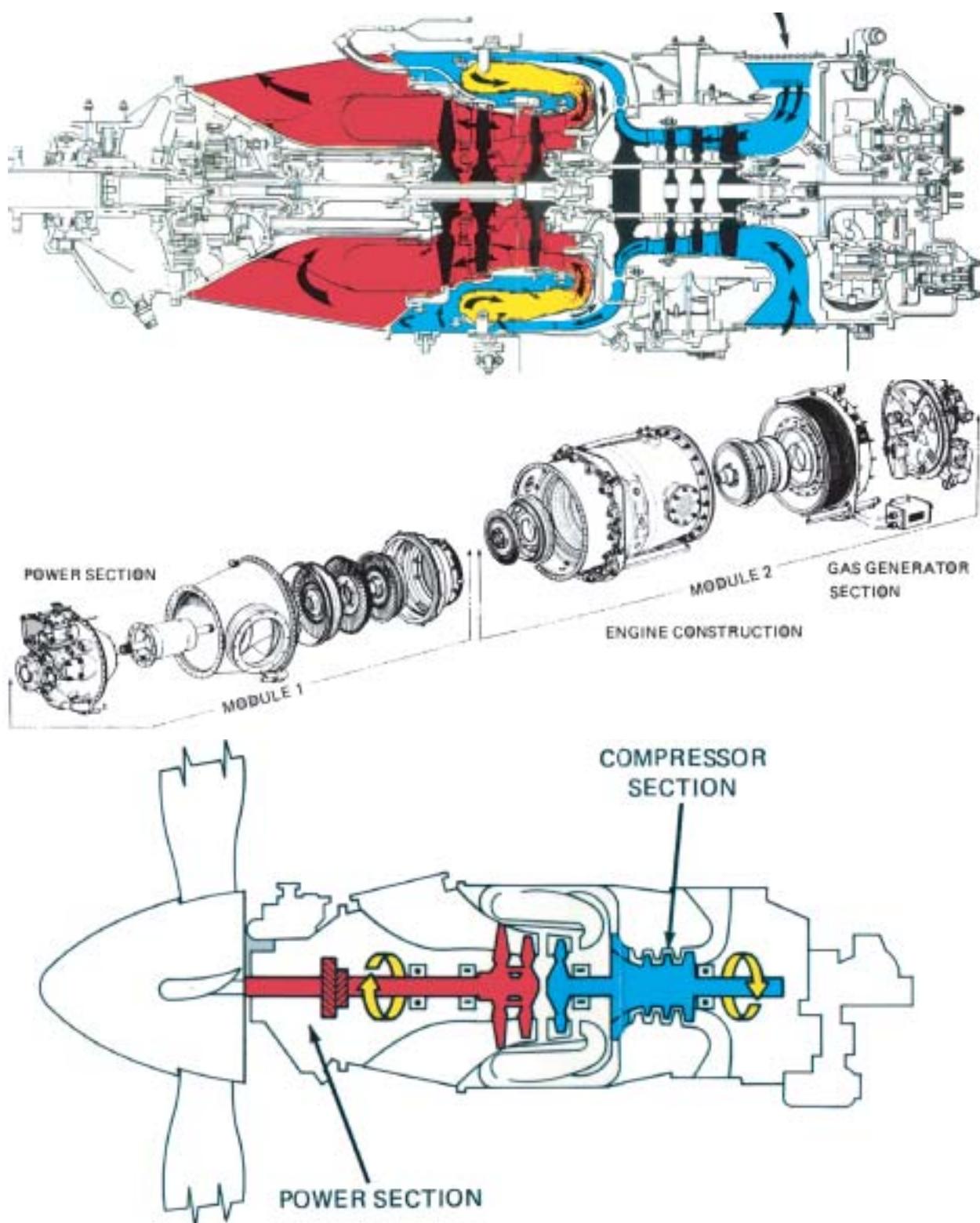
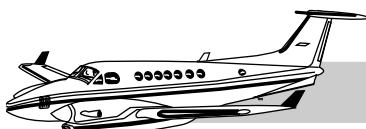


Figure 7-5 Engine Gas Flow and Stations

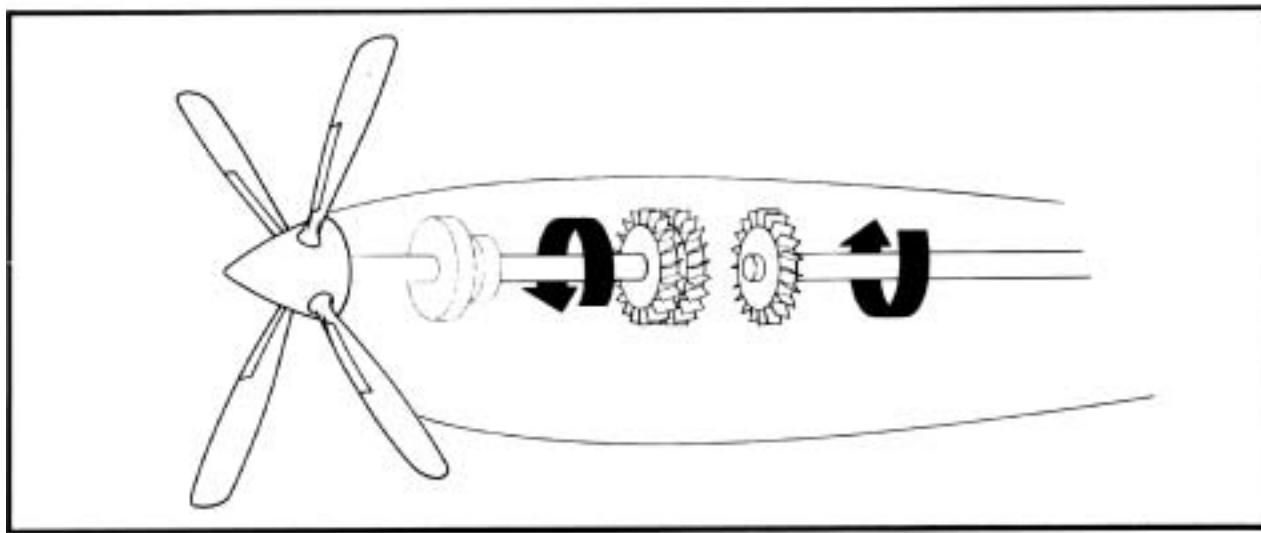


Figure 7-6 Compressor and Turbine Rotation

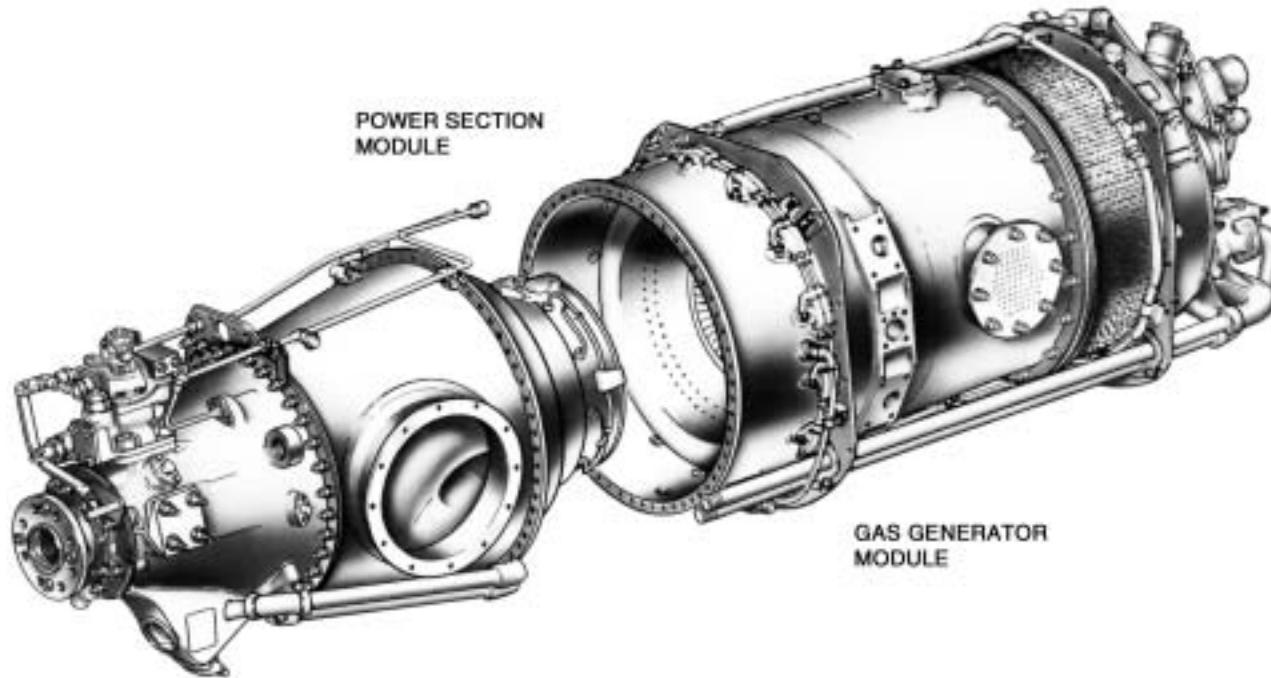


Figure 7-7 Engine Modular Concept



air then passes through straightening vanes to the annulus surrounding the combustion chamber liner.

The flow of air changes direction 180° as it enters and mixes with fuel in the combustion chamber. The combustion chamber liner has varying size perforations, which allow entry of compressor delivery air. Approximately 25% of the air mixes with fuel to support combustion. The remaining 75% is used to center the flame in the combustion chamber can and for internal cooling of the engine. The fuel/air mixture is ignited, and the resultant expanding gases are directed to the turbines. The unique location of the combustion chamber liner, utilizing flow reversal, eliminates the need for a long shaft between the compressor and the compressor turbine, thus reducing the overall length and weight of the engine.

For ease of starting, fuel is injected into the combustion chamber liner through 14 simplex nozzles arranged in two sets, which are supplied by a dual fuel manifold consisting of primary and secondary transfer tubes and adapters. For starting only, the fuel/air mixture is ignited by two spark igniters, which protrude into the liner. After starting, the igniters are turned off since combustion is self-sustaining.

The resultant gases expand from the liner, reverse direction in the exit duct zone, and pass through the compressor turbine inlet guide vanes to the single-stage compressor drive turbine. The guide vanes ensure that the expanding gases impinge on the turbine blades at the correct angle with minimum loss of energy. The expanding gases are then directed forward to drive the power turbine section.

Approximately 60% of the energy extracted from the combustion gases is extracted by the compressor turbine with the balance taken by the power turbines. The dual-stage power turbine, consisting of inlet guide vane and turbines, drives the propeller shaft through a reduction gearbox.

The compressor and power turbines are located in the approximate center of the engine, with their respective shafts extending in opposite directions. This feature simplifies the installation and inspection procedures. The exhaust

gas from the power turbines is directed through an annular exhaust plenum to the atmosphere through twin opposed exhaust ports provided in the exhaust duct.

COMPRESSOR BLEED VALVE

At low N_1 rpm, the compressor axial stages produce more compressed air than the centrifugal stage can use. A compressor bleed valve compensates for this excess airflow at low rpm by overboarding, or bleeding, axial-stage air ($P_{2.5}$) to reduce back pressure on the axial stages (Figures 7-5 and 7-8). This pressure relief helps prevent compressor stall of the axial stages.

The compressor bleed valve is a pneumatic piston which references the pressure differential between the axial and centrifugal stages. Looking forward, the valve is located at the 3 o'clock position. The function of this valve is to prevent compressor stalls and surges in the low N_1 rpm range.

At low N_1 rpm, the valve is open. At takeoff and cruise, above approximately 90% N_1 rpm, the bleed valve will be closed. If the compressor bleed valve were to stick closed at low N_1 speeds, compressor stalls would result as the engine attempts to accelerate to takeoff power. If the valve were to stick open at high N_1 speeds, ITT would be higher than normal and torque considerably lower than normal (at similar N_1 rpm), thereby reducing power output as the engine becomes temperature-limited at reduced torque. Therefore, at both low speeds and high speeds, proper compressor bleed valve operation is critical to normal engine operation.

JET-FLAP INTAKE SYSTEM

A unique feature of the PT6A-60A engine is its efficient utilization of $P_{2.5}$ air. In most other PT6A engines, it is ported overboard. In this engine, it is incorporated in a "jet flap" system. A jet flap, or slot, is machined into one side of each hollow strut, which secures the accessory section to the compressor section of the engine.

The jet flap intake system (Figure 7-8) functions as a variable inlet guide vane without variable geometry. Each hollow core provides a passage-

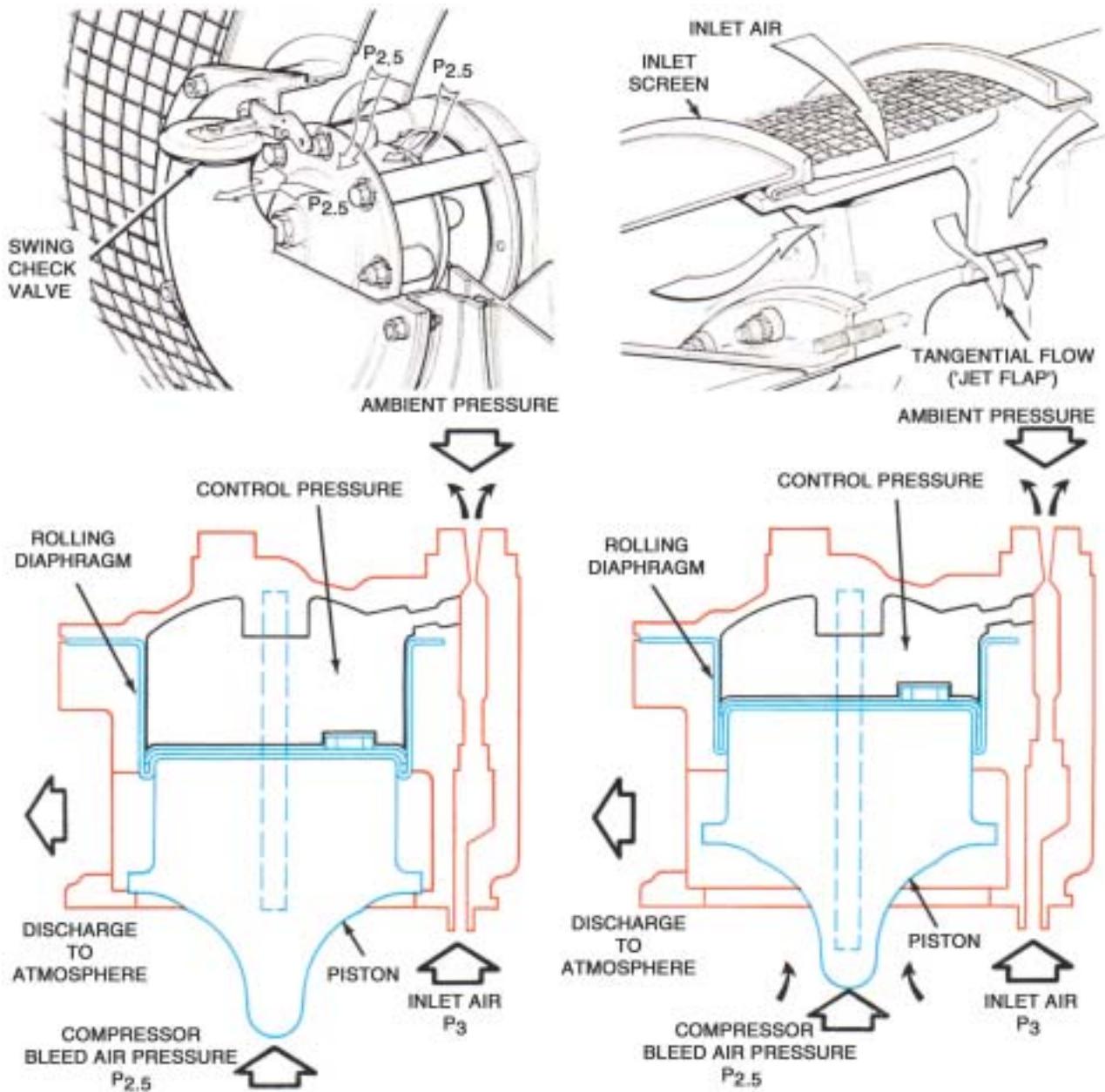
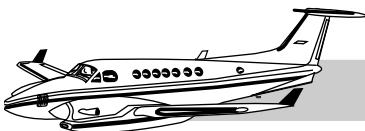


Figure 7-8 Jet-Flap, Compressor Bleed Valve, and Swing Check Valve

way for compressor interstage air ($P_{2.5}$) to exit through the narrow slot that is parallel to the engine centerline. The interstage air passing into the intake zone provides a swirl effect on inlet air entering the compressor. This “pre-swirl” effect improves low speed compressor characteristics, and allows the elimination of one of the compres-

sor bleed valves found in most of the PT6A series engines.

SWING CHECK VALVE

A swing check valve is located on the right side at the 3 o’clock position on the compressor bleed



valve cover. It is a plate valve hinged at the upper edge and capable of pivoting through approximately 90 degrees. The valve relieves excess P_{2.5} pressure that can't be used by the jet flap system when the compressor bleed valve is open.

IGNITERS

The combustion chamber has two spark-type igniters to provide positive ignition during engine start. These igniters are activated when starting as the IGNITION AND ENGINE START switch is moved to ON the position. While the engine is equipped with two igniters, it needs only one to start. The system is designed so that if one igniter is open or shorted, the remaining igniter will continue to function.

Once the engine is started, the igniters may be turned off when the IGNITION AND ENGINE START switch is moved to OFF since the combustion is self-sustaining.

The spark ignition provides the engine with an ignition system capable of quick light-ups over a wide temperature range. The system consists of an airframe-mounted ignition exciter, two individual high tension cable assemblies, and two spark igniters. It is energized from the aircraft nominal 28-volt DC supply, and will operate in the 9- to 30-volt range. The igniter control box produces up to 3,500 volts.

The engine start switches are located on the pilot's left subpanel (Figure 7-9). This subpanel contains the IGNITION AND ENGINE START switches and ENG AUTO IGNITION switches. The IGNITION AND ENGINE START switches have three positions: ON, OFF, and STARTER ONLY. The ON position activates both the starter and igniters. The STARTER ONLY position is a hold-down position, spring loaded to center (OFF). It provides motoring only, to clear the engine of unburned fuel. With the switch in this position, there is no ignition.

The ignition system features an automatic backup function for emergencies. This backup system is called AUTO IGNITION. The auto ignition switches should be moved to the ARM position in turbulence, precipitation, and icing

conditions. When auto-ignition is armed, if engine torque falls below approximately 17%, the igniters will automatically energize, and the IGNITION ON annunciators will illuminate. Therefore, the igniters would automatically attempt a restart had the engine flamed out during one of the above conditions.

ACCESSORY SECTION

All of the engine-driven accessories, except the propeller tachometer and propeller governors, are mounted on the accessory gearbox located at the rear of the engine (Figures 7-10 and 7-11). The accessories are driven from the compressor shaft (N₁) through a coupling shaft. The lubricating and scavenge oil pumps are mounted inside the accessory gearbox. Two scavenge pumps are externally mounted. The starter/generator, high pressure fuel pump, N₁ tachometer generator, and other optional accessories are mounted on pads on the rear of the accessory drive case. There are several such mounting pads, each with its own different gear ratio (Figure 7-12).

LUBRICATION SYSTEM

The PT6A engine lubrication system has a dual function (Figure 7-13). Its primary function is to cool and lubricate the engine bearings and bushings. Its second function is to provide oil to the propeller governor and propeller reversing control system. The main oil tank houses a gear-type engine-driven pressure pump, an oil pressure regulator, and an oil filter. The engine oil tank is an integral part of the compressor inlet case and is located in front of the accessory gearbox.

As oil is pumped from the tank, it passes through the pressure and temperature sensing bulbs mounted on or near the rear accessory case. The oil then proceeds to the various bearing compartments and nose case through an external oil transfer line below the engine. Scavenge oil returns from the nose case and the bearing compartments through the gear-type oil scavenge pumps in the accessory case through external oil transfer lines and through the external oil cooler below the engine.



The oil cooler is thermostatically controlled to maintain the desired oil temperature. Another externally mounted unit, the oil-to-fuel heat exchanger, uses hot engine oil to heat fuel before it enters the engine fuel system to prevent icing at the pump filter.

The oil tank is provided with a filler neck and integral quantity dipstick housing. The cap and dipstick are secured to the filler neck, which passes through the gearbox housing and accessory diaphragm and into the tank. The markings on the dipstick indicate the number of U.S. quarts of oil less than full (Figure 7-14).

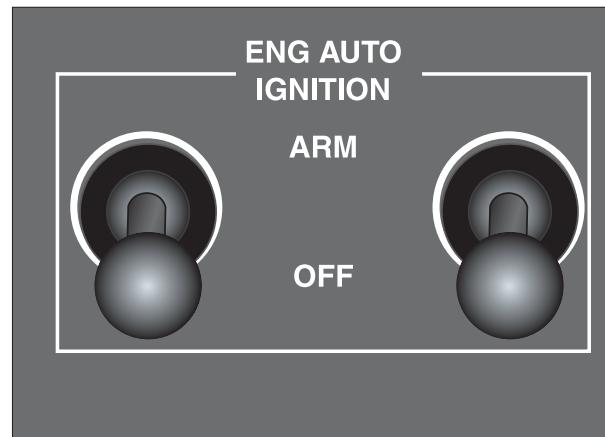
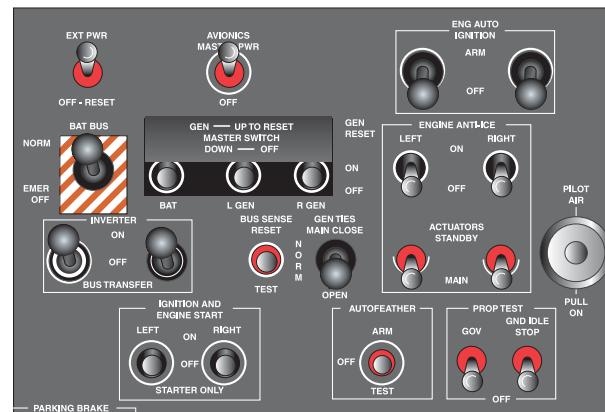
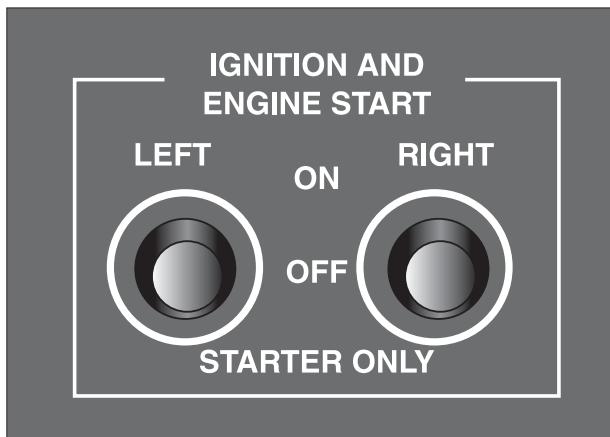
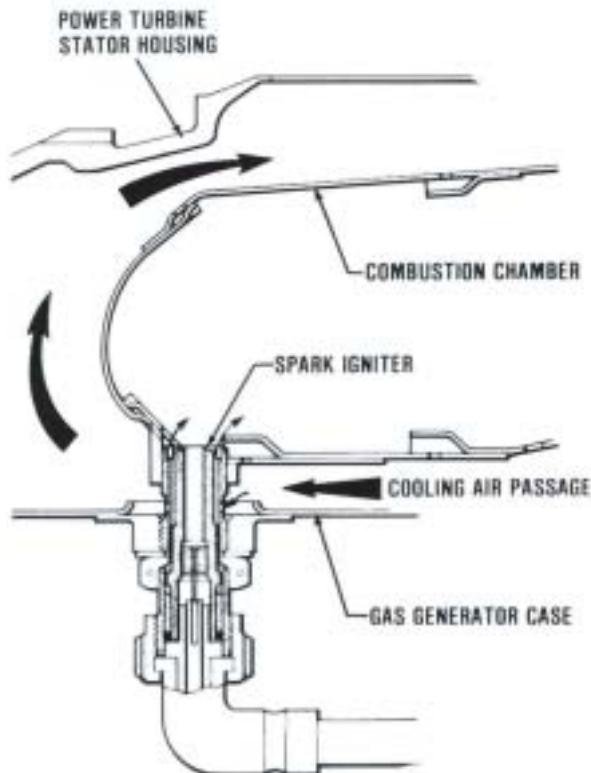


Figure 7-9 Engine Start and Ignition Switches

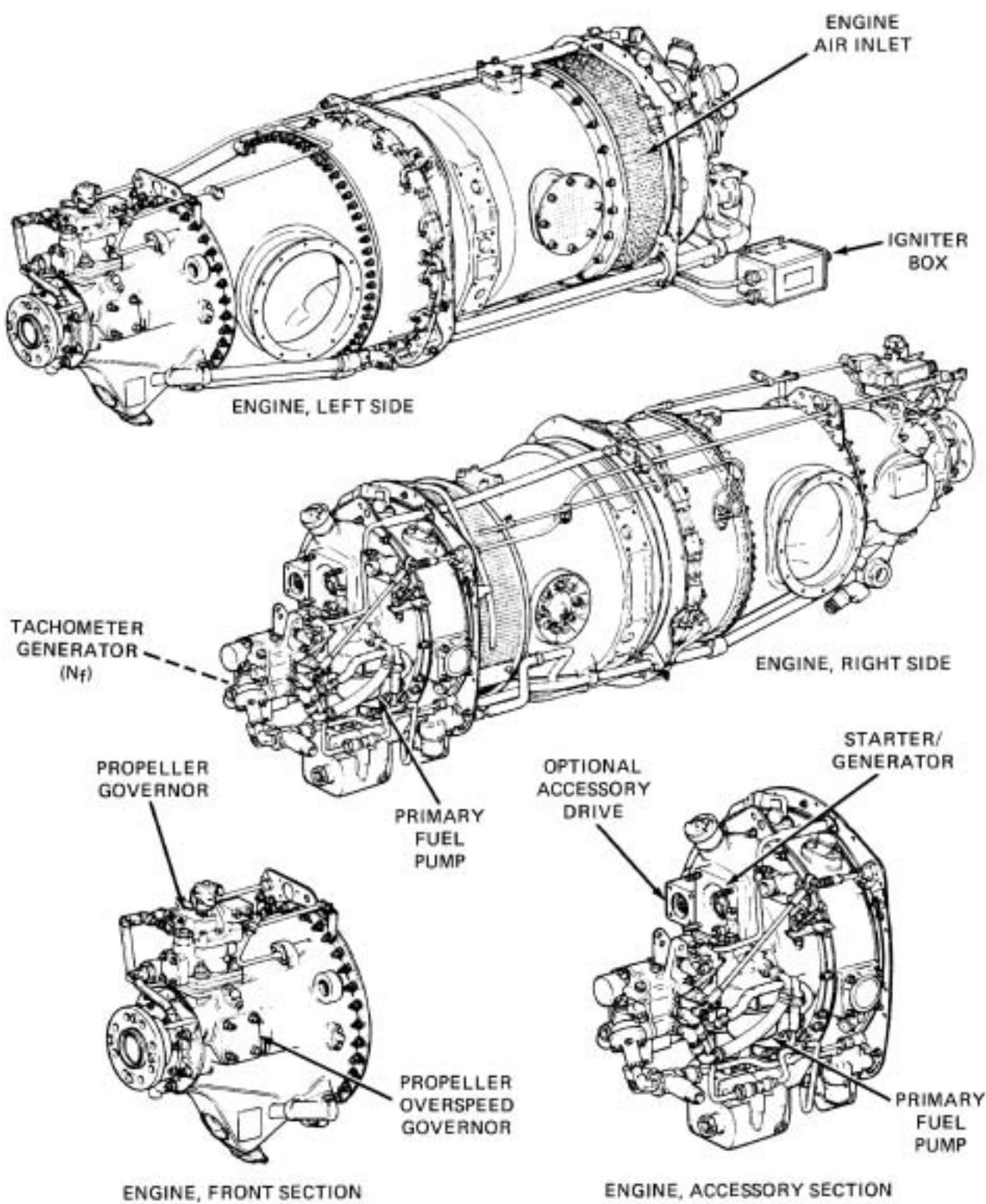
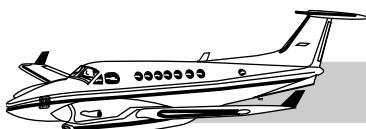
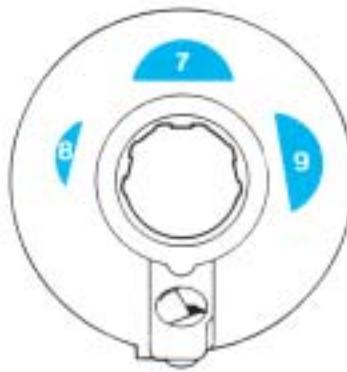


Figure 7-10 Typical PT6A Engine



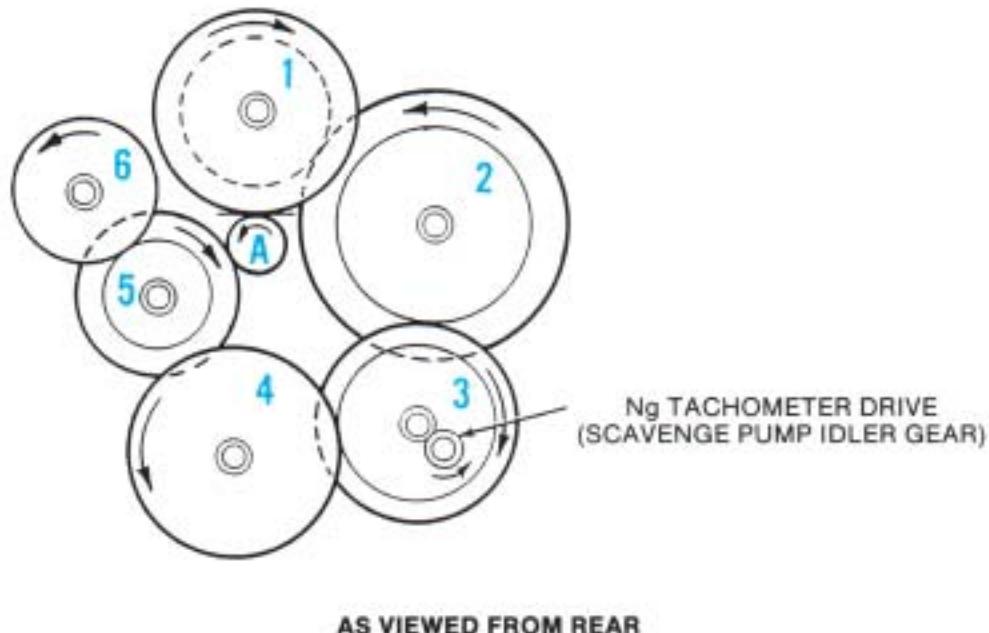
AS VIEWED FROM REAR
REAR ACCESSORY DRIVES
(100% N_g = 37,468 rpm)



AS VIEWED FROM FRONT
FRONT ACCESSORY DRIVES
(100% N_f = 29,894 rpm)

DRIVE PAD	ROTATION	RATIO	MAXIMUM TORQUE (in. lb.)		MAX. SPEED (RPM)
			CONTINUOUS	STATIC	
1 Starter-generator	CW	0.2931:1	170	1600	10980
2 Fuel Pump/FCU	CCW	0.1670:1			6260
3 Tachometer-generator (N_g)	CCW	0.1121:1	7	100	4200
4 Engine Driven Boost Pump	CCW	0.1019:1	60	800	3818
5 Optional Accessory Drive	CW	0.3207:1	135	800	12015
6 Optional Accessory Drive	CCW	0.2041:1	150	800	7650
7 Propeller Governor	CW	0.1405:1	50	850	4200
8 Tachometer-generator (N_f)	CW	0.1405:1	7	100	4200
9 Propeller Overspeed Governor	CW	0.1405:1	50	850	4200

Figure 7-11 Front and Rear Accessory Drive



DRIVE	RATIO	MAX. RPM
A Input (100% Ng)	1:1	37,500
1 Starter-Generator	0.2931:1	10,991
2 Fuel Pump/FCU	0.167:1	6,262
3 Internal Oil Scavenge and Pressure Pumps and Tacho-Generator	0.1121:1	4,203
4 External Oil Scavenge and Engine Driven Boost Pump	0.1019:1	3,821
5 Optional Accessory	0.3207:1	12,028
6 Optional Accessory	0.2041:1	7,654

Figure 7-12 Accessory Gearbox Geartrain

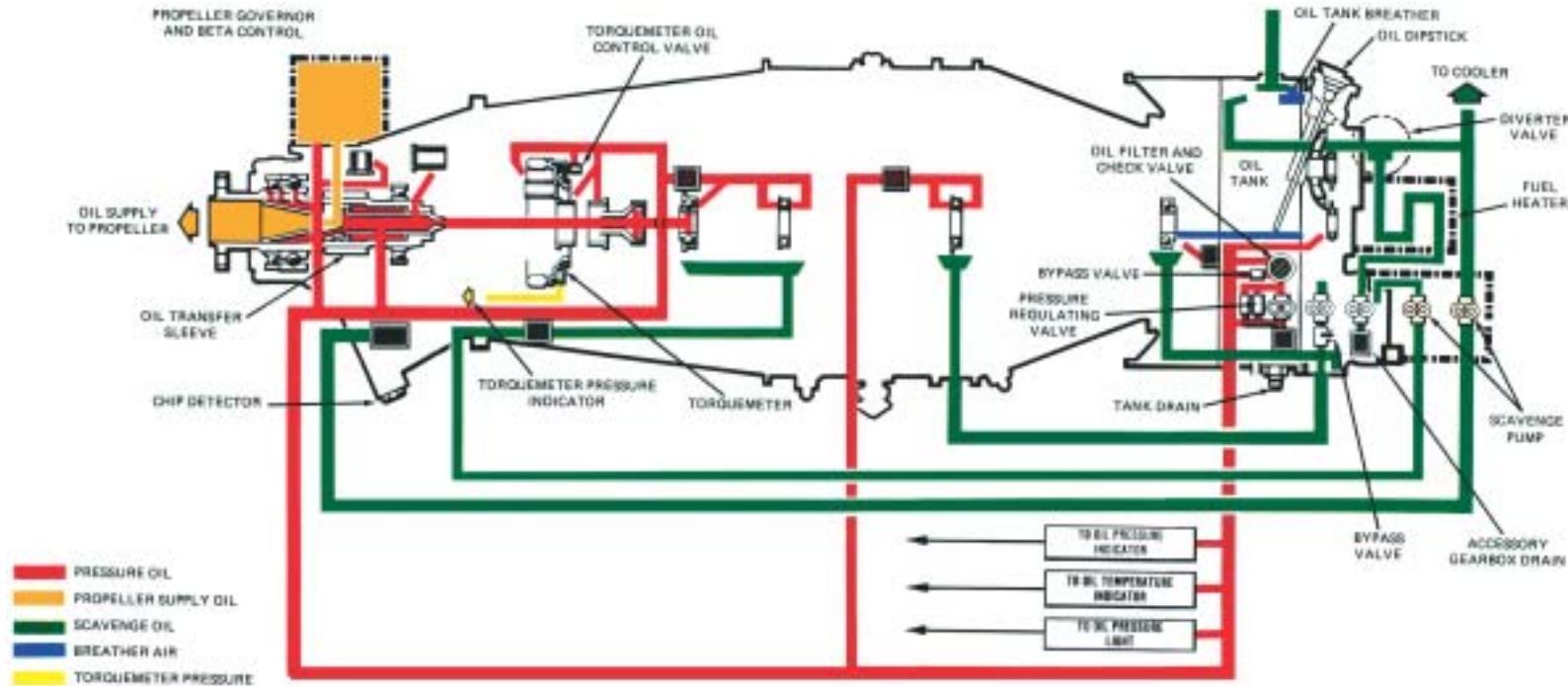
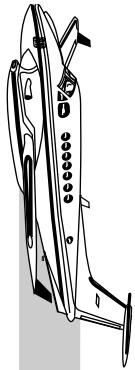


Figure 7-13 Engine Lubrication Diagram

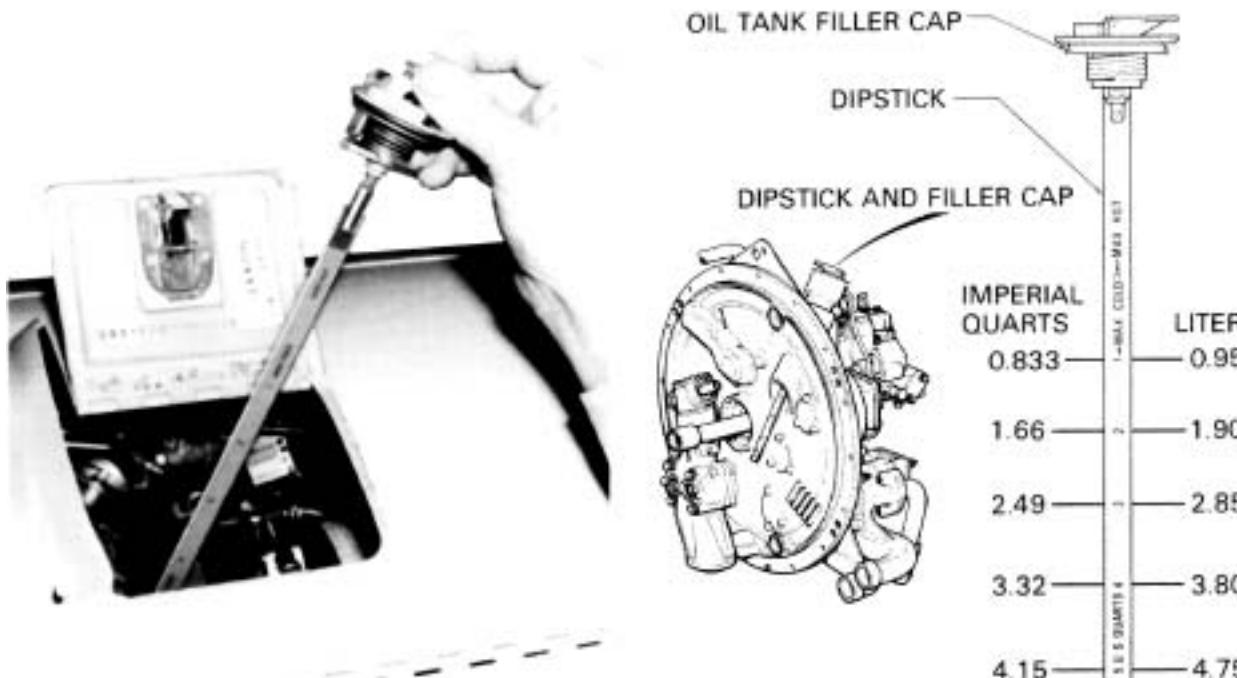


Figure 7-14 Engine Oil Dipstick

The engine oil system has a total capacity of four U.S. gallons including the 2.5 gallon oil tank. Maximum oil consumption is one quart every ten hours of operation. Normal oil consumption may be as little as one quart per fifty hours of operation.

Most PT6A engines will normally seek an oil level of from one to two quarts down on the dipstick with hot oil, and approximately one quart lower than that when the oil is cold. Do not overfill. When adding oil between oil changes, do not mix types or brands of oil due to the possibility of chemical incompatibility and loss of lubricating qualities. A placard inside the engine cover shows the brand and type of oil used in that particular engine.

Although the preflight checklist calls for checking the oil level, which is required, the best time to check oil quantity is shortly after shutdown, since oil levels are most accurately indicated at that time. Oil level checks during preflight may require motoring the engine for a brief period for an accurate level reading.

When gas generator speeds are above 72% N₁, normal oil pressure is between 90 and 135 psi.

Magnetic Chip Detector

A magnetic chip detector is installed in the bottom of each engine nose gearbox (Figure 7-15). This detector will activate a yellow light on the annunciator panel, L CHIP DETECT or R CHIP DETECT, to alert the pilot of possible oil contamination indicating possible or pending engine failure.

Illumination of a CHIP DETECT annunciator indicates possible metal contamination in the engine oil supply. Illumination of a CHIP DETECT annunciator is not in itself cause for an engine to be shut down. Engine parameters should be monitored for abnormal indications.

If parameters are abnormal, a precautionary shut down may be made at the pilot's discretion. After illumination of a CHIP DETECT annunciator, cause of the malfunction should be determined and corrected prior to the next flight.

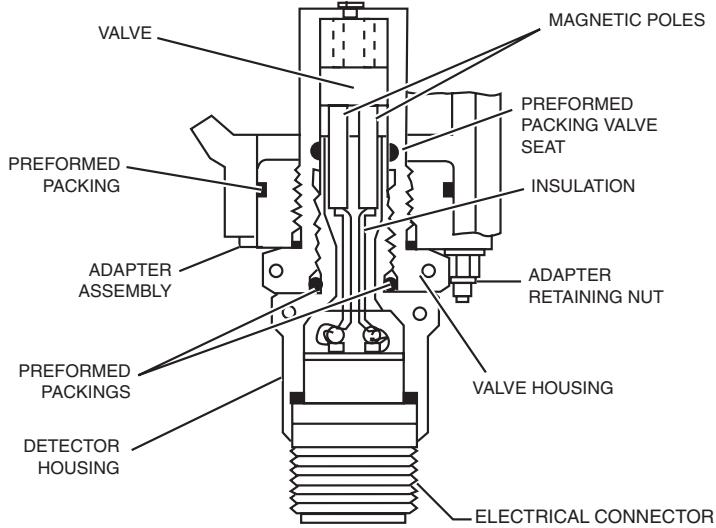
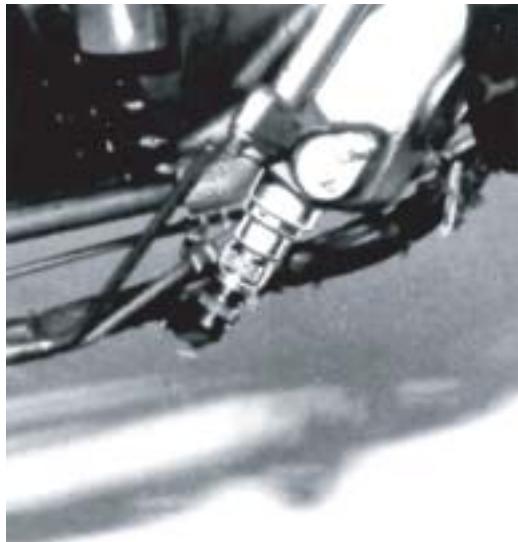


Figure 7-15 Magnetic Chip Detector

ENGINE FUEL SYSTEM

The fuel control system for PT6A-60A engines is a fuel governor that increases or decreases fuel flow to the engine to maintain selected engine operating speeds. The engine fuel control system consists of the main components shown in the block diagram (Figure 7-16). They are the engine driven boost pump, firewall fuel filter and pressure switch, oil-to-fuel heat exchanger, high-pressure fuel pump, fuel control unit, fuel flow transmitter and gage, minimum pressure flow valve, flow divider, and dual fuel manifolds with fourteen simplex nozzles.

The engine-driven boost pump operates when the gas generator shaft (N_1) is turning, to provide sufficient fuel head pressure (approximately 30 psi) to the high-pressure pump to prevent cavitation. The oil-to-fuel heat exchanger uses warm engine oil to maintain a desired fuel temperature at the fuel pump inlet to prevent icing at the pump filter. This is done automatically and requires no action by the pilot.

After fuel passes through the oil-to-fuel heat exchanger, it flows into the high-pressure, engine-driven fuel pump and on into the fuel control unit (FCU).

The high-pressure fuel pump is an engine-driven, gear-type pump with an inlet and outlet filter. Flow rates and pressures will vary with gas generator (N_1) rpm and FCU operation. The high-pressure pump supplies fuel to the fuel-receiving side of the FCU, up to a maximum pressure of 1,050 psi. Its primary purpose is to provide sufficient pressure at the fuel nozzles for a good spray pattern at all modes of engine operation.

A fuel-purge line is positioned at the output side of the high-pressure fuel pump, which constantly directs a small amount of fuel back to the gravity-feed line between the wing and nacelle tanks, to ensure that the FCU stays clear of vapors and bubbles.

Also located in the FCU is the pump unloading valve. This valve is controlled by the condition lever. It is either open to unload the pressure, or closed. There is no intermediate position of this valve.

The minimum pressurizing valve is located in conjunction with the flow divider. It blocks fuel flow during starts until fuel pressure builds sufficiently to maintain a proper spray pattern in the combustion chamber. About 100 psi is required to open the minimum pressurizing valve. The engine-driven high-pressure

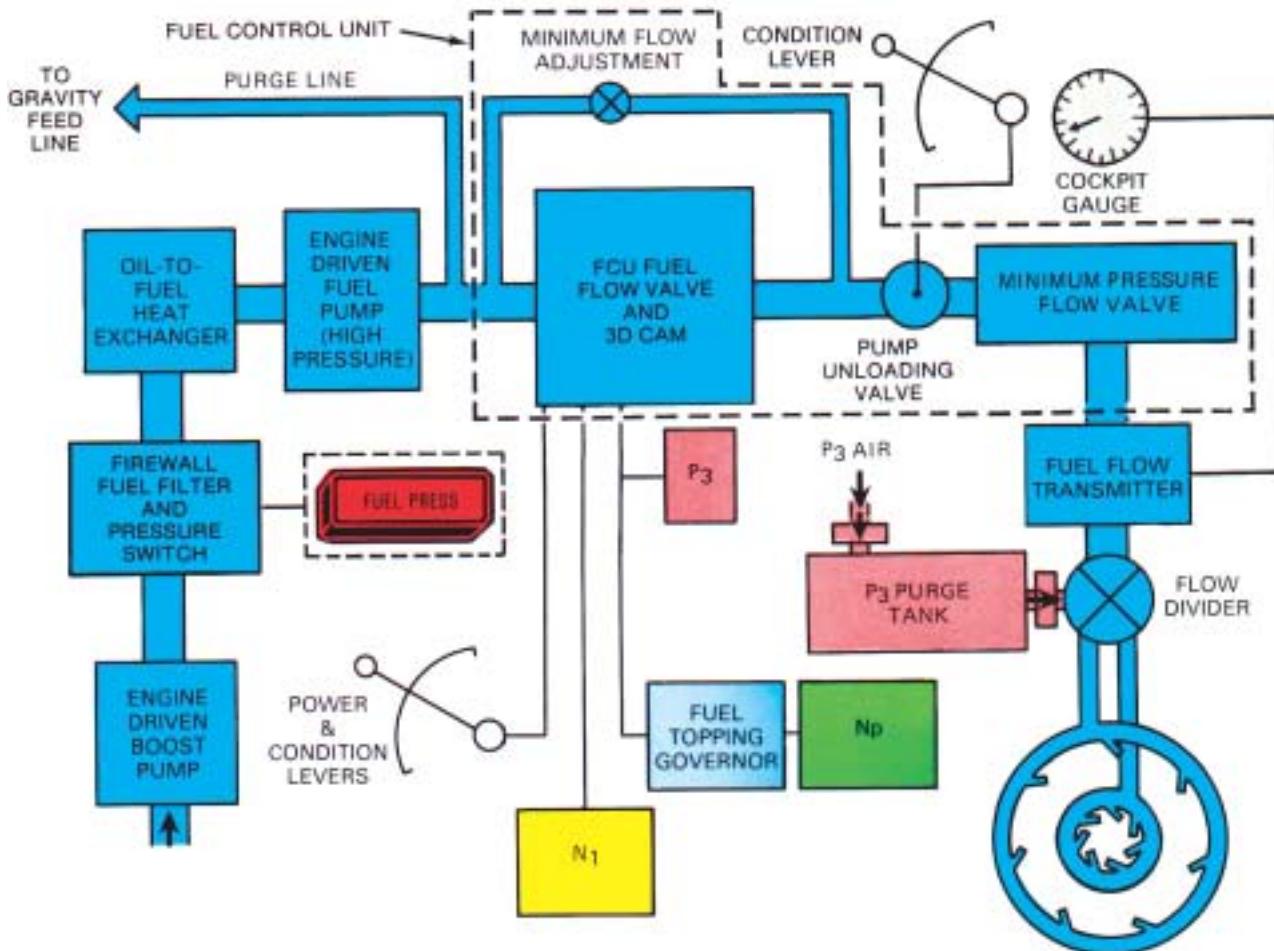


Figure 7-16 Simplified Fuel System Diagram

fuel pump maintains this required pressure. If the pump should fail, the valve would close and the engine would flame out.

For starting, fuel flows initially through the flow divider to the primary fuel spray nozzles in the combustion chamber. As the engine accelerates through approximately 35 to 40% N_1 , fuel pressure increases sufficiently to also supply the secondary fuel nozzles. At this time, all 14 nozzles are delivering atomized fuel to the combustion chamber. This progressive sequence of primary and secondary fuel nozzle operation provides cooler starts. During engine starts, there may be an increased acceleration in N_1 speed when the secondary fuel nozzles start delivering fuel.

Fuel Manifold Purge System

A fuel manifold purge system is provided to dispose of residual fuel in the flow divider and fuel manifold after engine shutdown. It consists of a P₃ pressure tank with connections for P₃ air input at one end and a discharge to the flow divider at the other end.

During normal engine operation, P₃ air enters the tank through a check valve to pressurize the tank. Fuel pressure against the discharge check valve prevents the air from escaping as long as the engine is running. Upon shutdown, as fuel pressure drops to zero, P₃ air escapes through the flow divider into the fuel manifold and nozzles, pushing any residual fuel into the combustion



chamber where it is burned. As a result, the pilot may notice a one- to two-second delay in initial engine spooldown after shutdown.

Fuel Control Unit

The fuel control unit (Figure 7-17), normally referred to as the FCU, has multiple functions, but its main purpose is to meter the proper fuel amount to the nozzles in all modes of engine operation. It is calibrated for starting flow rates, acceleration, and maximum power. The FCU compares gas generator speed (N_1) with the power lever setting and regulates fuel to the engine fuel nozzles. The FCU also senses compressor section discharge pressure, compares it to rpm, and establishes acceleration and deceleration fuel flow limits. A minimum flow adjustment, which is set to approximately 90 pounds/hour, guarantees sufficient fuel flow to sustain engine operation at minimum power.

FCU Operation

The fuel control unit (FCU) (Figure 7-17) is mounted on the rear flange of the fuel pump. A splined coupling between the pump and the FCU transmits a speed signal, proportional to gas generator shaft speed (N_1), to the governing section in the FCU. The FCU determines the fuel schedule for the engine to provide power required by controlling gas generator speed.

Engine power output is directly dependent upon gas generator speed (N_1), which is controlled by regulating the amount of fuel to the combustion section of the engine. Compressor discharge pressure (P_3) is sensed by the fuel control unit and is used to establish acceleration fuel flow limits. This fuel limiting function is used to prevent overtemperature conditions in the engine during starting and acceleration.

The fuel control consists of the following major components:

- A condition lever that selects the start, low idle, and high idle functions.

- A power lever that selects gas generator speed (N_1) between idle and maximum through the 3D cam, cam follower lever, and fuel valve.
- A flyweight governor that controls fuel flow to maintain selected speed.
- A pneumatic bellows that controls the acceleration schedule and acts to reduce gas generator speed in the event of propeller overspeed.

FCU operation is complicated, but it will be simplified and described briefly here. For detailed description and operation, refer to the *Pratt & Whitney Maintenance Manual* which applies to this engine.

The condition lever selects the LOW IDLE to HIGH IDLE N_1 speeds, while the power lever selects speeds between idle and maximum, 104% N_1 . This control lever positions a 3D cam in the FCU which, through a cam follower and lever, determines fuel flow corresponding to the selected N_1 speed.

The engine speed is controlled by the gas generator (N_1) governor which contains two flyweights mounted on a ballhead driven by the engine. The flyweight governor is the feedback element of the speed select system. It controls the "on-speed" condition by positioning the 3D cam as required in response to speed variations in the gas generator. As N_1 speed increases or decreases, the resulting flyweight action changes the 3D cam setting, which changes the fuel flow valve setting to maintain the selected N_1 speed.

The action of the N_1 governor maintains these forces in balance continually so the axial position of the 3D cam always represents engine speed. The cam follower and arm transmit motion of the 3D cam to the fuel valve. As the 3D cam moves upward, fuel flow to the engine is increased and N_1 speed increases. Downward movement of the 3D cam decreases fuel flow and N_1 speed. N_1 speed is therefore maintained by the N_1 governor in response to variations in power lever position. The governor adjusts fuel flow as required to maintain the requested N_1 speed.

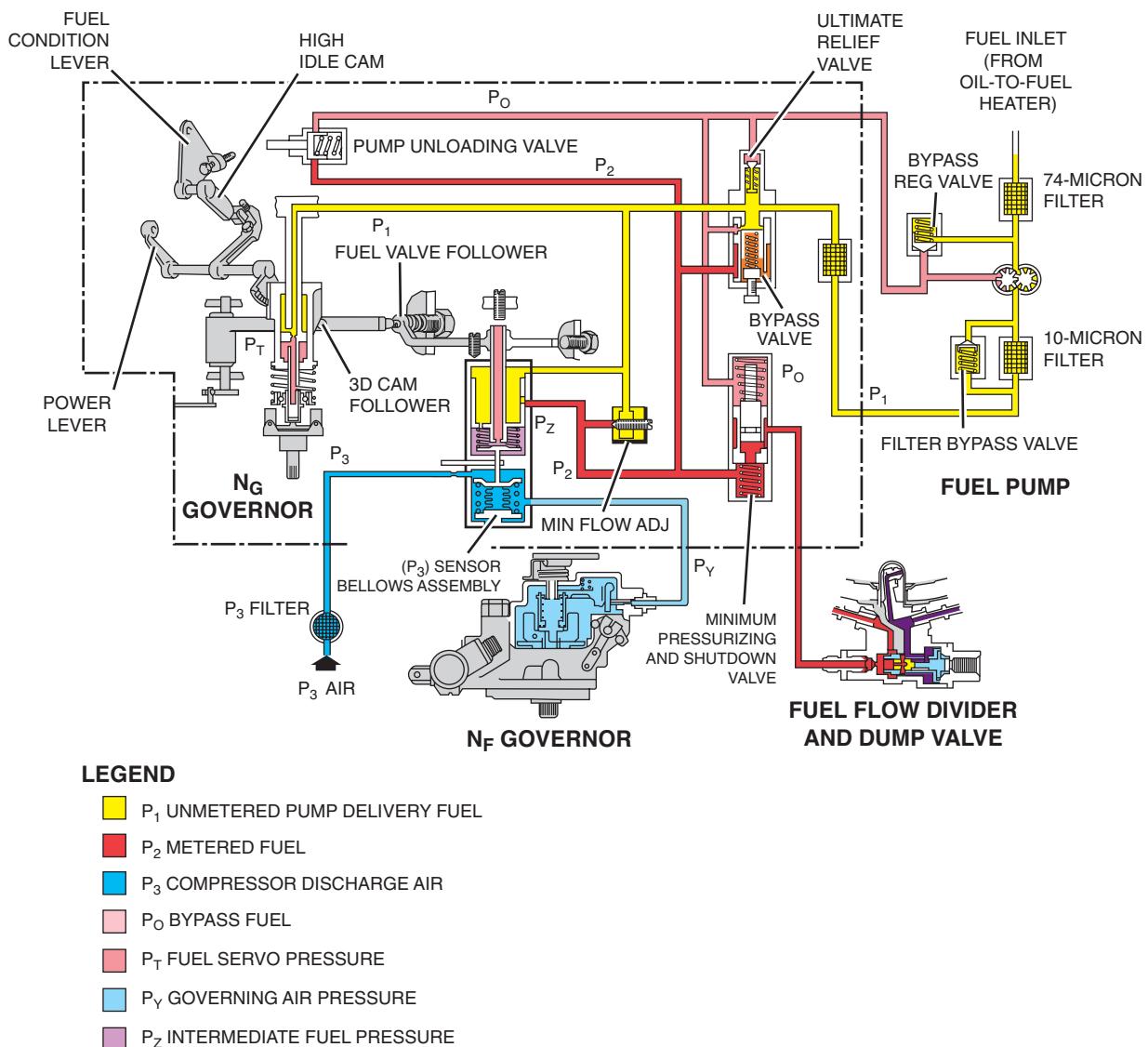
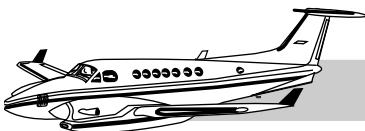


Figure 7-17 Simplified Fuel Control System



In an overspeed condition, increasing pressure by the governor flyweights moves the 3D cam downward, resulting in decreased fuel flow from the fuel flow valve to the engine. A balance point is reached when the N_1 speed is reduced to the selected speed, and the cam is stationary at the new speed position.

In an underspeed condition, decreasing pressure by the governor flyweights moves the 3D cam upward, resulting in increased fuel flow from the fuel flow valve to the engine until the system is in equilibrium again.

Compressor discharge pressure (P_3 air) is a second input affecting the fuel flow valve position during acceleration or deceleration to maintain the selected speed condition of the gas generator. An increase in P_3 causes the fuel flow valve to increase fuel flow in response to increased P_3 pressure until N_1 speed is stabilized. A decrease in P_3 causes the fuel flow valve to decrease fuel flow until N_1 speed is stabilized at the lower selected value.

Fuel Pressure Indicators

In the event of a primary engine driven boost pump failure, the respective FUEL PRESS red light in the warning annunciator panel (Figure 7-18) will illuminate and the master warning lights will flash. The FUEL PRESS light illuminates when outlet pressure at the engine-driven boost pump decreases below 10 psi. Switching on the standby fuel boost pump should increase fuel pressure above 11 (\pm 2) psi and extinguish the warning light.

CAUTION

Engine operation with the FUEL PRESS light on is limited to ten hours between overhaul or replacement of the engine-driven high-pressure fuel pump. Boost pump fuel pressure is needed to lubricate, cool and prevent cavitation of the high pressure fuel pump.

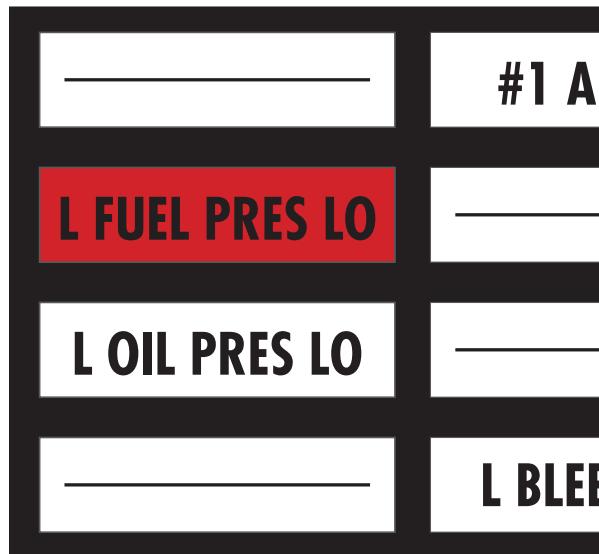


Figure 7-18 Fuel Pressure Annunciator

In the event of an engine-driven fuel pump (high pressure) failure, the engine will flame out since this high-pressure fuel is required to open the minimum pressurizing valve.

Fuel Flow Indicators

Fuel flow information is sensed by a transmitter in the engine fuel supply line between the fuel control unit and the flow divider, and is indicated on the fuel flow gage on the instrument panel (Figure 7-19). The gage indicates fuel flow in pounds-per-hour units times 100. Therefore when the needle indicates 2 on the dial, fuel flow is 200 pounds per hour. The fuel flow gages are DC-powered.

Fuel Additives

There are two approved fuel additives for the Model 300 and 350. An anti-icing additive conforming to the specification MIL-I-27686 is required when flying into known forecast conditions below -45°C . It should be blended in accordance with the procedures outlined in the Super King Air POH.



Figure 7-19 Fuel Flow Indicator

A fuel biocide called "Biobor JF" is also approved for prevention of micro-organism growth within fuel tanks and lines. It should be blended as outlined in the *Super King Air Maintenance Manual*.

ENGINE POWER CONTROL

Torque is controlled by the power lever acting on the gas generator governor (N_1). When the position of the power lever calls for more torque, the governor settings prevent the bleed-off of internal pressure and some of the P_3 air in the FCU. This moves the metering nozzle to allow the necessary fuel flow into the spray nozzles to meet the power condition called for.

The propeller lever adjusts the propeller governor to the desired propeller speed. The propeller will maintain the set speed by varying the blade angle as more or less torque is applied.

ITT AND TORQUEMETERS

Power management is relatively simple, with two primary operating limitations; temperature and torque. Engine torque and ITT operating parameters are affected by ambient temperature and altitude. During operation requiring maximum engine performance at cold temperature or low altitude, torque limits power; while at hot temperature or high altitude, ITT limits power. Whichever reaches its limit first, determines the power available.

ITT Gage

The ITT gage monitors the Interstage Turbine Temperature at Station 5 (Figure 7-20). ITT is a prime limiting indicator of the amount of power available from the engine under varying ambient temperature and altitude conditions. The normal operating range, indicated by the green arc on the gage, is 400°C to 820°C . These limits also apply to maximum continuous power. The maximum starting-only temperature of $1,000^{\circ}\text{C}$ is indicated by either a solid red or white line on the instrument. This starting limit of $1,000^{\circ}\text{C}$ is limited to five seconds. The ITT gages are self-energizing and do not require electrical power. The engines



INTERSTAGE TURBINE TEMPERATURE GAGE

- █ 400-820°C NORMAL OPERATING RANGE
- █ 820°C MAXIMUM CONTINUOUS LIMIT
- █ 1000°C MAXIMUM STARTING ONLY LIMIT

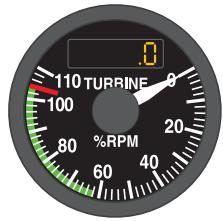


TORQUE METER

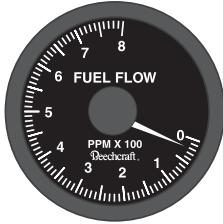
- █ 0 TO 100% NORMAL OPERATING RANGE
- █ 100% MAXIMUM LIMIT

PROPELLER TACHOMETER (N_p SPEED)

- █ 1450-1700 RPM NORMAL OPERATING RANGE
- █ 1700 RPM MAXIMUM LIMIT

GAS GENERATOR TACHOMETER (N_t SPEED)

- █ 62-104% NORMAL OPERATING RANGE
- █ 104% MAXIMUM LIMIT



NO LIMITATIONS MARKINGS



OIL TEMPERATURE SCALE

- █ 0-99°C NORMAL OPERATING RANGE
- █ 99-110°C CAUTION RANGE
- █ 110° MAXIMUM OIL TEMPERATURE LIMIT

OIL PRESSURE SCALE

- █ 60 PSI MINIMUM LIMIT
- █ 60-90 PSI CAUTION RANGE
- █ 90-135 PSI NORMAL OPERATING RANGE
- █ 200 PSI MAXIMUM LIMIT

Figure 7-20 Engine Instrument Markings



can be damaged if limiting temperatures indicated on the ITT gage are exceeded.

Torquemeter

The torquemeter, which is marked with the green arc from 0% to 100%, constantly measures rotational force applied to the propeller shaft (Figure 7-20). The maximum permissible sustained torque is 100%, indicated by the red radial at the top of the green arc on the instrument. Cruise torques vary with altitude and temperature.

Torque is measured by a hydromechanical torquemeter in the first stage of the reduction gearcase. Rotational force on the first-stage ring gear compresses oil in the torquemeter chamber. The difference between the torquemeter chamber pressure and reduction gear internal pressure accurately indicates the torque being produced at the propeller shaft. The torquemeter transmitter measures this torque and sends an AC electrical signal to the torquemeter on the cockpit instrument panel (DC [FL-121 and after]).

Gas Generator Tachometer

The gas generator (N_1) tachometer measures rotational speed of the compressor shaft, in percent rpm, based on 37,500 rpm at 100% (Figure 7-20). The outer dial reads in single units and the smaller inner dial reads in units of ten. The N_1 indicator is self-generating. The tachometer generator sensing unit, located in the engine accessory section, is geared to the N_1 indicator on the cockpit instrument panel to indicate the percent of N_1 revolutions. Maximum continuous gas generator speed is limited to 39,000 rpm, which is 104% on the N_1 indicator.

CONTROL PEDESTAL

The control pedestal extends between pilot and copilot (Figure 7-21). The three sets of control levers are, left to right: the power levers, propeller rpm and feather levers, and the condition levers

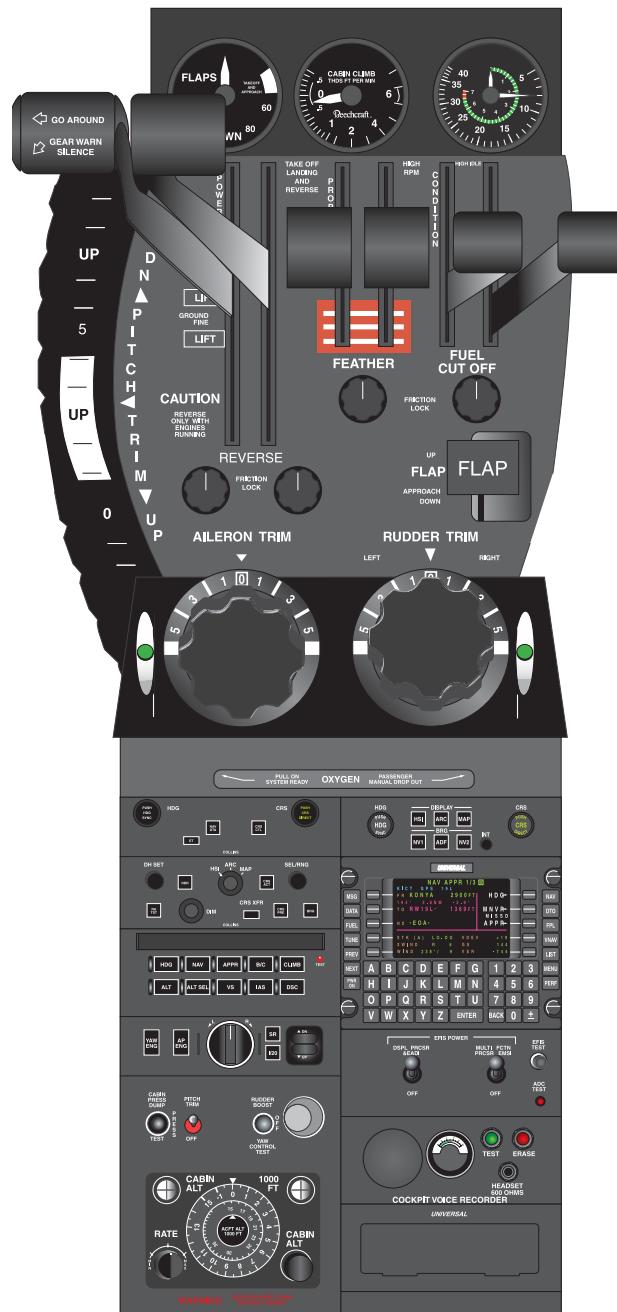
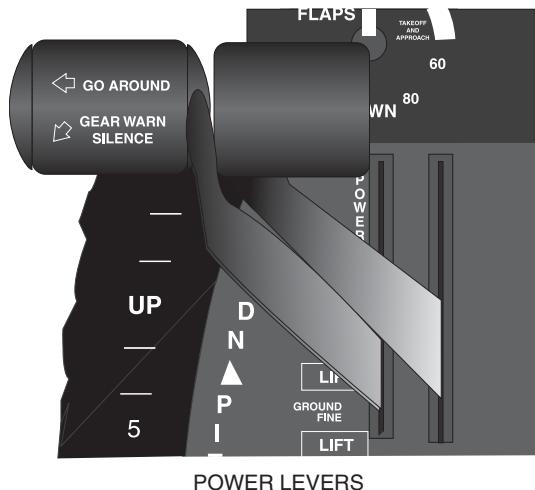


Figure 7-21 Control Pedestal



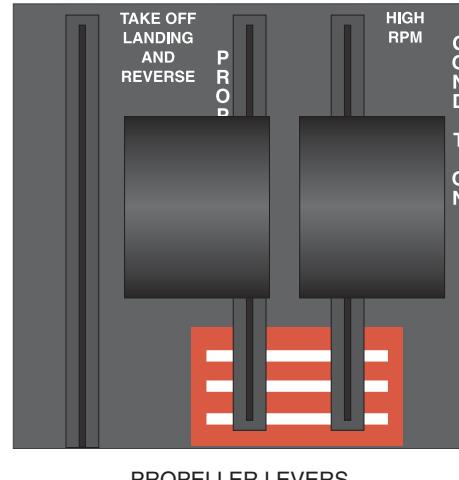
Power Levers

The power levers control engine power from idle to takeoff power by operation of the gas generator (N_1) governor in the fuel control unit (Figure 7-22). Increasing N_1 rpm results in increased engine power. The power levers have three control regions: forward thrust, ground fine, and reverse. When the levers are lifted over the IDLE gate and pulled back into the ground fine range, they hold engine power at the selected idle speed and control propeller blade angle. The ground fine range is normally used for taxi. When the levers are lifted over the GROUND FINE gate into the REVERSE range, they control engine power and propeller blade angle.



Propeller Levers

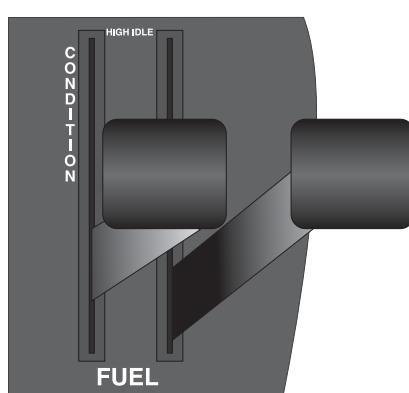
The propeller levers are conventional in setting the required rpms for takeoff and cruise positions (Figure 7-22). The normal governing range is 1,450 to 1,700 rpm. This airplane is equipped with both manual and automatic propeller feathering systems. To feather a propeller manually, pull the propeller lever back past the friction detent into the red and white striped section of the quadrant. To unfeather, push the lever back into the governing range. The propellers go to the feathered position when the engines shut down because of the loss of oil pressure in the propeller hub.



Condition Levers

The condition levers have three positions: FUEL CUTOFF, LOW IDLE, and HI IDLE (Figure 7-22). At FUEL CUTOFF position, fuel flow to the engines is cut off.

At LOW IDLE, engine gas generator speed (N_1) is a minimum of 62%, at HI IDLE it is 70%. The levers can be set between these two values for any proportional speed between 62% and 70% N_1 speed.



CONDITION LEVERS

Control Lever Operation

The engines are controlled from the cockpit by using the propeller, power, and condition levers. Both the power and condition levers are con-

Figure 7-22 Control Levers



nected to the N_1 governing section of the FCU. Either lever will reset the FCU to maintain a new N_1 rpm. For starting, the power levers are at IDLE position. Once the condition levers are moved to LOW IDLE, the fuel cutoff valve allows fuel to flow to the nozzles and the N_1 governor is set at LOW IDLE. The condition levers are continuously variable from LOW IDLE at 62% to HI IDLE at 70% N_1 . This variable idle speed with power levers at IDLE enhances engine cooling by maintaining a steady airflow through the engines.

With the condition levers at LOW IDLE, the power levers will select N_1 rpm from 62% to 104%, the maximum for takeoff. However, if the condition levers are at HI IDLE, the power levers can select N_1 rpm only from 70% to 104%.

Moving either the power levers or the condition levers changes N_1 rpm. As the power or condition levers are advanced, ITT, torque and fuel flow increases. These indicators are by products of the N_1 speed maintained by the FCU. With power levers in a fixed position, N_1 remains constant even in a climb or descent. However, ITT, torque and fuel flow will vary with altitude, ambient air temperature, and propeller setting.

ENGINE LIMITATIONS

Airplane and engine limits are described in the Limitations section of the *POH* (Figure 7-23). These limitations have been approved by the Federal Aviation Administration, and must be observed in the operation of the Beechcraft Super King Air 300/350. The ENGINE OPERATING LIMITS chart gives the major operating limits. The POWERPLANT INSTRUMENT MARKINGS chart lists the minimum, normal, and maximum limits.

During engine start, temperature is the most critical limit. The ITT starting limit of 1,000°C, represented on the ITT gage by either a solid red or white line, is limited to five seconds. During any start, if the indicator needle approaches the limit, the start should be aborted before the needle passes the solid white line (Figure 7-24). For

this reason, it is helpful during starts to keep the condition lever out of the LOW IDLE detent so that the lever can be quickly pulled back to FUEL CUTOFF.

Monitor oil pressure and oil temperature. During the start, oil pressure should come up to the minimum red line at 60 psi quickly, but should not exceed the maximum at 200 psi. During normal operation the oil temperature and pressure gages should be in the green arc normal operating range. The green arc extends from 90 to 135 psi. Fluctuations of plus or minus ten psi are acceptable.

Oil pressure between 60 and 90 psi is undesirable; it should be tolerated only for

ENGINE OPERATING LIMITS

The following limitations shall be observed. Each column presents limitations. The limits presented do not necessarily occur simultaneously. Refer to the Pratt & Whitney Engine Maintenance Manual for specific actions required if limits are exceeded.

OPERATING CONDITION	TORQUE % (1)	MAXIMUM ITT °C	GAS GENERATOR RPM %N ₁	PROP RPM N ₂	OIL PRESS PSI (2)	OIL TEMP °C (3)(4)
STARTING	---	1000 (5)	---	---	0 to 200	-40 to -40 (min)
IDLE	---	750 (6)	62 (min)	1050 (60 min)	60 (min)	-40 to +110
TAKEOFF	100	820	104	1700	90 to 135	0 to 110
MAX CONT	100	820	104	1700	90 to 135	0 to 110
CRUISE CLIMB	(7)	785	104	1700	90 to 135	0 to 110
MAX CRUISE	(7)	820	104	1700	90 to 135	0 to 110
MAX REVERSE	---	760	---	1650	90 to 135	0 to 99
TRANSIENT	156 (8)	850 (8)	104	1870	40 to 200 (5)	0 to 110

FOOTNOTES:

(1) Torque limit applies within range of 1000 - 1700 propeller rpm (N₂). Below 1000 propeller rpm, torque is limited to 62%.

(2) Normal oil pressure is 90 to 135 psig at gas generator speeds above 72%. With engine torque below 62%, minimum oil pressure is 60 psig at normal oil temperature (60° to 70° C). Oil pressures under 60 psig are undesirable. Under emergency conditions, to complete a flight, a lower oil pressure limit of 60 psig is permissible at reduced power level not exceeding 62% torque. Oil pressures below 60 psig are unsafe and require that either the engine be shut down or a landing be made as soon as possible, using the minimum power required to sustain flight. Fluctuations of plus or minus 10 psig are acceptable.

(3) A minimum oil temperature of 55°C is recommended for fuel heater operation at take-off power.

(4) Oil temperature limits are -40°C and +110°C. However, temperatures between 99°C and 110°C are limited to a maximum of 10 minutes.

(5) These values are time limited to 5 seconds.

(6) High ITT at ground idle may be corrected by reducing accessory load and/or increasing N₁ rpm.

(7) Cruise torque values vary with altitude and temperature.

(8) These values are time limited to 20 seconds.

Figure 7-23 Engine Limits Chart KA-300

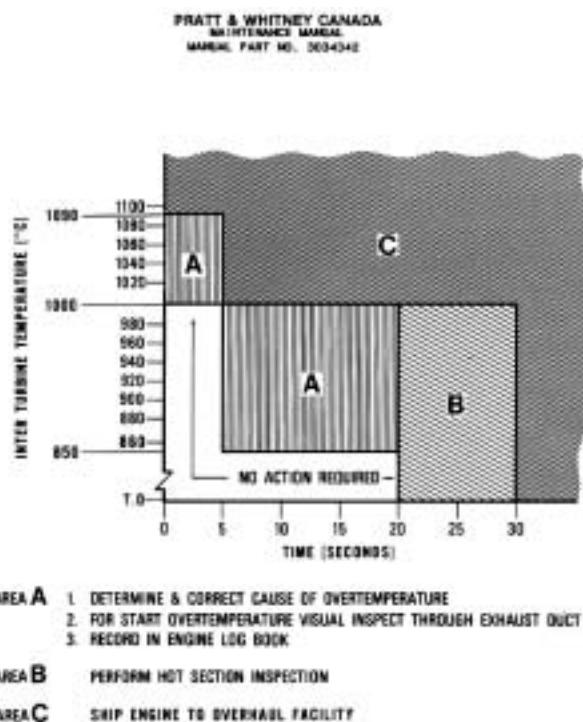


Figure 7-24 Overtemperature Limits

completion of the flight, and then only at a reduced power setting.

Oil pressure below 60 psi is unsafe; it requires that either the engine be shut down or that a landing be made as soon as possible, using minimum power required to sustain flight.

A minimum oil temperature of 55°C is recommended for oil-to-fuel heater operation at takeoff power. Oil temperature limits are -40°C and +110°C during IDLE, and 0°C to +110°C during normal operations. However, temperatures between +99°C and +110°C are limited to a maximum of 10 minutes.

During ground operations, ITT temperatures are critical and can be controlled by the N₁ rpm, air conditioning, P₃ use, and the generator load. With the condition levers at LO IDLE, high ITT can be corrected by reducing the generator and other N₁ loads, then increasing the N₁ rpm by advancing the condition levers. The air conditioner, for example, draws a heavy load on the right engine N₁, and may have to be temporarily turned off. At approximately 70% N₁ rpm, the HI IDLE condition lever position will normally

reduce the ITT. Once ITT is reduced below the IDLE limit of 750°C, the N₁ loads may be restored as desired as long as ITT stays below 750°C. During normal flight operations, the ITT should never be allowed to exceed the maximum continuous limit at the red line. This is 820°C on the PT6A-60A engine.

In the climb, torque will decrease and ITT may increase slightly. The cruise climb ITT limit is not placarded on the indicator. Torque, ITT, N₁, and propeller limits are the same in maximum cruise as they are for take-off; however, cruise torque values will vary with altitude and temperature.

Transient limits provide buffers for surges during engine acceleration. Torque and ITT have an allowable excursion duration of 20 seconds. A momentary peak of 156% and 850°C is allowed for torque and ITT respectively during acceleration.

The OVERTORQUE LIMITS chart (Figure 7-25) shows actions required if torque limits are exceeded under all conditions. If the torque limits are exceeded for more than a few minutes the gearbox can be damaged. The chart shows the specific limits and action required if they are exceeded.

The Generator Limits table in the POH shows limits for ground operation at various N₁ rpms. Any time these limits are exceeded, the accessory load must be reduced or N₁ increased to the limits shown in the table. A generator load of 75% is maximum for ground operation at an N₁ of between 62% and 70%. The Inflight limits are 100% generator load from sea level to 34,000 feet. Above 34,000 feet, maximum sustained generator load limit is 95%. Maximum continuous and maximum cruise share the same generator limits, but due to N₁ loading, certain limits must not be exceeded as indicated in the Before Takeoff (Final Items) checklist in the POH.

The overtemperature chart (Figure 7-24) shows the specific actions required if ITT limits are exceeded during starting conditions. For Area A, determine and correct the cause of overtemperature. If it was during a start, have the

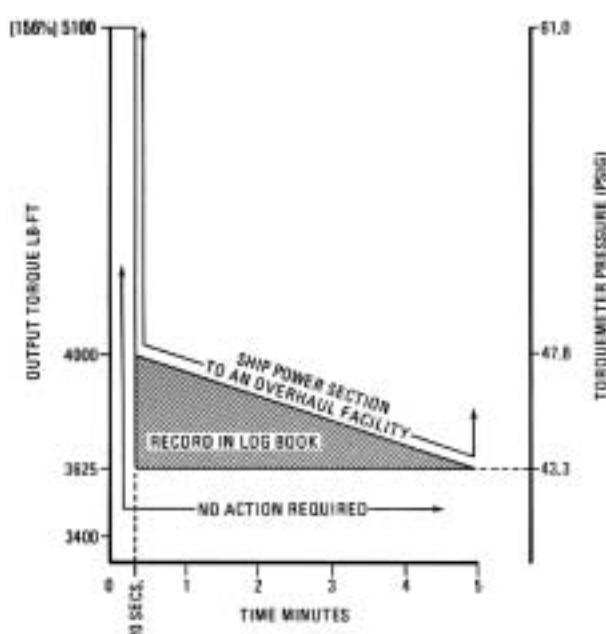


Figure 7-25 Overtorque Limits

engine visually inspected through the exhaust duct (Figure 7-26), then record the action in the engine log book.

Overtemperature in Area B will require that a hot section inspection be performed. During a hot section inspection, the combustion chamber and turbine areas and components are examined and replaced. Parts should be repaired or replaced as necessary. In Area C, overtemperatures may require that the engine be returned for overhaul. Exceeding ITT limits in this area for more than a few seconds may cause extensive engine damage.

Starter Operating Temperature Limits

The engine starters are time-limited during the starting cycle if for any reason multiple starts are required in quick sequence. The starter is limited to 30 seconds ON then five minutes OFF for cooling before the next sequence of 30 seconds ON, five minutes OFF. After the third cycle of 30 seconds ON, the starter must stay OFF for 30 minutes. If these limits are not observed, overheating may damage the starter.

The second starter cycle is used for clearing the engine of residual fuel.

Trend Monitoring

During normal operations, gas turbine engines are capable of producing rated power for extended periods of time. Engine operating parameters, such as output torque, interstage turbine temperature, compressor speed, and fuel flow for individual engines, are predictable under specific ambient conditions. On PT6A engines, these predictable characteristics may be taken advantage of by establishing and recording individual engine performance parameters. These parameters can then be compared periodically to predicted values to provide day-to-day visual confirmation of engine efficiency.

The engine condition trend monitoring system recommended by Pratt & Whitney is a process of periodically recording engine instrument readings such as torque, interstage turbine temperature, compressor speed, and fuel flow, correcting the readings for altitude, outside air temperature, and airspeed if applicable, and then comparing them to a set of typical engine characteristics. Such comparisons produce a set of deviations in interstage turbine temperature, compressor speed and fuel flow.

Data Collection

The trend monitoring procedure used specifies that flight data be recorded on each flying day, every five flight hours, or other flight period. Select a flight with long-established cruise, preferably at a representative altitude and airspeed.

With engine power established and stabilized for a minimum of five minutes, record the following data on a form similar to this in-flight engine data log (Figure 7-26):

- Indicated airspeed (IAS)—Knots
- Outside air temperature (OAT)—Degrees Centigrade



- Pressure altitude (ALT)—Feet
- Propeller speed (N_p)—rpm
- Torque (T_q)—Percentage
- Gas generator speed (N_g or N_1)—Percent of Gas Generator Speed
- Interturbine temperature (ITT)—Degrees Centigrade
- Fuel flow (WF)—Pounds Per Hour

PROPELLER DESCRIPTION

This section of the training manual presents a description and discussion of the propeller system. Location and use of propeller controls, principle of operation, reversing, and feathering

are included in this unit. The descriptions include the various governors low pitch stop system, autofeather system, and syncrophaser.

THE PROPELLER SYSTEM

This section on the description, operation, and testing of the propeller system on the Beechcraft Super King Air 300 and 350 is directed at increasing the pilot's understanding of the theory of operation of a constant-speed, full-feathering, reversing propeller, and helping to better understand the propeller system checks conducted as outlined in the *Pilot's Operating Handbook*.

Each engine is equipped with a conventional four-blade, constant speed, full-feathering, reversing propeller mounted on the output shaft of the reduction gearbox (Figure 7-27). The propeller pitch (Figure 7-28) is controlled by oil pressure through a reduction gear box driven propeller governor. An oil pump, which is part of the propeller governor, boosts engine oil pressure to

DATE	OAT (%)	PRESS ALT	IAS (KTS)	PROP SPEED		TORQUE (FT/LBS)	N ₁ (%)	ITT	FUEL FLOW	DELTA* NG	DELTA* ITT	DELTA* FF	OIL TEMP	OIL PRESS	ELECT LOAD
					LEFT										
					RIGHT										
					LEFT										
					RIGHT										
					LEFT										
					RIGHT										
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					RIGHT										
					LEFT										
					RIGHT										

*CALCULATED

Figure 7-26 In-Flight Engine Data Log



move the propeller toward the low pitch (high rpm) position and into reverse. Without oil pressure to counteract the counterweights and feathering springs, the propeller blades would move into feather. Counterweights and feathering springs move the propeller blades toward high pitch (low rpm) and into the feathered position. There are no high pitch stop locks. Therefore, the propeller feathers after engine shutdown.

The minimum low pitch propeller position is determined by a mechanically actuated hydraulic stop. GROUND FINE and REVERSE blade angles are controlled by the power levers in the GROUND FINE and REVERSE range by adjusting the low pitch stop position.

There are 3 governors that may control propeller rpm. They are the primary, overspeed, and fuel topping governors. The primary and overspeed governors use oil pressure to change propeller

blade angle so that the propeller rpm is adjusted or limited. The fuel topping governor limits fuel to limit propeller rpm. The propeller control lever adjusts the primary governor through its normal governing range of 1450 to 1700 rpm. Should the primary governor malfunction, the overspeed governor prevents the propeller speed from exceeding approximately 1768 rpm. In the event the propeller blade angle cannot be changed by either the primary or overspeed governor, the fuel topping governor will intervene. The fuel topping governor attempts to limit the propeller rpm to 106% of selected propeller rpm, when the power lever is in the forward thrust range. In the GROUND FINE and REVERSE ranges, the fuel topping governor is reset to approximately 95% of selected propeller rpm. This assures that the primary governor remains in an underspeed condition while in the REVERSE range on the ground.



Figure 7-27 Hartzell Propeller

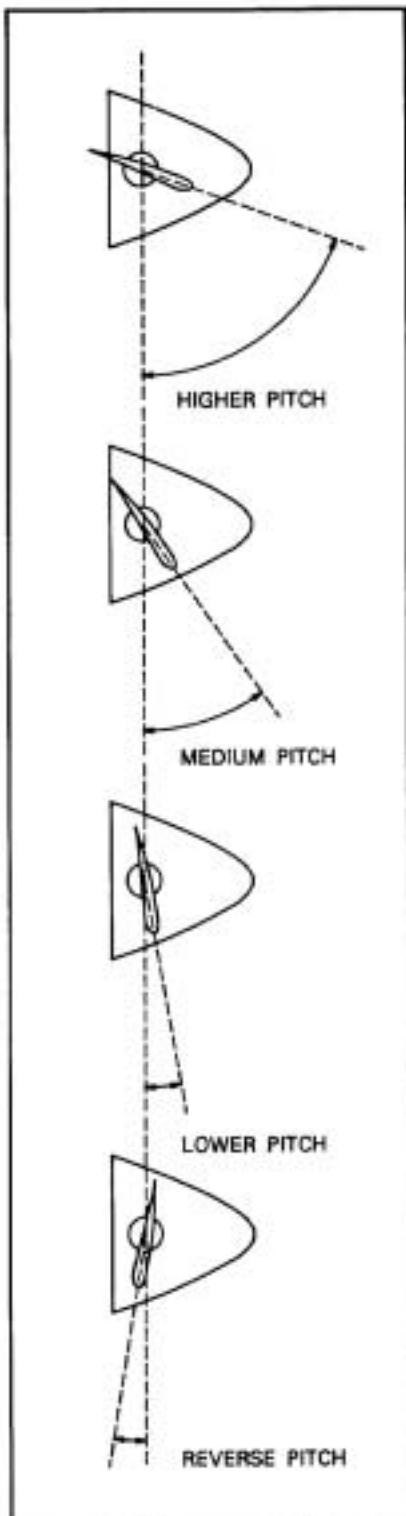


Figure 7-28 Propeller Pitch Diagram

BLADE ANGLE

Blade angle is the angle between the chord of the propeller and the propeller's plane of rotation. Blade angle is different near the hub than it is near the tip, due to the normal twist which is incorporated in a blade to increase its efficiency. In the propellers used on the Super King Air 300 and 350, the cord 42 inches out from the propeller's center has been selected as the position at which blade angle is measured. This position is referred to as the "42-inch station". All blade angles given in this section are based on the 42-inch station and are approximations (Figure 7-29).

PRIMARY GOVERNOR

The primary governor (Figure 7-30) is needed to convert a variable pitch propeller into a constant speed propeller. It does this by changing blade angle to maintain the propeller speed which the operator has selected. The primary governor can maintain any selected propeller speed from approximately 1450 rpm to 1700 rpm.

Suppose an airplane is in normal cruising flight with the propeller turning at 1500 rpm. If the pilot trims the airplane down into a descent without changing power, airspeed will increase. This decreases the angle of attack of the propeller blades, causing less drag on the propeller, thus beginning to increase its rpm. If this propeller has variable pitch capabilities and is equipped with a governor set at 1500 rpm, the governor will sense this "overspeed" condition and increase blade angle to a higher pitch. The higher pitch increases the blade's angle of attack, slowing it back to 1500 rpm, or "onspeed."

Likewise, if the airplane moves from cruise to climb airspeeds without a power change, the propeller rpm tends to decrease, but the governor responds to this "underspeed" condition by decreasing blade angle to a lower pitch, and the rpm returns to its original value. Thus the governor gives "constant speed" characteristics to the variable pitch propeller.

Power changes, as well as airspeed changes, cause the propeller to momentarily experience

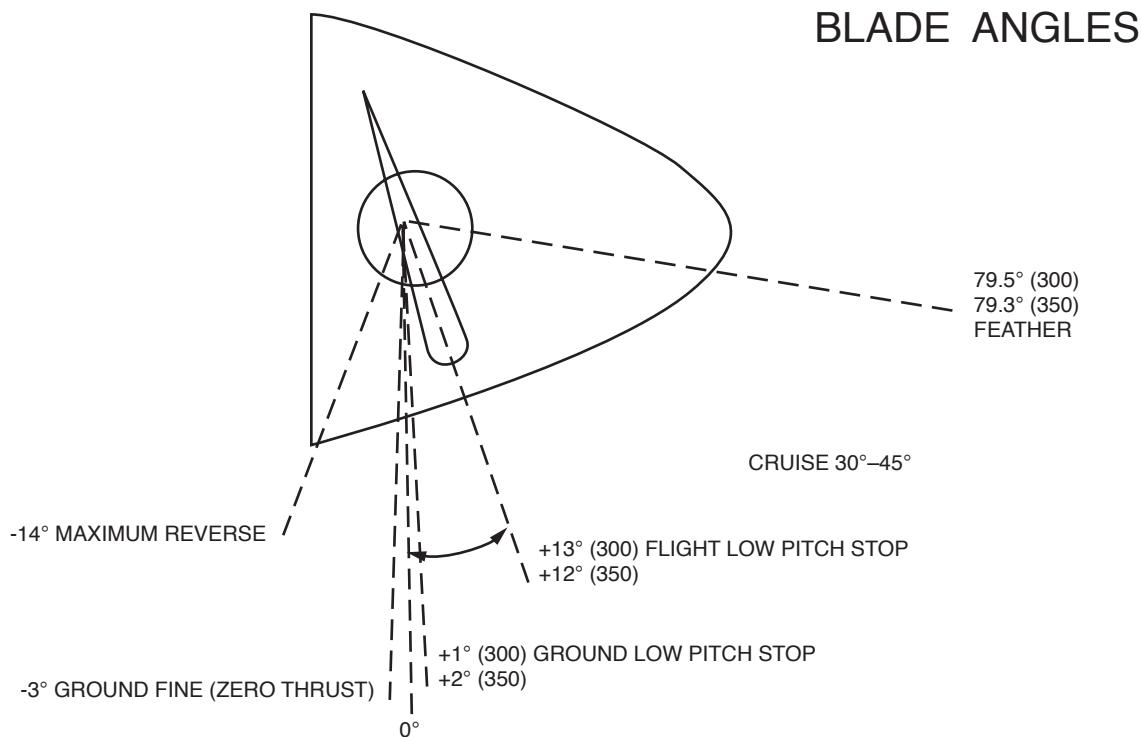


Figure 7-29 Propeller Blade Angle Diagram

overspeed or underspeed conditions, but again the governor reacts to maintain the onspeed condition. Due to the smooth action of the governor, the pilot will notice few, if any, of these minor adjustments.

Primary Governor Operation

The propeller levers adjust primary propeller governor settings between 1450 rpm and 1700 rpm. The primary propeller governor (Figure 7-31), mounted at the top of the engine reduction gearbox, can select any constant propeller rpm within the range of 1450 to 1700. The primary propeller governor adjusts propeller rpm by controlling the oil supply to the propeller mechanism.

An integral part of the primary propeller governor is the governor pump. This pump is driven by

the Np shaft and raises engine oil pressure from its normal range to approximately 375 psi. The greater the oil pressure sent to the propeller dome, the lower the propeller pitch and the higher the propeller rpm. Oil pressure is always trying to maintain a low pitch; however, the feathering springs and counterweights are trying to send the propeller to feather. Propeller control is a balancing act of opposing forces.

A transfer gland surrounds the propeller shaft. This transfer gland allows the oil to enter and exit the propeller dome as the propeller pitch is adjusted.

The primary propeller governor uses a set of rotating flyweights that are geared to the propeller shaft. The flyweights provide a comparison between the desired reference speed (as requested by the propeller levers in the cockpit) to how fast the propeller is turning.

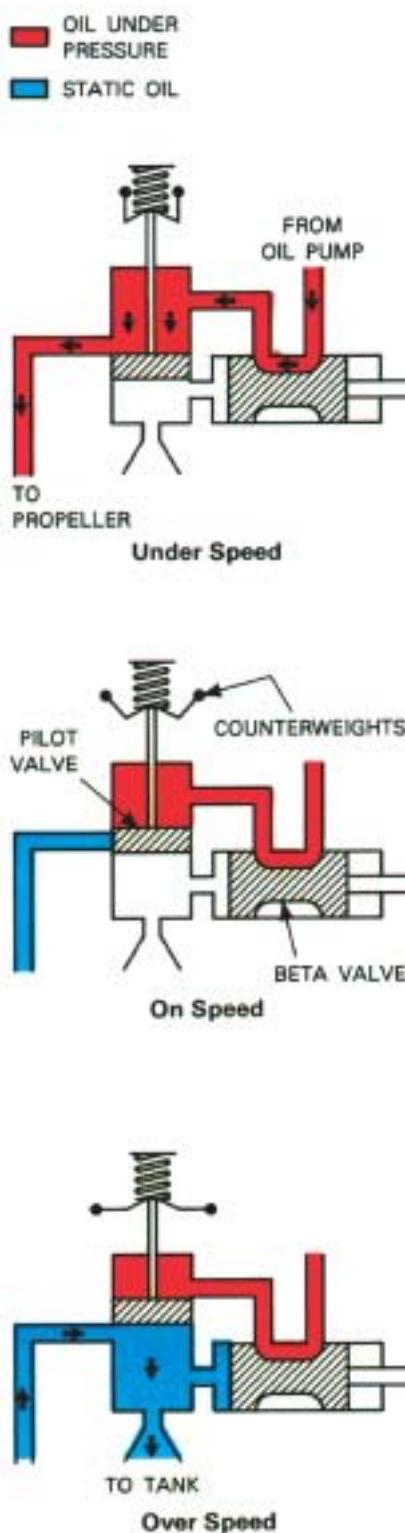


Figure 7-30 Primary Governor

These flyweights are connected to a free-floating pilot valve. The slower the flyweights are turning in relation to the desired reference speed, the lower the position of the pilot valve. If the propeller and flyweights turn faster, the additional centrifugal force makes the pilot valve rise inside the governor. The pilot valve position determines how much oil pressure is being sent to the propeller dome.

Propeller Onspeed

If a propeller rpm of 1500 is selected and the propeller is actually turning at 1500, the flyweights are in their center or “onspeed” condition (Figure 7-32). The pilot valve is in the middle position. This maintains a constant oil pressure to the propeller dome, which creates a constant pitch and constant rpm.

Propeller Overspeed

If the airplane enters a descent, or if engine power is increased without any change to the cockpit propeller levers, there will be a tendency for the airspeed to increase and the propeller to turn faster (Figure 7-33). The flyweights will, in turn, rotate faster. The additional centrifugal force will make the pilot valve rise. Notice that oil can now escape via the pilot valve. Lower oil pressure will result in a higher pitch and a reduction of propeller rpm. The propeller will then return to its original rpm setting. The flyweights will then slow down and the pilot valve will return to the middle position to maintain the selected propeller rpm.

Propeller Underspeed

If the airplane enters a climb, or if engine power is decreased without any change in the cockpit propeller controls, the airspeed will decrease and the propeller will tend to slow down (Figure 7-34). The flyweights in the propeller governor will also slow down, because of a loss in centrifugal force, and the pilot valve will move lower. This will allow more oil pressure to the propeller dome. Higher oil pressure will result in a lower pitch. This, in turn, will cause an increase in propeller rpm. As the propeller increases to its original rpm setting, the flyweights will speed up, and the pilot valve will return to its middle or “onspeed” position.

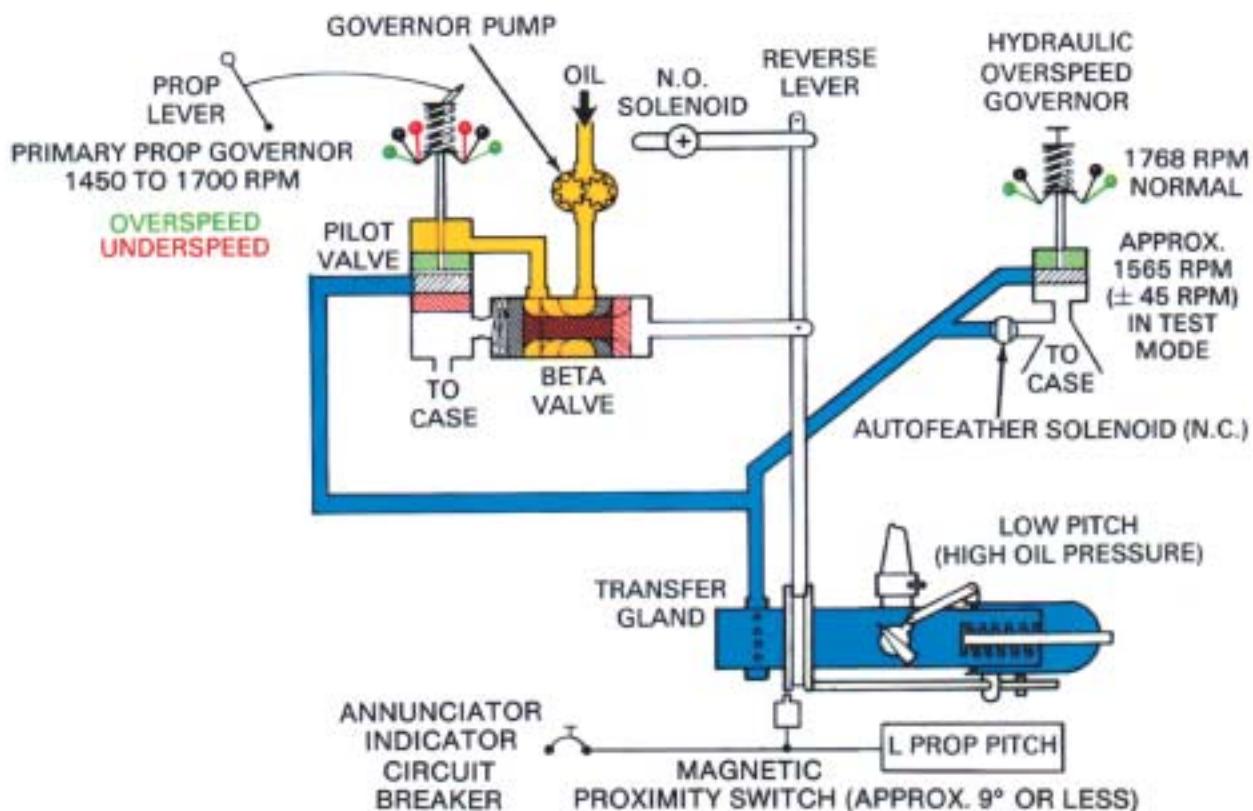


Figure 7-31 Propeller Complete System

The flyweights and pilot valve are always making small adjustments so that the propeller rpm is held constant by changing the propeller blade angles. The cockpit propeller lever adjusts where the “onspeed” condition will occur. The pilot can select any constant propeller rpm from 1450 to 1700 rpm. 1700 rpm is used for takeoff. Maximum Range Power and Recommended Cruise Power use 1500 rpm. In the event of a failure in the governor control linkage, an external spring on top of the governor will move the governor adjustment to 1700 rpm propeller speed.

If the blade angle could decrease all the way to 0° or reverse, the propeller would create so much drag that aircraft control would be dramatically reduced. The propeller, acting as a large disc, would create excessive drag and blank the airflow around the wing and tail surfaces. To prevent undesirable flight characteristics, a mechanism is provided to stop the governor from

selecting blade angles that are too low for safety. This mechanism provides for an adjustable low pitch stop. As the blade angle is decreased by the governor, the flight low pitch stop is eventually reached where the blade angle becomes fixed and cannot continue to a lower pitch. The governor is therefore incapable of restoring the onspeed condition, and propeller rpm decreases below the selected primary governor rpm.

Low Pitch Stop

There are situations where the primary governor cannot maintain the selected propeller rpm, such as final approach where power and airspeed are being reduced. With the progressive reduction of power and airspeed on final, the propeller and rotating flyweights will tend to go to the underspeed condition. In the underspeed condition the pilot valve will drop, increasing oil pressure to the dome, and the propeller pitch will decrease as power and airspeed are reduced.

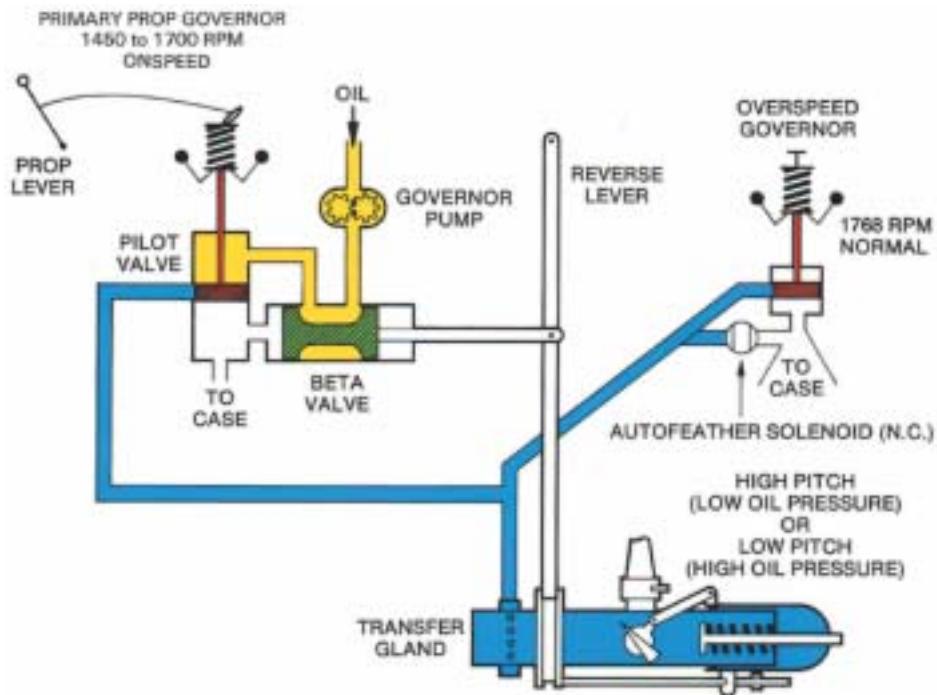


Figure 7-32 Propeller Onspeed Diagram

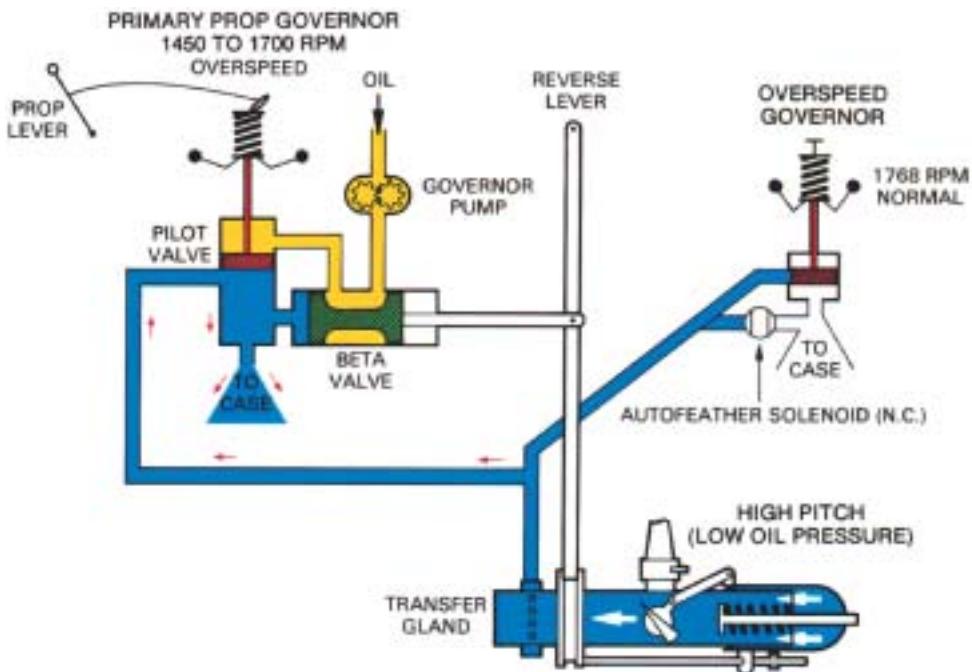


Figure 7-33 Propeller Overspeed Diagram

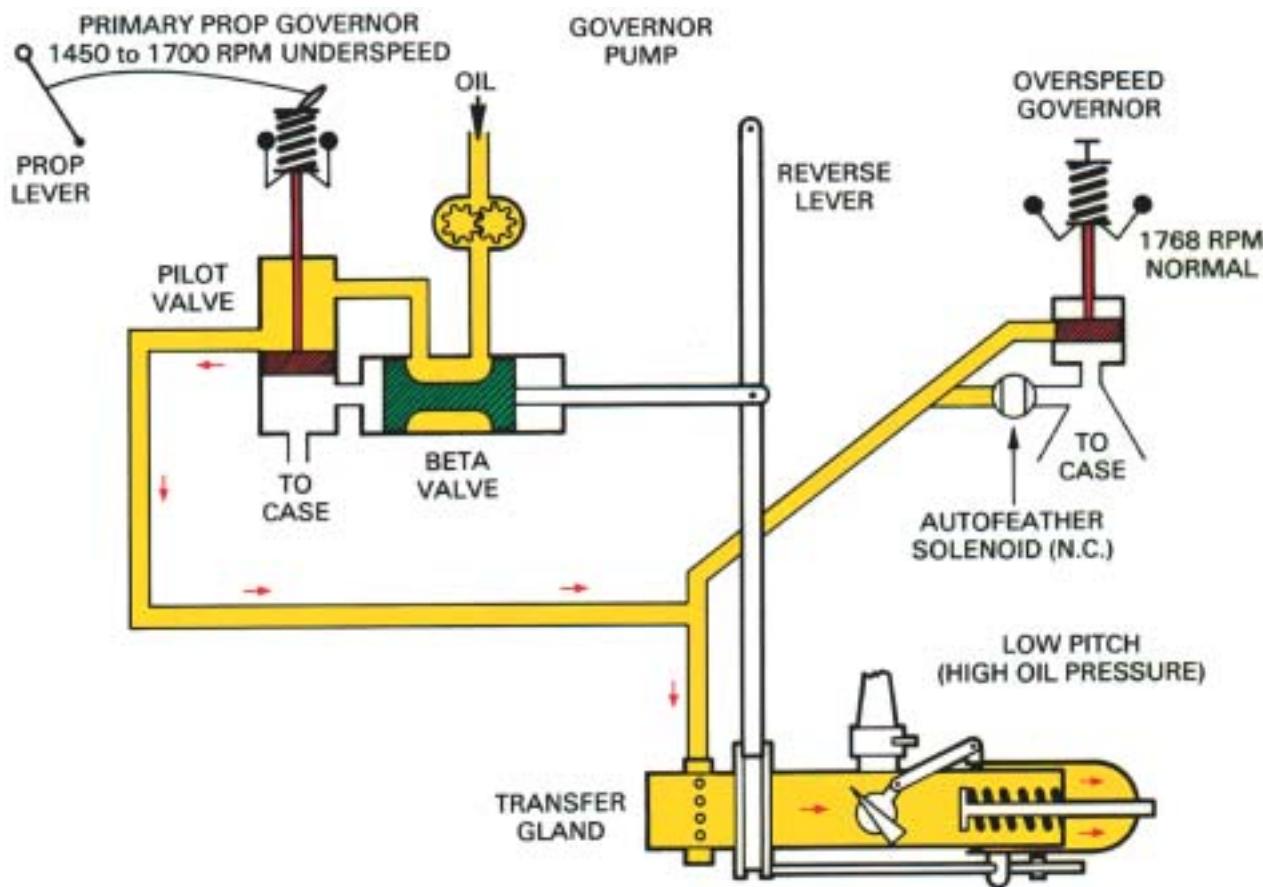


Figure 7-34 Propeller Underspeed Diagram

Assuming the propeller is not feathered or in the process of being feathered, whenever the propeller rpm is below the selected governor rpm, the propeller blade angle is at the low pitch stop. The low pitch stop will be the flight low pitch stop if the aircraft is in flight and it will be the ground low pitch stop if the aircraft is on the ground.

NOTE

This assumes that momentary periods of underspeed are not being considered. Rather, the propeller rpm is stabilized below the selected governor rpm.

For example, if the propeller control is set at 1500 rpm but the propeller is turning at 1400 rpm, the blade angle is at the low pitch stop.

On many types of airplanes, the low pitch stop is simply at the low pitch limit of travel, determined by the propeller's construction. But with a reversing propeller, the extreme travel in the low pitch direction is past 0°, into reverse or negative blade angles. Consequently, the low pitch stop on this propeller must be designed in such a way that it can be moved or repositioned when reversing is desired.



The low pitch stop is created by mechanical linkage sensing the blade angle. The linkage causes a valve to close which stops the flow of oil coming into the propeller dome. Since this oil causes low pitch and reversing, once it is blocked, a low pitch stop has been created. The low pitch stop valve, commonly referred to as the "beta" valve, is quite positive in its mechanical operation. Furthermore, the valve is spring-loaded to cut off the flow of oil and dump the oil out of the propeller dome in the event that control of the beta valve is lost due to linkage failure.

Low Pitch Stop Operation

Low pitch propeller position is determined by a mechanically created hydraulic stop (Figure 7-35). The propeller dome is connected by four spring-loaded polished rods to the feedback ring mounted behind the propeller. A carbon block riding in the feedback ring transfers the movement of the latter through the propeller reversing lever to the beta valve on the governor. The initial

forward motion of the beta valve blocks off the flow of oil to the propeller. Further motion forward dumps the oil from the propeller into the reduction gearbox sump. A mechanical stop limits the forward motion of the beta valve. Rearward movement of the beta valve does not affect normal propeller control. When the propeller is rotating at a speed slower than that selected, the governor pump provides oil pressure to the propeller dome and decreases the pitch of the propeller blades until forward motion on the feedback ring pulls the beta valve into a position blocking the supply of oil to the propeller, thus preventing further pitch reduction.

Ground Fine and Reverse Control

The position of the low pitch stop is controlled from the cockpit by the power lever. Whenever the power lever is at IDLE or above, this stop is set in flight at 13° on the 300, or 12° on the 350. On the ground, the low pitch stop is set at 1° on the 300, or 2° on the 350. Bringing the power

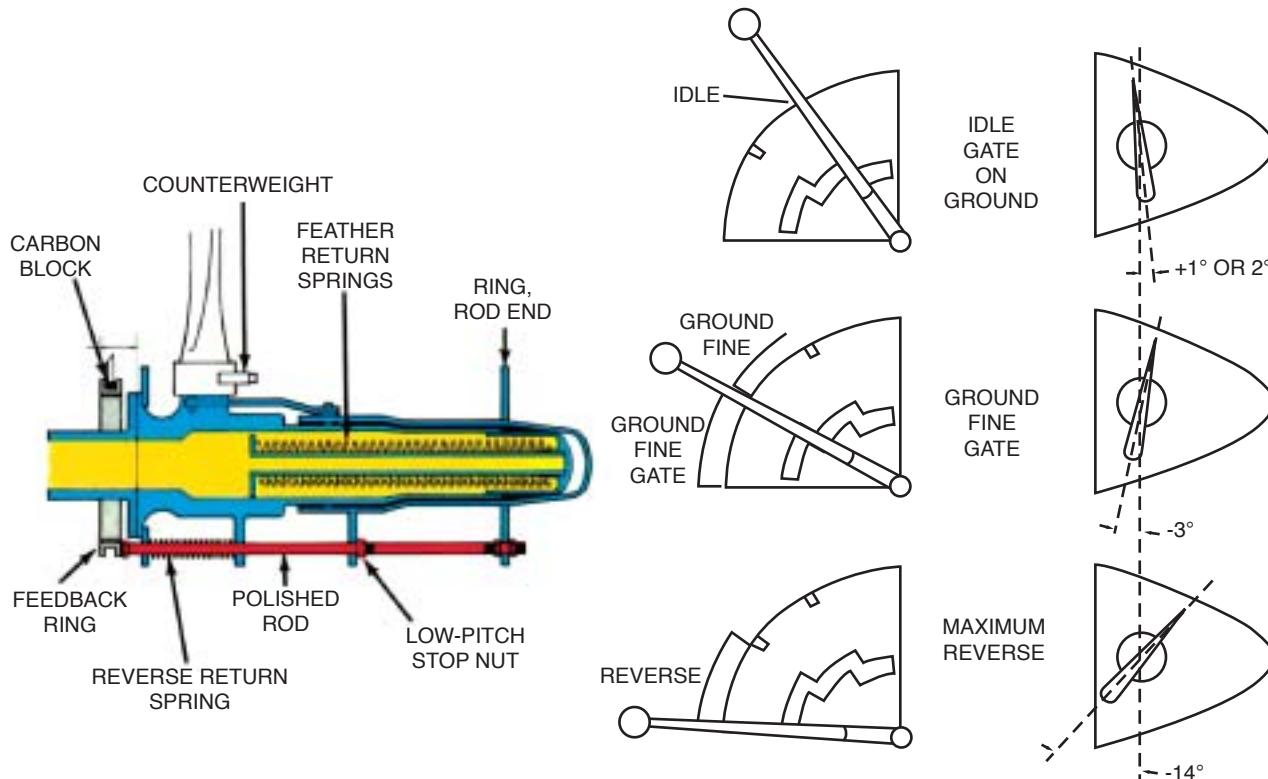


Figure 7-35 Low Pitch Stop Diagram



lever aft of IDLE progressively repositions the low pitch stop to smaller blade angles (Figures 7-36, 7-37, and 7-38).

The geometry of the power lever linkage through the cam box is such that power lever increments from IDLE to full forward thrust have no effect on the position of the beta valve. When the power lever is moved from IDLE into the GROUND FINE and REVERSE ranges, it pulls the reverse lever and beta valve aft, opening it, allowing oil pressure to increase to the propeller dome which will decrease the blade angle. The distance traveled by the propeller dome, as the blade angle decreases, is fed back to the beta valve through the polished rods, feedback ring, and finally the reverse lever to pull the beta valve forward. This will close it, stopping the oil flow into the propeller dome thereby stabilizing the blade angle at the selected position. The opposite will occur when the power lever is moved forward to IDLE. The power lever will push the reverse lever and beta valve fully forward, relieving oil pressure from the propeller dome which will increase the blade angle. The distance traveled by the propeller dome, as the blade angle increases, is fed back to the beta valve through the polished rods, feedback ring, and finally the reverse lever to push the beta valve aft. The beta valve will be pushed aft until the port, that is relieving oil pressure from the propeller dome, is closed. This will stabilize the blade angle at the selected position.

The region between the GROUND LOW PITCH STOP and -3° blade angle is referred to as the GROUND FINE range. In this range, the engine's compressor speed (N_1) remains at the value it had when the power lever was at IDLE (62% to 70%, based on condition lever position).

To enter the GROUND FINE range, the power lever must be lifted beyond the IDLE gate and then moved aft. With aft movement of the power levers in GROUND FINE, blade angle will move progressively from the GROUND LOW PITCH STOP to -3° (at the GROUND FINE gate).

The region between -3° and -14° blade angle is referred to as the REVERSE range. In this range, N_1 progressively increases to a maximum value of 87 ± 1 percent while blade angle decreases. To

enter the REVERSE range, the power lever must be lifted beyond the GROUND FINE gate and moved aft. With aft movement of the power levers in REVERSE, blade angle will progressively decrease from -3° (at the GROUND FINE gate) to -14° (at MAXIMUM REVERSE).

Prop Pitch Annunciators

The L PROP PITCH and R PROP PITCH annunciators indicate to the pilot that the propeller blade angle has decreased below the flight low pitch stop. These PROP PITCH lights are required by the FAA to alert the pilot any time the propeller pitch changes more than 8° below the flight low pitch stop without any direct pilot action such as occurs when the propeller blade angle moves from the flight low pitch stop to the ground low pitch stop on landing touchdown. The system uses a magnetic proximity sensor to sense the position of the feedback ring and thereby sensing the position of the propeller dome and blade angle.

Ground Fine and Reverse Control Operation

As the blade angle decreases to approximately 15° , four flanges extending from the propeller dome make contact with four low pitch stop nuts which are threaded onto four polished rods (Figure 7-35). As the propeller blade angle continues to decrease, the propeller dome pushes the nuts and the polished rods forward. As the rods move forward, the feedback ring is pulled forward. In turn the beta valve, inside the primary governor, is pulled into the oil cutoff position. The linkage is set to cut off the oil supply to the dome when the blade angle reaches the FLIGHT LOW PITCH STOP. This provides the propeller with a hydraulic low pitch stop for inflight operations.

To help prevent the propeller rpm from dropping below 1050, for operation on the ground, the low pitch stop is reset from the flight low pitch stop to the ground low pitch stop when the aircraft is on the ground. This will be discussed later in this section.

When a power lever is lifted up and over the IDLE gate into the GROUND FINE range, it is

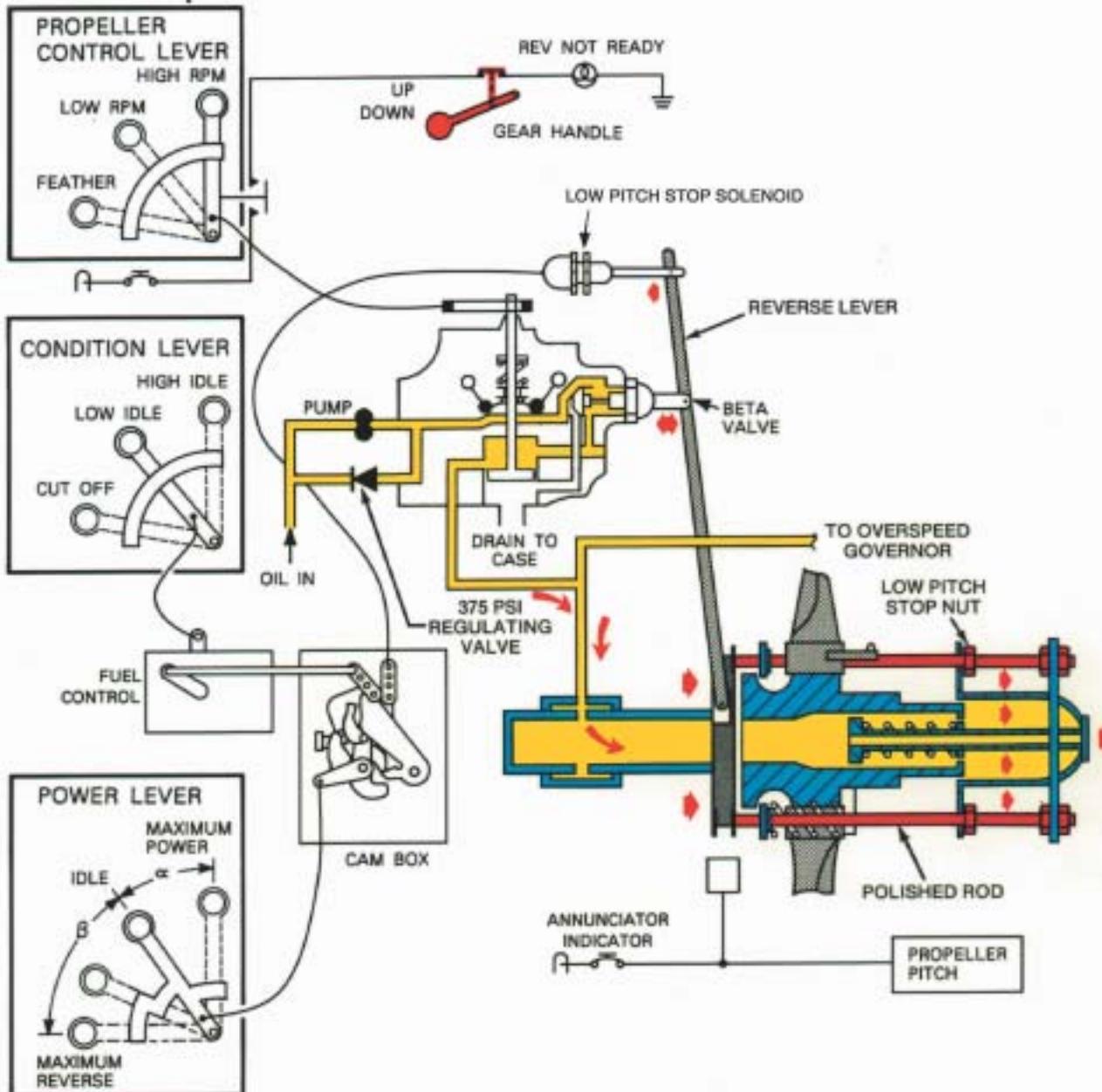


Figure 7-36 Ground Fine Range and Reverse Diagram

FOR TRAINING PURPOSES ONLY

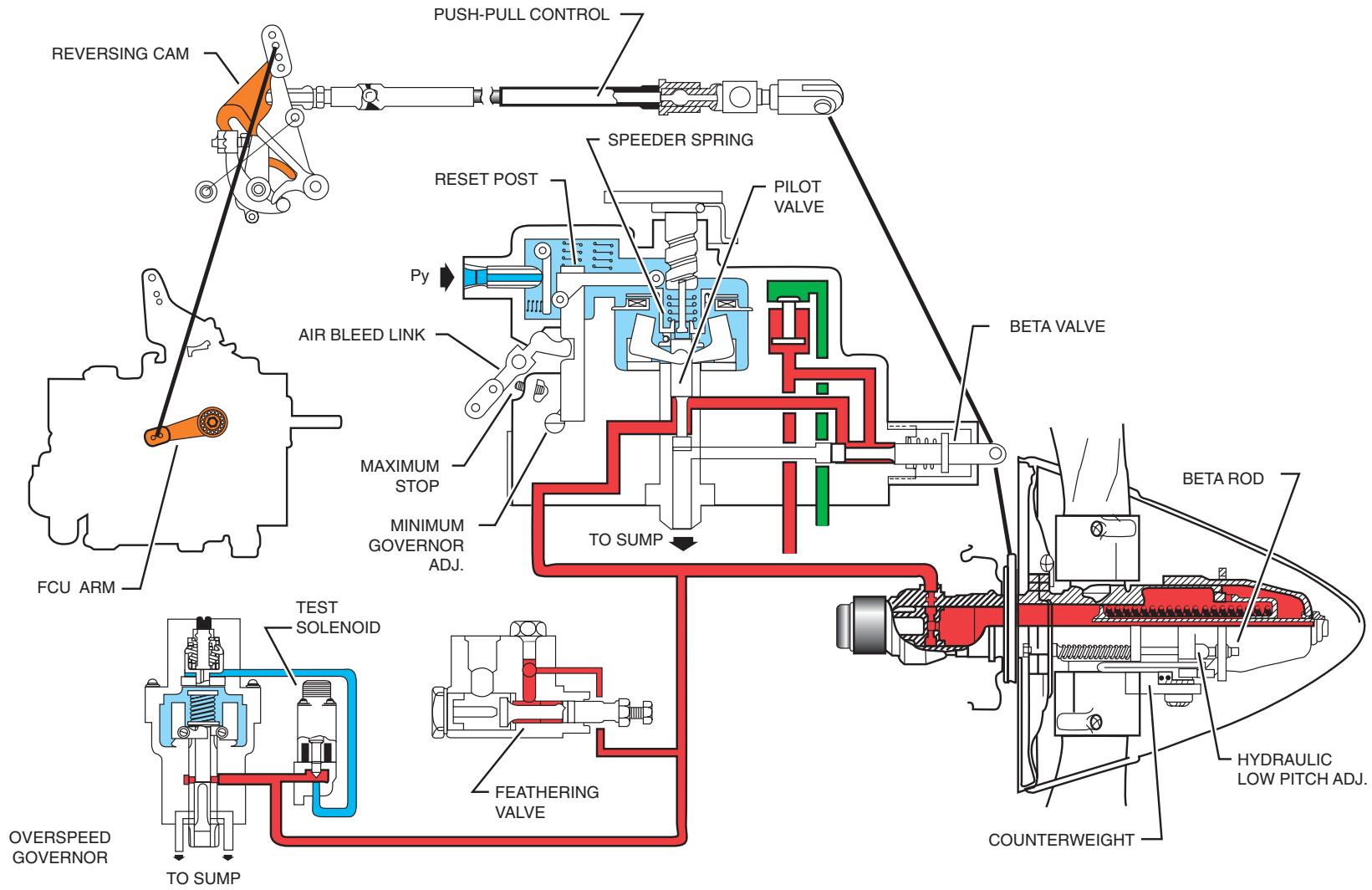
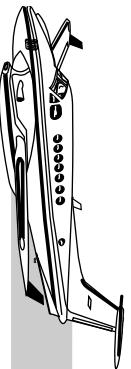


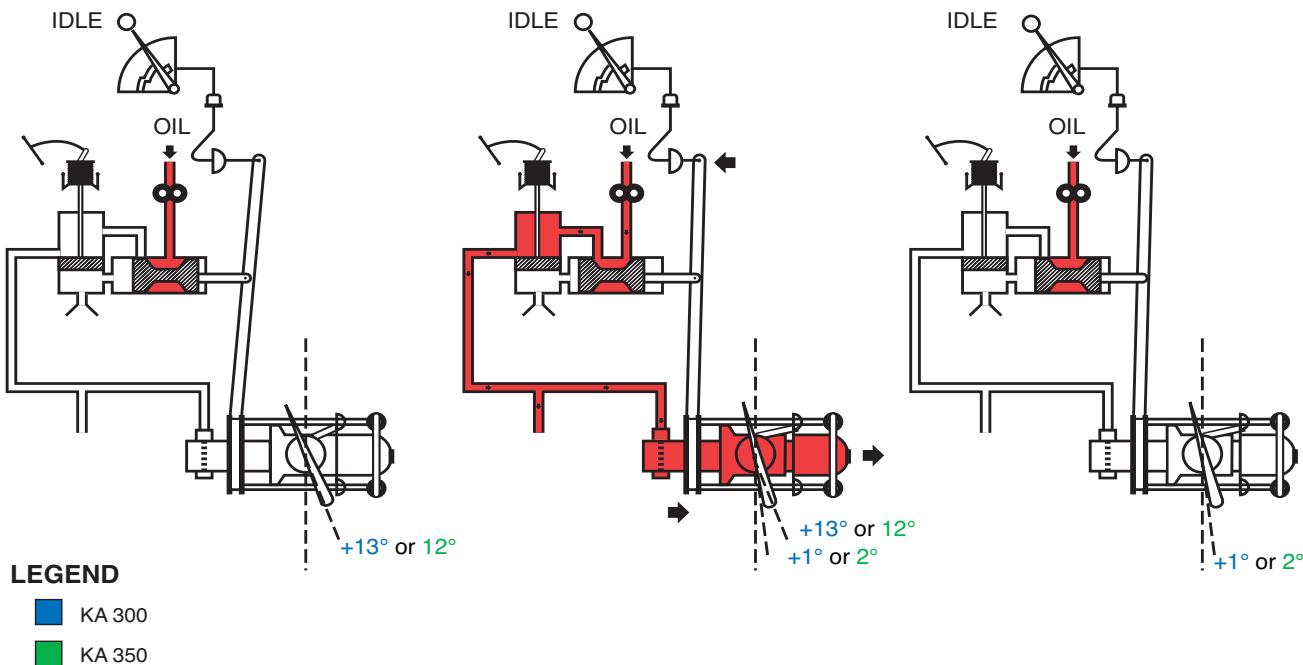
Figure 7-37 Propeller System Complete



KING AIR 300/350

PILOT TRAINING MANUAL

FlightSafety
International



FLIGHT IDLE

AS AIRCRAFT DECREASES AIR-SPEED, THE PROPELLER BLADE ANGLE DECREASES TO MAINTAIN PROPELLER GOVERNOR RPM AS THE BLADE ANGLE APPROACHES 13°, THE BETA VALVE IS PULLED INTO THE CUTOFF POSITION, CREATING A LOW PITCH STOP.

IN TRANSIT

AS THE AIRCRAFT TOUCHES DOWN, A SQUAT SWITCH ACTUATED ELECTRICAL SOLENOID REPOSITIONS THE BETA LEVER AFT, MOVING THE BETA VALVE INTO THE OPEN POSITION. THIS ALLOWS OIL TO FLOW TO THE PROPELLER DOME.

AS THE PROPELLER DOME FILLS WITH OIL, IT MOVES FORWARD, CARRYING THE BETA ROD AND LEVER ASSEMBLY, AND CONSEQUENTIALLY ROTATING THE PROPELLER BLADES TO A LOWER ANGLE.

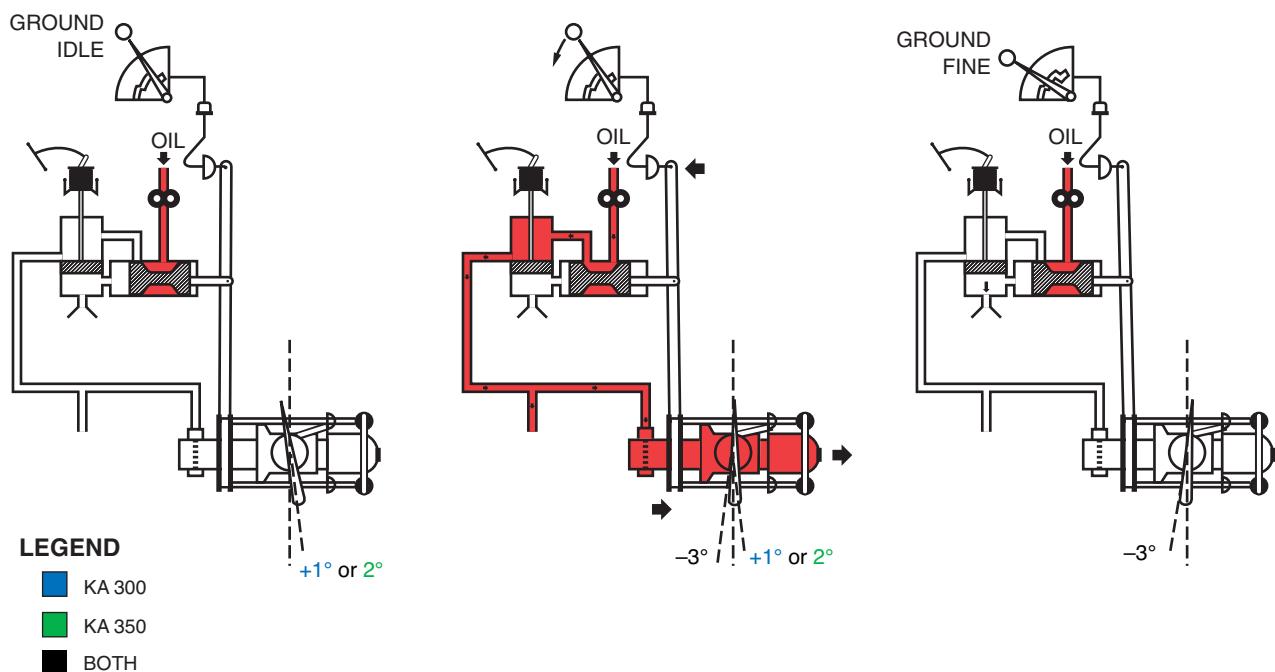
GROUND LOW PITCH STOP

AS THE PROPELLER DOME MOVES FORWARD, THE BETA VALVE REPOSITIONS THE LOW PITCH STOP. OIL IS AGAIN TRAPPED IN THE PROPELLER DOME, EFFECTIVELY CREATING THE GROUND LOW PITCH STOP.

Figure 7-38 Propeller Positioning—Flight Idle to Ground Low Pitch Stop (1 of 3)

pulling aft on the top of the reverse lever. As the reverse lever moves aft, the beta valve is pushed aft, re-establishing oil flow to the propeller dome which moves the propeller blade angle below the ground low pitch stop. As the propeller blade angle continues below the ground low pitch stop, the propeller dome and feedback ring continue

forward, eventually pulling the beta valve forward to the oil cutoff position. When a power lever is moved forward, away from or out of the GROUND FINE or REVERSE ranges toward IDLE, it pushes the reverse lever forward, which in turn pulls the beta valve fully forward. This is far enough to open a port so that oil is dumped



THE GROUND FINE IS A RANGE IN WHICH OPTIMUM AIRCRAFT CONTROL AND ENGINE PERFORMANCE ARE MAINTAINED DURING TAXI. ONCE THE AIRCRAFT IS ON THE GROUND, THE PILOT MOVES THE POWER LEVER AFT.

THIS MOVES THE BETA LEVER AFT AND REPOSITIONS THE BETA VALVE TO THE OPEN POSITION ALLOWING OIL TO FLOW TO THE PROPELLER DOME. AS THE PROPELLER DOME FILLS WITH OIL, IT MOVES FORWARD, CARRYING THE BETA ROD AND LEVER ASSEMBLY, FURTHER ROTATING THE PROPELLER BLADES TO A LOWER ANGLE.

THIS ACTION MOVES THE BETA VALVE TO THE CLOSED POSITION, TRAPPING OIL IN THE PROPELLER DOME, EFFECTIVELY CREATING A STOP AT ZERO THRUST OR GROUND FINE.

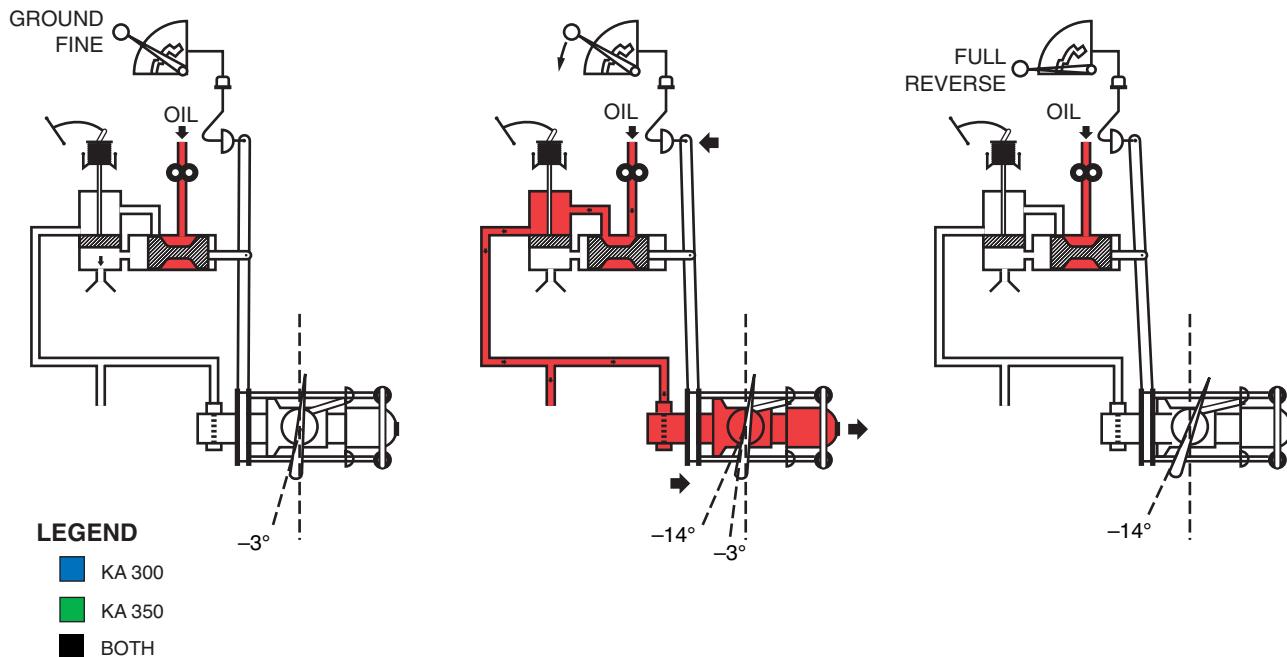
Figure 7-38 Propeller Positioning—Ground Low Pitch Stop to Ground Fine Gate (2 of 3)

from the propeller dome to the nose case. The propeller blade angle then increases until the polished rods and feedback ring, which are moving aft with the propeller dome, have pushed the reverse lever and the beta valve far enough aft to cut off the oil that is flowing from the propeller dome to the case. As a result, the position of the low pitch stop is controlled by the power levers. When the power levers are set at idle or above, while on the ground, the low pitch stop is set at 1° on the 300 or 2° on the 350. When a power

lever is moved aft of IDLE, the low pitch stop is repositioned to a blade angle of less than 1° or 2°.

Propeller Resonance

To avoid propeller resonance, maintain the propeller rpm above 1050 or below 400 during ground operations. The most severe resonance is in the range of 850 to 900 rpm, and is especially severe with a quartering tail wind. Sustained operation in this rpm zone exposes the propeller to increased resonance and stress.



GROUND FINE GATE

TO ENTER THE REVERSE RANGE FROM GROUND FINE, THE PILOT MUST LIFT UP ON THE POWER LEVER AND MOVE THE POWER LEVER AFT.

IN TRANSIT

AGAIN, THIS MOVES THE BETA LEVER AFT, MOVING THE BETA VALVE TO THE OPEN POSITION, ALLOWING OIL TO FLOW TO THE PROPELLER DOME, MOVING IT FORWARD. AS THE PROPELLER DOME MOVES FORWARD, IT CARRIES THE BETA ROD AND LEVER ASSEMBLY, FURTHER ROTATING THE PROPELLER BLADES TO A NEGATIVE ANGLE.

FULL REVERSE

THIS ACTION MOVES THE BETA VALVE TO THE CLOSED POSITION, TRAPPING OIL IN THE PROPELLER DOME, EFFECTIVELY CREATING A STOP FOR FULL REVERSE.

Figure 7-38 Propeller Positioning—Ground Idle to Full Reverse (3 of 3)

Since reduced blade angle results in an increased rpm without excessive thrust, the blade is automatically decreased from the flight low pitch stop upon landing touchdown. This decreased blade angle is called the ground low pitch stop.

Flight and Ground Low Pitch Stops

For normal operation on the ground, when the propeller is not in feather, the propeller rpm should be maintained above 1050 rpm. With the engine idling, operation with the propeller feathered is permissible because the speed is below the resonance rpm range. However, sustained

operation in feather on the ground should be avoided because excessive heat may build up in the nacelle, nose avionics area, and fuselage. Both a flight low pitch stop and a ground low pitch stop are utilized in this propeller system. In flight, the minimum blade angle is approximately 13° on the 300 or 12° on the 350 with the power levers at IDLE. While on the ground, the minimum blade angle is approximately 10° on the 300 or 2° on the 350 at IDLE. When the right main gear touches the ground on landing, the propeller blade automatically changes from the flight low pitch stop to the ground low pitch stop through the action of a ground low pitch stop solenoid on



each side. The ground low pitch stop solenoid acts on the top of the reverse lever in the same manner as the power levers when selecting GROUND FINE and REVERSE.

The right landing gear safety switch normally controls the low pitch stop. During the landing flare at idle power the propeller blade angle will be at the flight low pitch stop. Upon touchdown the landing gear safety switch causes the propeller blade angle to immediately decrease to the ground low pitch stop. As a backup for the right landing gear safety switch, a switch is also installed in the power quadrant to ensure the ground low pitch stop solenoids are activated when either or both of the power levers are lifted at the idle gate. The propeller blade angle remains at the ground low pitch stop until the power lever is moved aft of the IDLE gate.

When a ground low pitch stop solenoid is energized, after the primary governor is in an underspeed condition, the solenoid will pull aft on the reverse lever a small distance which, in turn, will push the beta valve aft, opening it, allowing oil pressure to increase to the propeller dome. The propeller dome will move forward and the blade angle will decrease. As the blade angle decreases, the propeller dome pulls the polished rods, feedback ring, reverse lever, and beta valve forward. When the blade angle reaches approximately 1° on the 300 or 2° on the 350, the beta valve has been pulled far enough forward to cut off and stabilize the oil pressure to the dome which stabilizes the blade angle at the ground low pitch stop.

The electric circuits of the ground low pitch stop solenoids are protected by the Prop Gov Test circuit breaker located on the right circuit breaker panel in the cockpit. Voltage to the ground low pitch stop solenoid is reduced from approximately 28 volts to approximately 14 volts, two seconds after the solenoids are energized. This is accomplished by resistors so that excess heat does not build up on the solenoids.

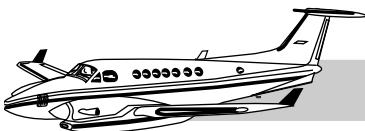
Keep in mind that even though the low pitch stop may have been changed to an angle less than that of the flight low pitch stop, this only affects the

actual blade angle after it has reached the flight low pitch stop. It follows, then, that as long as the primary governor stays onspeed (the indicated propeller rpm remains the same as that selected with the propeller control levers), selecting GROUND FINE or REVERSE with the power lever should not cause the propeller blade angle to move below the flight low pitch stop. Only when the primary governor stays in an underspeed condition (the indicated propeller rpm is less than that selected with the propeller control levers) should the propeller blade angle move below the flight low pitch stop or the ground low pitch stop, as the case may be, when the power lever is moved aft of the IDLE gate.

When the propeller blade angle is above the flight low pitch stop, the feedback ring is fully aft and the beta valve is pulled aft or open. The feedback ring is connected to the bottom of the reverse lever, the beta valve is connected to the middle of the reverse lever, and the reversing linkage from the power lever as well as the ground low pitch stop solenoid is connected to the top of the reverse lever. The reversing linkage and the ground low pitch stop solenoid is fully extended forward at this time. If the pilot attempts to select GROUND FINE or REVERSE, in this situation, the power levers would attempt to pull the reversing linkage and the reverse lever aft in an attempt to move the beta valve aft to open it. However, the beta valve is already aft and open, and the feedback ring cannot be moved forward by the reverse lever. This prevents the reverse lever and the reversing linkage from moving aft, and if forced, damage will occur to the reversing linkage.

King Air 300 Ground Low Pitch Stop System Operation On Takeoff

On the 300, the right landing gear safety switch will normally break the circuit to the ground low pitch stop solenoids as the aircraft lifts off on takeoff. The solenoids then allow the reverse levers and the beta valves to move forward a small fixed amount. However, this will have no effect on the flow of oil into the propeller dome because the beta valves -original positions were fully aft. The small forward movement of the



beta valves on lift off is not enough to affect the oil flow to the propeller domes when the primary governors are onspeed and the blade angles are above 13° (Figure 7-39).

King Air 350 Ground Low Pitch Stop System Operation On Takeoff

As the power levers are advanced on the King Air 350, a microswitch on each power lever breaks the circuit to the respective ground low pitch stop solenoid above 68 to 70 percent N₁. The result is that the blade angle changes from the ground low pitch stop of +2° to the flight low pitch stop of +12°. The propeller rpm decreases momentarily, because of the increased rotational drag, when the blade angle changes. This helps to prevent surging as power is added (Figure 7-40).

WARNING

Do not lift the power levers at the IDLE gate in flight. Doing so will energize the ground low pitch stop solenoids and cause the blade angle, if the primary governor is in an underspeed condition (the indicated propeller rpm is less than that selected with the propeller control levers), to decrease from the flight low pitch stop to the ground low pitch stop. This will cause excessive drag and the aircraft will develop a high sink rate.

CAUTION

Attempting to pull the power levers into the GROUND FINE and REVERSE ranges with the propellers in feather will cause damage to the reversing linkage of the power lever. Also, pulling the power levers into GROUND FINE and REVERSE on the ground with the engines shut down will damage the reversing system.

OVERSPEED GOVERNOR

The overspeed governor (Figure 7-41) provides protection against excessive propeller speed in the event of a primary governor malfunction. Since the PT6's propeller is driven by a free turbine (independent of the engine's compressor), overspeed can occur rapidly if the primary governor fails. It is located on the left side of the propeller reduction gearbox.

The overspeed governor is set at approximately 1768 rpm. The operation of the overspeed governor is very similar to that of the primary governor. However, there are 2 major differences. The pilot cannot select a particular speed, like he can with the primary governor, except when the overspeed governor is tested; and the overspeed governor can only reduce oil pressure to the propeller dome. The increasing of oil pressure to the dome can only come through the primary governor.

If a propeller overspeeds up to the value of the overspeed governor, we can safely assume that the blade angle is too low for the amount of power applied to the propeller. It is also assumed that the reason for the low blade angle is too much oil pressure in the propeller dome. If a propeller's speed reaches approximately 1768 rpm, the overspeed governor's flyweights are rotating fast enough to overcome the preset speeder spring tension. The flyweights will move out and, in turn, pull the overspeed governor's pilot valve up. This will allow oil pressure to be dumped from the propeller dome back to the case, thereby increasing the propeller blade angle and slowing the propeller down.

From a pilot's point of view, a propeller tachometer stabilized at approximately 1768 would indicate failure of the primary governor and proper operation of the overspeed governor. For pretakeoff check purposes, the set point of this governor can be rescheduled down to approximately 1565 rpm using the governor test switch on the pilot's left subpanel.

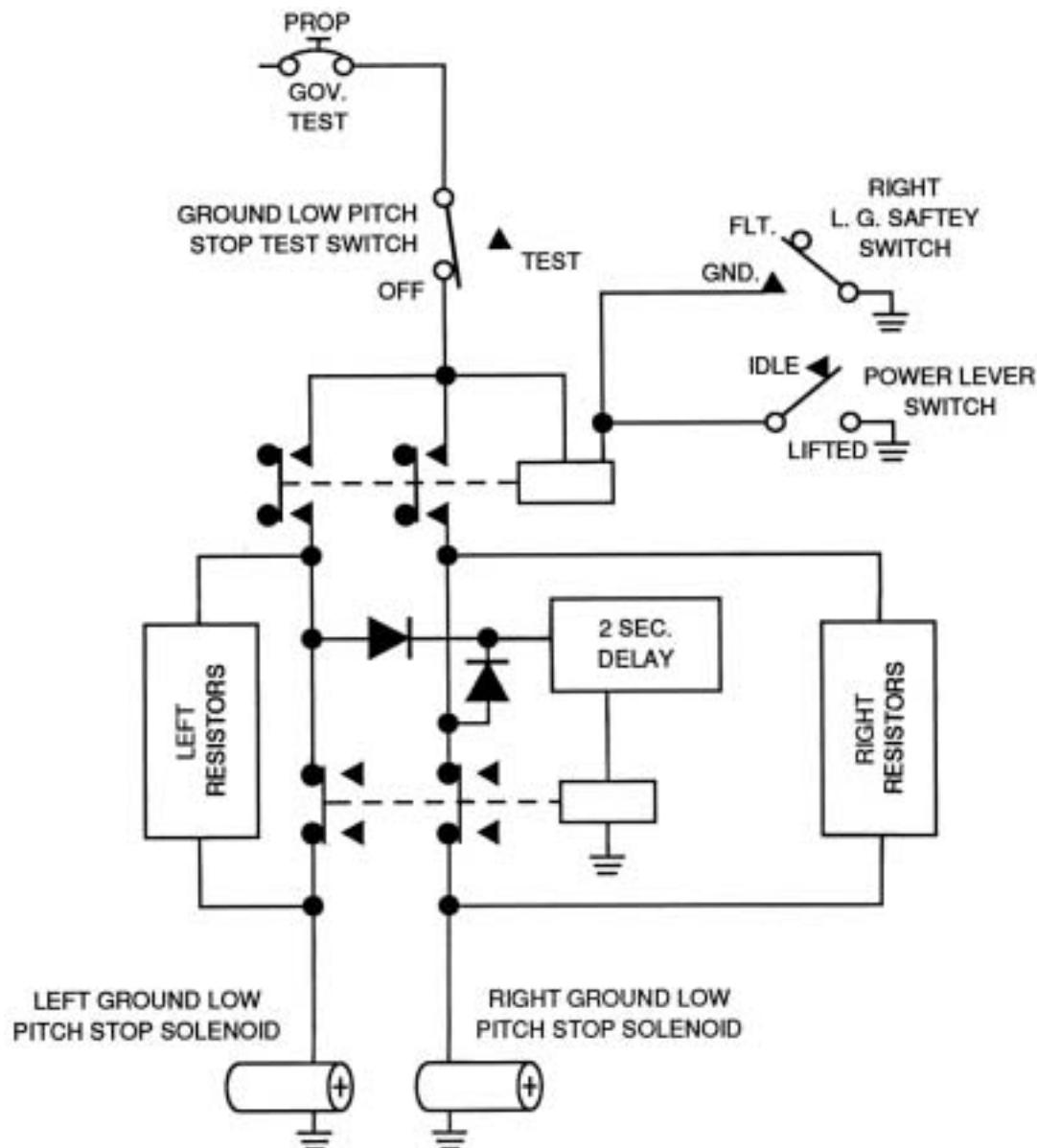


Figure 7-39 King Air 300 Ground Idle Stop Electrical Circuit

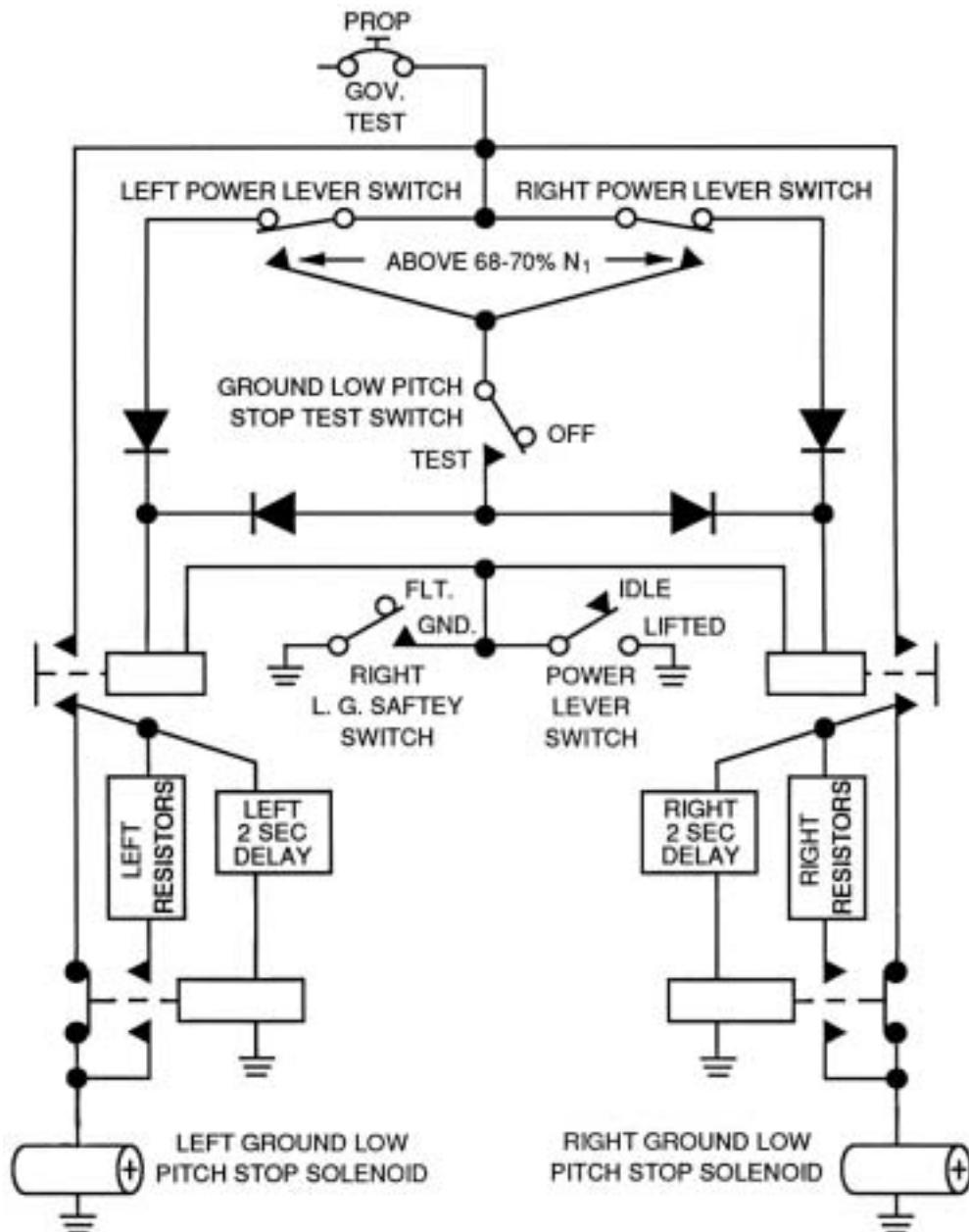
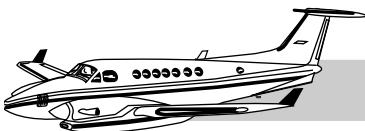


Figure 7-40 King Air 350 Ground Idle Stop Electrical Circuit



FUEL TOPPING GOVERNOR

The primary propeller governor contains a fuel topping governor (FTG) designed to help protect against propeller overspeeding in the event that the primary or overspeed governors have no effect on blade angle, such as in the case of a frozen propeller hub assembly. The FTG attempts to limit propeller speed by limiting the amount of fuel flowing to the fuel nozzles. It does this by reducing the P₃ air pressure in the FCU (fuel control unit). This will ultimately reduce the power applied to the propeller shaft and hopefully limit propeller rpm. The value at which the FTG attempts to limit propeller rpm, when the power lever is in the forward thrust range, is approximately 106 percent of selected propeller rpm. When the power lever is in the GROUND FINE or REVERSE range, the value at which the FTG attempts to limit propeller rpm is approximately 95 percent of selected propeller rpm. This assures that the primary

governor remains in an underspeed condition while in the REVERSE range on the ground.

POWER LEVERS

The power levers (Figure 7-42) are located on the power lever quadrant (first two levers on the left side) on the center pedestal. They are mechanically interconnected through a cam box to the fuel control unit, reverse lever, beta valve, and the fuel topping governor. The power lever quadrant permits movement of the power lever in the forward thrust range from IDLE to maximum thrust and in the GROUND FINE or REVERSE range from IDLE to maximum reverse. Gates in the power lever quadrant at the IDLE and GROUND FINE positions prevent inadvertent movement of the lever into the GROUND FINE or REVERSE range. The pilot must lift the power levers up and over these gates to select the GROUND FINE or REVERSE range

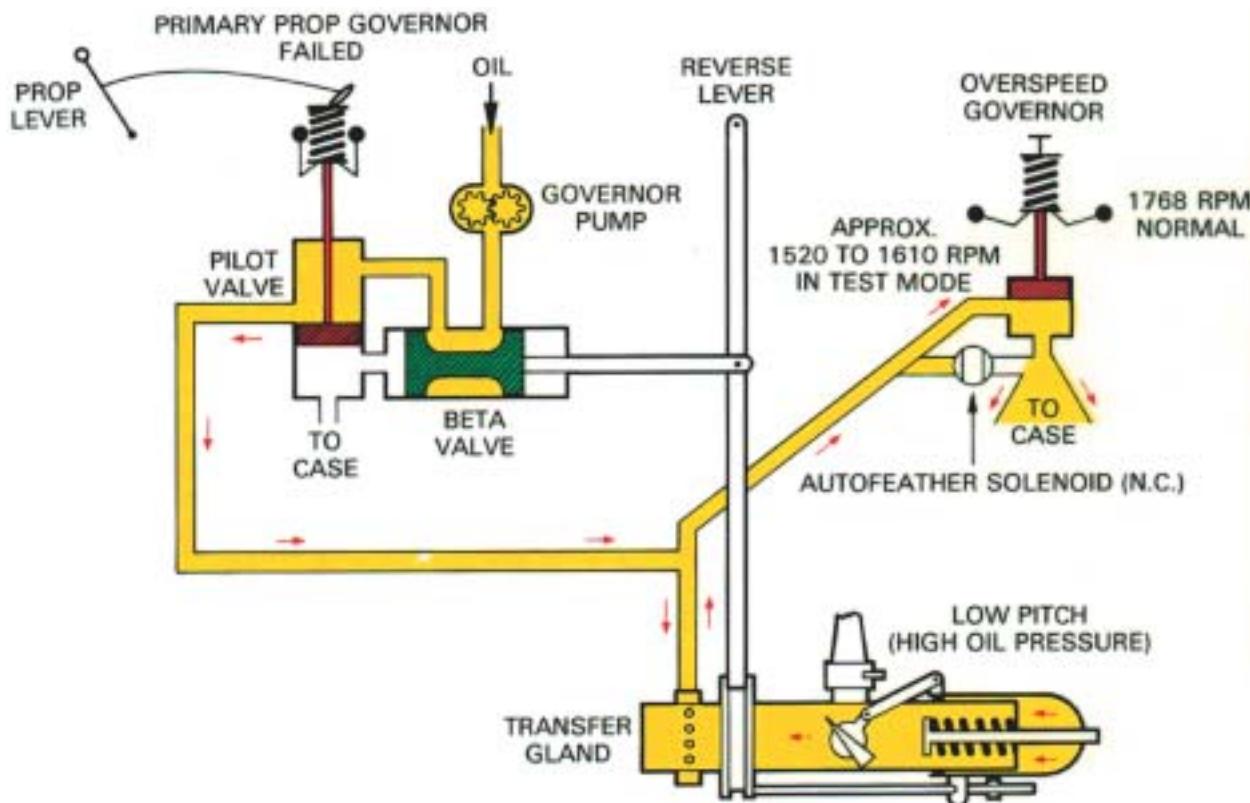


Figure 7-41 Overspeed Governor Diagram

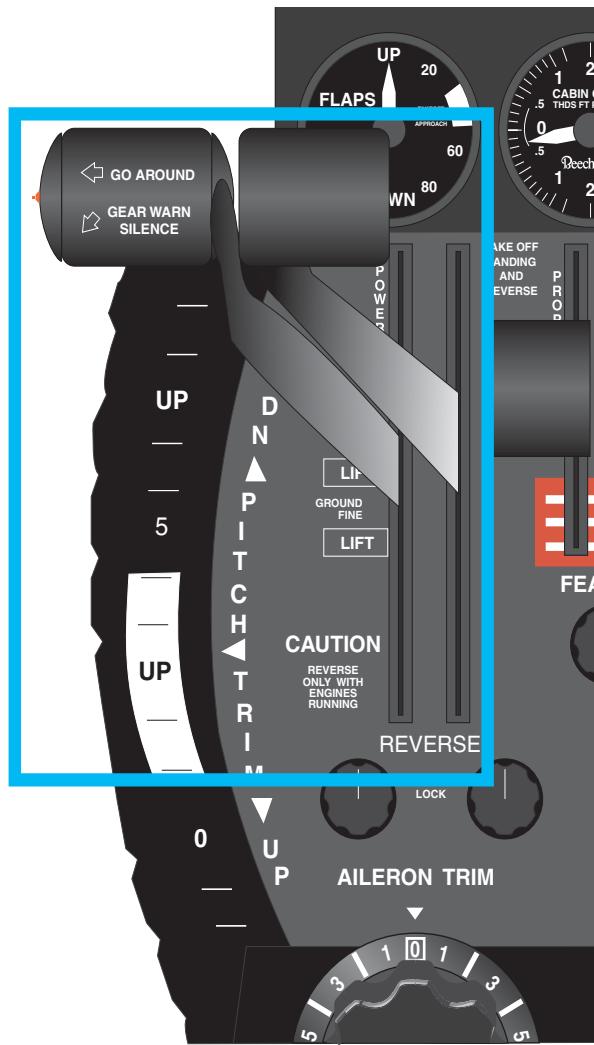
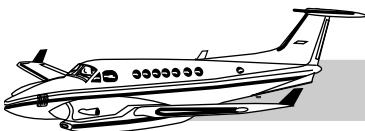


Figure 7-42 Power Levers

The function of the power levers in the forward thrust range is to select a gas generator rpm through the FCU (fuel control unit). THE FCU will then set a fuel flow that will produce and maintain the selected N_1 rpm.

In the GROUND FINE range, the power lever is used to: (1) select a propeller blade angle proportionate to the aft travel of the lever, thus reducing residual propeller thrust, (2) reset the fuel topping governor from its normal 106 percent to approximately 95 percent of selected propeller rpm. N_1 rpm is not affected in the GROUND FINE range.

In the REVERSE range, the power lever functions to (1) select a propeller blade angle proportionate to the aft travel of the lever, (2) select an N_1 that will sustain the selected reverse power, and (3) reset the

fuel topping governor from its normal 106 percent to approximately 95 percent of selected propeller rpm. Therefore, propeller rpm in the GROUND FINE or REVERSE range is a function of N_1 and power lever position, and may be limited by the FTG acting through the FCU to limit fuel flow and, consequently, propeller rpm in relation to power lever position.

PROPELLER CONTROL LEVERS

Propeller rpm, within the primary governor range of 1450 to 1700 rpm, is set by the position of the propeller control levers (Figure 7-43). These levers, one for each engine, are located between the power levers and the fuel cutoff levers on the center pedestal. The full forward position sets the primary governor at 1700 rpm. In the full aft position forward of the feathering detent, the pri-

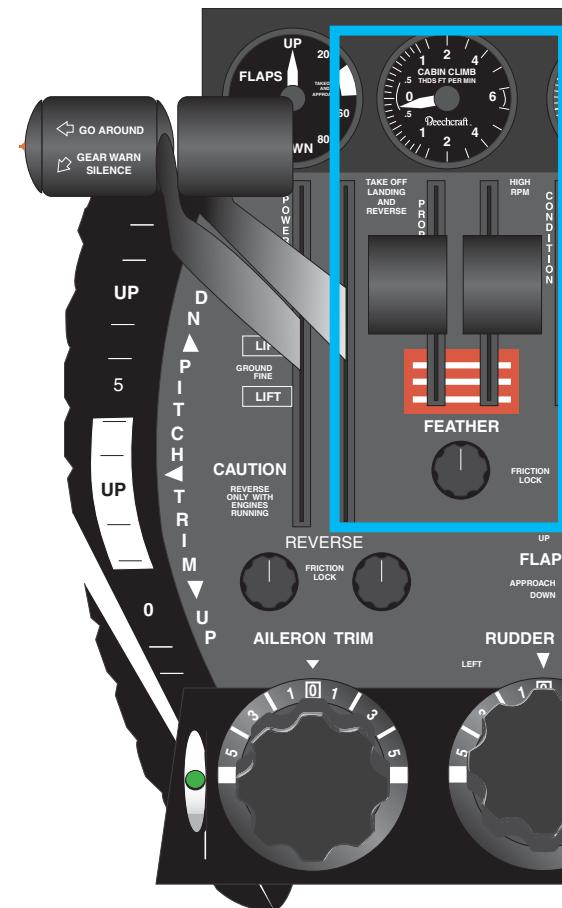


Figure 7-43 Propeller Control Levers



mary governor is set at 1450 rpm. Intermediate propeller rpm positions can be selected by moving the propeller levers to select the desired rpm as indicated on the propeller tachometer. These tachometers read directly in revolutions per minute. A detent at the low rpm position prevents inadvertent movement of the propeller lever into the FEATHER detent, indicated by the red and white stripes across the lever slots in the quadrant.

PROPELLER FEATHERING

The propeller can be feathered by moving the propeller lever full aft into the feather detent. This opens the feathering dump valve. All oil pressure quickly drains from the propeller dome and the propeller feathers. In this type of turbine engine, the propeller shaft and N₁ shaft are not connected. Thus, the propeller can be feathered with the engine running at idle power without damaging the engine or gearbox. Without the autofeather system armed, and the engine is shut down in cruise flight, the propeller will stay onspeed and in sync unless it is manually feathered or all oil pressure is lost. At the feather position, the propeller lever positions the feathering dump valve to dump oil pressure from the propeller dome. This allows the counterweights and springs to position the propeller blades at the feather position.

AUTOFEATHER SYSTEM

The automatic feathering system provides a means of immediately dumping oil from the propeller dome, thus enabling the feathering spring and counterweights to feather the propeller blades in the event of an engine failure. The system is controlled by a switch on the subpanel, placarded AUTOFEATHER ARM-OFF TEST. When the switch is in the ARM position (Figure 7-44), the completion of the arming phase occurs when both power levers are advanced above 88% N₁ and both torque indications are above 17%, at which time both the right and left autofeather annunciators indicate a fully armed system.

The annunciator lights are green, placarded L AUTOFEATHER and R AUTOFEATHER. When the switch is in the ARM position, the system will be inoperative as long as either power lever is retarded below 88% N₁ position.

Autofeather is required to be operable for all flights and armed for takeoff, climb, and approach. When the autofeather switch is in the ARM position, and starting with both power levers above 88% N₁ as well as both torque indications above 17%, the feathering springs and counterweights will move the propeller blades, for a particular engine, toward feather after two things happen:

1. The torque manifold oil pressure for that engine has dropped below the torquemeter value of 17%, which disarms the opposite engine autofeather electrical circuit and is evidenced by the extinguishing of its autofeather annunciator.
2. The torque continues to drop below 10%, which feathers its propeller and is evidenced by the extinguishing of its autofeather annunciator.

If the torque for the last operating engine drops below 17% or either power lever is retarded below 88% N₁, the autofeather system will be completely disarmed.

When the autofeather switch is in the TEST position (Figure 7-45), it works exactly as explained above except that:

1. The power levers do not have to be above 88 percent N₁. However, both torque indications must start from above 17 percent, preferably 22%, because the TEST position on the switch bypasses both power lever switches.
2. When an engine failure is simulated with a power lever (Figure 7-46), the associated autofeather annunciator will flash off and on when the power lever is pulled to IDLE, because there is still idle power.



King Air 300 and 350 Autofeather System Test

The system may be tested as follows (Figure 7-47):

1. POWER LEVERS—APPROXIMATELY 22% TORQUE. 22% torque is a power setting sufficient to simulate normal operation of both engines.
2. AUTOFEATHER Switch—Hold to TEST. This action completes a circuit bypassing the 88% autofeather arming switches inside the pedestal, allowing the test to be performed at power settings below the normal operating range of the system. Both autofeather annunciators should illuminate when the switch is moved to TEST.

3. POWER LEVERS—Retard individually. “Engine-out” situations are now simulated by retarding each engine power lever individually. At approximately 17% torque, the autofeather annunciator should extinguish for the engine opposite the one being retarded. This check verifies that the autofeather system disarms itself on the “operative” engine.

Continuing to retard the power lever will result in initiation of the feathering action at approximately 10% torque. At this time the propeller on the simulated inoperative engine will try to feather, indicated by a blinking annunciation light and fluctuating torque. The autofeather annunciation will cycle on and off with each fluctuation of torque as the propeller moves in and out of feather.

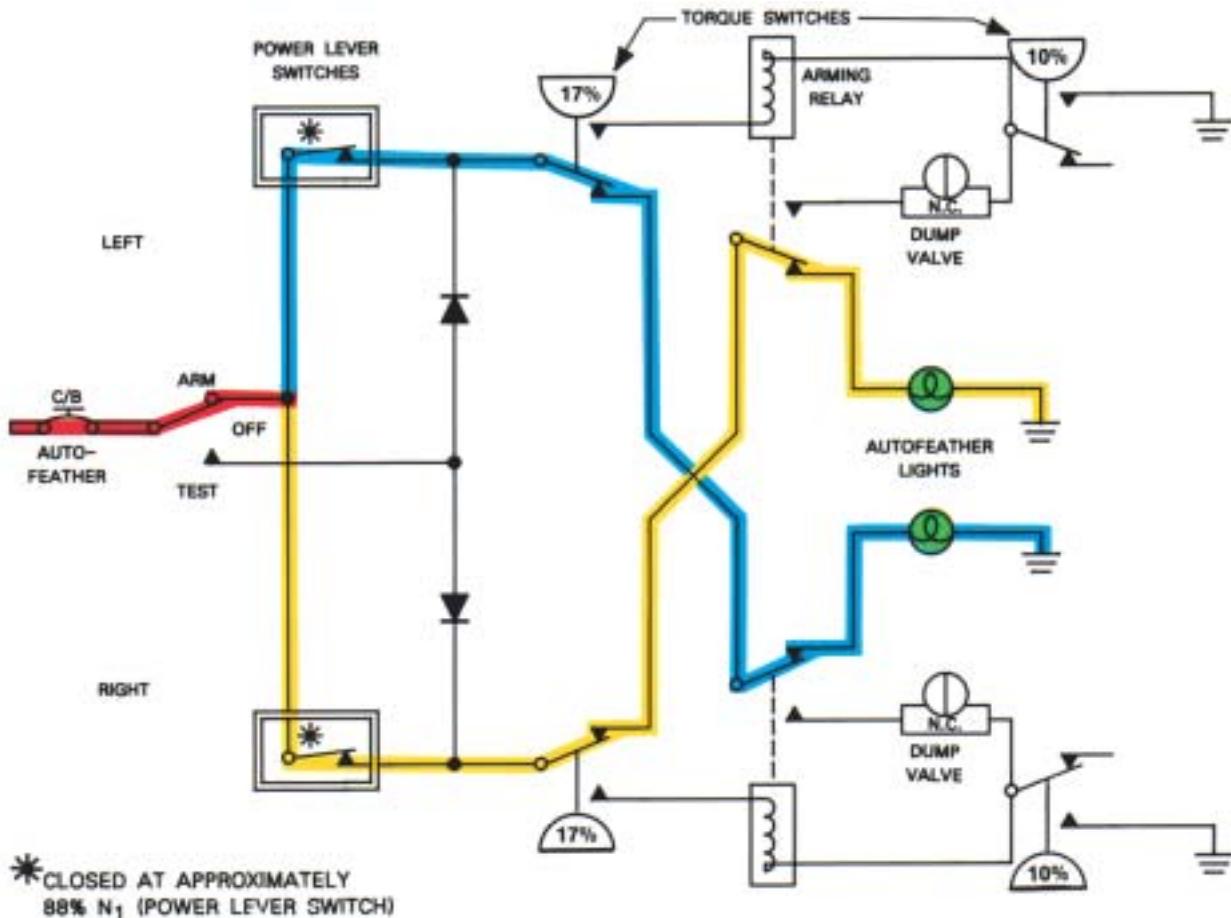


Figure 7-44 Autofeather Diagram—Armed

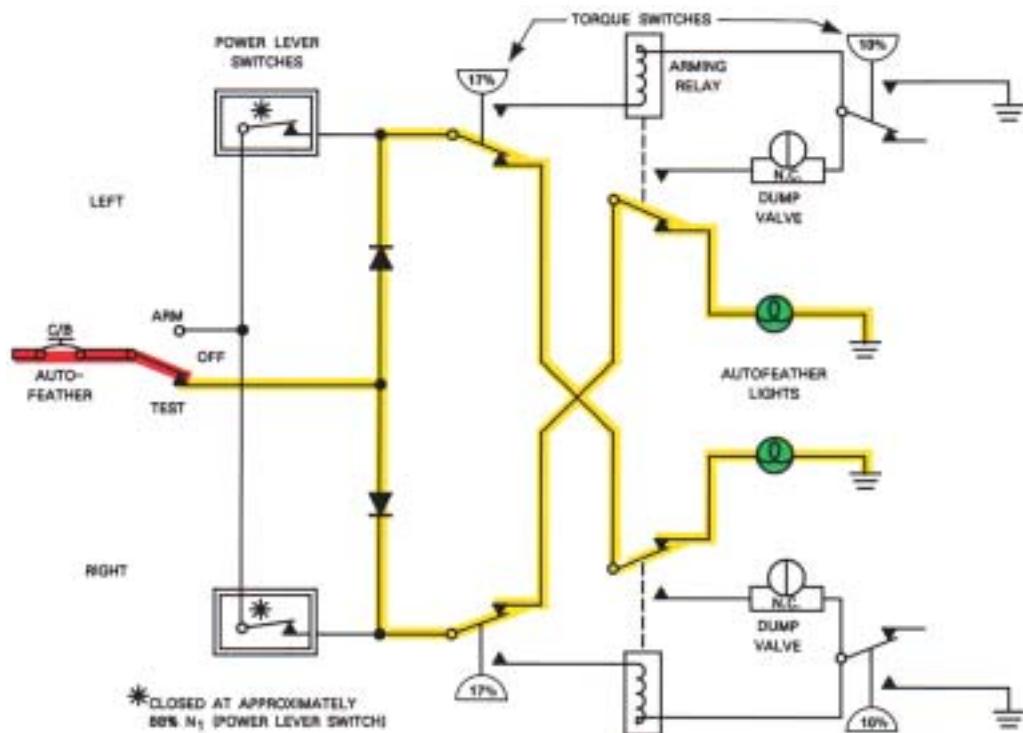


Figure 7-45 Autofeather Diagram—Test

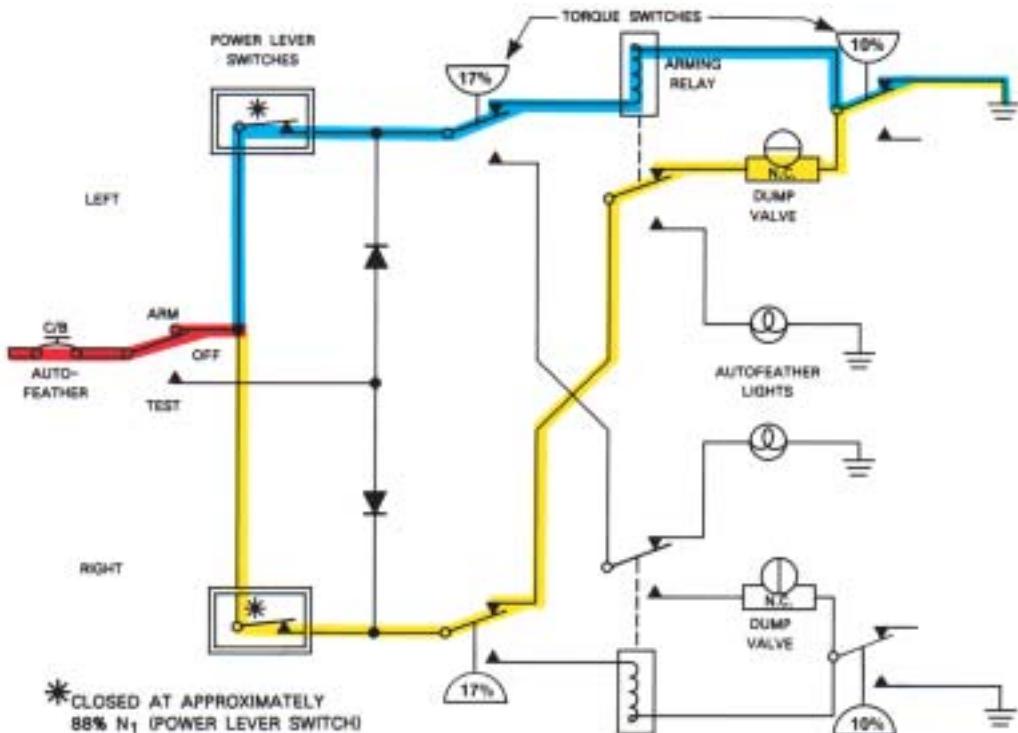


Figure 7-46 Autofeather Diagram—Left Engine Failure Armed



4. POWER LEVERS—Both retarded to IDLE. Upon completion of Step “3” for each power lever, check that with both levers retarded while holding the autofeather switch to TEST, that neither propeller feathers. The loss of both autofeather lights verifies this disarming of both systems due to the intentional reduction of power. Upon completion of this test, arm the system for takeoff.

Then, with the condition levers in LOW IDLE, propeller feathering (manual, as compared to AUTOFEATHER) is checked. In this free-turbine engine the propeller may be allowed to completely feather with the compressor operating at LOW IDLE with no engine damage sustained. Operation on the ground and in feather for extended periods of time may overheat the fuselage and possibly damage nose-mounted avionics because hot exhaust gases are not being blown aft by the propeller's air blast.

PROPELLER SYNCHROPHASER

The propeller synchrophaser (Figure 7-48) automatically matches the rpm of the two propellers and maintains the blades of one propeller at a predetermined relative position with the blades of the other propeller. The purpose of the system is to reduce propeller beat and cabin noise from unsynchronized propellers.

Synchrophaser Operation

The synchrophaser system is an electronic system, certified for use during all flight operations including takeoff and landing. It is not a designated master-slave system as it functions to match the rpm of the slower propeller to the faster propeller and establish a blade phase relationship between them to reduce cabin noise to a minimum.

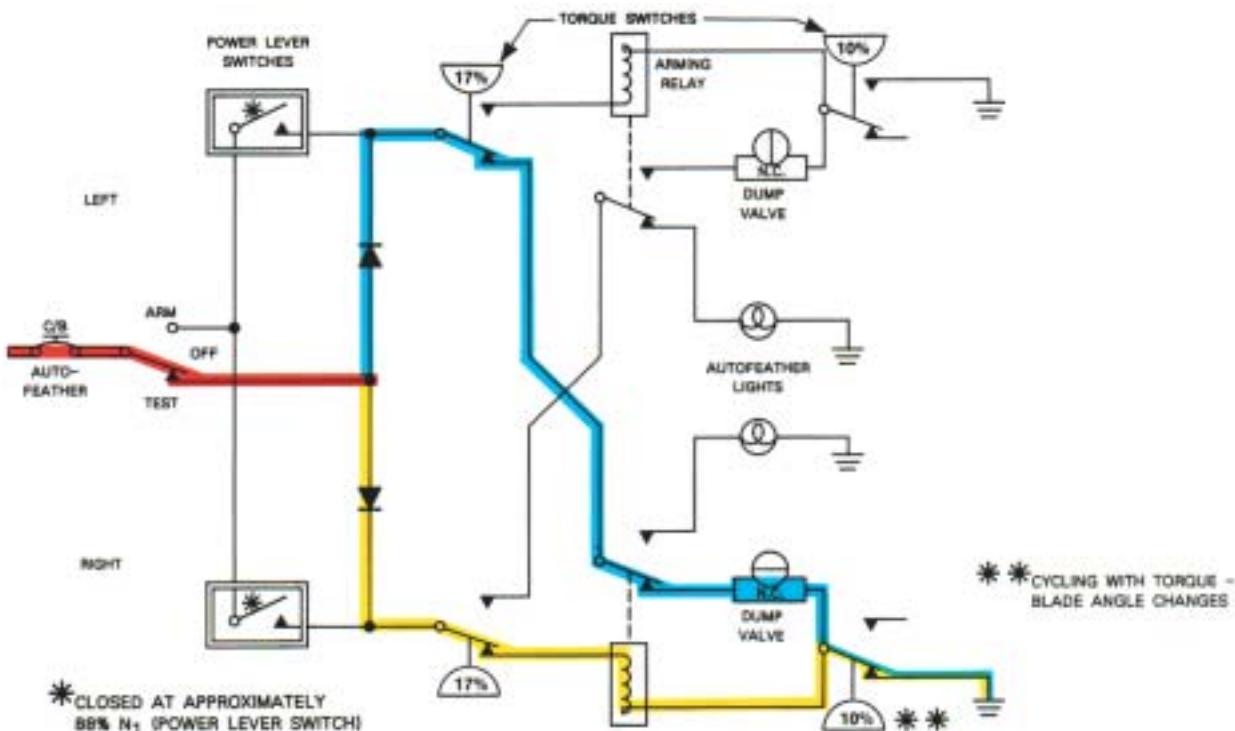


Figure 7-47 Autofeather Test Diagram (Right Engine)—Low Power and Feathering

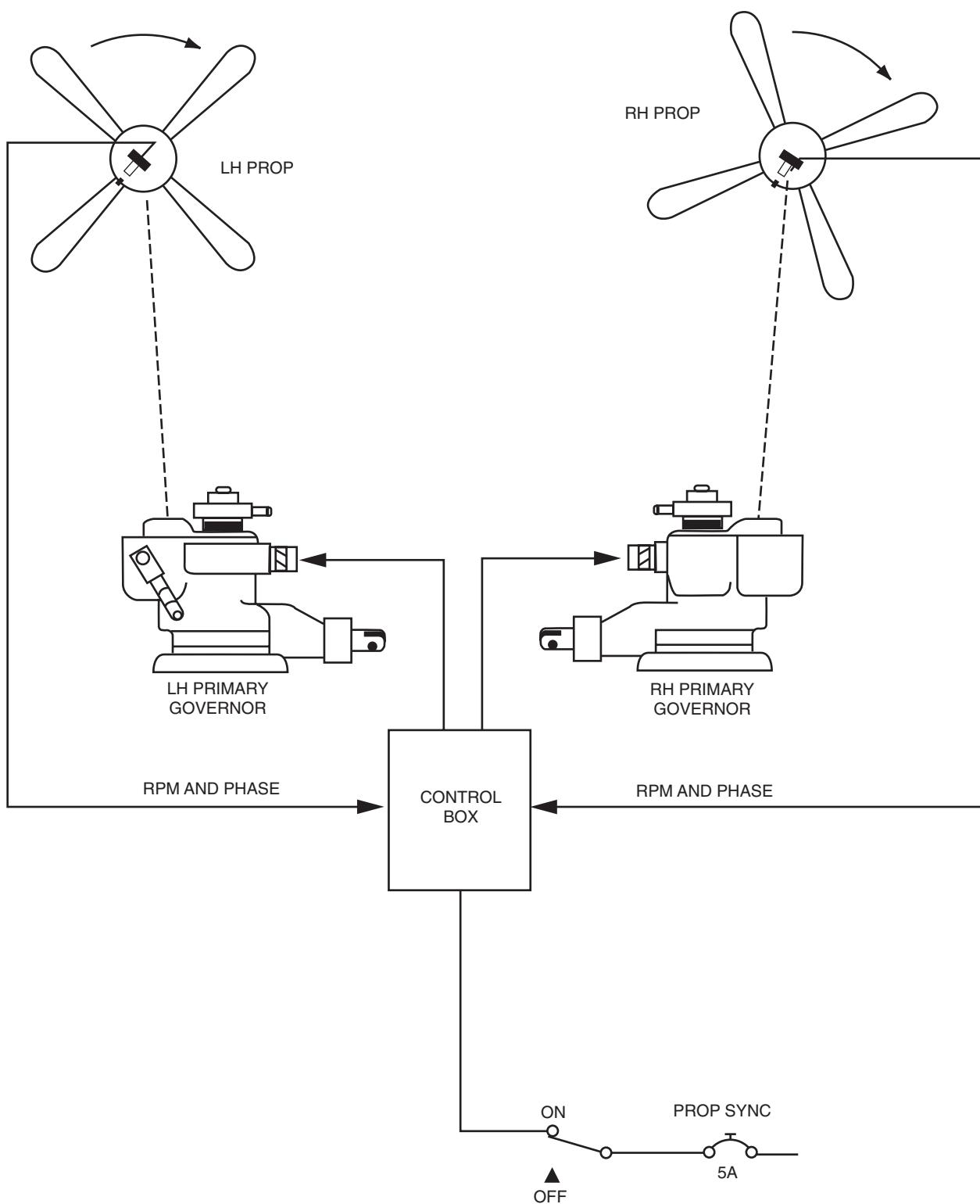


Figure 7-48 Propeller Synchrophaser System



The synchrophaser has a limited range of authority in relation to the primary governor setting. The maximum increase possible is approximately 20 rpm. In no case will the rpm fall below that selected by the propeller control lever. Normal governor operation is unchanged but the synchrophaser will continuously monitor propeller rpm and reset either governor as required.

Propeller rpm and position is sensed by a magnetic pickup mounted adjacent to each propeller spinner bulkhead. This magnetic pickup will transmit electrical pulses once per revolution to a control box installed forward of the pedestal.

The control box converts any pulse rate differences into correction commands, which are transmitted to coils mounted close to the flyweights of each primary governor. By varying the coil voltage, the governor speed settings are biased until the propeller rpms exactly match. A toggle switch on the 300 or a pushbutton on the 350 installed adjacent to the synchroscope turns the system on. In the synchrophaser OFF position, the governors operate at the manual speed settings selected by the pilot. To operate the synchrophaser system, synchronize the propellers manually or establish a maximum of 20 rpm difference between the propellers, then turn the synchrophaser on. The system may be on at all times unless a malfunction is indicated.

To change rpm with the system on, adjust both propeller controls at the same time. If the synchrophaser is on but does not adjust the propeller rpms to match, the system has reached the end of its range. Increasing the setting of the slow propeller, or reducing the setting of the fast propeller, will bring the speeds within the limited synchrophaser range. If preferred, turn the synchrophaser switch off, resynchronize manually, and turn the synchrophaser on.

Propeller Synchroscope

A propeller synchroscope is located to the left of the oil pressure/ temperature indicators and gives the status of propeller synchronization. The face of the synchroscope has a black and white cross pattern which can spin either left or right. If the right propeller rpm is greater than the left, the face turns clockwise or right. With the left propeller rpm greater than the right, the face turns counterclockwise or left. No rotation of the face indicates that both propellers are synchronized.



CHAPTER 8

FIRE PROTECTION

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CHAPTER 8

FIRE PROTECTION



INTRODUCTION

On the King Air 300 and 350 fire detection and fire-extinguishing systems provide fire protection in both engine compartments. Detection is automatic, while extinguishing is manually activated.

DESCRIPTION

ENGINE FIRE DETECTION SYSTEM

An engine fire detection system (Figures 8-1 and 8-2) is installed to provide immediate warning in the event of a fire in either engine compartment. The main element of the system is a temperature-sensing tube looped continuously around the engine and terminating in a responder unit. This

responder unit is mounted in the engine accessory area at approximately the ten o'clock position. The responder contains two sets of contacts. The integrity switch contacts are for the continuity test functions of the fire detection circuitry and the alarm switch contacts complete the circuit to activate the fire warning system when the detector tube senses an overtemperature condition in critical areas around the engine.

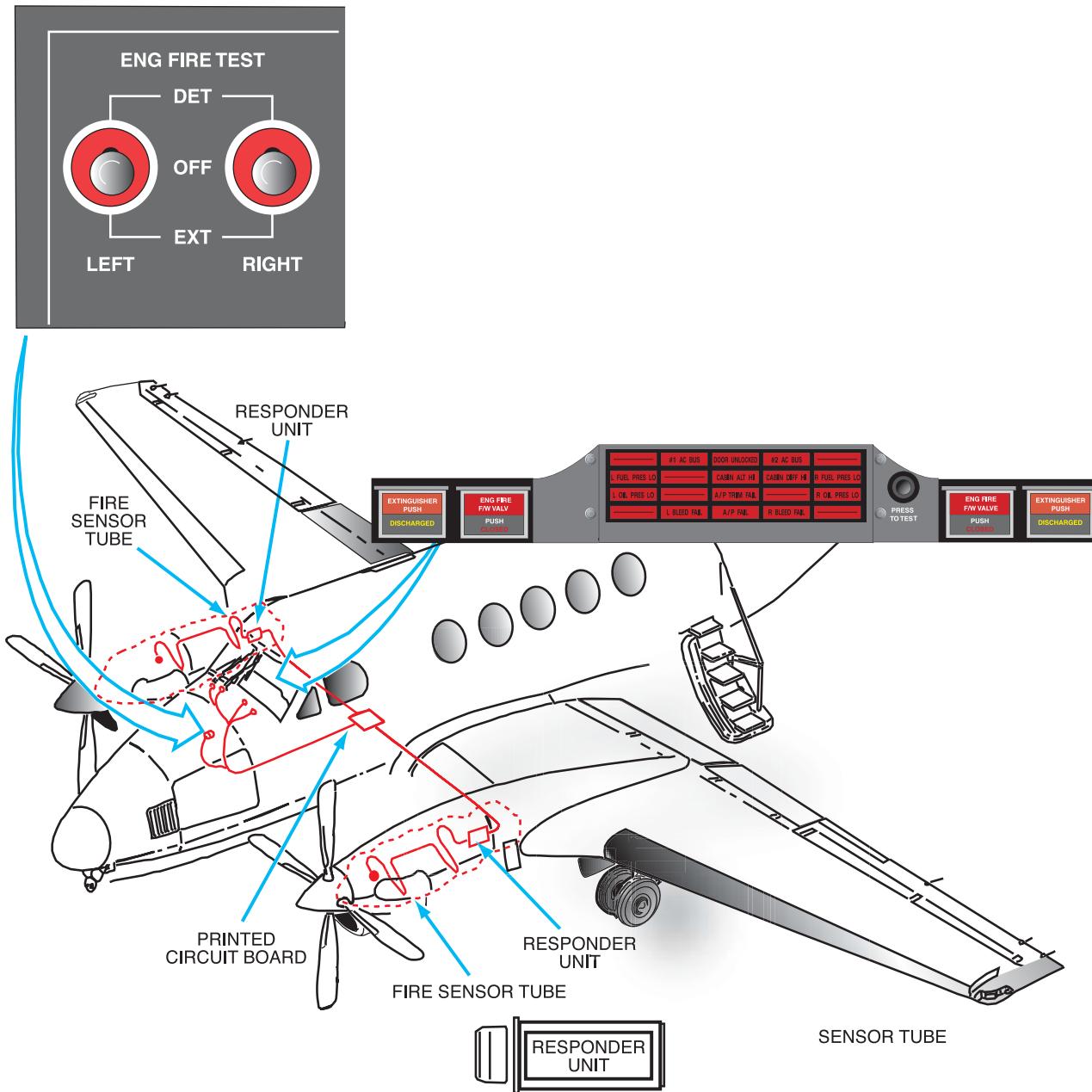


Figure 8-1 Engine Fire Detection System FL-122 and After



Signals from each responder unit are transmitted to a printed circuit board mounted forward of the main spar underneath the center aisle floor. A signal will activate a red warning light in the respective firewall fuel valve pushbutton, which is located on the glareshield just below the warning annunciator panel.

These pushbutton-type indicators have split lenses and serve the dual functions of monitoring the position of the firewall fuel valve and giving a visual warning of an overtemperature condition in either engine area.

The annunciator in the firewall fuel valve pushbutton, placarded "L" or "R FW VALVE CLOSED," indicates the firewall fuel valve position.

When the battery switch is on, this annunciator can indicate three situations.

- FW VALVE CLOSED annunciator extinguished—Firewall fuel valve is open.
- FW VALVE CLOSED annunciator illuminated—Firewall fuel valve is closed.
- FW VALVE CLOSED annunciator flashing—Firewall fuel valve position does not agree with the firewall fuel valve switch position.

The red lens, placarded "L" or "R ENGINE FIRE," illuminates when the fire detector tube senses an overtemperature condition of sufficient magnitude to activate the alarm switch contacts of the responder unit. The L or R ENGINE FIRE annunciator triggers the master warning flasher.

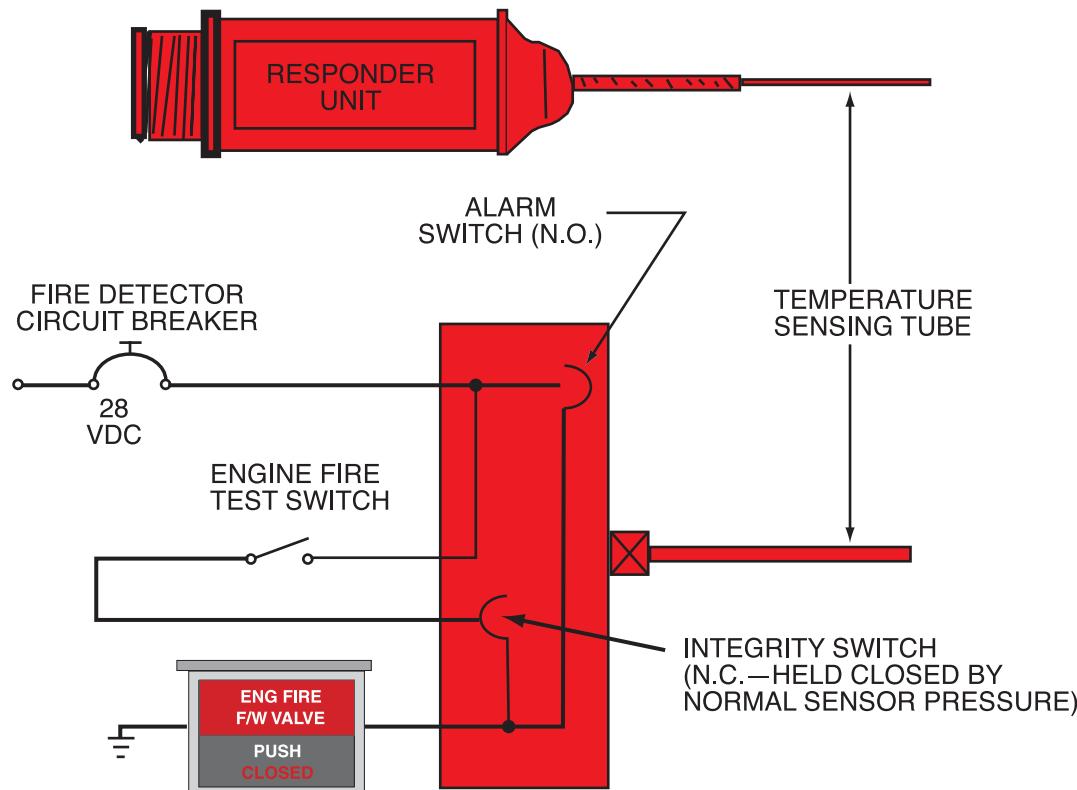
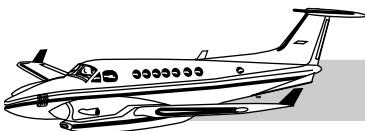


Figure 8-2 Engine Fire Detection System Simplified Schematic



The sensor tube consists of a hermetically sealed, corrosion-resistant, stainless-steel outer tube filled with an inert gas and an inner hydride core that is filled with an active gas. The gases within the tubes form a pressure barrier that keeps the contacts of the responder integrity switch closed for the fire detection system test.

As the temperature around the sensor tube increases, the gases within the tube begin to expand. When the pressure from the expanding gases reaches a factory preset point, the contacts of the responder alarm switch close, completing a circuit to the printed circuit board for the fire detector. From the printed circuit board, the signal is routed to the red L or R ENGINE FIRE light in the firewall fuel valve pushbutton, which will illuminate.

The fire detector is preset at the factory to activate the alarm when any of the following conditions are met:

- Any one-foot section of the tube is heated to 800°F.
- The average temperature of the entire tube reaches 360°F.
- The temperature in the accessory compartment reaches 545°F.
- The temperature in the hot section compartment reaches 540°F.

NOTE

The fire sensor tube section in the plenum chamber area is 2 feet long and is not a fire or overheat zone. Activation temperature in this zone is approximately 900°F.

Fire Detection System Test

Two toggle switches, located on the copilot's left subpanel, are used to test the integrity of the fire detection system. These switches, placarded "ENG FIRE TEST, DET-OFF-EXT, LEFT-RIGHT," are three-position, center-off type.

When either switch is placed in the DET position, electrical current flows from a 5-amp circuit breaker, located on the right circuit-breaker panel and placarded "RIGHT" or "LEFT FIRE DET," through the engine fire detector circuitry to activate the red L or R ENG FIRE light in the firewall fuel valve pushbutton, located on the glareshield, directly under the warning annunciator panel (300 and 350 prior to FL120) or adjacent to warning annunciator panel (FL121 and after). If either annunciator fails to illuminate during the test for that side a malfunction with the sensor tube, responder unit, annunciator, or electrical portion of the system is indicated. The malfunction must be corrected prior to flight.

ENGINE FIRE-EXTINGUISHING SYSTEM

A fire extinguisher supply container is mounted on brackets aft of the main spar in each wheel well of each main landing gear (Figure 8-3). Each cylinder is charged with 2.50 pounds of bromotrifluoromethane (CBrF₃) pressurized to 450 to 475 psi at 72°F. The line from the container runs along the side of the nacelle and branches into eight spray tubes strategically located about the engine to diffuse the extinguishing agent in the event of a fire. Four of the nozzles are positioned to discharge into the engine exhaust area and the other four discharge into the accessory area. Once activated, the entire supply of extinguishing agent is discharged for that side.

The fire extinguisher pushbuttons, used to activate the system, are located on the glareshield at each end of the warning annunciator panel. The power for these switches is derived from a 5-amp circuit breaker, placarded "RIGHT" or "LEFT ENGINE FIRE EXT," located on the hot battery bus. The push-to-activate pushbuttons incorporate two indicator lights.

The electrical circuit is routed from the 5-amp circuit breaker to the firewall fuel valve pushbutton first. The firewall fuel valve pushbutton must be in the closed position to complete the circuit to the fire extinguisher pushbutton. The fire extinguisher may be discharged any time the cir-

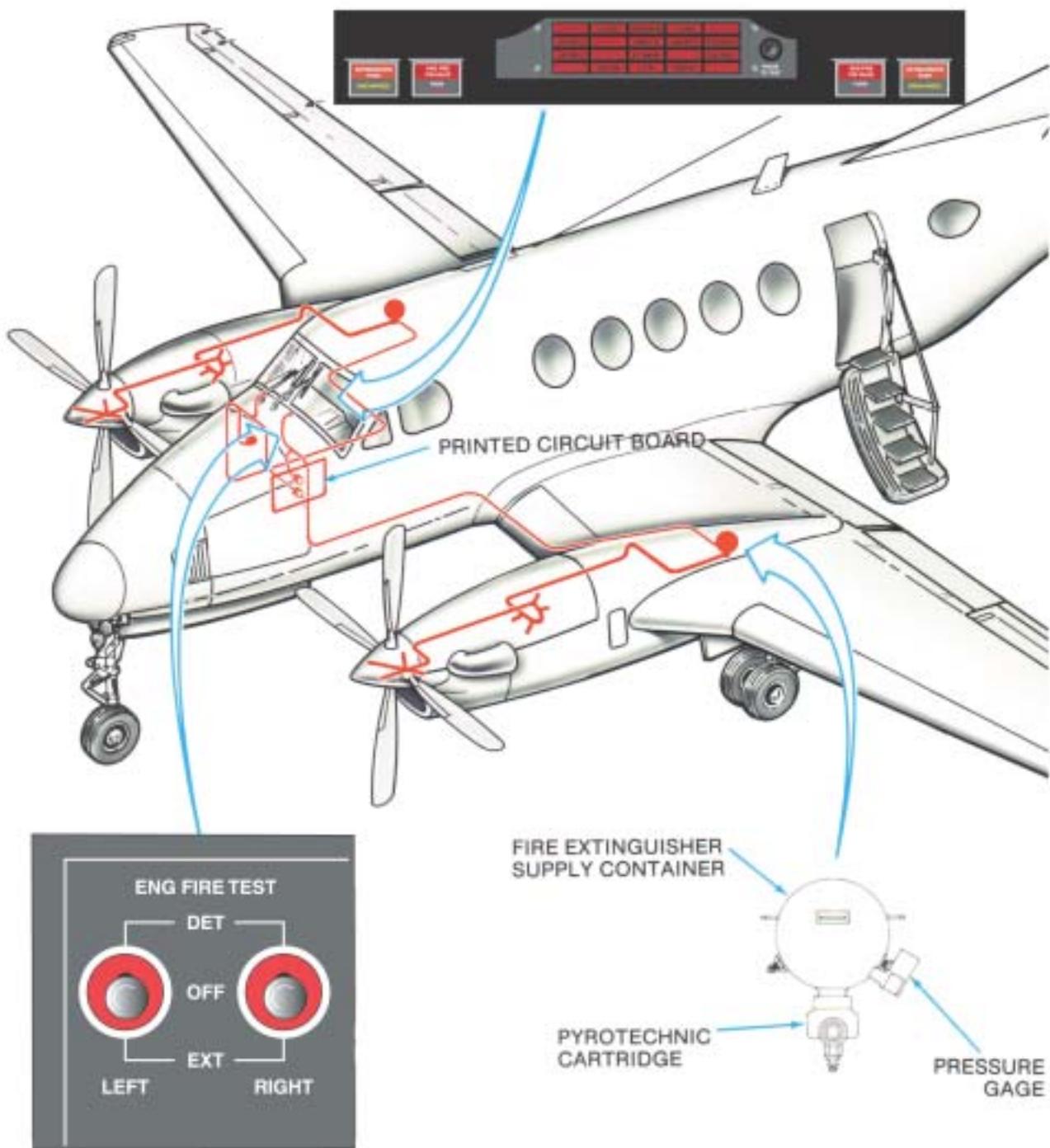


Figure 8-3 Fire-Extinguishing System FL-121 and After



cuit is complete, regardless of battery switch position. A red annunciator in the upper half of the fire extinguisher pushbutton, placarded "EXTINGUISHER PUSH," indicates that the circuit is complete.

The amber DISCHARGED annunciator in the lower half of the pushbutton uses a sensor to monitor the condition of the pyrotechnic cartridge in the container. As long as the sensor circuit is intact, the amber annunciator will remain extinguished. When the cartridge is fired and the sensor circuit is broken, the annunciator will illuminate and will remain illuminated until the cartridge is replaced, regardless of battery switch or firewall fuel valve pushbutton position.

Engine Fire Extinguisher System Test

Two toggle switches located on the copilot's left subpanel are used to test the functions of the fire extinguishing system. These switches, placarded "ENG FIRE TEST, DET-OFF-EXT, LEFT-RIGHT" are three-position, center-off type.

In the EXT position of the test switch for a particular side, a circuit is completed from the 5-amp circuit breaker on the hot battery bus, through the EXT test switch, EXTINGUISHER PUSH annunciator, cartridge on the container, and then to ground. Illumination of the EXTINGUISHER PUSH annunciator indicates that the cartridge circuit is in good order.

At the same instant, the cartridge sensor is also checked when the EXT test switch breaks the cartridge sensor circuit for that side. As a result, the DISCHARGED annunciator should also illuminate, indicating that the cartridge sensor is in good order.

Failure of either annunciator to illuminate on a particular side during a test for that side indicates a malfunction of the engine fire extinguisher system. The malfunction must be corrected prior to flight.

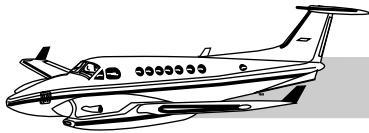


CHAPTER 9

PNEUMATICS

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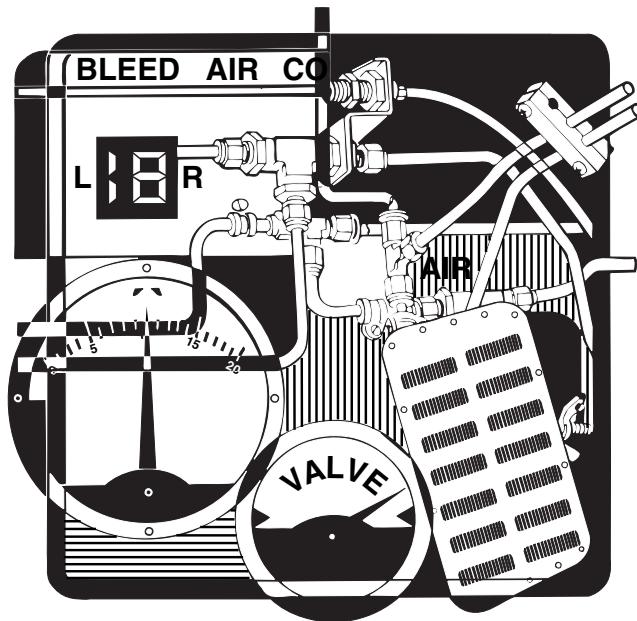
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CHAPTER 9

PNEUMATICS



INTRODUCTION

Many small but important tasks are accomplished by the aircraft pneumatic and vacuum systems. This section identifies these systems, their controls, and proper system utilization.

DESCRIPTION

The Pneumatic and Vacuum Systems section of the training manual presents a description and discussion of pneumatic and vacuum. The sources for and uses of each and acceptable

gage readings are discussed. The method used for detection of system leaks is described. Normal and abnormal/ emergency operations are discussed.

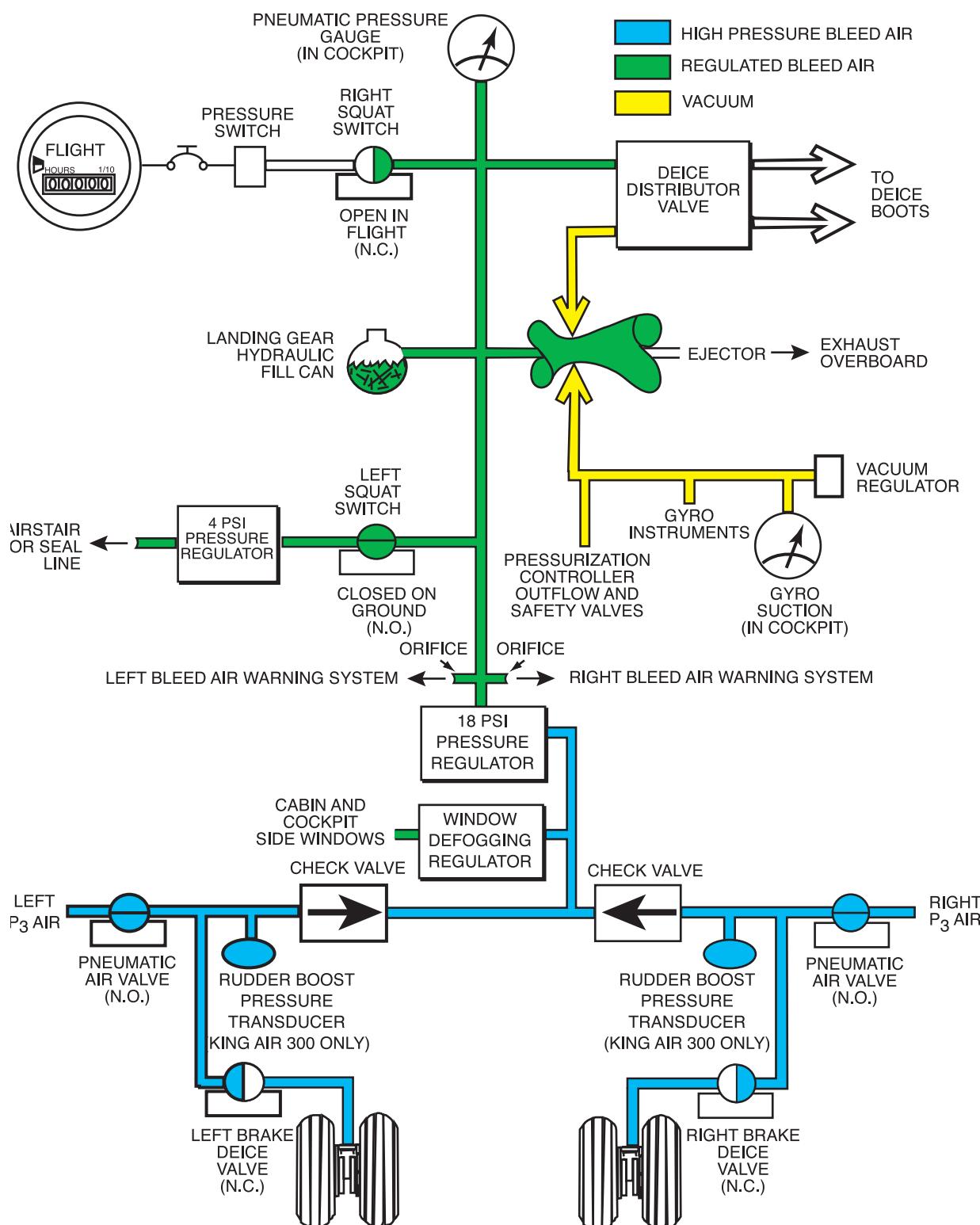


Figure 9-1 Pneumatic System Diagram



ENGINE BLEED AIR PNEUMATIC SYSTEM

The pneumatic and vacuum systems (Figure 9-1) in the Super King Air 300 and 350 provide support for several operations on the airplane. These operations include flight hourmeter and hydraulic fill can pressure, surface deice, rudder boost (300 only), brake deice, bleed air warning, window defogging, and the door seal. Pneumatic pressure is used to create a vacuum source for the air-driven gyros, pressurization control and deflation of the deice boots.

High-pressure P_3 bleed air from each engine compressor is routed through the normally-open, firewall-mounted shutoff valves; into the fuselage and then is regulated to 18 psi. It supplies pressure for the pneumatic systems and provides a vacuum source (Figure 9-1). Vacuum is derived from a bleed air ejector.

Normally, one engine can supply sufficient bleed air for all pneumatic and vacuum systems. During single-engine operation, a check valve in the bleed air line from each engine prevents loss of pressure back through the supply line on the side of the inoperative engine.

A pneumatic pressure gage, located on the copilot's right subpanel, indicates air pressure available to the pneumatic manifold in pounds per square inch. To the left of the pressure gage is a suction gage, calibrated in inches of mercury, which indicates instrument vacuum.

PNEUMATIC AIR SOURCE

P_3 bleed air at 90 to 120 psi pressure is obtained from both engines and flows through pneumatic lines to a tee junction located in the fuselage. Check valves are installed to prevent reverse flow during single-engine operation.

Downstream from the tee the P_3 air passes through an 18 psi regulator which incorporates a relief valve set to operate at 21 psi in case of regulator failure. This regulated pneumatic air supplies pressure to inflate the surface deicers and door seal, to the flight hourmeter, bleed air failure warning and hydraulic systems, and to provide flow for the vacuum ejector.

Bleed air is extracted from the fourth stage of the engine compressor at a maximum temperature of approximately 700° F. It has cooled to approximately 70° above ambient temperature by the time it reaches the tee in the fuselage due to heat transfer in the pneumatic plumbing.

Ordinarily, the pneumatic system pressure regulator, which is mounted under the right seat deck immediately forward of the main spar, will provide 18 ± 1 psi with the engine running at 70% to 80% N_1 . The pneumatic pressure gage on the copilot's right subpanel is provided to allow monitoring of system pressure (Figure 9-2).

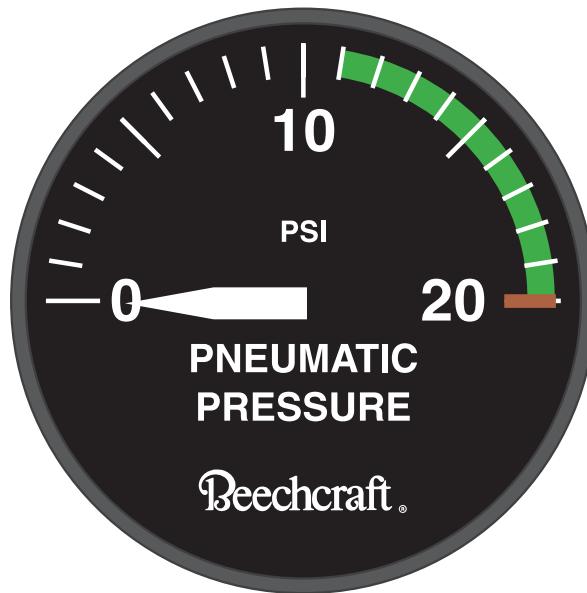


Figure 9-2 Pneumatic Pressure Gage

VACUUM AIR SOURCE

Vacuum is obtained from pneumatic air passing through a vacuum ejector. The ejector consists of a pressure line that constricts forming a venturi. At the apex (point of lowest pressure), the vacuum line is attached. As P_3 air passes through the ejector, it draws air from the attached "vacuum" line, thereby creating the suction for the vacuum system.



The ejector is capable of supplying from 15 inches of mercury (Hg) vacuum at sea level to 6 inches Hg vacuum at 31,000 feet. It supplies vacuum regulated 4.3 to 5.9 inches Hg for the deice boots, air-driven gyro instruments and the pressurization control system. Even with one engine running at 70% to 80% N₁, the vacuum gage normally should read approximately 5.9(+0,-0.2) inches Hg at sea level (Figure 9-3).

switches (Figure 9-4) which are placarded OPEN, ENVIR OFF, and PNEU & ENVIR OFF. When the switches are in the OPEN position, both environmental flow control valves and pneumatic instrument air valves are open. When the switches are in the ENVIR OFF position, the environmental flow control valves close and the pneumatic instrument air valves remain open. In the PNEU & ENVIR OFF position, all valves are closed.

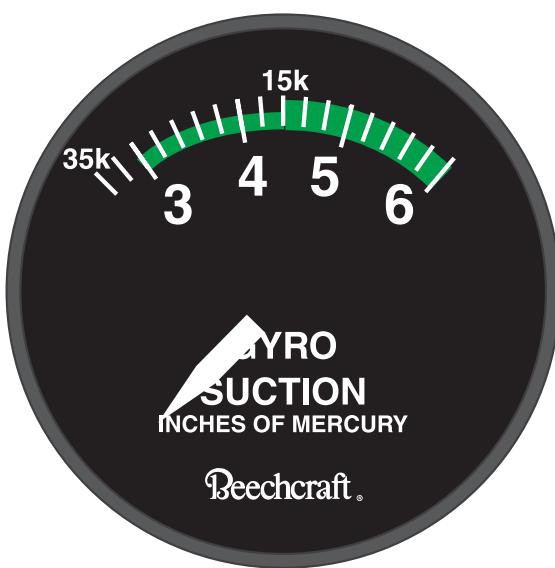


Figure 9-3 Gyro Suction Gage

The vacuum line for the instruments is routed through a regulator that is designed to maintain sufficient vacuum for proper operation of the instruments. The vacuum regulator is mounted in the nose compartment on the left side of the pressure bulkhead. The valve is protected by a foam filter. A filter located in the nose compartment provides a filtered and sheltered air source for the air-driven gyros.

BLEED AIR CONTROL

Bleed air entering the cabin, used for pressurization and environmental functions, is controlled by the two BLEED AIR VALVE

ENGINE BLEED AIR WARNING SYSTEM

The engine bleed air warning system provides a visual indication of a rupture in the bleed air lines so that the bleed air valves from the affected engine can be shut off before heat from the escaping air damages the skin and structure adjacent to the break in the line (Figure 9-5).

Ethylene vinyl acetate (EVA) tubes are in close proximity with the bleed air lines (environmental and pneumatic) leading from the engines to the cabin (Figure 9-6). These EVA tubes are pressurized by 18 psi pneumatic air tapped off from the pneumatic manifold. The system has two pressure switches mounted under the cockpit floorboard, one for each side. Excessive heat on the tubing, caused by a ruptured bleed air line, will melt the tubing. When the tubing melts, the air escapes and pressure inside the tubing decreases. When it drops below 2 psi, the normally-closed switch in the line closes, completing the circuit to the respective BL AIR FAIL annunciator in the warning panel. When the indication of a bleed air failure becomes evident, all bleed air for that side must be turned off by placing the respective BLEED AIR VALVE switch in the PNEU & ENVIR OFF position. With the switch in this position, both the environmental and pneumatic shut-off valves close, stopping bleed air flow at the engine firewall. This action will not extinguish the BL AIR FAIL light.

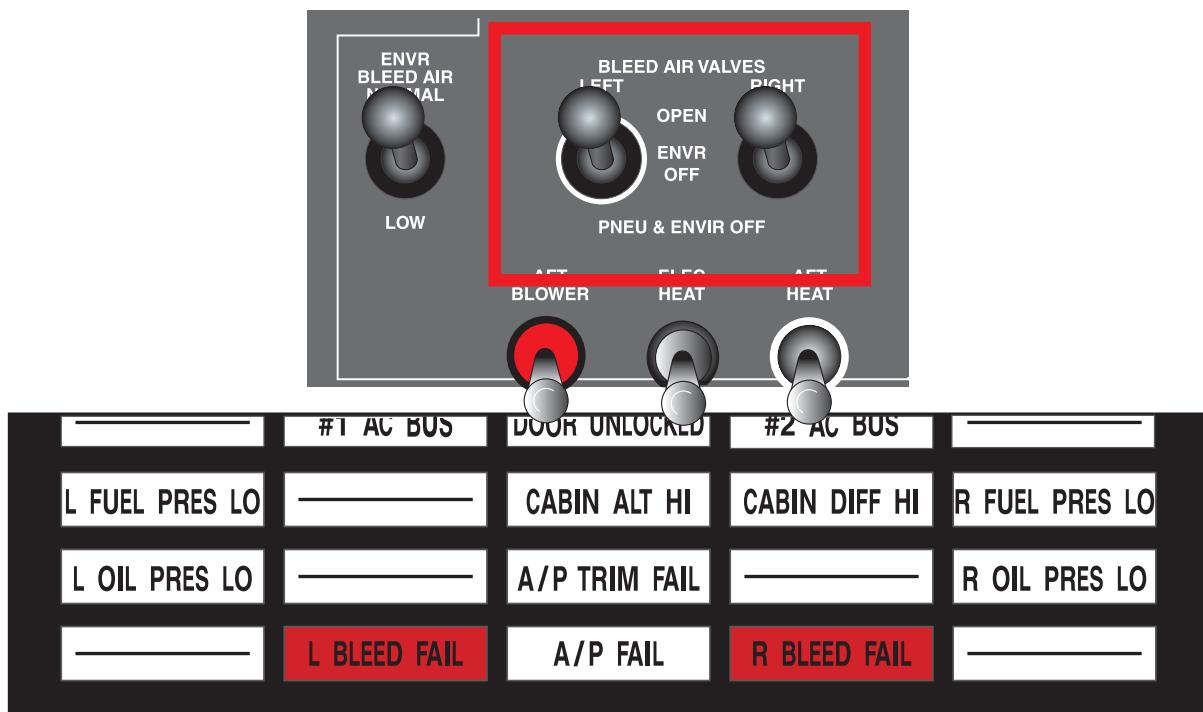


Figure 9-4 Bleed Air Valve Controls

CABIN DOOR SEAL

The entrance door to the cabin uses air from the pneumatic system to inflate the door seal after the airplane lifts off the ground. Bleed air is tapped off the manifold downstream of the 18 psi pressure regulator. The regulated air then passes through a 4 psi regulator and to the normally-open valve that is controlled by the left landing gear safety switch. When the airplane lifts off, the landing gear switch closes, which opens the valve to the door seal, and the door seal inflates.

HYDRAULIC FILL CAN PRESSURE

Pneumatic air pressure is injected into the landing gear hydraulic fill can. It provides positive pressure to the reservoir to minimize vaporization of hydraulic fluid and to provide positive feed to the hydraulic pump. When the engines are

shut down, the pneumatic pressure in the hydraulic fill is bled off through a small orifice.

CABIN AND COCKPIT SIDE WINDOW DEFOG SYSTEM

Cabin window and cockpit side window defogging is provided through the use of engine bleed air anytime one or both engines are running. A pressure regulator mounted under the floorboard regulates the bleed air pressure for window defog. From the regulator the air is routed to the right of center and forward where the air lines divide. At this point part of the air is routed to the sides of the airplane and up the sidewall to just below the windows where it tees and runs fore and aft under the cabin windows. Beneath each cabin window there is another tee in the system which delivers the air to the inside surface of the exterior window through two tubes.

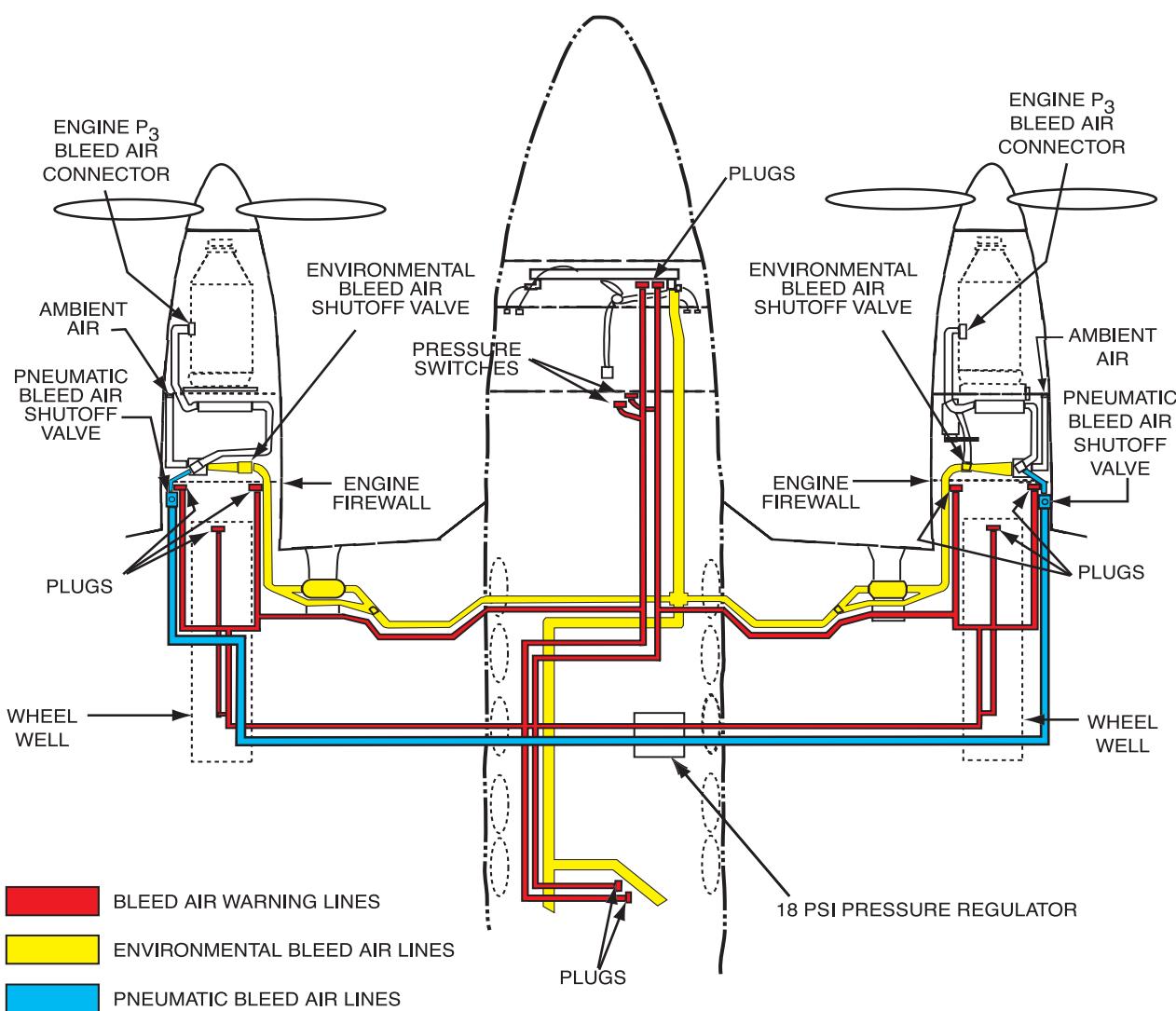


Figure 9-5 Bleed Air Warning System Diagram

Some 300s have cabin window defogging and some do not. Cabin window defogging is an

option on all 350s, and all 300s and 350s have cockpit side window defogging.

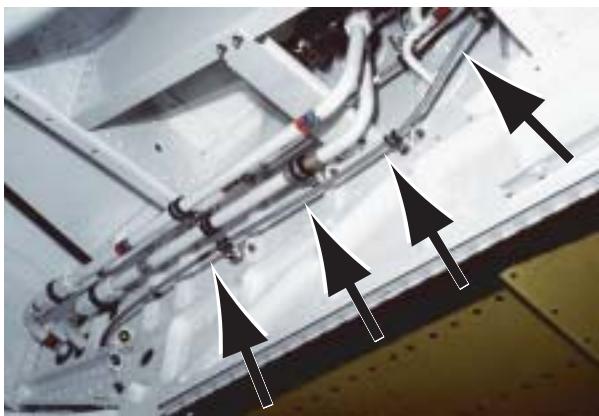


Figure 9-6 Bleed Air Warning Plastic Tubing

CAUTION

Caution must be used when removing or installing the emergency exits to avoid damaging the bleed air lines. Make certain the bleed air lines are properly connected when installing the hatches.

Air for the cockpit side windows is routed forward from the pressure regulator to the cockpit below the floorboard. At this point the line tees, running to the sides of the airplane and up the sidewall to the windows. The warm dry air is now directed to the inside surface of the windows through a manifold in the window frame.



CHAPTER 10

ICE AND RAIN PROTECTION

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CHAPTER 10

ICE AND RAIN PROTECTION



INTRODUCTION

Flight in known icing conditions requires knowledge of conditions conducive to icing, and of all anti-ice and deice systems available to prevent excessive ice from forming on the airplane. This section identifies these systems, their controls and best usage.

DESCRIPTION

The Ice and Rain Protection section of the training manual presents a description and discussion of the airplane ice and rain protection systems. The purpose of this section is to acquaint the pilot with all the systems available for flight in icing or heavy rain conditions. All of the anti-ice, deice and rain protection systems in this airplane are described showing location, controls, and how they are used. Procedures in case of malfunction in any system, and information concerning preflight deicing and defrosting are included.

ICE PROTECTION SYSTEMS

The Beechcraft Super King Air 300 and 350 is FAA approved for flight in known icing conditions when the required equipment is installed and operational (Figures 10-1 and 10-3). The Kinds of Operations Equipment List, contained in the Limitations section of the *Pilot's Operating Handbook*, lists the necessary equipment (Figures 10-2 and 10-4). The ice and rain protection controls are grouped on the pilot's subpanels, except the windshield wiper control which is on the overhead panel (Figure 10-5).



Figure 10-1 King Air 300 Anti-Icing and Deicing Components

SYSTEM and/or COMPONENT	VFR DAY				
	VFR NIGHT				
IFR DAY					
IFR NIGHT					
ICING CONDITIONS					
ICE AND RAIN PROTECTION					
1. Alternate Static Air System	1	1	1	1	1
2. Engine Auto-Ignition System Including Annunciator	2	2	2	2	2
3. Engine Auto-Ice System Including Annunciator	2	2	2	2	2
4. Heated Fuel Vent	0	0	2	2	2
5. Heated Windshield	0	0	0	0	2
6. Pitot Heat	0	0	2	2	2
7. Pneumatic Pressure Indicator	0	0	1	1	1
8. Stall Warning Heat (Lift Transducer and Mounting Plate)	0	0	0	0	1
9. Surface Deicer System	0	0	0	0	1
10. Propeller Deicer System	0	0	0	0	1
11. Wing Ice Light (Left)	0	0	0	0	1

Figure 10-2 King Air 300 Ice and Rain Protection Required Equipment

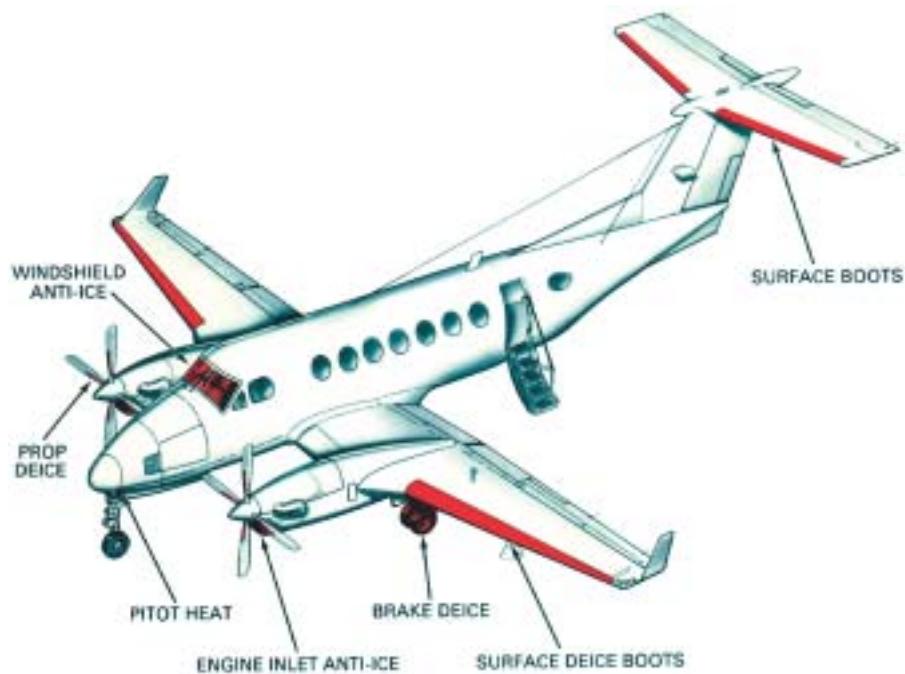


Figure 10-3 King Air 350 Anti-Icing and Deicing Components

SYSTEM and/or COMPONENT	VFR DAY				
	VFR NIGHT				
IFR DAY					
IFR NIGHT					
ICING CONDITIONS					
ICE AND RAIN PROTECTION	1	1	1	1	1
1. Alternate Static Air System	2	2	2	2	2
2. Engine Auto-Ignition System Including Annunciator	2	2	2	2	2
3. Engine Auto-Ice System Including Annunciator	0	0	2	2	2
4. Heated Fuel Vent	0	0	0	0	2
5. Heated Windshield	0	0	2	2	2
6. Pitot Heat	0	0	1	1	1
7. Pneumatic Pressure Indicator	0	0	0	0	1
8. Stall Warning Heat (Lift Transducer and Mounting Plate)	0	0	0	0	2
9. Surface Deicer System	0	0	0	0	1
10. Propeller Deicer System	0	0	0	0	1
11. Wing Ice Light	0	0	0	0	1

Figure 10-4 King Air 350 Ice and Rain Protection Required Equipment

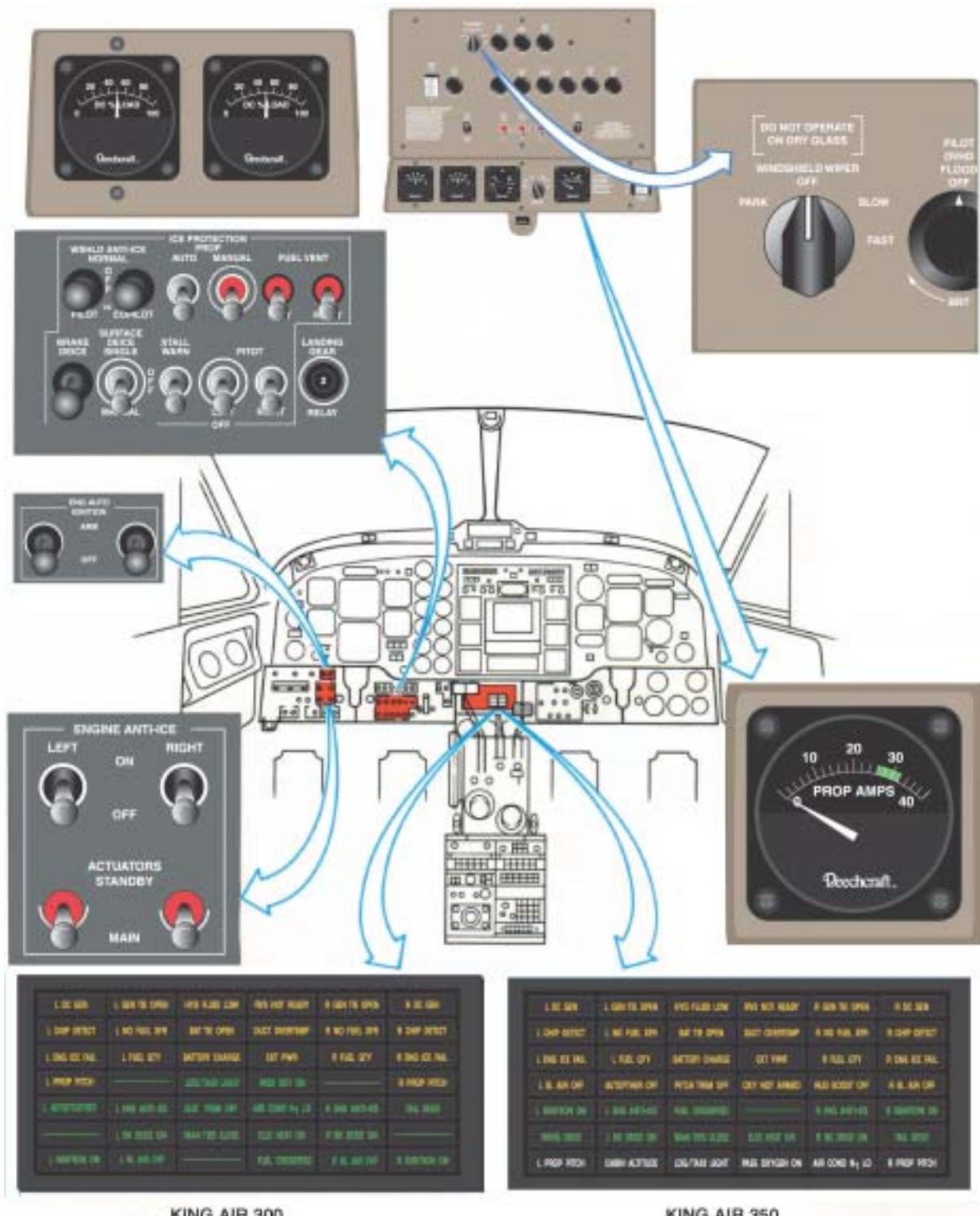


Figure 10-5 Ice and Rain Protection Controls



GENERAL DESCRIPTION AND OPERATION

The airplane is equipped with a variety of ice and rain protection systems that can be utilized during operation under inclement weather conditions. There are twelve anti-ice/deice/rain protection systems:

- Engine Inlet Lip Heat
- Inertial Separators (Ice Vanes)
- Auto-Ignition
- Windshield Anti-Ice
- Windshield Wipers
- Side Window Defog
- Propeller Deice
- Fuel System Anti-Ice
- Pitot Heat
- Stall Warning Heat
- Surface Deice (Leading-Edge Boots)
- Brake Deice (Optional)

The leading-edge lip of the engine air inlet is continuously anti-iced by engine exhaust air.

Engine ice protection is provided by an inertial separation system utilizing electric motors and actuators. Should the main motor fail, a standby motor is available.

An auto-ignition system provides positive engine ignition during periods of low engine power/engine flame-out.

Electric heating elements embedded in the windshield provide protection against ice formation, while air from the cabin heating system minimizes windshield fogging.

Heavy-duty windshield wipers for both the pilot and copilot provide increased visibility during rainy flight conditions.

Cabin window and cockpit side window defogging is provided through the use of engine bleed air anytime one or both engines are running.

The propellers are protected against icing by electro-thermal boots on each blade that automatically cycle to prevent ice formation.

Fuel is heated by an oil-to-fuel heat exchanger immediately prior to its entrance to the fuel control unit. One fuel vent in each wing fuel system is heated by electric elements which are powered through circuit breakers on the copilot's circuit breaker panel. They are controlled by switches on the pilot's right subpanel.

Heating elements in the pitot masts prevent the pitot opening from becoming clogged with ice. These heating elements are connected to the airplane electrical system through circuit breaker switches on the pilot's right subpanel.

Stall warning heat consists of electrical elements in the stall vane and plate on the left wing leading edge. It is controlled by a circuit breaker switch on the pilot's right subpanel.

Pneumatic deicer boots on the wings and horizontal stabilizers prevent ice formation during flight. Regulated bleed air pressure and vacuum are cycled to the pneumatic boots for the inflation/deflation cycle. The selector switch that controls the system permits automatic single-cycle operation or manual operation.

Optional brake deicers may be installed which use pneumatic air to prevent ice and slush buildup on the wheel brakes and between the main landing gear tires.

ENGINE INLET LIP HEAT

The lips around both engine cowling air inlets are heated by hot exhaust gases to prevent ice formation (Figure 10-6). A scoop in the engine's left exhaust stack deflects a small portion of the hot exhaust gases downward into the hollow lip tube that encircles the air

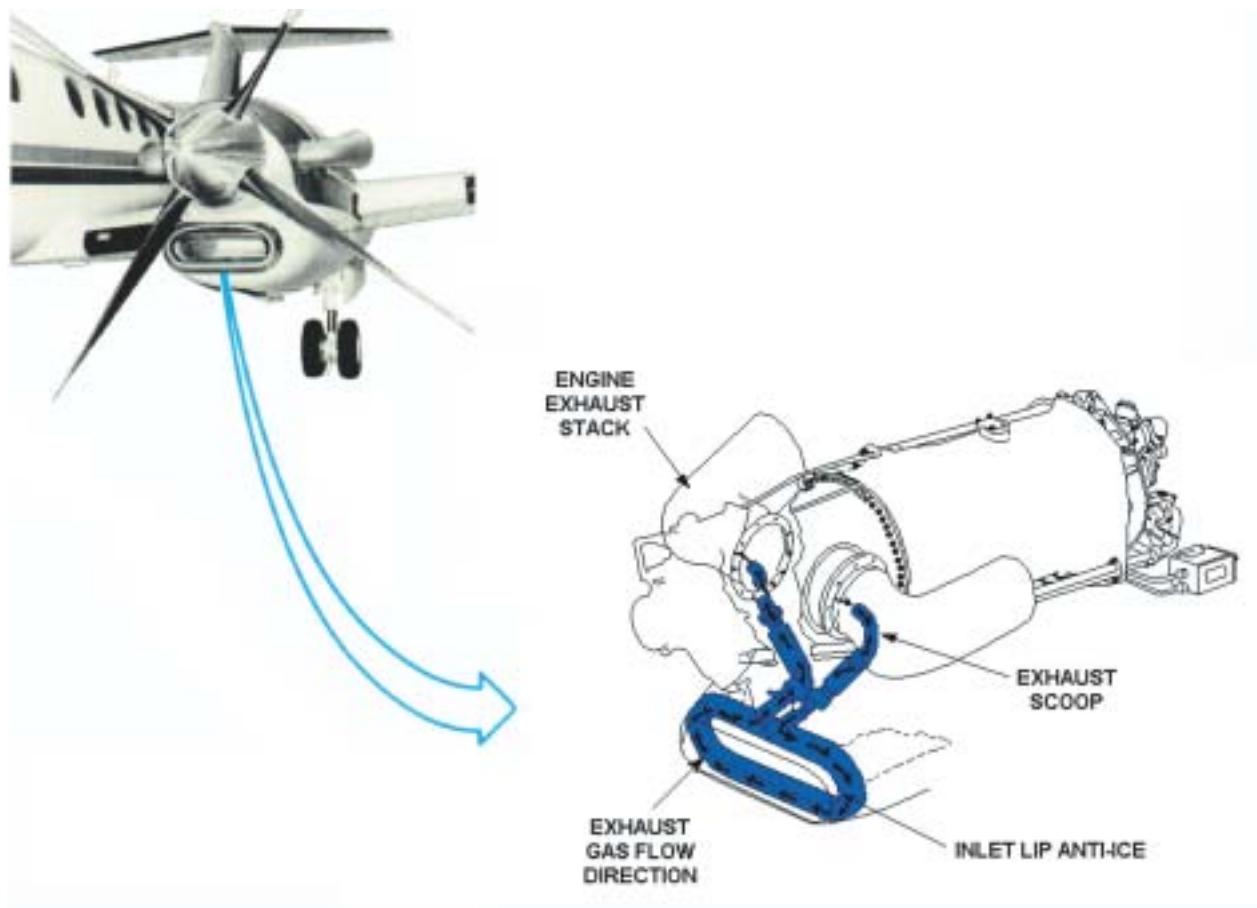


Figure 10-6 Engine Inlet Lip Heat

inlet. The gases are expelled through a line into the right exhaust stack where they move out with the engine exhaust gases. Engine exhaust air will flow through the inlet heat duct whenever the engine is running.

ENGINE INERTIAL SEPARATORS

A vane system of inertial separators is installed on each engine to prevent ice or other foreign objects, such as dust or sand, from entering the engine compressor intake, and to prevent ice accumulation on the engine inlet screen. Two movable vanes, forward and aft, are positioned as shown in Figure 10-7 for normal flying conditions.

Under icing conditions with the ENGINE ANTI-ICE switch ON, the forward ice vane is positioned to create a venturi effect and introduces a sudden turn where the airflow enters the engine intake (Figure 10-8). The aft vane in the lower aft end of the air duct is open.

As contaminated air enters the cowling inlet, it is accelerated by the venturi effect. Due to their greater mass, and therefore greater momentum, the frozen moisture particles, dust and/or sand particles accelerate past the screen area and are discharged overboard through the aft vane opening. Simultaneously, most of the air molecules make the sudden turn which frees them of the ice particles and other contaminants. The “filtered” air then enters the engine through the inlet screen.

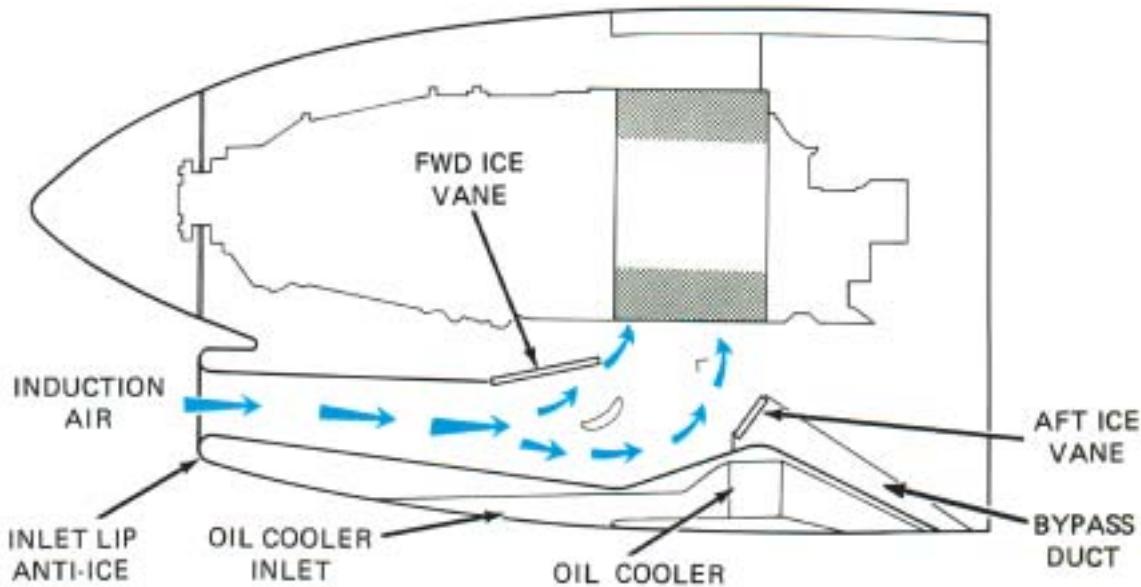


Figure 10-7 Inertial Separator in Retract Position

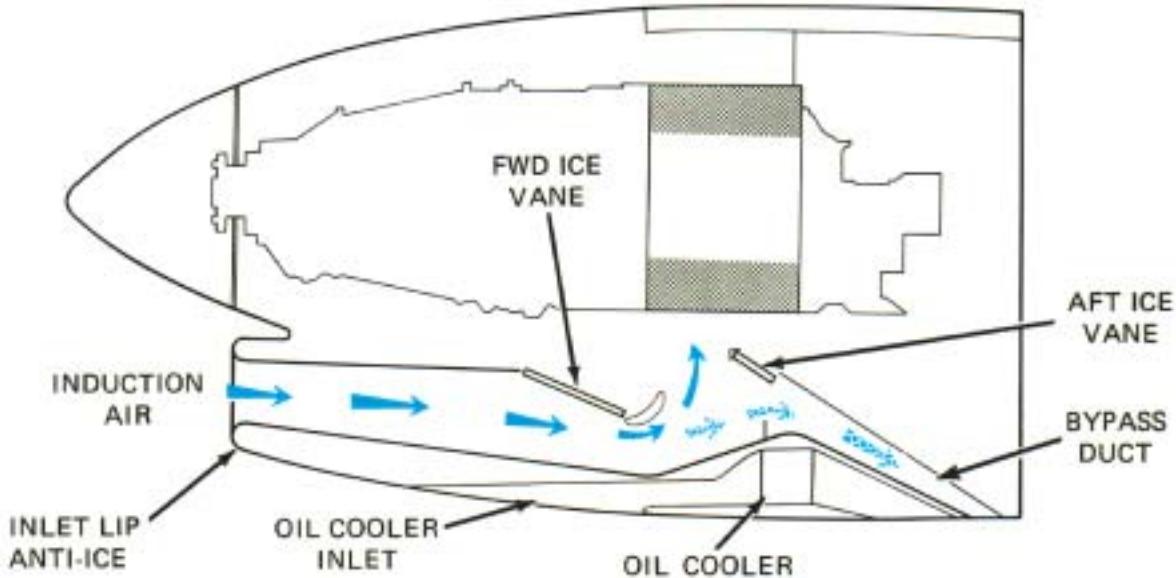


Figure 10-8 Inertial Separator in Extend Position



ICE VANE CONTROLS

The inertial separator doors are extended or retracted through mechanical linkage connected to electric motors and actuators. The motors are energized through switches in the ICE PROTECTION group located on the pilot's left subpanel (Figure 10-9). The switches are placarded: ENGINE ANTI-ICE - LEFT - RIGHT - ON - OFF and ACTUATORS - STANDBY - MAIN. The ENGINE ANTI-ICE switches extend the doors in the ON position, retract them in the OFF position and can be used in all normal flight operations.

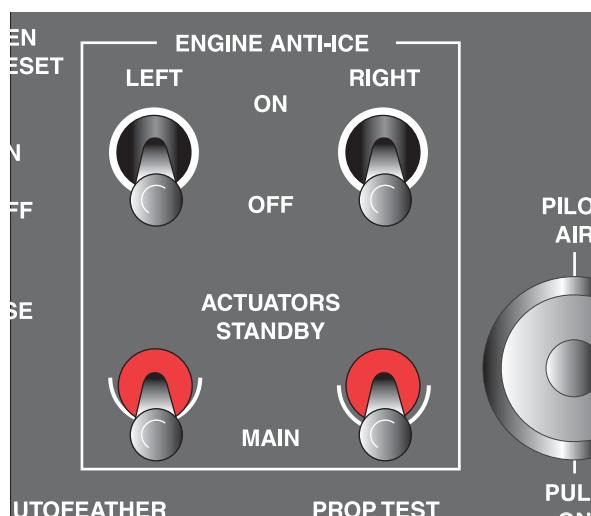


Figure 10-9 Ice Vane Controls

The ACTUATORS switches allow selection of either the MAIN or STANDBY actuator motors. These dual motors provide a redundant system. Although the motors are called main and standby, the only difference is their power source; therefore they can be used interchangeably. The main motors are powered by the triple-fed bus, while the standby motors are powered by their corresponding generator bus.

The inertial separators (ice vanes) shall be extended whenever there is visible moisture at ambient temperatures of +5°C or below. When the ice vanes are extended, two green advisory annunciators will be illuminated.

Since the airflow into the engine will be restricted, there may be a decrease in torque and a slight increase in ITT. A decrease in overall cruise performance can be expected with the ice vanes extended. When the ice vane doors are retracted, the annunciators will be extinguished and the ITT and torque will be restored.

The inertial separators (ice vanes) shall be retracted for takeoff operations in ambient temperatures of above +10°C on the King Air 350 only.

The vanes have only two positions; there are no intermediate positions. The system is monitored by L and R ENG ANTI-ICE (green) and L and R ENG ICE FAIL (yellow) annunciators (Figure

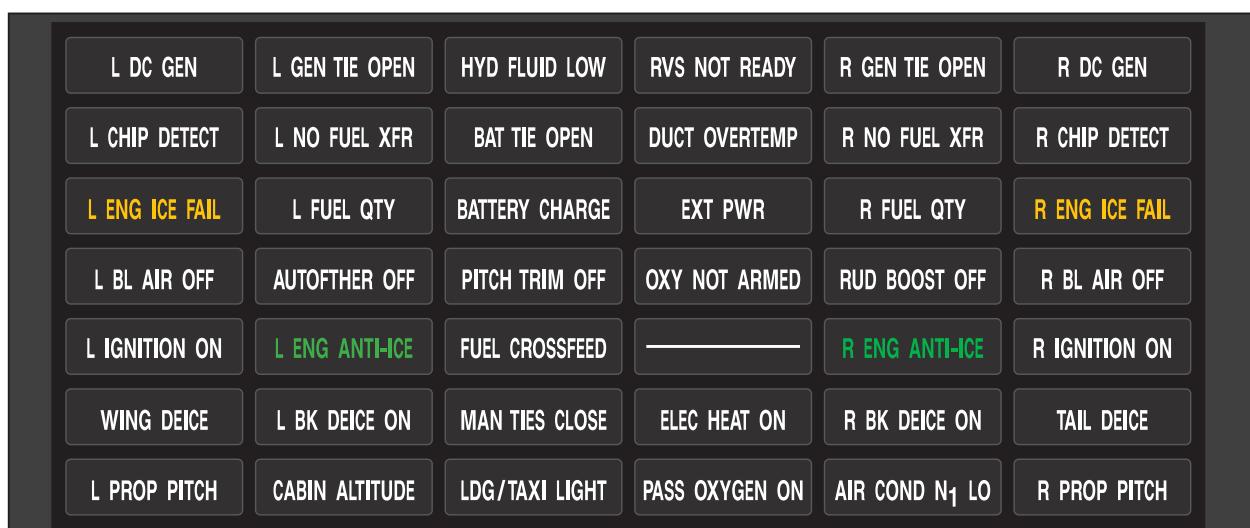


Figure 10-10 Caution Annunciators



10-10). Illumination of the L or R ENG ANTI-ICE advisory annunciator indicates that the corresponding system is extended.

Illumination of the L or R ENG ICE FAIL annunciator indicates that the corresponding system did not move to the desired position. The yellow L or R ENG ICE FAIL annunciator circuit compares the ENGINE ANTI-ICE switch position to the separator door microswitches, checking for both extension and retraction. After a 35-second delay, the annunciator will be illuminated if the switch position and microswitches do not agree. Immediate illumination of the L or R ENG ICE FAIL annunciator indicates loss of electrical power to the motor. In either event, the other actuator (MAIN or STANDBY) should be selected.

ENGINE AUTO-IGNITION SYSTEM

The engine auto-ignition system provides electrical power to the igniters in the event of an engine power loss. The system must be armed during periods of turbulence and penetration of icing or precipitation conditions. When armed, auto-ignition will automatically attempt to reignite the engine should engine flameout occur.

If engine torque falls below approximately 17% for any reason, electrical power is provided to energize the engine igniters. When this happens, the green IGNITION ON annunciator (Figure 10-11) on the caution/advisory panel will be illuminated, indicating that the ignition system has

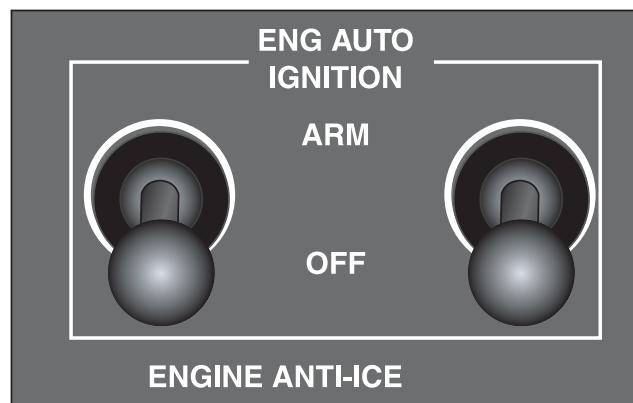


Figure 10-11 Engine Auto-Ignition System



been energized. The system should be turned off during ground operation to prolong igniter life.

The switches used to arm the auto-ignition system are located on the pilot's left subpanel, above the ice vane switches and just to the left of the pilot's control column.

The system is activated by moving the switches to the up or ARM position. Each switch must be lifted over a detent before it can be moved into or out of the ARM position. This lever-lock feature prevents inadvertent movement to the OFF position.

Refer to the Engine section of this training manual for additional information about auto-ignition.

WINDSHIELD ANTI-ICE

The pilot's and copilot's windshields each have independent controls and heating circuits (Figure 10-12). The control switch allows the pilot to select a HIGH intensity heat level or a NORMAL intensity heat level.

The windshields are composed of three physical layers. The inner layer is a thick panel of glass that acts as the structural member. The middle layer is a polyvinyl sheet which carries fine wire heating grids. The outer layer is a protective layer of glass bonded to the first two layers. The outside of the windshield is treated with a static discharge film called a NESA coating.

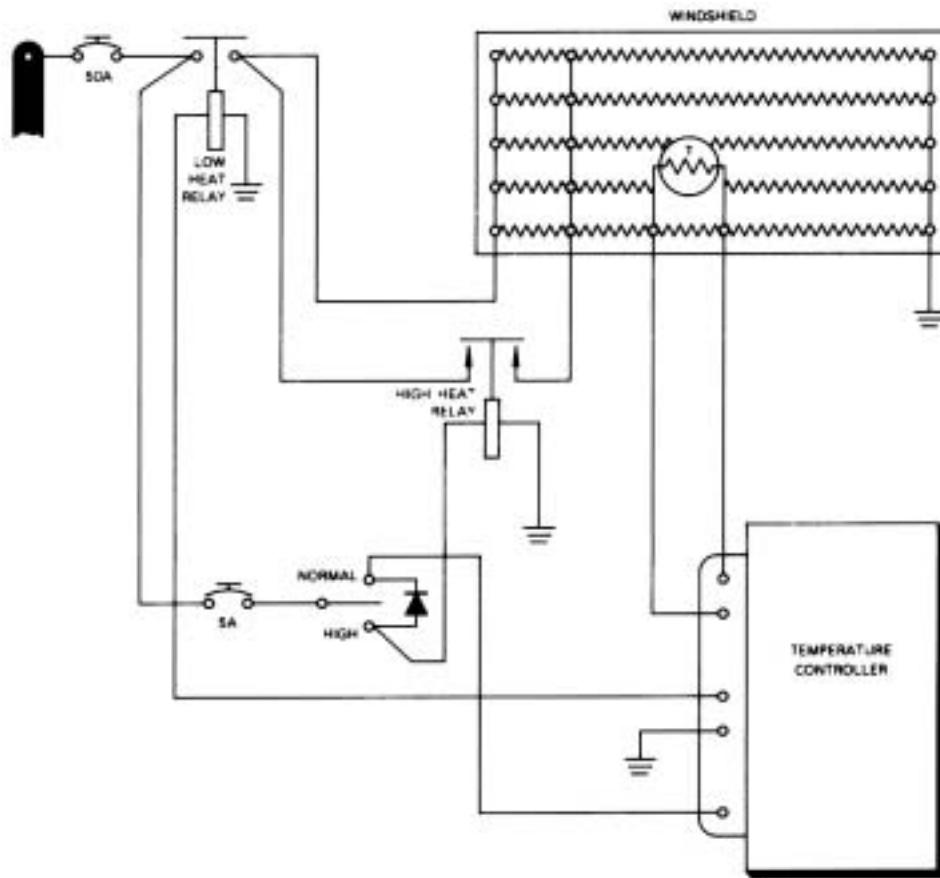


Figure 10-12 Windshield Anti-Ice Diagram



The windshields are protected against icing by electrical heating elements which consist of transparent material (stannic oxide) which has high electrical resistance incorporated in the laminations of windshields. The resistive material is arranged to provide primary heated surfaces and secondary heated surfaces.

Each is also fitted with electrical connections for temperature sensing elements. The heating elements are connected at terminal blocks in the corners of the glass to the wiring leading to the control switches mounted on the pilot's right subpanel.

Switches in the ICE PROTECTION group on the pilot's inboard subpanel (Figure 10-13) placarded:

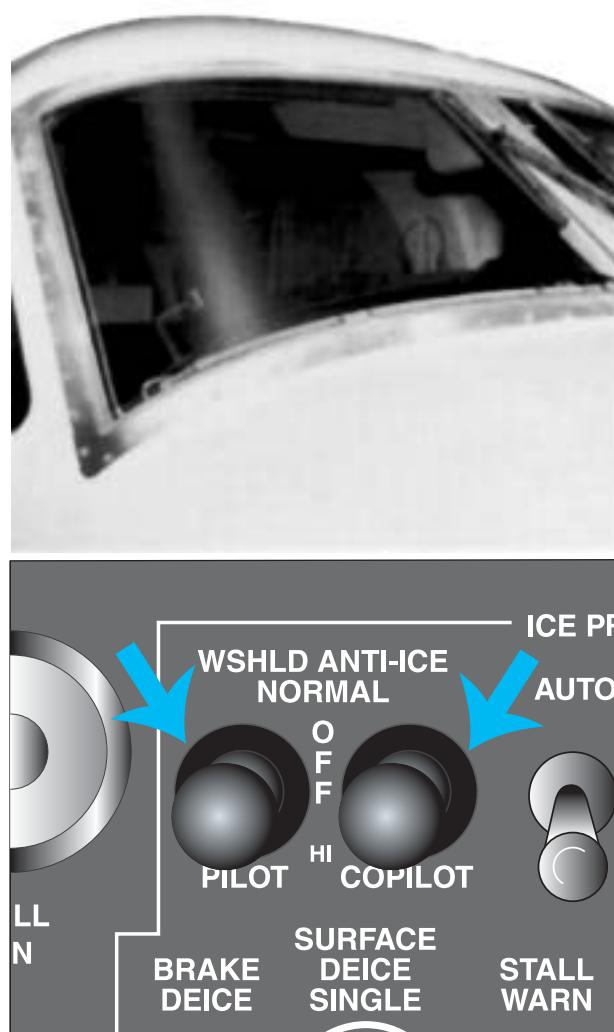


Figure 10-13 Windshield Anti-Ice Switches

WSHLD ANTI-ICE - NORMAL - OFF - HIGH - PILOT - COPILOT are used to control windshield heat. When the PILOT'S and COPILOT'S switches are in the NORMAL (up) position the secondary areas of the windshields are heated. When the switches are in the HIGH (down) position the primary areas are heated.

The primary areas are smaller areas and are heated faster to the same temperature as NORMAL position. Each switch must be lifted over a detent before it can be moved into the HIGH position. This lever-lock feature prevents inadvertent selection of the HIGH position when moving the switches from NORMAL to the OFF (center) position.

Windshield temperature is controlled automatically by a temperature-sensing element embedded in each windshield, and a temperature controller in each windshield circuit. The temperature controllers attempt to maintain the temperature of the windshield between 90 and 110°F.

When the NORMAL level of heating is selected, an automatic temperature controller senses the windshield and attempts to maintain it at approximately 90 to 110°F. It does so by energizing the power relay as necessary. In this mode, almost the entire windshield is heated (Figure 10-14).

When the HIGH level of heating is selected, the same temperature controller senses the windshield temperature and attempts to maintain it at 90 to 110°F. In this mode, however, the controller will also energize the high heat relay switch which applies the electrical heat to a more concentrated and more essential viewing area of the windshield. In HIGH only the outboard two-thirds of the windshield is heated (Figure 10-15).

The power circuit of each system is protected by 50-ampere circuit breakers located in the power distribution panel under the center aisle floor. Windshield heater control circuits are protected with 5-ampere circuit breakers, located on the copilot's circuit breaker panel.

Windshield heat may be used at anytime and in any combination, although it is best to turn it on prior to entering icing conditions. This



helps ensure that the windshield is sufficiently warm to prevent ice formation.

In addition, windshield heat may be used to help in windshield defrosting and defogging. Use of windshield heat may cause erratic operation of the magnetic compass because of the electrical field created by the heating elements.

CAUTION

In the event of windshield icing during sustained icing conditions, it may be necessary to reduce the airspeed in order to keep the windshield ice-free. 226 KIAS is the maximum speed for effective windshield anti-icing.

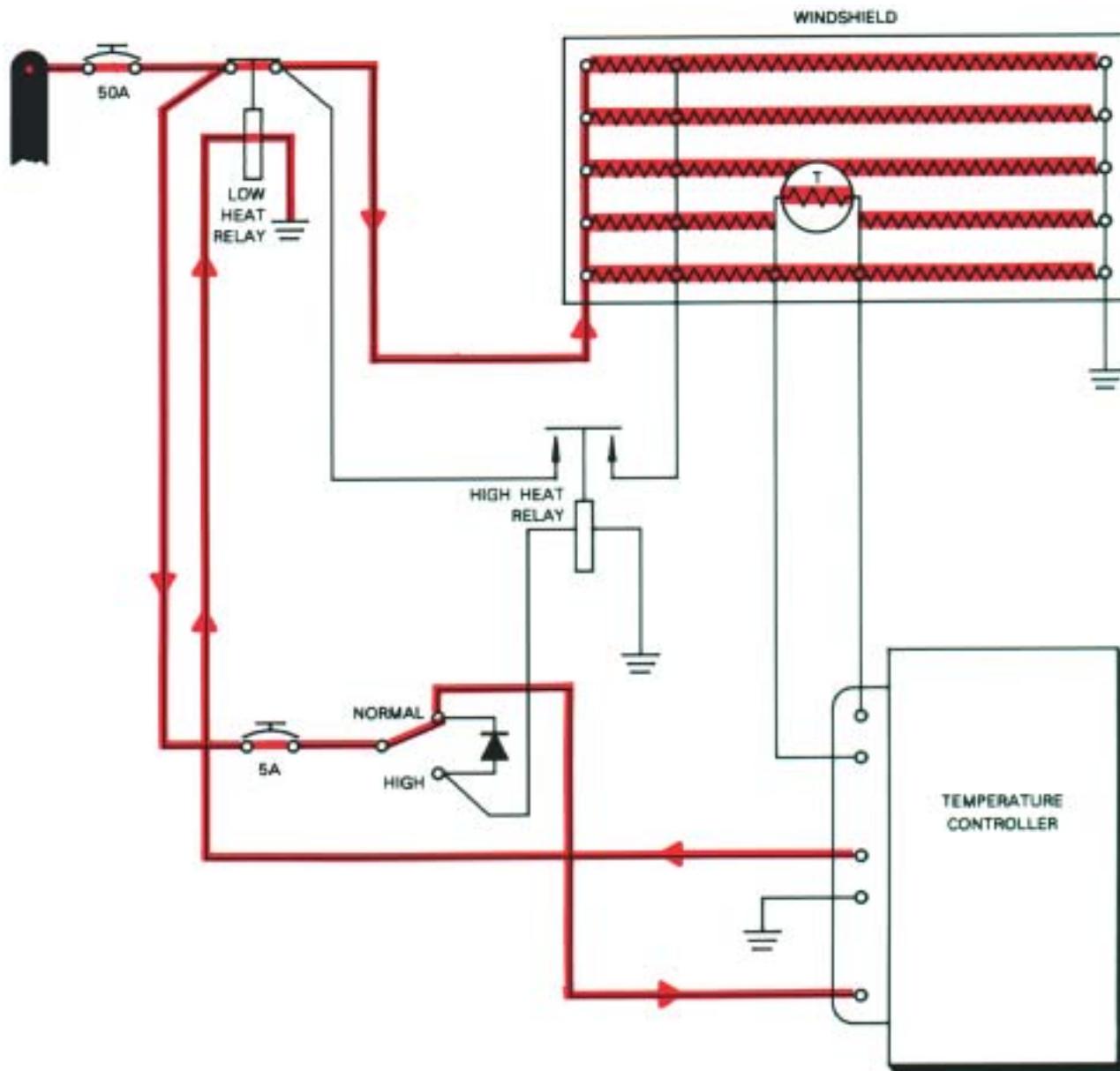


Figure 10-14 Windshield Anti-Ice Diagram—Normal Heat

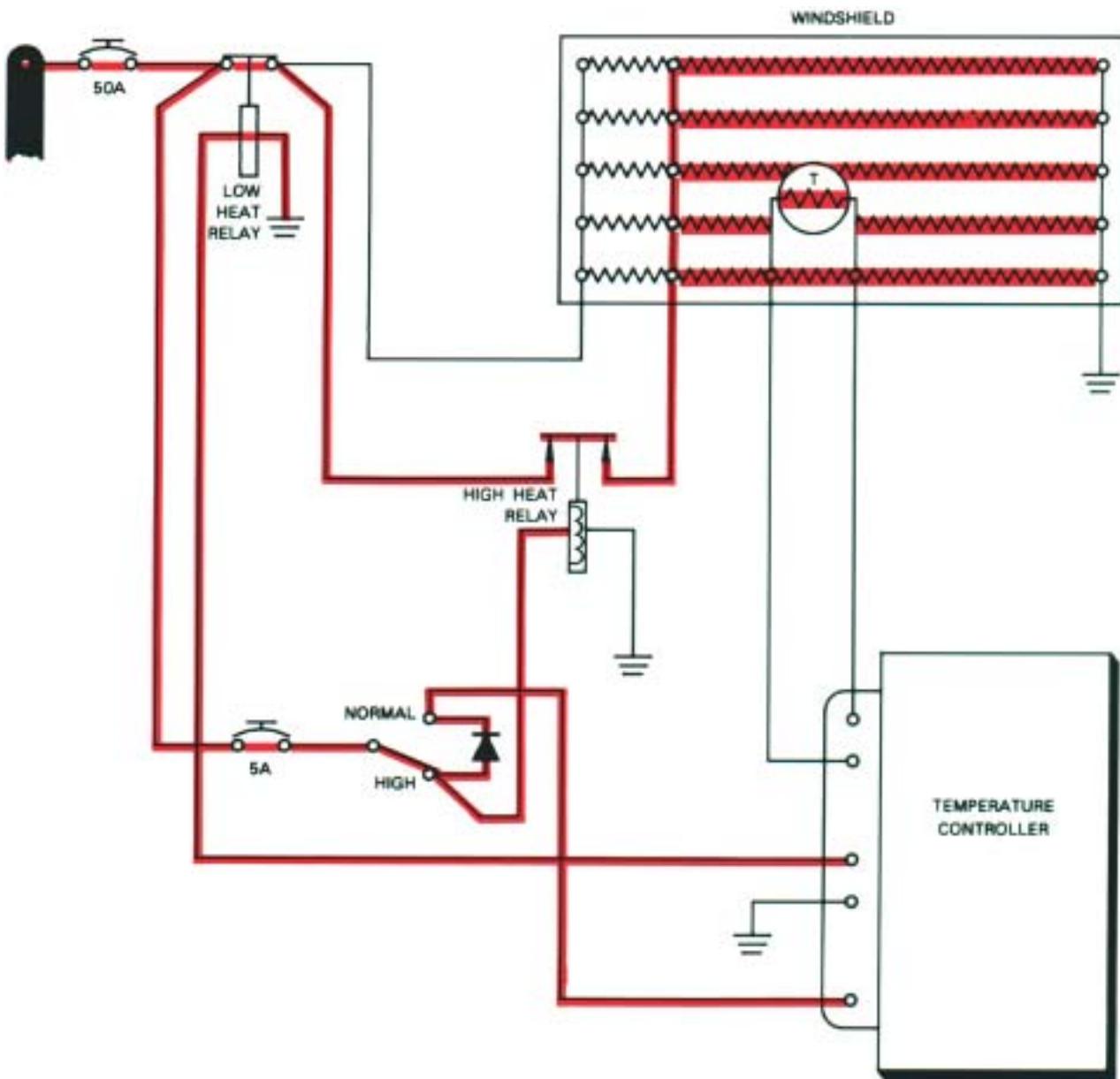


Figure 10-15 Windshield Anti-Ice Diagram—High Heat

WINDSHIELD WIPERS

Separate windshield wipers are mounted on the pilot's and copilot's windshields. The dual wipers are driven by a mechanism operated by a single electric motor, all located forward of the instrument panel.

The windshield wiper control is located on the overhead light control panel (Figure 10-16). It

provides the wiper mechanism with OFF, SLOW, FAST and PARK positions. The windshield wiper circuit breaker is on the copilot's circuit breaker panel in the WEATHER group. The wipers may be used as required, either on the ground or in flight, but must not be operated on a dry windshield, since they may scratch a protective electrostatic (NESA) coating on the outer layer of glass.



Figure 10-16 Windshield Wipers

SIDE WINDOW DEFOG

Cabin window and cockpit side window defogging is provided through the use of engine bleed air any time one or both engines are running.

A pressure regulator mounted under the floorboard regulates the bleed air pressure for window defog. From the regulator the air is routed to the right of center and forward where the air lines divide. At this point part of the air is routed to the sides of the airplane and up the sidewall to just below the windows where it tees and runs fore and aft under the cabin windows.

Beneath each cabin window there is another tee in the system which delivers the air to the inside surface of the exterior window through two tubes.

Some 300s have cabin window defogging and some do not. All 350s have cabin window defogging, and all 300s and 350s have cockpit side window defogging.

CAUTION

Caution must be used when removing or installing the emergency exits to avoid damaging the bleed air lines. Make certain the bleed air lines are properly connected when installing the hatches.

Air for the cockpit side windows is routed forward from the pressure regulator to the cockpit below the floorboard. At this point the line tees, running to the sides of the airplane and up the sidewall to the windows. The warm dry air is now directed to the inside surface of the windows through a manifold in the window frame.

PROPELLER DEICE SYSTEM

The propeller deice system includes: an electrically heated boot for each propeller blade, slip rings, brush assemblies, a timer, two on-off switches (one for automatic operation, one for manual operation), and an ammeter (Figures 10-17 and 10-18).

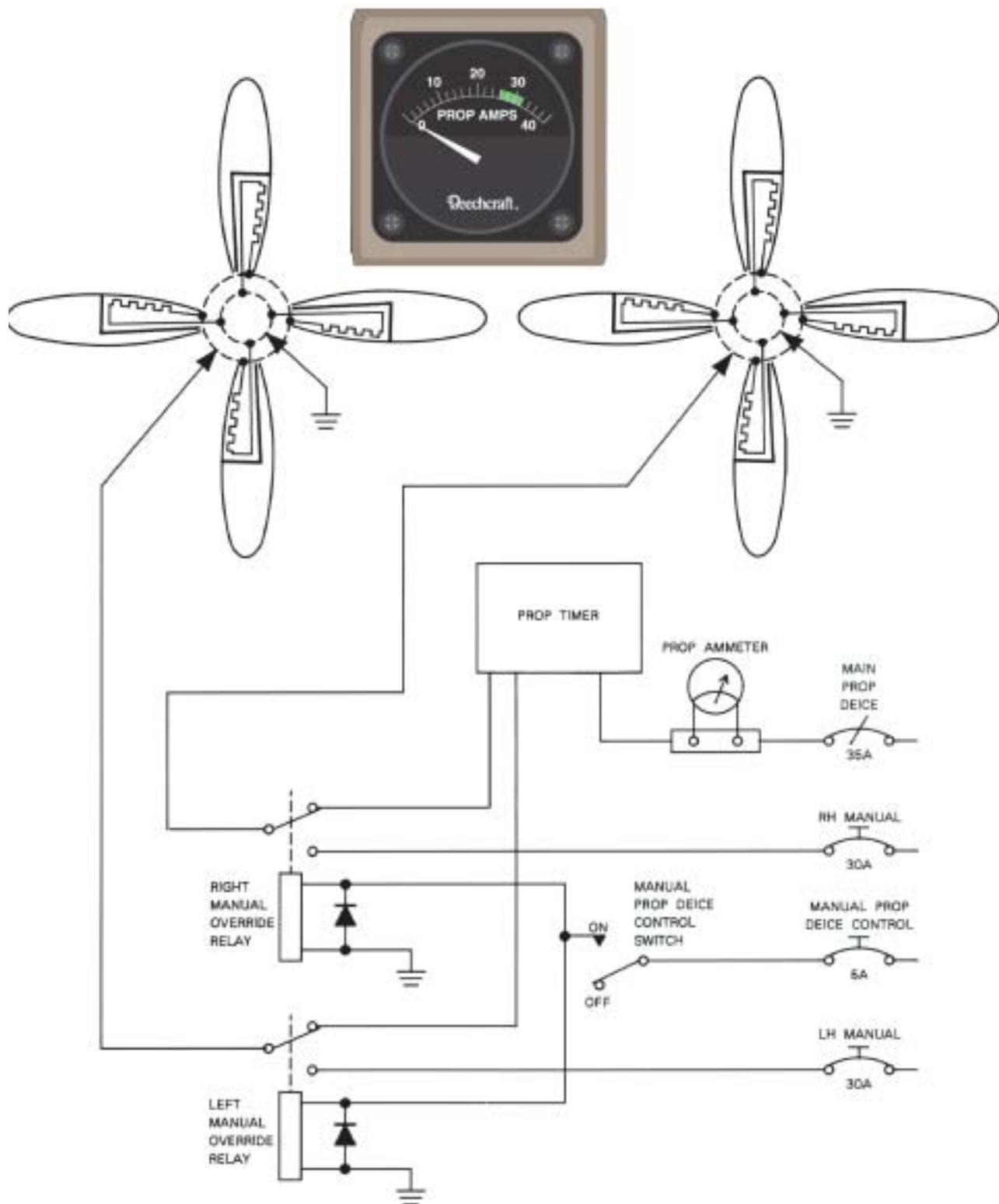
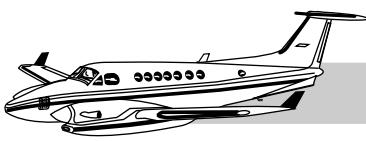


Figure 10-17 Propeller Deice System

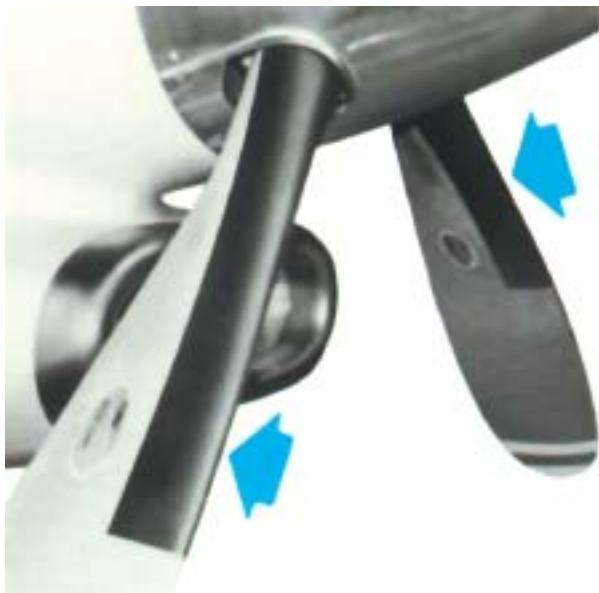


Figure 10-18 Propeller Deice Boots

Heat produced by the heating elements in the deice boots reduces the adhesion of the ice. The ice is then removed by the centrifugal force of the spinning propeller and the blast of the airstream.

CAUTION

Propeller deice must not be operated when the propellers are static or the slip rings around the propeller shaft will pit and burn, eventually becoming useless.

When the PROP-AUTO switch on the pilot's right subpanel is turned on, the ammeter registers the amount of current (normally 26 to 32 amperes) passing through the system. If the current rises beyond normal limits, the circuit breaker switch will shut off power to the deicer timer. Current flows from the timer through the brush assemblies to the slip rings, where it is distributed to the individual propeller deice boots.

Power to the deice boots is cycled in two 90-second phases. The first 90-second phase simultaneously heats all of the boots on one pro-

peller. The second phase heats the boots on the other propeller. The deice timer completes one full cycle every three minutes. As the deice timer moves from one phase to the next, a slight, momentary deflection of the propeller ammeter needle may occasionally be noted. Once the system is turned on for automatic operation, it cycles continuously.

NOTE

The heating sequences for the propeller deice boots are the sequences which are in evidence during normal operation; however, when the power is turned off, the timer moves ahead to the start of the cycle on the next propeller, so it may restart on either propeller.

A manual propeller deice system is provided as a backup to the automatic system. A control switch located on the pilot's right subpanel controls the manual override relays. When the switch is in the MAN position, power is supplied to all boots on both propellers. The manual override switch is momentary, and must be held in place until the ice has been dislodged from the propeller surface. The propeller ammeter will not indicate any load in the manual mode of operation, however, the loadmeters will indicate approximately a 10% increase in their loads.

FUEL SYSTEM ANTI-ICE

There are two anti-ice systems to protect fuel flow through the fuel lines to the engine (Figure 10-19). Without heat, moisture in the fuel could freeze, and diminish or cut off the fuel flow to the engines in freezing temperatures.

Ice formation in the fuel vent system is prevented by electrically heated vents in each wing. The fuel vent heat is operated by L and R FUEL VENT switches located in the ICE PROTECTION group on the pilot's right subpanel. They are powered by the left and right generator buses and should be turned on whenever ice is anticipated or encountered.

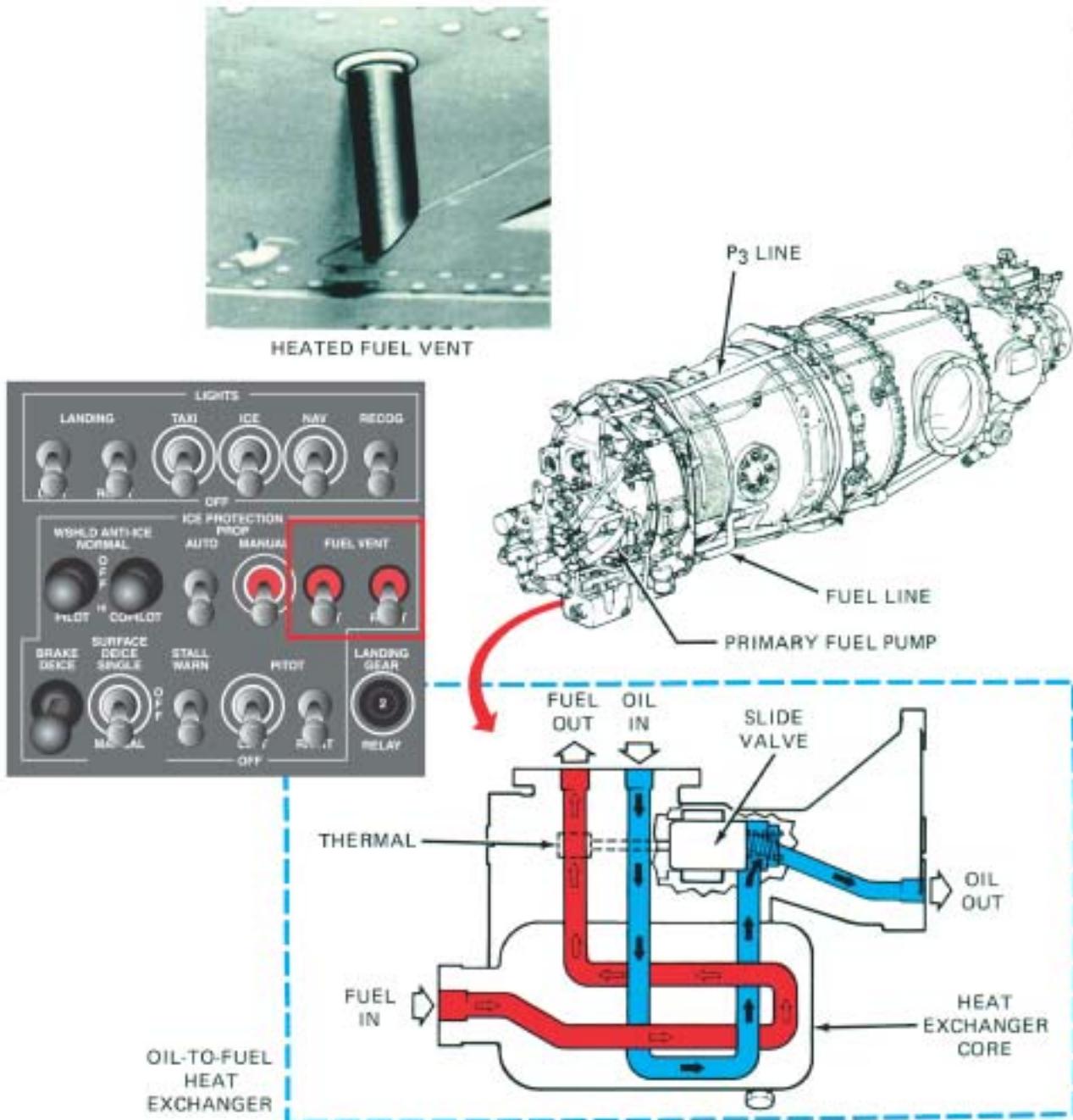


Figure 10-19 Fuel System Anti-Ice



The fuel control unit is protected against ice by an oil-to-fuel heat exchanger, mounted on the engine's accessory section. An engine oil line is located next to the fuel line within the heat exchanger. Heat transfer occurs through conduction, melting ice particles which may have formed in the fuel. This operation is automatic whenever the engines are running. Refer to the Fuel section of this training manual for more details about fuel heat, and the Limitations section of the *POH* for temperature limitations concerning the oil-to-fuel heat exchanger.

PITOT HEAT

Two pitot masts, located on the nose of the aircraft, contain electric heating elements to ensure proper airspeed indication during icing conditions (Figure 10-20). Pitot heat is controlled by two circuit breaker switches located on the pilot's right subpanel. The two-position switches placarded PITOT, LEFT - RIGHT are off when down and on when up. The left (pilot's) pitot heat is powered by the triple-fed bus; the right (copilot's) pitot heat is powered by the right generator bus.

The pitot heat system should not be operated on the ground, except for testing or for short intervals to remove snow or ice from the mast. Pitot heat should be turned on for takeoff and can be left on in flight during icing conditions, or whenever icing conditions are expected. If during flight at altitude there is a gradual reduction in airspeed indication, there may be pitot icing. If turning pitot heat on restores airspeed, leave it on because icing conditions exist. With many pilots, it is standard practice to use pitot heat during all flights at high altitude and/or cold temperature to prevent pitot icing.

STALL WARNING VANE

The stall warning vane and plate are provided with electric heat elements to insure against freeze-up during icing conditions (Figure 10-21). The stall warning plate is activated by a two-position switch, located on the pilot's right subpanel. The down position is off, and the up position is on. The stall warning heat is powered by the right generator bus.

Even though the heating elements protect the lift transducer vane and face plate from ice, a

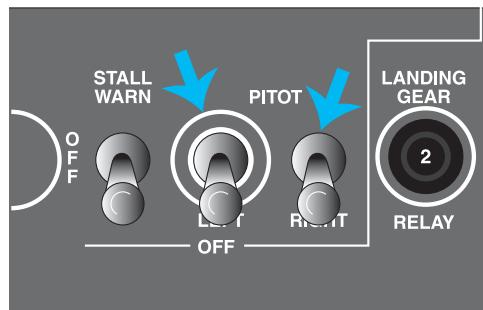


Figure 10-20 Pitot Mast and Heat Controls

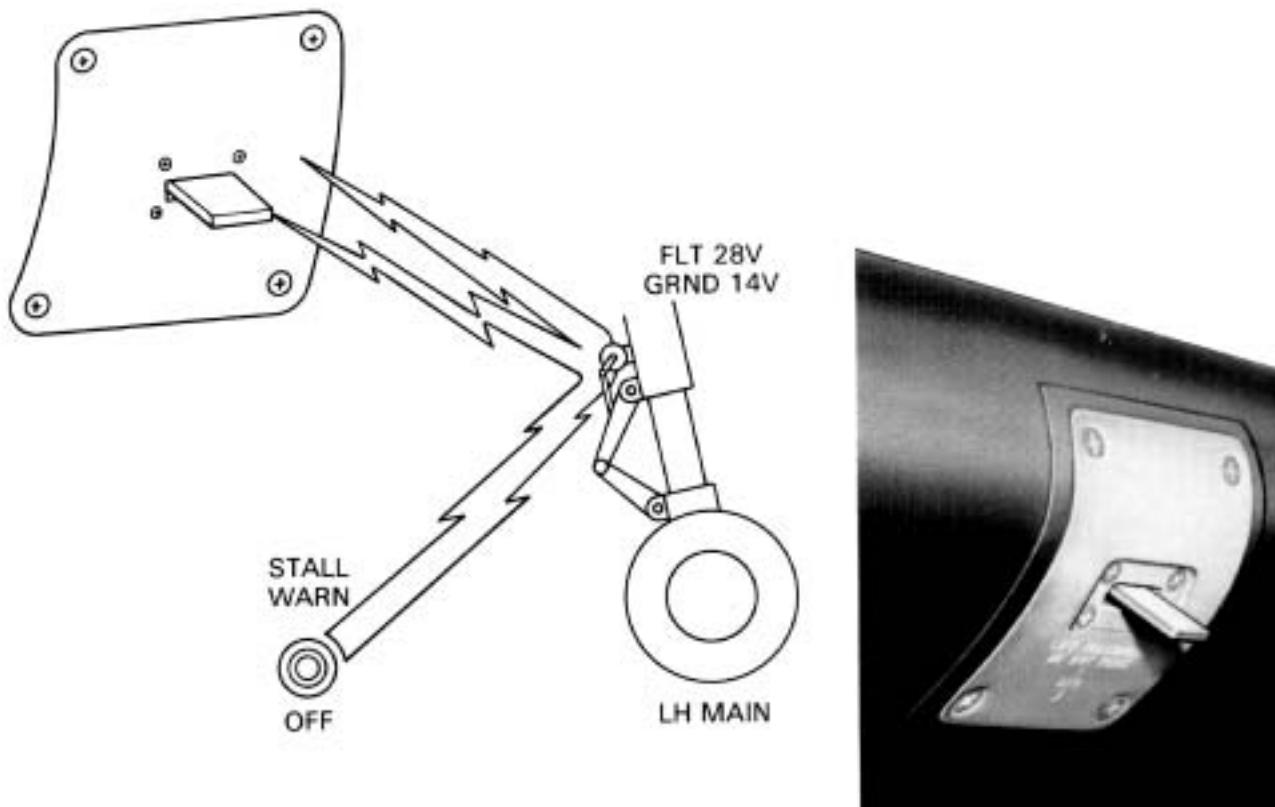


Figure 10-21 Stall Warning Vane and Heat Control

buildup of ice on the wing may change or disrupt the airflow and prevent the system from accurately indicating an imminent stall. Remember that the stall speed increases whenever ice accumulates on any airplane.

A safety switch on the left landing gear limits face plate and stall vane current flow to approximately 14 volts to prevent overheating while the airplane is on the ground. In flight, after the left strut extends, the full 28-volt current is applied to the stall warning face plate and stall vane.

SURFACE DEICE

The leading edges of the wings and horizontal stabilizer are protected against an accumulation of ice by inflatable boots (Figure 10-22). Ice removal is accomplished by alternately inflating and deflating the deice boots (Figure 10-23). Pneumatic system pressure (Figure 10-24) inflates the boots; system vacuum deflates the boots and holds them down

while not in use. Inflation and deflation phases are controlled by a deice distributor valve.

Each wing has an inboard and an outboard boot. The horizontal stabilizer has one boot on each of the left and right segments. The vertical stabilizer is not deiced, nor does it need to be, since no adverse effects were found during icing certification.

A three-position switch on the pilot's right sub-panel, placarded SURFACE DEICE - SINGLE - OFF - MANUAL, controls deicing operations (Figure 10-25). The switch is spring-loaded to - return to OFF from SINGLE or MANUAL. When SINGLE is selected, the distributor valve opens to inflate the wing boots. After an inflation period of approximately 6 seconds, an electronic timer switches the distributor to deflate the wing boots, and begins a four-second inflation cycle of horizontal stabilizer boots. Once these boots have deflated, the cycle is complete. Total cycle time is approximately 10 seconds.



Figure 10-22 Surface Deice Boot Installation

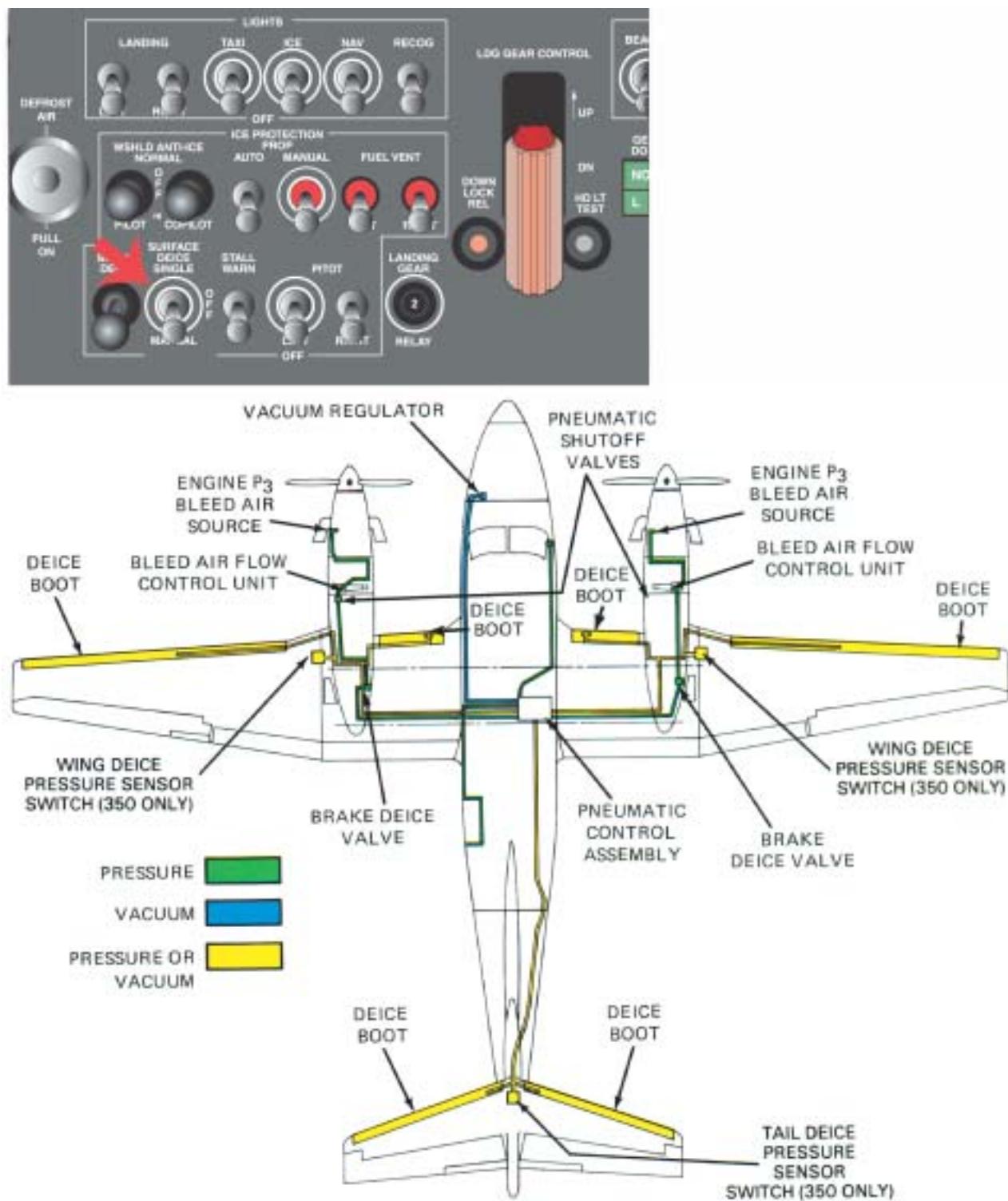


Figure 10-23 Surface Deice System Diagram

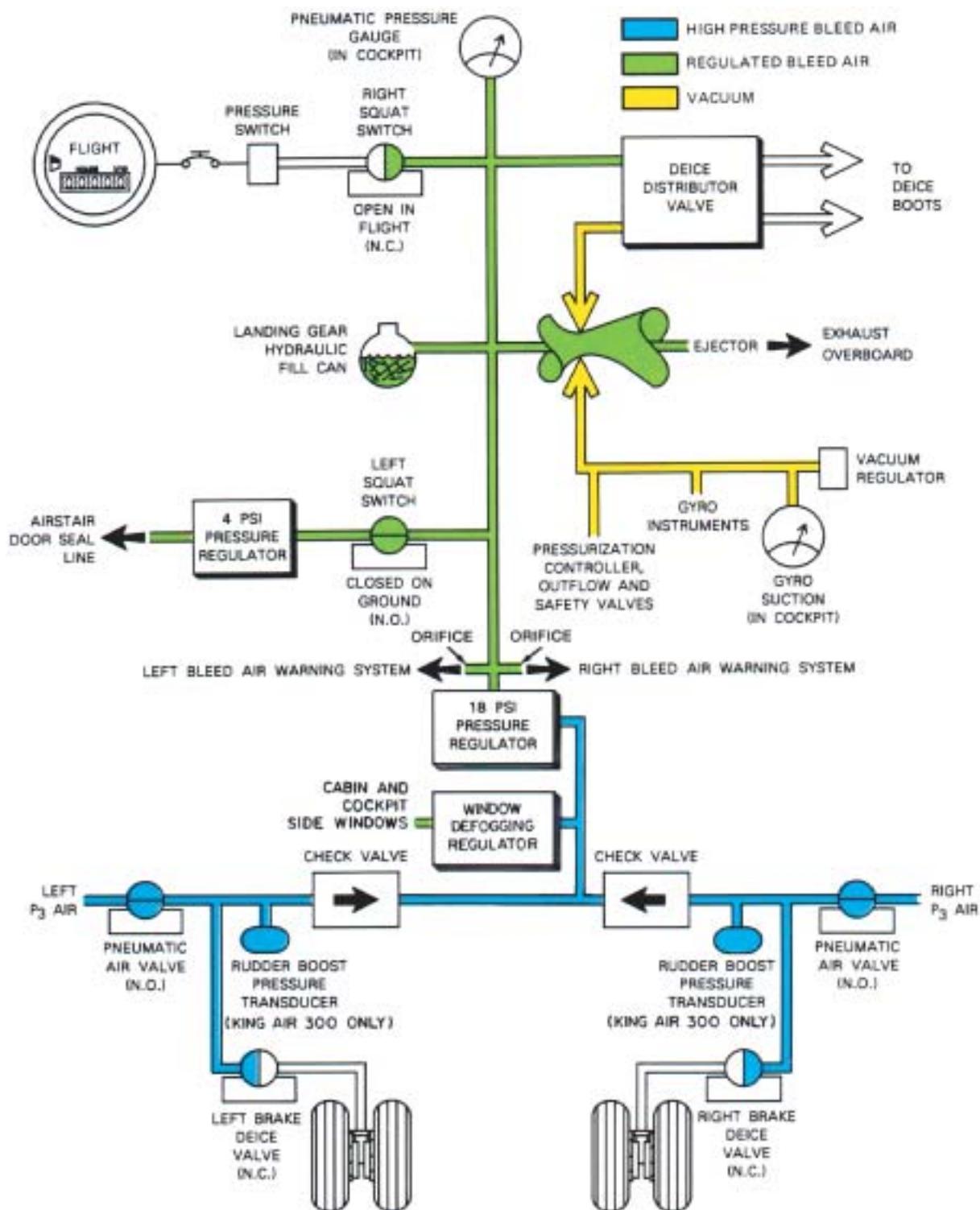


Figure 10-24 Pneumatic System Simplified Diagram

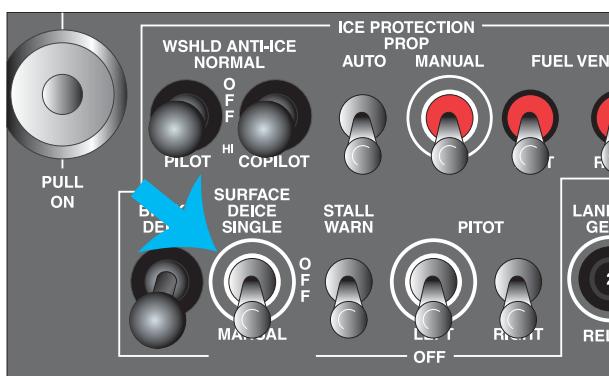


Figure 10-25 Surface Deice Controls

When the switch is held in MANUAL, all the boots will be inflated simultaneously. The boots remain inflated until the switch is released to OFF. After the cycle, the boots remain in the vacuum hold-down condition until again actuated by the switch.

Electrical power to the boot system is required for the distributor valve to inflate the boots in either single-cycle or manual operation. With a loss of this power, vacuum will hold them against the leading edge surfaces.

A SURF DEICE circuit breaker (Figure 10-26), located on the copilot's sidepanel, receives power from the triple-fed bus. It provides the electrical

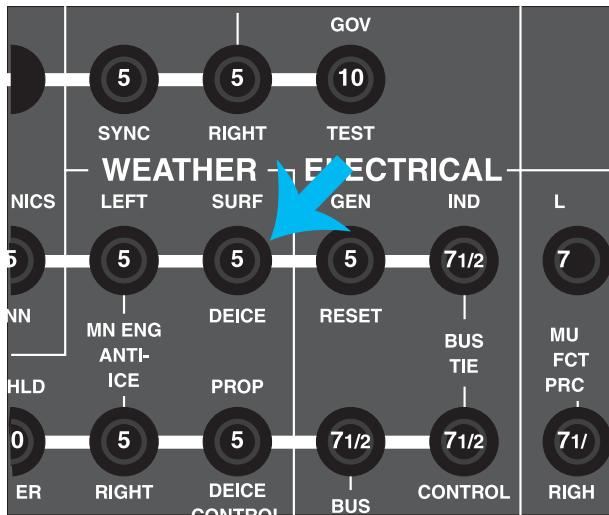


Figure 10-26 Surface Deice Circuit Breaker

power for both single cycle and manual surface deice system operation.

On the King Air 350, a green WING DEICE annunciation illuminates whenever the wing deice boots are inflated. There are 2 pressure sensing switches that sense the deice boot pressure, one for each wing. Both of these switches must sense 14-15 psi wing deice boot pressure to illuminate the WING DEICE annunciator. A green TAIL DEICE annunciation illuminates whenever the tail deice boots are inflated. One pressure sensing switch senses the tail deice boot pressure. When there is 14-15 psi tail deice boot pressure, this pressure switch will cause the TAIL DEICE annunciator to illuminate.

For most effective deicing operation, allow at least 1/2 inch, but no more than 1 inch, of ice to form before attempting ice removal. Very thin ice may crack and cling to the boots instead of shedding. Subsequent cyclings of the boots will then have a tendency to build up a shell of ice outside the contour of the inflated boot, thus making ice removal efforts ineffective. In addition, if the ice is allowed to build up to a depth greater than one inch, removal with the deice boots may be impossible.

BRAKE DEICE SYSTEM

Optional brake deicers may be installed on the main wheels (Figures 10-27 and 10-28). Their purpose is to prevent ice and slush from building up between the wheels and freezing the brakes. Heated air for brake deicing is supplied by bleeding air from the compressor of each engine. Brake deice is plumbed from the pneumatic system that provides air for other systems, such as surface deice and instrument vacuum operation.

This engine P₃ air is routed by a line on the outboard side of each nacelle to a solenoid-operated shutoff valve on the right side, inside each main gear wheel well. From the shutoff valve, the hot air is plumbed through an insulated line down the back of the main gear strut, to a distributor manifold which is attached to the brake piston and axle assembly. The hot P₃ air is directed on the brakes through orifices around the circumference of each ring of the distributor manifold.

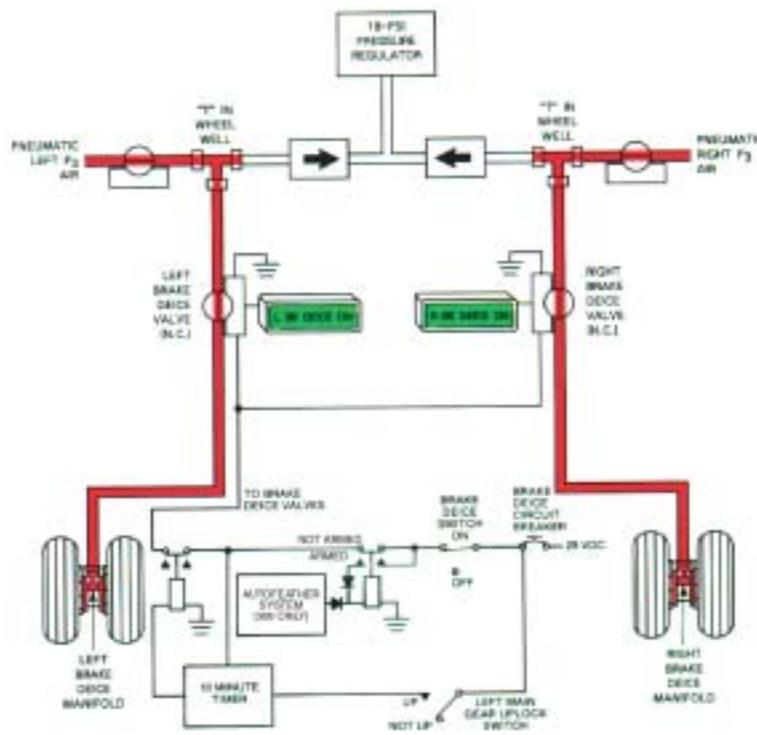


Figure 10-27 Brake Deice Schematic (System On)

The brake deice system is controlled by an on-off lever-lock switch mounted on the pilot's right subpanel (Figure 10-29). When this switch is up, power from the right generator bus is supplied through a 5-ampere circuit breaker in the copilot's sidepanel to a control module located under the center aisle floorboard, immediately aft of the partition between the cockpit and cabin. This module supplies current to open the normally closed solenoid shutoff valves in each wheelwell, allowing the hot bleed air to enter the brake deice lines. When the shutoff valves open, a signal illuminates the L and R BRAKE DEICE ON (green) annunciators on the advisory panel.

When the airplane is in flight with the control switch for the brake deice system switched on, a circuit is completed through the left main gear uplock switch to a timing circuit in the control module. This timing circuit will cycle the deice system off after 10 minutes of operation by closing the solenoid valves in the wheelwells. This

shuts off P₃ airflow to the brakes so that adjacent components in the wheelwell will incur no damage through overheating. The system cannot again be activated until the landing gear is cycled from the up and locked position.

On the King Air 300, the brake deice system is also wired through the autofeather system. When both autofeather annunciations illuminate, indicating the autofeather system is armed, the brake deice system is locked out. As a result, brake deice is not available during the test, takeoff climbout, or during the approach phase when the autofeather annunciations are illuminated. When the autofeather annunciations extinguish, indicating the autofeather system is disarmed, the BRAKE DEICE switch must be cycled off and on to restart the brake deice system.

The brake deice system is not to be continuously operated above 15°C ambient temperature.

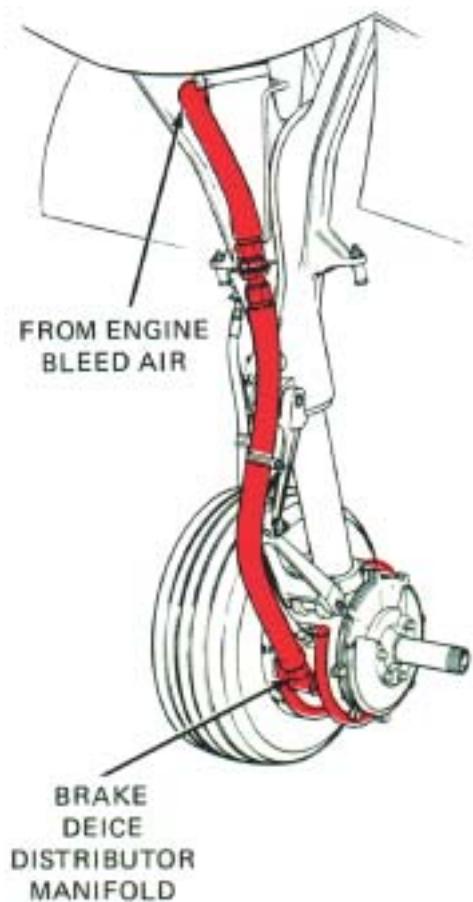


Figure 10-28 Brake Deice

PRECAUTIONS DURING ICING CONDITIONS

There are some precautions which prevail during winter or icing conditions. An airplane needs special care and inspection before operation in cold or potential icing weather. In addition to the normal exterior inspection, special attention should be paid to areas where frost and ice may accumulate.

Pilots should be familiar with the potential harm a harmless looking, thin layer of frost can cause. It is not the thickness of the frost that matters, it is the texture. A slightly irregular surface can substantially decrease proper airflow over the wings and stabilizers. Never

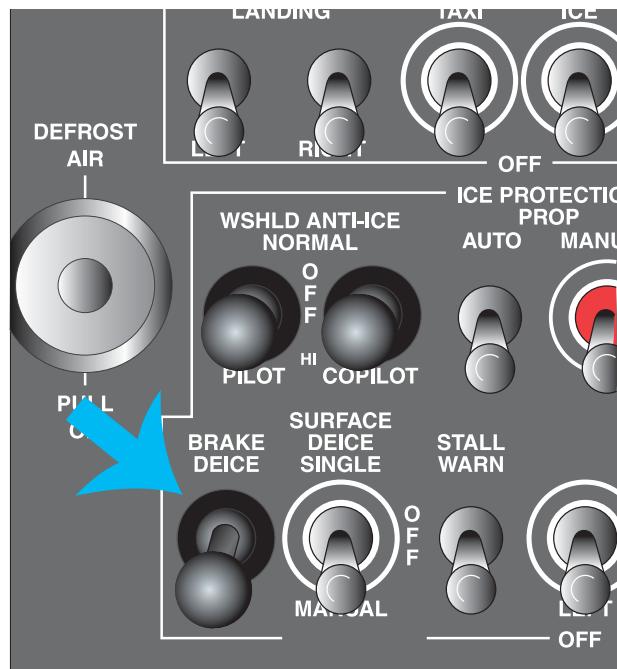


Figure 10-29 Brake Deice Controls

underestimate the damaging effects of frost. All frost should be removed from the leading edges of the wings, stabilizers and propellers before the airplane is moved.

The windshield, control surfaces, hinges, pitot masts, fuel tank caps and vents should also be free of frost. Deicing fluid should be used when needed.

Fuel drains should be tested for free flow. Water in the fuel system has a tendency to condense more readily during winter months, and if left unchecked, large amounts of moisture may accumulate in the fuel tanks. Moisture doesn't always settle at the bottom of the tank. Occasionally a thin layer of fuel gets trapped under a large mass of water which may deceive the tester. Make sure a good-sized sample of fuel is taken.

It is also important to add only the correct amount of anti-icing additive to the fuel. A higher concentration of anti-icer does not insure lower fuel freezing temperatures, and may hinder the airplane's performance. Consult the Normal Procedures section of the *Pilot's Operating Handbook* to determine the correct blend.



The brakes and tire-to-ground contact should be checked for lockup. No anti-ice solution containing oil-based lubricant should be used on the brakes. If tires are frozen to the ground, use undiluted defrosting fluid or a ground heater to melt ice around tires, then move the airplane as soon as the tires are free. Heat applied to tires should not exceed 160° F or 71° C.

Tie-downs for propellers should be installed to insure against damage to internal engine components not lubricated when the engine is not operating. Spinning propellers can also be a source of danger to crew, passengers and ground support personnel. Propeller blades held in their tie-down position channel moisture down the blades, past the propeller hub and off the lower blade more effectively than in other positions or when left spinning. During particularly icy ground conditions, the propeller hubs should also be inspected for ice and snow accumulation. The propellers should be turned by hand in their normal rotation direction to insure they are free prior to engine start.

Pitot masts should always be covered while the airplane is resting. Once the covers are removed make sure both masts and drains are free of ice or water. Faulty readings could be obtained if they are clogged.

Brake deice may be turned on before taxi to help rid the brake mechanisms of accumulated ice. If brake deice is used, place the condition levers in HIGH IDLE. Keep the flaps retracted to avoid throwing snow or slush into the flap mechanisms and to minimize the possibility of damage to the flap surfaces. When taxiing in extremely icy conditions, ensure the tires are rolling, not just sliding on the icy surface. The brake deice system is not to be continuously operated above 15°C ambient temperature.

Auto-ignition should be left off until immediately before takeoff. This will help to prolong the service life of the igniter units.

Snow, slush or standing water on the runway degrade airplane performance whether landing or taking off. During takeoff, more runway is needed to achieve necessary takeoff speed, while landing roll is longer because of reduced braking effective-

ness. When conditions are slushy or snowy, brake deice should be activated after establishing cruise flight and again, if necessary, before landing once the gear is down.

Only the surface and propeller deice systems are true deicers. The rest are really anti-icers and should be used to prevent the formation of ice, not to melt ice already present. Accumulated ice on even the best equipped airplane will degrade its performance and cause underestimation of time and fuel calculations used for flight planning. A minimum speed of 140 KIAS is necessary to prevent ice formation on the underside of the wing, which cannot be adequately removed by the deice system.

Due to distortion of the wing airfoil, stalling airspeeds should be expected to increase as ice accumulates on the airplane. For the same reason, stall warning devices are not accurate and should not be relied upon. Maintain a comfortable margin of airspeed above the normal stall airspeed when ice is on the airplane.

While in flight, the engine ice vanes must be extended and the appropriate annunciator lights monitored when freedom from visible moisture is not assured and the OAT is +5°C or below.

The inertial separators (ice vanes) shall be retracted for takeoff operations in ambient temperatures of above +10°C on the King Air 350 only.

During flight in icing conditions, fuel vent heat, pitot heat, prop deice, windshield heat, and stall warning heat should all be ON. In the event of windshield icing, it may be necessary to reduce airspeed below 226 KIAS.

If either BLEED AIR FAIL light illuminates in flight, the bleed air switch on the affected engine must be closed to the PNEU & ENVIR OFF position. This will isolate the brake deice system on that side; therefore, the brake deice system must be selected to OFF. BLEED AIR FAIL lights may momentarily illuminate during simultaneous wing boot and brake deice operation at low N₁ speeds. If lights immediately extinguish, they may be disregarded.



CHAPTER 11

AIR CONDITIONING

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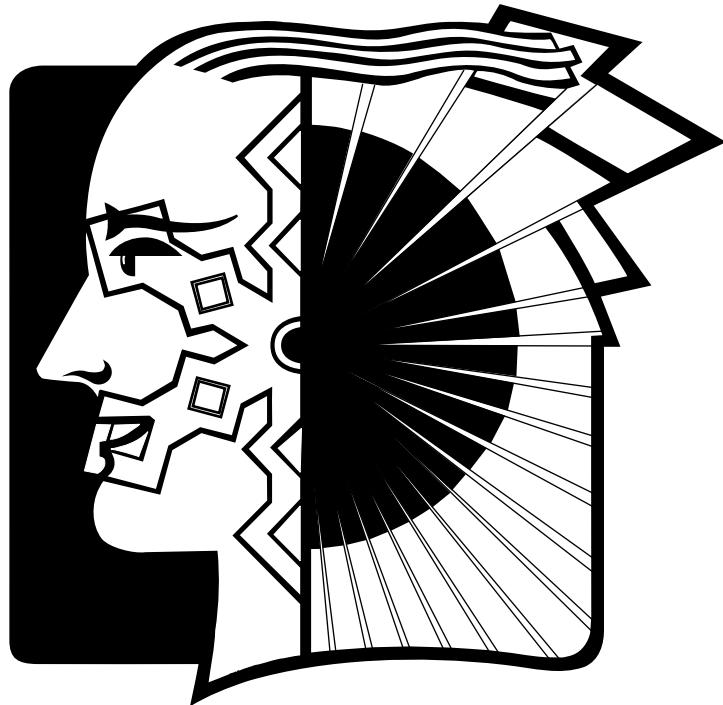


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CHAPTER 11

AIR CONDITIONING



INTRODUCTION

Since passenger comfort and safety are of prime importance, the task in this section is to describe how to operate the environmental systems effectively and within their limitations.

DESCRIPTION

The Environmental System section presents a description and discussion of the air-conditioning, heating, and fresh-air systems. Each system includes general description, principle of operation, controls, and procedures.

ENVIRONMENTAL SYSTEM

The Super King Air 300 and 350 environmental system consists of bleed air pressurization, venti-

lation, heating and cooling systems, and their associated controls. The pressurization and environmental systems operate in conjunction with each other and as separate systems to maintain the desired cabin pressure altitude and air temperature. This system (Figures 11-1 and 11-2) uses engine bleed air for cabin pressurization and cabin heating. Electric heating elements in the environmental ducting provide supplemental heat on the ground. A freon air conditioning (a/c) system is provided to supply additional cooling to the airplane cabin.

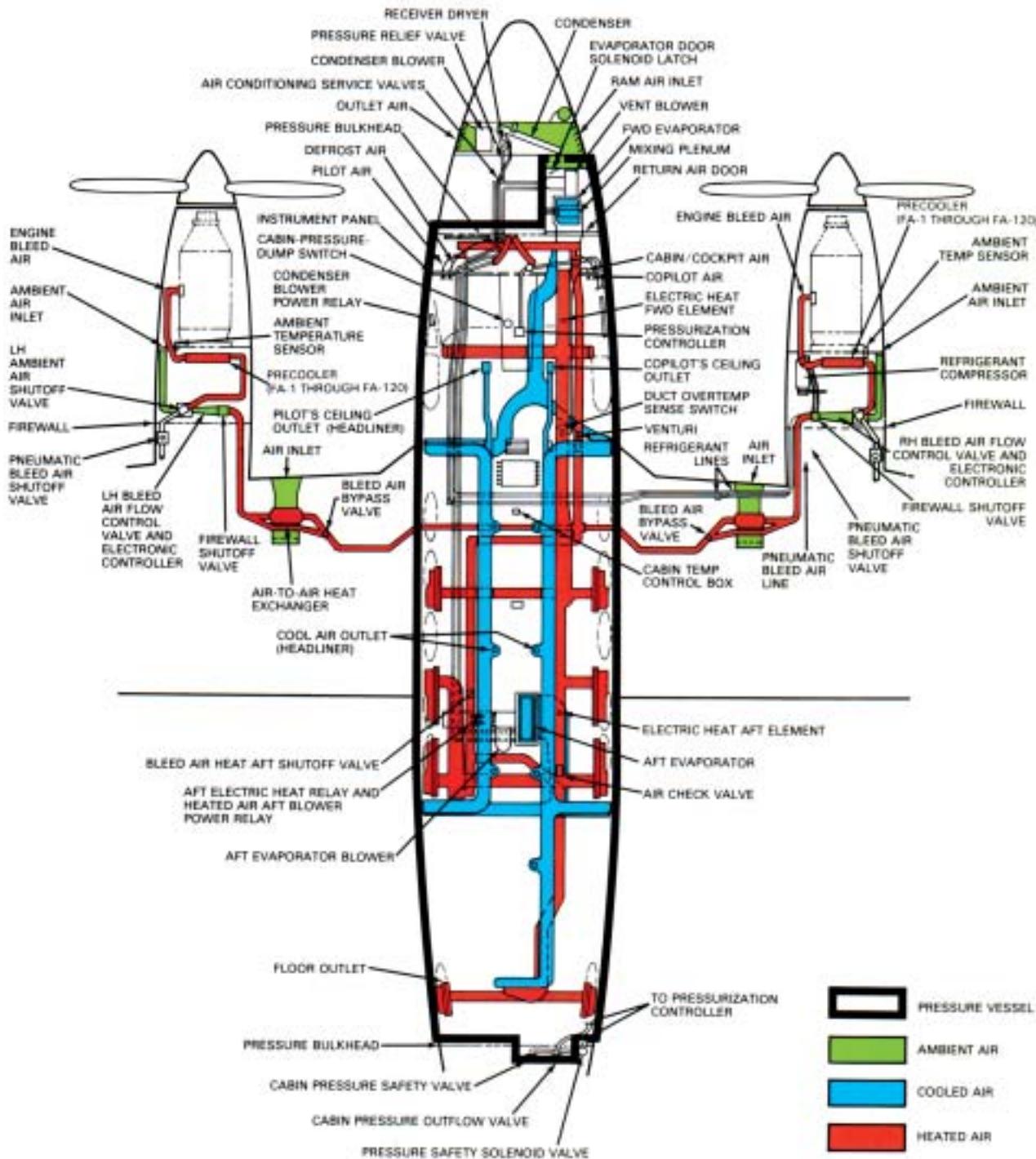
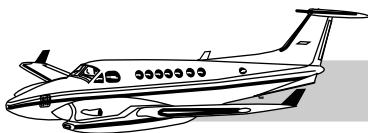


Figure 11-1 King Air 300 Pressurization and Environmental Distribution System

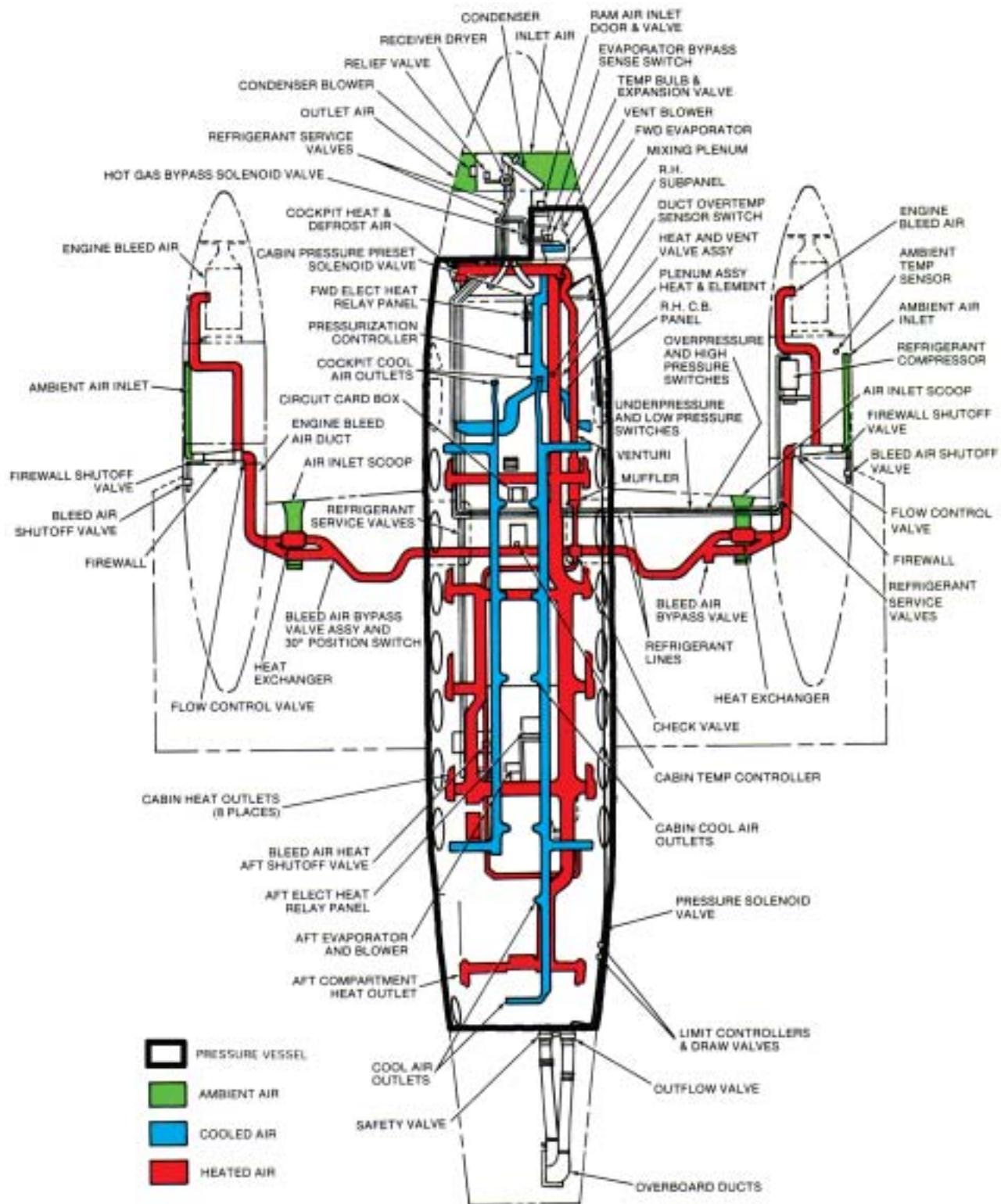


Figure 11-2 King Air 350 Pressurization and Environmental Distribution System



The cockpit and cabin compartments are pressurized and either heated or cooled through a common ducting arrangement. Ventilation can be obtained during nonpressurized flight through the ram air (evaporator) door (Figure 11-3) in the nose compartment.

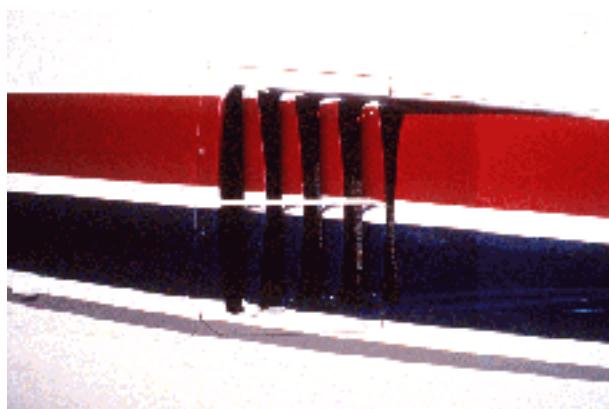


Figure 11-3 Ram Air Scoop

ENVIRONMENTAL CONTROLS

The ENVIRONMENTAL switch section on the copilot's left subpanel (Figure 11-4) provides for automatic or manual control of the system. This section contains all major controls of the environmental function: bleed air valve switches, forward vent blower control switch, aft evaporator blower control switch, electric heat switch, aft bleed air heat switch, manual temperature switch for control of the cabin temperature control (bypass) valves in the air-to-air heat exchangers, cabin temperature selector and the cabin temperature mode selector switch for selecting automatic heating or cooling, manual heating or cooling, or off.

Four additional manual controls on the main instrument subpanels may be utilized for partial regulation of cockpit comfort and temperature balance between the cockpit and cabin when the cockpit partition doors are

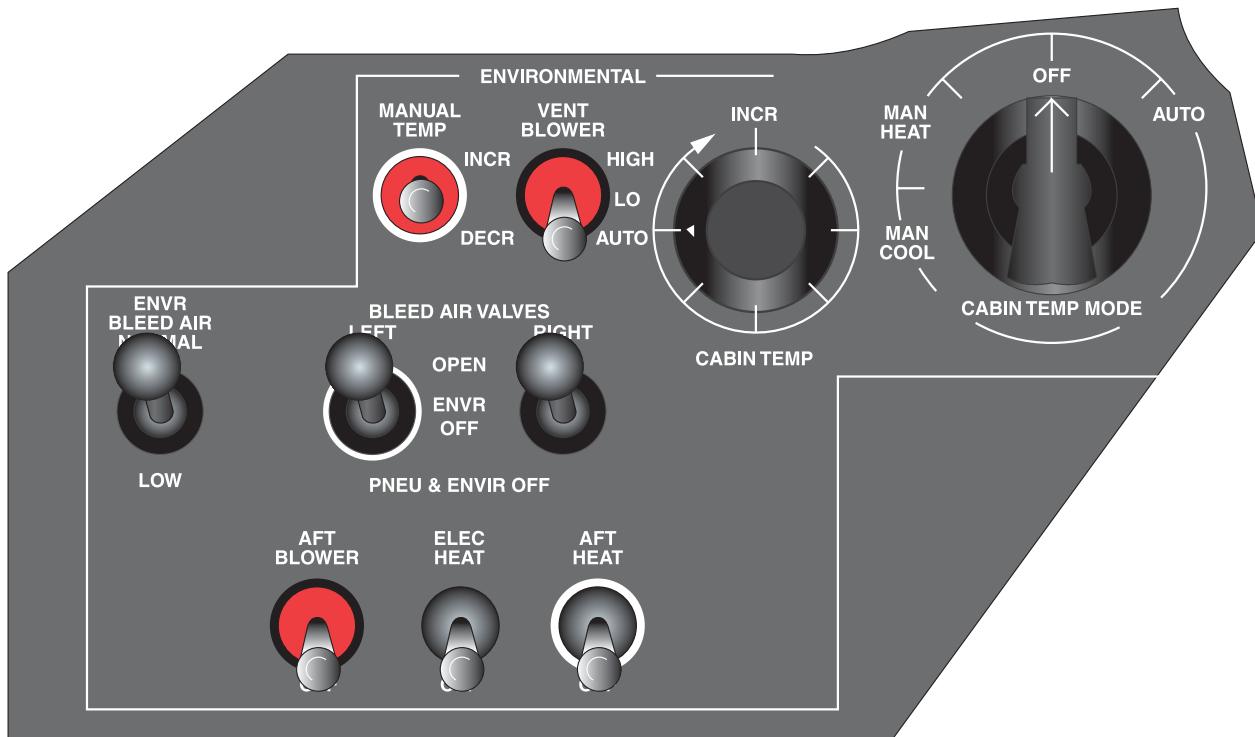
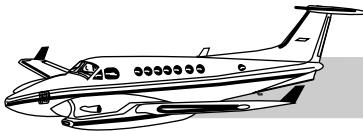


Figure 11-4 ENVIRONMENTAL Group Switches and Knobs



open. They are: PILOT AIR, DEFROST AIR, CABIN/COCKPIT AIR, and COPILOT AIR control knobs. The full out position of these controls will provide maximum airflow to the cockpit, and the full in position will provide maximum airflow to the cabin.

The overhead eyeball outlets, located in the cockpit and cabin headliner, provide recycled cabin air and cool conditioned air (when the air conditioner is operating). Since this air is always cooler than the air coming from the floor outlets in all temperature modes of operation, the eyeball outlets should be adjusted during preflight to allow for specialized flight profiles. For instance, prior to short low altitude flights in summer, all the overhead outlets should be fully open for maximum cooling. However, for high-altitude, night, and/or flights in cold weather, the overhead outlets should all be closed for maximum heating in the cabin.

BLEED AIR CONTROL

Bleed air entering the cabin is controlled by the two switches placarded BLEED AIR VALVES – OPEN – ENVIR OFF – PNEU & ENVIR OFF (Figure 11-5). When a BLEED AIR VALVE switch is in the OPEN position, the environmental flow control unit and pneumatic instrument air valve are open. When the switch is in ENVIR OFF, the environmental flow control unit is closed and pneumatic instrument air valve is open; in PNEU & ENVIR OFF, both are closed.

Environmental bleed air volume is controlled by the switch placarded ENVIR BLEED AIR – NORMAL

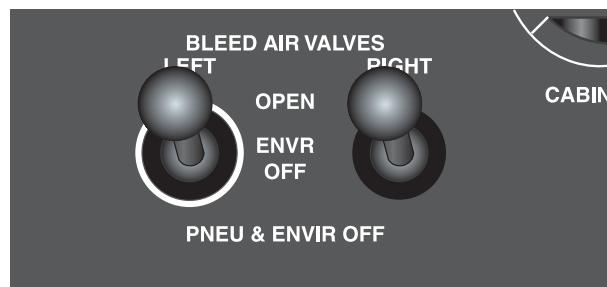


Figure 11-5 BLEED AIR VALVES Switches

– LOW (Figure 11-6) in the ENVIRONMENTAL controls group on the copilot's left subpanel. The LOW position must be selected for takeoff when ambient temperature is above 10° C. NORMAL may be used for increased heating or if increased pressurization air flow is required.

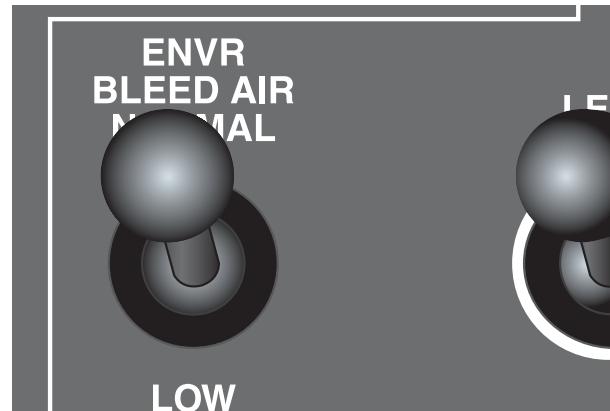


Figure 11-6 ENVIR BLEED AIR Switch

AUTOMATIC TEMPERATURE CONTROL

When the CABIN TEMP MODE selector switch (Figure 11-7) on the copilot's left subpanel is in AUTO, the heating and air conditioning systems operate automatically. A temperature selector, placarded CABIN TEMP – INCR (Figure 11-8), provides regulation of the temperature level in the AUTO mode. This temperature selector, in conjunction with a temperature sensor located in the bleed air duct, and a temperature sensor in the cabin installed under the headliner access panel in line with the second cabin window, are connected to a control box by means of a balanced bridge circuit. The control box regulates incoming environmental bleed air temperature by sending heat or cool commands to the bleed air bypass valves.

If a warmer cabin temperature has been selected, the automatic temperature control monitor adjusts the bypass valves one at a time to allow heated air to bypass the air-to-air heat exchangers in the center wing sections. Warm environmental air is then delivered to the cabin. Once the

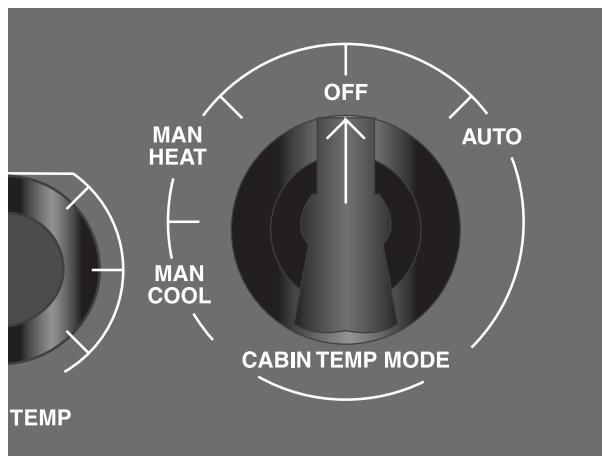


Figure 11-7 CABIN TEMP MODE Selector Switch

desired temperature is achieved, the automatic temperature control system constantly modulates the bypass valves to maintain proper incoming bleed air temperature.

When the automatic control drives the environmental system from a heating mode to a cooling mode, the bypass valves move toward the cool position (bleed air passes through the air-to-air heat exchanger). When the left bypass valve reaches the 0° or full cold position, the air conditioning system is activated, provided the right engine speed is above 62% N₁. When the left bypass valve re-opens more than 30° (toward the heat position), the air conditioning system is turned off.

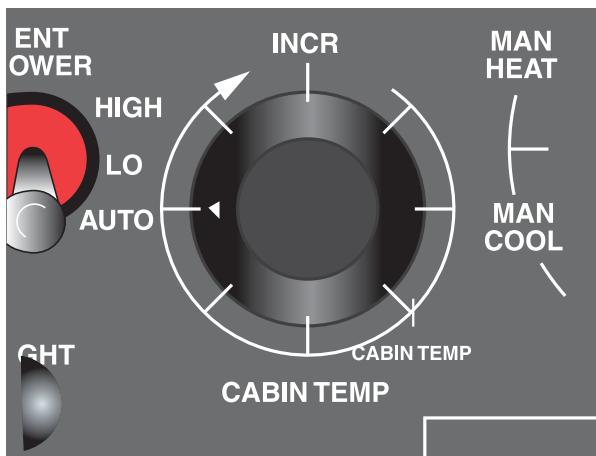


Figure 11-8 CABIN TEMP Level Control

MANUAL TEMPERATURE CONTROL

When the CABIN TEMP MODE selector is in MAN HEAT or MAN COOL, regulation of cabin temperature is accomplished manually by momentarily holding the MANUAL TEMP switch (Figure 11-9) to either INCR or DECR position, as desired. When released, this switch will return to the center (no change) position. Moving this switch to the INCR or DECR position results in modulation of the bypass valves in the environmental air lines in the wing center sections. Allow approximately 30 seconds per valve (one minute total time) for the valves to move to the full heat or full cold position. Only one valve moves at a time. Movement of these valves adjusts the amount of environmental air routed through the air-to-air heat exchanger which will adjust the temperature of incoming cabin environmental air accordingly (Figure 11-10). The cabin temperature will vary according to the position of the bypass valves whether the air conditioner is operating or not.

When the CABIN TEMP MODE selector is in MAN COOL position, the air conditioner system will operate provided the left bypass valve is within 30° of full cold position and operation is not prohibited by system protection controls. System protection controls will prevent compressor operation if refrigerant pressure is too high or too low, if the ambient temperature is below approximately 50°F, or if the right engine speed is below 62% N₁.

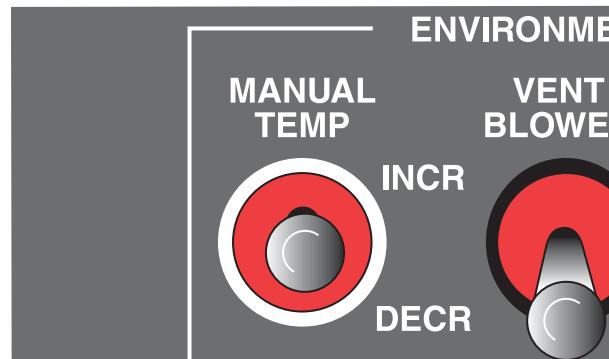


Figure 11-9 MANUAL MODE Switch

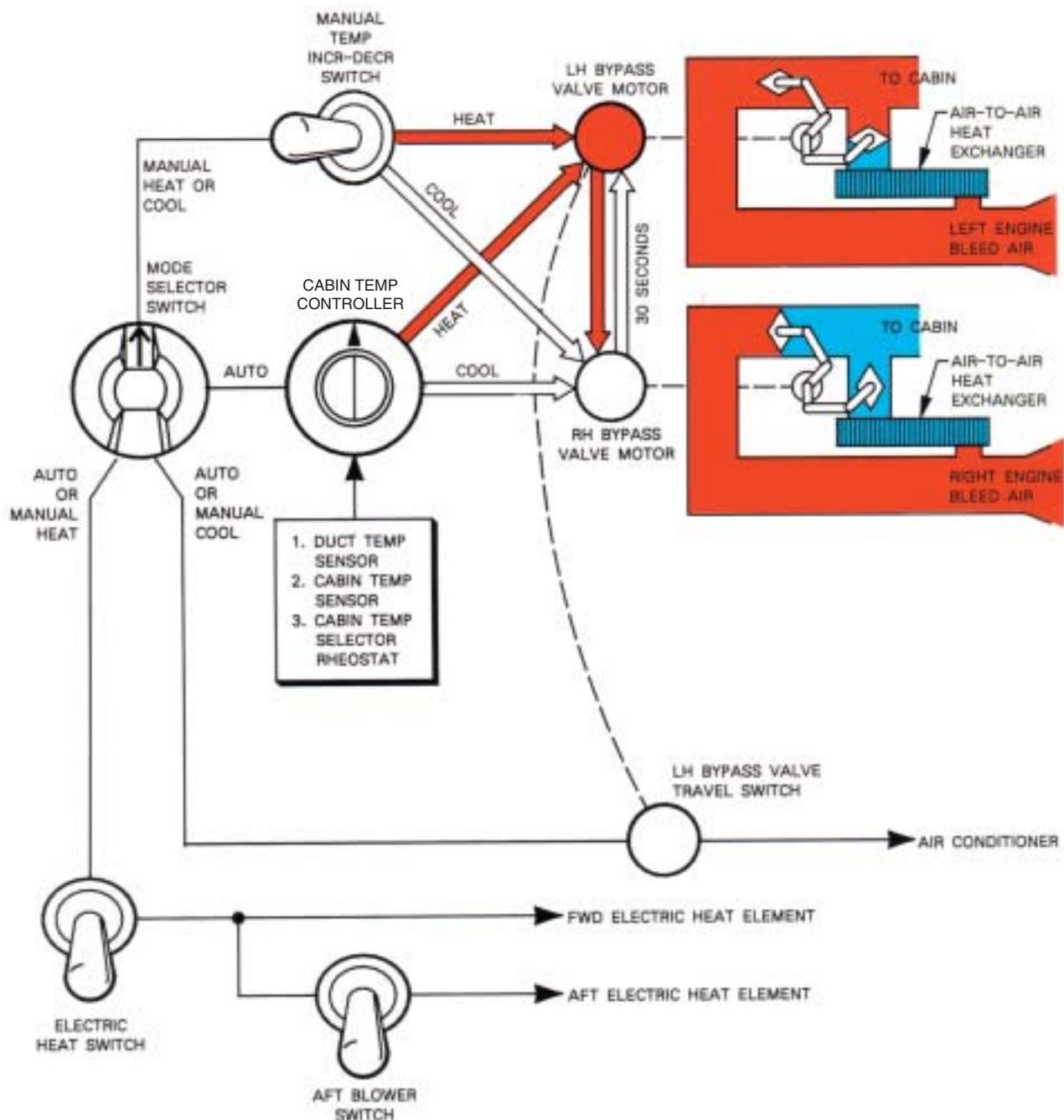
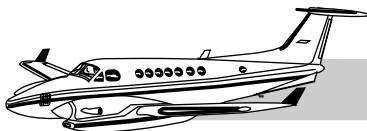


Figure 11-10 Air-to-Air Heat Exchanger Diagram



VENT BLOWER

The forward vent blower is controlled by a switch in the ENVIRONMENTAL group (Figure 11-11) placarded VENT BLOWER – HI – LO – AUTO. When this switch is in AUTO, the forward vent blower will operate at low speed when the CABIN TEMP MODE selector switch is in any position other than OFF (i.e., MAN COOL, MAN HEAT, or AUTO).

When the VENT BLOWER switch is in AUTO and the CABIN TEMP MODE selector switch is in OFF, the blower will not operate. Any time the VENT BLOWER switch is in LO, the forward vent blower will operate at low speed even if the CABIN TEMP MODE selector switch is OFF. Whenever the VENT BLOWER switch is in HI, the forward vent blower will operate at high speed regardless of the CABIN TEMP MODE selector switch position (i.e., MAN COOL, MAN HEAT, OFF, or AUTO).

AFT BLOWER

An aft blower is installed under the aft cabin immediately forward of the airstair door under the center aisle floorboard. The aft blower draws in cabin air and recycles it directly back into the aft floor and overhead outlets. This blower operates at one speed only. It is turned on and off by the AFT BLOWER switch in the ENVIRONMENTAL group (Figure 11-12) on the copilot's

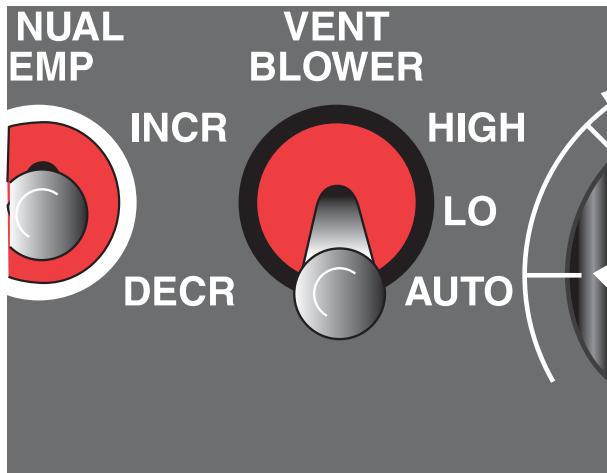


Figure 11-11 VENT BLOWER Switch

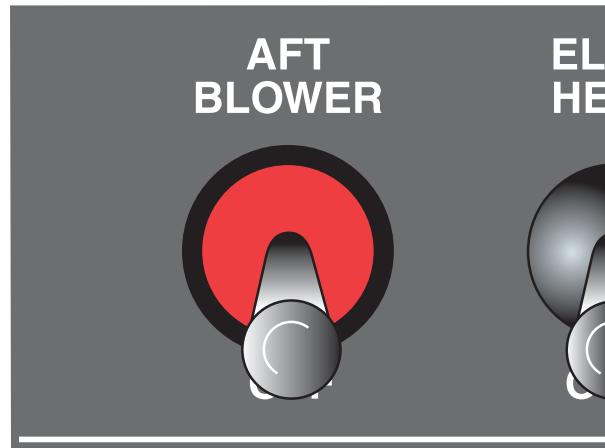


Figure 11-12 AFT BLOWER Switch

left subpanel. The blower is independent of any other control.

The aft blower is intended to distribute heat produced by the aft electric heat element or cooled air from the aft evaporator.

UNPRESSURIZED VENTILATION

Fresh air ventilation is provided from two sources. One source, available in both pressurized and unpressurized modes, is engine bleed air. This air mixes with recirculated cabin air in the forward mixing plenum and enters the cabin through the glare-shield "eyeball" outlets and floor registers (Figures 11-13, 11-14, and 11-15). Air ducted to each individual overhead eyeball



Figure 11-13 Glareshield "Eyeball" Outlets



outlet can be directionally controlled by moving the eyeball in the socket, while volume can be regulated by twisting the outlet to open or close the damper. There are no individual controls for the floor outlets.



Figure 11-14 Cabin "Eyeball" Outlets

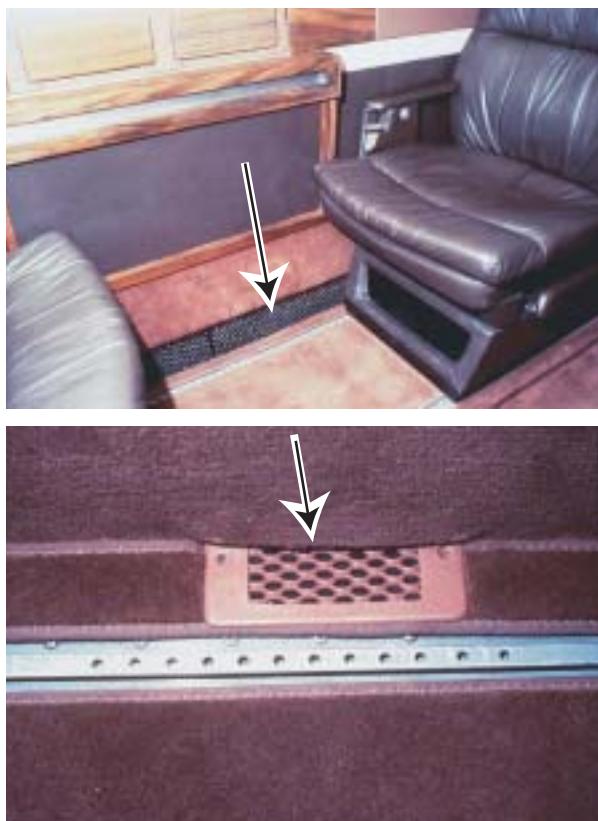


Figure 11-15 Cabin and Cockpit Floor Outlets

The second source of fresh air, available in the unpressurized mode only, is ambient air obtained through an evaporator (ram air) door from the condenser section (Figure 11-16) in the nose of the airplane. During pressurized operation, spring tension, an evaporator door solenoid magnet, and cabin pressure hold the evaporator door closed. During unpressurized flight, ram air will open the door if the solenoid magnet releases, allowing ambient air into the forward evaporator compartment. Once opened in flight, the evaporator door may not close until the aircraft lands since ram air pressure may hold it open unless the aircraft is repressurized. Ambient air then mixes with recirculated cabin air, flows through the forward vent blower, through the forward evaporator (if air conditioner is operating, the air will be cooled), into the mixing plenum (Figure 11-17). From there, unpressurized air is distributed to the overhead and floor outlets

BLEED AIR HEATING SYSTEM

Air for cabin pressurization, heating the cabin and cockpit, and operating the pneumatic and vacuum systems is obtained by bleeding P₃ air from the compressor of each engine. When air is compressed, its temperature is increased. Therefore, the bleed air extracted from each engine is hot. This heat is utilized to warm the cabin. Normally, bleed air from one engine is sufficient to continue to provide adequate air for pressurization, heating, and for the pneumatic and vacuum systems should the other engine fail.

Engine P₃ bleed air is ducted from the engine (through a nacelle precooler only on King Air 300s FA-1 through FA-120) to the flow control unit mounted on the firewall. The bleed air and ambient air from the cowling intake are mixed together by the flow control units, and are routed aft through the firewall along the inboard side of each nacelle, and inboard along the center wing section forward of the main spar.

When the left landing gear safety (squat) switch is in the "on ground" position, the ambient air valve (Figure 11-18) in each flow control unit is closed. Consequently, undiluted P₃ air is delivered to the environmental duct when the airplane is on the ground. The exclusion of ambient air

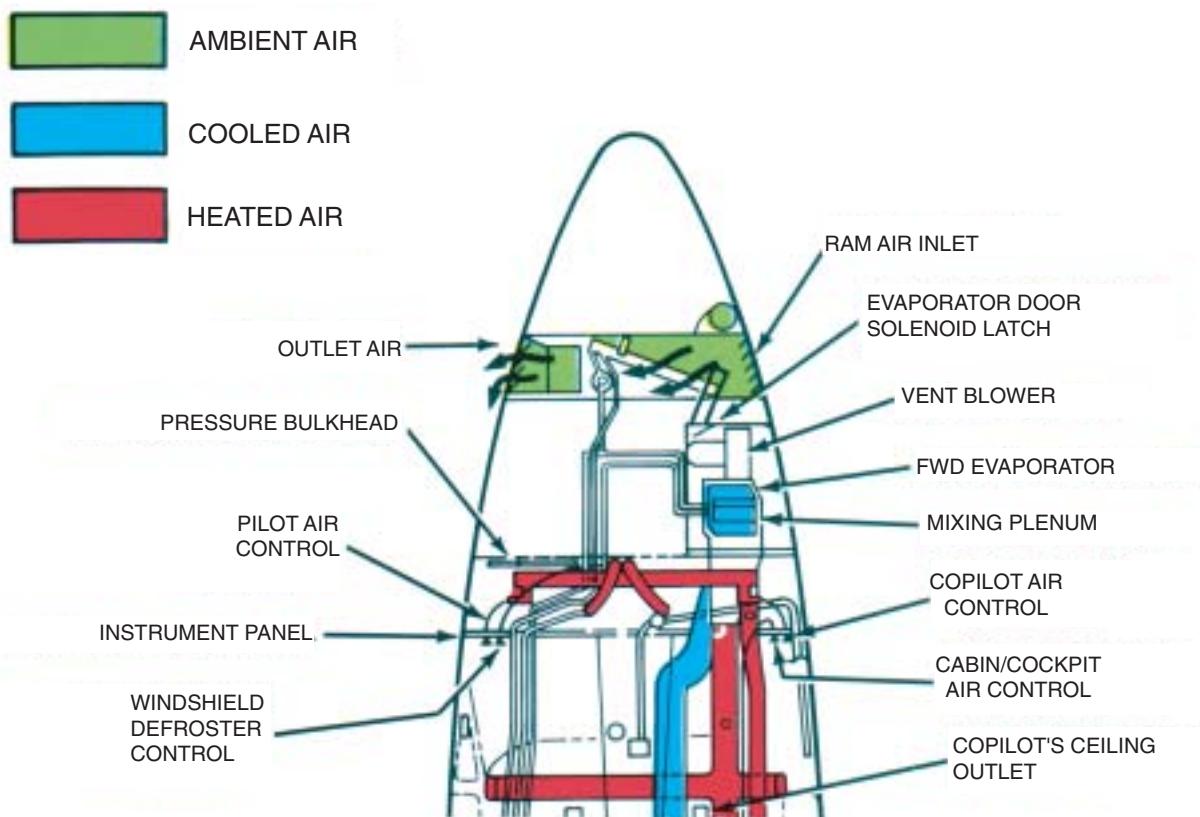


Figure 11-16 Fresh Air Source (Unpressurized Mode)

allows faster cabin warmup during cold weather operation. In flight, when ambient temperature is above approximately 50°F, the ambient air valve is open, which mixes ambient air with P_3 air in the flow control unit which increases environmental air flow. This enhanced environmental bleed air mixture is then routed into the cabin.

During warm weather ground operation, environmental air into the cabin can be shut off by placing the BLEED AIR VALVE switches on the copilot's left subpanel to the ENVIR OFF position. In pressurized flight, since the flow control valve must be open to maintain pressurization, a lower environmental air flow rate may be desirable when maximum cooling of the cabin is desired. The bleed air flow rate can be reduced by approximately 50% when the ENVIR BLEED AIR switch is set to LOW.

Heat in the environmental air may be either retained for cabin heating or dissipated for cool-

ing purposes as it passes through the air-to-air heat exchangers in each center wing section on its way to the fuselage. An air intake on the leading edge of the center wing section brings ram (outside ambient) air into the air-to-air heat exchanger to reduce the temperature of the cooling fins in the heat exchanger. The temperature of the environmental air flowing through the heat exchanger is lowered as its heat is transferred to the cooling fins. Depending upon the position of the bypass valves, the environmental air mixture will be routed either through or around the heat exchanger. If the environmental air entering the cabin is too warm for cabin comfort, the bypass valves (see Figure 11-9) motor to send more environmental bleed air through the air-to-air heat exchanger. If the cabin temperature is too cold, the bypass valves motor toward the bypass position to bypass more environmental bleed air around the heat exchangers (to retain its natural heat). The posi-

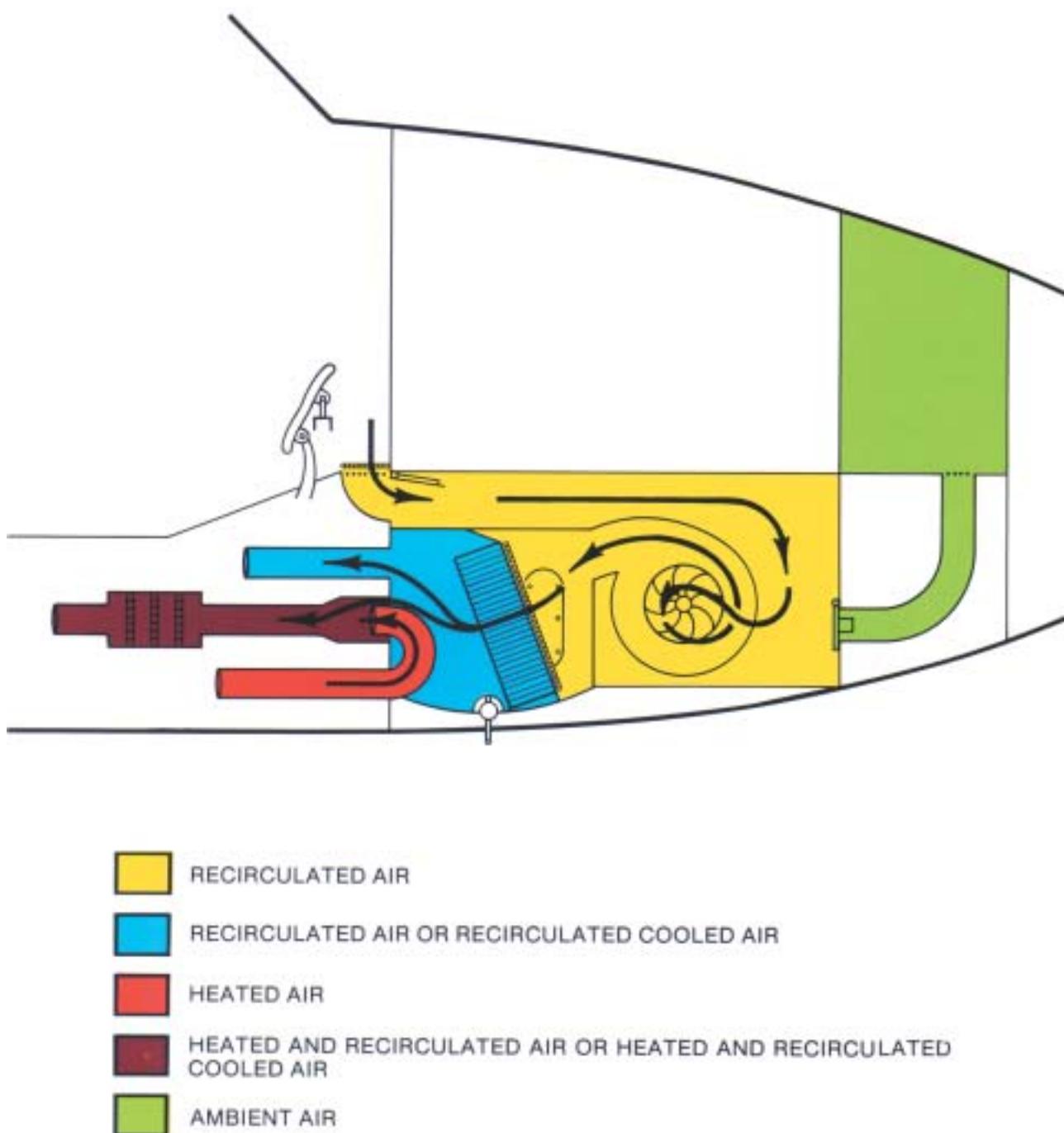
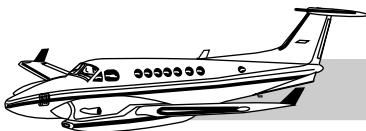


Figure 11-17 Mixing Plenum

tion of the damper in the bypass valves is determined by the controls in the ENVIRONMENTAL group on the copilot's left subpanel.

After passing through the heat exchanger, the ram (outside ambient) air is ducted overboard through louvers on the underside of the wing.

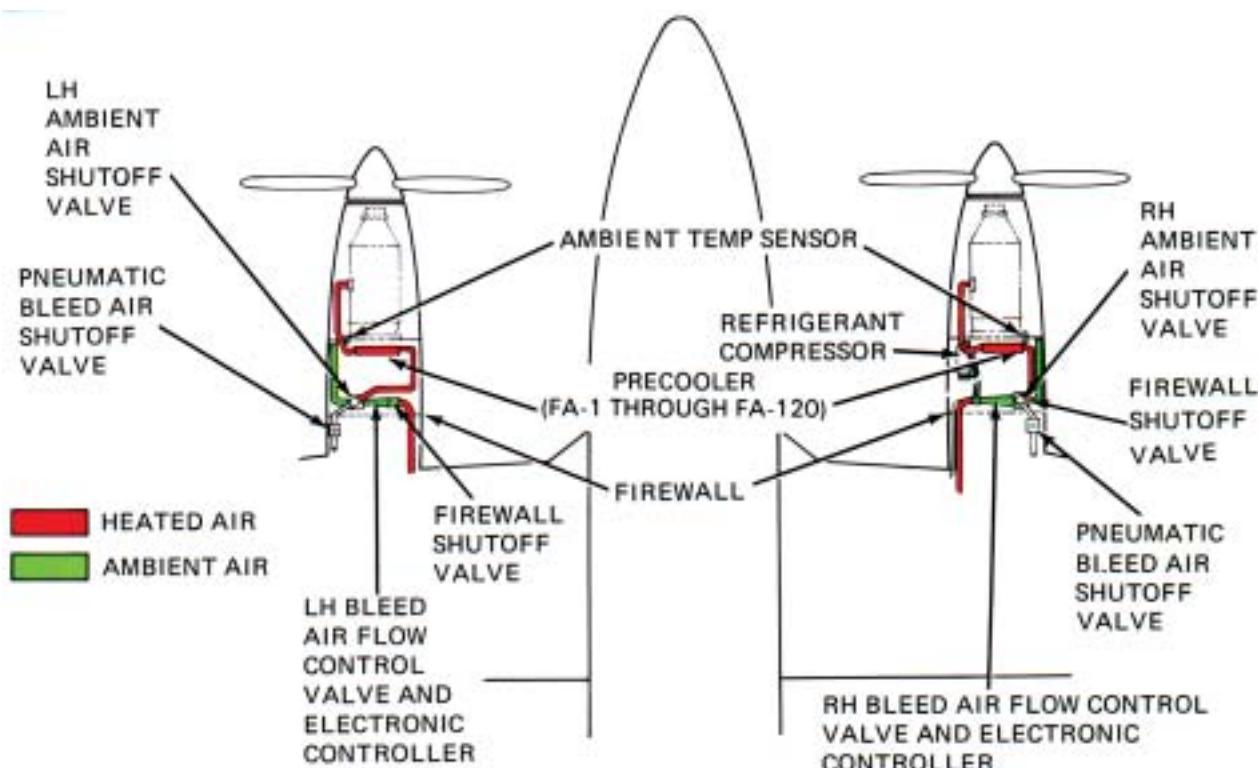


Figure 11-18 Ambient and Bleed Air Flow Forward of Firewalls

Upon exiting both left and right center wing ducts, the “conditioned” environmental air from each side is ducted together at a cross-junction in the center fuselage section. From there it goes forward through a single muffler. The muffler is located on the right side of the cabin under the floorboard forward of the main spar, and helps ensure quiet operation of the environmental bleed air system. The air mixture is then ducted from the muffler into the mixing plenum, located under the copilot’s floorboard.

When the aft bleed air heat control switch (Figure 11-19) is set to AFT HEAT, some environmental air flows from the aft side of the cross junction through the aft bleed air heat valve and directly to the aft floor outlet ducts to provide additional bleed air heat to the aft cabin area. The motor-driven valve, located on the left side of the cabin under the floorboard, opens when the AFT HEAT switch is turned on, and closes when it is turned OFF.

A partition divides the mixing plenum into two sections. One section supplies the defrost and floor outlet ducts, while the other supplies the headliner duct. Both sections receive recirculated cabin air from the forward vent blower. The recirculated air passes through the forward evaporator so it will be cooled if the air conditioner is operating. In the event that the forward vent blower becomes inoperative, some air will still be circulated due to the duct design in the discharge side of the mixing plenum.

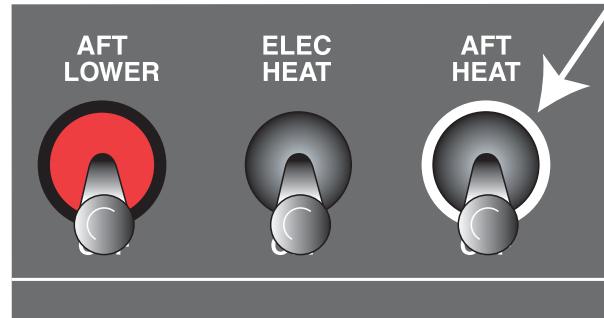
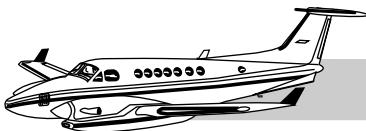


Figure 11-19 AFT HEAT Switch



The environmental air duct is routed into the mixing plenum, then curves back to discharge the environmental air toward the aft end of the floor duct section of the mixing plenum. Forward of the discharge end of the environmental air duct (see Figure 11-16), a small portion of this warm air can be tapped off, ducted up through the top of the mixing plenum, and delivered to the defrost/pilot/copilot heat duct, which is located below the instrument panel.

The DEFROST AIR control knob (Figure 11-20) is located on the pilot's subpanel, just below and inboard of the control column. This knob controls a butterfly valve at the forward side of the defrost/pilot/copilot heat duct which allows warm air to be delivered to the windshields and to glareshield-mounted eyeball outlets for the pilot and copilot. In addition, a mechanically-controlled damper in each duct permits the volume of airflow to be regulated to the crewmember's feet. The pilot's damper is controlled by the PILOT AIR (Figure 11-21) knob, located on the pilot's left subpanel, just below and outboard of the control column. The copilot's damper is controlled by the COPILOT AIR (Figure 11-22) knob, located on the copilot's subpanel just below and outboard of the control column.

The majority of the air in the environmental air duct is discharged into the floor outlet duct section of the mixing plenum. The floor outlets receive the warmest air and are located adjacent to the sidewalls on the floor of the passenger compart-

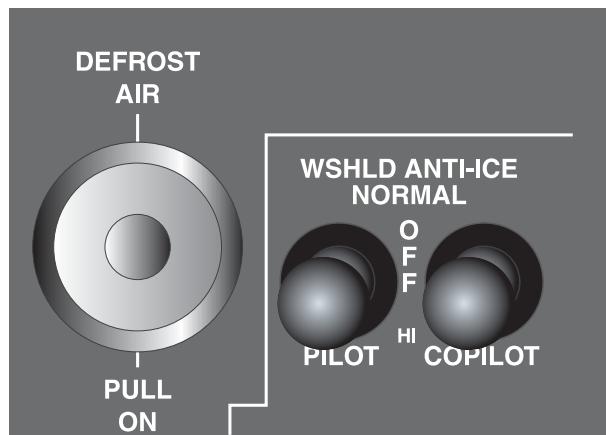


Figure 11-20 Air Control Knobs—Defrost Air

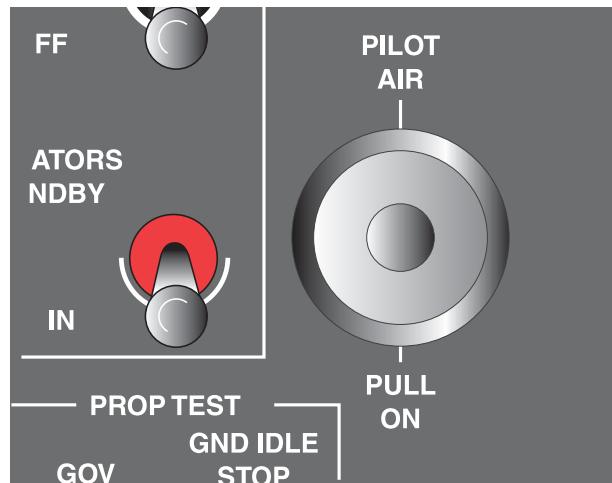


Figure 11-21 Air Control Knobs—Pilot Air

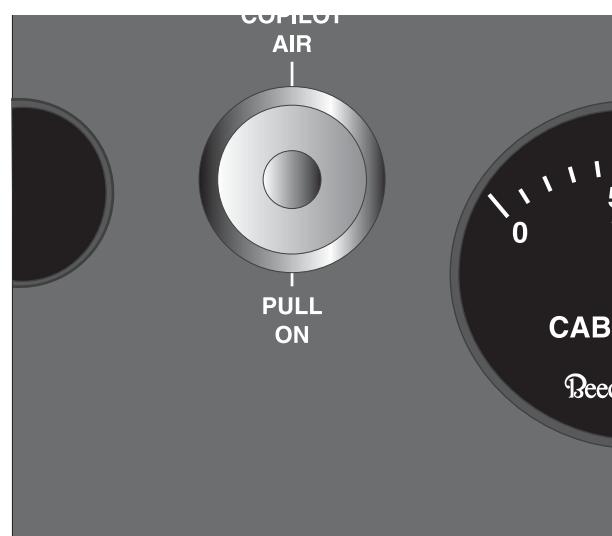
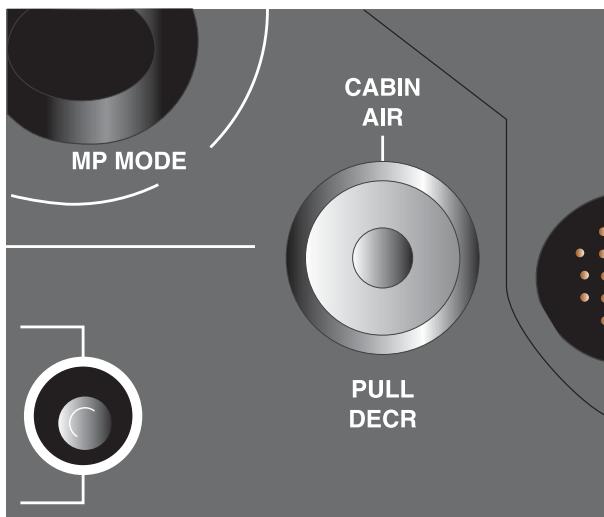


Figure 11-22 Air Control Knobs—Copilot Air

ment and the flight compartment. The headliner ducts receive cooler air because it is a recycled mixture. They distribute air to overhead eyeball outlets located in the cockpit and cabin above each of the normal crew/ passenger seat positions.

A butterfly valve installed in the ducting is controlled by the CABIN AIR control knob (Figure 11-23) on the copilot's subpanel, just below and inboard of the control column. When this knob is pulled out to the stop, only about 50% of the air will be permitted to pass through the valve, thereby increasing the amount of air available to the flight compartment. When this knob is

**Figure 11-23** Air Control Knobs—Cabin Air

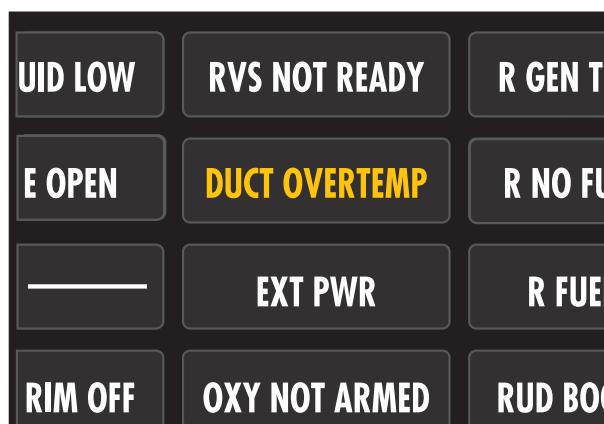
pushed fully in, the valve is open and most of the air in the duct will be directed to the floor outlet registers in the cabin.

If the air temperature inside the main duct aft of the mixing plenum becomes excessive, an over-temperature sensor inside the duct causes the yellow DUCT OVERTEMP caution annunciator (Figure 11-24) to illuminate. This sensor is located aft of the mixing plenum and forward of the butterfly valve. See the Abnormal Procedures section of the *POH* for more information about DUCT OVERTEMP.

ELECTRIC HEAT

When the airplane is on the ground, bleed air heat can be supplemented by the forward and aft electric heat elements (Figure 11-25). Electric heat is operated by a switch in the ENVIRONMENTAL group on the copilot's left subpanel (Figure 11-26), placarded ELEC HEAT - OFF. Due to electrical power requirements, a ground power unit must be used in conjunction with electric heat when warming the cabin prior to engine start. It may be used in conjunction with the MAN HEAT or AUTO temperature control modes only. In order to receive full heating capacity, the AFT BLOWER switch should be on during electric heat operation. Electric heat is unavailable in flight

The electric heat system can operate when the CABIN TEMP MODE switch is set to either AUTO or MAN HEAT, but control of electric heat is separate from the automatic or manual temperature controls (i.e. adjustment of the bypass valves) for bleed air heat. A control switch placarded ELEC HEAT energizes the heater power relays for the forward and aft electric heat elements. When the temperature in the main heating duct is less than the cutout point, the overtemperature (power) thermistors are closed, and the heater power relays allow power to the electric heat elements. The aft blower should be energized when the electric heat is turned on for maximum heating capacity. With ELECTRIC HEAT selected and the aft blower turned off, only the forward heat element will be energized. With the aft blower turned on also, the aft heater control relays energize, thereby allowing power to the aft heat element. Since both heat elements (forward and aft) are identical in capacity, output is double in this configuration. The same blowers that distribute cool air also distribute heat produced by the electric elements.

**Figure 11-24** DUCT OVERTEMP Caution Light

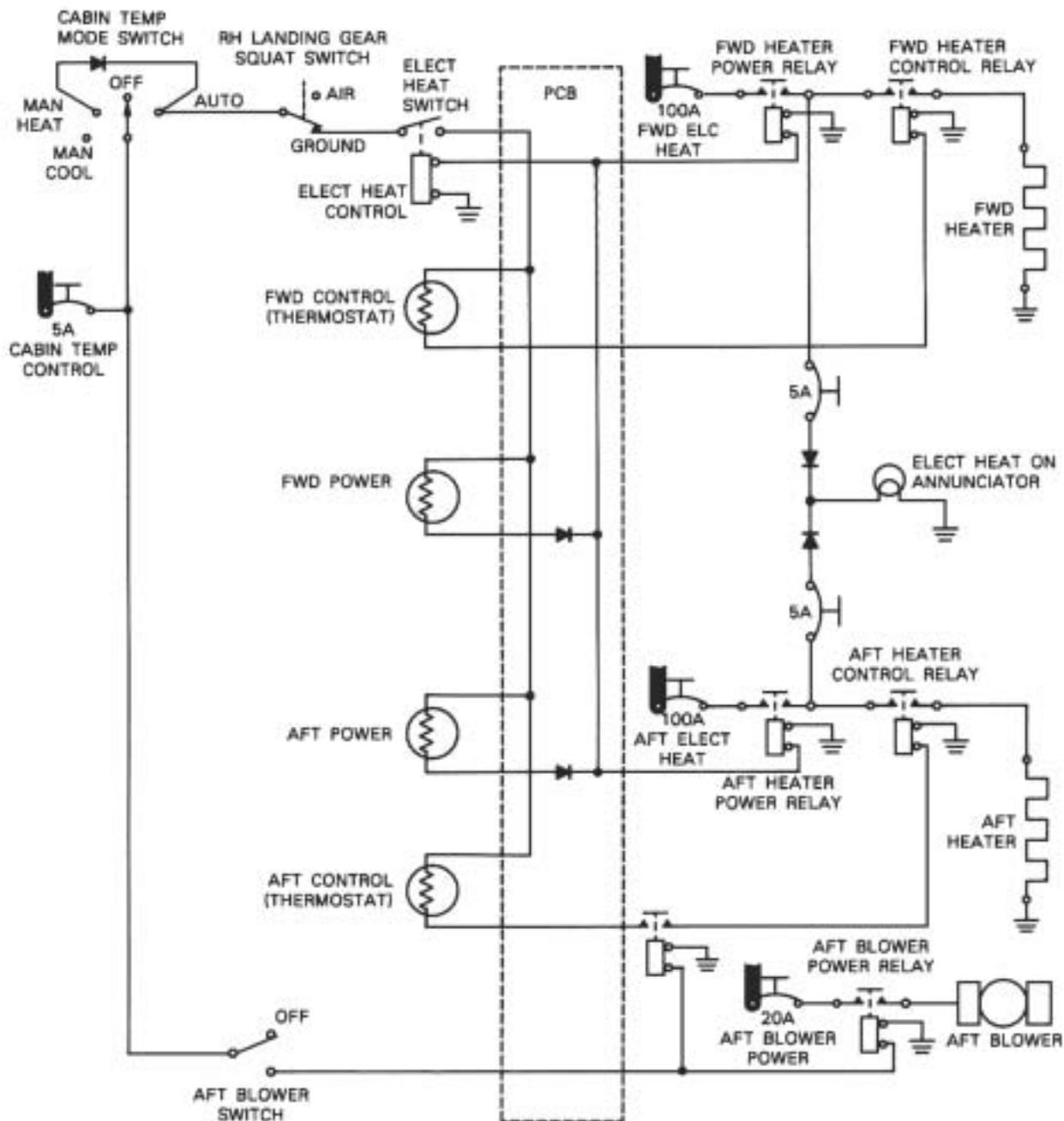
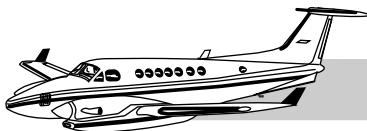


Figure 11-25 Simplified Electric Heat Schematic

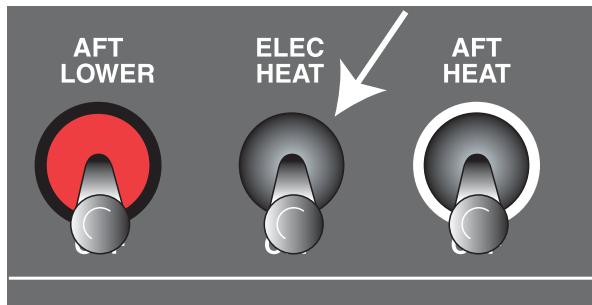


Figure 11-26 ELEC HEAT Switch

NOTE

The electric heat elements will use approximately 200 amps when both heat elements are operating.

The blowers should be operated for fifteen seconds after turning off the ELEC HEAT switch.

On some King Air 300s only, there are temperature control thermistors used to provide thermostatic control over each of the heat elements. They maintain system temperature within normal limits by cycling their respective elements on and off as required without interrupting power to the other element or the ELEC HEAT ON advisory annunciator light.

On some King Air 300s and 350s, there are over-temperature (power) thermistors, located in the duct about 1 – 2 inches upstream of the heat elements which provide a safety cutoff of power to the electric heat elements. When an overheat condition exists (such as one caused by a blower failure), a thermistor in the overheated duct opens, deenergizing the electric heat power relay circuit. The forward and aft heater power relays and the solenoid-held control switch are then deenergized, and power is removed from the heat elements. The ELEC HEAT switch solenoid releases and the switch springs OFF.

When the airplane is in flight (weight is removed from the right main squat switch), control power

is removed from the electric heat circuit, making electric heat unavailable.

COOLING SYSTEM

When outside ambient temperature is below approximately 50° F, the environmental bleed air system normally can maintain the cabin at comfortable temperatures. However, above this temperature a supplementary cooling system may be required. Supplementary cabin cooling is provided by a freon air conditioning system consisting of- a belt-driven, engine-driven compressor, installed on the right engine accessory pad; refrigerant plumbing; an N₁ speed switch; high- and low-pressure protection switches; a condenser coil; a condenser blower; forward and aft evaporators (aft optional); forward and aft evaporator blowers; a receiver/dryer; an expansion valve and hot gas bypass valve; and a cabin heat control valve. Plumbing from the compressor is routed through the right wing inboard leading edge to the fuselage, then forward (Figure 11-27) to the condenser coil, receiver/dryer, expansion valve, and evaporator, which are all located in the nose of the airplane.

The forward vent blower recirculates cabin air through the forward evaporator and mixing plenum, and into the headliner ducts. When the cooling mode is operating, refrigerant will be circulating through the evaporator and the recycled cabin air leaving it will be cooled. A majority of it enters the headliner duct. This air is discharged through eyeball outlet nozzles in the cockpit and cabin. Each nozzle is movable so the airstream can be directed as desired. When the nozzle is twisted, a damper opens or closes to regulate airflow volume.

Some cool air will enter the floor outlet duct, but in order to provide cabin pressurization, warm environmental bleed air will also enter the floor outlet duct any time either bleed air valve is open. Therefore, pressurized air discharged from the floor registers will always be warmer than that discharged at the overhead outlets, no matter what temperature mode is in use.

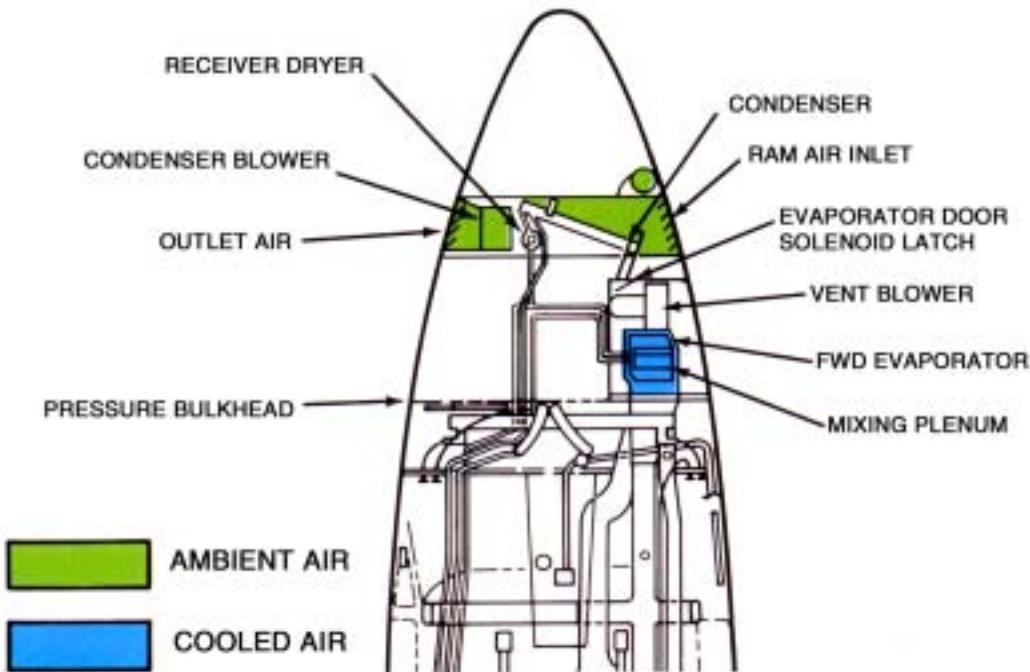


Figure 11-27 Cooling System Components in Nose

An aft blower (Figure 11-28) and optional aft evaporator are installed in the fuselage center aisle equipment bay behind the rear spar. Refrigerant will flow through the aft evaporator any time it flows through the forward evaporator, however, it provides additional cooling only when the aft blower is operating. The aft blower recirculates cabin air through the aft evaporator and delivers it to the aft overhead and floor outlets, including the baggage compartment.

A condenser blower in the nose section draws ambient air through the condenser when the air conditioner is operating on the ground. This blower shuts off when the nose landing gear is up. Since the aircraft is then in flight, the condenser is cooled by the flow of ram air that enters through the inlet in the nose.

The receiver/dryer and sight gage (Figure 11-29) are located in the upper portion of the condenser compartment. They can be viewed by removing the upper compartment access panel, located on top of the nose section just left of the centerline. The forward evaporator utilizes a solenoid-oper-

ated, hot gas control valve to prevent icing. A 33°F thermal switch on the forward evaporator controls the valve. The over- and underpressure limit switches and N₁ engine speed switch are provided to prevent compressor operation outside of established limitation parameters. The N₁ speed switch will prevent the flow of electric current to the compressor clutch when the engine speed is below approximately 62% N₁. When the N₁ speed switch is below 62% and there is a demand for air conditioning, the green AIR COND N₁ LOW advisory annunciator (Figure 11-30) will illuminate.

The following selections provide maximum air conditioner output and minimum bleed air input to the cabin consistent with pressurization requirements. First, select the CABIN TEMP MODE switch to AUTO and turn the CABIN TEMP rheostat full counterclockwise. Then select VENT BLOWER to HIGH and AFT BLOWER on. If the aircraft is on the ground or

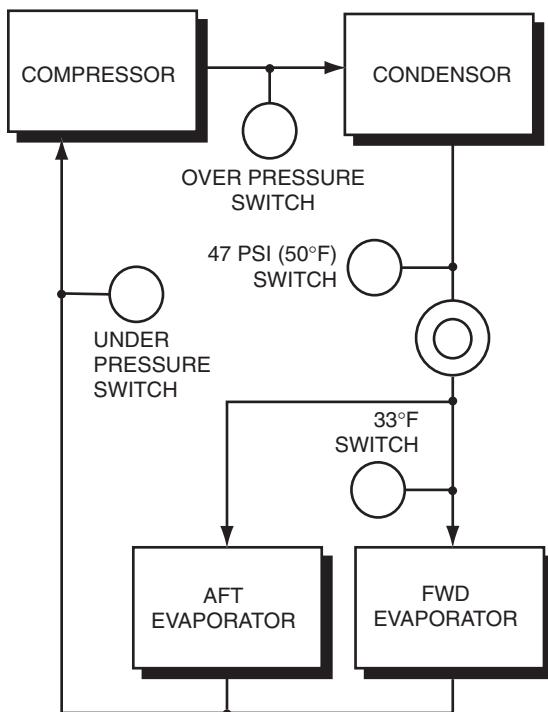


Figure 11-28 Simplified Air Conditioning Diagram

unpressurized flight is desired, the BLEED AIR VALVE switches should be selected to ENVIR OFF. When pressurized flight is desired, the BLEED AIR VALVE switches must be ON but select the ENVIR BLEED AIR switch to LOW.

ABNORMAL INDICATIONS

If a BLEED AIR FAIL warning annunciation illuminates (Figure 11-31), it could be caused by a bleed air leak in either environmental or pneumatic air. Turn the appropriate BLEED AIR VALVE switch to PNEU & ENVIR OFF. Refer to Chapter 9, “Pneumatics,” of this training manual and the *POH* for more details.

A DUCT OVERTEMP caution annunciation illumination is caused by environmental bleed-air temperature exceeding approximately 300° F. The duct overtemperature sensor is separate from the electric heat thermistors and is

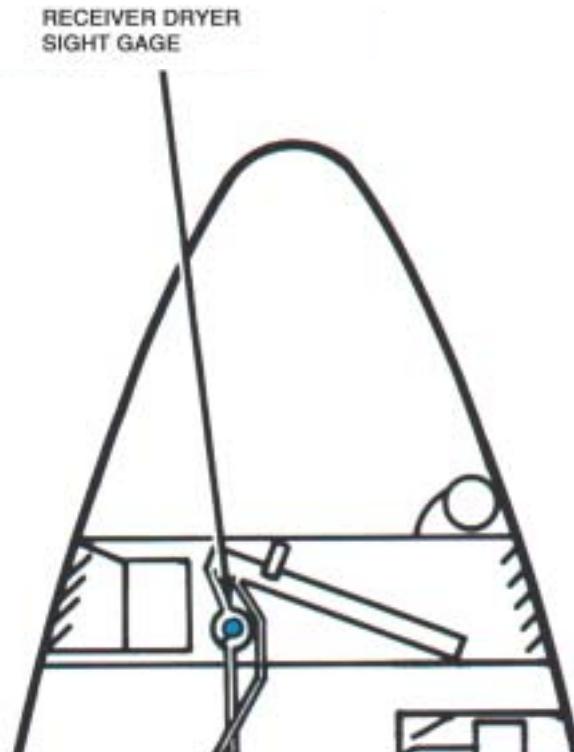


Figure 11-29 Receiver/Dryer Sight Gage

located approximately one foot downstream of the mixing plenum in the floor duct. Historically, this light indication has been caused by a restriction in the normal airflow within the ducts such as when the CABIN AIR control handle is pulled most of the way out. Combine this with an abrupt power increase (such as during takeoff), and the DUCT OVERTEMP annunciator illuminates as the restriction causes duct air temperature to climb above the sensor threshold. Refer to the *POH* for more details on the recommended procedure. When



Figure 11-30 AIR COND N₁ LOW Advisory Light (Green on the 300 or White on the 350)



Figure 11-31 Bleed Air Fail Warning Annunciators

an overheating occurs with electric heat on, the thermistor protection within the system should shut it off. In the event of a failure of the thermistors, the overheating will be detectable by smell, sight and/or a DUCT OVERTEMP annunciator. Turn the ELEC HEAT switch OFF. Complete the appropriate procedure in the *POH*.

If the ELEC HEAT switch will not remain in the up position, leave the switch OFF and check the CABIN TEMP circuit breaker in the ENVIRONMENTAL section of the copilot's circuit-breaker panel. With this breaker out, electric heat is unavailable due to loss of control power (see Figure 11-25). If the breaker is set, have the cause investigated.

NOTE

Electric heat is only available if external power is utilized or at least one engine generator is operating. Electric heat is unavailable in flight.

The AIR COND N₁ LOW annunciator informs the pilot that the environmental system is requesting air conditioning but the right engine is idling too slow. The air conditioner compressor cutout switch has disconnected the electric clutch for the a/c compressor, rendering air conditioning unavailable. Readjust the right condition lever to maintain 62 to 64% N₁ speed and the light should extinguish.

NOTE

The minimum engine idle speed and the air conditioner compressor cutout switch have the same values. Therefore, it may save wear and tear on the system if minimum engine idle speed

were adjusted by a mechanic to approximately 1% N₁ above the cutout switch threshold.

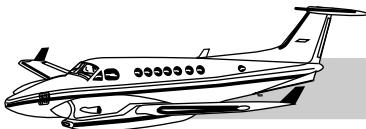
Constant cycling of this annunciator will cause pressure surges inside the a/c plumbing. This surging may cause one of the other system protection devices to actuate, thereby deactivating the system. When either the over- or underpressure switch senses a fault, it trips a resettable switch in the nose wheel well, which illuminates amber when tripped. The pilot should attempt only one reset of the switch in the nose wheel well. If the switch trips a second time, have an authorized service center investigate the cause.

The amber pushbutton switch may be reset by the following procedure:

1. The battery switch must be on and the generator bus ties closed. External power may be utilized but is not necessary. The engines should be shut down (not just feathered) due to the proximity of the spinning propeller to the location of the reset switch.
2. Air conditioning must be requested. Select MAN COOL on the CABIN TEMP MODE switch, then hold the INCR DECR switch to DECR position for one minute. An alternate method to request a/c would be to select AUTO on the CABIN TEMP MODE switch and to turn the CABIN TEMP - INCR rheostat fully counterclockwise.
3. Depress the amber switch in the nose wheel well. The light should extinguish.

CAUTION

Repeated attempts to reset the a/c system without determining the cause of the failure may cause serious damage to the system.



CHAPTER 12

PRESSURIZATION

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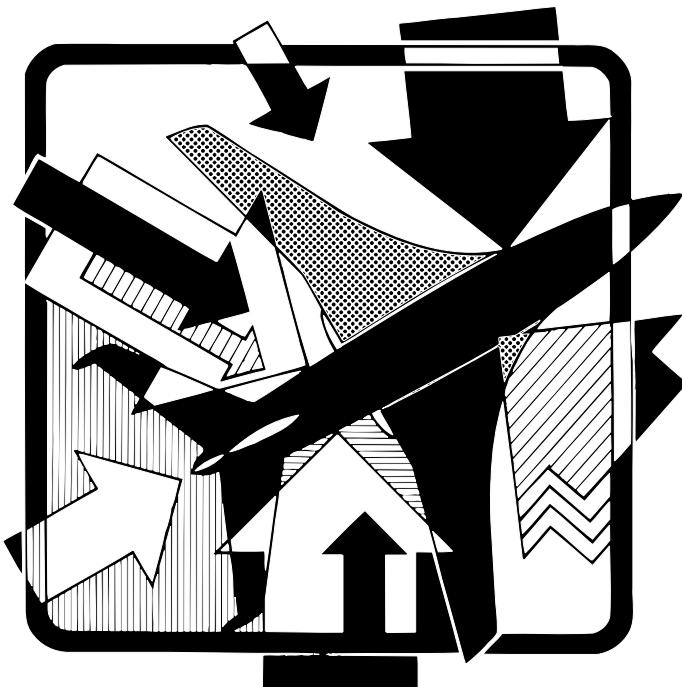
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CHAPTER 12

PRESSURIZATION



INTRODUCTION

Pressurization is desirable in an airplane because it allows the altitude of the cabin to be lower than the altitude of the airplane, thus decreasing or eliminating the need for supplementary oxygen. In this section the pilot learns how the system operates, is controlled, and how to handle malfunctions of the system.

DESCRIPTION

The Pressurization System section of the training manual presents a description of the pressurization system. The function of various major components, their physical location, and operation of the pressurization system controls are discussed. Where necessary, references are made to the environmental system as it affects pressurization. Common malfunctions and limited troubleshooting are discussed.

PRESSURIZATION SYSTEM

The pressurization system in the Super King Air 300 and 350 is designed to provide a cabin environment with sufficient oxygen for normal breathing, regardless of the airplane altitude, up to its design ceiling. As the airplane altitude increases, outside ambient air pressure decreases until, at approximately 12,500 feet, it cannot support normal respiration. The pressurization system maintains a proportionally lower altitude inside the cabin. The differential between the



cabin pressure and the outside ambient air pressure is measured in pounds per square inch of differential (psid).

The cabin, or “pressure vessel,” is that portion of the aircraft designed to withstand the pressure differential (Figure 12-1). In the Super King Air 300 and 350, the pressure vessel extends from the forward pressure bulkhead, between the cockpit and nose section, to the rear pressure bulkhead, just aft of the cabin baggage compartment, with the exterior skin making up the outer seal. Windows are round for maximum strength. All cables, wire bundles, and plumbing passing through the pressure vessel boundaries are sealed to reduce leaks.

As the Cabin Altitude For Various Airplane Altitudes Graph (Figure 12-2) shows, whenever cabin altitude and airplane altitude are equal, no pressure differential exists. Whenever cabin pressure is the greater of the two, the pressure differential is positive. If cabin pressure is less than that of outside ambient air, the pressure dif-

ferential is negative. Maximum differential is defined as a measure of the highest differential pressure the airplane structure can safely withstand for an extended period of time.

The Super King Air 300 and 350 pressure vessel will maintain and is designed to withstand a normal maximum positive differential pressure of up to 6.6 psid (6.5 ± 0.1 , -0.0). This provides for a cabin altitude of approximately 2,700 feet at an airplane altitude of 20,000 feet, an 8,700 foot cabin altitude at 31,000 feet, and a 10,200 foot cabin altitude at 35,000 feet. Since the airplane structure is not designed to withstand a negative differential, the minimum allowable differential pressure is 0.

The pressurization and heating/air conditioning systems (Figures 12-3 and 12-4) operate in conjunction with each other or as separate systems to maintain the desired cabin pressure altitude and cabin air temperature. Occupied compartments are pressurized, heated or cooled through a common ducting arrangement.



KING AIR 300



KING AIR 350

Figure 12-1 Cabin Pressure Vessel

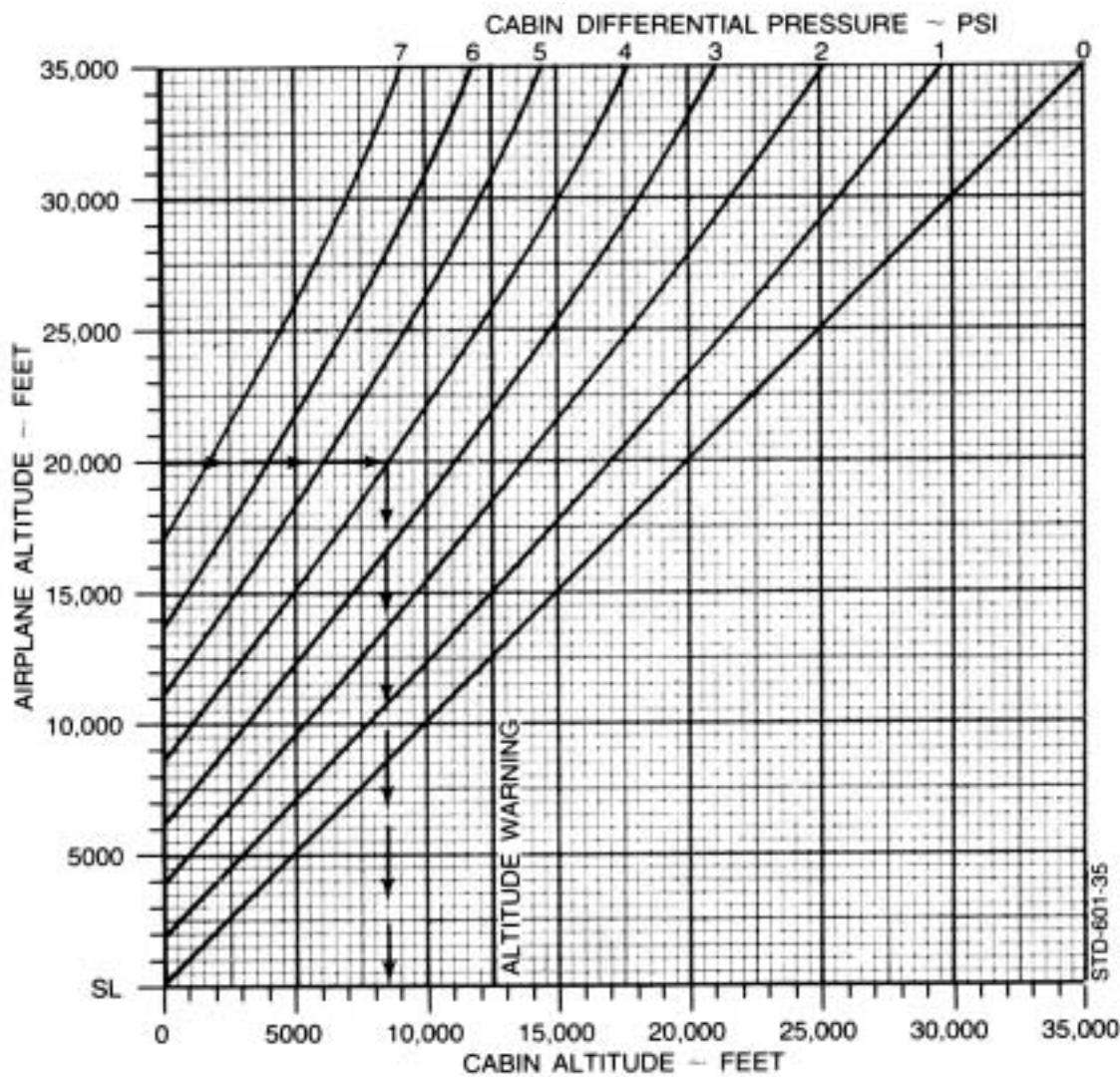


Figure 12-2 Cabin Altitude for Various Airplane Altitudes Graph

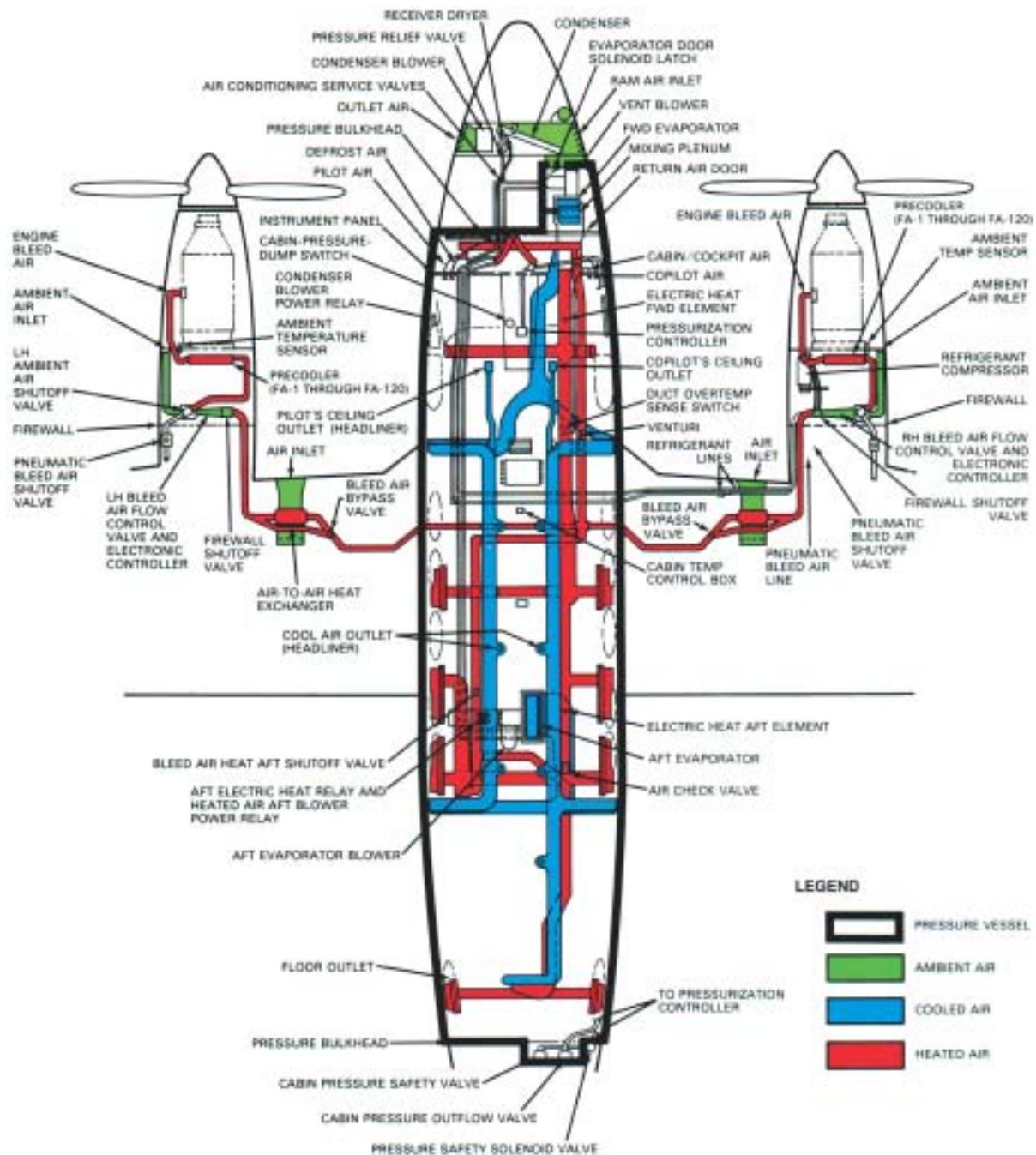


Figure 12-3 King Air 300 Pressurization and Environmental Distribution System

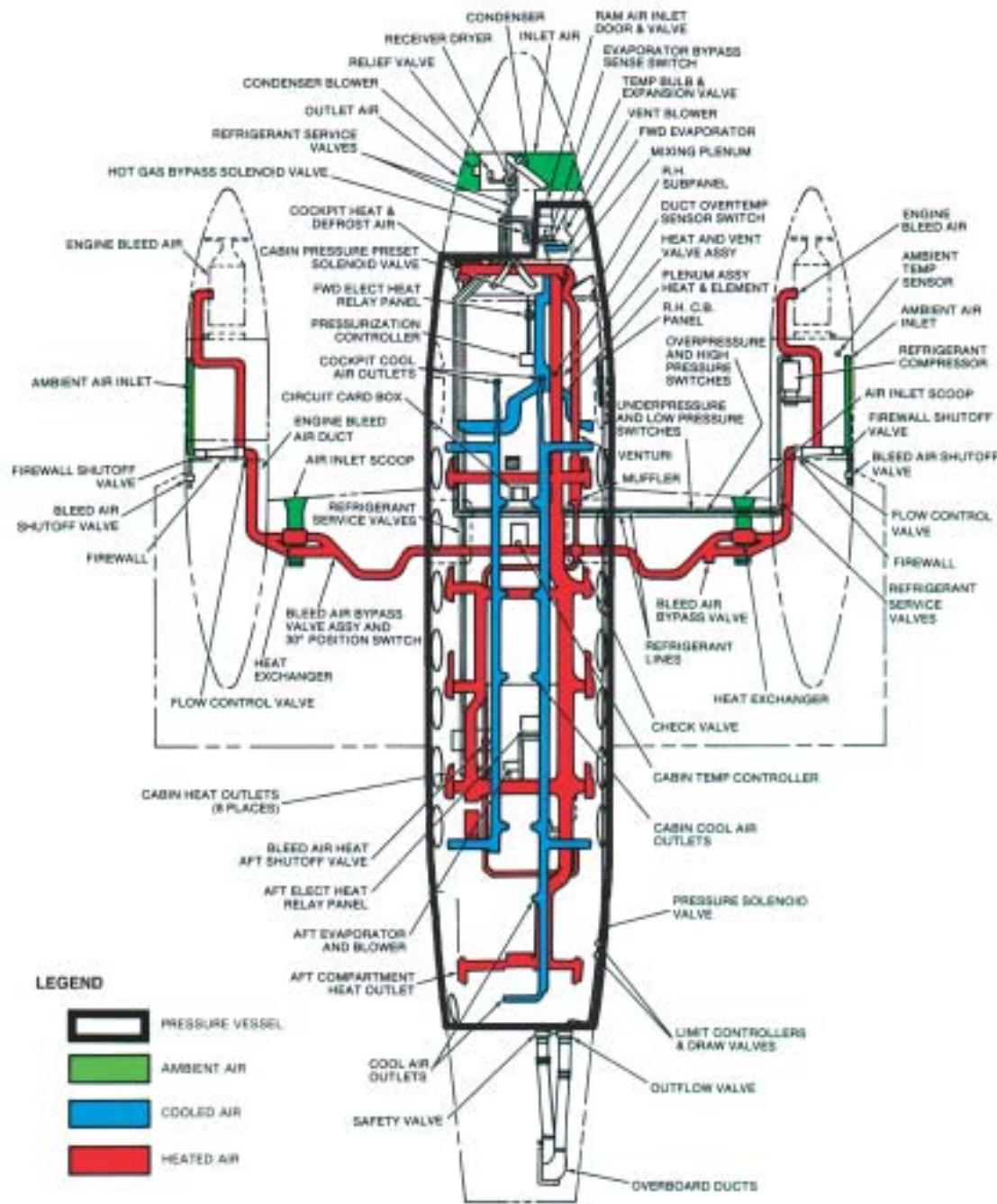


Figure 12-4 King Air 350 Pressurization and Environmental Distribution System



AIR DELIVERY SYSTEM

P₃ bleed air from the compressor section of each engine is the source of inflow air for pressurizing the cabin. A precooler heat exchanger on King Air 300 serials FA-1 through FA-120 (Figure 12-5), mounted on the backside of engine oil cooler, lowers the P₃ air temperature approximately 50°F before the air enters the flow control unit. There is no thermostatic control of temperature by the precooler. This precooler may be bypassed per an optional service bulletin from Beech King Air 300s after FA-121 and all King Air 350s do not have the precooler installed.

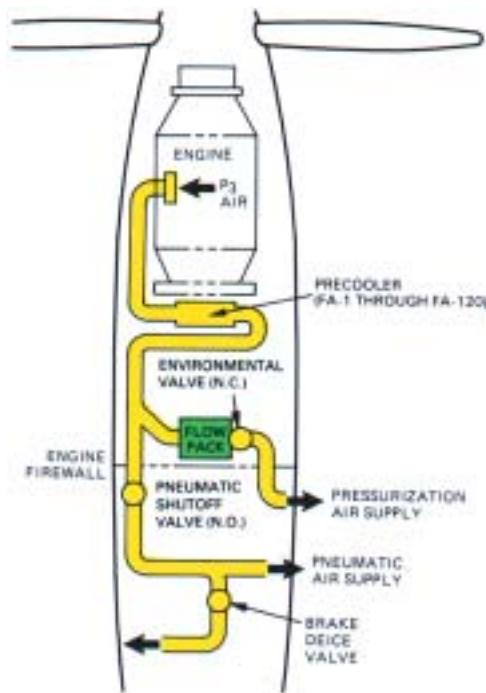


Figure 12-5 Bleed Air Inflow

Flow control units (Figure 12-6), located in each engine nacelle, control the bleed air/ambient air mixture which provides the proper air input for the environmental and pressurization functions.

The air mixture, called pressurization (or environmental) air, flows from the flow control unit through the environmental shutoff valve. This

valve is regulated by a normally-closed solenoid controlled by switches placarded BLEED AIR VALVES - LEFT (or) RIGHT - OPEN - ENVIR OFF - PNEU & ENVIR OFF in the ENVIRONMENTAL controls group (Figure 12-7) on the copilot's subpanel.

When a switch is positioned to either ENVIR OFF or PNEU & ENVIR OFF, the corresponding environmental shutoff valve is closed and no bleed air can pass through the flow control unit or enter the cabin. When a BLEED AIR VALVE switch is in the OPEN position, the corresponding solenoid valve is electrically held open, opening the environmental shutoff valve. The air mixture can then flow through the valve and into the cabin.

The volume of air flow is controlled by the switch placarded ENVIR BLEED AIR - NORMAL - LOW in the ENVIRONMENTAL controls group on the copilot's left subpanel. When this switch is in LOW position, pressurization air input to the cabin is approximately 50 percent of the input when the switch is in NORMAL. When ambient temperatures are above 10°C, the switch is required to be in LOW during takeoff and landing and is usually left in LOW during ground operation. The NORMAL position may be selected shortly after takeoff for maximum pressurization efficiency and cabin comfort at cruise altitudes. The switch should also be in the LOW position before landing.

Air flows out of the pressure vessel through the outflow valve and/or the safety valve (Figure 12-8 and 12-9), located on the aft pressure bulkhead. A silencer on each valve ensures quiet operation. Pressure within the cabin and the rate of cabin pressure changes are normally regulated by modulation of the outflow valve, which controls the rate at which air can escape from the pressure vessel. Additional cabin pressurization safety protection and cabin pressurization dump capability are provided by the safety valve.

Flow Control Unit

Electronic flow control units (Figure 12-6) control the mass flow of both ambient and bleed air

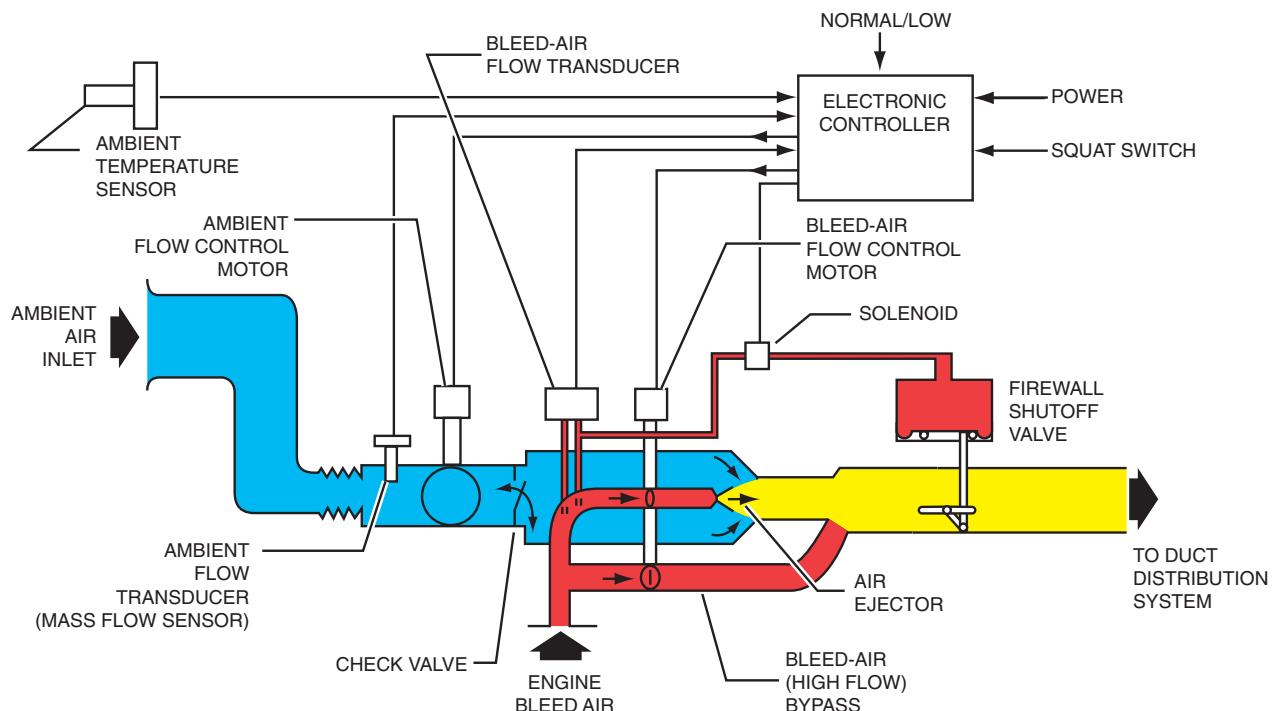


Figure 12-6 Flow Control Unit Schematic

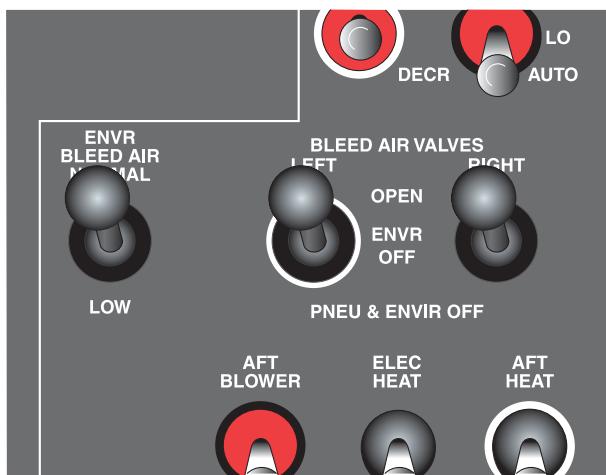


Figure 12-7 Bleed Air Valve Switch

into the cabin. Each unit consists of an ambient temperature sensor, an electronic controller, and an environmental air control valve assembly, interconnected by a wire harness.

The control valve assembly consists of a mass flow transducer, an ambient flow motor and modulating valve, a cheek valve that prevents the bleed air from escaping through the ambient air intake, a bleed air flow transducer, a bleed air flow motor and modulating valve (including bypass line), an air ejector, a flow control solenoid valve, and an environmental shutoff valve.

After engine start-up when the flow control unit is energized, the bleed air modulating valve closes. When it is fully closed, it actuates the bleed air shaft switch, signaling the electronic controller to open the solenoid valve. This enables P_3 bleed air to pressurize the environmental shutoff valve, causing it to open. The bleed air shaft continues to open until the desired bleed air flow rate to the cabin is reached. (The flow rate is sensed by the bleed air flow transducer and controlled by the electronic controller per the input of the ambient temperature sensor.)

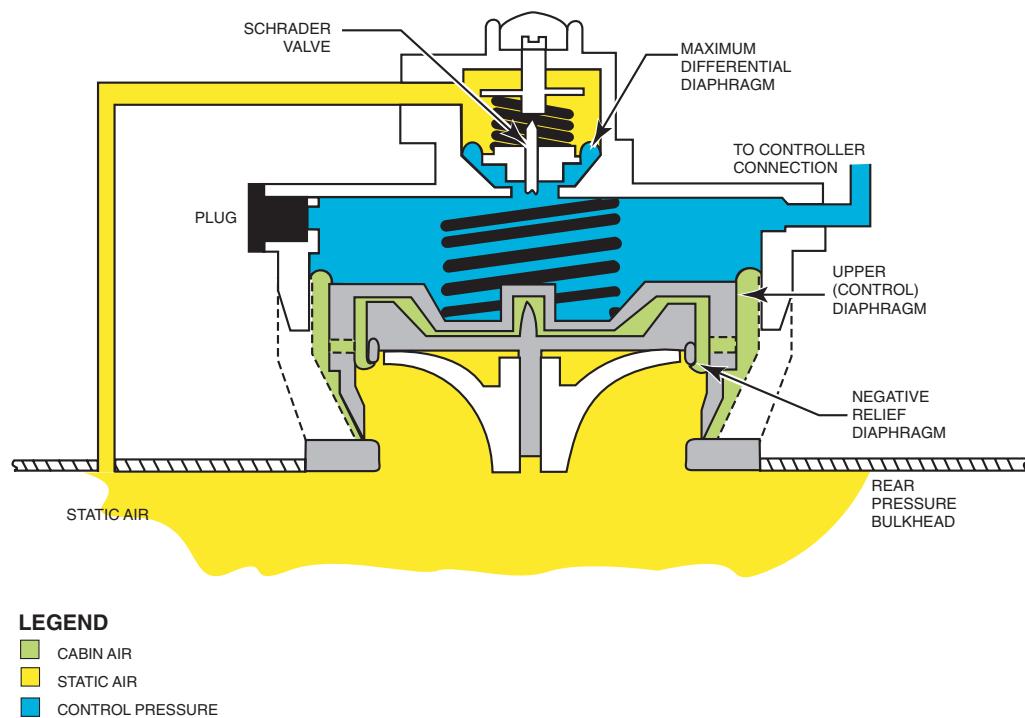


Figure 12-8 Cabin Air Outflow Valve Schematic

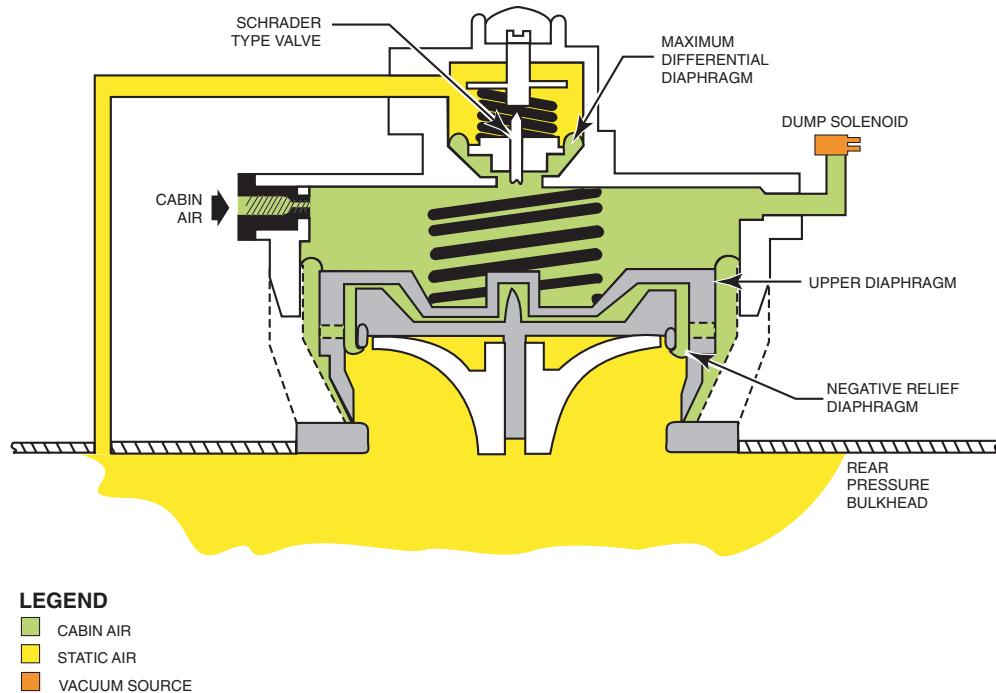


Figure 12-9 Cabin Air Safety Valve Schematic



During ground operation (as sensed by the left main gear safety switch), the ambient flow modulating valve is closed to prevent the addition of undesired hot or cold ambient air into the cabin environment. After takeoff, the left main gear safety switch signals the electronic controller, which opens its ambient flow modulating valve (Figure 12-12). Approximately 4 to 6 seconds later, the right controller opens its ambient air valve until the desired ambient flow rate is reached. By increasing the volume of airflow into the pressure vessel in stages, excessive pressure bumps during takeoff are avoided.

As the aircraft enters a cooler environment, ambient air flow is gradually reduced and bleed air flow gradually increased to maintain a constant inflow and to provide sufficient heat for the cabin. At approximately 50°F ambient temperature, ambient airflow is completely closed off and the bleed air valve bypass section is opened, as necessary, to allow more bleed air flow past the fixed flow passage of the air ejector.

A two-position ENVIR BLEED AIR switch adjusts the flow of P₃ bleed air through the flow control unit. With the switch in NORMAL, maximum flow of P₃ bleed air will be allowed through the flow control unit. With the switch in LOW, approximately one-half of the P₃ bleed air flow is eliminated by almost completely closing the bleed air bypass valve and slightly closing the normal bleed air valve. In addition, all signals to the bleed air flow control motor are biased by

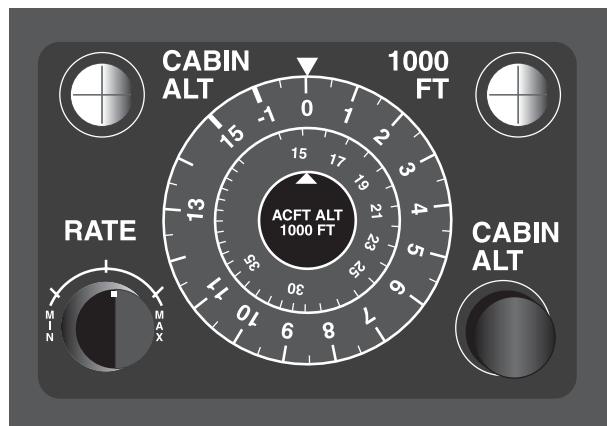


Figure 12-10 Pressurization Controller

approximately one-half to maintain the desired bleed air/ambient air mixture as requested by the sensors within the unit.

The flow control units regulate the rate of air flow to the pressure vessel. The bleed air portion is variable from approximately 5 to 14 pounds per minute (mass airflow, not volume) depending upon ambient temperature and the position of the ENVIR BLEED AIR switch. Each flow control unit provides one-half of the total airflow. On the ground, since ambient air is not available, cabin inflow is variable and limited by ambient temperature. In flight, ambient air provides the balance of the constant airflow volume of 10 to 14 pounds per minute. With the ENVIR BLEED AIR switch in LOW, however, the normal flow rate will be reduced to approximately one-half or 5 to 7 pounds per minute total with both flow control units operating.

Outflow Valve

A vacuum-operated outflow valve (Figure 12-8) is mounted adjacent to the safety valve on the aft pressure bulkhead. It is intended to serve three functions:

1. On the King Air 300 and 350, it serves to meter the outflow of cabin air in response to vacuum control forces from the controller.

On the King Air 350, the outflow valve will meter the outflow of cabin air up to approximately 13,500 feet cabin pressure altitude. This cabin altitude limit is accomplished by means of an altitude limit controller installed in the line connecting the outflow valve to the pressurization controller and another one installed in the line connecting the safety valve to the dump solenoid.

2. It provides maximum pressure differential relief. The outflow valve contains a pre-adjusted relief valve set to ensure that differential pressure does not exceed 6.6 psid, providing protection against over-pressurization.
3. It provides negative pressure differential relief. The outflow valve contains a negative differential pressure relief diaphragm.



This diaphragm opens the valve anytime the outside pressure becomes greater than the cabin pressure.

Metering of cabin air outflow, or outflow modulation, is controlled by the outflow valve. Controller pressure, which is metered suction from the vacuum system, is introduced into the center chamber (blue) of the outflow valve. This suction is regulated to provide the force against the internal spring that holds the valve closed. When this suction is increased (lower chamber pressure), the valve opens. Conversely, as suction is decreased, the outflow valve closes. In normal pressurized flight, the outflow valve is constantly being adjusted to maintain the desired cabin altitude as engine power and cabin/ambient pressure differential changes occur. Once cabin pressure stabilizes at the desired altitude, the center chamber pressure (blue) inside the outflow valve stabilizes at a slightly lower pressure (approximately 0.05 psi) than actual cabin pressure (orange). This differential holds the outflow valve open at the precise setting appropriate to the air inflow to the cabin.

The cabin is protected from exceeding maximum differential (6.6 psid) by the outflow valve. Ambient (static) pressure (yellow) is compared to controller chamber pressure (blue) in Figure 12-8. When maximum differential is reached, pressure is applied to the Schrader relief valve against spring tension in the top of the outflow valve case. If this differential increases, the Schrader valve opens and releases controller pressure air to the static line, thereby reducing pressure in the center chamber (blue) and modulating the outflow valve open. As the outflow valve opens, more cabin air is allowed to escape, raising the cabin altitude and reducing the differential between ambient (static) altitude and cabin altitude.

The outflow valve also protects the cabin against possible negative differentials. As cabin pressure approaches ambient (static) pressure, pressure in the negative relief diaphragm decreases relative to static pressure immediately to the right (outside) of the valve. In a negative differential situation, static pressure exceeds cabin pressure,

forcing the valve against the internal spring and opening the valve. Static air is then able to backflow into the cabin thereby equalizing the pressure on both sides of the valve.

Safety Valve

A vacuum-operated safety valve (Figure 12-9) is mounted adjacent to the outflow valve on the aft pressure bulkhead. It is intended to serve three functions:

1. It serves as the dump valve.

On the King Air 300, the safety valve opens completely to relieve all pressure differential whenever the CABIN PRESS switch is positioned in DUMP, or when the switch is in PRESS and there is weight on the left landing gear safety switch.

On the King Air 350, the safety valve opens to relieve pressure differential up to approximately 13,500 feet cabin pressure altitude whenever the CABIN PRESS switch is positioned in DUMP, or when the switch is in PRESS and there is weight on the left landing gear safety switch. This cabin altitude limit is accomplished by means of an altitude limit controller installed in the line connecting the safety valve to the dump solenoid and another one installed in the line connecting the outflow valve to the pressurization controller.

2. It provides maximum pressure differential relief. The safety valve contains a preadjusted relief valve set to ensure that differential pressure does not exceed 6.6 psid, providing protection against over-pressurization.
3. It provides negative pressure differential relief. The safety valve contains a negative differential pressure relief diaphragm. This diaphragm opens the valve anytime the outside pressure becomes greater than the cabin pressure.



When the safety valve is used as a dump valve, vacuum evacuates the main chamber (orange) through the dump solenoid. The resulting pressure drop causes the valve to open (much like what occurs when low controller pressure opens the outflow valve), allowing cabin air to escape, and raising cabin altitude. This is known as “dumping” cabin pressure. Dumping cabin pressure will cause the cabin to depressurize completely on the 300 or up to approximately 13,500 feet on the 350.

Negative differential protection in both the outflow and safety valves ensures that the cabin differential remains at zero (unpressurized) and not less than outside atmospheric pressure.

The maximum pressure differential function of the safety valve works identically to the outflow valve; however, due to slightly less controller pressure (blue) than cabin pressure (orange) when cabin altitude is stable, the safety valve usually senses maximum differential slightly before the outflow valve does. Once one of the valves senses maximum differential, it is sufficient to protect the cabin. The second valve may never attain its maximum differential point and will continue its normal function.

Cabin Controller

The cabin controller modulates the outflow valve to regulate cabin pressure as requested by the pilot. It compares current cabin pressure to desired pressure and utilizes suction from the vacuum system to provide the required dynamics. As vacuum is applied to the controller, the balancing forces move toward equalization. Cabin altitude (pressure) stabilizes at the desired altitude when the forces are in balance.

Two controls on the face of the controller allow pilots to set the system. The CABIN ALT knob controls the altitude setting in the controller “window,” and the RATE knob controls the speed at which cabin altitude equalizes the setting in the window.

Inside the controller (Figure 12-10) an aneroid bellows provides a reference point for all control-

ler action. When the CABIN ALT knob is turned, it moves the bellows and a calibration spring connected to the rate diaphragm separating the two chambers in the controller. The position of the diaphragm regulates the amount of suction applied to the upper chamber through a metering valve. The upper chamber is also vented to the cabin through an orifice and a filter.

Cabin air provides input, telegraphing when cabin altitude equals selected altitude. When upper and lower chamber pressures are equal and are approximately 0.05 psi less than cabin pressure, the metering valve holds steady suction to the controller to stabilize controller and cabin pressure. This slightly lower controller pressure is required to hold the outflow valve open (against its internal spring tension) to maintain a dynamic balance between air inflow and cabin outflow.

When the two chamber pressures are not equal, vacuum provides the dynamics to adjust them to higher or lower pressures, as required, to comply with desired cabin pressure altitude. The rate at which the chambers equalize their pressures is regulated by the RATE knob. Turning this knob adjusts a needle valve which changes the size of the opening in the line between the two chambers.

A request for the cabin altitude to climb (move toward a lower pressure) results in a temporary reduction in pressure in the upper chamber of the controller, commanding the outflow valve to open. Conversely, a request for the cabin altitude to descend (move toward a higher pressure) results in a temporary increase in controller upper chamber pressure, causing the outflow valve to close.

The preset solenoid (Figure 12-11) allows the pilot to adjust the cabin controller prior to take-off by cutting off the vacuum to the controller. Once airborne, the solenoid opens to admit vacuum and the controller begins to readjust cabin altitude as requested by the CABIN ALT set knob. With the CABIN PRESS switch in PRESS, the preset solenoid is energized (closed) on the ground and de-energized (open) in normal flight. Refer to the Pressurization Control segment in this chapter for more details.

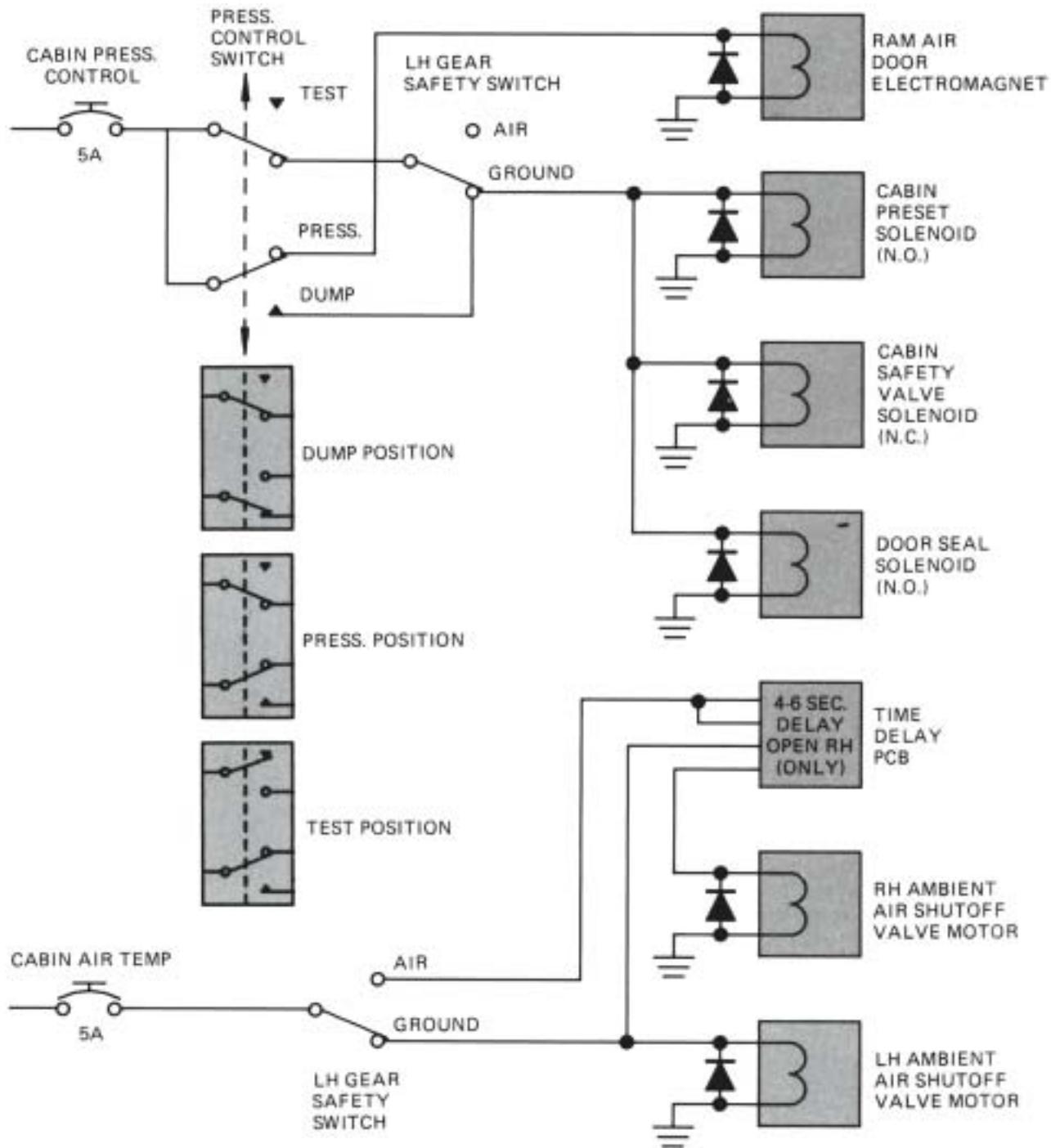


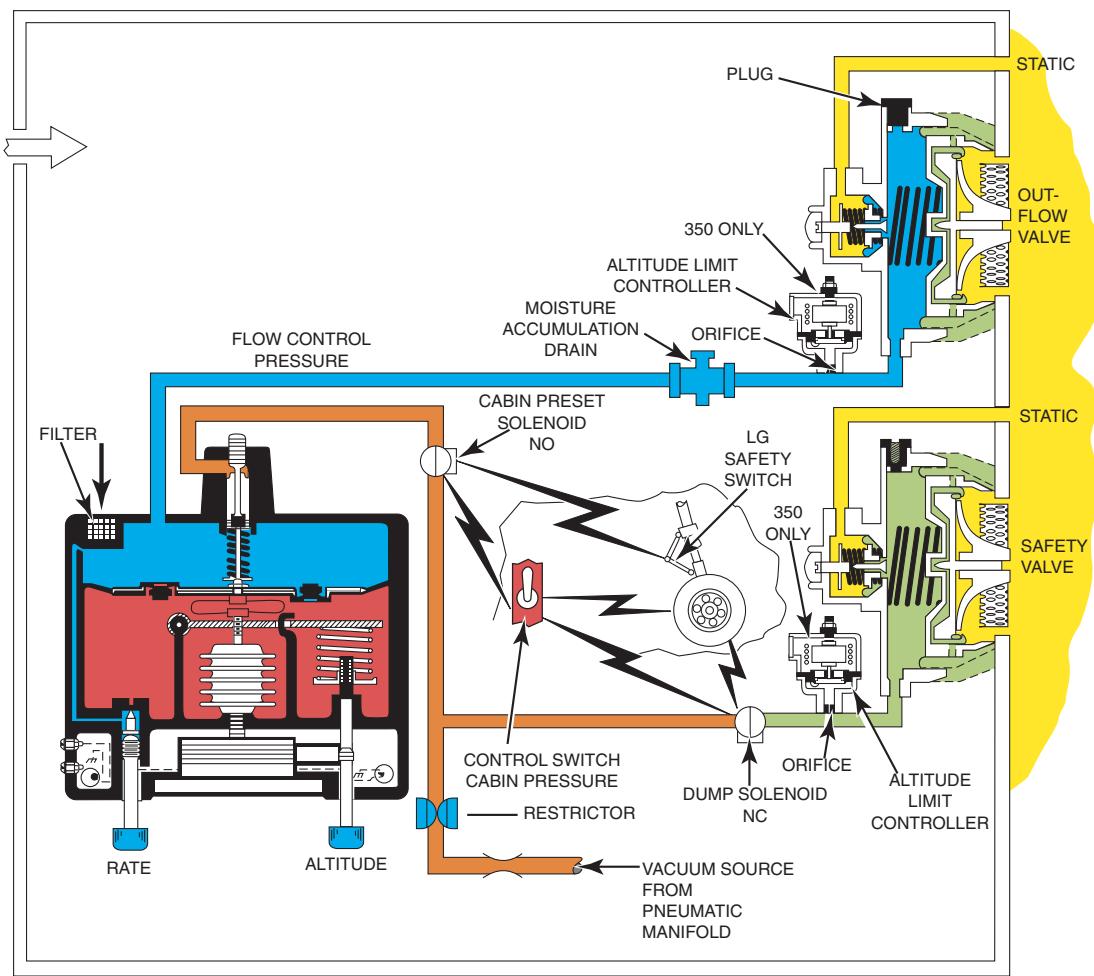
Figure 12-11 Pressurization Control Schematic (Electrical)



PRESSURIZATION CONTROL

An adjustable cabin pressurization controller (Figure 12-12), mounted in the cockpit pedestal commands modulation of the outflow valve. A

dual-scale indicator dial is mounted in the center of the pressurization controller. The outer scale (CABIN ALT) indicates the cabin pressure altitude which the pressurization controller will maintain. The inner scale (ACFT ALT) indicates



LEGEND

- [Green square] CABIN AIR
- [Orange square] VACUUM SOURCE
- [Yellow square] STATIC AIR
- [Blue square] CONTROL PRESSURE
- [Red square] INTERNAL PRESSURE

Figure 12-12 Pressurization System Schematic



the maximum ambient pressure altitude at which the airplane can fly without causing the cabin pressure altitude to climb above the value selected on the outer scale (CABIN ALT) of the dial.

The cabin pressure altitude is selected by turning the cabin altitude selector knob, placarded CABIN ALT, until the desired setting on the CABIN ALT dial is aligned with the index mark at the top of the case. The cabin altitude selected may be anywhere from minus 1,000 to 15,000 feet pressure altitude.

Both scales rotate together when the cabin altitude selector knob is turned. They are in a fixed relationship of 6.5 psid. As long as the aircraft altitude remains below the altitude specified on the ACFT ALT scale (indicated by the arrow below the scale), the cabin will seek the requested altitude. The airplane altitude indicated is the altitude at which a differential of 6.5 psi occurs. If the aircraft climbs above the altitude marked by ACFT ALT, the cabin will be at maximum permissible differential and the cabin will climb at a rate proportional to the aircraft's climb rate to maintain maximum differential.

The rate control selector knob is placarded RATE - MIN - MAX. The rate at which the cabin pressure altitude changes from the current value to the selected value is controlled by the rate control selector knob. The rate of change selected may be from approximately 150 to approximately 2,000 feet per minute.

There are two gages in the cockpit that indicate the status of the cabin pressurization system. The cabin altimeter (Figure 12-13) is located forward of the condition levers. It indicates actual cabin pressure altitude (outer scale) and cabin differential (inner scale). Immediately to the left of the cabin altimeter is the cabin vertical speed (CABIN CLIMB/DESCENT) indicator (Figure 12-14), which continuously indicates the rate at which the cabin pressure altitude is changing.

The cabin pressure switch (Figure 12-15), located forward of the pressurization controller on the pedestal, is placarded CABIN PRESS - DUMP - PRESS - TEST. The switch is normally in PRESS position. When the switch is in PRESS and the aircraft is on the ground, the outflow valve is closed



Figure 12-13 Cabin Altimeter

and the safety valve is open allowing all input air to escape and keeping the cabin depressurized. When the aircraft takes off, the safety valve closes and the outflow valve modulates open, as directed by the cabin pressurization controller, to regulate cabin pressure. Therefore, when the CABIN PRESS switch is in PRESS and the pressurization system is operating normally, the cabin will be depressurized on the ground and will pressurize in flight as directed by the pressurization controller.



Figure 12-14 Cabin Climb Indicator

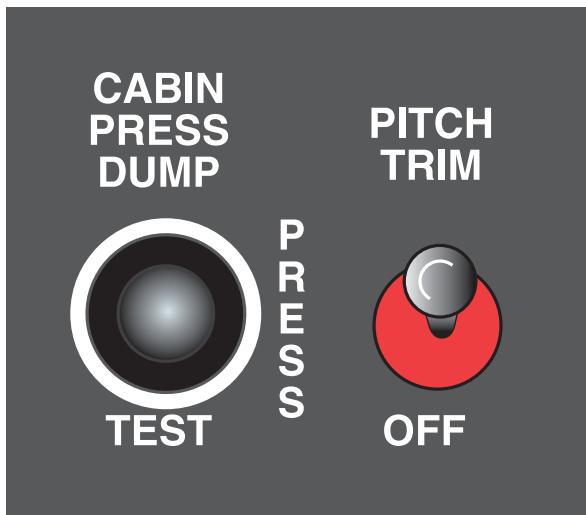


Figure 12-15 Cabin Pressure Switch

The other two positions of the switch will cause the cabin to either pressurize or depressurize regardless of aircraft position. With the switch in DUMP (lever-lock forward position), the aircraft will depressurize because the safety valve opens. Although the outflow valve simultaneously closes as a result, the aircraft depressurizes rapidly because the safety valve opens completely. With the switch held in TEST, the aircraft will pressurize because all of the outflow controls are positioned to the “normal” in-flight position. The switch is spring-loaded from TEST back to PRESS.

With the CABIN PRESS switch in PRESS, electrical power is available through the PRESS CONTROL circuit breaker (Figure 12-16) and the CABIN PRESS switch. The ram air (fresh air) door solenoid electromagnetically holds the door closed. In addition, when the airplane is on the ground, electric power is sent through the left main gear safety (squat) switch to energize the preset solenoid, the dump solenoid and the door seal solenoid. As a result, the preset solenoid closes, cutting off the vacuum source to the pressurization controller, allowing the outflow valve to spring closed. When electrical power opens the dump solenoid, vacuum is introduced to the safety valve

causing it to open. When the door seal solenoid is energized (closed), pneumatic air pressure is removed from the door seal, deflating it.

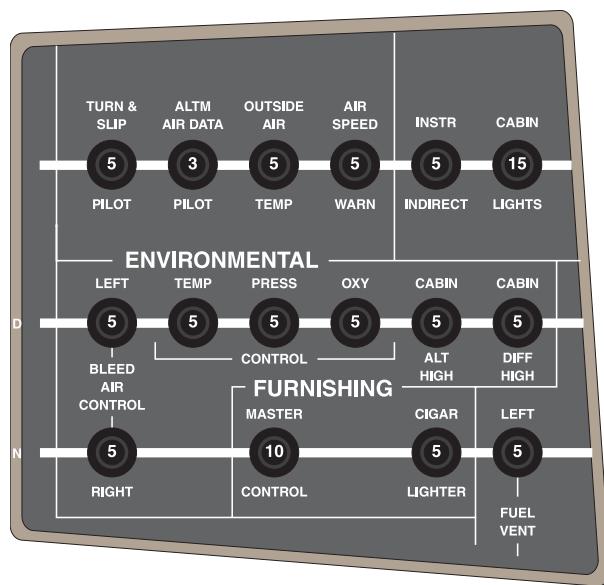


Figure 12-16 Environmental System Circuit Breakers

When the aircraft takes off, the left main squat switch opens the circuit to the preset, dump and door seal solenoids allowing them to spring to their normal positions. As a result, the outflow valve opens, the safety valve closes and the door seal inflates. Since cabin air outflow is now regulated the cabin may be pressurized.

Notice the electrical similarity between positioning the CABIN PRESS switch in DUMP and the normal situation (with the switch in PRESS) when the aircraft is on the ground. The differences are that in DUMP, the ram air door is not held closed electrically, and the other three solenoids are energized to the depressurized (on ground) position.

There is also an electrical similarity between the TEST position of the CABIN PRESS switch and the normal in-flight situation (with the switch in PRESS). The only difference is that the CABIN PRESS switch interrupts power upstream of the squat switch; therefore it “overrides” the squat switch.



and allows checking or testing of pressurization while on the ground. Also notice that there will be no effect on the system by placing the switch in TEST (from PRESS) in flight.

Preflight Check

During runup, the pressurization system may be checked using the cabin pressure switch. With both BLEED AIR VALVE switches OPEN, adjust the CABIN ALT knob so that the CABIN ALT dial indicates an altitude 1000 feet BELOW field pressure altitude. Rotate the rate control selector knob to place the index between the twelve and one o'clock positions. The ENVIR BLEED AIR switch can be either in NORMAL or LOW; however, the check may be completed more quickly with the switch in NORMAL and the condition levers HI IDLE. Hold the CABIN PRESS switch to TEST and check the cabin altitude indicator for a descent indication. This may take twenty to forty seconds. Release the pressurization switch to the PRESS position when pressurizing is confirmed. If any doubt exists that the system is operating normally, hold the switch in TEST until the cabin altitude stabilizes at the selected altitude.

Prior to takeoff, the CABIN ALT knob should normally be adjusted so that the ACFT ALT scale on the dial indicates an altitude approximately 500 - 1000 feet above the planned cruise pressure altitude, and the CABIN ALT scale indicates an altitude at least 500 feet above the takeoff field pressure altitude. The RATE knob should be adjusted as desired; setting the index mark between the twelve and one o'clock positions will provide the most comfortable cabin rate of climb. The cabin pressure switch should be checked to ensure that it is in the PRESS position, and the BLEED AIR VALVE switches should be OPEN for takeoff. The ENVIR BLEED AIR switch should be in LOW for takeoff when ambient temperature is above 10°C.

Climb and Cruise

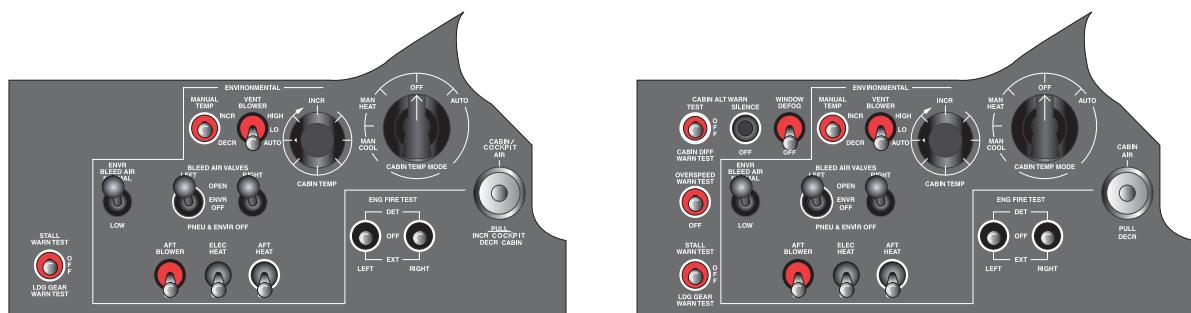
After takeoff, when time permits, the ENVIR BLEED AIR switch should be moved to NORMAL. A smaller pressure transient or "bump" will result when the pressure differential is small. As the airplane climbs, the cabin climbs at the selected rate of

change until it reaches the selected pressure altitude. The pressurization system then maintains this cabin altitude. If the airplane climbs to an altitude higher than the value indexed on the ACFT ALT scale on the face of the controller, the cabin-to-ambient pressure differential will reach the pressure relief setting of the outflow and safety valves. One of these valves (usually the safety valve) will then override the cabin pressurization controller in order to limit the cabin-to-ambient pressure differential to the normal maximum working pressure differential of 6.6 psid. During cruise operation, if the flight plan calls for an altitude increase, reselect the new flight altitude plus 500 - 1000 feet on the ACFT ALT dial.

Pressurization Annunciators

On the King Air 300, if the cabin pressure altitude should reach a value of 12,500 feet, a pressure-sensing switch will close, illuminating the red ALT WARN annunciator light to warn the pilot of operation requiring the use of oxygen. Refer to the Oxygen System section in this training manual for more details.

On the King Air 350, if the cabin pressure altitude should reach a value of 10,000 to 10,500 feet, a pressure-sensing switch will close, illuminating a white status annunciator labeled CABINJ ALTITUDE and also causing a tone to sound. The tone may be canceled. If the cabin pressure altitude should reach a value of 11,500 to 12,000 feet, a pressure-sensing switch will close, illuminating the red CABIN ALT HI annunciator light and also causing a tone to sound to warn the pilot of operation requiring the use of oxygen. This tone may also be canceled. If the cabin pressure differential exceeds 6.9 PSID, a red CABIN DIFF HI annunciator illuminates to warn the pilot that the maximum cabin pressure differential of 6.6 has been exceeded. These warning systems may be tested by using the CABIN ALT TEST and CABIN DIFF WARN TEST switch on the copilot's left subpanel (Figure 12-17). When the switch is pushed to the CABIN ALT TEST position, the CABIN ALTITUDE and CABIN ALT HI annunciators will illuminate. The warning tone will also sound. When the switch is pushed to the CABIN DIFF TEST position, the CABIN DIFF HI annunciator will illuminate.



300 COPILOT'S SUBPANEL

350 COPILOT'S SUBPANEL

Figure 12-17 Copilot's Left Subpanel

Descent

During enroute descents and in preparation for landing, the CABIN ALT selector should be set as appropriate for the lower altitude. For enroute descents, set the ACFT ALT to 500 - 1000 feet above the level-off flight altitude

unless it results in a CABIN ALT less than destination field pressure altitude. On normal descents, the CABIN ALT should be set to indicate a cabin altitude of approximately 500 feet above the landing field pressure altitude (Figure 12-18), and the rate control selector should be

CLOSEST ALTIMETER SETTING	ADD TO AIRPORT ELEVATION
28.00	+2,400
28.10	+2,300
28.20	+2,200
28.30	+2,100
28.40	+2,000
28.50	+1,900
28.60	+1,800
28.70	+1,700
28.80	+1,600
28.90	+1,500
29.00	+1,400
29.10	+1,300
29.20	+1,200
29.30	+1,100
29.40	+1,000
29.50	+ 900
29.60	+ 800
29.70	+ 700
29.80	+ 600
29.90	+ 500
30.00	+ 400
30.10	+ 300
30.20	+ 200
30.30	+ 100
30.40	0
30.50	- 100
30.60	- 200
30.70	- 300
30.80	- 400
30.90	- 500

Figure 12-18 Pressurization Controller Setting for Landing



adjusted as required to provide a comfortable rate of descent for the cabin. The airplane rate of descent should be controlled so that the airplane altitude does not catch up with the cabin pressure altitude until the cabin pressure altitude reaches the selected value and stabilizes.

As the airplane descends to and reaches the cabin pressure altitude, the negative-pressure relief function modulates the outflow and safety valves toward the full open position, thereby equalizing the difference between ambient and cabin pressures. When the airplane continues to descend below the preselected cabin pressure altitude, the cabin will be unpressurized and will follow the airplane rate of descent to touchdown.

PRESSURIZATION PROFILES

The following pressurization situations are described in order to illustrate operation of the system using normal flight situations. In each case, the given conditions will be outlined on the profile diagram.

Situation 1

Climb from sea level to FL310, then descend to a field pressure altitude of 1500 feet (Figure 12-19).

Conditions:

- Aircraft climbs at 2000'/min to FL200, then 1000'/min to FL310
- Cabin climbs at 500'/min
- Aircraft descends at 1500'/min
- Cabin descends at 500'/min

Controller setup before takeoff—Prior to takeoff, set the inner dial (ACFT ALT) on the pressurization controller to FL315 (500 feet above cruise altitude) which will provide a 9000-foot altitude on the outer dial (CABIN ALT). Set the rate knob between the twelve and one o'clock positions.

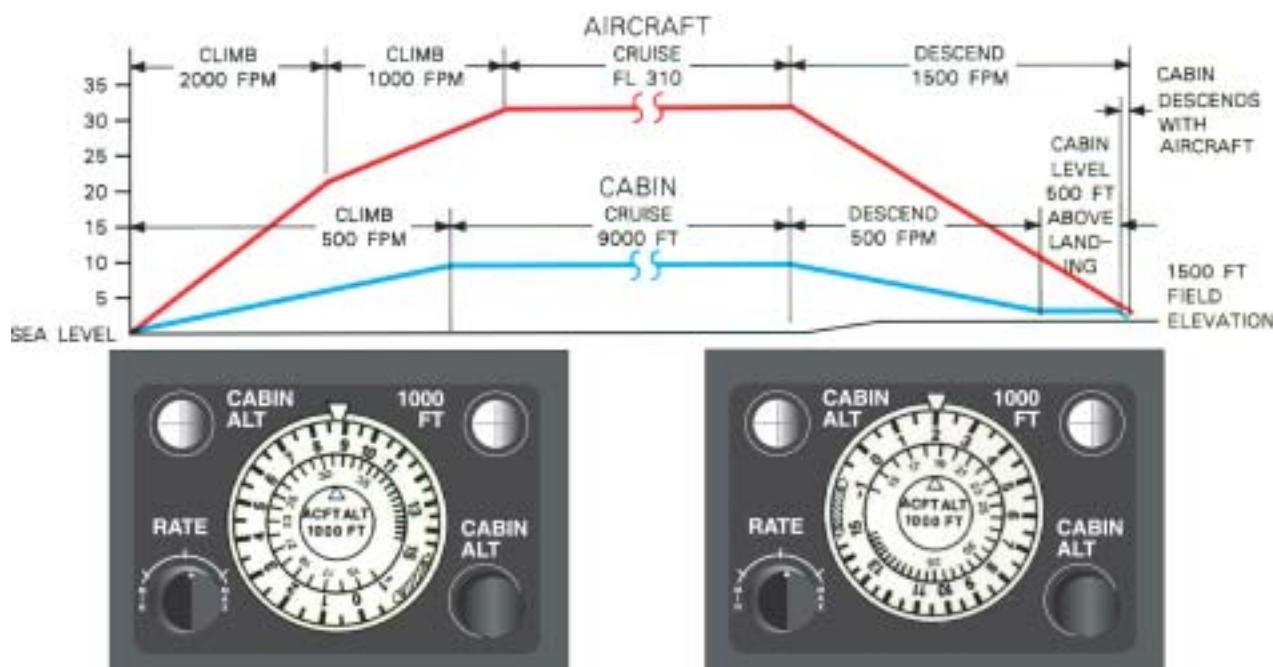


Figure 12-19 Situation 1



Operation—As the aircraft climbs to FL310, which will take approximately 21 minutes, the cabin climbs to 9000 feet in approximately 18 minutes, thus the cabin always stays “ahead” of the aircraft during the climb. (Staying ahead means that maximum differential is never achieved since the cabin climb rate is sufficient to make the final altitude on its own.)

Controller setup for descent—As the aircraft starts to descend, set the outer dial (CABIN ALT) to 2000 feet pressure altitude (500 feet above field pressure altitude). The rate knob should stay in the 12 to 1 o'clock position.

Operation—As the aircraft descends to the field, which will take approximately 19 minutes, the cabin descends to 2000 feet, which takes approximately 14 minutes. As the aircraft passes 2000 feet, the cabin descends unpressurized to 1500 feet with the aircraft.

Remarks—All settings are normal and the system reacts properly.

Situation 2

Climb from sea level to FL310, then descend to a field pressure altitude of 1500 feet (Figure 12-20).

Conditions:

- Aircraft climbs at 2000'/min to FL200, then 1000'/min to FL310
- Cabin climbs at 500'/min
- Aircraft descends at 1500'/min
- Cabin descends at 500'/min

Controller setup—Same as Situation 1 except set ACFT ALT dial to FL310 (same as cruise altitude) which will put cabin at max. differential when aircraft gets to final altitude. Set it as in Situation #1 for the descent.

Operation—Everything is normal until the cabin gets to max. differential which then causes pressure bumps in the cabin. The condition normalizes on descent.

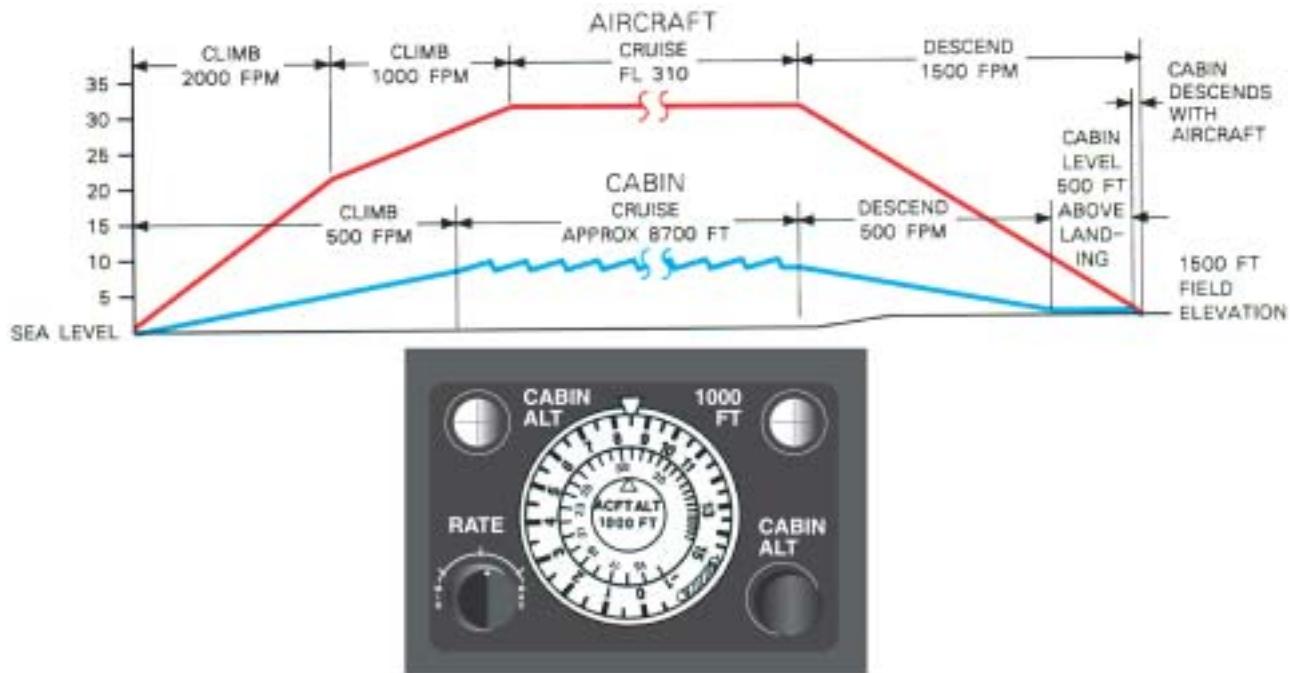


Figure 12-20 Situation 2



Remarks—By not setting the pressurization controller properly, cabin pressure bumps are likely with resultant discomfort for passengers. The ACFT ALT dial should be set to at least 500 feet above aircraft cruise altitude.

Situation 3

Climb from sea level to FL310, then descend to a field pressure altitude of 1500 feet (Figure 12-21).

Conditions:

- Aircraft climbs at 2000'/min to FL200, then 1000'/min to FL310
- Cabin climbs at 500'/min
- Aircraft descends at 1500'/min
- Cabin descends at 500'/min

Controller setup—Same situation as 1 except set CABIN ALT dial to landing field pressure altitude prior to takeoff. No readjustment required for descent.

Operation—Everything is normal until max. differential is reached as the aircraft passes approximately FL190. As the aircraft continues to climb, the cabin climbs at a rate proportional to but less than the aircraft's rate since it remains on max. differential. The cabin will be subject to pressure bumps while on max. differential. Once the aircraft starts to descend, the condition normalizes.

Remarks—By setting the controller for landing prior to takeoff, a similar problem to that in Situation 2 occurred with resultant passenger discomfort.

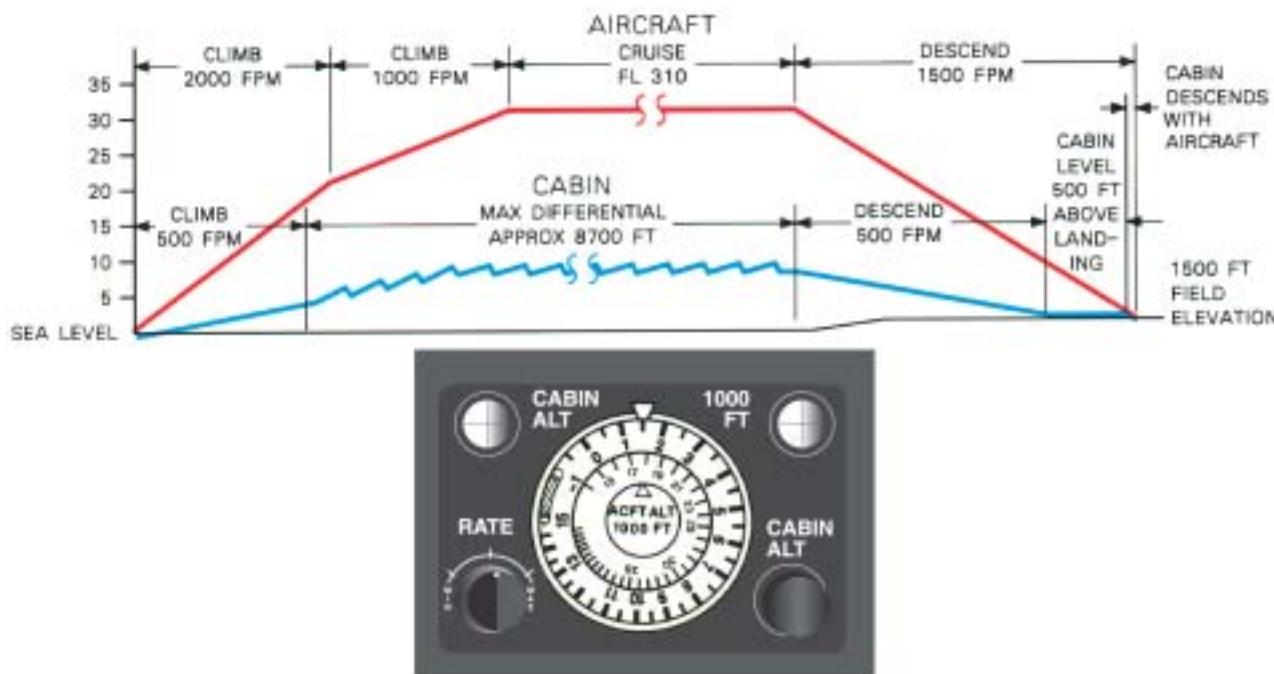


Figure 12-21 Situation 3



Situation 4

The aircraft was held to 5000 feet for 15 minutes during climb, then cleared to FL310. The aircraft was given a segmented descent to FL250 before given final descent for landing (Figure 12-22).

Conditions:

- Aircraft climbs at 2000'/min to 5000', levels off for 15 minutes then continues to FL310.
- Cabin climbs at 500'/min to 4500', levels off until reset, then climbs at 500'/min to 9000'.
- Aircraft descends at 1500'/min with a 10-min level off at FL250 before continuing down for landing.
- Cabin descends at 500'/min with a brief level off at 5900' until being reset for landing, then descends at 500'/min.

Controller setup before takeoff—Set CABIN ALT dial to 500 feet below (4500') the aircraft

intermediate level off altitude (5000 feet). This prevents the cabin altitude from catching up to the aircraft altitude during climb.

When finally cleared to FL310—Set the ACFT ALT dial to 500 feet above the assigned flight level as in Situation 1.

When cleared down to FL250—Set the ACFT ALT dial to 500 feet above the newly assigned flight level.

When cleared for final descent—Set the CABIN ALT dial to 500 feet above field pressure altitude.

Operation—As the aircraft climbs to 5000 feet, the cabin climbs to 4500 feet, thereby maintaining a slight pressurization differential. When the aircraft climbs to FL310, the controller is reset as in Situation 1 and the cabin will climb accordingly. When the aircraft starts to descend to FL250, the cabin starts down and levels at approximately 5900 feet. This allows the cabin pressurization to stay on the controller and not go

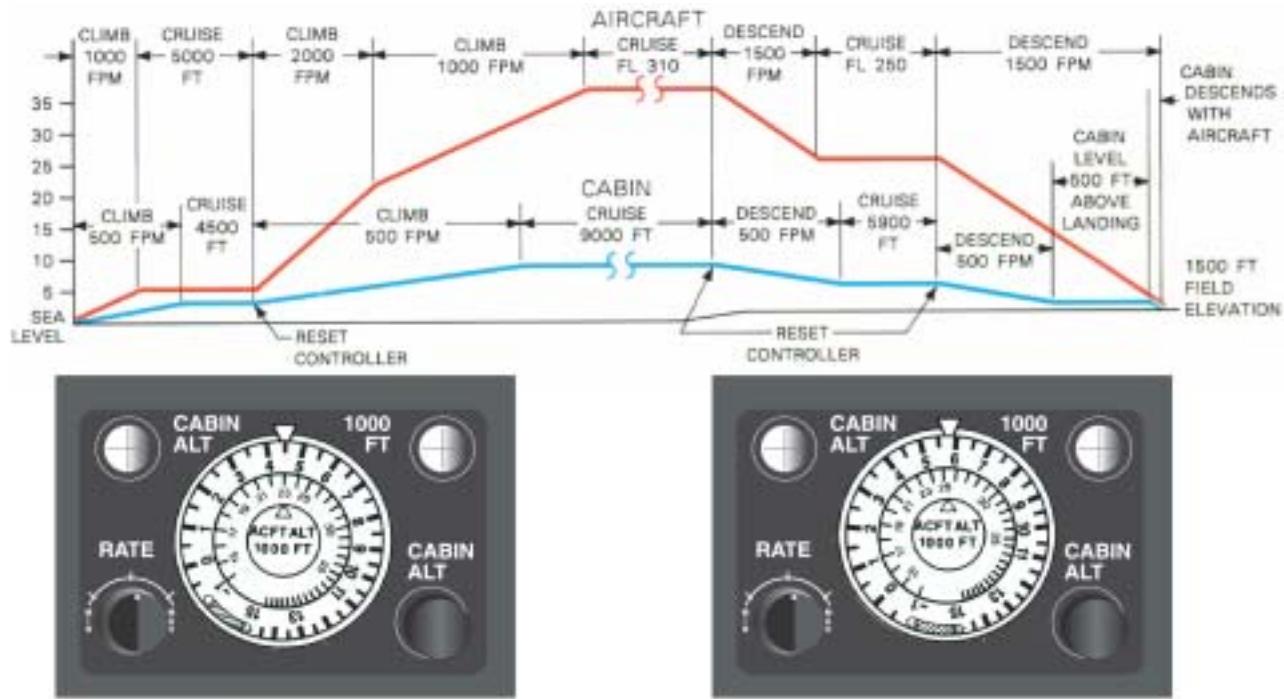


Figure 12-22 Situation 4



to max. differential which avoids bumps as in Situation 2. Once commencing final descent, everything will proceed as normal in Situation 1.

Remarks—An alternate method in this situation is to set the rate knob to minimum until cleared to the final altitude or for final descent, at which time the rate knob is reset for normal situations as in Situation 1. One problem with this alternate method is that it disrupts the normal balance between the outflow and safety valves on takeoff and will result in a pressure bump shortly after takeoff. In addition, it is possible, although unlikely, that the cabin pressurization could still catch up to the aircraft. Using either method in this situation, forgetting to reset the controller will result in problems similar to those discussed in Situations 2 and 3.

Situation 5

Depart from an airport at a higher elevation (in this case 6000 feet), fly at an altitude of 16,000

feet, and land at a sea level airport 20 minutes later (Figure 12-23).

Conditions:

- Aircraft climbs at 2000'/min to 16,000'
- Aircraft levels off for approx 5 min
- Cabin climbs and descends at 500'/min

Controller setting before takeoff—Set the CABIN ALT dial to 500 feet above takeoff field elevation (6000 feet).

Operation—Once the aircraft is stable in the climb and cabin altitude stabilizes at 6500 feet, reset controller for landing, usually within a few minutes after takeoff.

Controller setting in flight—Set the CABIN ALT dial to 500 feet above landing field pressure altitude.

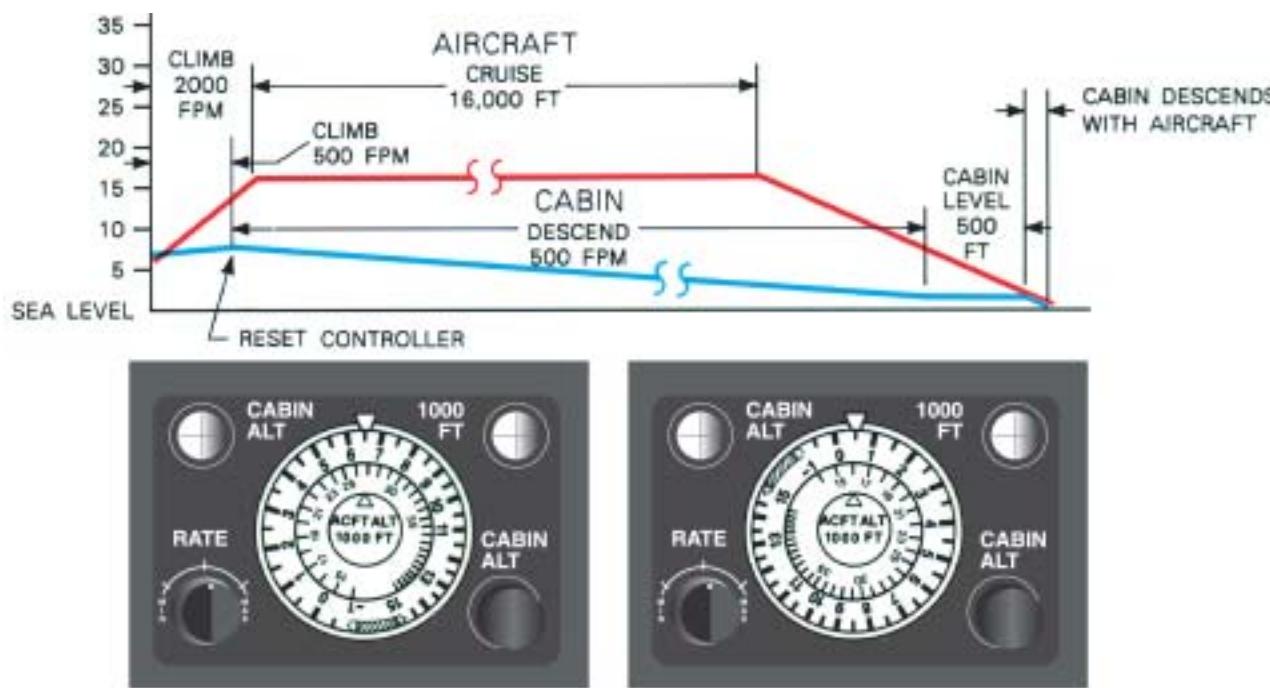


Figure 12-23 Situation 5



Operation—The aircraft will continue to climb to altitude while the cabin starts to descend to 500 feet pressure altitude. Aircraft levels at cruise, then descends for landing. By the time the aircraft is ready for landing, the cabin altitude is level at 500 feet (this assumes aircraft altitude vs. cabin altitude does not exceed 6.6 psid).

Remarks—It is important to set the cabin pressurization controller for a cabin altitude above takeoff field pressure altitude. If it is set lower, a pressure bump will be experienced shortly after liftoff since the balance between the outflow and safety valves will be disrupted. Once cabin pressurization is stabilized after takeoff, the controller may be reset for landing, provided cruise altitude does not exceed the altitude in the ACFT ALT window. If it does, normal controller procedures described earlier apply.

MALFUNCTIONS AND TROUBLESHOOTING

The pressurization system in the Super King Air 300 and 350 is derived from the proven and highly reliable system used in other King Airs. It is well engineered for safety, comfort, reliability and ease of operation. Pilot controls are simple and straightforward and workload is minimal. The pilot has sufficient controls readily available to either regain control or minimize the effect of most problems.

With loss of pressurization in flight, follow the procedures outlined in the Emergency Procedures section of the *POH*. Once the situation in the aircraft has stabilized, and if extreme care and good judgement are utilized, other corrective action may be taken using the techniques and procedures discussed in this section as a guide.

WARNING

For crew and passenger safety, pressurization troubleshooting should be accomplished below an aircraft altitude of 10,000 feet MSL whenever possible. In addition, only minimal troubleshooting should be attempted with passengers onboard.

It should also be noted that as the cabin climbs past 12,500 feet pressure altitude, the passenger oxygen masks should deploy. If they do not, the passenger manual dropout handle should be pulled. See the Oxygen section of this workbook for more details about the oxygen system.

Most pressurization malfunctions will show up shortly after takeoff. They will show three general symptoms: rapid pressurization toward maximum differential, lack of pressurization (i.e., the cabin climbs at the same rate as the aircraft), and cabin leakdown (i.e., the cabin leaks pressurization slowly—500 feet/minute at low-pressure differentials and faster at high-pressure differentials.) The first two symptoms are generally caused by controller, control system, or outflow/safety valve malfunctions. The third is normally caused by air inflow problems.

Air inflow problems could be caused by a malfunction within the flow control units. Since it is unusual for both flow control units to fail simultaneously, one probably failed earlier and went undetected.

A second cause of air inflow problems on takeoff can be the BLEED AIR VALVE switches. They could have been left in the ENVIR OFF or PNEU & ENVIR OFF. If they are OPEN, an electrical failure to the switches would cause these normally closed valves to close (see Figure 12-7). In this case, the pilot should check the bleed air control circuit breakers on the copilot's circuit-breaker panel.

If cabin altitude descends rapidly shortly after takeoff, it is caused by closed outflow and safety valves (see Figures 12-8 and 12-9). The safety valve normally closes shortly after takeoff, but the outflow valve should modulate open as directed by controller pressure. The problem lies somewhere in the plumbing and/or components in the outflow system.

Possible sources of this problem include: a stuck preset solenoid, a cracked pressurization controller, a diaphragm failure in the pressurization controller, an open moisture drain, disconnected or leaking plumbing, a cracked outflow valve, or a failed diaphragm in the outflow valve.



In this situation, the pilot should turn off both bleed air valves, which will stop the P₃ bleed air inflow and depressurize the cabin at its leak rate. Once the cabin is stabilized, cycling the CABIN PRESS switch to TEST may free a stuck preset solenoid. The moisture drain can be checked through its access panel on the lower right sidewall of the baggage compartment. Any additional troubleshooting should be accomplished on the ground.

Failure of the cabin to pressurize shortly after takeoff is indicative of inflow and/or outflow malfunctions. If the cabin altitude climbs with the airplane, it indicates an outflow problem caused by an open outflow and/or safety valve. The outflow valve may have prematurely opened due to the preset solenoid opening on the ground. In this situation, the controller prematurely thought the aircraft took off. Controller pressure will hold the outflow valve wide open until the aircraft "catches up" to the controller pressure. This may not occur until the aircraft climbs through the altitude in the CABIN ALT window on the pressurization controller. If this condition is suspected, the pilot can turn the rate knob full counterclockwise to the minimum position, which will minimize any additional change in controller pressure. An alternative action would be to select a lower cabin altitude with the selector; however, it is important to remember to reselect the proper setting once the cabin pressurizes normally.

If the safety valve remained open after takeoff, cabin altitude would climb together with the aircraft. This could be caused by: the CABIN PRESS switch in DUMP, failure of the left main squat switch, failure of the dump solenoid, or a stuck safety valve (see Figure 12-9). The pilot should first check to ensure the CABIN PRESS switch is in PRESS (see Figure 12-15). If it is already in PRESS, move the switch to TEST, which will override the squat switch which may have malfunctioned. If the cabin begins to pressurize, hold the switch in TEST until the cabin differential pressure exceeds 0.5 psid, then pull

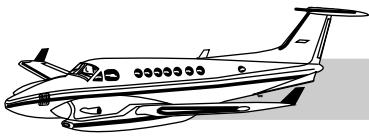
the PRESS CONTROL circuit breaker on the copilot's circuit breaker panel. Normal pressurization control will be resumed, but the electrical dump functions will not be available.

NOTE

With the PRESS CONTROL circuit breaker pulled out, the electrical functions related to cabin dump are deenergized. Therefore, pressurization dump with the CABIN PRESS switch or upon landing touchdown will be disabled until the circuit breaker is reset.

If the cabin does not pressurize with the CABIN PRESS switch in TEST, then the problem is beyond the capability of in-flight troubleshooting and must be repaired when the aircraft lands.

Symptoms such as unusual or excessive pressure bumps on takeoff or any time the controller is reset indicate sticking outflow/safety valves, possibly due to buildup of tars and nicotine. The filters associated with this system may also be dirty. A symptom of dirty filters is a large difference (over 200 feet/minute) in the cabin climb vs. the cabin descent with the rate knob in the same position. These valves and filters should be checked at regular maintenance inspections; however, inspection intervals of these components may be shortened when unusual conditions (heavy smoking, dusty atmosphere) exist.



See Chapter 14, “Landing Gear and Brakes,” for information on the hydraulic power systems.



CHAPTER 14

LANDING GEAR AND BRAKES

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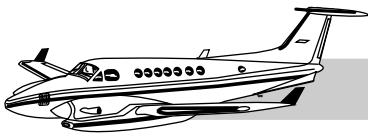


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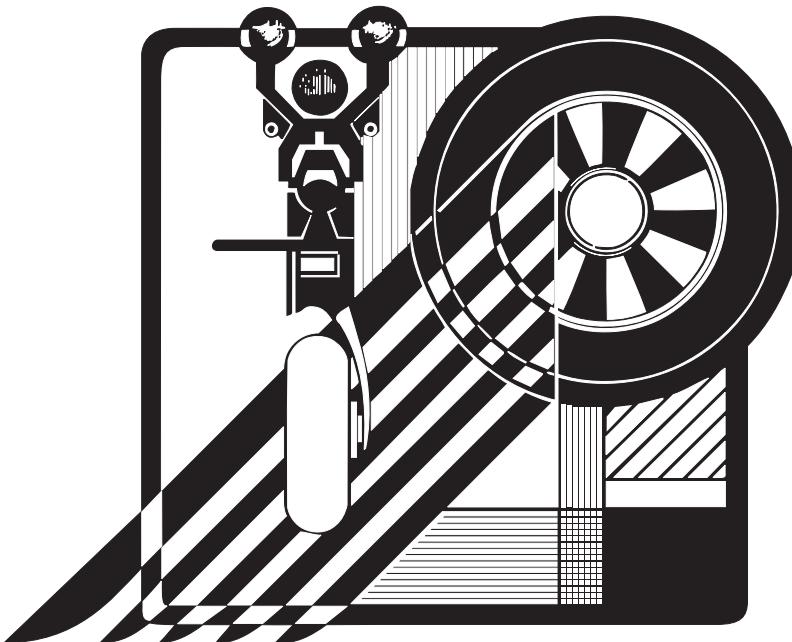
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CHAPTER 14

LANDING GEAR AND BRAKES



INTRODUCTION

An understanding of the landing gear system will aid the pilot in proper operation of the landing gear and in handling abnormal procedures. This chapter, in addition to describing the system, identifies inspection points and abnormal conditions. An understanding of the brake system will help the pilot operate the brakes safely and with minimum wear. This chapter also describes the brake system and points out operating and servicing procedures.

DESCRIPTION

The landing gear system information presents a description and discussion of the landing gear system, landing gear controls and limits. The indicator system and alternate landing gear extension are also described.

The wheel brake system information presents a description and discussion of the wheel brake system. Correct use of the brakes and parking brakes, brake system description, and what to look for when inspecting brakes are also detailed.



LANDING GEAR SYSTEM

The landing gear system on the Super King Air 300 and 350 is a retractable, electrically powered, hydraulically actuated, tricycle gear system. When the landing gear is fully retracted, it is completely enclosed by the gear door assemblies. The system is operated by a control switch handle on the pilot's right subpanel. An alternate means of landing gear extension is a manually-operated hand pump.

LANDING GEAR COMPONENTS

The landing gear assembly (main and nose) consists of a shock strut, torque knee (scissors), drag leg, actuator, a shimmy damper, brake assembly, wheel, and tire. The brake assemblies are located on the main gear assemblies (Figure 14-1); the shimmy damper is mounted on the nose gear assembly (Figure 14-2).

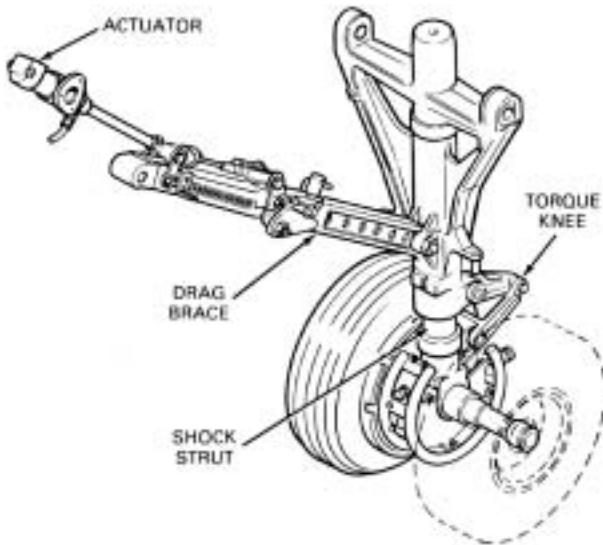


Figure 14-1 Main Landing Gear Assembly

The upper end of the drag legs and two points on the shock strut assemblies are attached to the airplane structure. When the gear is extended, the drag braces become rigid components of the gear assemblies. Hydraulic actuators extend and retract the gear assembly.

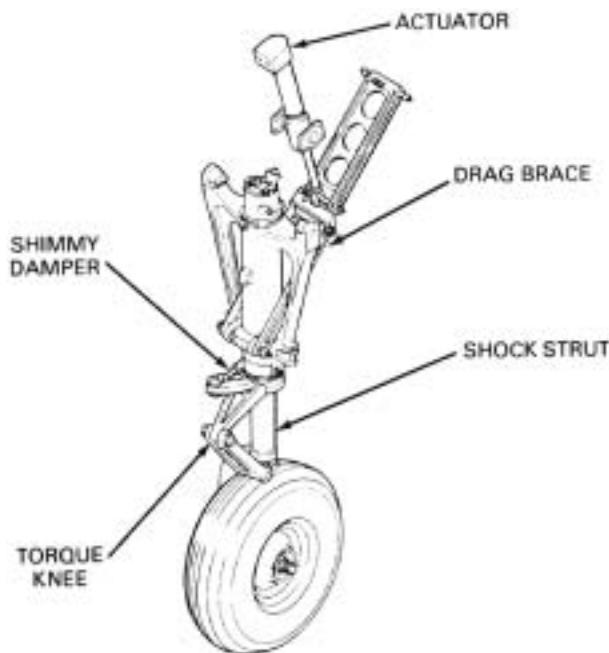


Figure 14-2 Nose Gear Assembly

The landing gear incorporates air/oil shock struts that are filled with both compressed air and hydraulic fluid. Airplane weight is borne by the air charge in the shock struts. At touchdown, the lower portion of each strut is forced into the upper cylinder; this moves fluid through an orifice, further compressing the air charge and thus absorbing landing shock. Orifice action also reduces bounce during landing.

At takeoff, the lower portion of the strut extends until an internal stop engages on the main gear. On the nose gear, the lower portion of the strut extends until the torque knee prevents further extension.

A torque knee connects the upper and lower portion of the shock strut. It allows strut compression and extension but resists rotational forces, thereby keeping the wheels aligned with the airplane longitudinal axis. On the nose gear assembly, the torque knee also transmits steering motion to the nosewheel, and nosewheel shimmy motion to the shimmy damper.



The shimmy damper, mounted on the right side of the nose gear strut, is a balanced hydraulic cylinder that bleeds fluid through an orifice to dampen nosewheel shimmy.

WHEEL WELL DOOR MECHANISMS

The landing gear doors consist of nose gear doors and two sets of main gear doors. Landing gear doors are mechanically actuated by gear movement during extension and retraction (Figure 14-3).

The main and nose gear doors are hinged at the sides and are spring-loaded to the open position. As the landing gear is retracted, a roller on each side of the nose engages a cam assembly on each door, and draws the doors closed behind the gear. The reverse action takes place and spring-loading takes effect as the gear is extended. When the cam has left the roller, springs pull the linkage over-center to hold the doors open.

NOSEWHEEL STEERING

Direct linkage to the rudder pedals permits nosewheel steering when the nose gear is down (Figure 14-4). One spring-loaded link in the system absorbs some of the force applied to the interconnected rudder pedals until the nosewheel is rolling. At this time the resisting force is less, and more pedal motion results in more nosewheel deflection.

Since motion of the pedals is transmitted via cables and linkage to the rudder, rudder deflection occurs when force is applied to the rudder pedals. With the nose landing gear retracted, some of the force applied to the rudder pedals is absorbed by the spring-loaded link in the steering system, so that there is no motion at the nosewheel, but rudder deflection still occurs. The nosewheel is self-centering upon retraction.



Figure 14-3 Main Gear Door Mechanism

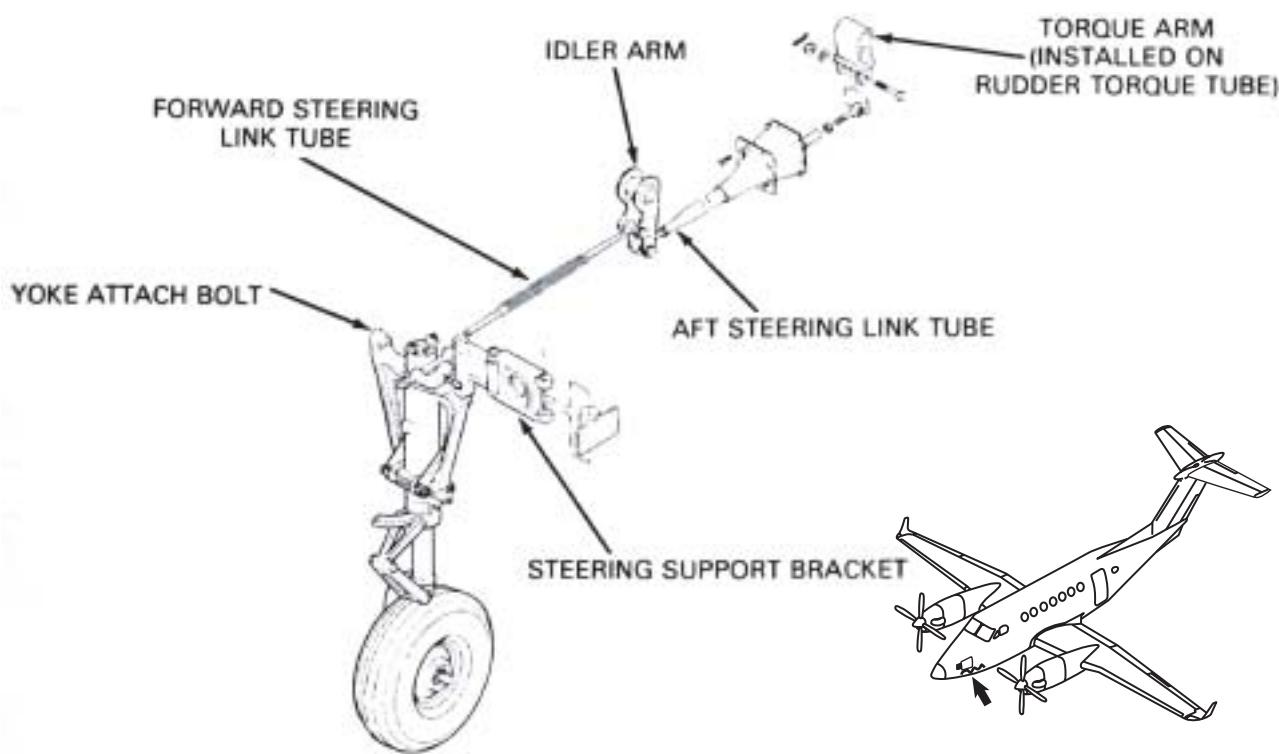
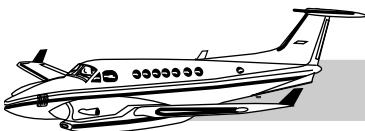


Figure 14-4 Nosewheel Steering Mechanism

When force on the rudder pedal is augmented by a main wheel braking action, the nosewheel deflection can be considerably increased. Nosewheel steering provides 12° to the left and right of center. Castering provides an additional 36°, for a total possible deflection of 48° left and right.

HYDRAULIC LANDING GEAR EXTENSION AND RETRACTION

The retractable tricycle landing gear is electrically controlled and hydraulically actuated. The system utilizes folding braces, called drag braces, that are locked in place when the gear is fully extended. The landing gear is held in the up position by hydraulic pressure.

Nose gear downlock is accomplished by an internal mechanical lock in the nose gear actuator and the nose gear drag brace assembly. "J" hooks fitted to each main gear drag brace provide a positive mechanical downlock for the main gear.

System hydraulic pressure is supplied by a hydraulic power pack (Figure 14-5). A reservoir located in the left center wing section provides hydraulic fluid to the power pack. The reservoir incorporates a dipstick to provide a visual check of fluid level.

An electrically actuated selector valve controls the flow of hydraulic fluid to the individual gear actuators. The selector valve receives electrical power through the landing gear control switch.

Accidental retraction of the landing gear is prevented through safety switches located on both main landing gears.

The nose and main landing gear assemblies are extended and retracted by a hydraulic power pack in conjunction with hydraulic actuators (Figure 14-6). The hydraulic power pack is located in the middle of the left wing center section, just forward of the main spar.

One hydraulic actuator is located at each landing gear. The power pack consists of: a hydraulic pump, a 28-VDC motor, a two-section fluid reser-

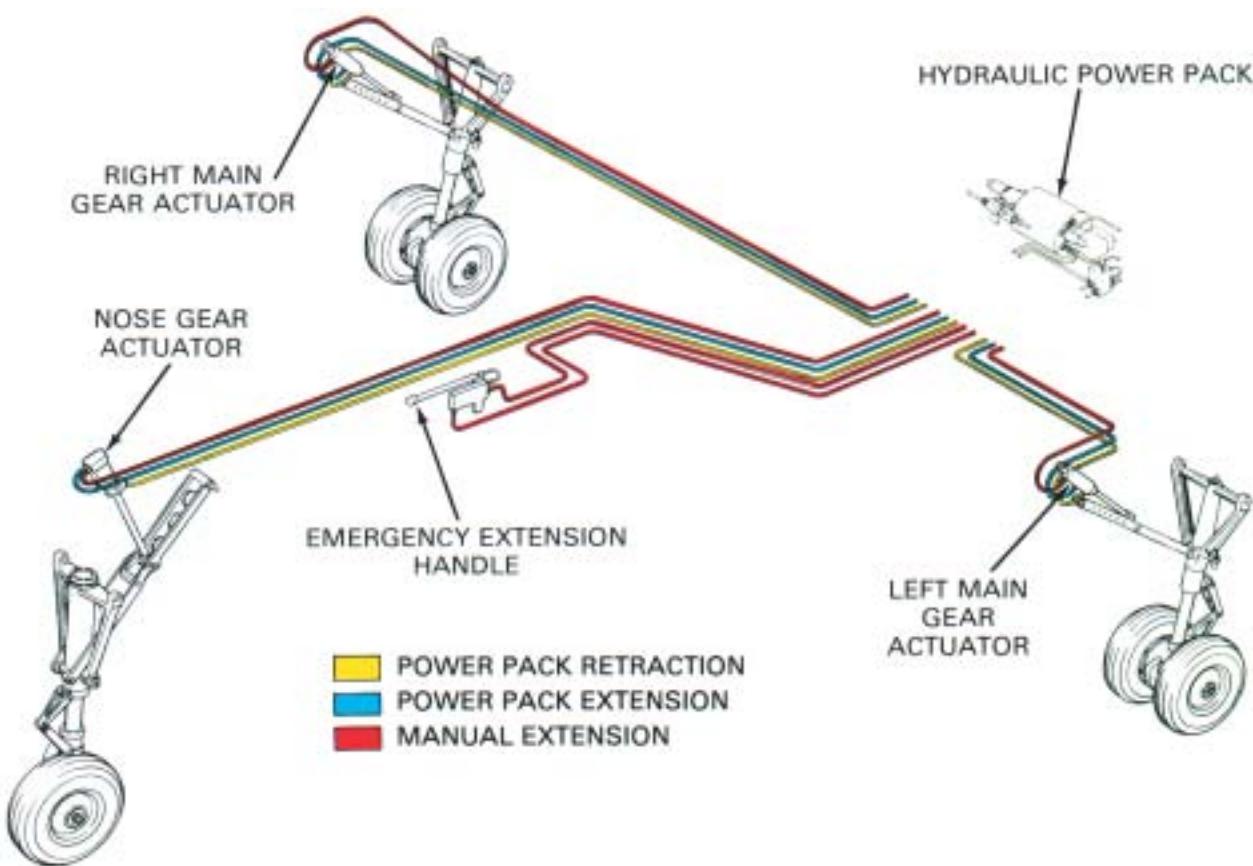


Figure 14-5 Hydraulic Landing Gear Plumbing Schematic

voir, filter screens, a gear selector valve, an up selector solenoid and a down selector solenoid, a fluid level sensor, and an uplock pressure switch. For manual extension the system has a hand-lever-operated pump. The pump handle is located on the floor between the pilot's seat and the pedestal.

Three hydraulic lines (one for normal extension and one for retraction, routed from the power pack, and one for manual extension routed from the hand pump) are routed to the nose and main gear actuators. The normal extension lines and the manual extension lines are connected to the upper end of each hydraulic actuator. The hydraulic lines for retraction are fitted to the lower ends of the actuators. Hydraulic fluid under pressure (generated by the power pack pump and contained in the accumulator) acts on the piston faces of the actuators (which are attached to folding drag braces) resulting in the extension or retraction of the landing gear.

When the actuator pistons are positioned to fully extend the landing gear, an internal mechanical lock in the nose gear actuator and the nose gear drag brace assembly lock the nose gear in the down position. In this position, the internal downlock mechanism in the nose gear actuator will position the actuator downlock switch to interrupt current to the nose gear part of the pump motor control circuit.

A notched J-hook, lock link, and lock link guide attachments fitted to each main gear drag brace provide a positive downlock action for the main gear. A downlock switch on each J-hook assembly will interrupt its part of the pump motor control circuit when the respective main gear is down and locked. The motor will continue to run until all three landing gear are down and locked.



HYDRAULIC FLUID LEVEL INDICATION SYSTEM

A HYD FLUID LOW caution annunciator (Figure 14-7) in the caution/advisory panel will illuminate (yellow) whenever the hydraulic fluid level in the landing gear power pack reservoir is low. A fiber-optic sensing unit mounted on the motor end of the power pack provides the necessary switching circuitry to illuminate the low fluid light.

The annunciator and associated circuitry is tested by pressing the HYD FLUID SENSOR TEST button located on the pilot's right subpanel. The test button sends a signal to the fiber-optic sensor in the hydraulic fluid reservoir which causes the sensor to "sense" a low fluid level. The sensor then sends a low fluid signal to the annunciator panel. A five-second delay can be expected before the annunciator light illuminates during the test and also before the annunciator extinguishes when the button is released.

When a low fluid level is indicated, attempt to lower the landing gear normally upon reaching the destination. If it fails to extend completely, follow the "Landing Gear Manual Extension" procedure in the Abnormal Procedures section of the *POH*.

CONTROL CIRCUITS

The landing gear hydraulic power pack motor is controlled by the landing gear control switch handle placarded LDG GEAR CONTROL-UP-DN, located on the pilot's right subpanel (Figure 14-8). The control switch handle must be pulled out of a detent before it can be moved from either the UP or DN position.

When the LDG GEAR control is moved to the UP position in flight, a control circuit is completed to the gear selector valve up solenoid. This moves the gear selector valve so that fluid can flow to the retraction side of the system. A control circuit is also completed to the pump motor relay which will cause the pump motor (or hydraulic power pack) to operate.

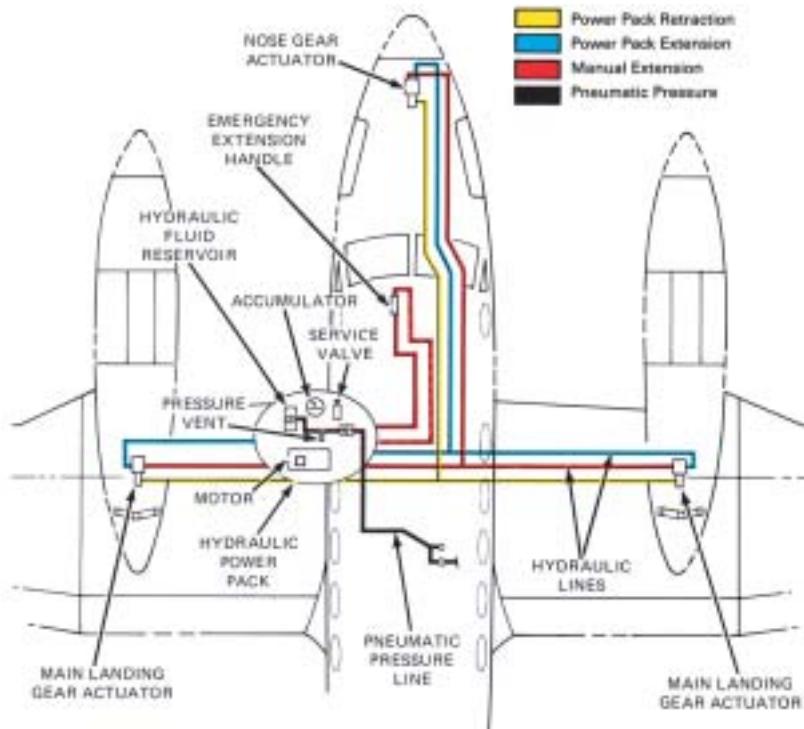


Figure 14-6 Hydraulic Landing Gear Schematic



Figure 14-7 Hydraulic Fluid Low Caution Annunciator

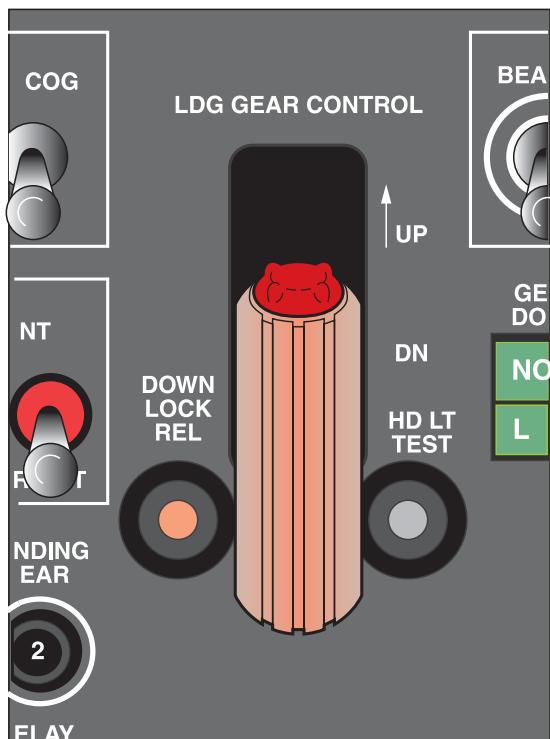


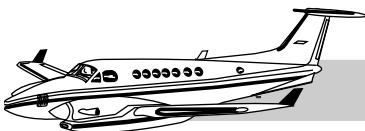
Figure 14-8 Landing Gear Control Switch Handle

The nose gear actuator will unlock and begin to retract the nose gear when 200 to 400 psi of hydraulic pressure is applied to the retract port of the nose gear actuator. The main gear will begin

to retract after the main gear actuators unlock the respective J-hooks on the main gear braces. Hydraulic system pressure performs the uplock function, holding the landing gear in the retracted position. When hydraulic pressure reaches approximately 2,775 psi, the uplock pressure switch will cause the landing gear relay to open and interrupt current to the pump motor. The same pressure switch will cause the pump to actuate and repressurize the system should the pressure drop to approximately 2,375 psi.

When the LDG GEAR control is moved to the DN position in flight, a control circuit is completed to the gear selector valve down solenoid. This moves the gear selector valve so that fluid can flow to the extension side of the system. A control circuit is also completed to the pump motor relay which will cause the pump motor (or hydraulic power pack) to operate.

When the nose gear is down and locked, the internal downlock mechanism in the nose gear actuator will position its actuator downlock switch to interrupt current to the nose gear part of the pump motor control circuit. A downlock switch on the J-hook assembly of each main gear drag brace will interrupt its part of the pump motor control circuit when the respective main gear is down and locked. The pump motor will continue to operate until all three gear are down and locked.



The landing gear control circuit is protected by a 2-ampere circuit breaker located on the pilot's right subpanel. Power for the pump motor is supplied through the landing gear motor relay and a 60 ampere relay circuit breaker, both of which are located next to the pump motor in the middle of the left wing center section, just forward of the main spar. The motor relay is energized by current from the 2-ampere circuit breaker.

If the pump motor fails to shut off after 13 to 15 seconds, a timer will activate the logic relay, which will open the pump motor relay and stop the motor. This action also shorts out the 2-ampere control circuit and trips the LANDING GEAR RELAY circuit breaker on the pilot's right subpanel.

Safety switches (Figure 14-9) called "squat" switches, on the main gear torque knees, open control circuits when the oleo strut is compressed. This is done to prevent gear retraction on the ground.

When the left squat switch senses that the aircraft is on the ground, it opens the control circuit to the

landing gear selector valve up solenoid to help prevent inadvertent gear retraction on the ground. When the right squat switch senses that the aircraft is on the ground, it opens the control circuit to the landing gear selector valve up solenoid, the landing gear motor control relay, and the landing gear handle lock solenoid. This further helps to prevent inadvertent gear retraction on the ground.

The landing gear handle lock prevents the LDG GEAR CONTROL handle from being placed in the UP position when the airplane is on the ground. It automatically unlocks when the airplane leaves the ground because the right squat switch closes and a circuit is completed through the solenoid that moves the lock. In the event of a malfunction of the handle lock solenoid or the squat switch circuit, the handle lock can be overridden by pressing downward on the red DOWN LOCK REL button. The release button is located immediately left of the LDG GEAR CONTROL handle.

The LDG GEAR CONTROL handle should never be moved out of the DN detent while the airplane is

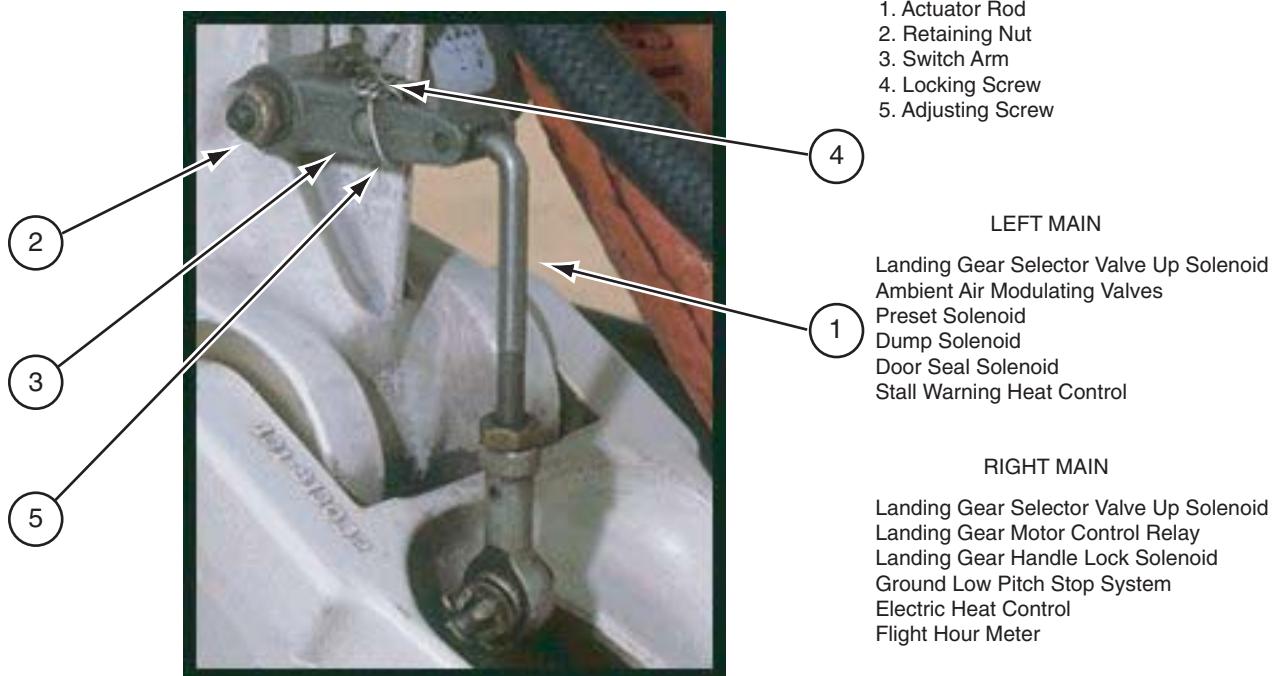


Figure 14-9 Safety Switches



on the ground. If it is, the landing gear warning horn will sound and the red gear-in-transit lights in the LDG GEAR CONTROL handle will illuminate (provided the MASTER SWITCH is ON), warning the pilot to return the handle to the DN position. The landing gear should not retract on the ground because the squat switches also prevent activation of the gear select valve and power pack motor.

POSITION INDICATORS

Landing gear position is indicated by an assembly of three lights in a single unit located on the pilot's right subpanel (Figure 14-10). The unit has a light transmitting cap that is marked as follows: NOSE-L-R. One light in each main segment, and two in the nose segment when illuminated, make each segment appear green to indicate that the gear is down and locked (Figure 14-11). Absence of illumination indicates that the gear is not down and locked (Figure 14-11). The

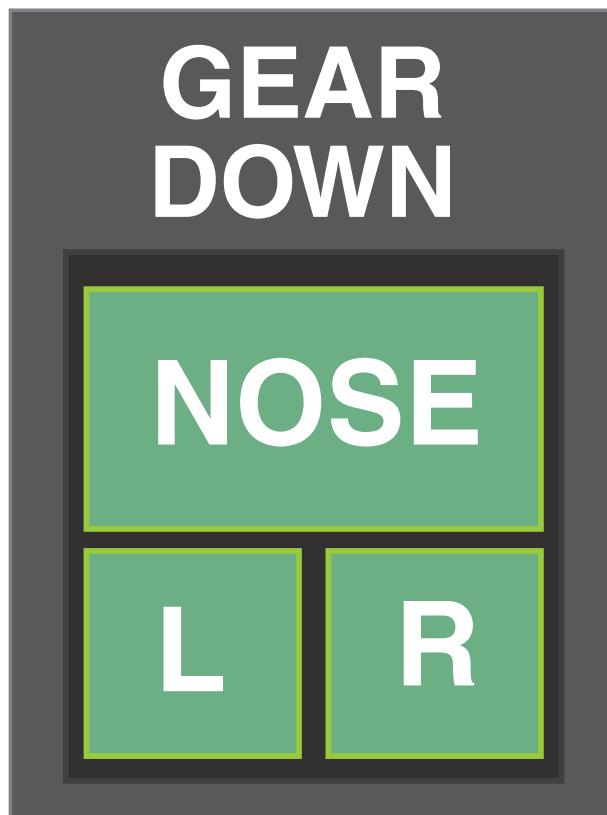


Figure 14-10 Gear Position Indicator

green position indicator light bulbs may be checked by depressing on the light housing.

Two red indicator lights, located in the LDG GEAR CONTROL handle, illuminate to show that the gear is in transit (Figure 14-12) unsafe, or unlocked. Gear UP is indicated when the red lights go out and the gear is obviously retracted. Gear DOWN is indicated when all three green gear position indicator annunciators are illuminated and the red lights extinguish. The red lights in the handle also illuminate when the landing gear warning horn is actuated.

The red control handle lights may be checked by pressing HDL LT TEST button located adjacent to the LDG GEAR CONTROL handle. They may also be tested in conjunction with the gear warning horn with the STALL WARN TEST-OFF LDG GEAR WARN TEST switch on the copilot's left subpanel.

Upon gear retraction, when the gear reaches the fully-retracted position, each gear strut actuates its respective up-position switch to open the in-transit light circuit. Conversely, upon gear extension, when the gear moves from the fully-retracted position, each strut releases its respective up switch and illuminates the in-transit light (Figures 14-14, 14-15, and 14-16).

The in-transit light extinguishes when both main gear drag brace J-hook assemblies and the nose gear drag brace position their respective downlock switches to the down and locked position. In this position, the main gear downlock switches illuminate the respective gear down-position lights. The nose gear drag brace downlock switch and the internal actuator downlock switch must both be in the down and locked position to illuminate the nose gear down-position light.

The landing gear in-transit light will indicate one or all of the following conditions:

1. The landing gear handle is up and the airplane is on the ground with weight on the landing gear (the warning horn will also be operating).

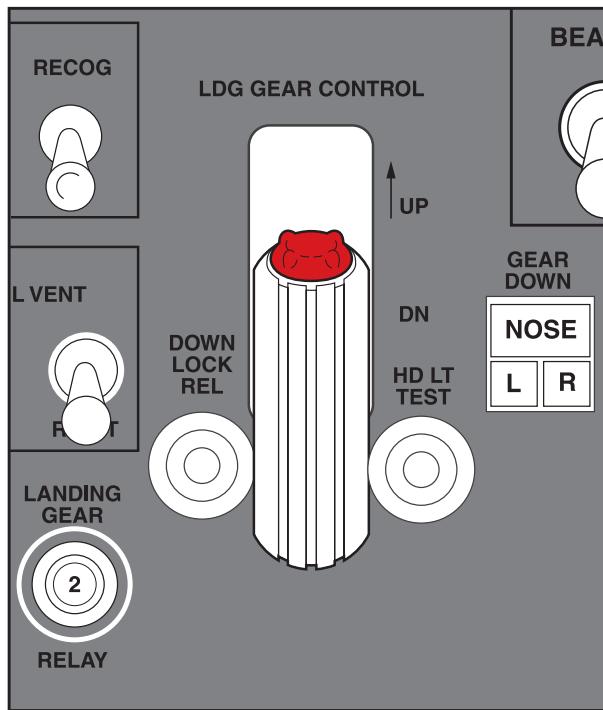


Figure 14-11 Landing Gear Controls

2. One or both power levers are retarded below approximately $86 \pm 1\% N_1$ and one or more of the landing gear is not down and locked (the warning horn will also be operating).
3. Any one or all landing gears are not in the fully retracted or in the down-and-locked position.
4. The warning horn has been silenced and will not operate.

Thus, the function of the landing gear in-transit light is to indicate that the landing gear is in transit, the position of the landing gear does not match that of the handle, or the landing gear warning horn has been silenced and not rearmed. The light remains on when the horn is silenced.

LANDING GEAR WARNING SYSTEM

The landing gear warning system is provided to warn the pilot of an unsafe condition within the

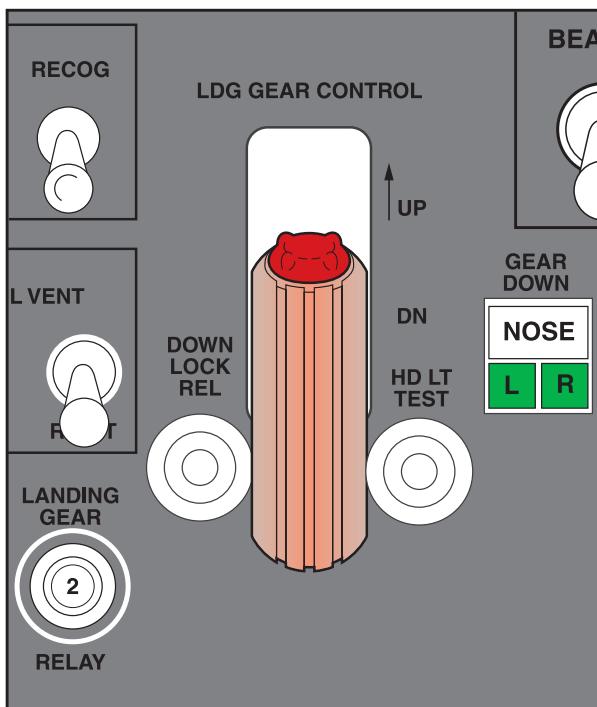


Figure 14-12 Gear Position Indicator—No Illumination (Nose)

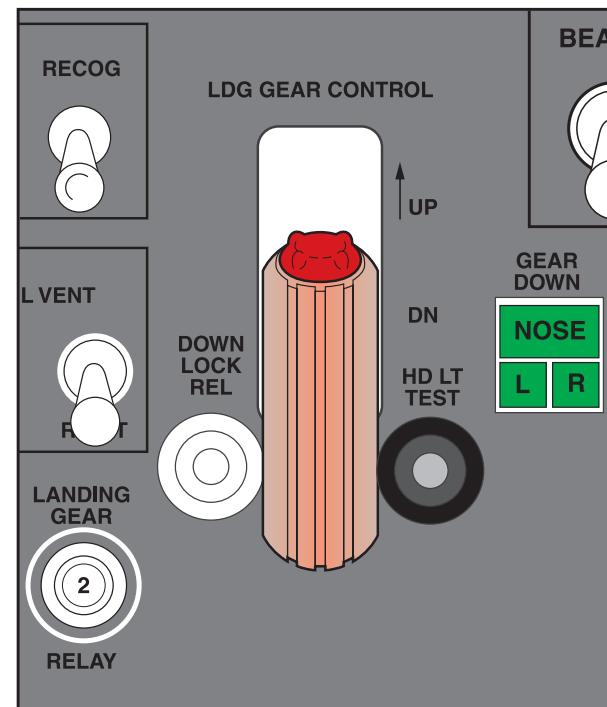


Figure 14-13 Landing Gear Control Switch Handle Light Test

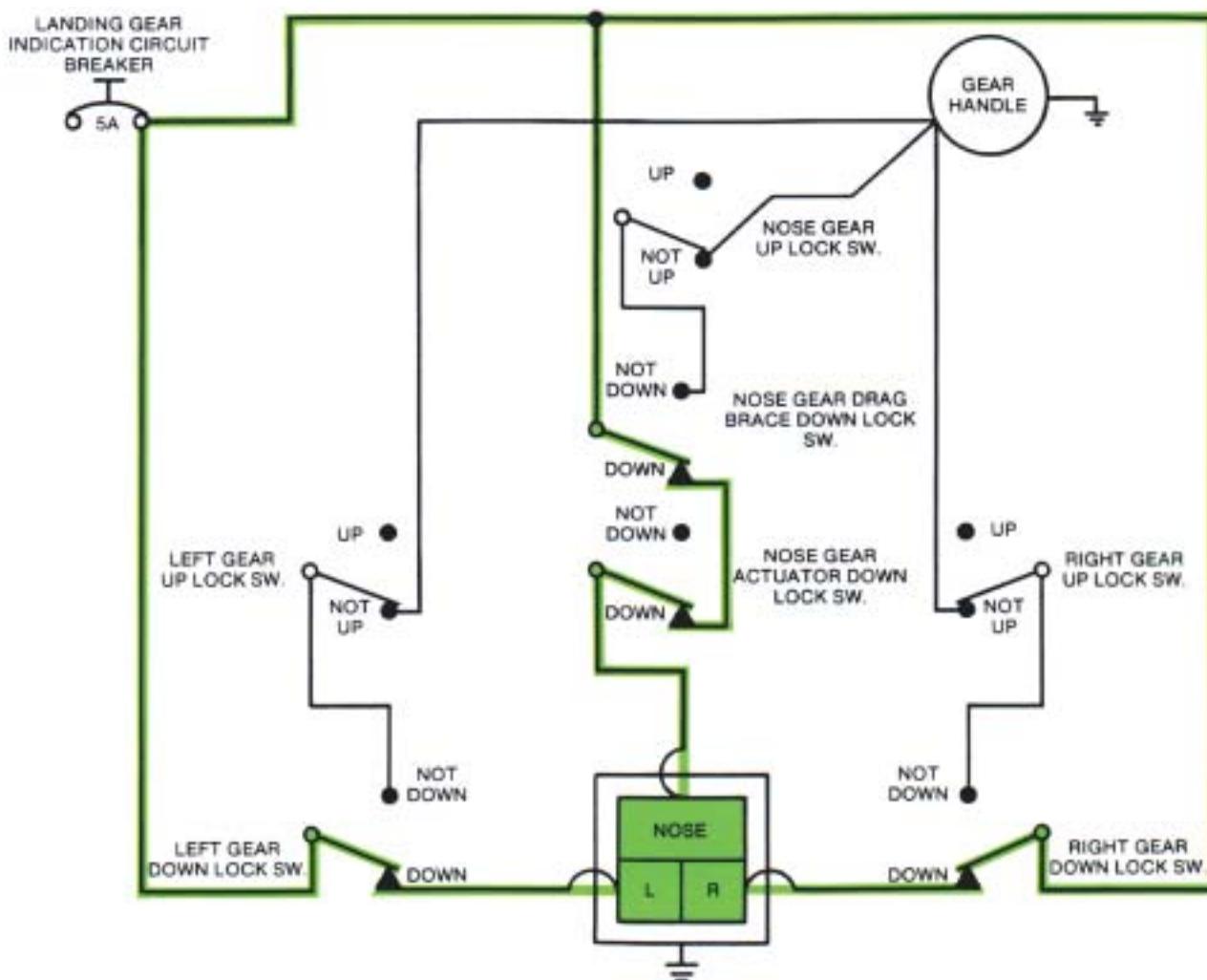


Figure 14-14 Landing Gear Position Indication—Gear Extended

landing gear system. When the airplane is on the ground, if the LDG GEAR CONTROL handle is not in the DN position, the landing gear warning horn sounds and the red gear-in-transit lights in the LDG GEAR CONTROL handle illuminate. Place the LDG GEAR handle DN to silence the horn and extinguish the handle light.

In flight, various warning modes are available, depending upon the position of the flaps. With the flaps in the UP or APPROACH position and either or both power levers retarded below approximately 86% N₁, the warning horn will sound and the LDG GEAR CONTROL handle lights will illuminate.

The horn can be silenced by pressing the GEAR WARN SILENCE button adjacent to the LDG GEAR CONTROL handle (and/or the silence button on the left power lever); the lights in the LDG GEAR CONTROL handle cannot be canceled. The landing gear warning horn will be rearmed if the power levers are sufficiently advanced.

With the FLAPS beyond APPROACH position, the warning horn and landing gear control handle lights will be activated regardless of the power settings; neither can be canceled.

**Table 14-1 LANDING GEAR WARNING HORN OPERATION**

GEAR POSITION	FLAPS	POWER	HORN	SILENCE MODE
Not down	Up	Above 86% N ₁	No	N/A
Not down	Up	Below 86% N ₁	Yes	Silence button
Not down	Approach	Below 86% N ₁	Yes	Silence button
Not down	Beyond approach	Any	Yes	Lower gear

Landing gear warning horn operational data is shown in Table 14-1 above.

HYDRAULIC SCHEMATICS

The hydraulic gear schematics shown are for gear extended, retraction, retracted, extension, hand pump extension, and maintenance retraction (Figures 14-17, 14-18, 14-19, 14-20, 14-23, and 14-24).

Figure 14-17 is the condition of the system after a normal gear extension. Electrical power comes from the 2-amp LDG GEAR CONTROL circuit breaker through the landing gear control switch and on to the three downlock switches. Each gear is down and locked so these three switches are open and no electrical power passes through them. However, power continues to be provided to the hydraulic selector valve to hold it in the down position.

The service valve must be down for electrohydraulic action (up or down) and for manual extension of the landing gear.

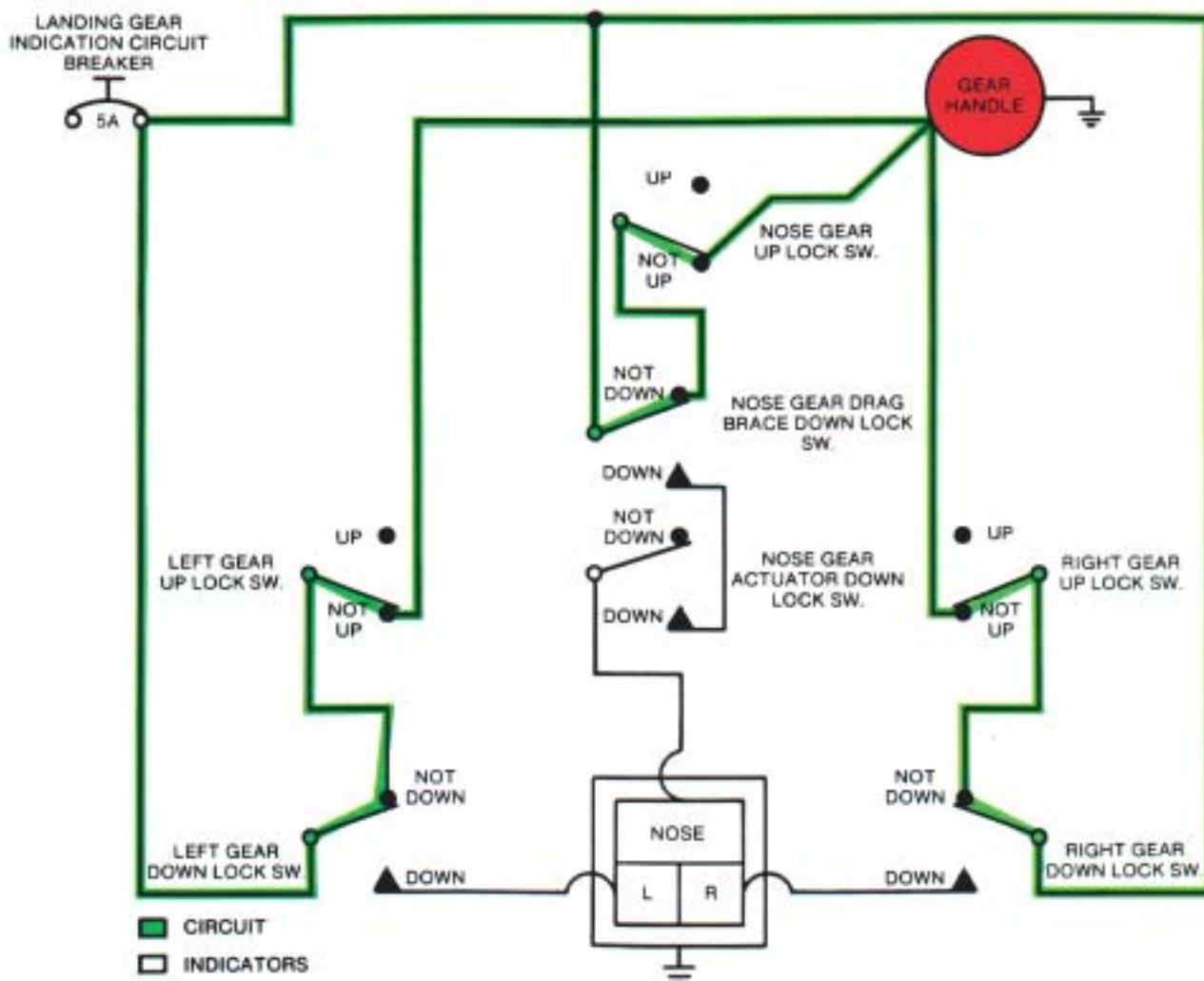


Figure 14-15 Landing Gear Position Indication—Gear In Transit

LANDING GEAR RETRACTION

When the LDG GEAR control is moved to the UP position in flight, a control circuit is completed to the gear selector valve up solenoid (Figure 14-18 and 14-19). This moves the gear selector valve so that fluid can flow to the retraction side of the system. A control circuit is also completed to the pump motor relay which will cause the pump motor (or hydraulic power pack) to operate.

The nose gear actuator will unlock and begin to retract the nose gear when 200 to 400 psi of

hydraulic pressure is applied to the retract port of the nose gear actuator. The main gear will begin to retract after the main gear actuators unlock the respective J-hooks on the main gear drag braces.

After approximately six seconds, the retraction cycle is complete. Once the landing gear reaches full-up travel, each actuator physically bottoms out. The pressure on the retract line builds rapidly until pressure reaches approximately 2,775 psi. The uplock pressure switch opens at this time, breaking the power circuit to the pump motor, and stopping the hydraulic

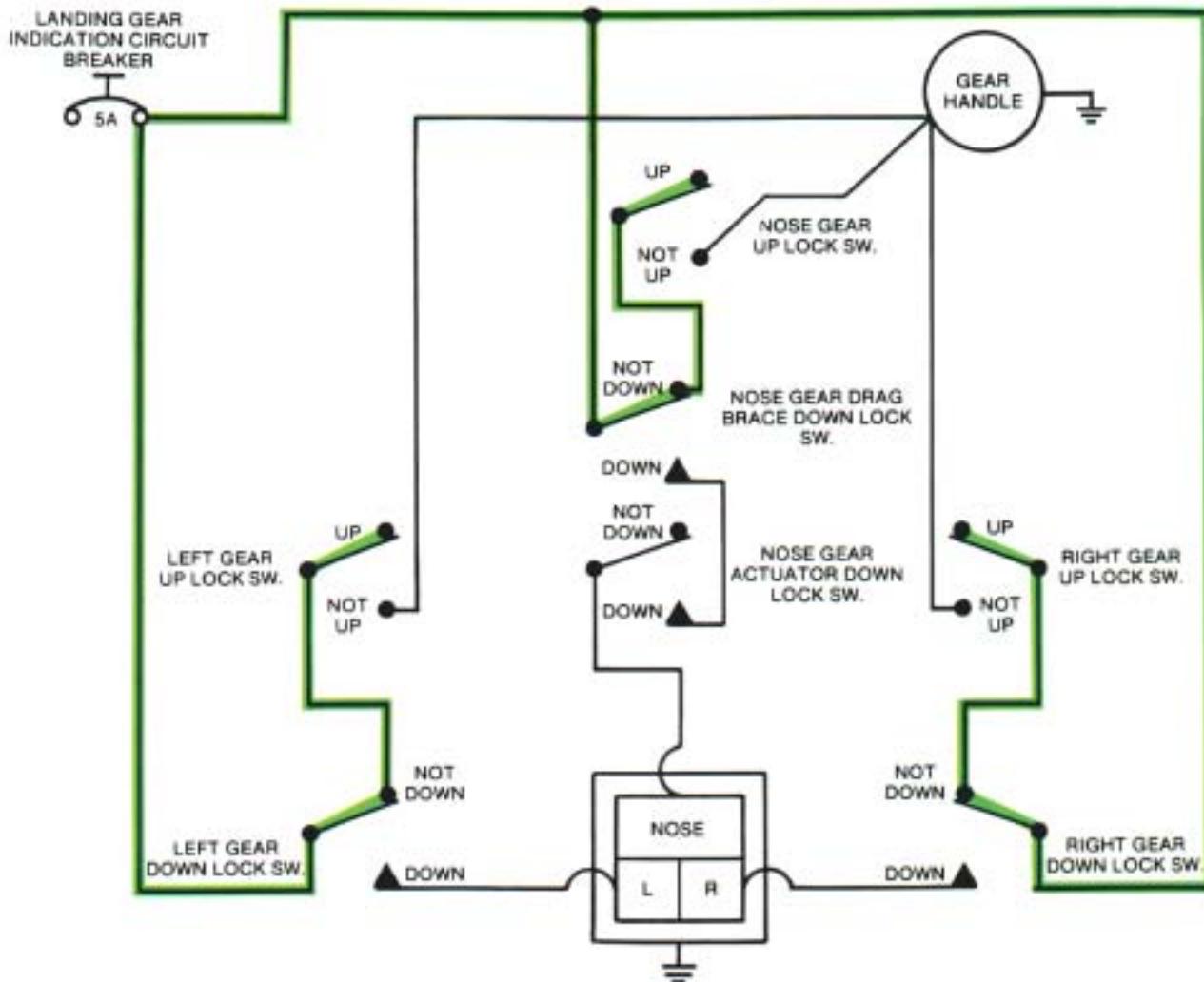


Figure 14-16 Landing Gear Position Indication—Gear Up

pump. This pressure switch will close periodically when pressure drops to approximately 2,375 psi, due to the normal system pressure leak-down, and reenergize the pump to restore needed uplock pressure. Consequently, when the gear is retracted, pressure will be maintained between approximately 2,375 to 2,775 psi to keep the landing gear in their retracted position. An accumulator precharged to 800 psi, located in the left-hand wing inboard of the nacelle, is designed to aid in maintaining the system pressure when the gear is up.

LANDING GEAR EXTENSION

When the LDG GEAR control is moved to the DN position in flight, a control circuit is completed to the gear selector valve down solenoid (Figure 14-20). This moves the gear selector valve so that fluid can flow to the extension side of the system. A control circuit is also completed to the pump motor relay which will cause the pump motor (or hydraulic power pack) to operate.

After approximately six seconds, the extension cycle is complete. When the nose gear is down

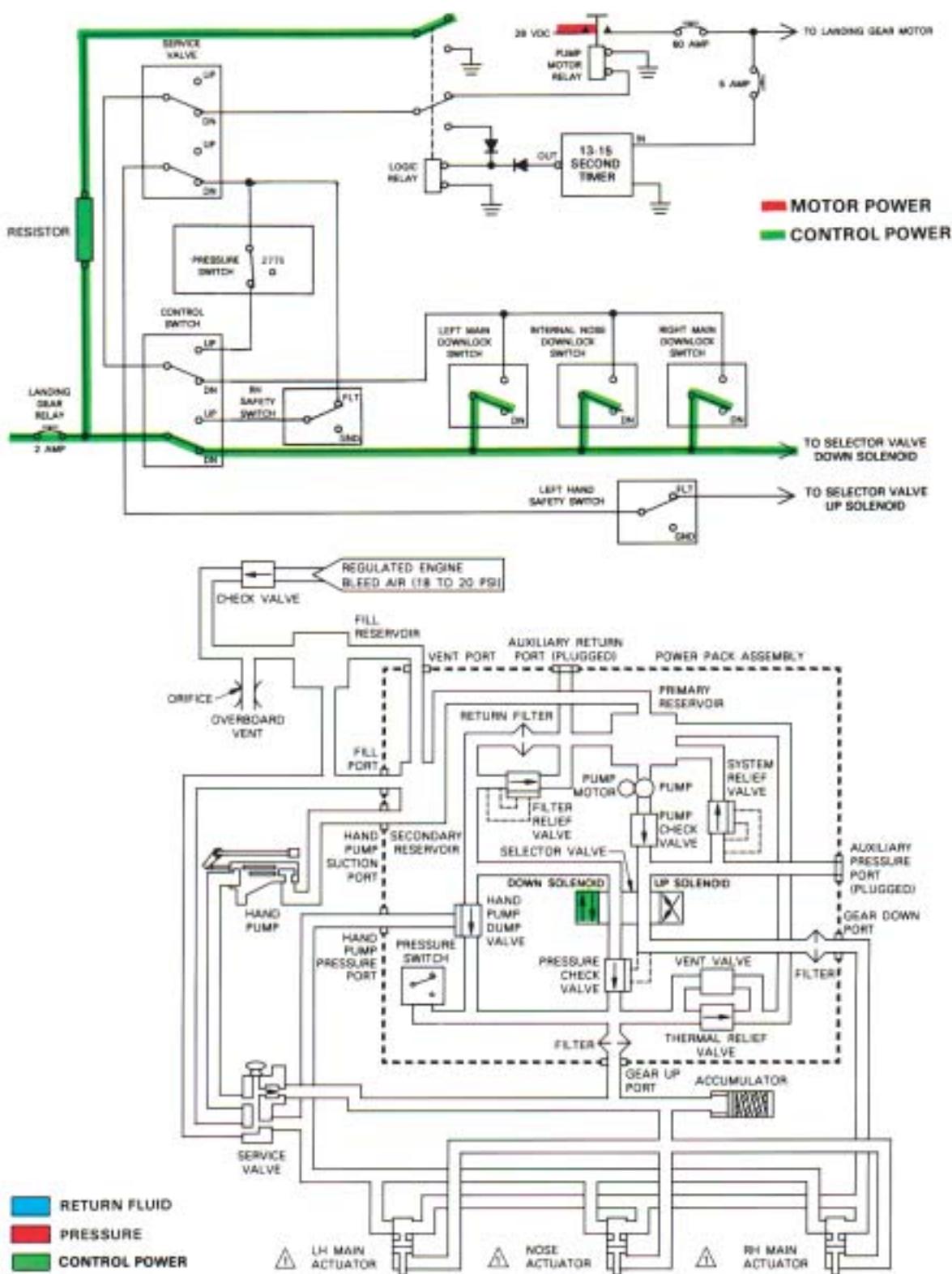


Figure 14-17 Landing Gear Extended Schematic

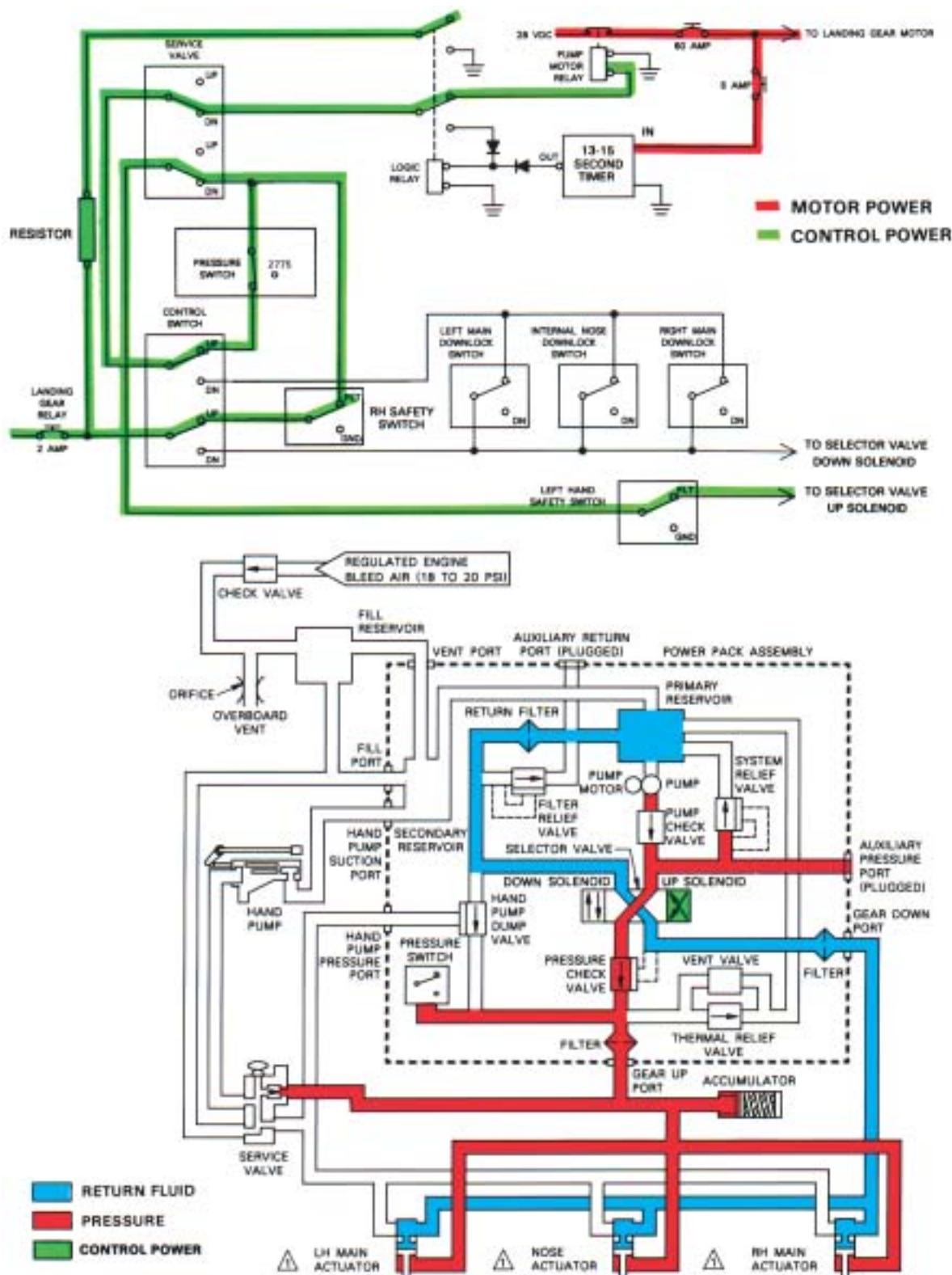


Figure 14-18 Landing Gear Retraction Schematic

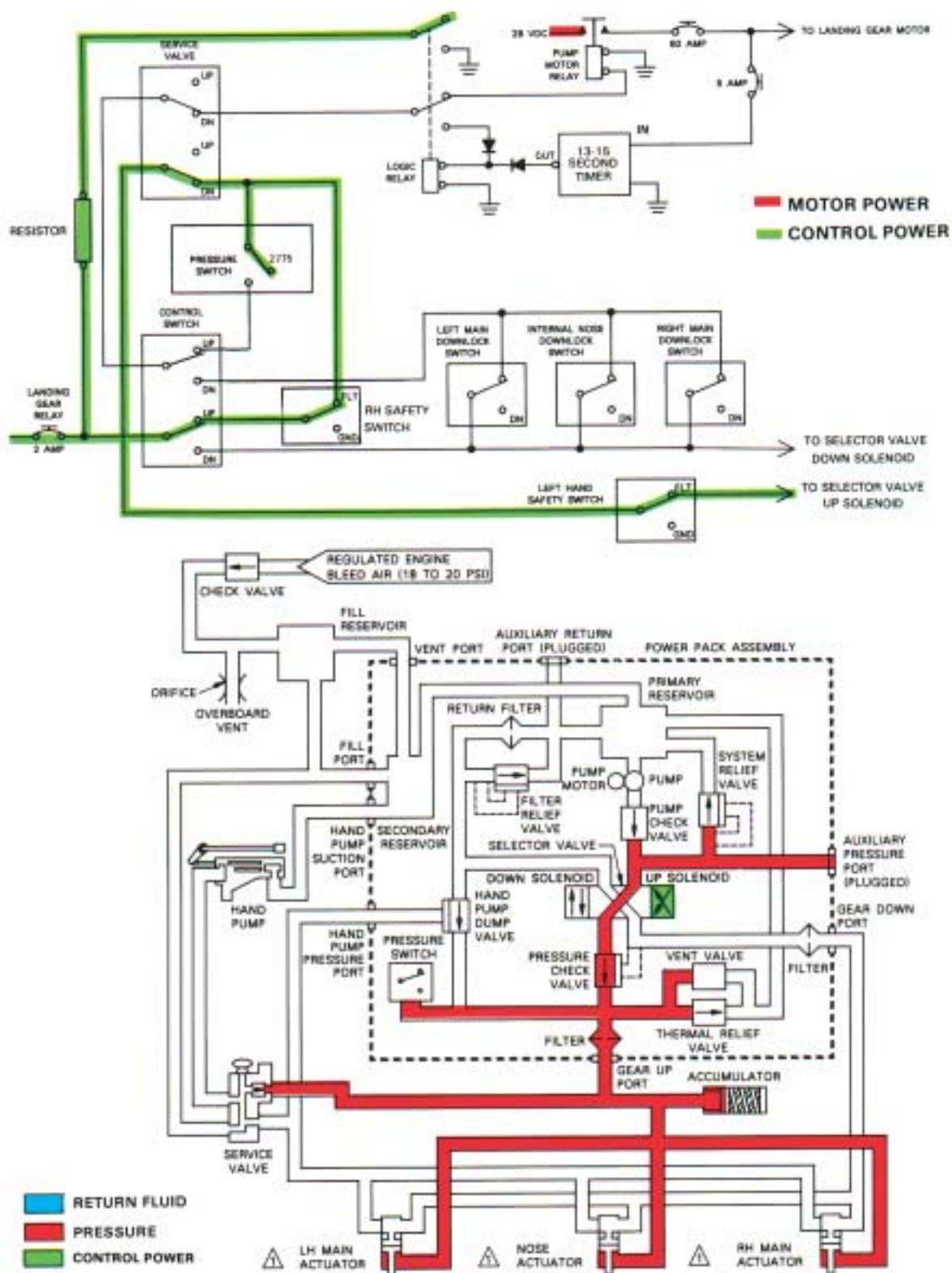


Figure 14-19 Landing Gear Retracted Schematic

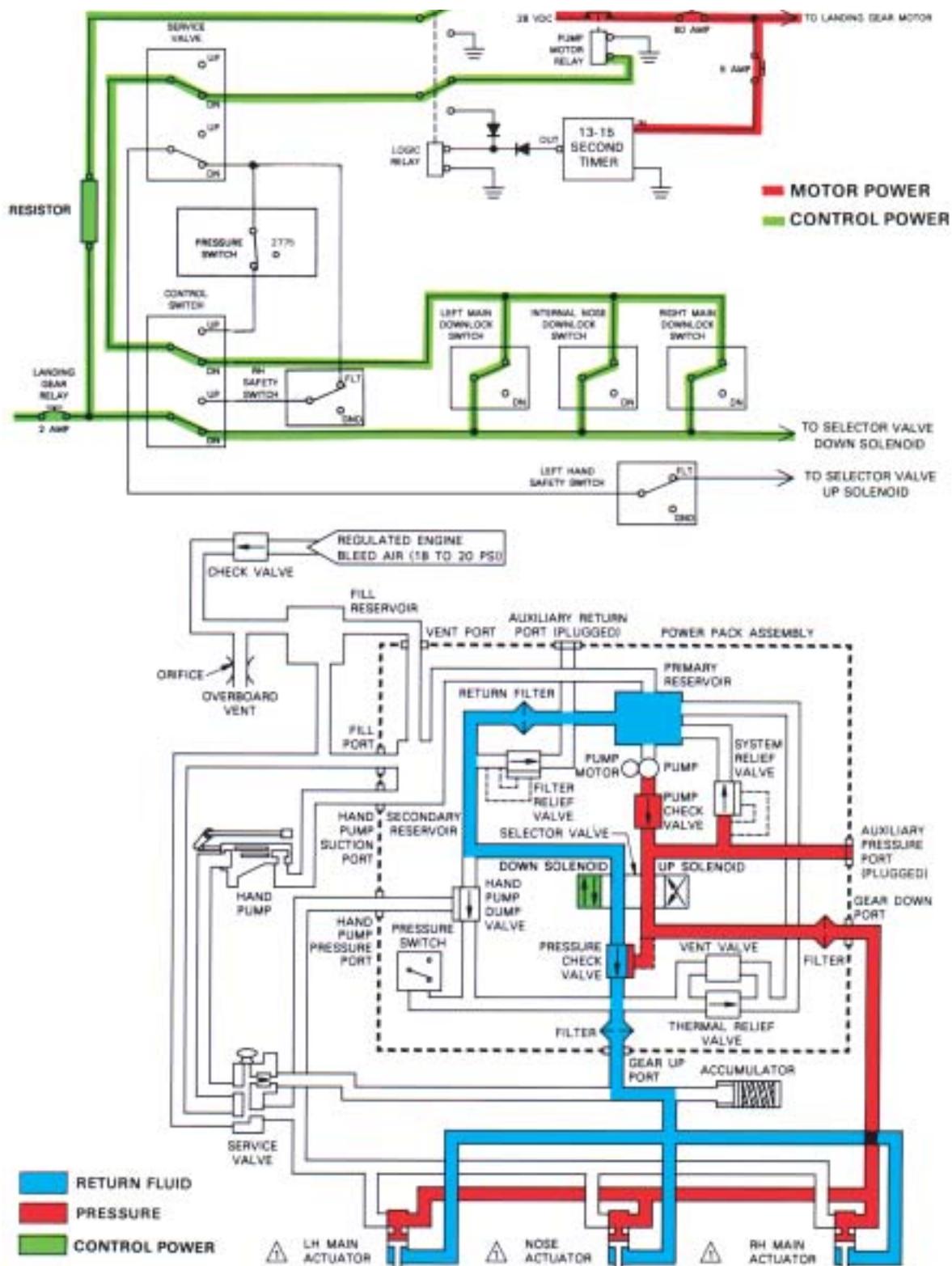


Figure 14-20 Landing Gear Extension Schematic



and locked, the internal downlock mechanism in the nose gear actuator will position its actuator downlock switch to interrupt current to the nose gear part of the pump motor control circuit. A downlock switch on the J-hook assembly of each main gear drag brace will interrupt its part of the pump motor control circuit when the respective main gear is down and locked. The pump motor will continue to operate until all three gear are down and locked. The J-hooks on each main gear drag brace and an internal mechanical lock in the nose gear actuator, lock the gear down.

HAND PUMP EXTENSION

A hand-pump handle, placarded LANDING GEAR ALTERNATE EXTENSION (Figure 14-21), is located on the floor between the pilot's seat and the pedestal. The pump is located under the floor, below the handle, and is used when normal extension of the gear is incomplete.



Figure 14-21 Landing Gear Alternate Extension Placard

To engage the system, pull the LANDING GEAR RELAY circuit breaker (Figure 14-22), located below and to the left of the LDG GEAR CONTROL handle and ensure that the LDG GEAR CONTROL handle on the pilot's sub-panel is in the DN position. Remove the pump

handle from the securing clip and pump the handle up and down until the green NOSE - L - R gear down indicator lights illuminate and further resistance is felt. Stow the pump handle back in the retaining clip when the gear is down and locked (Figure 14-23).

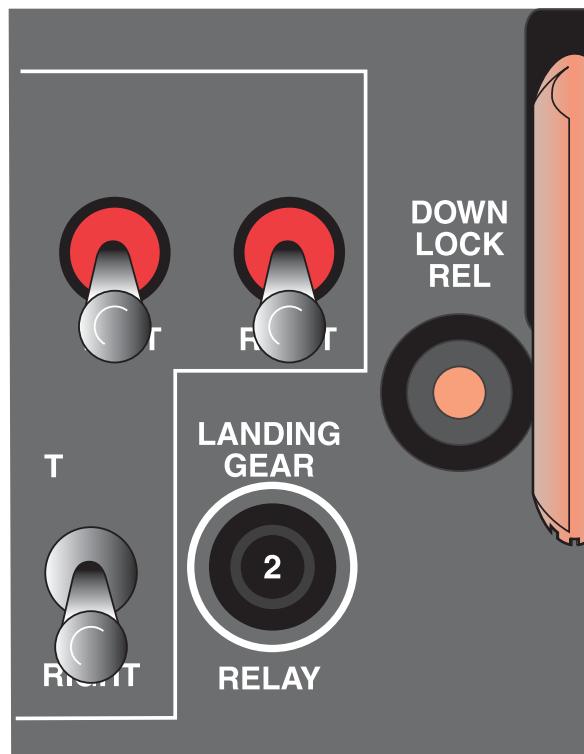


Figure 14-22 Landing Gear Relay Circuit Breaker

WARNING

If for any reason the green GEAR DOWN lights do not illuminate (e.g., in case of an electrical system failure, or in the event an actuator is not locked "down"), continue pumping until sufficient resistance is felt to ensure that the gear is down and locked. Do not stow pump handle. The landing gear cannot be manually retracted in flight.

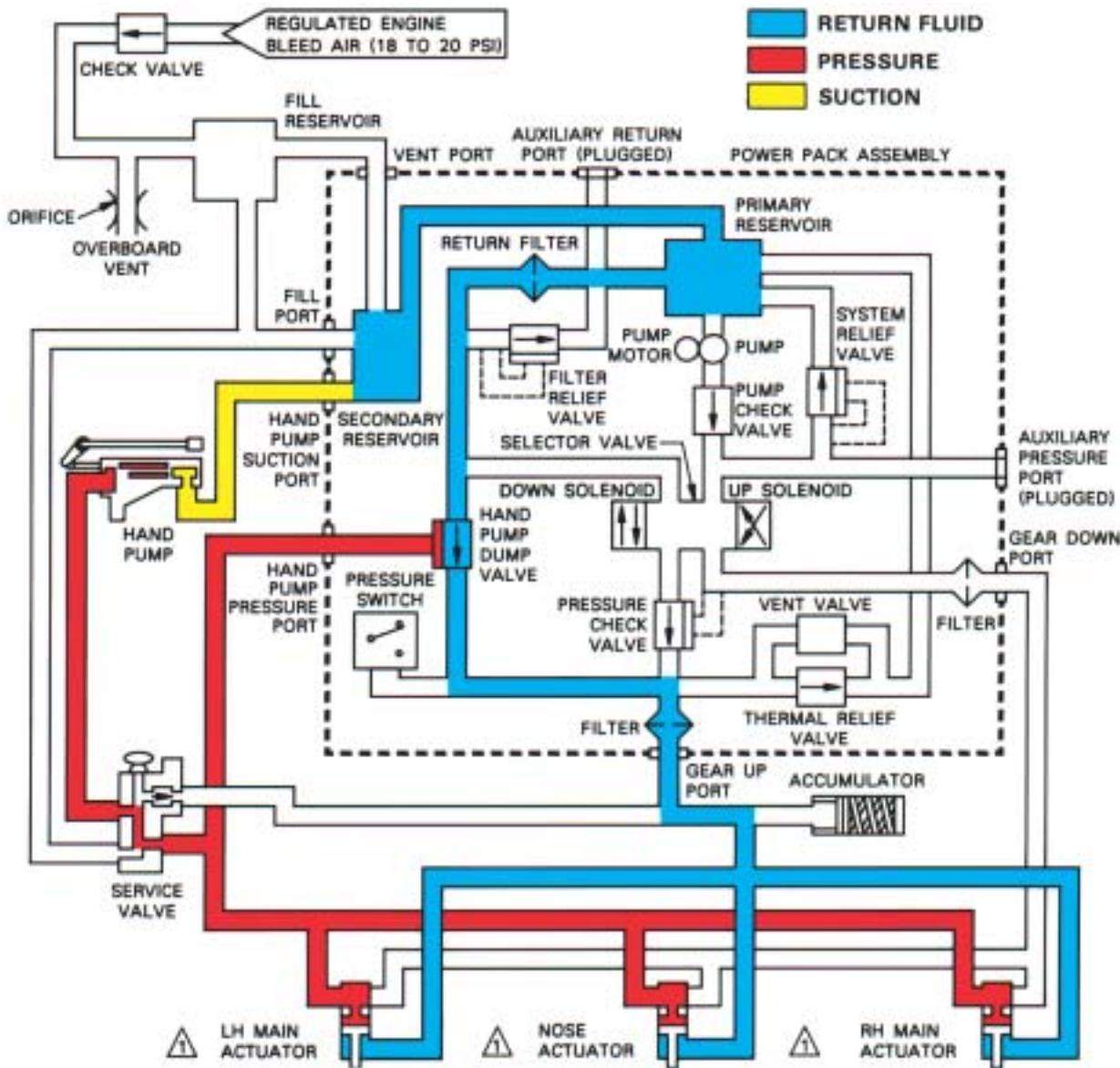
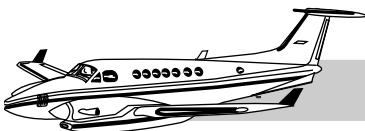


Figure 14-23 Hand Pump Extension

WARNING

After an emergency landing gear extension has been made, do not move any landing gear controls, or reset any switches or circuit breakers until the airplane is on jacks, since the failure may be in the gear-up circuitry or the hydraulic system and the gear might

retract on the ground. The landing gear cannot be retracted manually.

Refer to the "Normal Procedures" section of the *POH* for information pertaining to practice manual landing gear extensions.

After a practice manual extension of the landing gear, the gear may be retracted hydraulically by pushing the **LANDING GEAR RELAY** circuit breaker in and moving the **LDG GEAR CON-**



TROL handle to the UP position provided the manual extension handle is properly stowed.

HAND PUMP RETRACTION

A service valve, located forward of the power pack assembly, may be used in conjunction with the hand pump to raise the gear for maintenance

purposes (Figure 14-24). With the aircraft on jacks and an external electrical power source attached, unlatch the hinged retainer and pull up on the red knob located on top of the service valve. The hand pump can then be used to raise the gear to the desired position. After the required maintenance has been performed, push the red knob down, and use the hand pump to lower the gear.

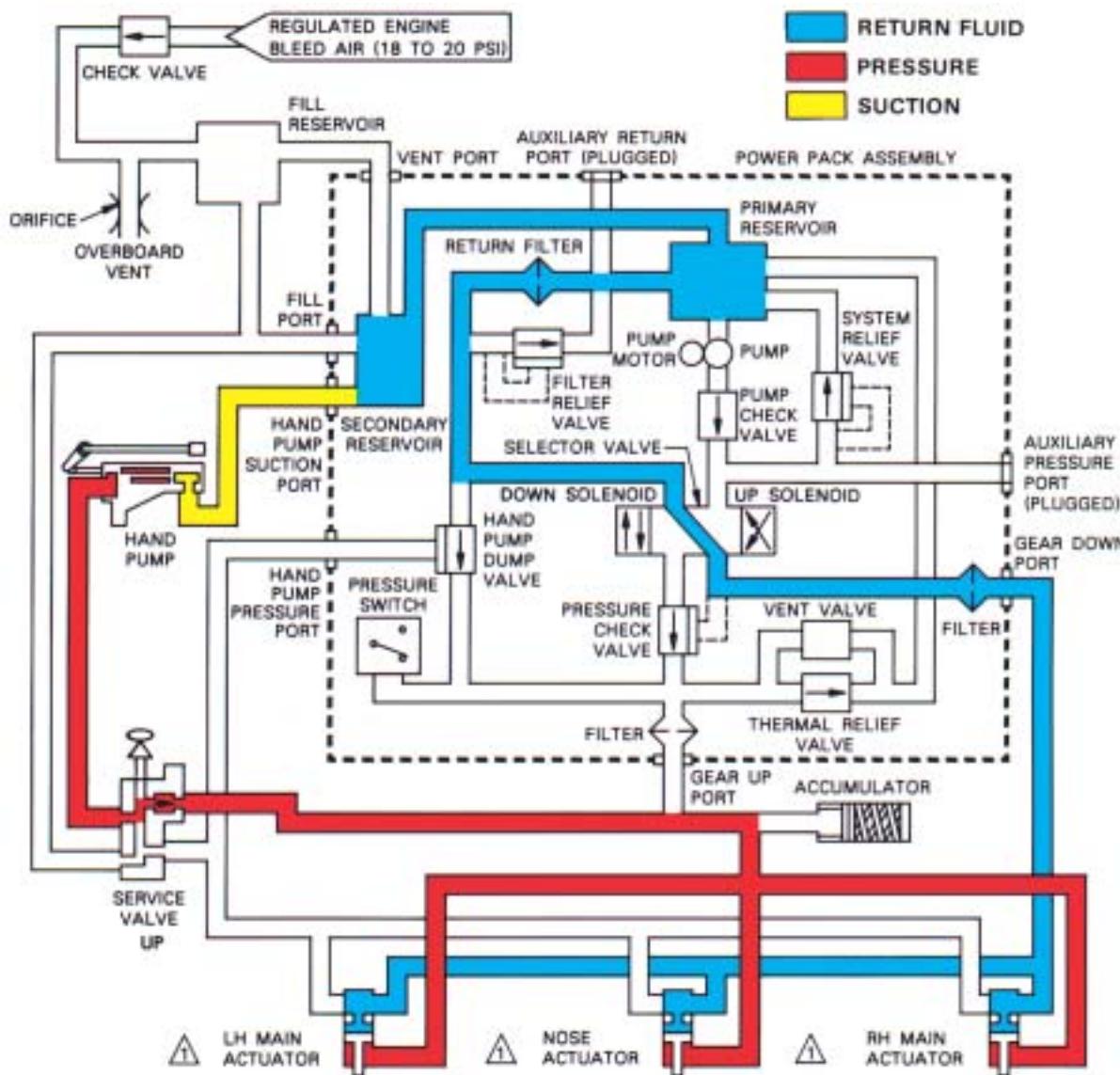


Figure 14-24 Hand Pump Retraction (Maintenance)

**CAUTION**

If the red knob on the service valve is pushed down while the landing gear is retracted with electrical power on and the landing gear control handle DN, the landing gear will extend immediately.

A fill reservoir, located just inboard of the left nacelle and forward of the front spar, contains a cap and dipstick assembly to facilitate maintenance of the system fluid level. A line plumbed to the upper portion of the fill reservoir is routed overboard to act as a vent.

TIRES

The nose landing gear wheel is equipped with a 22 x 6.75 x 10, 8-ply-rated tubeless tire. Each main landing gear wheel is equipped with a 19.5 x 6.75 x 8, 10-ply-rated, tubeless tire. Check the *Pilot's Operating Handbook* for correct tire pressure.

SHOCK STRUTS

Shock struts should always be properly inflated. Do not over- or under-inflate, and never tow or taxi an aircraft when any strut is flat. Correct inflation is placarded on the main and nose struts or refer to the *Maintenance Manual*.

LANDING GEAR OPERATING LIMITS

Landing gear cycles (1 up - 1 down) are limited to one every 5 minutes for a total of 6 cycles followed by a 15 minute cool-down period (Tables 14-2 and 14-3).

Table 14-2 KING AIR 300 AIRSPEEDS

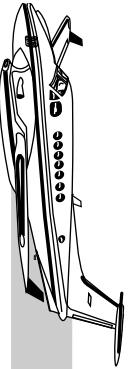
AIRSPEED	KIAS	REMARKS
Maximum gear operating speed (V_{LO}) • Extension • Retraction	181 163	Do not extend or retract the landing gear above these speeds
Maximum gear extended speed (V_{LE})	181	Do not exceed this speed with the landing gear extended.

Table 14-3 KING AIR 350 AIRSPEEDS

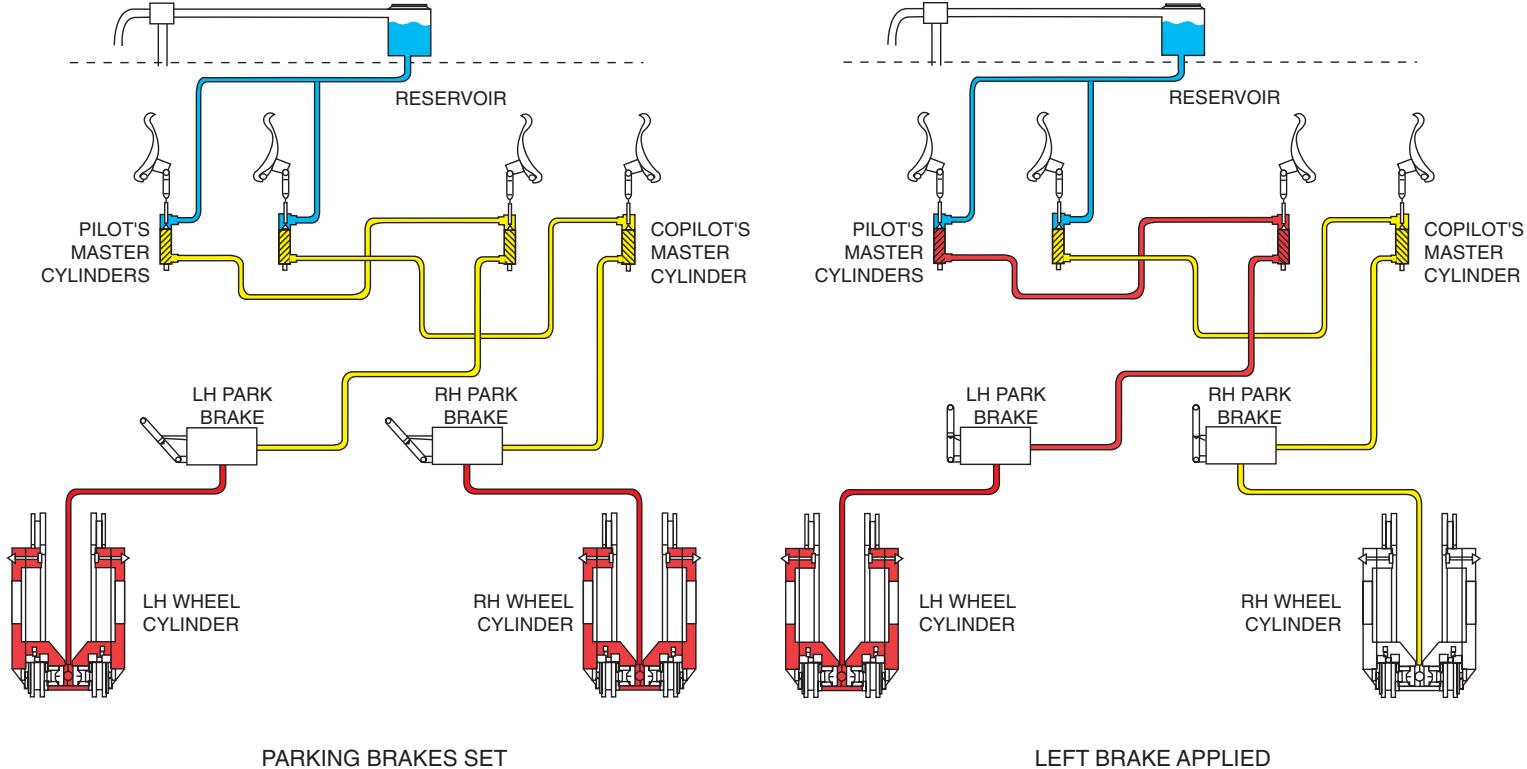
AIRSPEED	KIAS	REMARKS
Maximum gear operating speed (V_{LO}) • Extension • Retraction	184 166	Do not extend or retract the landing gear above these speeds.
Maximum gear extended speed (V_{LE})	184	Do not exceed this speed with the landing gear extended.

BRAKE SYSTEM

The Super King Air 300 and 350 use a non-assisted hydraulic brake system (Figure 14-25). The main landing gear wheels are equipped with multi-disc dual hydraulic brakes. These brakes are actuated by toe pressure on the rudder pedals by either the pilot or copilot. Depression of either set of pedals compresses the piston rod in the master cylinder attached to each pedal. Hydraulic pressure resulting from the movement of the pistons in the master cylinders is transmitted through flexible hoses and fixed aluminum tubing to the disc brake assemblies on the main landing gear. This pressure forces the brake pistons to press against the linings and discs of the brake assembly.



FOR TRAINING PURPOSES ONLY



LEGEND

- FLUID UNDER PRESSURE
- SUPPLY FLUID
- STATIC FLUID

Figure 14-25 Brake System Schematic



Each rudder pedal is attached to its own master cylinder. The pilot's and copilot's right rudder pedals control the brake in the right main landing gear. Similarly, the pilot's and copilot's left rudder pedals control braking in the left main gear. This arrangement allows differential braking for taxiing and maneuvering on the ground.

The dual brakes are plumbed in series. The pilot's master cylinders are plumbed through the copilot's master cylinders, thus allowing either set of pedals to perform the braking action and eliminating the need for shuttle valves. Neither set of brake pedals can override the other.

As with any airplane, proper traction and braking control cannot be expected until the landing gear is carrying the full weight of the airplane. Use extreme care when braking to prevent skidding and the resulting flat sections on tires caused by skidding. Braking should be smooth and even all the way to the end of ground roll.

Three automatic brake adjuster assemblies are installed in each brake piston housing to maintain the proper brake clearance and compensate for brake lining wear. The automatic brake adjusters reduce brake drag, thereby allowing unhampered roll. They also tend to exhibit a softer pedal and a somewhat longer pedal stroke.

PARKING BRAKE

The parking brake utilizes the regular brakes and a set of valves. Dual parking brake valves are installed adjacent to the rudder pedals between the master cylinders of the copilot's rudder pedals and the wheel brakes. The control for the parking brake valves is on the lower left-hand corner of the pilot's left subpanel. To set the parking brake, depress and hold the brake pedals to maintain pressure in the brake system, then depress the button in the center of the parking brake control, and pull out the control handle. This procedure closes both parking brake valves simultaneously.

NOTE

When preparing to use the parking brake, ensure the aircraft has come to a complete stop.

The parking brake valves should retain the pressure previously pumped into the system. The parking brake can be released from either the pilot's or copilot's side by depressing the brake pedals (to equalize the pressure on both sides of the valves) and simultaneously pushing the parking brake handle in (to allow the parking brake valves to open).

To avoid damage to the parking brake system, tires and landing gear, the parking brake should be released and wheel chocks installed if the airplane is to be left unattended. There are two reasons for this: ambient temperature changes can expand or contract the brake fluid causing excessive brake pressure or brake release, and the airplane may need to be moved by ground personnel in the pilot's absence.

NORMAL USE OF THE BRAKES

The wheel brakes can be highly effective in stopping the airplane when the situation requires; however, normal operation usually requires minimal brake use.

The propellers are more effective than the brakes immediately after touchdown, during landing rollout. As the airplane decelerates, the brakes become most effective. Since most landing situations are not runway length critical, a smooth, comfortable deceleration can be achieved with almost exclusive use of the propellers until passing approximately 40 knots ground speed. At that time, light wheel brake pressure will provide sufficient deceleration for a comfortable transition to taxiing clear of the runway.

CAUTION

When runway length is critical during landing, maximum braking techniques should be employed. Consult the "Performance" section of the *POH* for requirements and landing distances.

Speed can normally be controlled primarily with the use of the propellers in the GROUND FINE range during taxi. The brakes should be used when necessary for supplemental control. Turning can be accomplished by nosewheel steering with an occa-



sional assist by tapping on one brake to help initiate turns of shorter radius. When minimum radius turns are required, intermittent heavy pressure may be necessary on the inboard brake.

CAUTION

Do not hold steady pressure on the inboard brake while turning. It may induce a twisting action on the strut, inflicting serious damage to the strut assembly.

BRAKE SERVICE

Brake fluid is supplied to the master cylinders from a reservoir accessible through the nose avionics compartment door (Figure 14-26). The brake fluid reservoir is located on the upper corner of the left side of the nose avionics compartment.

Brake system servicing is limited primarily to maintaining the hydraulic fluid level in the reservoir. A dipstick is provided for measuring the fluid level. When the reservoir is low on fluid, add a sufficient quantity of MIL-H-5606 hydraulic fluid to fill the reservoir to the full mark on the dipstick. Check all hydraulic connections for signs of seepage and correct if necessary.

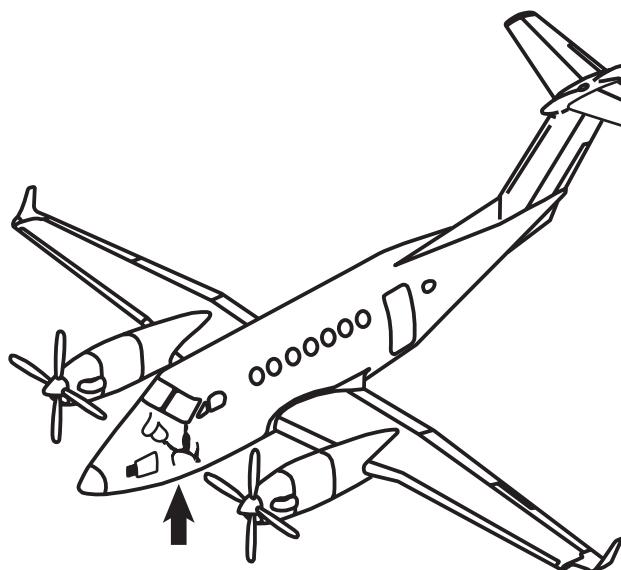


Figure 14-26 Brake Fluid Reservoir



BRAKE WEAR LIMITS

To check the brakes for wear, measure the distance between the segmented carrier and lining assembly and the piston housing (Figure 14-27). When this distance (with the parking brake set) measures 0.200 inch or more, the brake is ready for a lining inspection per the *Maintenance Manual*.

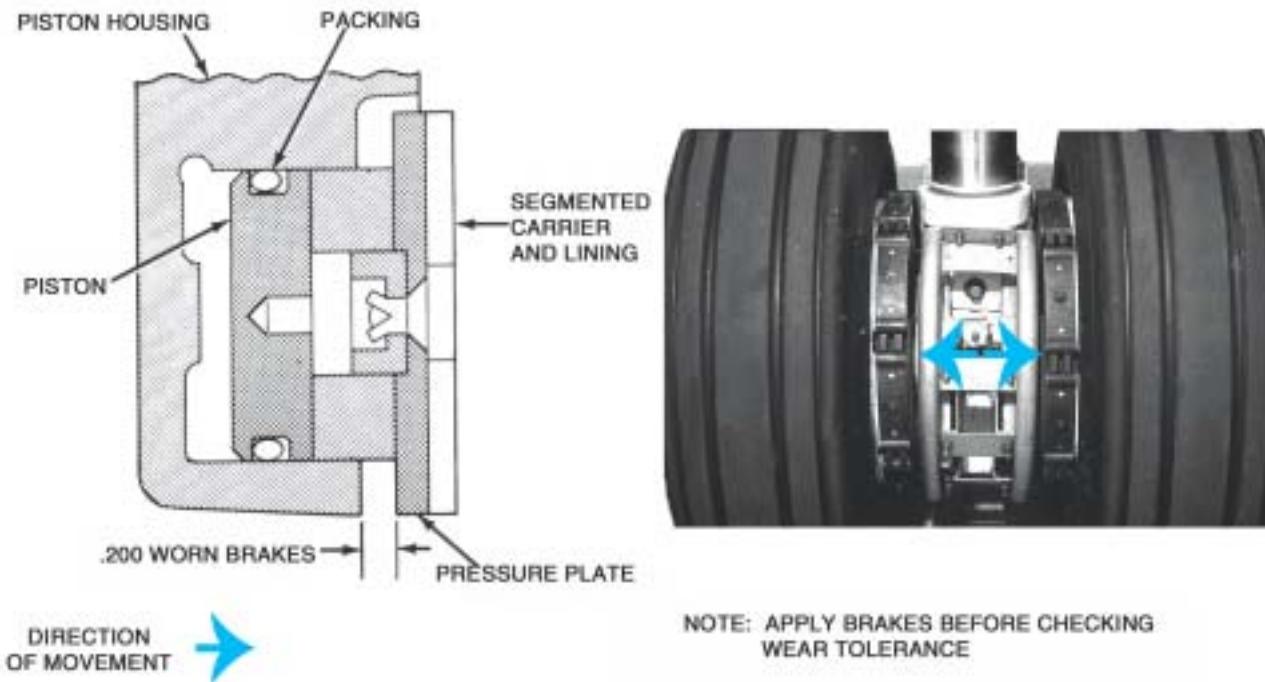


Figure 14-27 Brake Wear Diagram

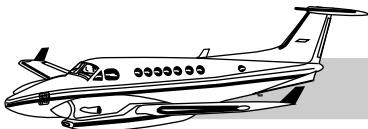


CHAPTER 15

FLIGHT CONTROLS

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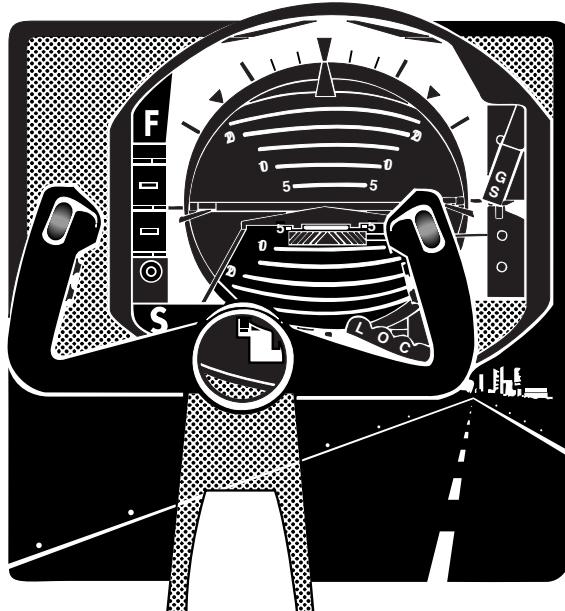
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CHAPTER 15

FLIGHT CONTROLS



INTRODUCTION

Familiarization with the flap system operation and limits is necessary to provide optimum performance in takeoff, approach, and landing modes. This training unit identifies and describes flap action so the pilot will understand proper operation, system controls, and limits.

DESCRIPTION

The Flap System section of the training manual presents a description and discussion of the flap system.

The four-segment Fowler-type flap system, its controls and limits are considered with reference to operation as outlined in the *Pilot's Operating Handbook*.



FLAP SYSTEM

The Super King Air 300 and 350 have a four-segment, Fowler-type flap system. The flaps, with

two segments on each wing, are driven by an electric motor through a gearbox mounted on the forward side of the rear spar (Figure 15-1).

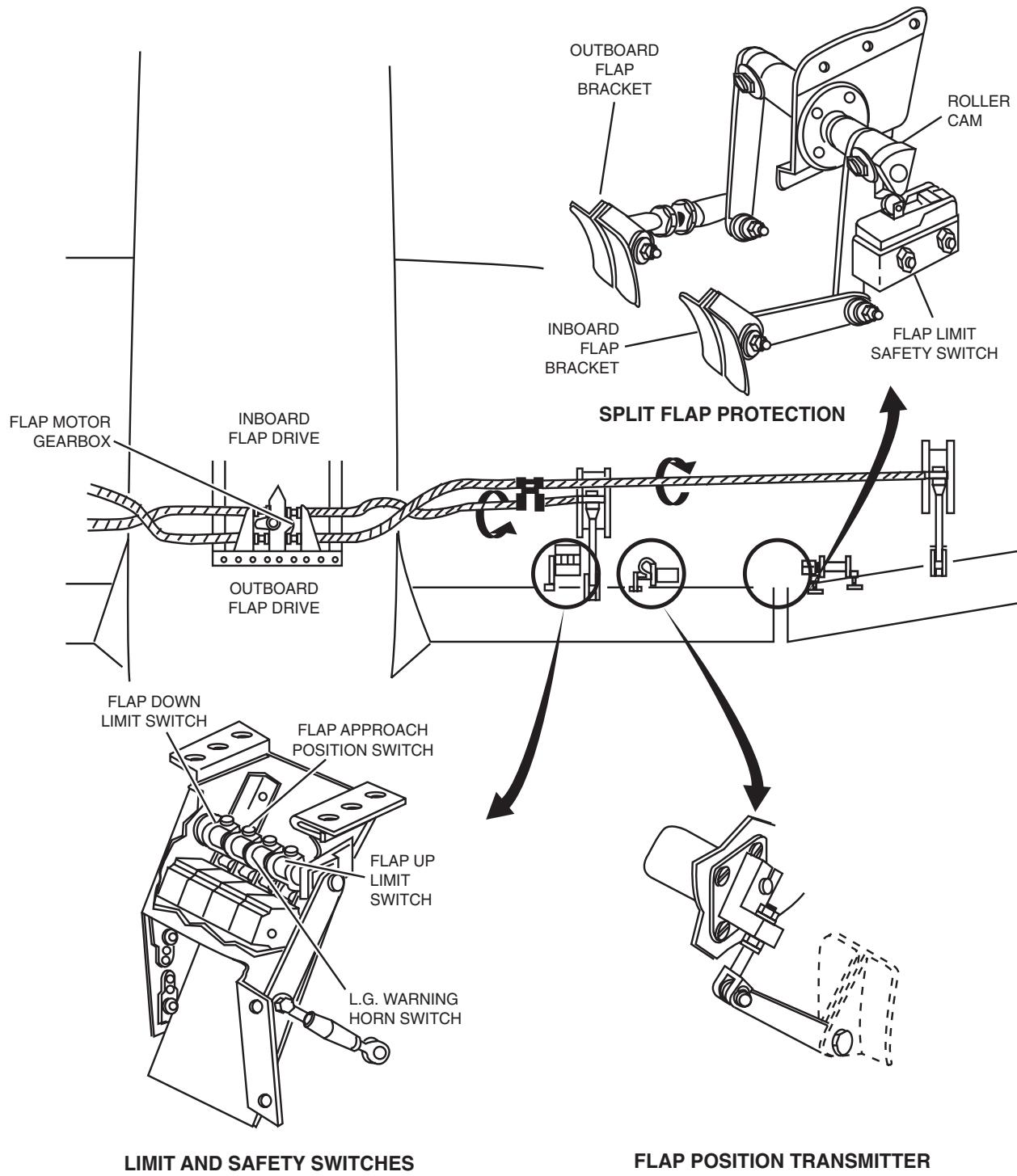


Figure 15-1 Flap Control System



The gearbox drives four flexible driveshafts connected to jackscrew actuators at each flap segment. A safety mechanism is provided to disconnect power to the electric motor in the event of a failure, which may cause one segment to be three to six degrees out of phase with the others. The motor incorporates a dynamic braking system. Two sets of motor windings help prevent overtravel of the flaps.

The flaps are controlled by a sliding lever handle and switch located just below the condition levers on the pedestal (Figure 15-2). The flap control has a position detent provided to select 40% flaps for takeoff or approach. Full flap deflection, or 100%, is equal to approximately 35° of flap travel.



Figure 15-2 Flap Control Lever

Flap deflection, from 0% (up) to 100% (down), is registered in percentage on an electric indicator on top of the pedestal, below the caution/advisory annunciator panel (Figure 15-3). The indicator is operated by a potentiometer driven by the right inboard flap segment. Flap position limit switches are also driven by the right inboard flap.

On the King Air 300, from the UP position to the APPROACH position, the flaps cannot be stopped at an intermediate point. Between APPROACH and DOWN, the flaps may be stopped as desired by moving the handle to

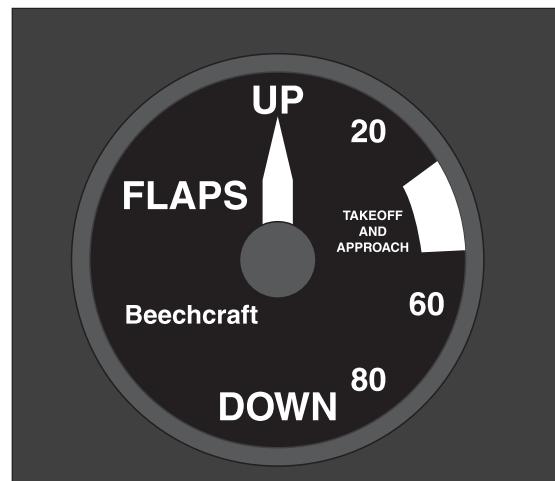


Figure 15-3 Flap Position Indicator

DOWN until the flaps have moved to the desired position, then moving the handle back to APPROACH. Similarly, the flaps may be raised to any position between DOWN and APPROACH by raising the handle to UP until the desired setting is reached, then returning it to the APPROACH detent. The APPROACH detent acts as a stop for any position greater than 40%. Moving the handle from DOWN to APPROACH will not retract the flaps. When the handle is moved from APPROACH to UP, the flaps retract completely and cannot be stopped at an intermediate position.

On the King Air 350, the flaps may be selected to the UP, APPROACH, or DOWN position. The flaps cannot be stopped at an intermediate point between these positions. Whenever one of the 3 positions is selected, the flaps will move to the selected position.

The flap motor power circuit is protected by a 20-ampere circuit breaker, placarded "FLAP MOTOR," located on the copilot's circuit breaker panel (Figure 15-4). A 5-ampere circuit breaker, placarded "FLAP IND & CONTROL," for the flap control circuit is also located on this panel.

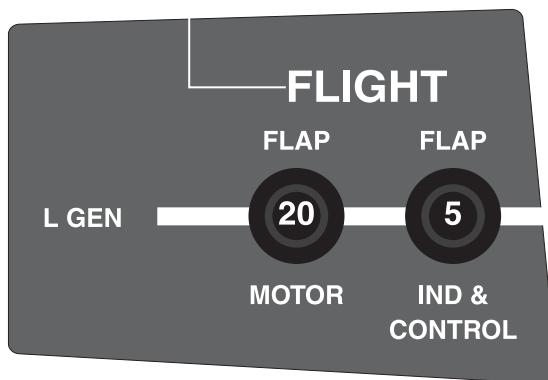


Figure 15-4 Flap System Circuit Breakers

LANDING GEAR WARNING SYSTEM

The landing gear warning system is provided to warn the pilot that the landing gear is not down and locked during specific flight regimes. The warning horn will sound continuously when the flaps are lowered beyond the APPROACH (40%) position, regardless of the power lever setting, until the landing gear is extended or the flaps are retracted. The effect of flap position on the landing gear warning system is discussed more completely in the Landing Gear section of this training manual.

ASYMMETRICAL FLAP PROTECTION

Split-flap protection is provided by an asymmetrical flap switch system. This switch is rigged to shut off the flap motor for any out-of-phase condition of approximately three to six degrees between the adjacent flap segments on each wing. This switch is spring-loaded to the normally-open position, but is rigged so that the roller cam holds the switch in its momentary (closed) position. This provides electrical continuity to the flap motor when the outboard and inboard flap segments on both sides are parallel and in phase with one another.

FLAP AIRSPEED LIMITS

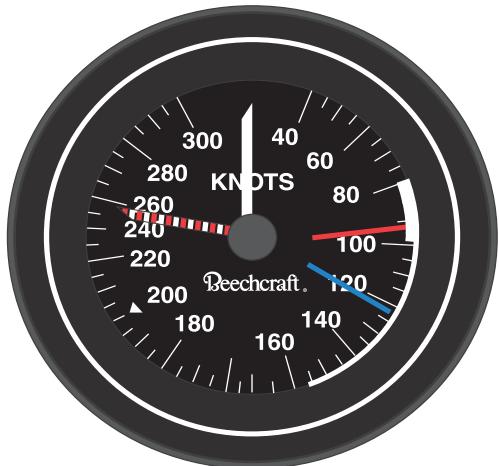
Airspeed indicator markings (Figure 15-5) show the maximum speeds and operating range of the flaps (V_{FE}). The white triangle indicates maximum speed permissible with flaps extended to approach. The upper limit of the narrow white arc is the maximum speed permissible with flaps extended beyond approach. The upper limit of the wide white arc is the flaps-up, zero-thrust stall speed. Normally flaps should be extended when flying below this airspeed. The lower limit of the wide white arc is the flaps-down, zero-thrust stall speed.

NOTE

All illustration needles may not reflect normal indications.

Lowering the flaps will produce these results:

- Attitude—Slight nose up
- Airspeed—Reduced
- Stall speed—Lowered
- Trim—Very little (if any) adjustment required to maintain altitude (once initial ballooning effect is accounted for).

**300 AIRSPEED INDICATOR**

259 KIAS MAXIMUM (V_{MO}) or .58 MACH

NOTE
MAXIMUM ALLOWABLE AIRSPEED (RED STRIPED)
POINTER IS SELF ADJUSTING WITH ALTITUDE

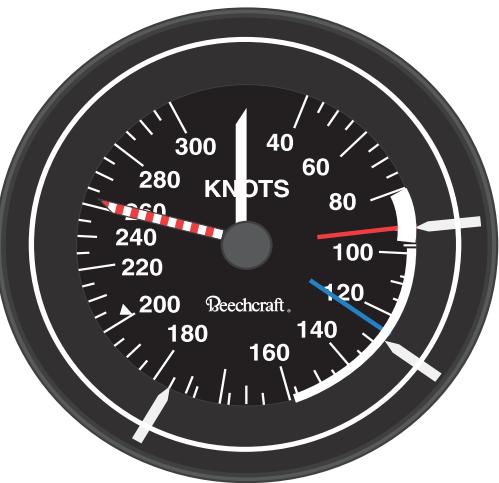
92 KIAS MINIMUM SINGLE-ENGINE CONTROL
SPEED (V_{MCA})

122 KIAS SINGLE ENGINE BEST RATE OF
CLIMB SPEED (V_{YSE})

81-100 KIAS FULL FLAP OPERATING RANGE

100-157 KIAS (NARROW) FLAPS UP STALL SPEED
(V_S) TO MAXIMUM LIMIT FLAPS MORE THAN 40%

200 KIAS MAXIMUM FLAPS TO/AT APPROACH SPEED

**350 AIRSPEED INDICATOR**

263 KIAS MAXIMUM (V_{MO}) or .58 MACH

NOTE
MAXIMUM ALLOWABLE AIRSPEED (RED STRIPED)
POINTER IS SELF ADJUSTING WITH ALTITUDE

94 KIAS MINIMUM SINGLE-ENGINE CONTROL
SPEED (V_{MCA})

125 KIAS SINGLE ENGINE BEST RATE OF
CLIMB SPEED (V_{YSE})

81-96 KIAS FULL FLAP OPERATING RANGE

96-158 KIAS (NARROW) FLAPS UP STALL SPEED
(V_S) TO MAXIMUM LIMIT FLAPS MORE THAN 40%

202 KIAS MAXIMUM FLAPS TO/AT APPROACH
SPEED

Figure 15-5 Airspeed Limits



CHAPTER 16

AVIONICS

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CHAPTER 16

AVIONICS



INTRODUCTION

Avionics systems, as a vital part of the airplane, are becoming more sophisticated and complex. These systems lighten the pilot load, particularly during IFR operations. It is therefore important for the flight crew to understand how the various NAV/COMM systems function, and how to use them effectively. This section describes the standard avionics installation and how it operates.

DESCRIPTION

Avionics controls, along with the weather radar, are mounted on an isolation panel in the center of the instrument panel, easily accessible to the pilot or copilot (Figure 16-1).

Individual audio switches, across the top of the panel, control audio to the speakers and headphones. There are separate sets of controls for the pilot and copilot so that each can select audio from any NAV or COMM receiver.

In its standard configuration, the Beechcraft Super King Air 300 and 350 come equipped with a complete IFR avionics package, including weather radar, compass system, dual audio systems, and dual flight instrumentation. The flight deck can be optionally equipped with virtually any combination of avionics desired.

Most King Air 350 and some King Air 300 aircraft are equipped with the Collins FCS-65C flight control system category 1 with the Collins EFIS-85B (14) (three-tube system) electronic flight instrument system. Other systems available

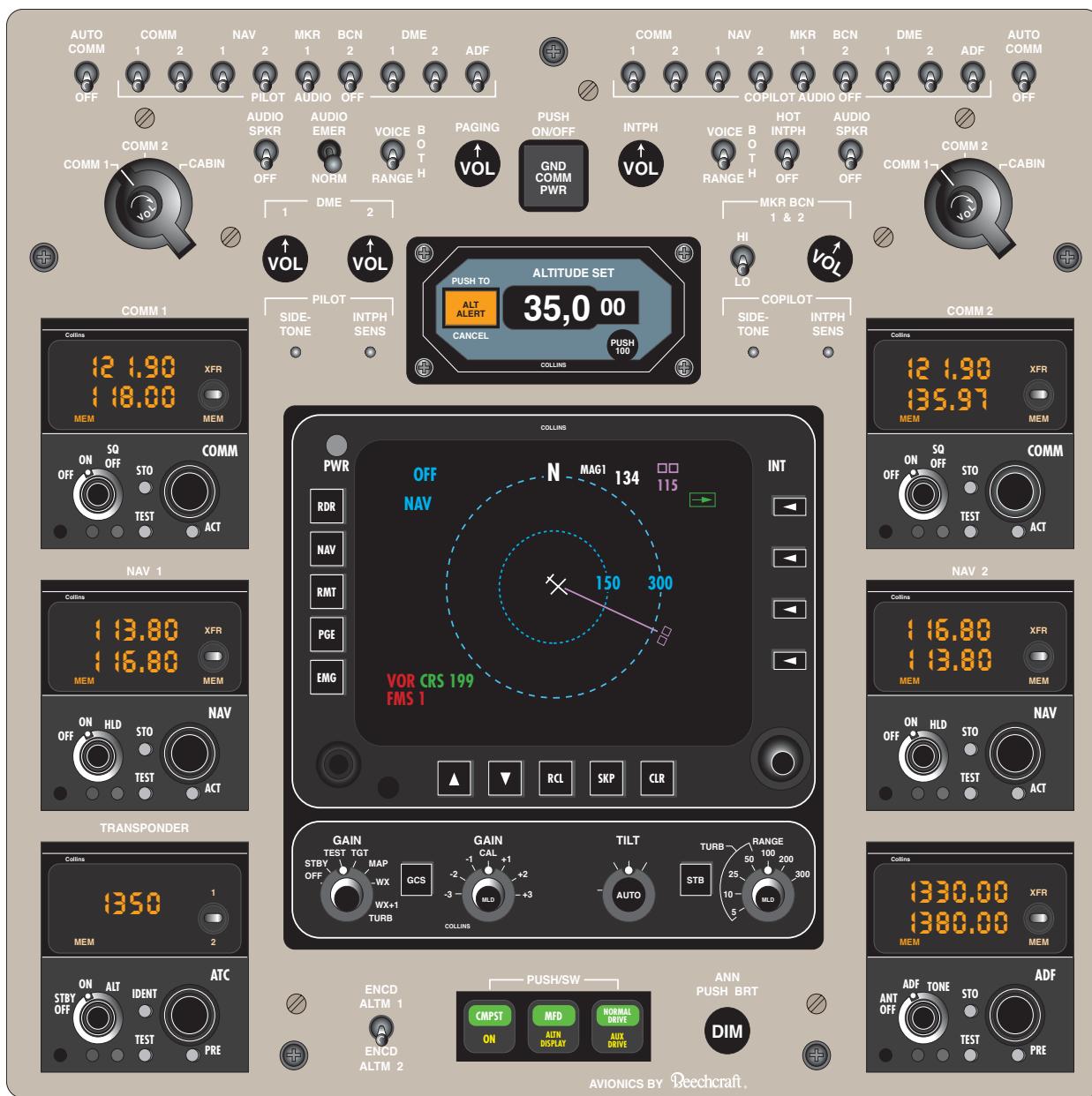
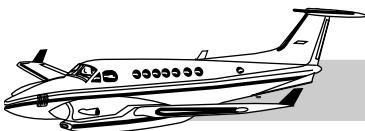


Figure 16-1 NAV/COMM Control Panel

include the King KFC 400 automatic flight control system Category 1 with Bendix EFS 10 electronic flight instrument system (three- or four-tube EFIS) and the Sperry SPZ-4000 automatic flight control system Category 1 and Sperry EDZ-601/EDZ-801/EDZ-603/EDZ-803 electronic flight instrument system (EFIS).

Most King Air 350 aircraft are equipped with the universal UNS-1 (B+, C, D, K, or M) flight management systems. Other aircraft are equipped with GPS units from other manufacturers.

Refer to the *Pilot's Operating Manual* supplements and the manufacturer's operating manuals for your specific operating instructions.



AVIONICS POWER DISTRIBUTION

All avionics equipment may be turned on and off by the AVIONICS MASTER PWR switch (Figure 16-2). In the event that this switch fails, power may be restored by pulling the avionics master circuit breaker, located in the upper right-hand corner of the main circuit-breaker panel.

The Super King Air 300 and 350 have three avionics buses (Figure 16-3) to feed DC power to the various types of avionics equipment. To determine specifically what equipment is being fed from a specific bus or power source, refer to the wiring diagram entitled "DC Power Distribution" which is supplied with each airplane. There are, however, some general rules of thumb which usually apply.

For example: Due to the large number of individual receivers and transmitters, a Beech avionics master switch is installed to provide an easy method for switching them ON or OFF.

As shown in Figure 16-4, the avionics master switch, when placed in the OFF position, applies power to the relays, causing the contacts to be

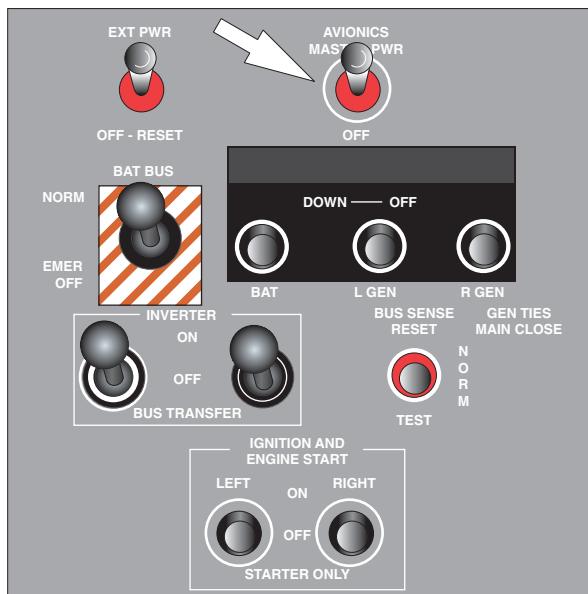


Figure 16-2 Avionics Master Switch

disconnected between the 28-volt source buses and the three avionics power buses.

NOTE

Figures 16-5 and Figure 16-6 show methods of restoring power to avionics equipment.

It should now be apparent that if the avionics switch were to fail in the labeled OFF position, one would pull the circuit breaker labeled MASTER POWER (Figure 16-6) in the avionics circuit-breaker section in order to restore power to the avionics equipment: Alternately, when the avionics master switch is placed in the ON position (Figure 16-5), it is actually cutting power to the relays, allowing the contacts to close, delivering power to the avionics buses.

- Items numbered one (e.g., COMM 1, NAV 1, etc.) are fed by the No. 1 avionics bus, which in turn is fed from the electrical system triple-fed bus. It is important to note that in the event of a dual genera-

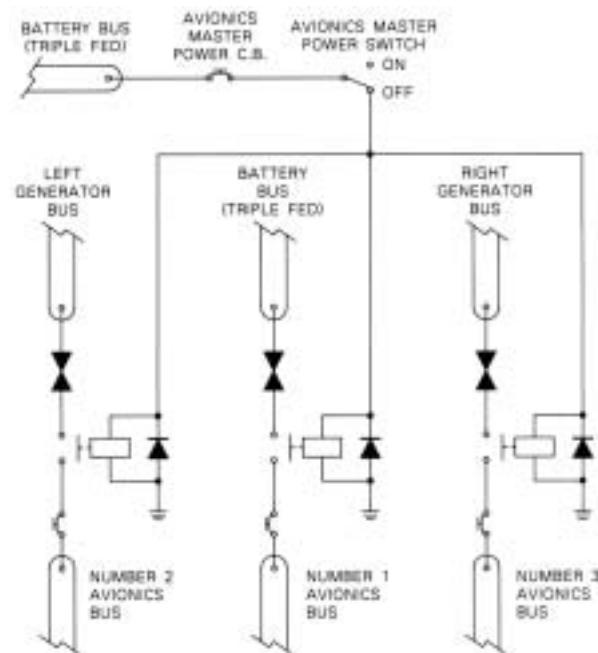


Figure 16-3 Avionics Buses

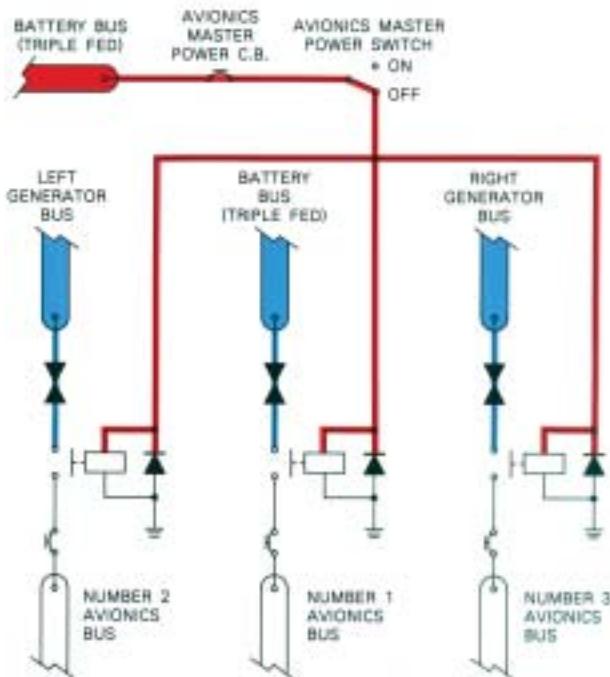


Figure 16-4 Avionics Master Switch OFF

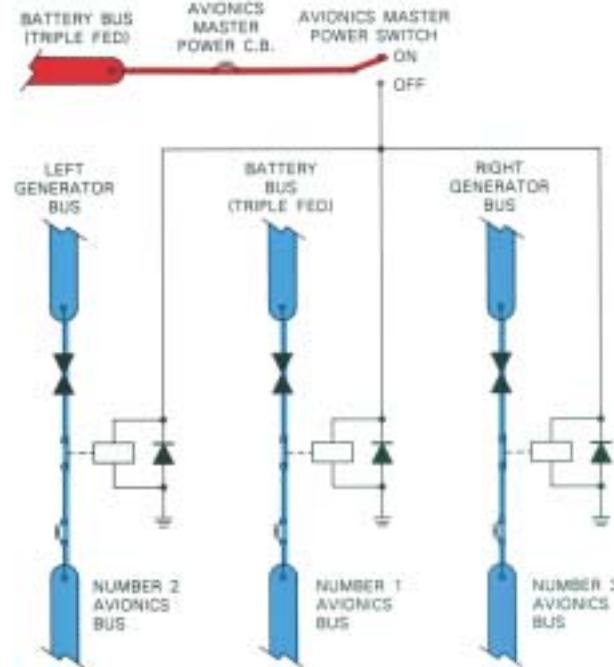


Figure 16-5 Avionics Master Switch ON

tor failure, the items fed by the No. 1 avionics bus would continue to operate for a limited period of time, being fed directly by the battery.

- Items numbered two (e.g., COMM 2, NAV 2, etc.) are fed by the No. 2 avionics bus. The No. 2 avionics bus is fed by the left generator bus.
- Additional avionics items which are not fed by the previous buses are fed by the No. 3 avionics bus. The No. 3 avionics bus is fed by the right generator bus.

When the triple-fed bus and both generator buses are powered, all three avionics buses may be powered.

However, assume the need to make a quick ground check of COMM 2, prior to starting engines, and without an APU connected. In this

situation, manually close the bus tie with the GEN TIE switch located on the pilot's outboard subpanel.

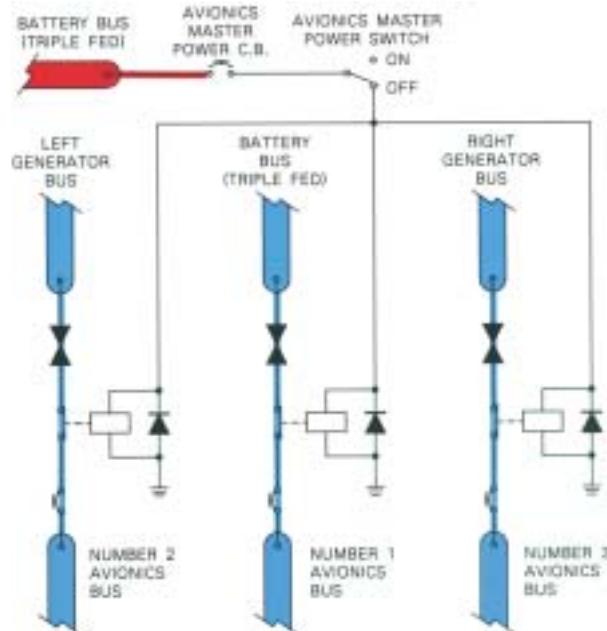


Figure 16-6 Alternate Method of Feeding Power to the Avionics Bus



As a general rule of thumb, an APU should be considered essential for running avionics equipment on the ground. For electronic flight instrument system (EFIS) equipped airplanes, the avionics equipment and one of the inverters require approximately 50 amperes of current from the battery. This amount of current drain would deplete the battery in a short period of time.

Also, especially with EFIS equipment installed, it is desirable to have the avionics nose compartment doors removed to allow sufficient cross ventilation and cooling of the equipment—particularly during practice sessions with the avionics equipment which exceed 15 minutes in duration.

COLLINS PRO LINE II EQUIPMENT

The most common radio package installed on the Super King Air currently is the Collins Pro Line II package. Although other systems are available, since this package is the most common, it is the only one addressed in this section.

A new series of Collins CTL control heads is used for the NAV, COMM, ADF, and TRANSPONDER. The ADF-60 and TDR-90 (ADF and TRANSPONDER units respectively), however, are retained from the earlier Collins Pro Line System.

The Pro Line II family presently consists of a VHF COMM (VHF22), a VOR/LOC/GS/MB receiver (VIR-32), and a DME (DME42). These units employ many state-of-the-art features, including extensive self-diagnostic capabilities and multiple frequency storage. Some of the features of this equipment will be described here. For additional information, see the current *Collins Pro Line II Pilot's Guide*.

The COMM and NAV units have many features in common therefore, we will use the features of the NAV (VIR-32/CTL-32) as a building block for the COMM, which will be described next. Features common to both will be described under

the NAV explanation; differences will be pointed out under the COMM explanation.

NAVIGATION SYSTEMS

VIR-32 Navigation Receiver

General

Two VIR-32 navigation receiver systems are installed in the B300 aircraft. The VIR-32 navigation receiver provides 200, 50-kHz spaced VOR/LOC channel receiver from 108.00 MHz through 117.95 MHz, and 40 GS receiver channels automatically paired with localizer channels. It also includes a marker beacon receiver.

The VIR-32 provides VOR, LOC, and GS deviation outputs, NAV valid discretes, magnetic bearing to the station, to/from information, marker beacon lamp signals, and VOR/LOC and marker beacon audio outputs.

Display Interface

The No. 1 NAV receiver (Figure 16-7) supplies digital data to the display processor unit (DPU) and the multifunction processor unit (MPU). The No. 2 NAV receiver supplies digital data to the MPU. This NAV data is used for display of the navigation information on the pilot's EHSI, EADI and MFD. The pilot's EFIS system can display the onside NAV information and the cross-side NAV information via the cross-side data buses between the DPU and the MPU. The display control panel (DCP) is used to select the navigation receiver used as the source for display. The No. 2 NAV receiver also provides analog signals to the copilot's HSI processor unit (HPU) for display on the copilot's EHSI. Each NAV receiver also provides a bearing signal to each RMI for display of VOR station bearing. Also, the bearing signal from the No. 1 NAV is supplied to the HPU for No. 1 NAV bearing display on the copilot's EHSI.

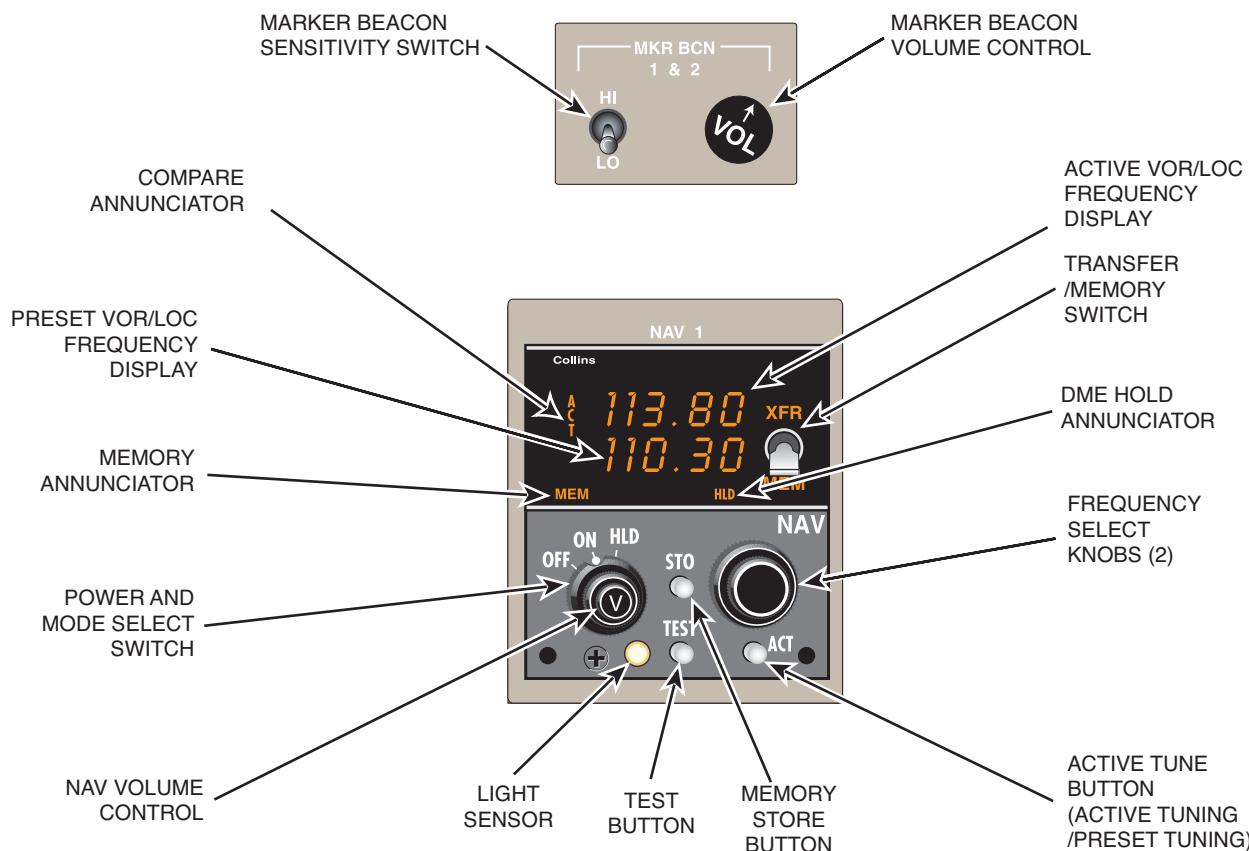


Figure 16-7 VIR-32 NAV Receiver Controls/Displays

Locations

The VIR-32 receivers are located in the right avionics bay (No. 1 on the bottom and No. 2 on the top). There are several antennas: the two VOR/localizer antennas are mounted on either side of the vertical stabilizer, the single glideslope antenna is just below the radar antenna; and the marker beacon antenna is on the bottom of the airplane below the cockpit area. Both VIR-32 receivers share all the antennas.

Self Test

During self test, the VIR-32 provides VOR, ILS, and marker beacon test outputs. When in self test, the CTL-32 modulates the intensity of the upper and lower displays from minimum to maximum to annunciate that self test is in progress. The DME will also be placed in self test at the same time.

The following paragraphs provide the procedures required and the results to be expected when performing the self test.

VOR Self Test

- Select a VOR frequency on the CTL-32 NAV control (108.20 MHz will do. A specific frequency is not required for test.). A signal on frequency will not interfere with the Self test. Select VOR1 or VOR2 (as required) as the active course sensor on the EHSI. Rotate the course select knob to approximately 0°.
- Push and hold the TEST button on the CTL-32.
- The active course sensor VOR1 or VOR2 annunciator on the EHSI will turn red.



- After approximately two seconds, the VOR1 or VOR2 annunciator will turn green, the EHSI lateral deviation bar will approximately center, and a TO indication will appear. The RMI pointers connected to the VIR-32 will indicate approximately 0° magnetic bearing.
- Release the TEST button. The VIR-32 will return to normal operation after approximately 15 seconds, even if the TEST button is held.

ILS (Localizer and Glideslope) Self Test

- Select a localizer frequency on the CTL-32 NAV Control. 108.10 MHz will do. A specific frequency is not required for test. Select LOC1 or LOC2 (as required) as the active course sensor on the EHSI.
- Push and hold the TEST button on the CTL-32.
- The active course sensor LOC1 or LOC2 annunciator on the EHSI will turn red and the red GS flag will come into view.
- After approximately three seconds, the LOC1 or LOC2 annunciator will turn green and the GS flag will go out of view, the EHSI lateral deviation bar will deflect right approximately two-thirds of full scale, and the glideslope pointer will deflect down approximately two-thirds of full scale.
- Release the TEST button. The VIR-32 will return to normal operation after approximately 15 seconds, even if the TEST button is held.

Marker Beacon Self Test

The marker beacon assembly is tested automatically when the TEST button on the CTL-32 is pushed and either a VOR or localizer frequency is selected. For No. 1 NAV receiver, proper opera-

tion of the marker beacon assembly is indicated by all three-marker annunciators on the EADI cycling through in order. For No. 2 NAV receiver, the indication will be the three marker annunciators flickering at 30Hz. In addition, a tone will also be present in the marker beacon audio output.

Diagnostic Display

An extensive self test diagnostic routine is also initiated in the VIR-32 when the TEST button on the CTL-32 NAV control is pushed (Figure 16-8).

The diagnostic routines are intended as an extension of the self test capability. The operator should first observe the deviation indicators and associated flags for the proper self test responses. If an out-of-limit condition exists, then the problem may be verified in more detail by the diagnostics.

For the first two or three seconds immediately after the TEST button on the CTL-32 is pushed, a 2-digit diagnostic code may be displayed in the lower window based on the conditions existing immediately before the TEST button was pushed.

If an out-of-limit condition is detected during self test, that 2-digit code will also be displayed on the CTL-32 along with the word “DIAG” (diagnostic), “FLAG,” or four dashes (“----”) in the upper window. The four dashes will be displayed along with the code “00”, indicating normal operation, no trouble found.

“FLAG” will be displayed along with a two-digit code when something is abnormal, but a failure has not occurred (i.e., low signal level, etc.). “DIAG” is displayed along with a two-digit code to indicate a failure has been detected in the VIR-32.

Completion of self test is indicated when the NAV control displays either the normal active and preset frequencies in the upper and lower windows, respectively, or a two-digit code.

Diagnostic data can also be read on the EFIS system when the EFIS system diagnostic CSDB data page is displayed.

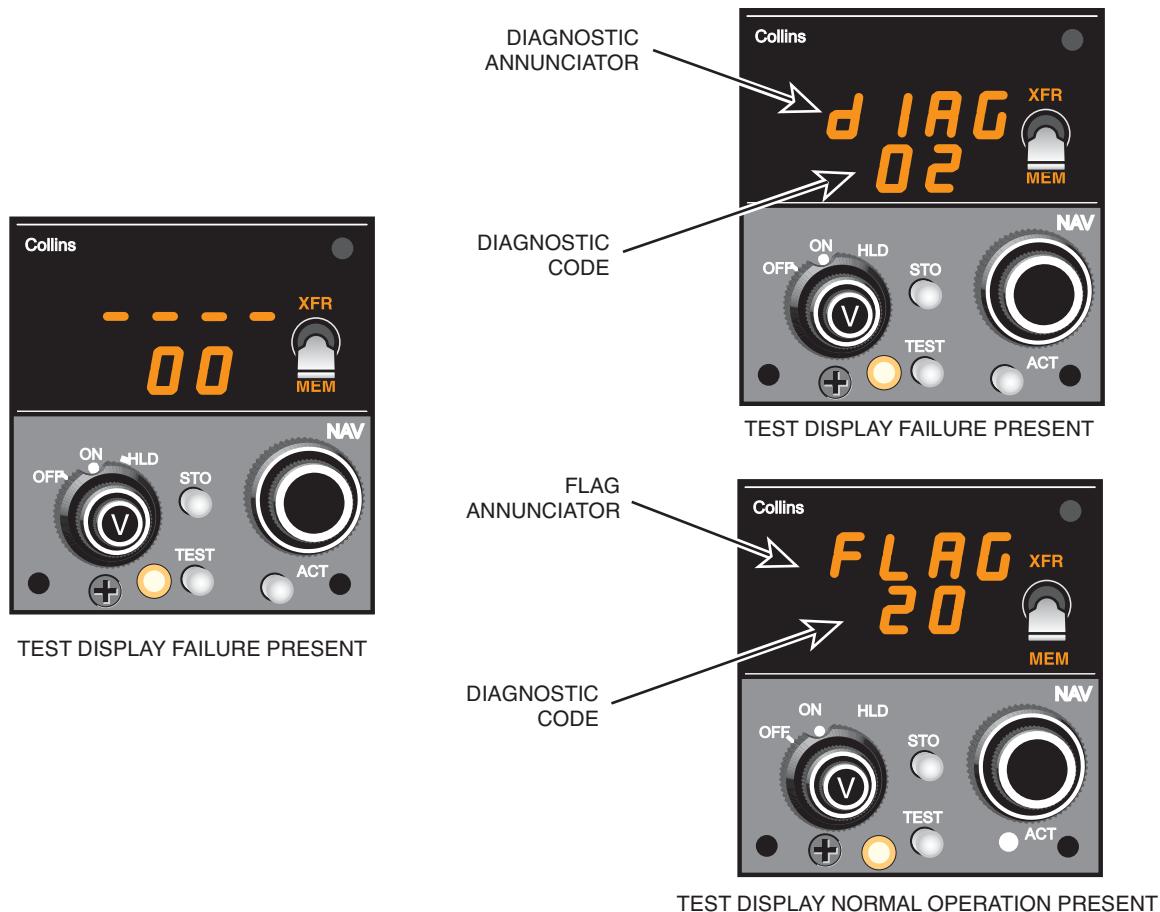


Figure 16-8 VIR-32 NAV Receiver Self Test Displays

Table 16-1 CTL-32 NAV/DME CONTROL, CONTROLS AND INDICATIONS

CONTROL OR DISPLAY	FUNCTION/DESCRIPTION
Active frequency display	The active frequency (frequency to which the VIR-32 is tuned) and diagnostic messages are displayed (upper window) in the upper window.
Preset frequency display	The preset (inactive) frequency and diagnostic messages are displayed in the lower window (lower window).
Compare annunciator	ACT momentarily illuminates when frequencies are being changed. If ACT continues to flash, the actual tuned frequency is not identical to the frequency shown in the active display.



Table 16-1 CTL-32 NAV/DME CONTROL, CONTROLS AND INDICATIONS (CONT.)

CONTROL OR DISPLAY	FUNCTION/DESCRIPTION
Annunciators	The NAV control contains MEM (memory) and HLD (hold) annunciators.
MEM	The MEM annunciator illuminates whenever a frequency is displayed in the lower window.
HLD	The HLD annunciator indicates that the DME is held to the active frequency at time of selection. The upper window displays the NAV frequency and the lower window displays the DME hold frequency. Tuning of active frequencies can take place during this time.
Volume control	The volume control is concentric with the power and mode switch and controls the VIR-32 audio output volume.
Power and mode switch	The NAV control power and mode switch contains three detented positions. The positions are: OFF/ON/HLD. The OFF/ON positions switch system power. The HLD position allows the NAV frequency to be changed but holds the DME to the current active NAV frequency.
Light sensor	The built-in light sensor automatically controls the display brightness. The ANN PUSH BRT control knob/push button can be used to override the automatic dim controls and force the display to go to full bright.
XFR/MEM switch	This switch is a three-position, spring-loaded toggle switch. When moved to the XFR position, the preset frequency is transferred up to the active display and the NAV/DME retunes. The previously active frequency becomes the new preset frequency and is displayed in the lower window. When this switch is moved to the MEM position, one of the four stacked memory frequencies is loaded into the preset display. Successive pushes cycle the four memory frequencies through the display (...2, 3, 4, 1, 2, 3....).
Frequency select knobs	Two concentric knobs select the preset or active frequency displays. The larger knob changes the two digits to the left of the decimal point in 1-MHz steps. The smaller knob changes the two digits to the right of the decimal point in 0.05-MHz steps. Frequencies roll over at the upper and lower limits. The two frequency select knobs are independent of each other such that the upper and lower limit rollover of the 0.1-MHz digit will not cause the 1.0-MHz digit to change.
ACT button	Push the ACT button for approximately two seconds to enable the frequency select knobs to directly retune the VIR-32 and DME. The bottom window will display dashes and the upper window will continue to display the active frequency. Push the ACT button a second time to return the control to the normal 2-display tune/preset mode of operation. The active tuning feature is not affected by power removal. If active tuning is selected (one push of the ACT button) and power is removed from the control, active tuning will still be enabled the next time power is reapplied to the control.

**Table 16-1 CTL-32 NAV/DME CONTROL, CONTROLS AND INDICATIONS (CONT.)**

CONTROL OR DISPLAY	FUNCTION/DESCRIPTION
STO button	The STO button allows up to four preset frequencies to be selected and entered into the control's nonvolatile memory. To store a frequency, simply toggle the MEM switch until the upper window displays the desired channel number (CH 1 through CH 4), rotate the frequency select knobs until the lower window displays the frequency to be stored, and push the STO button twice within five seconds. After approximately five seconds, the control will return to the normal two-display tune/preset mode of operation.
TEST button	Push the TEST button to initiate the radio self testself test diagnostic routine. (In the case of the VIR-32 NAV receiver, self testself test is active only while the TEST button is pushed or about 15 seconds maximum. In the case of the DME-42 transceiver, the self test routine requires about 10 seconds for completion.)

Table 16-2 VIR-32 SELF TEST DIAGNOSTIC CODES

CODE	DESCRIPTION	DISPOSITION
ALL MODES		
00	No fault found	
02	RAM test failed	Unit unusable (μ P problem)
003	No serial data to unit	Unit unusable (CTL problem)
04	No serial freq word	Unit unusable (CTL problem)
05	Invalid NAV freq	CTL may be tuned to DME channel
06	Microprocessor fault	Unit unusable
09	Microprocessor fault	Unit unusable
10	Microprocessor fault	Unit unusable
11	A/D fault	Unit unusable
12	A/D failed accuracy test	Unit unusable
13	+13-VDC power supply fault	Unit unusable
14	-13-VDC power supply fault	Unit unusable
15	VOR sin \pm /LOC D/A fault	VOR/LOC unusable
16	VOR cos \pm /GS D/A fault	VOR/GS unusable
VOR MODE ONLY		
17	VOR smo unlocked	VOR receiver inoperative
18	VOR AFC not locked	VOR unusable (no rf signal)



Table 16-2 VIR-32 SELF TEST DIAGNOSTIC CODES (CONT.)

CODE	DESCRIPTION	DISPOSITION
19	Low 30-Hz reference signal	VOR unusable
20	Low 30-Hz reference signal	VOR unusable
21	400-Hz power supply unusable	VOR OBI and OBS unusable
22	OBI sin out of tolerance	VOR OBI and OBS unusable
23	OBI cos out of tolerance	VOR OBI and OBS unusable
24	OBS return out of tolerance	VOR OBS unusable
ILS MODE ONLY		
25	LOC smo unlocked	LOC receiver inoperative
26	LOC signal level low	LOC unusable
27	LOC device out of tolerance	LOC device unreliable
28	28 GS smo unlocked	GS receiver inoperative
29	GS signal too low	GS unusable
30	GS device out of tolerance	GS device unreliable
ALL MODES		
32	Marker beacon fault	Observe marker lamps for fault

DME-42 DISTANCE MEASURING EQUIPMENT

General

The B300 aircraft have two DME-42 systems with IND-42 DME indicators installed (Figures 16-9 and 16-10). The DME is a three-channel transceiver that measures slant range (line-of-sight) distance from the airplane to a ground station, computes relative closure rate and time to station, and decodes the station identifier from the 252 DME channels available. The DME-42 transceiver is controlled by a CTL-32 control shared with the VIR-32 NAV receiver. The DME audio output is applied to the airplane audio system, and digital bus outputs are applied to the IND-42 indicators and the EFIS systems.

Display Interface

Each DME transceiver's information is displayed on an IND-42 and the onside EFIS system EHSI (Figures 16-11 and 16-12). The No. 1 DME transceiver sends information to the pilot's IND-42A DME indicator and to the display processor unit (DPU) and multifunction processor unit (MPU). The No. 2 DME transceiver sends information to the copilot's IND-42A, the HSI processor unit (HPU), and to the MPU. Each EHSI display shows the DME information in the top section of that display. The pilot's EFIS system can display the onside DME information and the cross-side NAV information via the cross-side data buses between the DPU and the MPU. The display control panel (DCP) is used to select the NAV receiver as the source for display and, in doing so, the associated DME receiver is also used as the source for display.

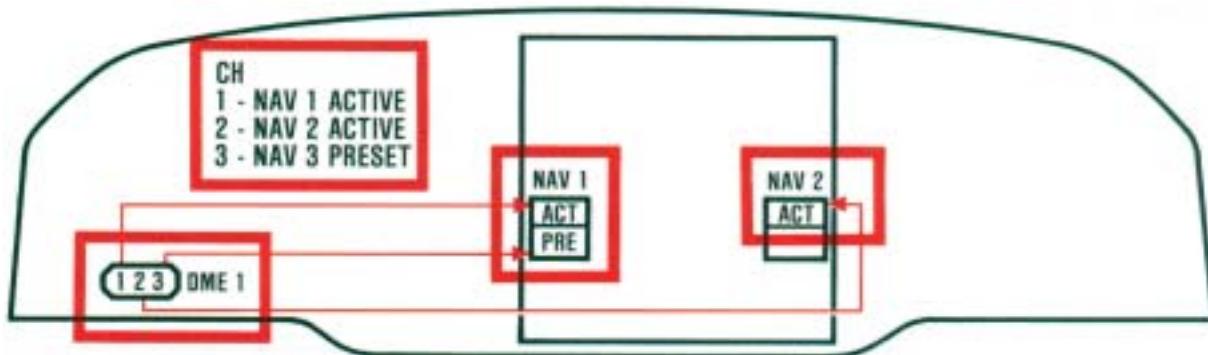


Figure 16-9 Single DME Installation (Typical)

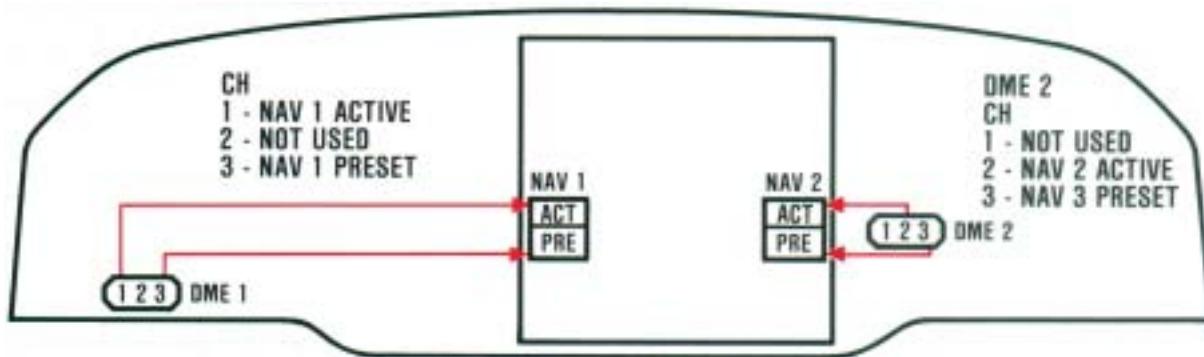


Figure 16-10 Dual DME Installation (Typical)

Self Test

- Turn power on to the DME, NAV and EFIS systems.
- Ensure that VOR or LOC is selected as the NAV source on the EHSI.
- On the CTL-32, select ON. Use the frequency select knobs and select the frequency for any DME or VORTAC station that is within range.
- Read the distance to the station on the IND-42A/C and the left side of the EHSI

display. Read the station ID next/above to the distance to station display.

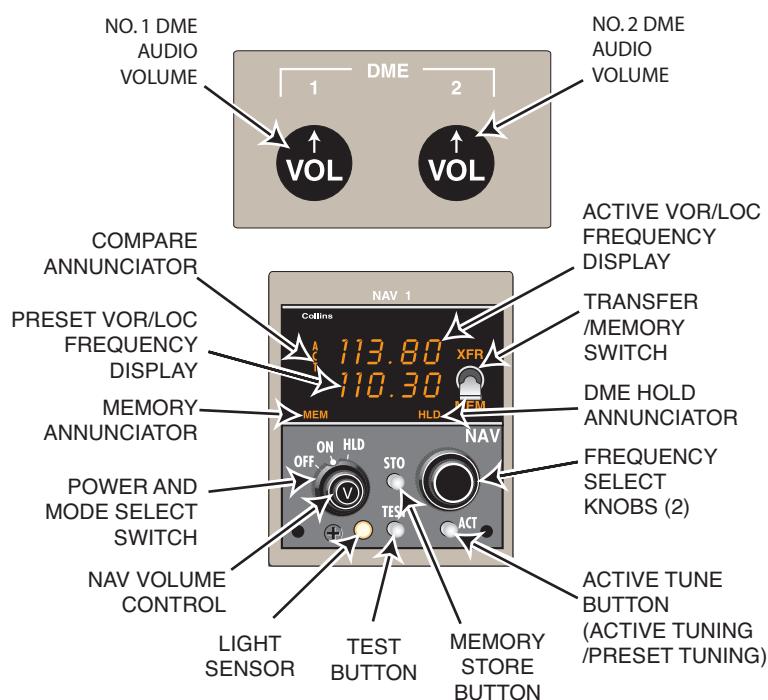
NOTE

The DME can require at least 30 seconds, and as much as two minutes, to properly decode the station ident.

- On the CTL-32, push TEST.

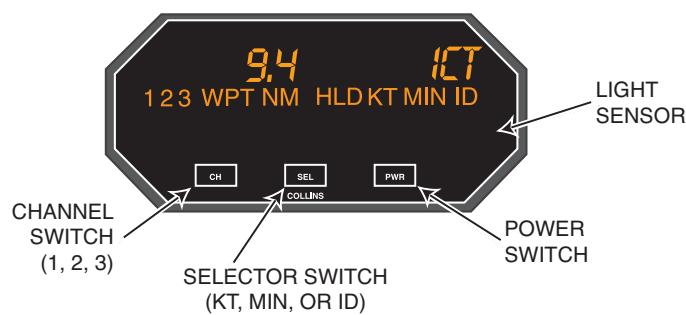
On the IND-42 the following happens:

- Initially the IND-42A/C display modulates in intensity between minimum and maximum brightness.



CTL-32 NAV CONTROL

Figure 16-11 DME-42 Transceiver Controls/Displays



IND-42C (COPILOT'S DISPLAY)

Figure 16-12 DME-42 Systems IND-42 Display



- LH display on IND-42A/C shows a test distance of 100.0 (nmi). After about 10 seconds, the RH display shows an “AOK.”
- Listen to the DME audio and note that the audio is a Morse code A O K.
- Push SEL to annunciate KT and read 100 (knots) in the RH window.
- Push SEL again to annunciate MIN and read 60 (minutes) on right side of the window.
- Push SEL again to annunciate ID again.

NOTE

If the 10-second self test expires before reaching this point, select TST again and continue with the test.

On the pilot’s EHSI display the following happens:

- Dashes will appear in the DME and TTG/GSP display
- The left side of the EHSI display shows a test distance of 100 (nmi). After about 10 seconds, the left side of the EHSI display shows an “AOK.”
- Listen to the DME audio and note that audio is a Morse code A O K.
- Rotate the navigation select switch on the DSP-84 display select panel until

the groundspeed position (GSP) reads 100 (knots) on the right side of the EHSI display.

- Rotate the navigation select switch to the time to-go (TTG) position and read 60 (minutes) on right side of the EHSI display.

On the copilot’s EHSI display the following will appear:

- The DME display will read 0.0 with a small T next to display to indicate the DME is in test.

NOTE

If the 10-second self test expires before reaching this point, select TST again and continue with the test.

If there are any detected faults in the system on the IND-42A/C, a diagnostic code will appear in place of the “AOK” display. The EFIS display will only show dashes for a fault. Refer to the following pages for a description of the diagnostic codes.

Locations

The two DME receivers are found on the avionics rack located in the aft tail section. No. 1 is in the lower unit; No. 2 is the upper unit.

The DME antennas are easy to confuse with the transponder antennas. The DME No. 1 antenna is the farthest aft L-Band antenna and the DME No. 2 antenna is the farthest forward L-Band antenna on the bottom of the fuselage.

**Table 16-3 DME-42 SELF TEST DIAGNOSTIC CODES**

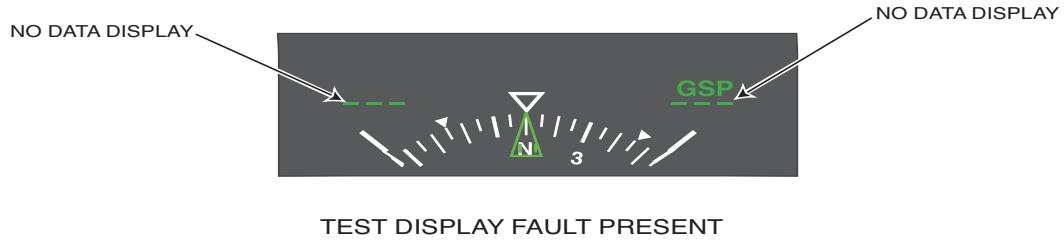
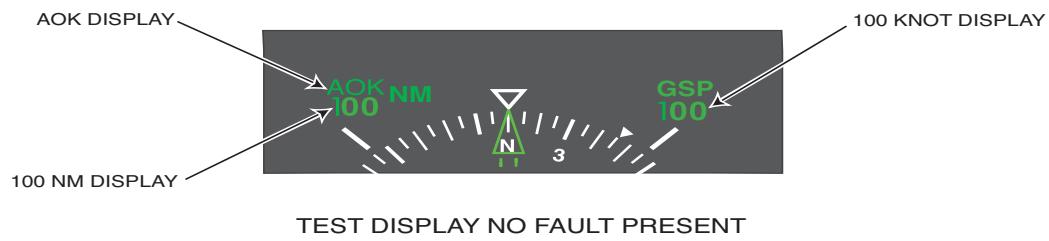
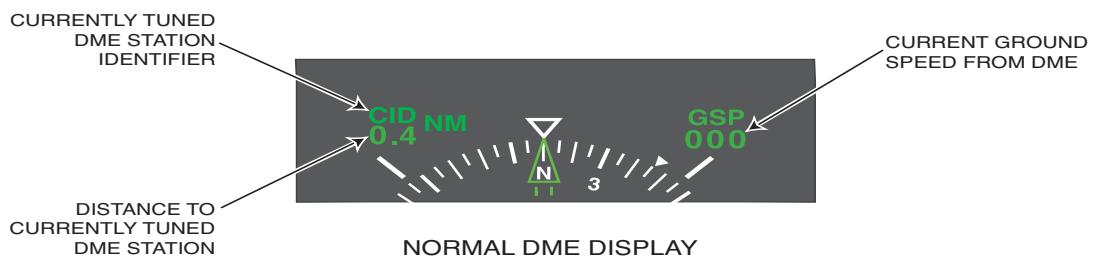
DME-42 DIAGNOSTICS	
CODE	DESCRIPTION
01	Power supply
02	Synthesizer
03	Transmitter
04	Video processor
05	Receiver
06	Distance processor
07	Microprocessor ROM
08	Microprocessor RAM
IND-42A/C DIAGNOSTICS	
CODE	DESCRIPTION
90	Bus failure (no data)
91	TEST mode fail (no response)
92	Distance word missing
93	VEL, TTS word missing
94	IDENT word missing
95	Microprocessor RAM
96	Microprocessor ROM

Table 16-4 IND-42A/C DME INDICATOR, CONTROLS AND INDICATIONS

CONTROL/INDICATOR	FUNCTION/DESCRIPTION
Numeric display	The numeric display presents the NM (distance) and diagnostic code.
Alphanumeric display	The alphanumeric display presents the KT (velocity), MIN (time-to-station), ID (2-, 3-, or 4-letter station identifier), and diagnostic identifier.
Power (PWR) switch	The latching push-on/push-off PWR switch controls the power applied to the IND-42.

**Table 16-4 IND-42A/C DME INDICATOR, CONTROLS AND INDICATIONS**

CONTROL/INDICATOR	FUNCTION/DESCRIPTION
Mode selector (SEL) switch alphanumeric	The nonlatching pushbutton SEL switch selects the information to be displayed in the display. (When power is initially applied, NM (distance) is shown in the numeric display and ID (DME station identifier) is shown in the alphanumeric display. Pressing the SEL switch will sequentially select KT (velocity), MIN (time-to-station), and ID (2-, 3-, or 4-letter station identifier). KT, MIN, and ID are shown in the alphanumeric display and NM (distance) is continuously shown in the numeric display, provided the DME is locked on a signal.
Channel (CH) switch (IND-42A only)	The momentary pushbutton CH switch sequentially selects the information from the next DME channel and lights the appropriate channel annunciator 1, 2, or 3. The copilot's IND-42C will always power up on channel 2.
Annunciators	The annunciators provide an indication of which DME channel is selected, system operational information, and units of measure. The following list describes the annunciators.
ANNUNCIATOR DESCRIPTION	
1 2 3	Sequentially controlled by the channel (CH) button to indicate which DME channel is providing the information being displayed in the numeric and alphanumeric displays.
NM	Automatically illuminates after power on when valid DME data is available. Indicates that the numbers displayed in the numeric display are slant range DME distance in nautical miles.
HLD	Indicates that DME hold has been selected on the CTL-32 NAV control. When in HLD, KTS and MIN, will revert to ID after approximately five seconds.
KT	Indicates that the value displayed in the alphanumeric display is the computed rate of change of DME. Distance.
MIN	Indicates that the value displayed in the alphanumeric display is the computed time-to-station in minutes.



UPPER SECTION OF EHSI DISPLAY

NOTE: NAV MUST BE SELECTED AS THE ACTIVE DISPLAY FOR THE ABOVE DISPLAYS TO BE PROVIDED BY THE DME.

Figure 16-13 DME-42 DME Transceiver Displays and Self Test Displays

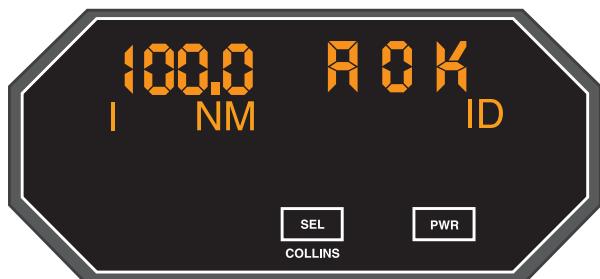
IND-42A SELFTEST DISPLAY
NO FAULTS FOUNDIND-42C SELFTEST DISPLAY 03
(TRANSMITTER) FAULT FOUND

Figure 16-14 IND-42 Self Test Displays

COMMUNICATIONS SYSTEMS

VHF-22A VHF COMM TRANSCEIVER

General

Two VHF-22A VHF communications transceivers are installed in the B300 aircraft to provide two-way airborne VHF AM communications on the 720 channels from 118.000 MHz through 136.975 MHz with 25 KHz increments. The VHF-22A receives audio from the audio system

for transmission and sends received audio to the audio system.

Some of the operating features of the VHF-22A transceiver are described below.

VHF-22A Equipment Turn On

The VHF-22A transceiver and the CTL-22 COMM control are turned on by rotating the power and mode switch on the COMM control to the ON position (Figure 16-15). When the transceiver is first turned on, it sounds a brief tone while the microprocessor checks its own memory. If there is a memory defect, the tone is continuous to indicate that the transceiver can neither receive nor transmit.

After the memory check, the CTL-22 displays the same active and preset frequencies that were present when the equipment was last turned off.

NOTE

If two short 800-Hz tones are heard, the transceiver has detected an internal fault. Push the TEST button on the CTL-22 to initiate self test and display the diagnostic code. If the CTL-22 fails after equipment turn on, the transceiver remains tuned to the last active frequency. With a failed CTL-22, cycling the power will cause the transceiver to beep twice and automatically change frequency to 121.5 MHz.

Adjust the volume level and perform a quick squelch test by setting the mode switch on the CTL-22 to SQ OFF and adjusting the volume level with background noise. After a comfortable listening level has been established, return the mode switch to the ON or NORM (depending on the control) position. All background noise should disappear unless a station or aircraft is transmitting on the active frequency.

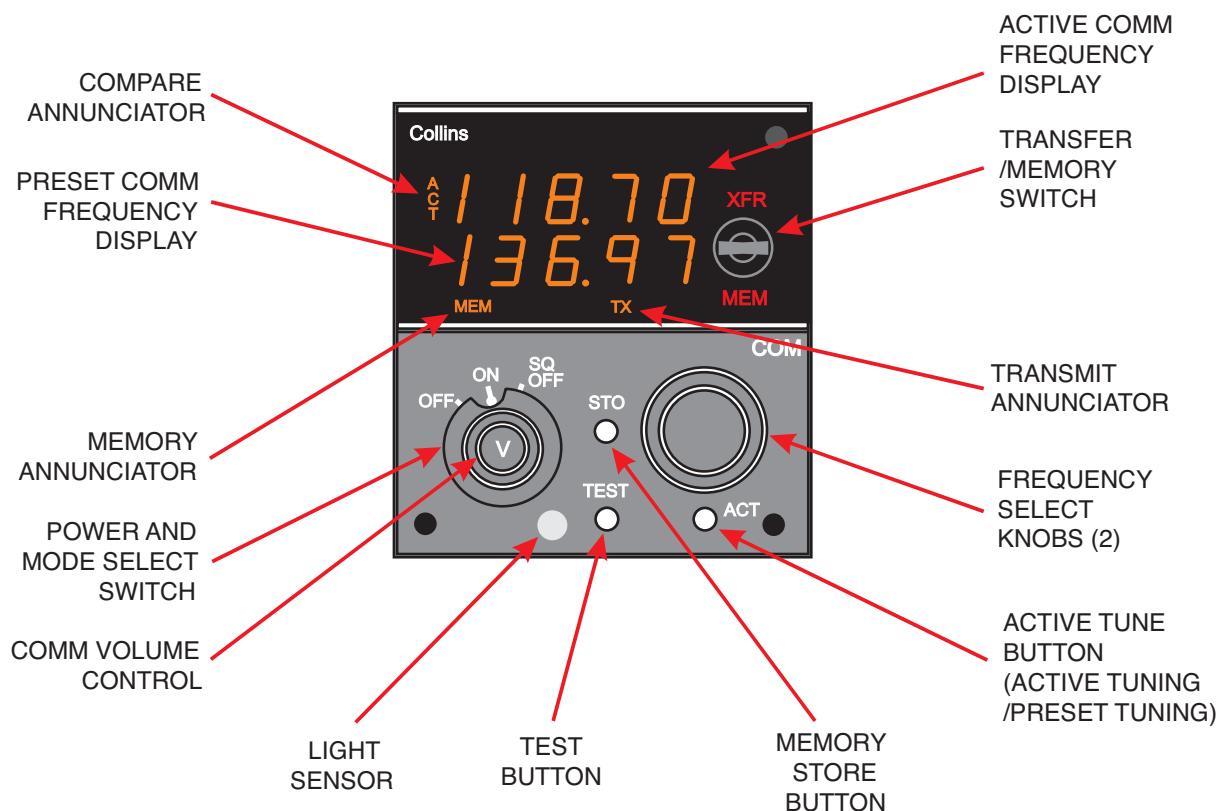


Figure 16-15 VHF-22 COMM Radio Controls/Displays

Stuck Mic Protection

Each time the PTT (push-to-talk) button is pushed the microprocessor in the transceiver starts a two-minute timer. The TX annunciator on the CTL-22 is illuminated whenever the transmitter is on. If the transmitter is still on at the end of two minutes, the microprocessor turns it off. Most intentional transmissions last much less than one minute; a two-minute transmission is most likely the result of a stuck PTT button. This timing feature protects the ATC channel from long-term interference.

Overtemperature Protection

The microprocessor regularly monitors the temperature of the transmitter. If the transmitter gets too hot during a transmission, the microprocessor stops the transmission, and sidetone ceases at that instant. When the PTT button is released,

two beeps will be heard. At this time, pressing the TEST button on the CTL-22 will display the overtemperature diagnostic code.

The shutdown temperature is +160°C (+320°F), well above normal operating temperature even in very hot conditions.

As long as the temperature remains above the limit, the microprocessor will not respond to a normal push of the PTT button. If you must transmit, you can override the protection by rapidly keying the PTT button twice, holding it in on the second push.

Locations

The VHF-22A transceivers are located in the right avionics bay; No. 1 is on the bottom, No. 2 on the top. There are two COMM antennas: No. 1 is on top of the airplane, and No. 2 is on the bottom.



Self Test

An extensive self test diagnostic routine can be initiated in the transceiver by pushing the TEST button on the CTL-22 COMM control (Figure 16-16). During self test, the CTL-22 modulates the intensity of the upper and lower displays from minimum to maximum to annunciate that self test is in progress. Several audio tones will be heard from the audio system while the self test routine is being executed. At the completion of the self test program, the CTL-22 usually displays four dashes (“----”) in the upper window and “00” in the lower window. This indicates normal operation, no fault detected. If any out-of-limit condition is found, the CTL-22 displays “DIAG” (diagnostic) in the upper window and a 2-digit diagnostic code in the lower window. Refer to the diagnostic codes table for a list of the diagnostic codes that can be displayed on the CTL-22.

Table 16-5 VHF-22A DIAGNOSTIC CODES

CODE	INTERPRETATION
21	Tuning voltage out of limit at highest receive
22	Tuning voltage out of limit at 118 MHz
23	Local oscillator output below limit
24	No-signal AGC voltage too high
25	Inadequate AGC voltage increase with RF signal
26	Excessive reflected RF power

Table 16-5 VHF-22A DIAGNOSTIC CODES

CODE	INTERPRETATION
00	No fault found
01	5 VDC below limit
02	5 VDC above limit
03	12 VDC below limit
04	12 VDC above limit
05	Synthesizer not locked
07	Noise squelch open without signal
08	Noise squelch not open with signal
12	BCD frequency code invalid
13	2/5 frequency code invalid
14	Serial message invalid
15	Frequency out of range
16	Forward power below limit
17	Transmitter temperature excessive

**Figure 16-16 VHF-22 COMM Radio Self Test Displays**



Table 16-6 CTL-22 COMM CONTROL, CONTROLS AND INDICATIONS

CONTROL OR DISPLAY	FUNCTION/DESCRIPTION
Active frequency display	The active frequency (frequency to which the VHF-22A is tuned) and diagnostic messages are displayed in the upper window.
Preset frequency display	The preset (inactive) frequency and diagnostic messages are displayed in the lower window.
Compare annunciator	ACT momentarily illuminates when frequencies are being changed. If ACT continues to flash, the actual radio frequency is not identical to the frequency shown in the active window.
Annunciators	The COMM control contains MEM (memory), and TX (transmit) annunciators. The MEM annunciator illuminates whenever a frequency is displayed in the lower window. The TX annunciator illuminates whenever the VHF-22A is transmitting.
Power and mode switch	The OFF/ON positions switch system power to turn the system on or off. The SQ OFF position disables the receiver squelch circuits, so you should hear noise. Use this position to set the volume control or, if necessary, to try to receive a very weak signal which cannot break squelch.
Light sensor	The built-in light sensor automatically controls the display brightness. The ANN push BRT control knob/push button can be used to override the automatic dim controls and force the display to go to full bright.
XFR/MEM switch	This switch is a three-position, spring-loaded toggle switch. When moved to the XFR position, the preset frequency is transferred up to the active display and the VHF-22A retunes. The previously active frequency becomes the new preset frequency and is displayed in the lower window. When this switch is moved to the MEM position, one of the six stacked memory frequencies is loaded into the preset display. Successive pushes cycle the six memory frequencies through the display (...2, 3, 4, 5, 6, 1, 2, 3....).
Frequency select knobs	Two concentric knobs control the preset or active frequency display. The larger knob changes the three digits to the left of the decimal point in 1-MHz steps. The smaller knob changes the two digits to the right of the decimal point in 50-kHz steps (or in 25-kHz steps for the first two steps after the direction of rotation has been reversed). Numbers roll over at the upper and lower frequency limits.
ACT button	Push the ACT button for approximately two seconds to enable the frequency select knobs to directly retune the VHF-22A. The bottom window will display dashes and the upper window will continue to display the active frequency. Push the ACT button a second time to return the control to the normal two-display tune/preset mode of operation. The active tuning feature is not affected by power removal. If active tuning is selected (one push of the ACT button) and power is removed from the control, active tuning will still be enabled the next time power is reapplied to the control.
STO button	The STO button allows up to six preset frequencies to be selected and entered into the control's nonvolatile memory. To store a frequency, simply toggle the MEM switch until the upper window displays the desired channel number (CH 1 through CH 6), rotate the frequency select knobs until the lower window displays the frequency to be stored, and push the STO button twice within five seconds. After approximately five seconds, the control will return to the normal two-display tune/preset mode of operation.
TEST	Push the TEST button to initiate the radio self test diagnostic routine. The transceiver performs a complete self test routine requiring about five seconds.



AUDIO SYSTEM (350)

General

The audio system consists of an audio control panel, two flight compartment speakers with jacks for pilot and copilot headphones and microphones, dual audio amplifiers, a passenger speaker amplifier, an aural warning tone generator, and an interphone amplifier (Figure 16-17).

The audio control panel provides control over both transmission and reception of all communication and navigation equipment installed in the airplane. ON/OFF switches, selector switches, and volume controls are provided for pilot and copilot control of each individual audio system.

The two DB 438 audio control amplifiers operate independently for the pilot and copilot systems. Both units are mounted on a shelf in the upper nose avionics compartment adjacent to the passenger speaker amplifier and the aural warning tone generator. Both audio control units are remotely controlled electronic units, which operate on command from switches and potentiometers on the audio control panel of the avionics instrument panel in the flight compartment. The audio amps provide audio from the radio receivers to the headphones and speakers and also provide interphone communication between the crew. Microphone audio is recorded at all times on the cockpit voice recorder.

The DB 240 passenger speaker amplifier provides chime tones and paging audio to the cabin speakers. The DB 240 is a remotely-controlled electronic unit, which operates from flight compartment controls.

The DB 600 aural warning tone generator generates aural warning tones to the cockpit headphones and speaker for stall warning, landing gear warning, etc. The aural warning tone generator is a remotely-controlled electronic unit which operates on command from fault detection equipment.

The audio system consists of independent audio control systems for the pilot and copilot. As part of the audio control system, MIC key PTT switches are provided on respective control wheels. In addition, MIC and headphone jacks are provided for use at each pilot's and copilot's station.

Audio Control Amplifier DB 438

The DB 438 audio control amplifier is a remotely-controlled electronic unit, which operates on command from switches and potentiometers in the cockpit. The DB 438 is capable of switching up to 14 receivers, four sidetone inputs, and an interphone line. Five unswitched and unmuted inputs are also provided to accommodate an aural warning input and four auxiliary inputs. Not all inputs and outputs are used in the B300 aircraft. Headphone and selected speaker outputs are isolated from each other and the cockpit voice recorder output is the summation of headphone, speaker, and microphone audio. Microphone audio is recorded on the cockpit voice recorder at all times (i.e. the microphone to cockpit voice recorder audio channel is unswitched and unmuted). Receivers and operating modes are electronically switched. Ground signals, supplied by manual switches in the cockpit, enable the electronic switches in the DB 438.

The DB438 audio amp is powered by 28 VDC and in the event of a power failure internal relays will deenergize, bypassing the following audio from it's input to the headset output: COMM 1, COMM 2, Sidetone 1, Sidetone 2, and the aural warning tones. In the event of an internal audio amp failure, a remote audio emergency switch can be activated to remove power from the internal relay to bypass the audio, as for a power failure. All these signals will be mixed at the headphone output.

Amplifiers are provided for the headphone and speaker outputs of the DB 438. The speaker output can be turned off, but the headphone output is always active. These outputs are open and short-circuit protected. An electronic attenuator is employed in the speaker and headphone channels for master volume control. The attenuator

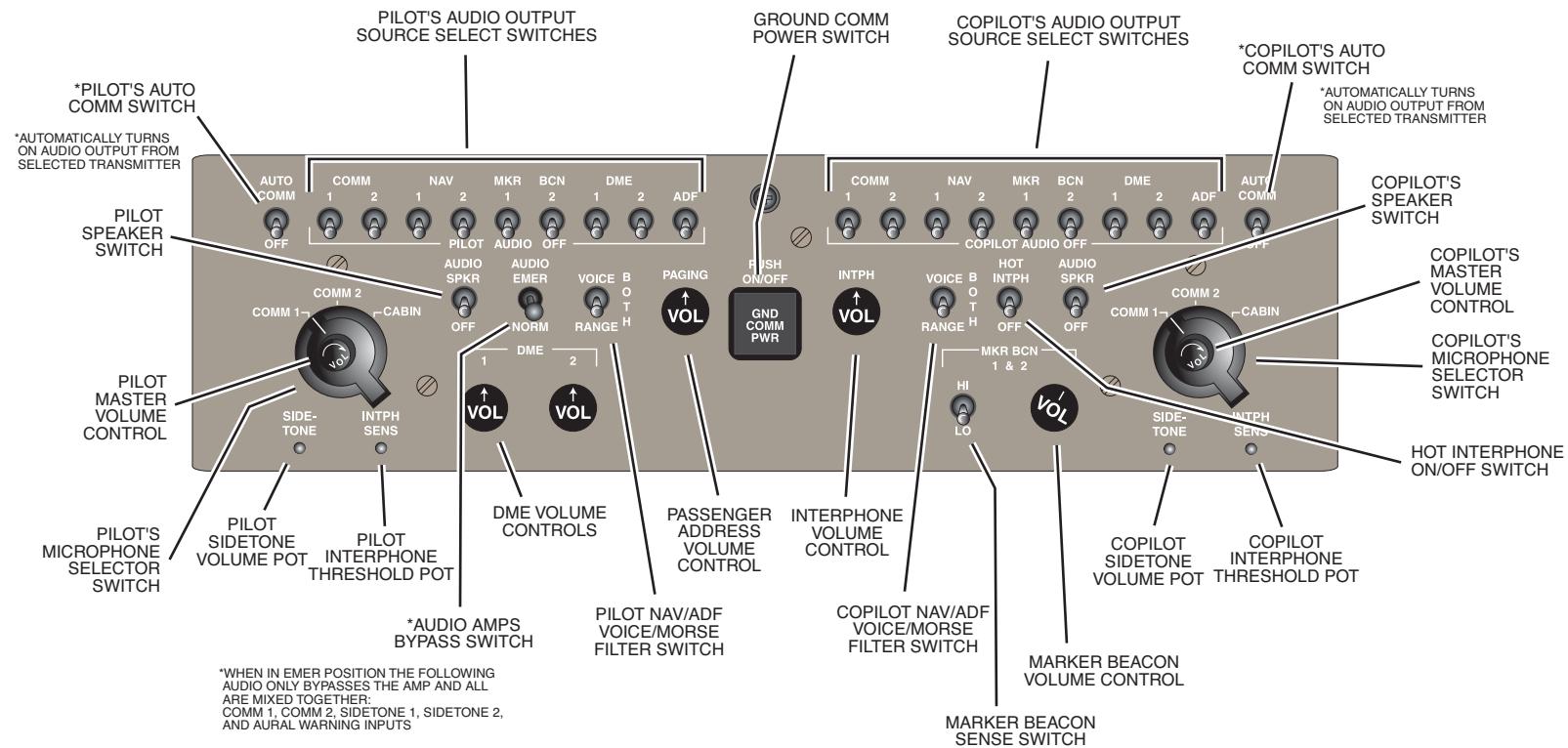
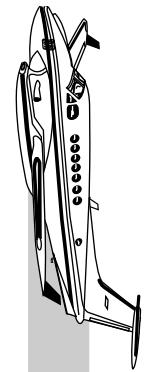
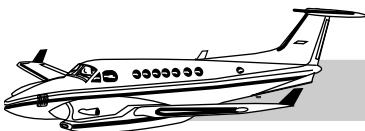


Figure 16-17 B300 (350) Audio Panel Controls



controls the level of all audio except the sidetone, auxiliary and aural warning inputs. A second attenuator is provided to set the sidetone audio level. The speaker/headphone and sidetone attenuators are controlled by separate cockpit controls.

The microphone input is used for interphone operation and supplies the bias current for microphone operation. The interphone system is activated by a remote hot interphone switch. A voice-activated switch is incorporated, in series with the microphone input, to reduce interphone noise. This feature allows hot interphone operation without ambient noise being coupled into the amplifier when the pilot is not talking. The microphone to cockpit voice recorder audio channel does not employ the voice activated switch circuitry. An electronic attenuator is employed in the interphone channel and is controlled by separate cockpit interphone control. Also, a cockpit control provides an input to the audio amplifier to set the threshold of the voice activating switch.

When the transmitter PTT input is active, the DB 438 opens the microphone to the interphone circuit and mutes both the headphone and speaker audio. Sidetone is made available when the transmitter PTT input is active and not controlled by the mute circuits.

Selective voice or code filtering is provided on four of the receiver audio inputs, intended for ADF and NAV audio. The filter function is electronically-controlled by ground signals from the cockpit voice-off-range switch.

Passenger Speaker Amplifier (DB 240)

The DB 240 passenger speaker amplifier provides chime tones and paging audio to the cabin speakers. The DB 248 is a remotely-controlled electronic unit which operates on command from cockpit controls and fault detection equipment.

Inputs from the pilot's and copilot's microphone are enabled by ground signals from the MIC key switch (PTT) switches via the MIC select switches to the DB 240. The circuitry is arranged to give the pilot priority in paging, the pilot PTT signal opens the copilot's microphone input. Microphone bias current is provided by the passenger speaker amplifier during paging.

The DB 240 incorporates a speaker amplifier to drive the cabin speakers during PA operation. The speaker amplifier output is internally protected against open and short circuits. An electronic attenuator is employed to control paging volume and is controlled by the cockpit paging volume potentiometer.

A musical chime tone is provided in the cabin speaker output to alert the passengers of a change in the fasten seat belt or no smoking sign status. The chimes are activated by inputs received from cockpit switches. The chime tone does not pass through the electronic attenuator.

Aural Warning Tone Generator (DB 600)

The DB 600 aural warning tone generator is a remotely-controlled electronic unit, which operates on command from fault-detection equipment to generate an appropriate warning tone. The



aural warning output is a summation of all activated warning tone generators within the DB 600. There are six, distinctly different tones available from the Model 600. Each tone generator is activated by ground signals at the associated control inputs.

The aural warning tones provided by the DB 600 are:

- Landing Gear Warning Tone
- Stall Warning Tone
- Autopilot Disconnect Warning Tone
- Altitude Alert Warning Tone
- Decision Height Warning Tone
- Overspeed Warning Tone

Parallel aural warning outputs are provided for the pilot's and copilot's audio control amplifiers. The aural warning output is a summation of all activated warning tone generators within the DB 600 and is both open and short circuit protected.

Audio System Signal Flows

Audio System Inputs Interface

Microphone audio is input to the audio control panel for switching to the various transmitters and interphone system. The microphone audio comes from both the pilot's and copilot's system and can be input from three sources. The sources are the hand microphone and a selectable input from either the headset boom microphone or the oxygen mask microphone. The MIC select switch is located on the overhead panel and microphone jacks are located on each side of the cockpit sidewall. The boom and oxygen mask microphone have MIC key switches (PTT) for transmitting located on the control wheels. The hand microphone has a key switch as part of the microphone assembly.

The switched microphone audio from the audio panel is sent to the appropriate transmitter COMM 1, COMM 2, HF (if installed) and PA amp DB 240. The microphone key ground is also sent to the selected transmitter or PA amp to activate the transmit operation.

During radio transmission, sidetone is generated by the transmitting unit and supplied to the DB 438 audio amps to allow the crew to monitor the actual audio that is being transmitted. A screwdriver adjustable sidetone volume pot located on the audio panel can be used to adjust the sidetone level.

Audio System Output Interface

The audio amps drive the respective speaker and headphones. The headphone output is operational at all times, the speaker can be turned on and off with the speaker switch. A master volume control is provided for the speaker and headphones.

Switches on the audio control panel provide a ground into the audio amp to turn the respective audio input on or off.

The VOICE–BOTH–RANGE switch works in conjunction with both the ADF and NAV receivers. When in the VOICE position, the voice portion of the audio will be heard, but not the Morse code station identification. When in the RANGE (IDENT) position, only the Morse code station identification will be heard, not the voice portion. When in the BOTH position, both the voice and range portions of the audio will be heard.

The interphone connection allows the audio signal to travel between the audio amps when the interphone is active.

The passenger speaker amp drives the cabin speakers. The PA amp receives inputs from the pilot's and copilot's microphone for PA announcements. A PA volume control (Paging) on the audio control panel adjusts cabin speaker volume. Chime tones are also generated and provide to the cabin speakers for no smoking and fasten seat belts signs.



Dedicated outputs from each audio amp and a connection from one of the speaker outputs are sent to the CVR for recording. The CVR records all audio received and transmitted through the audio system.

GROUND COMMUNICATIONS AND AUDIO SYSTEM POWER

GENERAL

A ground communication system is installed in the B300 aircraft to provide the crew the ability to operate COMM 1 radio without powering up the main electrical system. With all electrical power off in the aircraft pushing the GRD COMM PWR switch will activate the COMM 1 radio and the pilot's and copilot's audio amps. The annunciator in the switch will illuminate to indicate system operation. Pressing the GND COMM PWR switch off or turning the battery master switch or external power switch on accomplishes system deactivation. The annunciator in the GND COMM PWR switch will extinguish.

NOTE

Ensure the ground communication power switch is off before leaving the airplane so a drain on the battery will not occur.

SYSTEM OPERATION

The ground communication electric power comes directly from the battery when ground COMM is selected on. Control of the system consists of a push-on/push-off solenoid-held annunciator switch located on the audio control panel on the instrument panel. Circuit power comes from the HOT BATTERY BUS. Activation of the system allows operation of the No. 1 COMM system and the pilot's and copilot's audio amps. Audio is provided to the headphones and speakers. When the battery

master switch or external power is applied and turned on, the ground COMM system will automatically be disconnected. Pressing the GND COMM PWR switch will also turn off the ground COMM system.

In the B300 aircraft the pilot's audio system is powered off the triple-fed bus, while the copilot's is powered off the right generator bus, and will only be active with an engine running (generator buses powered) or ground power connected.



KING AIR 300 DB-415 AUDIO SYSTEM

The majority of the Super King Air 300s built to date have the DB-415 audio system installed. Although other optional audio systems may be installed, the standard DB-415 system is the only one which will be described in this section.

The avionics system has dual DB-415 audio systems which are totally independent of each other (Figure 16-18). The only exception to this rule is that there is only one emergency/normal switch on the radio panel which serves both the pilot's and copilot's audio systems. Therefore, if the emergency/normal switch is in EMER position, both systems will be in that mode of operation.

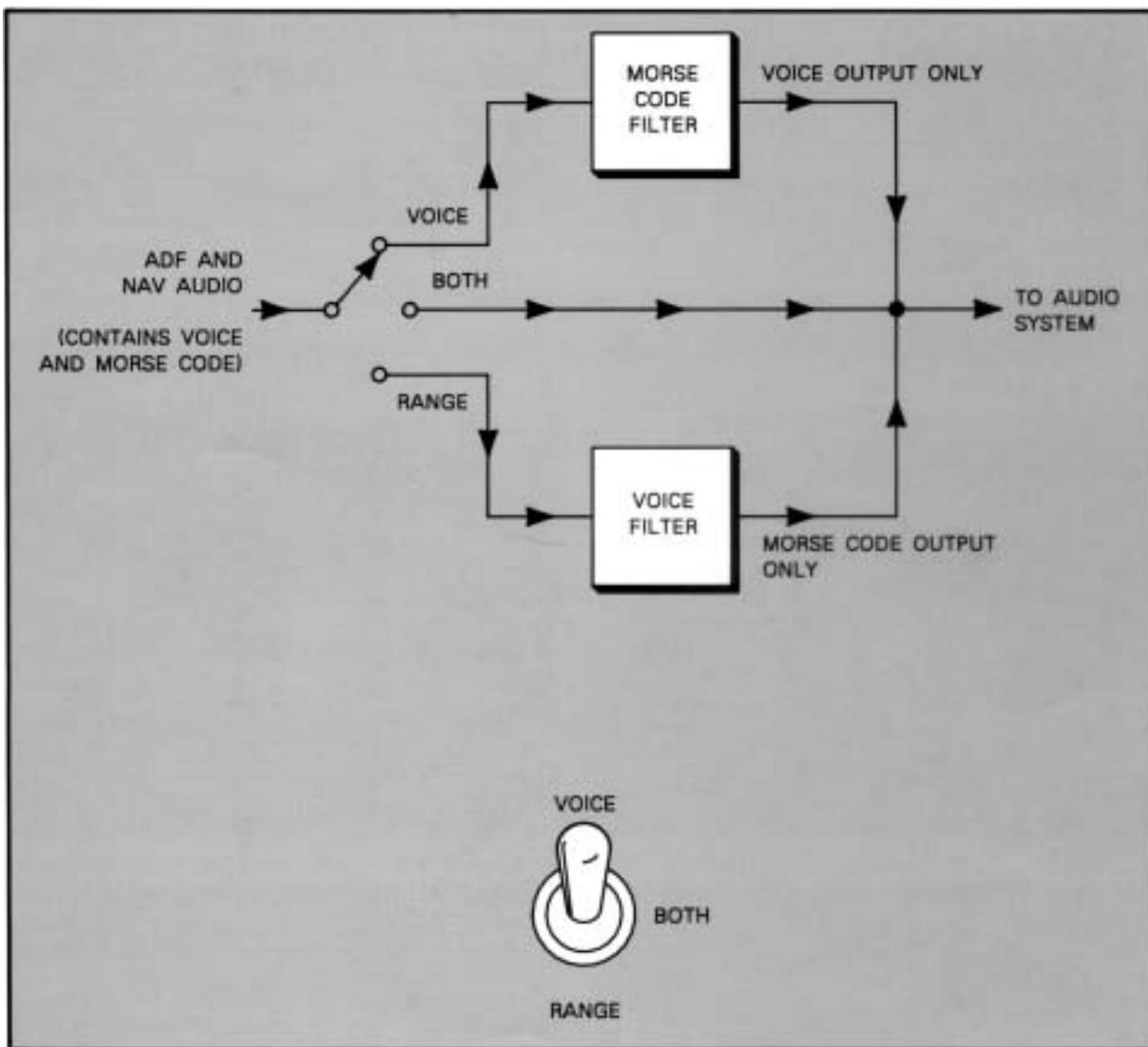


Figure 16-18 King Air 300 DB-415 Audio Diagram



Normal Operation

Under "normal" circumstances, the normal/emergency switch should remain in the NORM position (Figure 16-19). The following operating rules apply when in the normal mode of operation (Figure 16-20). Rules will only be listed for the pilot's audio system; however, they apply equally to the copilot's audio system:

- The volume control on the microphone selector switch regulates the volume level for both the pilot's speaker and headphones.
- The speaker switch turns the speaker ON and OFF.
- The headphones are operational at all times as long as they are plugged into the headphone jack.
- The speaker and headphone audio channels are independent of each other and failure of one does not necessarily imply a failure of the other.

- To select any audio source (e.g., Comm 1, ADF, etc.) turn ON the appropriate audio selector switch.
- The switch labeled VOICE – BOTH – RANGE (see Figure 16-19) works in conjunction with both the ADF and NAV receivers. When in the VOICE position, the voice portion of the audio will be heard and not the Morse code station identification. When in the RANGE (ident) position, only the Morse code station identification will be heard, not the voice portion. When in the BOTH position, both the voice and range portions of the audio will be heard.

If the pilot's audio system has failed entirely, the pilot may still listen to audio through the copilot's speaker. If this is undesirable for any reason, or if both the pilot's and copilot's audio systems have failed, place the emergency/normal switch in the EMER position.

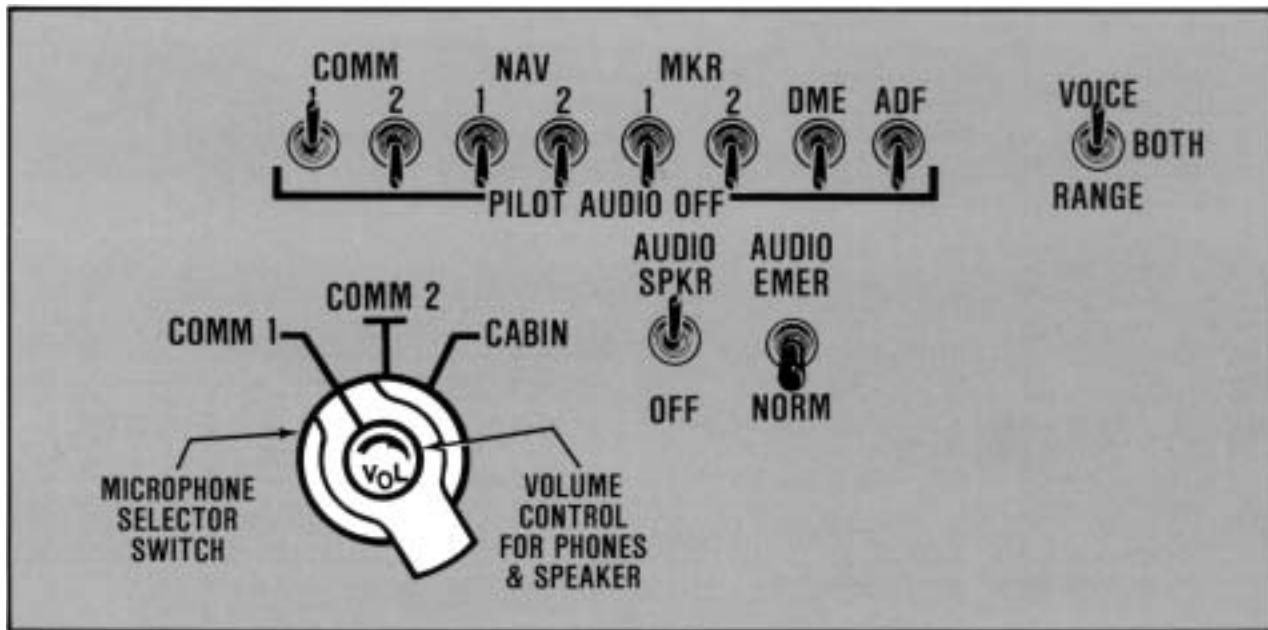


Figure 16-19 King Air 300 DB-415 Audio Switching Panel (Typical)

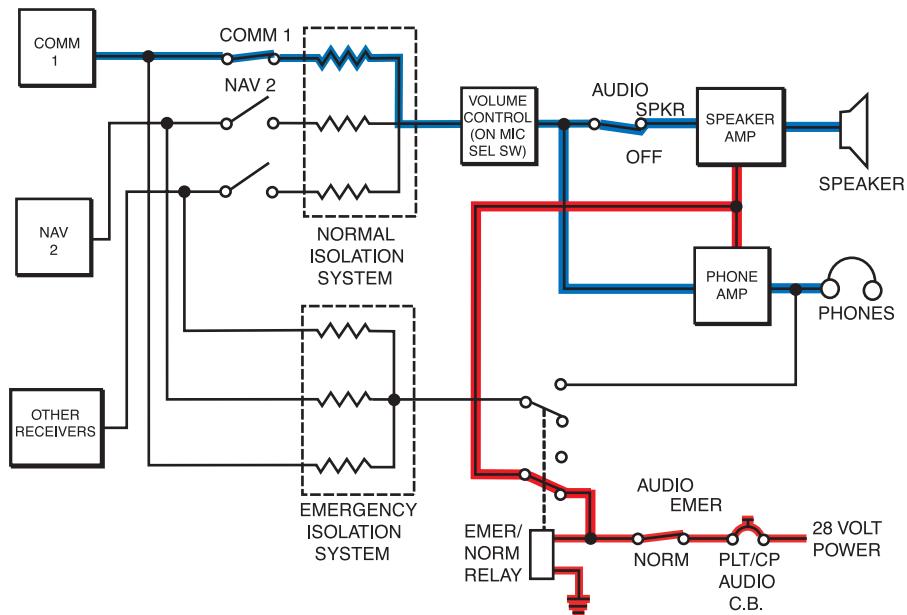


Figure 16-20 King Air 300 DB-415 Normal Operation Schematic

Emergency Operation

When in the emergency mode of operation, the following operating rules apply (Figure 16-21):

- All audio sources (Comm 1, NAV2, ADF, etc.) are connected directly to the headphones.

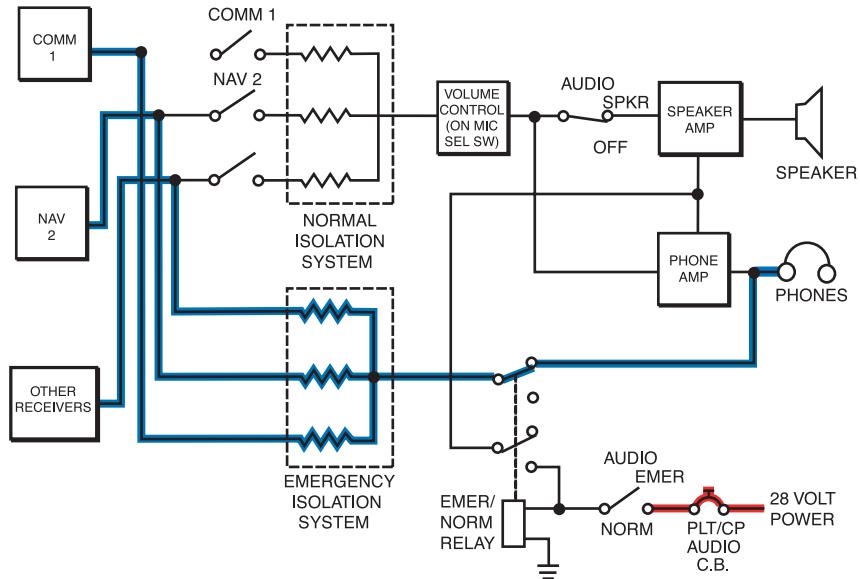


Figure 16-21 King Air 300 DB-415 Emergency Operation Schematic



- To eliminate any specific audio source, turn down the volume control on that audio source (e.g., NAV1). This rule does not apply to Comm 1 and Comm 2.
- The volume control located on the microphone selector switch has no function in the emergency mode.
- If the emergency/normal switch should fail for any reason, the audio systems may still be placed into the emergency mode of operation by pulling the two circuit breakers labeled PLT AUDIO and COPLT AUDIO located directly beneath the avionics master circuit breaker, on the main circuit breaker panel.

ADF-60A AUTOMATIC DIRECTION FINDING SYSTEM

General

The B300 aircraft can have either single ADF receiver installed or dual ADFs (Figure 16-22). The ADF-60A system receives transmissions from a selected ground station, indicates relative bearing to that station, and provides audio for

determining station identification and listening to voice announcements. The ground station must be within the normal operating range of 190 KHz to 1749.5 KHz. There are three functional modes of operation. In ANT mode, the ADF receiver functions as an aural receiver, providing only an aural output of the received signal. In ADF mode, it functions as an automatic direction finder receiver in which bearing-to-the-station is presented on an associated bearing indicator and an aural output of the received signal is provided. A TONE mode provides a 1000-Hz aural output tone, when a signal is being received, to allow identification of keyed CW signals.

Display Interface

The No. 1 ADF-60A supplies an ADF bearing signal to the display processor unit (DPU) and the multi-function processor (MPU). When the No. 2 ADF-60A receiver is installed it also supplies an ADF bearing signal to the multi-function processor (MPU). The EFIS system can display the onside ADF bearing information and the cross-side ADF bearing information via the cross-side data bus from the MPU to the DPU on the pilot's EHSI display. The display control panel (DCP) is used to select the ADF receiver used as the source

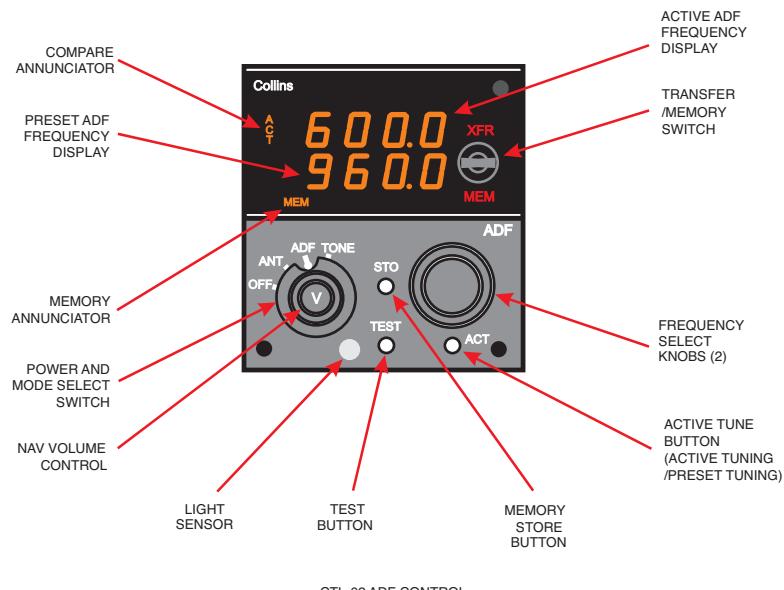


Figure 16-22 ADF-60A ADF Receiver Controls/Displays



for bearing display. In the case of a single ADF installed there is only an ADF selection, not ADF1 and ADF2 as in the dual ADF install. The ADF receiver also provides a bearing signal to each RMI and the HSI processor unit (HPU) for display of ADF station bearing on the RMIs and the copilot's EHSI display.

Locations

For the B300 aircraft, the No. 1 ADF-60A is mounted upside down in the right avionics bay with its CAD-62 mounted upside down in the left side of the nose avionics bay.

The ADF antennas are located on the bottom of the aircraft just forward of the wing.

Self Test

- Apply power to the ADF and RMI, and EFIS systems.
- Select the ADF to be tested bearing pointer on the RMI and EHSI.
- Set the control mode to ANT. This applies power to the system.
- Tune the control to any frequency, preferably one that is known to give a good signal.
- Note that the RMI pointer should indicate the correct relative bearing to the station. Adjust the audio volume, as necessary, for a comfortable listening level.

NOTE

If the RMI pointer remains parked, the system may not be receiving a reliable signal. In this case, try two or three other stations if possible.

- Push and hold the self test switch. Note the RMI and EHSI bearing pointer

rotates 90° counterclockwise. Release the self test switch.

NOTE

If the signal is weak or of poor quality, the bearing pointer can rotate rather slowly. If this happens with a station that historically has given good indication, check for antenna cable and/or connector defects (i.e., tears, breaks, etc). A degraded receiver sensitivity might give the same response.

- Tune to any CW station, if one is available. Set the mode to TONE and listen to the audio for a 1000-Hz tone identifying the CW station.

Diagnostic Display

When the CTL-62 is used with the CAD-62 to control the ADF-60A system, certain diagnostics codes can be displayed on the CTL-62 in the self test mode. The diagnostic codes relate to the CAD-62 operation, apart from one code "10" ADF not locked in ADF mode. The diagnostic display appears when the self test button pressed as above for the ADF self test. If the diagnostics detect no faults, the CTL-62 displays four dashes (---) and code 00. If the diagnostics detect a fault "DIAG" will be displayed along the fault code on the CTL-62 display (Figure 16-23).

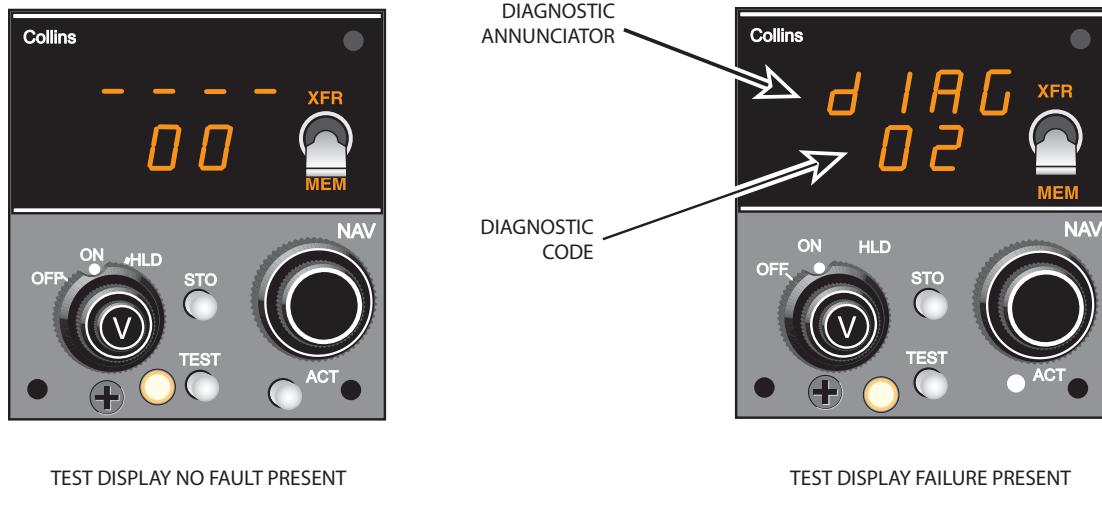
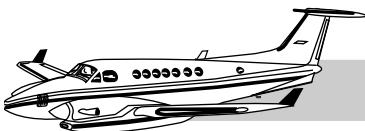


Figure 16-23 ADF-60A ADF Receiver Self Test Displays

Table 16-7 ADF-60A SELF TEST DIAGNOSTIC CODES

CODE	INTERPRETATION
00	No faults
02	RAM test failed (test failed at power-up)
03	No serial sync block received
04	No serial data block received for more than two seconds
05	Transponder label received in first block
06	I/O port 1 failure
07	I/O port 2 failure
08	Illegal ADF frequency
09	Serial input failed or stuck low
10	ADF not locked in ADF mode



Table 16-8 CTL-62 ADF CONTROL, CONTROLS AND INDICATIONS

CONTROL/INDICATOR	FUNCTION/DESCRIPTION
Active frequency display	The active frequency; the frequency to which the ADF-60A is tuned. In self test mode and if an out-of-tolerance condition is detected, the word "dIAG" is displayed in the upper window while the diagnostic code is displayed in the lower window.
Preset frequency display	The preset frequency is displayed in the lower window. In self test mode and if an out-of-tolerance condition is detected, the diagnostic code is displayed in the lower window.
Compare annunciator	ACT momentarily illuminates when frequencies are being changed. If the ACT annunciator continues flashing, the receiver is not tuned to the displayed active frequency.
Annunciators	The ADF control contains a MEM (memory) annunciator. The MEM annunciator illuminates whenever a frequency is displayed in the lower window.
Volume control	The volume control, is concentric with the power and mode switch and controls ADF audio volume.
Light sensor	The built-in light sensor automatically controls the display brightness. The ann push brt control knob/push button can be used to override the automatic dim controls and force the display to go to full bright.
Power and mode switch	The power and mode switch contains four detented positions. OFF The OFF position interrupts system power (Turns the ADF off). Selecting ANT, ADF, or TONE applies power to the ADF system and establishes the system mode of operation. ANT In ANT mode, the ADF receiver functions as an aural receiver, providing only an aural output of the received signal. ADF In ADF mode, it functions as an automatic direction finder receiver in which bearing-to-the-station is presented on an associated bearing indicator and an aural output of the received signal is provided. TONE TONE mode provides a 1000-Hz aural output tone when a keyed CW signal is being received.
XFR/MEM switch	This switch is a 3-position, spring-loaded toggle switch. When moved to the XFR position, the preset version units only frequency is transferred up to the active display and the ADF-60 retunes. The previously active frequency becomes the new preset frequency and is displayed in the lower window. When this switch is moved to the MEM position, one of the four stacked memory frequencies is loaded into the preset display. Successive pushes to the MEM position cycles the four memory frequencies through the display (...2, 3, 4, 1, 2, 3...). The frequency that was in the preset window is: 1. Maintained in memory if it was originally assigned as a stored frequency, or 2. Discarded if it was originally direct tuned either as a preset or active.
1/2 switch	The 1/2 switch connects the tuning knobs to either the upper window or lower window for tuning, 1 for upper and 2 for lower.

**Table 16-8 CTL-62 ADF CONTROL, CONTROLS AND INDICATIONS (CONT.)**

CONTROL/INDICATOR	FUNCTION/DESCRIPTION
Tuning	Normally, tuning is accomplished by entering a frequency into the preset window and then either units storing that frequency in memory (STO) or entering it into the active window (XFR) to tune the receiver. An alternate method is to press the ACT button for at least two seconds (this gives direct tuning access to the upper window) and insert the desired frequency directly into the active window.
Frequency select knobs	Two concentric knobs control the preset or active frequency displays. The larger knob changes the 1000's and 100's KHz digits. The smaller knob changes the 10's, units, and tenths KHz digits. Each detent of the larger knob changes the frequency in 100-KHz steps. Each detent of the smaller knob changes the frequency in 1-KHz steps with the exception that the first two detent positions following a change in rotational direction will cause a 0.5-KHz change. Rapid rotation of the smaller knob will cause frequency changes greater than 1 KHz as a function of the rate of rotation. Frequencies roll over at the upper and lower limits. The two frequency select switches are independent of each other such that the upper and lower limit rollover of the 10-kHz digit will not cause the 100-KHz digit to change.
ACT button	Push the ACT button for approximately two seconds to directly change the active display window from units only the frequency select knobs. The bottom window will display dashes. Push the ACT button a second time for about two seconds to return the control to the normal 2-display tune/preset mode of operation. The active tuning feature is not affected by power removal. If active tuning is selected (one push of the ACT button) and power is removed from the control, active tuning will still be enabled the next time power is reapplied to the control.
STO button	The STO button allows up to four preset frequencies to be selected and entered into the control's only nonvolatile memory. To store a frequency, simply toggle the MEM switch until the upper window displays the desired channel number (CH 1 through CH 4), rotate the frequency select knobs until the lower window displays the frequency to be stored, and press the STO button twice within five seconds. After approximately five seconds, the control will return to the normal 2-display tune/preset mode of operation.
TEST button	Push the TEST button to initiate the radio self test routine. Self test is active only while the TEST button is pushed. The display modulates in intensity while the TEST button is pushed.

TDR-94 MODE S TRANSPOUNDER

General

Two TDR-94 transponders are installed in the B300 aircraft with only one operating at any one time (Figure 16-24). The TDR-94 is a mode-A, mode-C, and mode-S transponder and is an integral part of the air traffic control radar

beacon system. It provides identification (mode-A) of the transponder equipped aircraft on the ATC ground controller's plan position indicator. The TDR-94 transponder also provide aircraft pressure altitude to the ground controller's indicator (mode-C). The transponders are sent altitude data information from the pilot's altimeter or the copilot's altimeter.

In normal mode-A or mode-C operation, the TDR-94 is interrogated by radar pulses from a

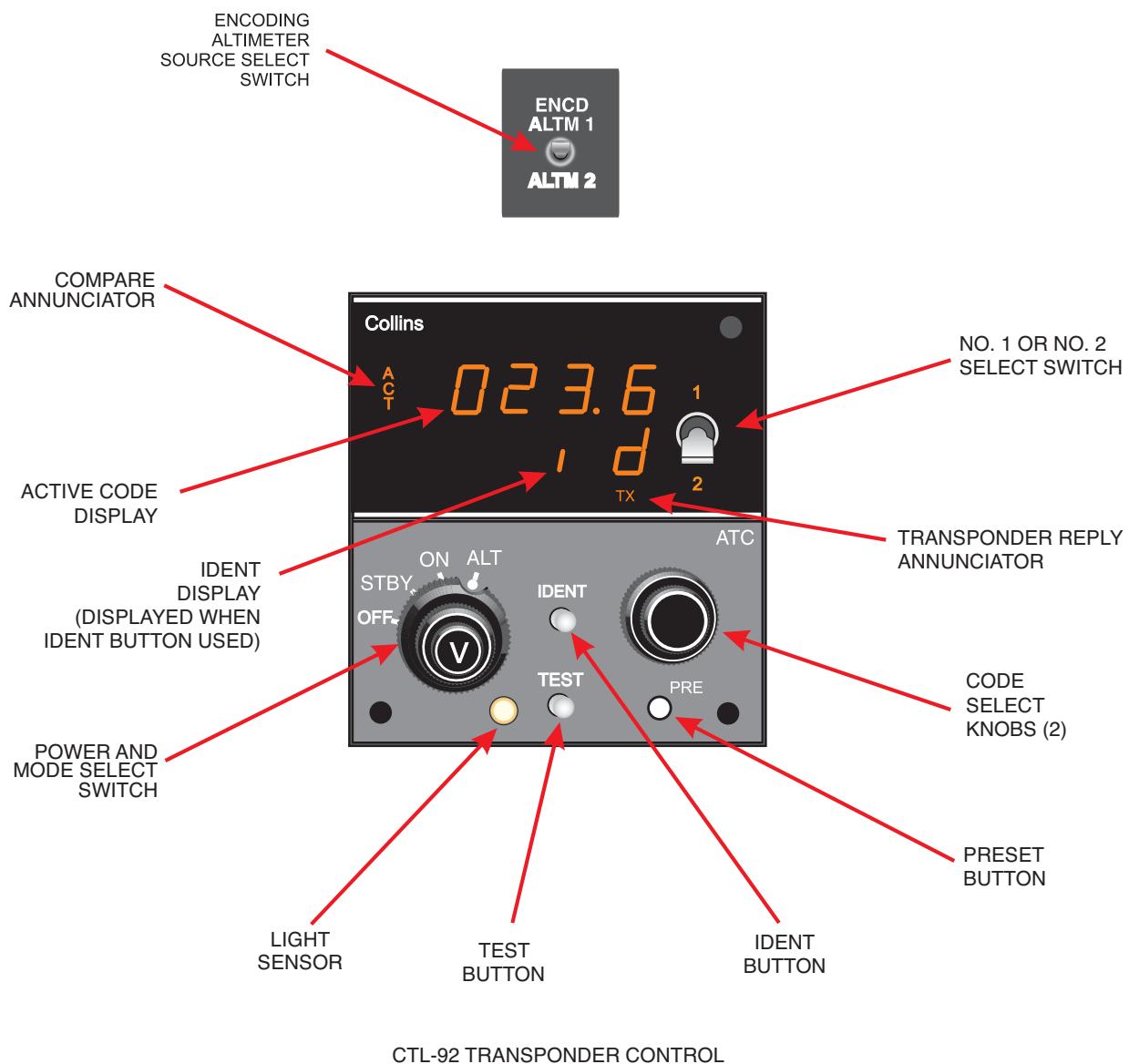


Figure 16-24 TDR-94 Transponder Controls/Displays

ground station and replies automatically with a series of pulses. These reply pulse groupings are coded to supply identification and altitude data.

The TDR-94/94D can operate in mode-S and provide a unique aircraft identification code as well as air-to-air and air-to-ground interrogation replies. The unit also has datalink capability, which allows it to perform additional air traffic control and aircraft separation assurance functions.

The TDR-94 transponder can also transmit an identification pattern when requested to “squawk ident” by the ground controller.

Refer to the Controls and Indications Table for information on operation of the transponders (Table 16-9).

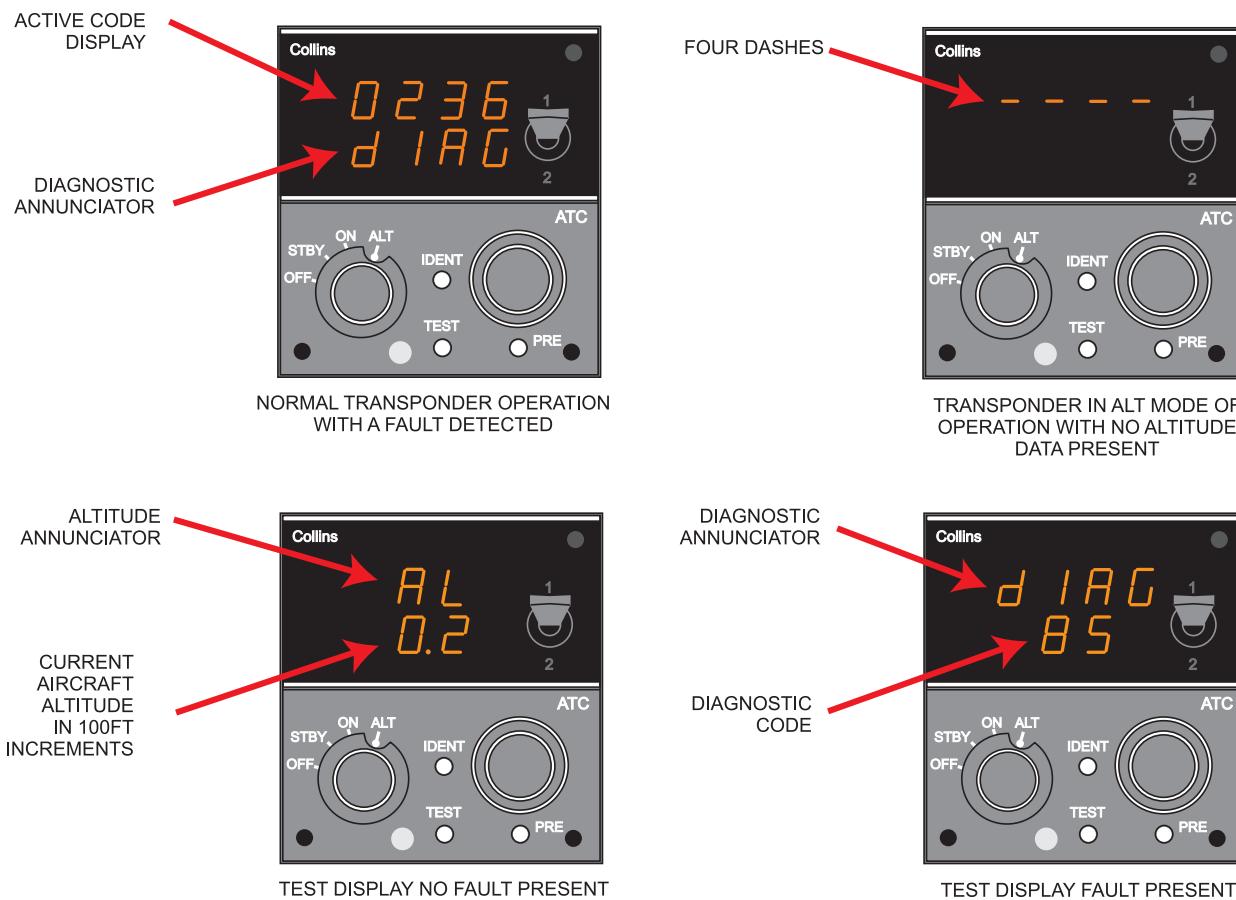
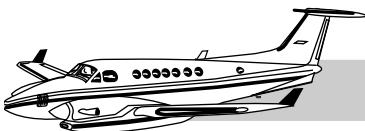


Figure 16-25 TDR-94 Transponder Self Test Displays

Locations

Both transponders are found in the tail section avionics shelf, No. 1 forward, No. 2 aft. The two transponder antennas are just aft of the ADF antennas, on the bottom of the fuselage.

Self Test

To carry out transponder self test, position the mode switch to ON and select the desired transponder to be tested. Set the desired code using the code select knobs. Push the TEST button on the CTL-92. During self test the CTL-92 display flashes from minimum to maximum brightness. If there are no diagnostic conditions detected,

uncorrected barometric altitude is displayed on the bottom display line and the annunciator "AL" on the top display line (Figure 16-25). If a diagnostic condition is detected in the TDR-94/94D during self test, the upper window displays the word "DIAG" while the lower window displays a 2-digit diagnostic code. To carry out self test on the opposite transponder, use the select switch on CLT-92 to select that transponder. If during normal operation a "fail/warn" condition is detected, the CTL-92 will display the "dIAG" message in the lower window without having pressed the TEST button. When this occurs, press the TEST button to view the associated diagnostic code. Refer to the *Collins Installation Manual* for diagnostic code interpretation.



Table 16-9 CTL-92 ATC CONTROL, CONTROLS, AND INDICATIONS

CONTROL OR DISPLAY	FUNCTION//DESCRIPTION
Upper display window	The ATC code (code with which the active transponder replies) and diagnostic messages are displayed in the upper display window. During normal operation, the CTL-92 has only a single display, the transponder code, shown in the upper window.
Lower display window	The lower display window is normally blank. It is active only during self test. If a fail/warn condition is detected, "DIAG" will be displayed. Press the TEST button to view the diagnostic code.
Compare annunciator	ACT momentarily illuminates when codes are being changed. If ACT flashes, the actual reply code is not identical to the code shown in the active code display.
Annunciator	The ATC control contains an TX (transmit) annunciator. The TX annunciator illuminates when the transponder replies to an interrogation.
Power and mode switch	<p>The ATC control power and mode switch contains four detented positions. The available positions are: OFF-STBY-ON-ALT.</p> <p>Power is removed in the OFF position and is applied when any of the other modes is selected.</p> <p>In the STBY mode, power is applied to the transponder but it is prevented from transmitting replies. STBY should be used only during taxi or when requested by ATC.</p> <p>The ALT position is the normal operating position and allows the transponder to reply to the interrogation pulses as well as transmitting uncorrected barometric altitude when the transponder is interrogated in mode-C.</p> <p>The ON position deletes the altitude code and is normally used when requested by ATC.</p>
1/2 switch	The 1/2 switch selects which of two transponders is active
Light sensor	The built-in light sensor automatically controls the display brightness. The ANN push BRT control knob/push button can be used to override the automatic dim controls and force the display to go to full bright.
Code select knobs	Two concentric knobs control the active code display. The larger knob changes the two more significant digits, and the smaller knob changes the two less significant digits. The less significant digit is incremented or decremented for each detent of the smaller knob if the knob is slowly turned. Rapid rotation of either knob will cause changes proportional to the rate of rotation. Rollover of the less significant digits will occur at 0 and 7, and will cause the more significant digits to be incremented or decremented. The left two digits and the right two digits are independent of each other. The various codes used for normal operation are listed in the <i>Airman's Information Manual</i> . Codes 7600 or 7700 are selected for in-flight emergency operation and will be annunciated by the codes flashing in the active code display for a couple of seconds before transmission begins.

**Table 16-9 CTL-92 ATC CONTROL, CONTROLS, AND INDICATIONS (CONT.)**

CONTROL OR DISPLAY	FUNCTION//DESCRIPTION
PRE button	Push and hold the PRE button while turning the code select knobs to select a pre-set code for storage. The preset code will be stored in nonvolatile memory and can be recalled by momentarily pressing the PRE button again.
IDENT button	The IDENT button causes the transponder to transmit a special identification pattern that is displayed on the ground controller's radar scope. This button should be pushed only when you are requested to "squawk ident" by the ground controller
TEST button	Push the TEST button to initiate the radio self test routine. In dual version units, the 1/2 switch determines which transponder responds to the test command.
Encoding altimeter select switch	This switch selects which altimeter, the pilot's (ALTM 1) or copilot's (ALTM 2), will provide encoding altimeter information to the transponders.
Transponder normal override select switch	This switch overrides the on-ground squat switch input to enable the transponder to output the in-air message in the mode-S information. For ground maintenance test use.
SELF TEST DISPLAY	
No failure	During self test, the active code display intensity will modulate from minimum to maximum. If the transponder is functioning properly and an altitude encoder is connected to the CTL-92 and operating, AL will be displayed in the upper window and the altitude in thousands of feet in 100-foot increments will be displayed in the lower window.
Failure	If an out-of-tolerance condition is detected, the upper window shows the word "DIAG" while the lower window shows a 2-character diagnostic code.

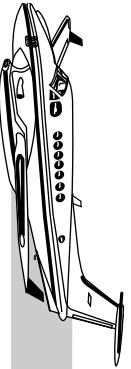
UNS-1K/L FLIGHT MANAGEMENT SYSTEM

General

Universal Avionics Systems Corporation's Flight Management System, the UNS-1K/L FMS, is a fully integrated navigation and flight management system designed to provide the pilot with centralized control for the aircraft navigation sensors, computer-based flight planning, and fuel management. The UNS-1K/L unit consists of a master computer and a 12-channel GPS sensor housed in one unit, the navigation computer unit (NCU) and a flat panel full color, LCD display with alphanumeric keys, function keys, line

select keys, flat panel control display unit (FPCDU) (Figure 16-26).

The UNS uses position information from the internal GPS receiver as well as the No. 1 DME, and No. 1 VOR. The data from these sensors is used to determine the best computed position. This position is used by the UNS for navigating the aircraft along the programmed flight plan. Flight plans are created using the Jeppesen database. The database locates waypoints along a desired route and then combines these waypoints to form a flight plan. The fuel management function of the UNS accepts input from the aircraft's fuel flow sensors as well as information supplied by the pilot. It uses this data to continuously update and display fuel management information during flight.



FOR TRAINING PURPOSES ONLY

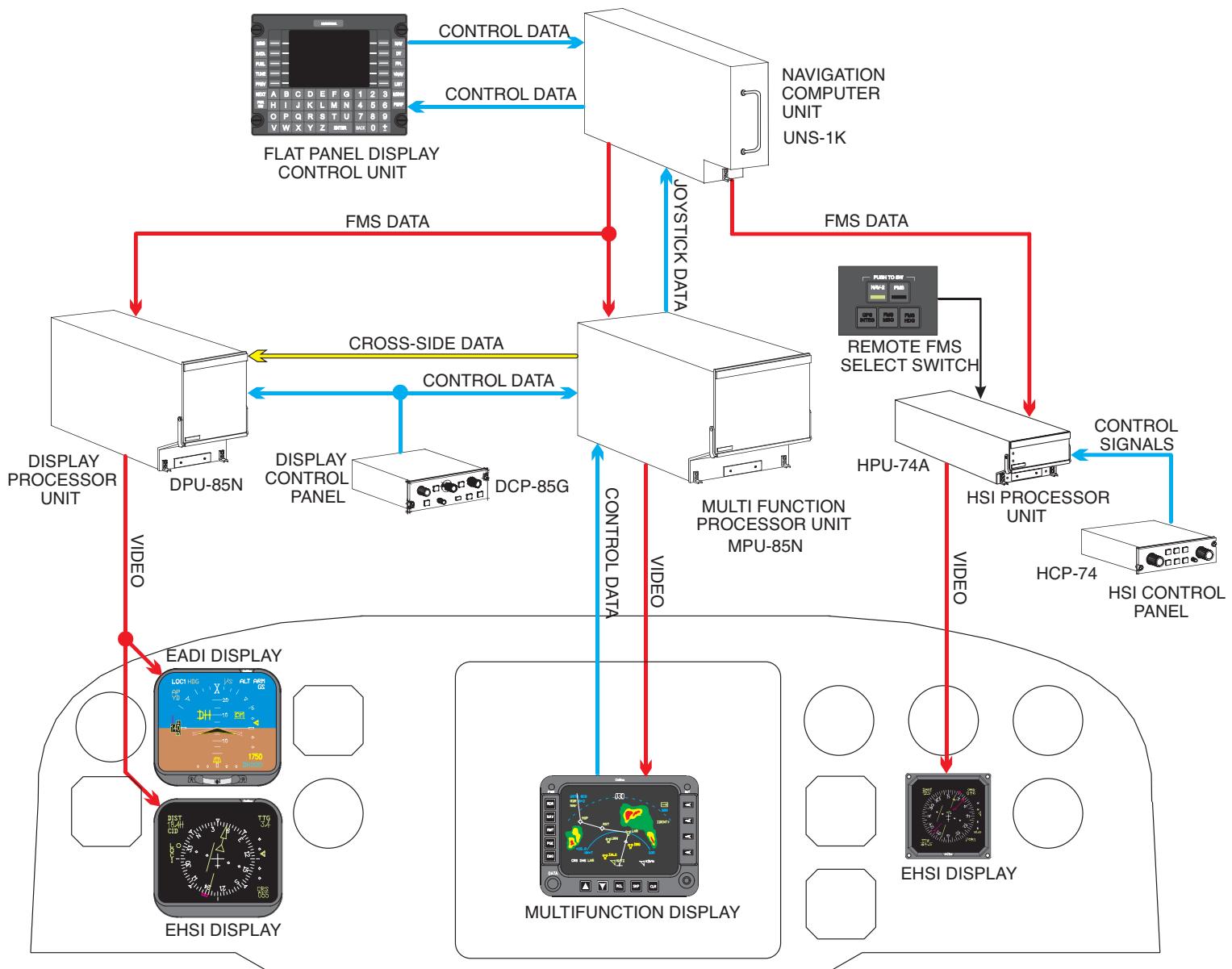


Figure 16-26 Flight Management System Display Interface



The system carries out lateral great circle navigation and provides a vertical navigational guidance display from aircraft departure to arrival. This includes enroute as well as terminal and approach capabilities.

In addition to the navigational inputs, the UNS also receives true airspeed and altitude information from the ADC-85 air data computer. The UNS receives fuel flow data from the aircraft fuel flow gages and the heading signal from No. 1 compass system.

The UNS-1K/L FMS system is approved for GPS as the primary means of navigation when outside the range of land based sensors and approved for approaches to runways using GPS and/or other land based sensors.

NAVIGATION DATABASE

The FMS contains a memory capacity up to 100,000 waypoints in a non-volatile flash RAM (no battery required). The stored navigation database provides the capacity for complete coverage for SIDs, STARs, approaches, high/low airways, navaids, IFR intersections, and airports with runways longer than 4,000 feet with IFR approach in the worldwide database, and 2,000 feet in the regional database. Worldwide or regionalized database subscription services are available. These services provide for update on a 28-day cycle. In addition, trip kits for other than the subscribed region are available upon request. Extensive memory space is also allocated for pilot defined data. The system can store in memory up to 200 pilot created flight plan routes comprised of up to 98 waypoints each, and up to 200 pilot defined waypoints, 100 arrivals/departures, 100 approaches, 100 runways, 100 airports, 100 alignment points and 25 radar waypoints. Once stored, pilot defined data is easily accessed and may be added to the flight plan with a few simple keystrokes. A data transfer unit (DTU) is used to update the navigation database on a 28-day cycle.

Best Computed Position

The best computed position of the UNS is determined by using position inputs from all available

navigation and radio sensors. The UNS searches the database to determine which DME stations are within range (approximately 300 NM) and then sequentially tunes each of those stations. By interrogating multiple DME stations, knowing the geographic coordinates of each station, and correcting the distance computation for slant range using station elevation and aircraft's altitude, the UNS FMS is able to compute the position of the aircraft. The best computed position of the UNS is a weighted average of various sensor inputs, heading information from the compass system, TAS derived from the air data computer and position information derived from the internal GPS sensor. These are integrated with DME derived position to arrive at the best-computed position. If the aircraft is out of DME range, the FMS will rely upon the long-range sensor (GPS) for position information. After the best computed position is obtained, secondary navigational functions such as course to waypoint, ETA, distance to waypoint, wind, and groundspeed are computed for display.

Sensor Monitoring and Integrity

A sensor “watchdog” automatically protects against a large error in a navigation sensor, which could cause the best-computed position to slip. This is accomplished continuously by monitoring the difference between the sensor, or second FMS computed position, and the UNS’s best-computed position. If any difference exceeds a preset value, the “watchdog” will activate the GPS INTEG (GPS integrity) annunciator to alert the pilot. The preset value changes for the different phases of flight, 2.0 nm enroute, 1.0 nm terminal, approach transition, and 0.3 nm final approach.

Navigation and Steering Outputs

Both the computed and raw sensor data is output for display to the flight crew on the UNS FPCDU. The UNS FMS provides desired track, bearing, cross-track, lateral deviation, vertical deviation, and related data to the EFIS system and autopilot system. This information is used for the EHSI and MFD displays and steering



commands from the autopilot/flight director system. The multifunction display (MFD) allows selection and display of flight plan waypoints on the MFD display and input of a joystick waypoint to the UNS FMS. The FMS provides roll steering commands for the autopilot/flight director system. The system anticipates leg changes and provides "smart turns" to eliminate "S" turning. The roll steering command is gain-scheduled based on altitude for gentle turns. The bank limit is configured at time of installation to match autopilot limits.

FUEL MANAGEMENT

The aircraft fuel flow gauges provide the UNS FMS with the inputs necessary to integrate real time fuel management information with the navigational functions. During the pre-departure phase, the pilot inputs the fuel on board the aircraft to initialize the fuel management functions. During flight, the UNS automatically updates the fuel on board gross weight. It also provides continuous estimates of the fuel requirements for the programmed flight plan based upon fuel flow and groundspeed.

SYSTEM CONFIGURATION

The UNS FMS is configured to its specific aircraft installation by the use of a "configuration module" which is part of each UNS aircraft installation. At the time of installation, the configuration module is programmed through keyboard inputs to completely define the sensor input ports, fuel flow type, air data type, EFIS interface, etc.

System Components

The basic UNS-1K/L installation consists of the navigation computer unit (NCU) (containing the navigation computer and the GPS sensor), flat panel control display unit (FPCDU) (containing the display and keyboard), a configuration module, a data transfer unit, and a GPS antenna.

UNS-1K/L Display

The UNS-1K/L utilizes a 4-inch active matrix liquid crystal color flat panel display. The high-resolution display provides alphanumeric characters, icons, and graphics capabilities to facilitate on screen data recognition. The display allows for data display on eleven lines with 24 characters each in two different sizes (Figure 16-27).

UNS-1K/L Keyboard

The full alphanumeric keyboard also contains dedicated function keys and, when used in conjunction with the ten line select keys, provides the operator with all the controls necessary to communicate with the internal navigation computer and all associated sensors. The keyboard allows manual data entry, system mode selection, and control (selection and direction) of navigation sensors.

UNS-1K/L Navigation Computer

The UNS-1K/L contains the system's powerful central processing unit (NAV computer), navigation sensor interface circuits, flight guidance system interface, and the Jeppesen database. The unit processes data from DME, VOR, air data, fuel flow, and the GPS navigation sensor. At installation, the system is configured with the number and type of long-range navigation sen-



Figure 16-27 UNS-1K Flight Management System Flat Panel Display Control Unit



sors, which will optimize the UNS-1K/L system to the aircraft's specific requirements. The GPS NAV sensor act in addition to the multi-channel scanning DME and VOR sensing capability integrated into the basic circuit.

Configuration Module

The configuration module is a serially connected EEPROM designed to store configuration data that is specific to the aircraft installation. The EEPROM is attached to the aircraft side of a UNS-1K/L NCU mounting rack rear connector. The user is able to view and initialize the aircraft configuration through specially designed menus that can be accessed through the UNS-1K/L keyboard. Subsequent modifications to the configuration are possible but restricted through software interlocks. The configuration module receives and transmits data via private serial communications buses.

Data Transfer Unit (DTU)

The data transfer unit (DTU) is a floppy disk drive housed within a protective case that is mounted in the pedestal for convenient access. It is designed to update the Jeppesen databases within minutes. The update is contained on 3-1/2 inch floppy disks or zip disks, which are distributed periodically. The DTU receives power from a dedicated circuit breaker and communicates directly with the UNS-1K through a digital bus.

The DTU may also load pilot-defined flight plans and checklists created with off-line flight planning software loaded into an IBM compatible computer. FMS in-flight data parameters can be written to a disk utilizing the DTU. The maintenance log stored in non-volatile memory also may be written to a disk.

GPS Antenna

The GPS antenna receives the satellites signals and is located on the roof of the cockpit for an unobstructed view of the satellites. The antenna is an active unit in that it amplifies the received

signal, the amplifier being powered from the GPS receiver through the coax cable.

Annunciators

Standard annunciations provided by the UNS FMS are: FMS HDG (heading), FMS XTK (selected cross-track), FMS MSG (message), WPT (waypoint), FMS APPR (approach) and GPS INTEG (GPS integrity). Annunciator outputs are in the form of discretes to drive remote annunciators and annunciator signals are also provided on the data buses to the EFIS systems.

UNS-1K/L System Outputs

The UNS-1K/L provides digital outputs to the EFIS-85B(14) system and the EHSI-74A system for the display of FMS data. Analog outputs are provided to the autopilot system and discretes to selected annunciators.

EFIS-85B(14) (Pilot's Displays)

For the EFIS-85 system ARINC 429 digital data is supplied to the DPU and MPU for display of FMS data. The EFIS displays will display all the required FMS annunciations on its displays apart from GPS INTEG which is a remote annunciator above the pilot's EADI display. The FMS displays are selected by selecting FMS 1 as the NAV sensor source for display on the EHSI. The EFIS system will display the lateral enroute deviation, VNAV vertical deviation and the pseudo ILS display during approach.

EHSI-74A (Copilot's Displays)

For the EHSI-74A system ARINC 561 digital data is supplied to the HPU for display on the EHSI. The EHSI display will display all the required FMS annunciations apart from FMS HDG, FMS MSG and GPS INTEG, for which there are remote annunciators below the copilot's EHSI. The FMS display is selected by remote annunciator switches to select between NAV 2 or FMS as the NAV sensor source displayed on the EHSI. The EHSI will only



display lateral deviation, there is no vertical deviation display.

APS-65H/J Autopilot

The flight director/autopilot receives roll steering commands for enroute navigation, the flight director/autopilot cannot use the VNAV commands. During the approach phase of operations (pseudo ILS) analog DC signals for lateral and vertical deviation is provided for a 3D approach. The flight director/autopilot will track these signals just as it would for an ILS approach. Relays are used to switch the lateral and vertical deviation signals from the DPU to the FMS NCU when FMS 1 is selected as the active NAV sensor on the EHSI.

ELECTRONIC FLIGHT INSTRUMENT SYSTEMS (EFIS)

INSTALLED EFIS SYSTEMS

The B300 (350) aircraft can be equipped with a Collins EFIS-85B(14) (3-Tube) electronic flight instrument system (EFIS) for the pilot's display and a Collins EHSI-74A electronic horizontal situation indicator for the copilot's display. There are other EFIS configurations available, but this is the most common EFIS configuration found in these aircraft (Figure 16-28).

EFIS-85B(14) System

In this system, attitude and navigation information are displayed on color CRTs which serve the functions of electronic attitude director indicator (EADI) and electronic horizontal situation indicator (EHSI). Additional information, such as weather radar, NAVAID/waypoint locations, flight control system (FCS) mode annunciation, autopilot/yaw damper engage status, decision height, and diagnostic messages may also be displayed. The system also includes a multifunction display (MFD) to provide the pilot additional information along with the primary flight displays.

The system gives a pilot the capability to display information available in the area of his central scan by allowing the pilot to select or deselect information depending on the regime of flight, and by providing the pilot a means of easily seeing the interrelationships of dynamically changing flight data.

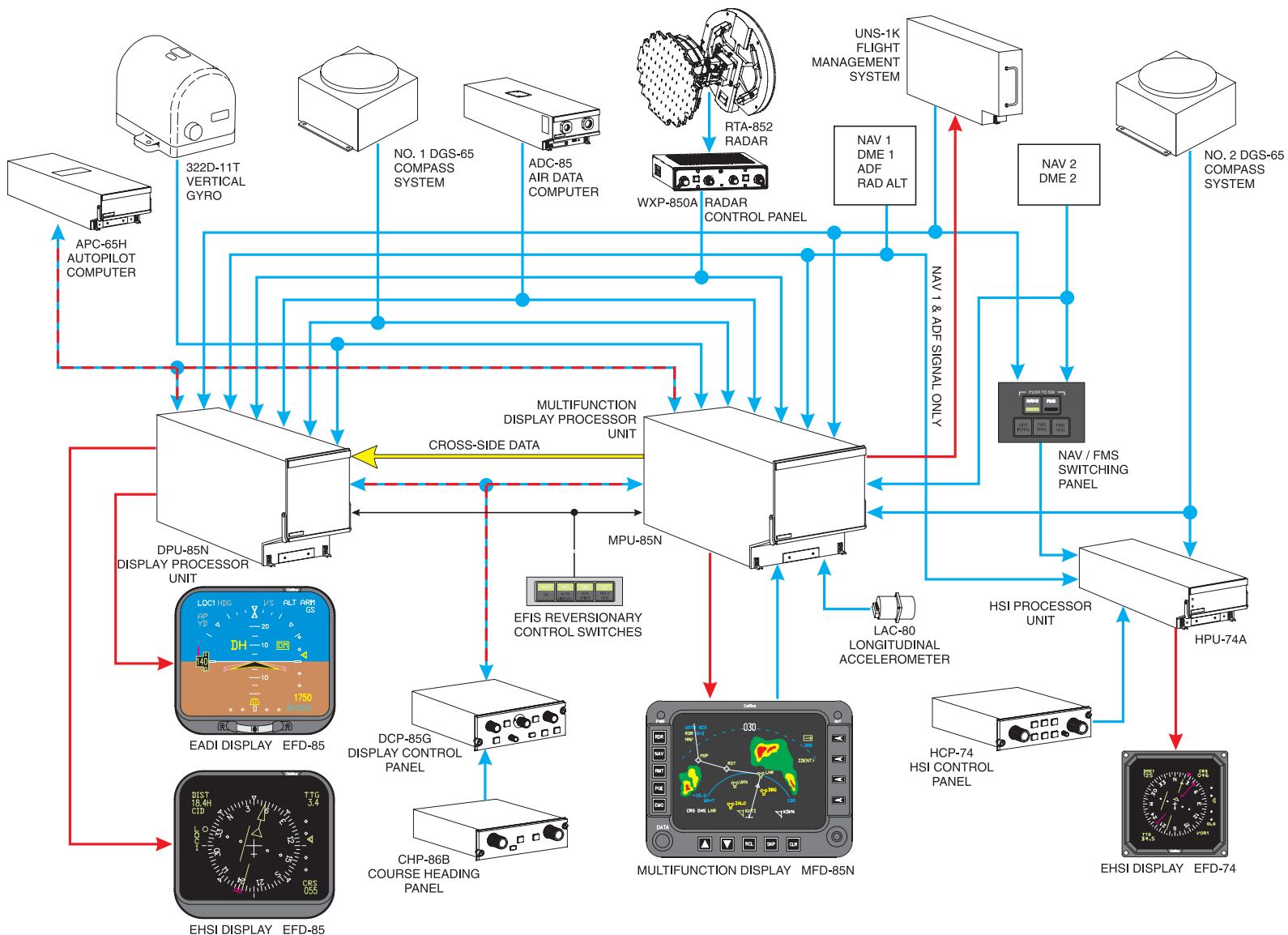


Figure 16-28 EFIS System EFIS-85B (14) and EHSI-74A



CONTROL PANELS

The two control panels associated with the EFIS system are the DCP-85G, and the CHP-86B (Figures 16-29 and 16-30).

DCP-85G Display Control Panel

The DCP-85G provides EHSI display sensor selection control, EHSI display format control, EADI and EHSI display dimming control, and EADI radio altitude decision height set capability (Figure 16-29). The DCP provides the pilot with the controls needed to select the various operating formats and functions for the electronic horizontal situation indicator (EHSI). The only other DCP controls associated with the EADI are the DH SET knob and the radio altim-

eter test (RA TST) pushbutton. In addition to the front panel controls, the DCP receives and stores course and heading information from the CHP in nonvolatile memory, and transmits it to the DPU-85N display processor unit. This same information is also transmitted to the MPU where it is processed for display on the MFD. Appropriate DCP mode status information (i.e., decision height, HSI mode, and status of the auxiliary data, bearing pointer, and course arrow/deviation bar) will be stored in nonvolatile memory when power is removed from the system. This information is used to initialize the EFIS system in the same configuration that it was in prior to power off.

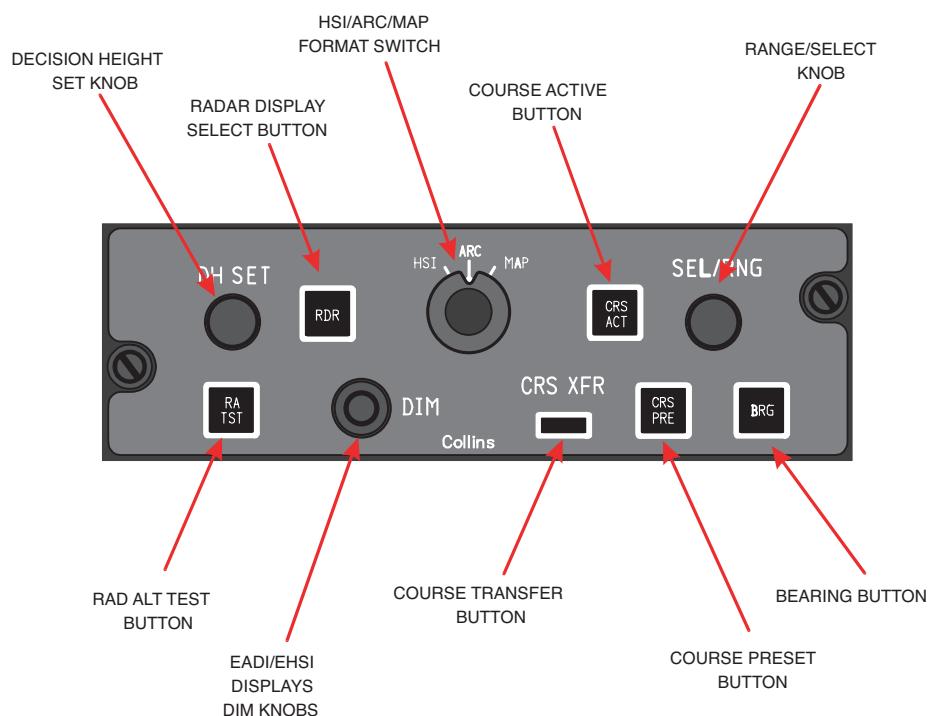


Figure 16-29 Display Select Panel Controls DCP-85G



CHP-86B Course Heading Panel

The CHP-86B provides course and heading select functions for the EFIS system (Figure 16-30). The CHP-86B provides a single CRS select control and a single HDG select control. The panels also provide for selection of course direct to and heading sync. Additionally, groundspeed (GSP), time-to-go (TTG), and elapsed time (ET) are selected from the CHP. An ET button for initiating and completing timing functions is also provided.

SELF TEST MODE

In addition to the on-line monitors, it is possible to place the EFIS system in a self test mode by the use of an EFIS test switch.

When the test mode has been activated, the button and knobs on the DCP select various submodes. Test functions available include:

Confidence Test

When the EFIS test switch is pressed an increment of $+10^\circ$ is added to pitch and roll on the EADI display (Figure 16-31). A $+20^\circ$ increment is added to the heading on the EHSI. The word

TEST in red characters is written across the EADI above the aircraft symbol while the test is in process (Figure 16-33). If the test switch is held for longer than four seconds, the 10° increments are removed and all EADI and EHSI flags are brought into view. This flag condition will remain until either the test switch is released or the RA TST button on the DCP is pressed to enter a sequence of diagnostic/maintenance routines (DMRS).

EADI Display Formats

The EADI shows the familiar attitude display with lateral and vertical computed steering signals needed to give the commands to intercept and maintain a desired flight path (Figure 16-32). The EADI provides the following display information:

Attitude Display

The pitch position is indicated by the sky-ground area and the pitch scale. The roll pointer moving in relationship to the roll scale indicates roll.

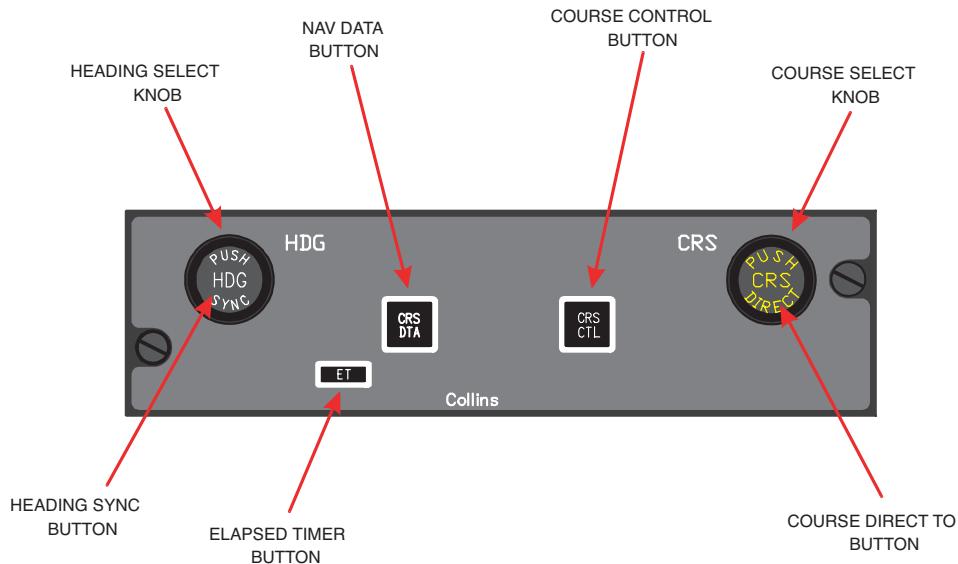
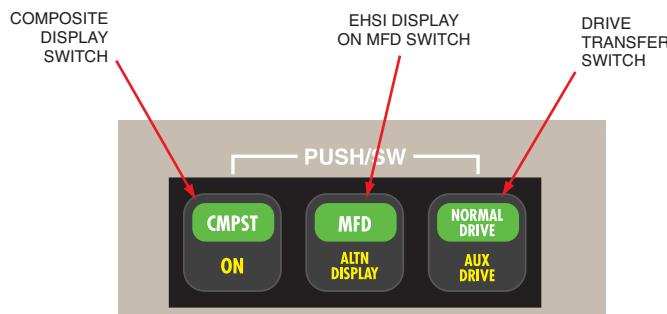
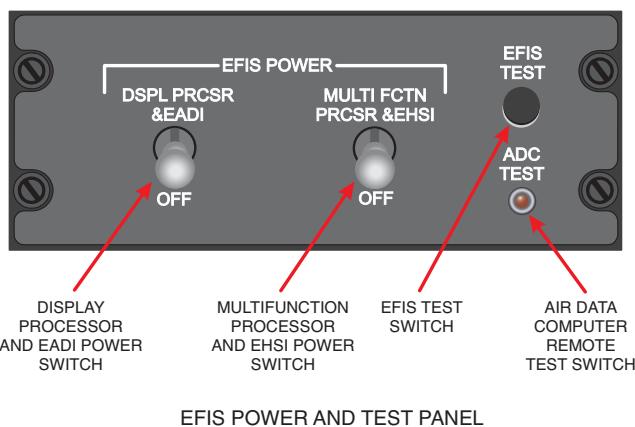


Figure 16-30 EFIS System Control Panel CHP-86B



EFIS REVERSIONARY SWITCH/ANNUNCIATOR CLUSTER (LATER S/N AIRCRAFT)

Figure 16-31 EFIS System Control Panels

Aircraft Symbol

The aircraft symbol is a stationary representation of the aircraft in reference to the sky ground display and the flight director command bars.

Flight Director Commands

The command bars appear when the flight director is being used and provide pitch and roll commands to capture and maintain a desired flight path. The aircraft symbol should be inside the command bars “vee” when tracking flight director commands.

FCS Mode Annunciation

The flight director vertical and lateral modes are annunciated along the top of the EADI, vertical modes to the right and lateral modes to the left.

Green annunciators indicate active modes and white indicates armed modes.

Autopilot/Yaw Damper Engage Status

The autopilot/yaw damper annunciators are displayed at the upper left side of the EADI. When the autopilot is engaged, a green AP annunciator and a green YD annunciator appears. A trim in motion annunciator TRIM will also appear when the trim is running on the right side of the screen.

Vertical Deviation

A scale and pointer on the right side of the display indicates glideslope, FMS VNAV or altitude hold deviation. The pointer symbol changes for the different sensor inputs. The glideslope vertical

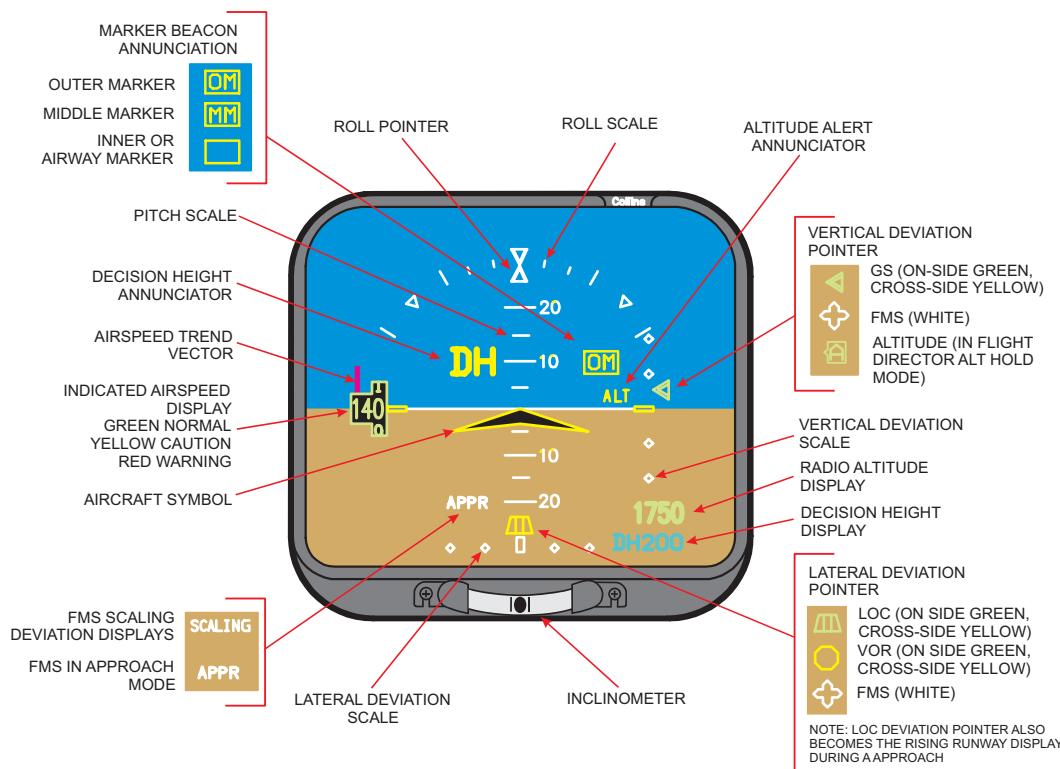


Figure 16-32 EFIS-85B (14) EADI Displays and Indications

deviation display is removed when LOC mode is not selected or during backcourse approaches.

Lateral Deviation

A scale and pointer at the bottom of the EADI display indicates deviation from a selected lateral navigation path. The sensors are selected on the EHSI and can be VOR/LOC or FMS. The pointer symbol changes to indicate either VOR/LOC or FMS is driving the display. When on approach, the pointer changes to a rising runway expanding as the aircraft descends, as detected by RAD ALT.

Radio Altitude

Radio altitude is displayed in the lower right corner as a digital readout. The RAD ALT display only appears when the radio altimeter system is within range or in self test mode.

Decision Height

Selected decision height is displayed in the lower right corner of the display and is set by the DH SET knob on the DSP. When decision height +50 feet is reached, the selected decision height digits flash. At decision height, a flashing annunciator DH appears on the EADI.

Marker Beacon

Marker beacon passage is indicated by the boxed OM (outer marker), MM (middle marker) and a yellow box for airway or inner marker appearing on the right upper side of the display.



Indicated Airspeed

IAS, as transmitted by the ADC-85, shows as a green numerical readout surrounded by a green T-shaped box on a black background at the left center of the EADI. At airspeeds below 30 knots, the numerical readout shows three green dashes instead of the numbers. If at the present acceleration, V_{MO} would be exceeded in 10 seconds or less, or if V_{MO} is exceeded, and at present acceleration will remain exceeded in 10 seconds or less, the IAS readout changes to yellow and the box changes to yellow and flashes. Exceeding $V_{MO} + 2$ knots, changes the IAS readout to red, the box to red, and flashing at twice the rate of the V_{MO} warning.

Airspeed Trend

The airspeed trend display has two parallel magenta bars extending upward or downward from the digital IAS readout. The bars extend upward for increasing airspeed and downward

for decreasing airspeed. The vector starts at 0.1 knots and stops at 1.3 knots when a horizontal line shows across the end of the bars to show that the trend indicator has reached its maximum limit. The airspeed trend only shows on the display if the IAS is 50 knots or greater. The trend vector is derived from actual IAS complemented with longitudinal acceleration and pitch.

Inclinometer

The inclinometer or slip indicator consists of a weighted ball in a liquid-filled tube. It is attached to the lower front of the EADI display and is used as an aid to coordinate turns.

Warning Flags

The following warning flags can appear if the EFIS system detects a fault (Figure 16-33):

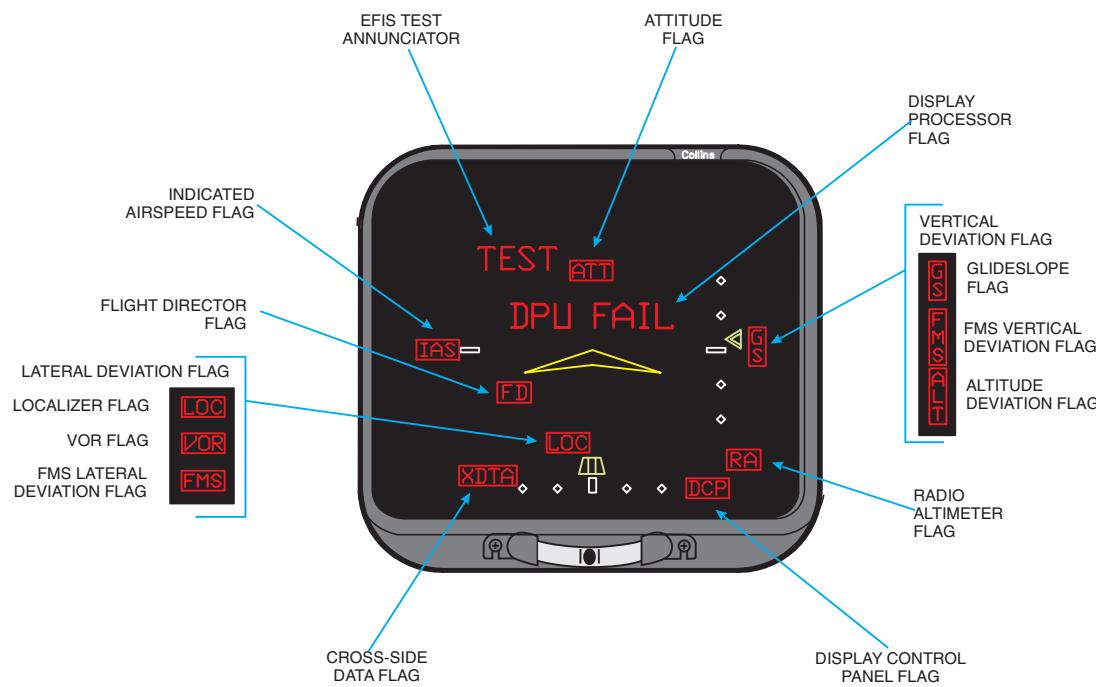
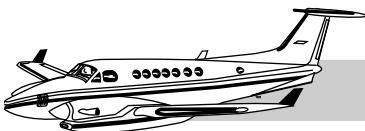


Figure 16-33 EFIS-85B (14) EADI Warning Flags



Attitude Flag

If a failure of the attitude sensor is detected, the pitch scale, roll scale, roll pointer, sky/ground display, and command bars disappear and a red box with the letters "ATT" inscribed appears above the aircraft symbol. The ATT flag remains in view until an alternate sensor is selected or until the fault is cleared.

Display or Multifunction Processor Unit (DPU or MPU) Flag

If the DPU or MPU fails, the non-flashing inscription "DPU FAIL" or "MPU FAIL" appears in red centered on the display depending on which unit is currently driving the display. If the flag remains in view more than five seconds, the entire display blanks except for the DPU or MPU flag.

Vertical Deviation Flag

If a failure of the glideslope receiver is detected, a red box with the letters "GS" inscribed appears near the vertical deviation scale. The pointer and scale are removed from view. Detection of a FMS VNAV failure turns off the scale and pointer, and shows a flag consisting of a red box with "FMS" inscribed.

Radio Altimeter Flag

If a failure of the radio altimeter system is detected, a red box with the letters "RA" inscribed replaces the radio altitude display. The DH set display and the DH annunciator is also removed from view.

Display Control Panel Flag

If a failure of the display select panel occurs, a red box with the letters "DCP" inscribed appears in the lower right corner replacing the DH set display. The EFIS-85% system continues to operate in the mode that was active prior to the DCP failure.

Lateral Deviation Flags

If a sensor failure is detected by loss of the valid or by an internal monitor, the appropriate red boxed letters LOC, VOR, or FMS appear above the scale. The scale and pointer are removed from view.

Cross-Side Data Flag

If a failure of the cross-side data bus occurs, a red box with the letters "XDTA" inscribed appears in the lower left of the EADI. Data from the cross-side sensors is no longer available, and any display driven by cross-side data is flagged.

Indicated Airspeed Flag

Detection of a failure of the digital air data system or invalid data turns off the IAS readout, trend vector and shows the IAS flag in a red box. The flag flashes for 10 seconds then stays on. The air data source self test shows the ADC TEST annunciator in red next to the airspeed display.

Flight Director Flag

If the flight director system fails, the command bars disappear and a red box with the letters "FD" inscribed appears at the lower left of the aircraft symbol.

Test Annunciator

The test annunciator appears when the EFIS system is in test mode.

EHSI Display Formats

The EHSI brings together numerous displays to show a map-like presentation of the aircraft's horizontal navigation situation. The EHSI shows aircraft heading and aircraft displacement relative to VOR radials, localizer, and glideslope beams. Lateral and vertical deviation from the FMS system may also be shown.

The EHSI can display three formats that are selected with the format switch on the DCP-85G display control panel. These are the HSI (full compass rose), ARC (approach or sector), and MAP (enroute or map) formats. The EHSI provides the following display information on the HSI, ARC, or MAP formats except where noted otherwise.



Aircraft Symbol

The delta-shaped symbol represents a plan view of the aircraft pointing vertically up the display.

Compass Display (HSI Format)

The compass display is a full 360 compass rose with the aircraft symbol at the center. Aircraft heading is read against the lubber line (Figures 16-34 and 16-35). During normal compass operation there is no annunciation.

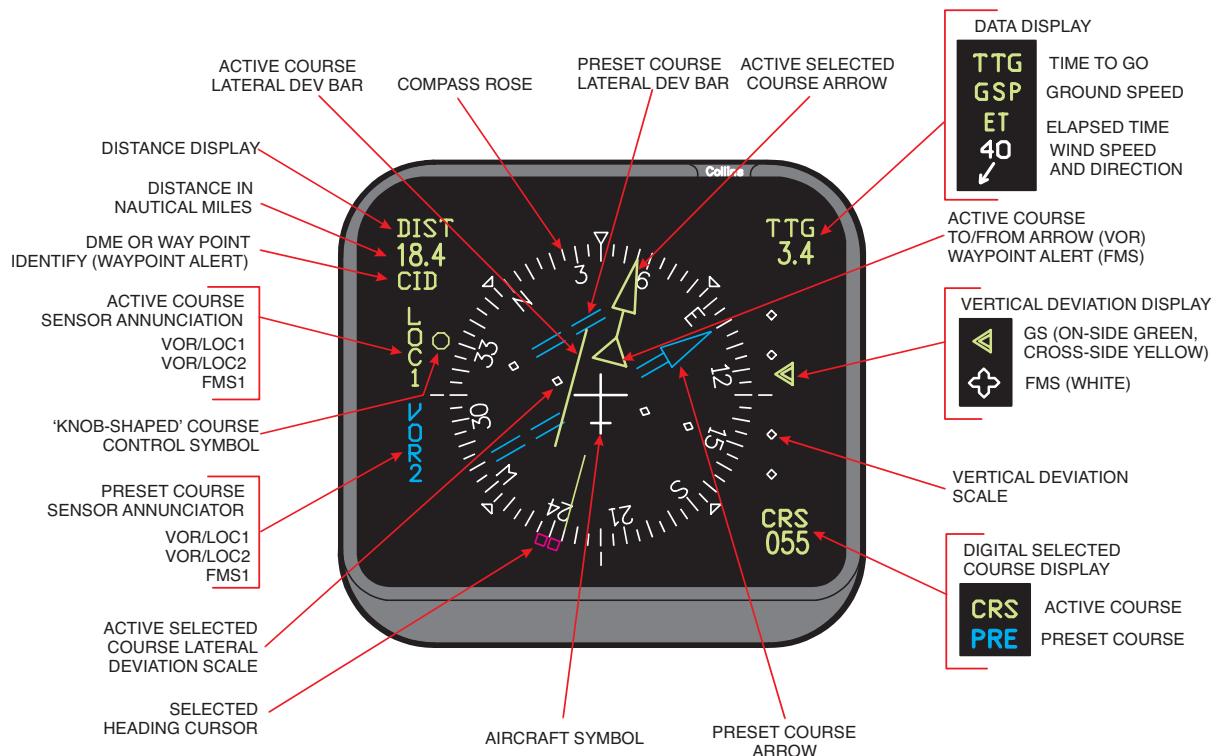


Figure 16-34 EFIS-85B (14) EHSI HSI Display and Indications

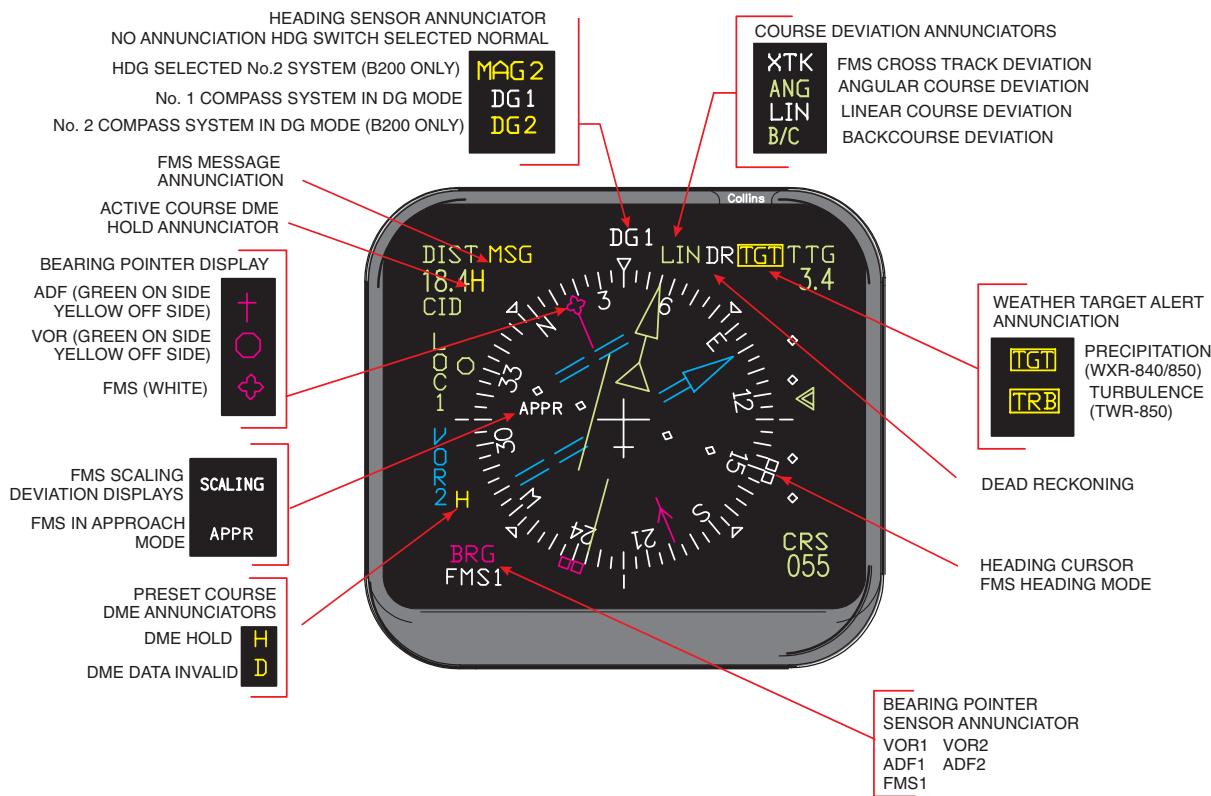
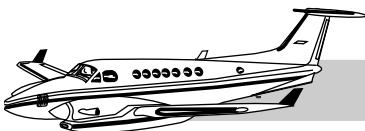


Figure 16-35 EFIS-85B (14) EHSI HSI Display and Indications

Compass Display (ARC and MAP Formats)

The compass display is an expanded sector of $\pm 30^\circ$ with the aircraft symbol placed at the bottom center of the display. All other functions of the compass display are the same as the HSI format (Figures 16-36 and 16-37).

Selected Heading Cursor

The selected heading is shown as two adjacent magenta rectangles with respect to the compass card. The HDG knob on the DCP is used to select the heading.

Active Selected Course Display

Active selected course is shown by the relationship of the course arrow with respect of the compass card. The active selected display is digitally repeated in the lower right corner of the display. The CRS knob on the DCP is used to select the active course in VOR/LOC mode when the course select knob is in ACT. In FMS mode the active course pointer is driven directly by the FMS system.

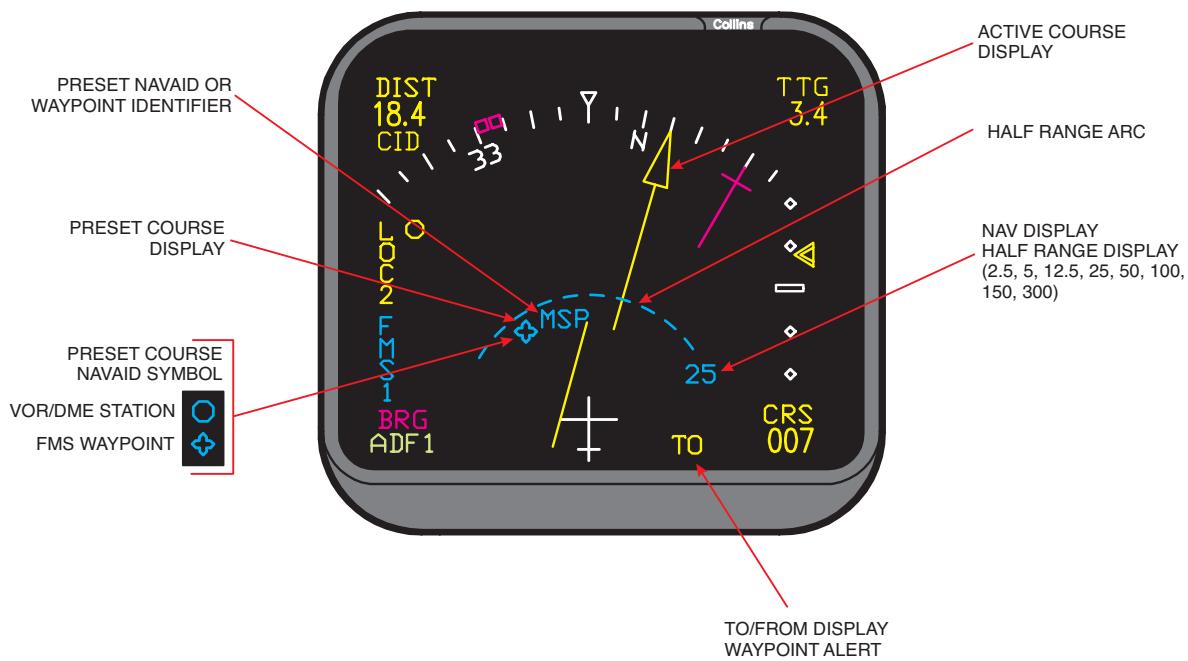


Figure 16-36 EFIS-85B (14) EHSI ARC Display and Indications

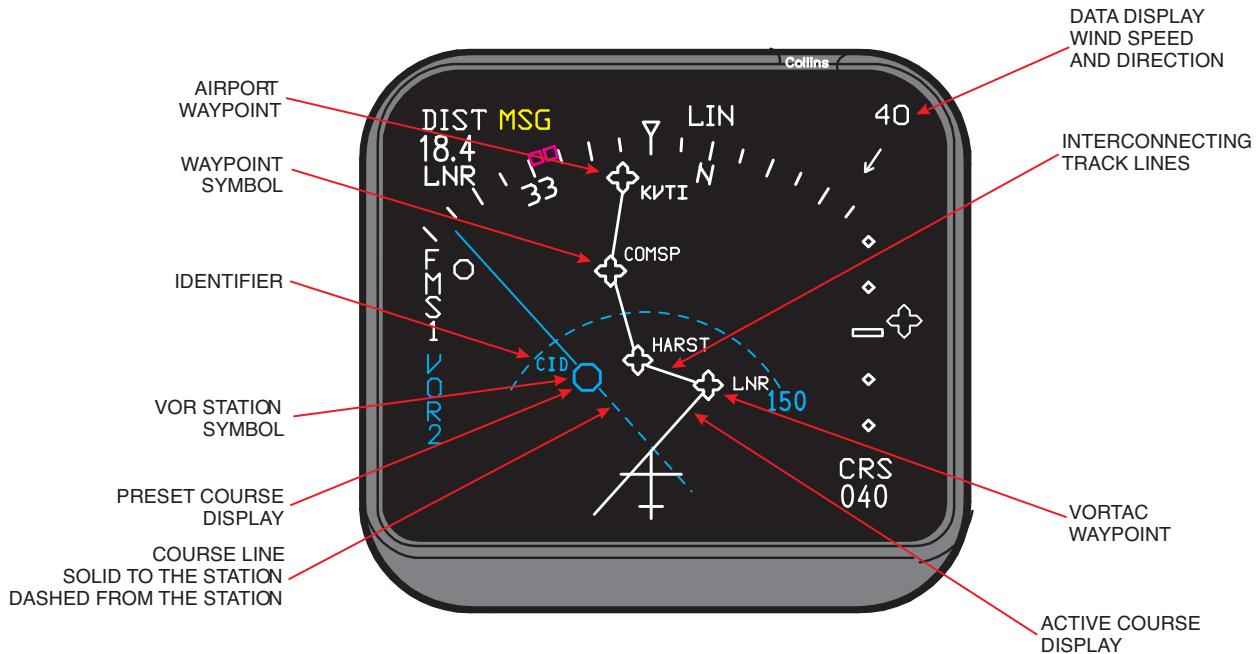


Figure 16-37 EFIS-85B (14) EHSI MAP Display and Indications



Active Course Lateral Deviation Display

The center portion of the active selected course arrow is the lateral deviation bar, which represents the centerline of the selected course. This bar moves laterally with respect to the course arrow and the white deviation dots indicate deviation from the selected course.

Annunciation of Lateral Deviation Type

LIN (linear) is annunciated when FMS is the active selected course indicating the deviation scale is indicating linear deviation from the center of track. VOR mode deviation can be angular or linear depending if DME distance is available, so the annunciation can be ANG (angular) or LIN. LOC mode is always angular with no annunciator. B/C is annunciated when the EFIS system automatically detects a backcourse approach and the deviation display has been reversed. XTK (cross track) is annunciated when the FMS system is flying an offset track.

Active Selected Course Navigation Sensor Annunciation

The active selected course navigation sensor is annunciated in the left side of the display. The annunciators are VOR/LOC 1, VOR/LOC 2, and FMS 1. The CRS buttons/knobs on the DCP selects the active course.

Active Selected Course To/From Arrow (HSI Format)

The to/from arrow indicates that the selected course is either TO or FROM the selected NAVAID when the active navigation sensor is VOR. When the active source is FMS the arrow indicates the course to the next waypoint and acts as the waypoint alert by flashing when approaching the waypoint. When LOC is the active source, the to/from arrow is removed.

Active Selected Course TO/FR Annunciation (ARC and MAP Formats)

To/from is indicated by the letters TO or FR appearing in the lower part of the display. The TO annunciator flashes to indicate waypoint alert.

Preset Course Arrow and Course Deviation Bar in HSI Format

Preset selected course is shown by the relationship of the magenta dashed course arrow with respect of the compass card. The CRS knob on the DCP is used to select the preset course in VOR/LOC mode when the course preset button is active. In FMS mode the preset course pointer is driven directly by the FMS system. The preset course indication can be turned off. No deviation scale or to/from information is provided with the preset course.

Preset Course Navigation Sensor Annunciation

The preset selected course navigation sensor is annunciated in the left side of the display below the active course annunciation. The annunciators are VOR/LOC 1, VOR/LOC 2, and FMS 1. The SEL knob on the DCP selects the preset course with the course preset button active.

Vertical Deviation Display

A scale and a pointer on the right side of the display indicates glideslope deviation or FMS VNAV (pseudo GS) deviation. The GS vertical deviation display is removed when LOC mode is not selected or during backcourse approaches.

Bearing Pointer Display and Sensor Annunciation

A single bearing pointer is provided to indicate bearing to the selected NAVAID or next waypoint. The bearing pointer sensor is indicated by a symbol placed on the end of bearing pointer for ADF, VOR or FMS. The bearing pointer is turned on and off by the bearing button on the DCP. The SEL (select) knob is used to select the sensor to drive the pointer. Annunciators in the lower left of the display indicate the selected sensor.



Distance Display and Identifier Display

Distance to the selected DME station and DME station ident are displayed below the letters DIST when VOR/LOC is the active source. When FMS is the active sensor displayed distance is to the next waypoint and the ident display is the next waypoint. An "H" annunciator appears next to the digits when the DME is in hold.

Navigation Data Display (GSP/TTG/ET)

The navigation data display indicates ground-speed and time-to-go as supplied by the active sensor source DME or GPS. The elapsed time is a counter from within the EFIS system. The navigation data select button on the CHP is used to select the desired data.

MSG Annunciation

The MSG annunciation comes from the FMS to indicate the FMS has a message for the pilot to read on the FMS control display unit.

Approach (APPR) and Scaling (SCALING) Annunciators

The approach annunciator will appear when the FMS has gone into the approach mode of operation and is providing pseudo ILS deviation signals on approach to a runway. The scaling annunciator appears when the FMS is adjusting the lateral deviation scale from enroute to terminal and then again from terminal to approach.

FMS Heading Cursor

When the FMS is in heading mode of, a white heading cursor appears on the compass rose to indicate the selected heading from the FMS.

Weather Radar Target Alert

The weather radar target alert can be a boxed TGT or a boxed TRB depending on the weather radar installed and the target alert provided from the weather radar.

ARC and ARC/WX (Approach or Sector) Format

The ARC format has an expanded compass segment across the top of the display with the aircraft symbol relocated to the bottom center of display. The active course display is the same as the HSI mode. The preset course VOR-DME/waypoint symbol with identifier or number, is placed in rho-theta position with respect to the aircraft symbol and selected range. The range knob on the DCP in the ARC format controls range displayed.

The weather radar display can be displayed by pressing the radar button on the DCP. WX range arc and mode annunciation is shown along with the weather radar returns. In the ARC/WX format the range is controlled by the radar control panel.

MAP and MAP/WX (Enroute or Map) Format

The MAP format also uses the expanded compass segment and places VOR and/or waypoint symbols for both active selected and preset course in proper rho-theta position with respect to the aircraft symbol and selected range.

The selected course lines are drawn through the symbols (active and preset) waypoint strings (up to five waypoints) can be shown with their numbers or identifiers next to the symbol and with the symbols connected by desired track lines. The range knob on the DCP in the MAP format controls range displayed and DME ident is now shown next to the VOR/DME symbols.

Weather radar display, range selection same as in ARC/WX format and the range is controlled by the radar control panel.



Warning Flags

The following warning flags can appear if the EFIS system detects a fault (Figure 16-38):

Distance Flag

If invalid distance data is detected from a DME or GPS, the distance digits and “NM” annunciation are replaced with dashes that are the same color as the active course display. If the digital bus monitor detects an inactive bus, the data or dashes are removed from the display.

Heading Flag

If a failure of the heading system occurs, a red box with the letters “HDG” inscribed appears to the left of the lubber line. If the heading synchro

or bus monitor detects a failure, the HDG flag appears and the compass card is frozen in position. Nothing is removed from the EHSI when a HDG flag appears.

Display or Multifunction Processor Unit (DPU) or (MPU) Flag

If the DPU or MPU fails, the nonflashing inscription “DPU FAIL” or “MPU FAIL” (depending on which unit is driving the display) appears in red centered on the display. If the flag remains in view more than five seconds, the entire display blanks except for the DPU or MPU flag.

Data Flags

If invalid time-to-go, or groundspeed is detected from a DME or GPS, the digits are replaced with

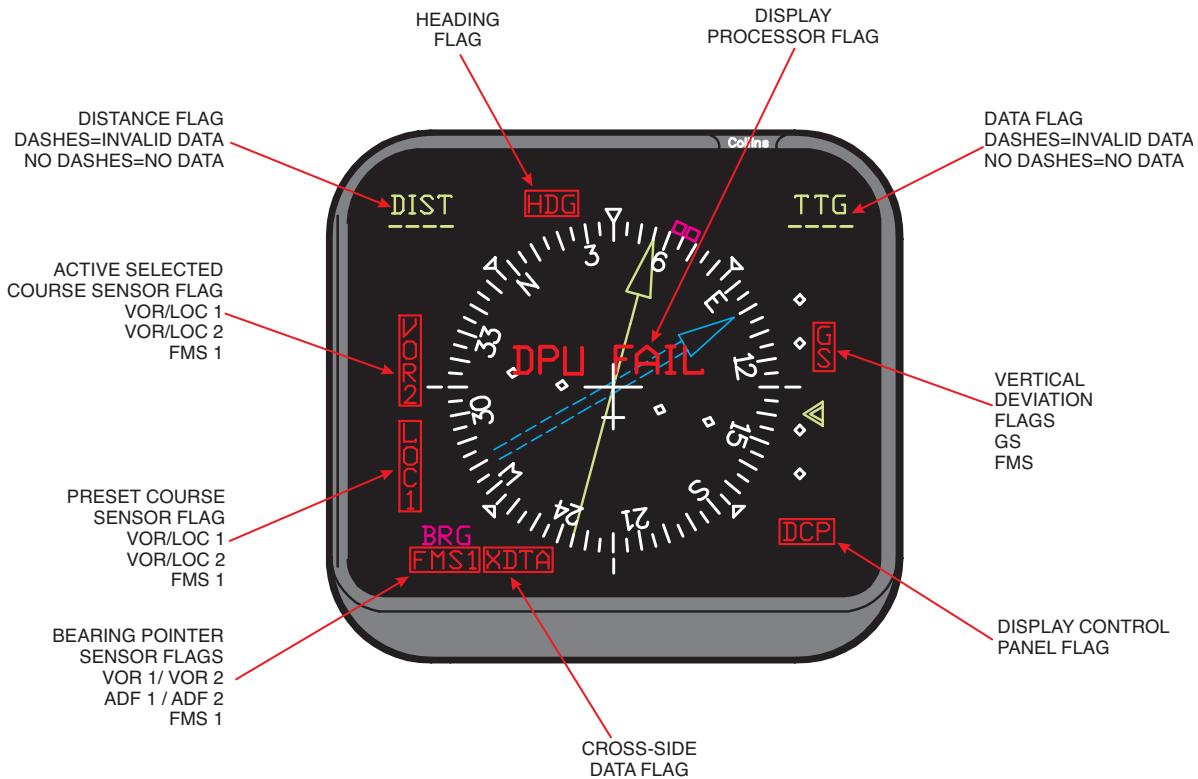


Figure 16-38 EFIS-85B (14) EHSI Warning Flags



dashes that are the same color as the active course display. If the digital bus monitor detects an inactive bus, the data or dashes are removed from the display.

Vertical Deviation Flag

If a failure of the glideslope receiver is detected, a red box with the letters "GS" inscribed appears and the scale and pointer are removed from view. Detection of a FMS VNAV failure turns off the scale and pointer and shows a flag consisting of a red box with "FMS" inscribed.

Display Control Panel Flag

If a failure of the display control panel occurs, a red box with the letters "DCP" inscribed appears in the lower right corner replacing the digital selected course readout. The EFIS-85% continues to operate in the mode that was active prior to the DSP failure.

Cross-Side Data Flag

If a failure of the cross-side data bus occurs, a red box with the letters "XDTA" inscribed appears in the lower left of the EHSI. Data from the cross-side sensors is no longer available, and any display driven by cross-side data is flagged.

Bearing Pointer Flag

If a bearing pointer sensor failure occurs, the sensor annunciation in the lower left corner of the EHSI becomes red and boxed, and the bearing pointer is removed from view.

Active Selected Course Flag

If a navigation sensor failure is detected while it is selected for the active selected course, the active selected course sensor annicator becomes red and boxed. If VOR valid signal is lost, nothing is removed from the display. If FMS valid signal is lost, the to/from display is removed. Loss of FMS or VOR bearing data removes the to/from display and centers the deviation bar. Loss of FMS track data freezes the course arrow. If they are being displayed, NAVAID symbols and connecting lines are also removed from view.

Preset Course Flag (HSI Format)

If a navigation sensor failure is detected while it is selected for the preset course, the preset course sensor annicator becomes red and boxed. The preset course arrow deviation bar is removed from view.

Preset Course Flag (ARC and MAP Formats)

If a navigation sensor failure is detected while it is selected for the preset course, the NAVAID symbol (ARC format) or NAVAID symbols and connecting lines (MAP format) are removed from view and the preset course navigation sensor annicator becomes red and boxed.

MFD-85B DISPLAY FORMATS

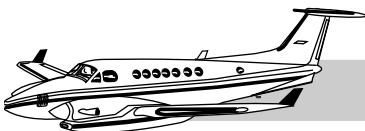
General

The MFD is a flight-planning unit used to increase the operational capabilities of the EFIS-85B system. Standard functions displayed include weather radar, three selections of pictorial navigation maps (heading up, north up with aircraft centered, and north up maximum view), and page data (normal and emergency checklists).

As an additional function, pages of extended data may also be displayed when selected from the remote menu.

If none of the format buttons on the left side of the MFD have been pressed, the display consists of the following:

- Aircraft symbol
- Dashed full-range arc
- Digital heading readout (actual aircraft heading)
- Heading type and sensor annicator



MFD Functions

Weather Radar

Pressing the RDR button on the MFD allows detected precipitation from the weather radar system to be displayed on the MFD (Figure 16-39). Weather mode, range, and other system operating parameters are selected on the WXP-850A.

When the RDR button on the MFD is pressed, the MFD shows a “heading up” navigation display with weather radar data superimposed. Weather radar data does not appear if the MFD is in the NAV mode and showing a “north up with aircraft centered” display, a “north up max view” display, or the “map plan” display.

Navigation Information

Three pictorial navigation displays (maps) may be selected for display from the NAV menu. The three displays are the “heading up”, the “north up with aircraft centered”, and the “north up maximum view” displays. All three displays (maps) are capable of simultaneously displaying the location of two VOR/DME stations and, from the FMS, a waypoint string of up to five waypoints, up to seven closest NAVAIDS, and up to seven closest airports. The NAV menu is used to select the navigation map for display.

The “heading up” map may be displayed with or without the overlay of weather radar information by alternately pressing the RDR button on the MFD (the weather radar system must also be in a weather detection mode). If RDR is selected

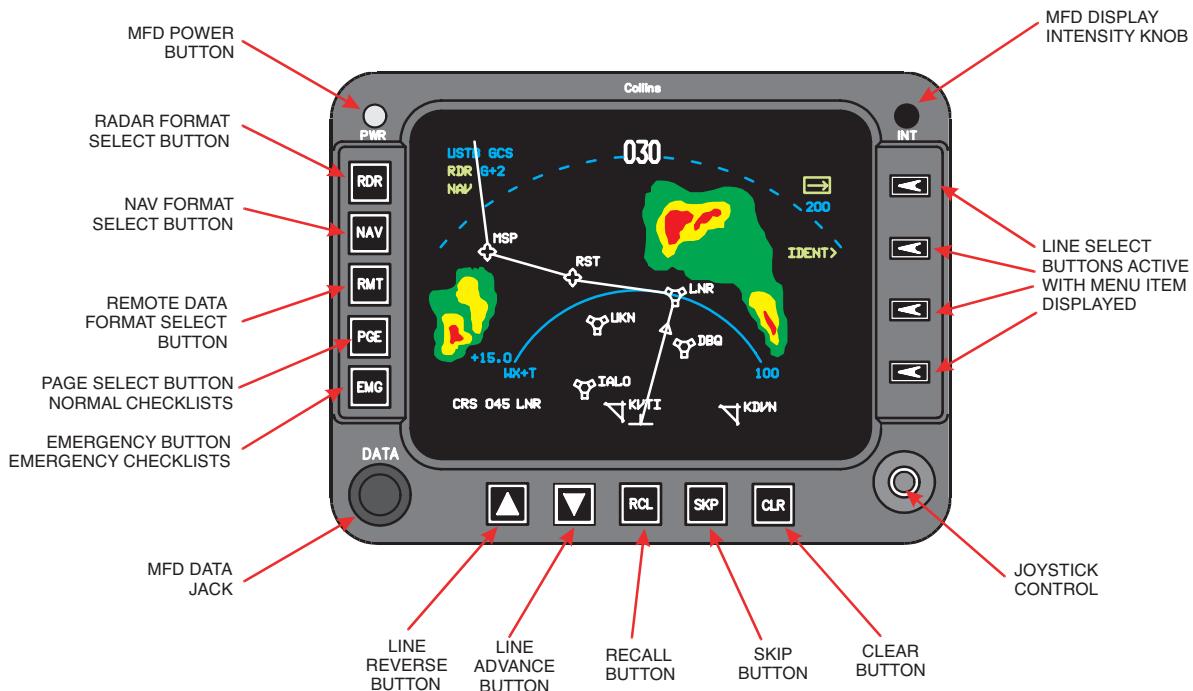


Figure 16-39 MFD Controls With NAV and Radar Display



while displaying the “north up with aircraft centered”, or the “north up maximum view” map, or “map plan”, the MFD automatically switches to the “heading up” map with weather radar information displayed. If RDR is then deselected, the MFD returns to the previously selected navigation map display.

Map range is set from the WXP when radar is selected. In NAV only modes, range is set by the line advance or line reverse keys at the bottom of the MFD front panel and the range arc is dashed. When entering the NAV only mode, the range is the last range selected in that mode. NAV in green is annunciated in the upper left corner of the display.

Heading Up

The heading up display has the aircraft symbol centered at the bottom of the display, and a digital heading readout centered at the top of the display (Figure 16-40). No compass arc is displayed.

North Up With Aircraft Center

The north up with center display always has true north at the top of the display (Figure 16-41). North up is annunciated with an N. The white aircraft symbol is located in the center of the display and rotates as the heading of the aircraft changes.

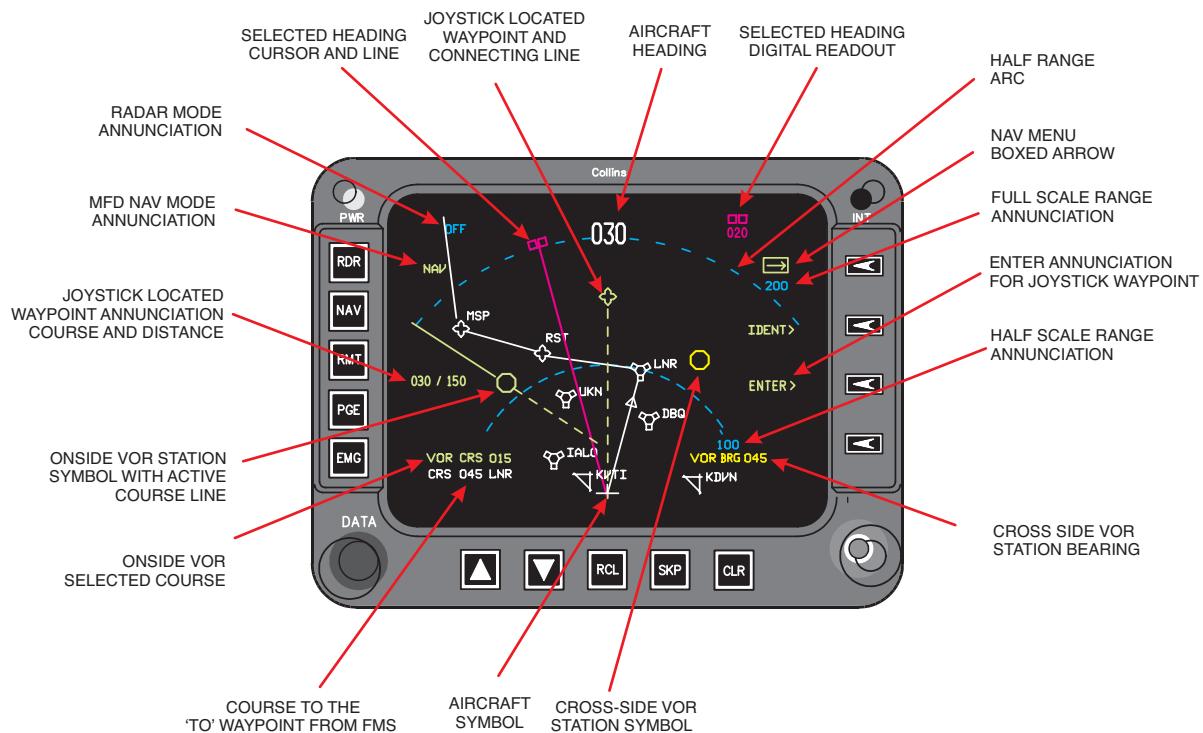


Figure 16-40 NAV Display Heading Up Format

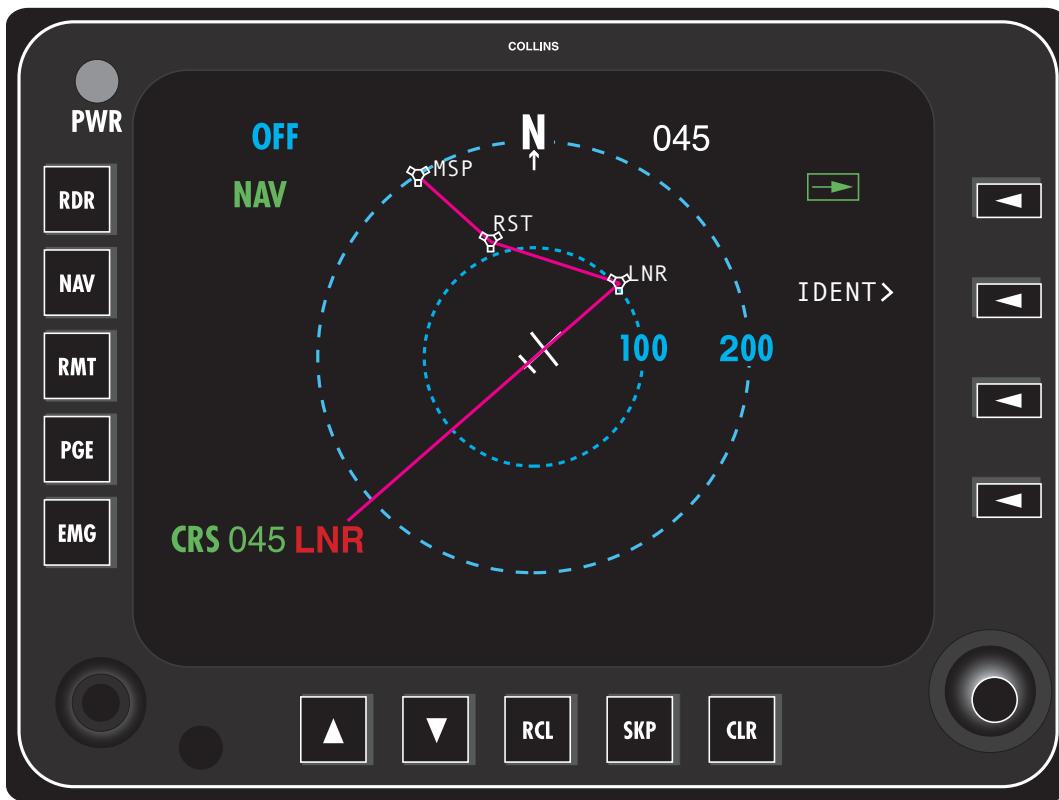


Figure 16-41 NAV Mode North Up With Aircraft Centered Format

North Up Maximum View

The north up maximum view display always has true north at the top center of the display (Figure 16-42). North up is annunciated with an N. The white aircraft symbol is positioned on an imaginary circle near the edge of the display so the greatest amount of area is shown in front of the aircraft.

NAV Menu

The NAV menu is used to select the format of the navigation display, the sensors being displayed on the navigation display and the heading display (Figure 16-43). Push the top line select key (the key on the right hand side of the display with the green boxed arrow pointing toward it) to turn on

the NAV menu mode. When NAV menu is being displayed the following selections can be made:

To select the NAV display mode format, push the following buttons to select the menu item next to that button.

- RDR button Radar display only
- NAV button Heading up display
- RMT button North up with aircraft center
- PGE button North up with maximum view

The display changes to show the selected format with the selected sensor(s) data.

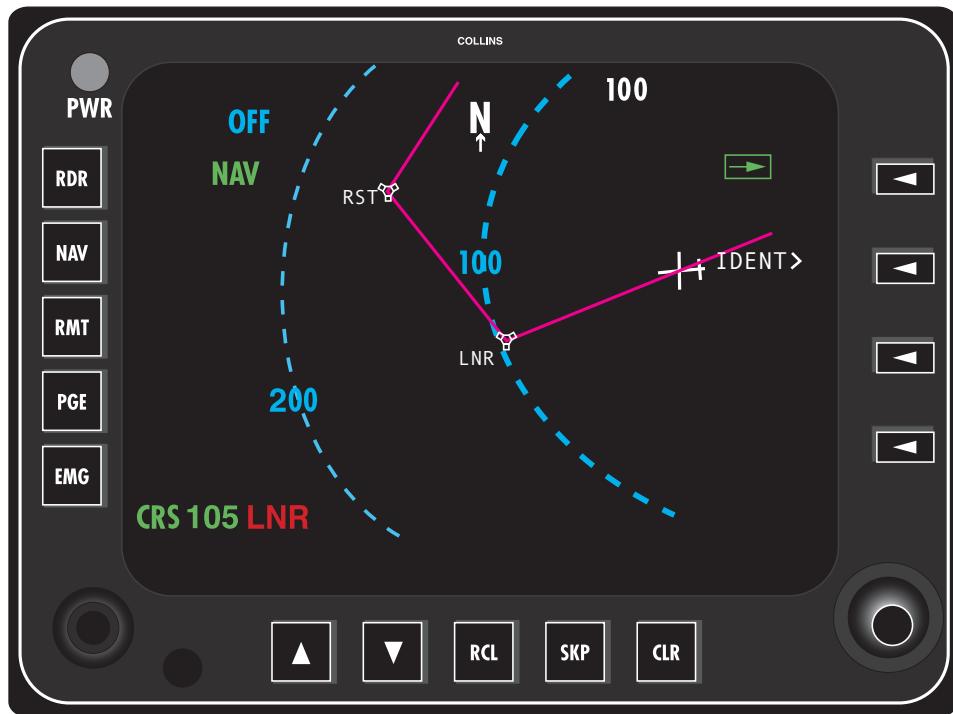


Figure 16-42 NAV Mode North Up Maximum View Format

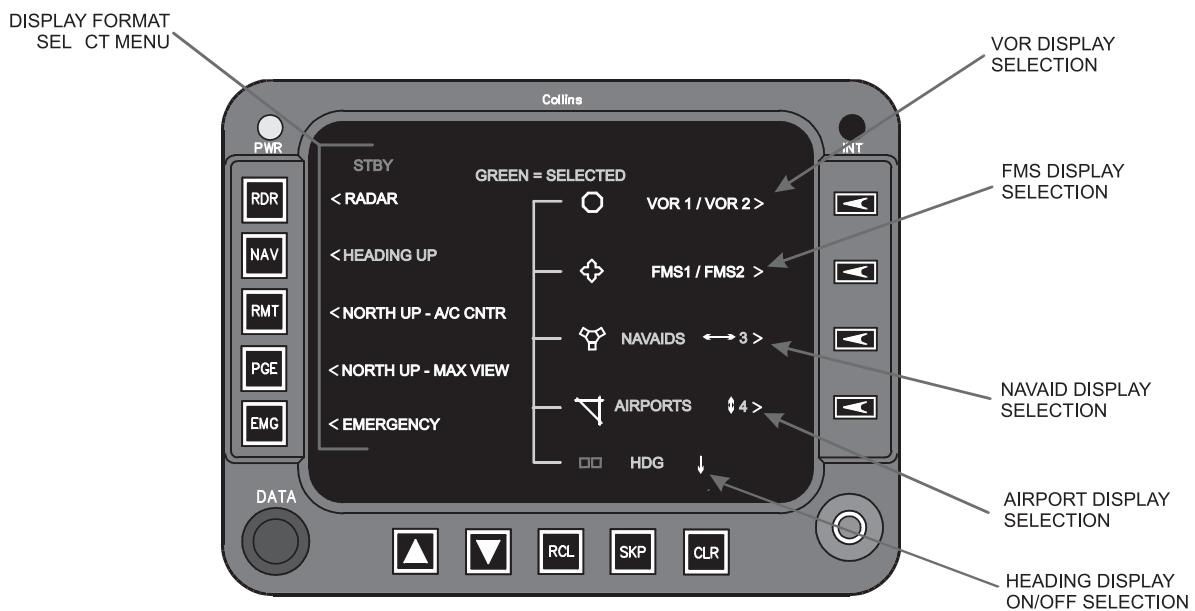


Figure 16-43 NAV Mode NAV Data Display



To select the desired sensor(s), push the line key next to the desired sensor(s) to select it to show its data on the display in the NAV mode. The sensor(s) selected are shown in green in the NAV menu and the sensors not selected are shown in white. For the VOR1/VOR2, and FMS1/ FMS2 keys, successive pushes alternate between the two selections.

For example the VOR selections:

- Both off (VOR1 and VOR2 both white)
- VOR1 selected to show data on the display (VOR1 green and VOR2 white)
- VOR2 selected to show data on the display (VOR2 green and VOR1 white)
- Both VOR1 and VOR2 selected to show data on the display (VOR1 and VOR2 both green)
- Both off

The FMS selections are the same but only one FMS is installed, so only FMS1 will display any information on the navigation displays.

To change the number of NAVAIDS displayed, push the joystick left or right. To change the number of airports displayed, push the joystick up or down. Up to seven NAVAIDS and seven airports can be shown on the display.

To select the heading display, push the CLR button (HDG on the display) to turn the heading bug on and off. A line connects the heading bug to the airplane symbol. The line rotates with the heading bug. In addition, a numerical readout of the selected heading shows below a heading bug symbol in the upper right corner of the display. Heading select operates exactly as on the EHSI with the CHP HDG knob controlling this function.

Page Data (Normal and Emergency Checklists)

The MFD can store up to 100 pages of user information in nonvolatile memory. This data may be arranged in chapters and can be designated as

either normal PGE (page) or EMG (emergency) data. All 100 pages can be input and modified by the user using the hand-held RDP-300 remote data programmer or the CEU-85 checklist entry unit.

Remote Data (RMT) Menu

Push the RMT button to show four labels, RMT1, RMT2, FLT INFO SUMMARY, and EXTENDED DATA, next to the line select keys. Push the appropriate line key to show the remote input on the MFD. In the B300 aircraft, no remote device is connected, so a message NO DATA FROM REMOTE DEVICE #X will show on the display after approximately five seconds.

FLT INFO SUMMARY Display

From the remote data menu, the flight information summary page may be selected to display certain information that may be useful to the flight crew. Information from the FMS is as follows: GS, ETE, DIST, ETD, ETA, XTK, DA and WIND. From the air data computer the information is TAS, RAT, IAS, and TEMP.

Extended Data Display

From the remote data menu, the extended data menu may be selected. It contains several pages of data useful to the flight crew or maintenance personnel. These are: VOR/DME, CRUISE, APPROACH, CATII, and DIAGNOSTICS. Push the RMT button followed by the select button next to EXTENDED DATA to show a menu with the first four page titles. Push the joystick up or down to select additional titles. Push the select key next to the desired page title to show the first page of data associated with that title. If there is more data associated with that title, an arrow shows in the lower right corner to show which way to push the joystick to show the additional data. Fixed display legends show in white and variable data shows in green.



EFIS REVERSIONARY SYSTEMS

The EFIS system provides system backup operational capability in the event of the failure of a system element.

Reversion Capabilities

The drive transfer system provides backup operation in the event of a display processor unit (DPU) failure and will restore the missing EADI and EHSI displays. The EFIS system carries out this function by the use of the multifunction processor unit (MPU) (Figure 16-44).

The DPU has a relay card that can switch the EADI and EHSI video from the DPU (internal within the DPU) to the MPU cross-side video signals when the relays are energized.

Power for the relay coils are supplied from the left MFD power circuit breaker and the ground for the relay coil is made with the drive transfer switch.

The DPU and MPU also incorporate a cross-side data bus that allows for data from the MPU to be sent to DPU for cross-side sensor display. The display control panel sends its control data to both the DPU and MPU so that when the system is in the reversionary modes of operation full control of the displays is available. Both the DPU and MPU send control reply data to the DCP.

For B300 (350) aircraft, the EFIS reversionary switches are backlit pushbutton selector switches.

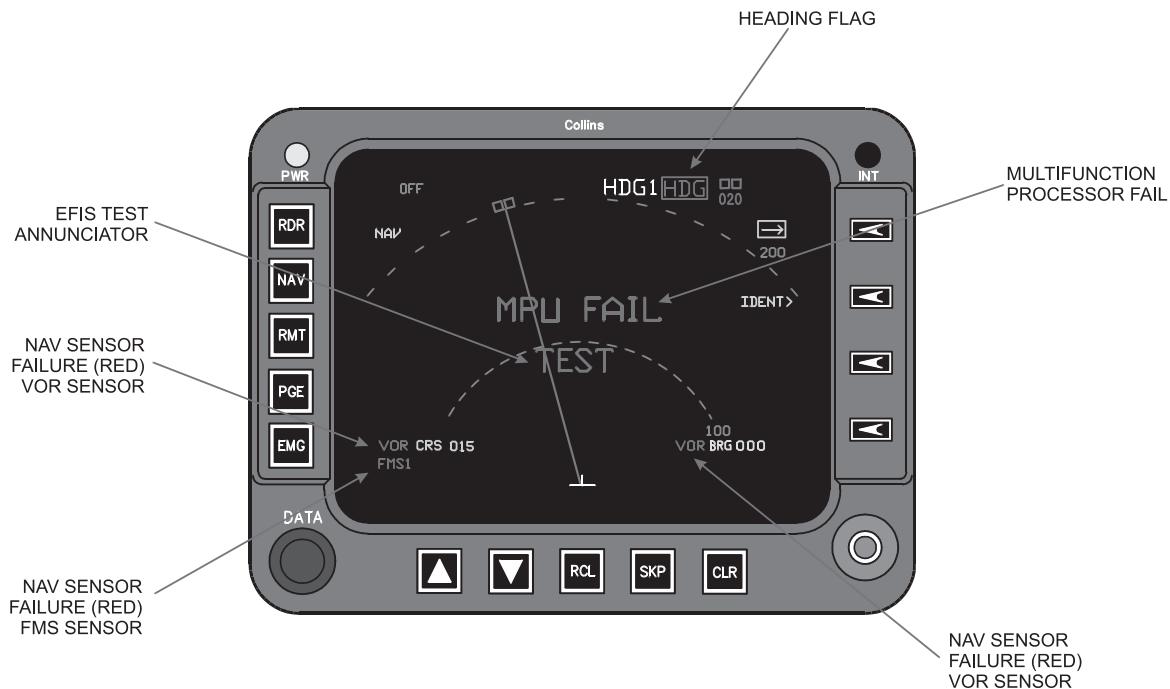


Figure 16-44 MFD Warning Flags



Composite Format Display

In the event of an EFD failure by the use of a composite display switch, the operating display can be used to display a composite format display (Figure 16-45). The composite display is a mixing of the EADI display with the EHSI display. The compass display is in the form of a small compass sector with the heading cursor and active course displayed in the lower half of the EADI display.

Normally, each EFD is configured as an EADI or EHSI display depending on its location in the instrument panel. With the use of the CMPST switch, a ground discrete is applied into the DPU EADI and EHSI composite display active input. When the DPU receives this ground, the DPU provides composite format video to both displays. In the event of a screen failure, the remaining screen would display the composite format display.

MFD Alternate Display Operation

In the event of an EHSI failure, the MFD display can be used to provide the EHSI display

(Figure 16-46). This mode is selected by using the MFD alternate display switch providing a ground into the MPU which then forces the MPU into the DPU mode and sends EHSI video to the MFD for display.

Drive Transfer System Operation

When the drive transfer switch is in the normal position, the DPU drives its EADI and EHSI screens. In the event of a DPU failure, the switch is selected to the AUX DRIVE position placing ground into the DPU, which activates the drive and transfer relays. The ground is also provided to the MPU, forcing it into the DPU mode. When the relay activates, cross-side video from the MPU drives the EADI and EHSI. The AUX DRIVE annunciator, which is part of the drive transfer switch, will illuminate to indicate drive transfer operation. If the MFD display is active, it will display an EHSI as it did for MFD alternate display.

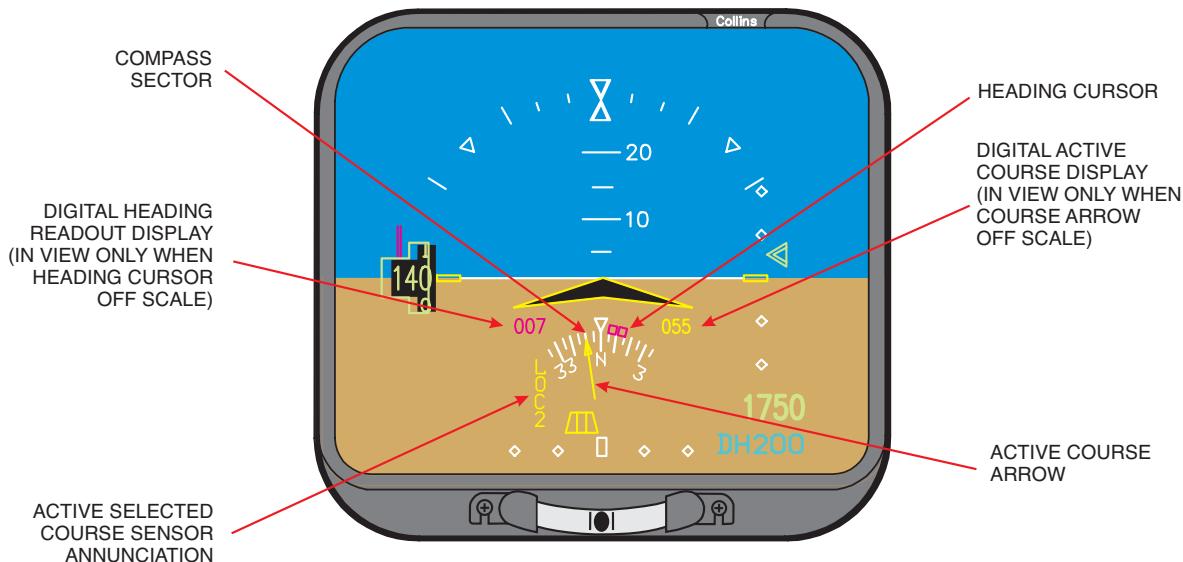


Figure 16-45 EFIS-85B (14) Composite Display



Figure 16-46 MFD Alternate Display (EHSI Display)

Drive Transfer and Composite Display Operation

With both drive transfer and composite display switches active, all displays are viewed as composite displays (Figure 16-47). The ground from the composite display switch is transferred by the drive transfer switch to the MFD EADI and EHSI composite display active input.



Figure 16-47 MFD Composite Display

EHSI-74A SYSTEM

General

A Collins EHSI -74A electronic horizontal situation indicator system is installed on the copilot's instrument panel. The EHSI-74A electronic horizontal situation indicator system consists of a panel-mounted EFD-74 electronic flight display (Figure 16-48), a pedestal-mounted HCP-74 HSI control panel (Figure 16-49), and a remote-mounted HPU-74A HSI processor unit. The EHSI utilizes a multicolor CRT display unit that presents a plan view of the airplane's horizontal navigation situation. Information displayed includes: indicated heading, selected heading, selected VOR, localizer, FMS course and deviation, glideslope deviation navigation source annunciation, to/from information, DME distance, groundspeed, time-to-go, and a bearing pointer that can be driven by VOR, ADF, and FMS sources as selected on the HSI control panel (Figures 16-50 and 16-51). The EHSI can also be operated in an ARC mode that provides a compass sector with alphanumeric data shown in the corners of the display (Figure 16-52), and in a MAP mode that shows bearing and distance to the selected NAVAID or waypoint (Figure 16-53).

The navigation sensor sources available to the EHSI-74A system are:

- NAV 2 /DME 2
- FMS

A remote switch panel assembly below the EHSI-74A display is used to select between these two navigation sources.

The bearing pointer displays can be driven by the following sensors:

- VOR1 / VOR2
- ADF1 / ADF2 (if installed)
- FMS

Bearing pointer display and sensor source are selected on the HSI control panel (HCP).

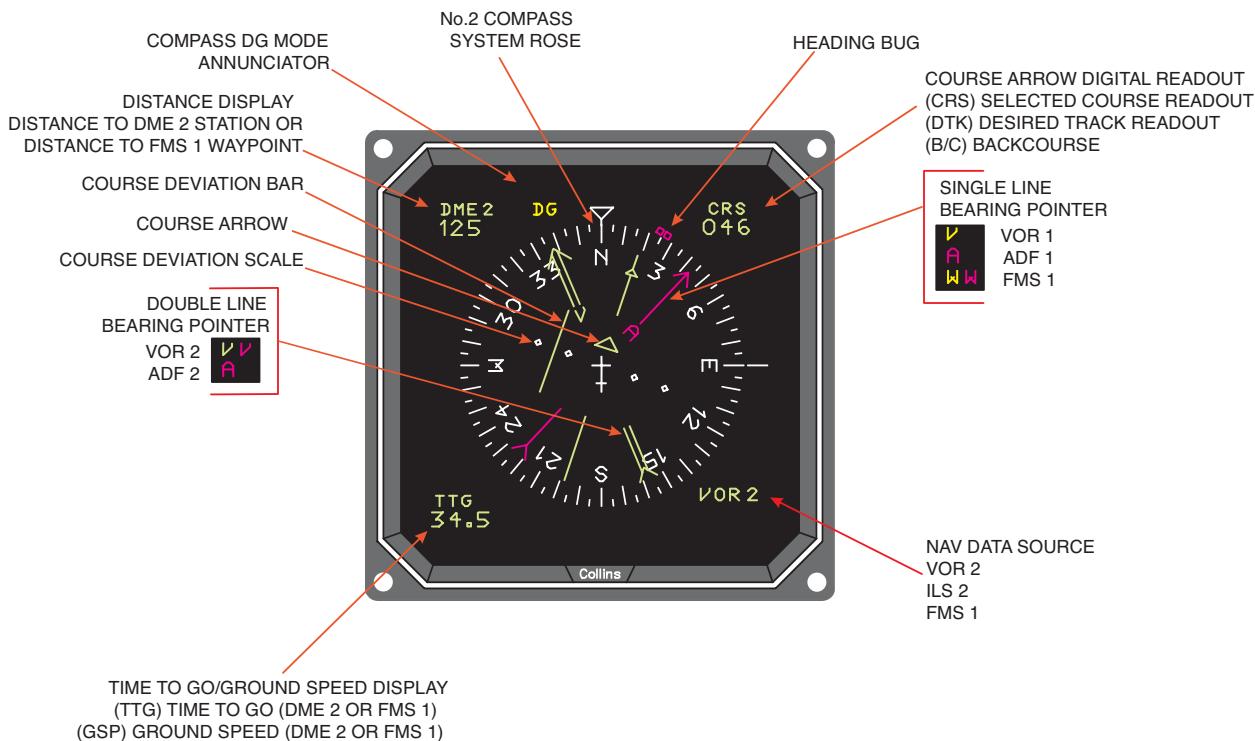
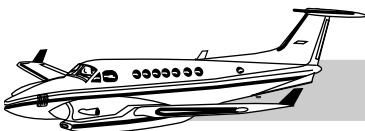


Figure 16-48 EHSI-74A In HSI Mode With NAV (VOR Tuned) Sensor Selected

The compass display is driven by the No. 2 compass system.

The following paragraphs provide brief descriptions of the units used in the EHSI-74A system.

Electronic Flight Display

The EFD-74 electronic flight display is a panel-mounted high-resolution multicolor CRT display unit that contains the CRT, deflection yoke, video and deflection amplifiers, and a high-voltage power supply. Video and deflection power supplies, along with the high-voltage power supply, provide the necessary voltages to operate the EFD-74. The primary power input to the EFD-74 is +28 VDC from the aircraft power bus.

HSI Control Panel

The HSI control panel provides switches and rotary controls for EHSI display mode control,

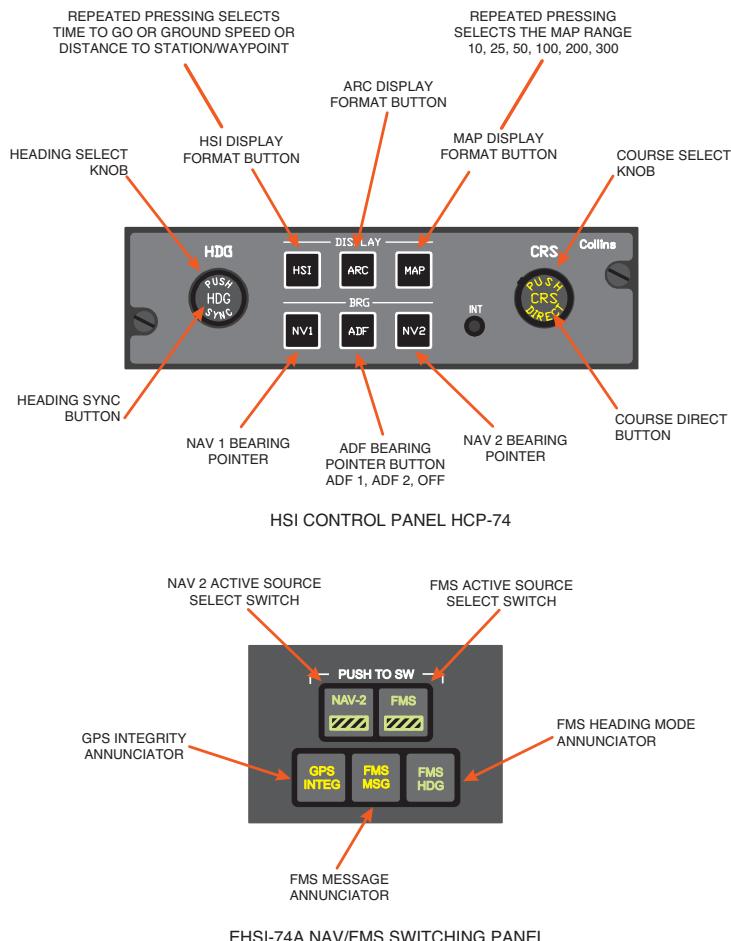
EHSI display dimming, bearing pointer selection, selected heading control, heading synchronization, selected course control, and direct to course.

HSI Processor Unit

The HPU generates the video and deflection signals required to draw the desired display on the EFD. It also contains the aircraft I/O interface, I/O processor, display generator, and a power supply. There are two indicators, HPU FLAG and EFD FLAG, on the front of the HPU.

HPU monitoring consists of various monitors in the display generator, input/output processor, memories, and power supply. When a failure is detected, the HPU FLAG indicator lights, and the FAIL legend appears on the CRT display.

The EFD FLAG indicator monitors operation of the deflection and video signals in the EFD. When a fault is detected (such as the display


Figure 16-49 EHSI-74A Controls

going blank), a signal is sent to the HPU to light the EFD FLAG (Figure 16-54).

The power supply provides all necessary voltages for the HPU-74A to operate. The primary power interface to the HPU-74A is +28 VDC from the aircraft power bus. The power supply also provides line drivers and monitoring of the video and deflection signals. Monitoring of the primary input power is also provided to ensure orderly start-up and shutdown of the HPU-74A in the event of power transients on the aircraft power bus.

EHSI-74 Operating Controls

The following section describes the functions of the controls used in the EHSI-74 electronic horizontal situation indicator system.

HCP-74 HSI Control Panel

The HCP-74 HSI control panel (Figure 16-49) provides the user with the controls needed to select the various operating modes and functions for the EHSI. The HCP-74 also provides the EHSI dimming control.

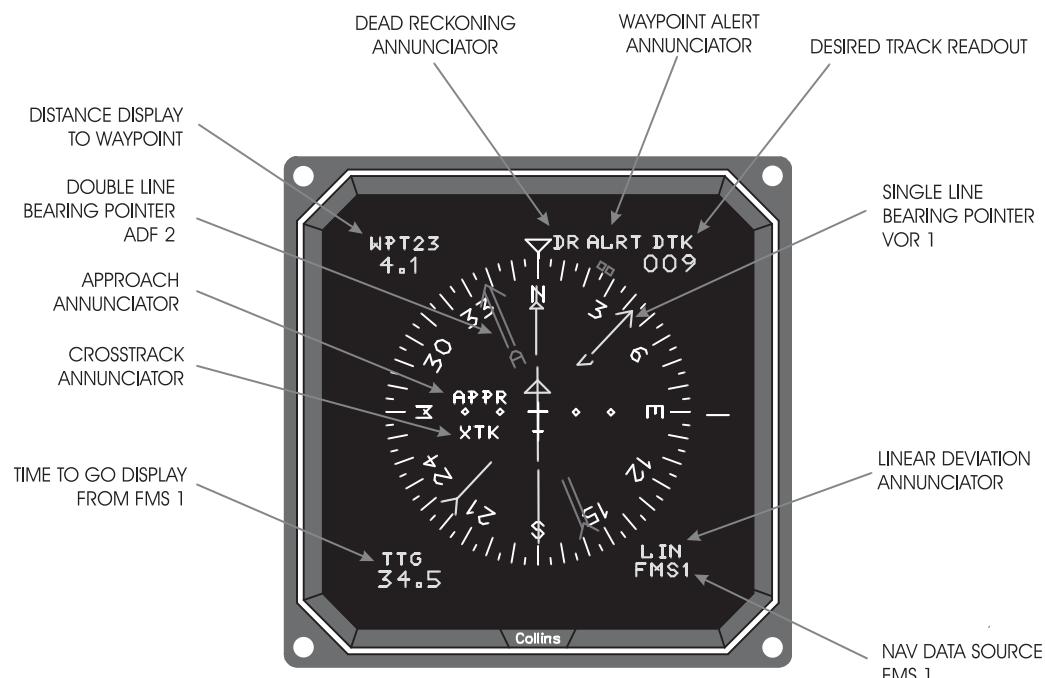


Figure 16-50 EHSI-74A In HSI Mode With FMS Sensor Selected

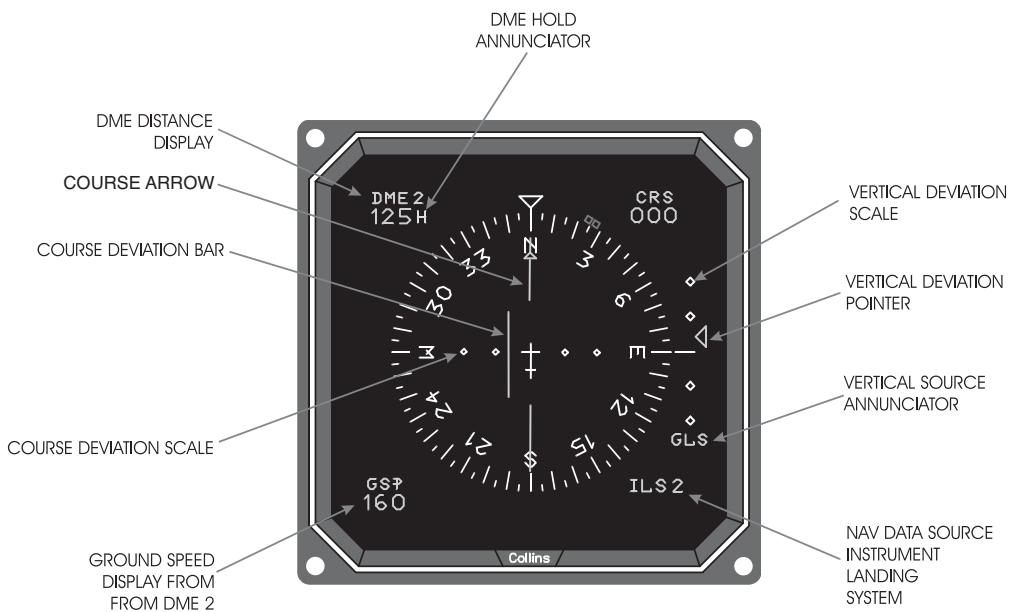


Figure 16-51 EHSI-74A In HSI Mode With NAV (LOC Tuned) Sensor Selected



Heading Select (HDG) Knob

The heading select knob controls the selected heading displayed on the EHSI. The knob provides 1° increments of selected heading when turned slowly. A variable rate sensor provides larger increments when the HDG knob is turned at a faster rate.

Heading Synchronization (PUSH HDG SYNC) Button

The heading knob includes a center SYNC push button which, when pressed, causes the heading bug to rotate and match the airplane heading under the lubber line on the EHSI.

Course Select (CRS) Knob

The course select knob controls the course arrow displayed on the EHSI. The knob provides 1° increments of selected course when turned slowly. A variable rate sensor provides larger increments when the CRS knob is turned at a faster rate. The CRS knob is disabled when an FMS is the navigation source.

Course Direct to (PUSH CRS DIRECT) Button

The course knob includes a center PUSH CRS DIRECT button that provides a “direct to” function. When the DIR button is pressed, the course arrow automatically rotates until the course is equal to the VOR bearing, this will provide a zero lateral deviation display. The PUSH CRS DIRECT button is disabled when the FMS or ILS is the navigation source.

DISPLAY Format Selection

The HSI, ARC, and MAP buttons provide control of the display formats. In the HSI position, the EHSI full compass rose format is displayed. The ARC position selects an expanded compass sector (approximately 80°) across the top of the display with alphanumerics displayed in the corners of the display (Figure 16-52). This mode gives the pilot information, similar to that presented in the full compass rose mode, in an enlarged easily readable format. The MAP mode adds a pictorial presentation of the navigation situation to the compass sector (Figure 16-53). Navigation data displayed

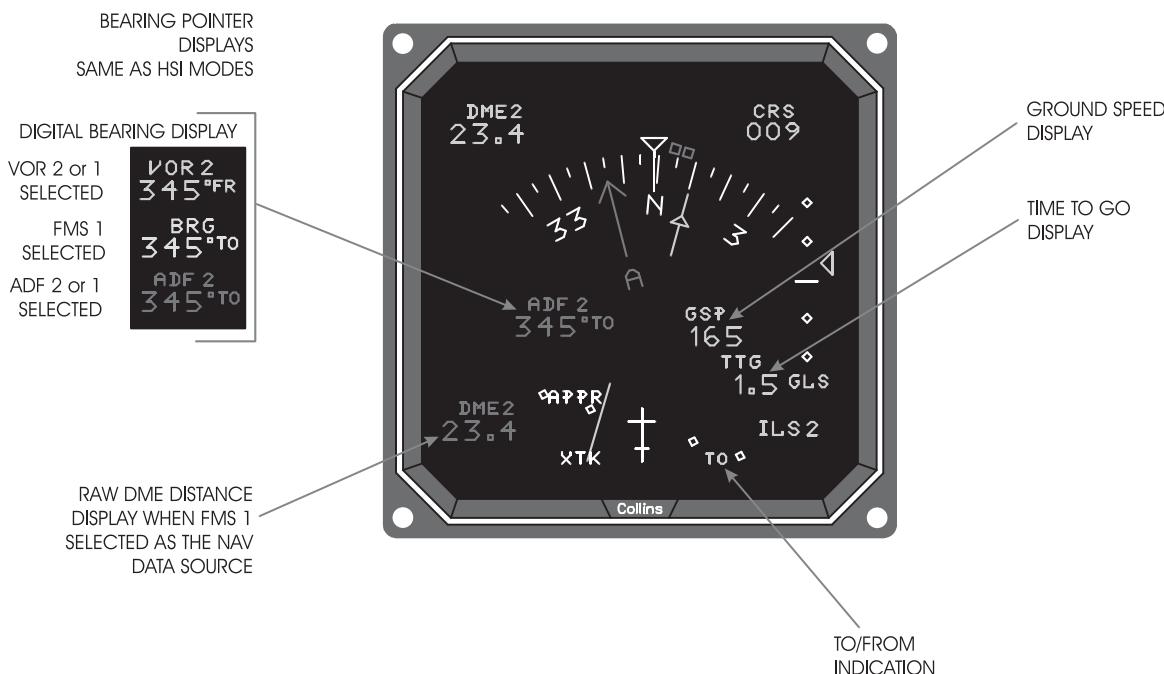


Figure 16-52 EHSI-74A in ARC Mode



includes VOR/DME and active waypoint location, and course line.

Repeated pressing of the HSI button will select the time-to-go, groundspeed, or distance to station/waypoint of the NAV data display. Repeated pressing of the MAP button in the MAP mode will select the map display range.

Bearing (BRG) Selection

The NV1, ADF, and NV2 buttons control which bearing pointer is displayed on the EHSI. For ADF, either pointer may be displayed or both pointers may be displayed at the same time by repetitive pushes of the ADF button. The third push of the ADF button removes the ADF bearing pointers. For NV1 and NV2, the bearing pointer is selected by pressing the desired push button. The bearing pointer is removed from the display by pressing its push button again or by selecting another bearing pointer. A bearing pointer will not appear when the navigation receiver is tuned to an ILS frequency. A special

case exists when an FMS is the navigation source. In this case, pressing the NV1 button the first time displays the waypoint bearing and pressing the NV1 button again removes the waypoint-bearing pointer and displays the VOR bearing pointer.

Intensity (INT) Control

The intensity of the EHSI display is controlled by rotation of the INT knob. Clockwise rotation increases the brightness of the display.

Remote Switch Panel

The remote switch panel contains two annunciator switches labeled NAV2 and FMS. These switches are used to select between NAV2 and FMS sensors. The selected sensor annunciator will be illuminated green when active. The FMS external annunciators used with the FMS display on the EHSI-74A display, are also located on this panel (Figure 16-49).

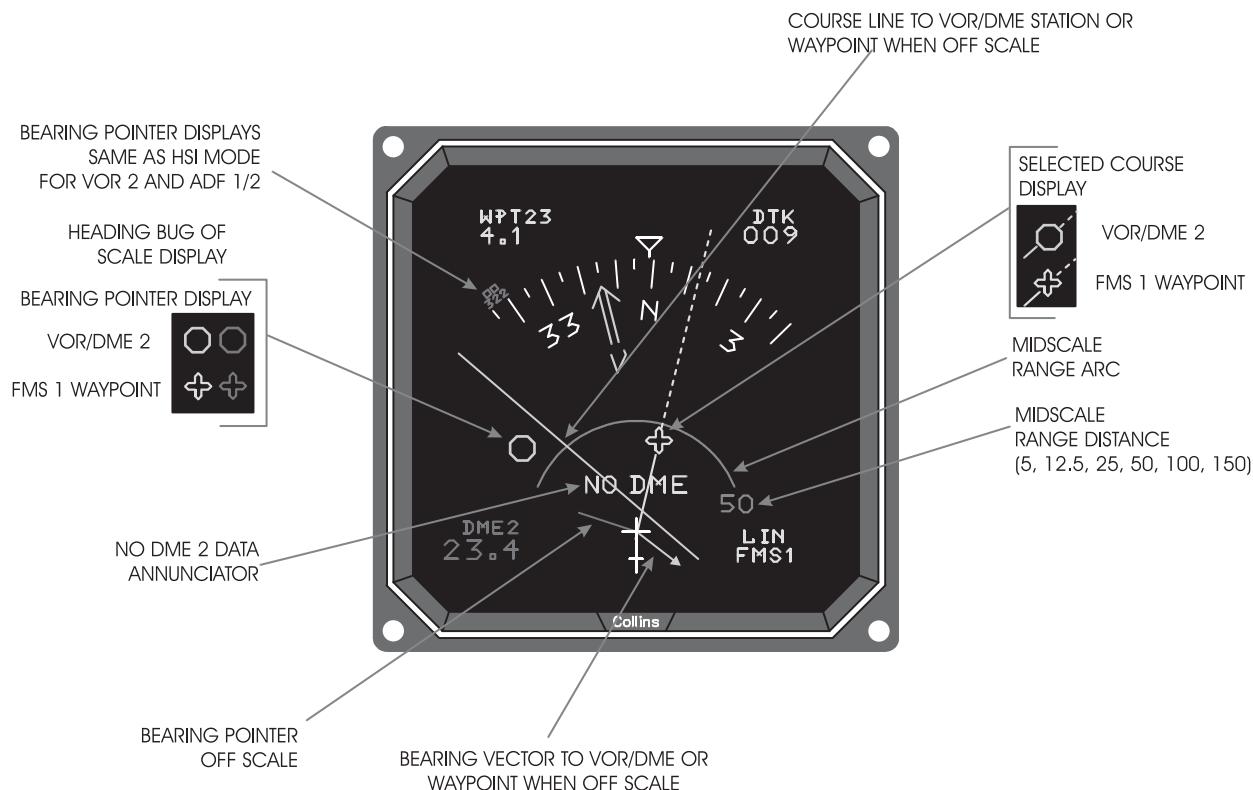


Figure 16-53 EHSI-74A In MAP Mode

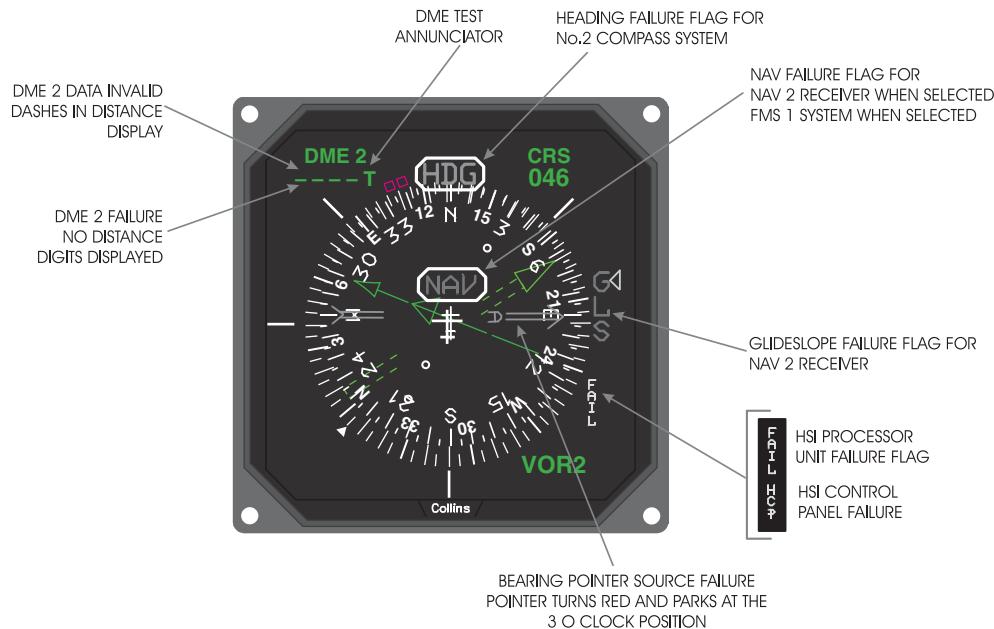


Figure 16-54 EHSI-74A Failure and Warning Flags

FCP-65 FLIGHT CONTROL PANEL

General

The FCP-65 flight control panel is used to select the operating modes of the flight director system (Figure 16-55). The mode select buttons are momentary-action, push-on/push-off buttons. Mode selection is annunciated on the flight control panel and the EADI display. Mode selection is interlocked to ensure that only compatible modes can be selected at the same time. When no lateral modes are selected, the steering display on the EADI is biased out of view except during and after selection of the go-around mode.

The FCP-65 mode (Table 16-10) lists the annunciators provided on the flight control panel. The FCP flashing mode annunciator

(Table 16-11) lists those mode annunciators that flash to indicate loss of a monitor signal while the mode is selected.

The TEST button is a momentary-action push-button. Airborne operation of the TEST button is limited so that it does not interfere with operation of the system. Pushing the TEST button while airborne causes a momentary lamp test followed by a display that indicates if any of the continuous monitor tests have failed. Operation of the TEST button while on the ground selects the system diagnostic mode consisting of a lamp test followed by a display that indicates if any of the continuous monitor tests have failed. Other test routines can be selected and performed on the ground as a maintenance aid.

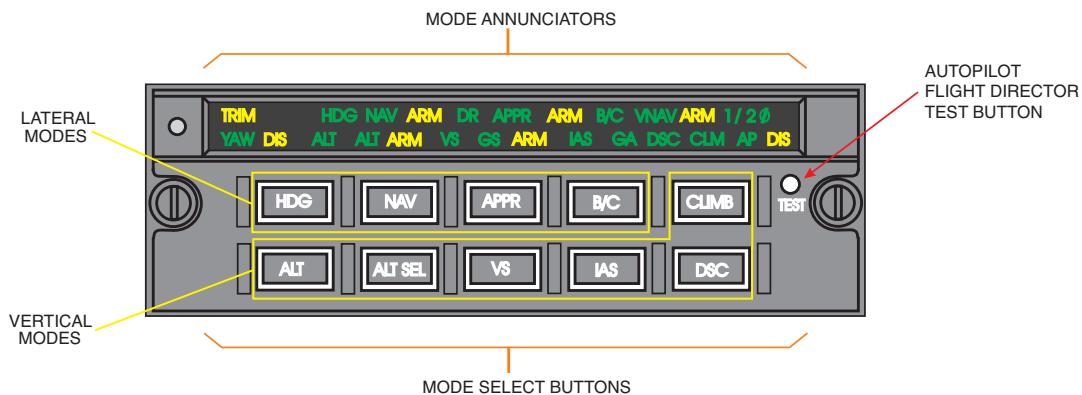
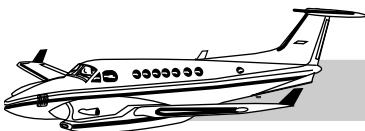
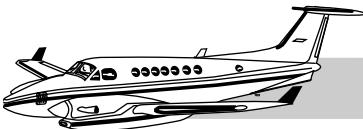


Figure 16-55 Flight Control Panel FCP-65 Controls and Indications

Table 16-10 FCP-65 MODE ANNUNCIATORS

ANNUNCIATOR	INDICATION	COLOR
AP	Autopilot engaged	Green
DIS (AP)	Autopilot disengaged	Amber
YAW	Yaw damper engaged	Green
DIS (YAW)	Yaw damper disengaged	Amber
HDG	Selection of heading mode	Green
NAV	Selection of navigation mode	Green
ARM (NAV)	System is armed to capture VOR, LOC, or FMS signal	Amber
DR	System is operating in dead reckoning mode	Green
APPR	Selection of approach mode	Green
ARM (APPR)	System is armed to capture VOR, LOC or FMS signal	Amber
VNAV	System is responding to a pseudo glideslope signal from the FMS (Automatic selection with approach and FMS as sensor input)	Green
ARM (VNAV)	System is armed to capture FMS pseudo glideslope signal	Amber
B/C	Selection of back-course localizer mode	Green
CLM	Selection of climb mode	Green
DSC	Selection of descent mode	Green
ALT	Selection of altitude hold mode	Green
ALT ARM	System is armed to capture a preselected altitude	Amber
VS	Selection of vertical speed hold mode	Green

**Table 16-10 FCP-65 MODE ANNUNCIATORS**

ANNUNCIATOR	INDICATION	COLOR
GS	GS System is responding to glideslope signal (Automatic selection with approach and LOC as sensor input)	Green
ARM (GS)	System is armed to capture glideslope signal	Amber
GA	Selection of go-around mode	Green
TRIM	Elevator trim system operating (trim in motion)	Amber
1/2 φ	Selection of half-bank mode	Green

Table 16-11 FCP-65 FLASHING MODE ANNUNCIATORS

FLASHING ANNUNCIATOR	INDICATION	COLOR
HDG	Loss of attitude or compass flag	Green
NAV	Loss of attitude, compass, NAV or FMS flag	Green
APPR	Loss of attitude, compass, NAV or FMS flag	Green
VNAV	Loss of attitude or FMS flag	Green
DIS (YAW)	Flashes for five seconds after yaw damper disengages	Amber
ALT	Loss of attitude flag or ADS validity	Green
ALT ARM	Loss of attitude or ADC-85 flag, or ADS validity	Green
VS	Loss of attitude flag or ADS validity	Green
GS	GS Loss of attitude or glideslope flag	Green
IAS	Loss of attitude flag or ADS validity	Green
DSC	Loss of attitude flag or ADS validity	Green
CLM	Loss of attitude flag or ADS validity	Green
GA	Loss of attitude flag	Green
DIS	Flashes for five seconds after autopilot disengages	Amber



APP-65A AUTOPILOT PANEL

The APP-65A Autopilot Panel provides the autopilot and yaw engage/disengage buttons, soft-ride and half-bank mode select buttons, turn knob, vertical control, and autopilot annunciators (Figure 16-56 and 16-57). Refer to the APP-65A annunciator table for a list of the autopilot annunciators (Table 16-12).

NOTE

All pushbuttons in the system require a positive push of at least 1/4 second to be recognized by the system. This delay prevents inadvertent operation of the system due to electrical interference.

The yaw and autopilot engage buttons are momentary-action, push-on/push-off buttons. Pushing the yaw engage (YAW ENG) button once activates the yaw channel of the autopilot and engages the rudder servo. Pushing the YAW ENG button a second time disengages the rudder servo. Pushing the autopilot engage (AP ENG) button once engages all autopilot servos including the rudder and trim servos. However, pushing the AP ENG button a second time disengages all servos except the rudder servo, which must be disengaged by pushing the YAW ENG button or the system disengage button located on the control wheel.

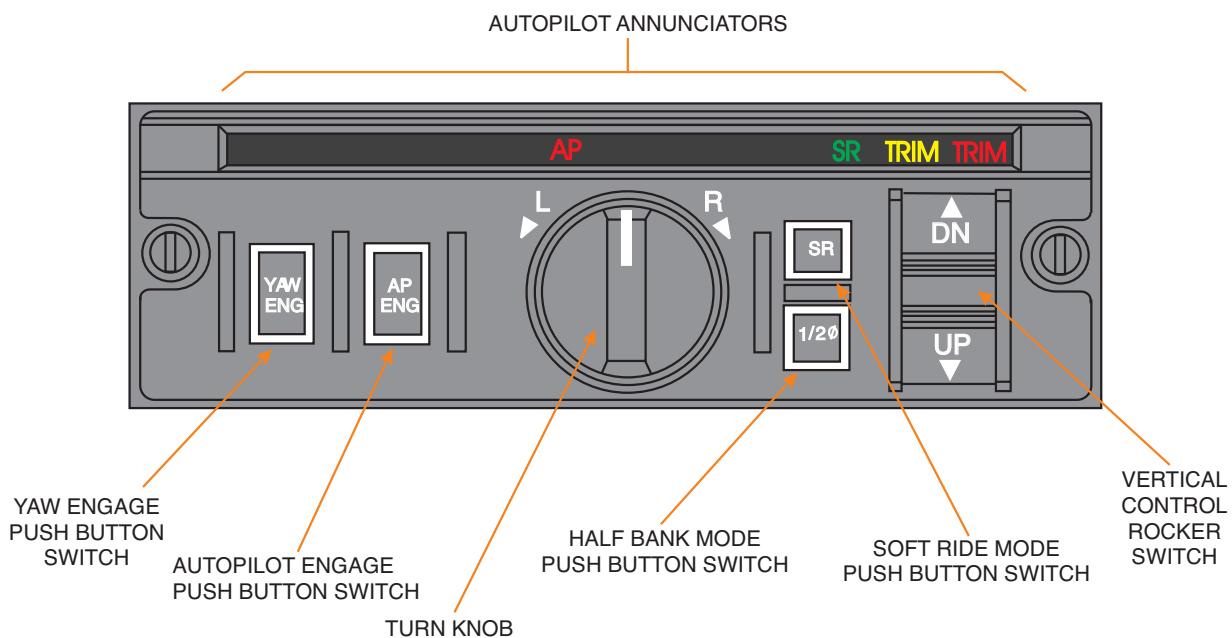


Figure 16-56 B300 (35C) Autopilot Controller APP-65A Controls and Indications

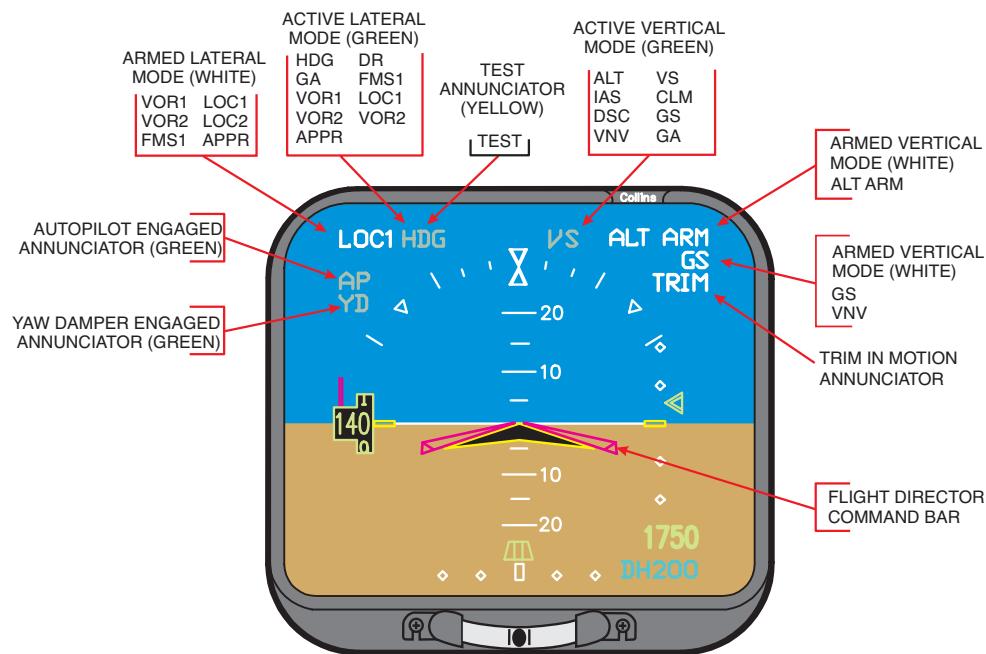


Figure 16-57 EADI APS-65 Autopilot/Flight Director Displays and Indications

The autopilot also selects the altitude preselect mode (ALT ARM) whenever the autopilot is engaged.

Each time the autopilot or yaw system is engaged, the computer goes through a preengage test routine during which the AP and TRIM failure annunciations are lit and the disengage circuits and internal monitors are checked. If the preengage test sequence is not completed satisfactorily, the system does not engage and an AP fail annunciation is displayed on the APP-65A and the annunciation warning panel. The computer continuously performs a self test routine while power is applied. Loss of a critical monitor or validity sig-

nal from either the air data sensor or vertical gyro, while the autopilot is engaged, causes the autopilot to disengage.

The soft-ride (SR) and half-bank (1/2) mode select buttons are also momentary-action, push-on/push-off buttons. Pushing the soft-ride button activates the soft-ride mode, which changes the gain of various circuits in the computer to provide increased passenger comfort during turbulent conditions. Pushing the half-bank button activates the half-bank mode, which limits the maximum commanded roll angle to 12.5°.

Table 16-12 APP-65A ANNUNCIATORS

ANNUNCIATOR	INDICATION	COLOR
SR	Selection of soft-ride (turbulence) mode	Green
1/2 φ	Selection of half-bank mode	Green
AP	Autopilot failure	Red
TRIM	Trim system failure	Red



The turn knob is a right/left bank angle control that has a center detent position at the 0° position. The control is not spring-loaded and remains at any position between the end stops when released. Operation of the control drops any previously selected lateral modes except the APPR mode. The bank angle commanded by the control is proportional to the displacement of the control from the detent position. The turn knob becomes inactive if it is out of the detent position when the SYNC button is pushed. The autopilot will engage with the turn knob out of the detent, but the turn knob is inactive until after it has been returned to the detent position.

The vertical control is a center-off spring-loaded rocker switch that provides manual control of the elevator channel when the autopilot is engaged and no vertical modes are selected, or provides incremental changes to the air data reference if a vertical mode is selected. If the vertical control is held for one second or more, the selected vertical mode is dropped and the system returns to the pitch hold mode.

AUTOPILOT SYSTEM EXTERNAL SWITCHES

Control Wheel Disengage Switches

A two-level disengage switch labeled DISC TRIM/AP YD is located on each control wheel and provides a convenient means to disengage both the autopilot and the yaw damper, when pressed to the first level. Pressing the switch to the second level will lock out the electric trim. Operation of the disengage switch causes all autopilot servos to disengage and all selected vertical modes to drop if no lateral modes are selected. The vertical modes can be reselected after the autopilot has disengaged and a lateral mode has been selected. Any selected lateral and vertical modes are retained and can be used in the flight director mode.

Pitch Sync and CWS Switches

The pitch synchronization and control wheel steering button located on each control wheel, is used to manually maneuver the aircraft without completely disengaging the autopilot when the autopilot is engaged, or to sync the command bars to the current aircraft attitude with only the flight director selected.

When the autopilot is engaged, depressing the PITCH SYNC & CWS button disengages the aileron and elevator servos. The aircraft can be maneuvered to any desired attitude while the PITCH SYNC & CWS button is depressed. When the PITCH SYNC & CWS button is released, the aileron and elevator servos reengage, the computer synchronizes to the new attitude, and control is returned to the previously selected modes. During the time the button is pressed the AP DIS annunciator will flash. Return to the selected modes is restrained to prevent any rapid maneuvers when the aircraft attitude varies from that commanded by the autopilot.

With the ALT, IAS, or VS flight director modes of operation selected, depressing the PITCH SYNC & CWS button and manually maneuvering the aircraft to a new reference altitude, airspeed, or vertical speed will cause the flight director command bars to sync to the new air data reference when the button is released.

When in pitch hold, depressing the PITCH SYNC & CWS button and manually maneuvering the aircraft to a new pitch attitude and releasing the button will cause the command bars to sync to the new pitch reference.

Go-Around Mode Switch (GA)

Go-around mode is activated by pushing the Go-around switch (GA) located on the left power lever. GA mode may be selected whenever a lateral mode is selected on the controlling FCP-65.

When GA mode is activated by the power lever switch, the autopilot will disengage, the autopilot disconnect aural warning sounds, and flight director command a wings-level, 7° nose-up attitude. Auto-pilot engagement, the CWS/PITCH SYNC button, or another mode selection cancels GA mode.



Trim Switches

The split trim switches on each control wheel are used to operate the electric trim system, which forms part of the autopilot installation. The inner trim switch is the primary method for autopilot disconnect. The YAW damp remains engaged.

Rudder Boost and Yaw Control Test Switch

The rudder boost and yaw control test switch is a three-position switch used to arm the rudder boost system and used to carry out the yaw control test. The switch is located on the pedestal. For more information on the rudder boost system refer to the rudder boost section in this chapter.

The yaw control test switch is used to check the functional integrity of the yaw damper system. The yaw control test is carried out by engaging the yaw damper and confirming the yaw damper is engaged by checking for the YD annunciator on the EADI. The YAW annunciator should illuminate on both the flight control panels. Also check that rudder pedals have added resistance with the rudder servo engaged. Move the switch to yaw control test position and check that the yaw damper has disengaged as indicated by all above annunciators turning off and the rudder pedals movement freeing up.

AUTOPILOT SYSTEM EXTERNAL ANNUNCIATORS

The autopilot installation includes the following annunciators:

- A/P FAIL — Autopilot computer or monitored input failure.
- A/P TRIM FAIL — Detected failure in the trim channel. This annunciator indicates that the electric trim is not responding to commands from the autopilot computer. The autopilot will not trip off line with the trim fail warning, but the pilot is alerted to the fact that elevator

could be out of trim and the elevator servo is holding a large static load.

- ELEC TRIM OFF/PITCH TRIM OFF — Electric elevator trim selected off.
- RUD BOOST OFF — Rudder boost system turned off (B300 only).

FLIGHT DIRECTOR AND AUTOPILOT OPERATION

The flight director and autopilot have a range of different modes that the system can operate in. The flight director modes are lateral modes and vertical modes, some of which have submodes of operation, as well.

The Lateral modes are:

- Heading
- Navigation
- Approach (Also a vertical mode when the glideslope or FMS pseudo glideslope signal used on a full ILS or GPS approach)
- Back Course

The Vertical modes are:

- Pitch Hold
- Altitude Hold
- Vertical Speed
- Indicated Airspeed
- Altitude Preselect
- Climb
- Descend

The altitude preselect mode is automatically selected anytime a lateral mode is selected. The flight director installation does not include these modes.



The autopilot only mode of operation is a roll hold mode. When the autopilot is engaged without any flight director modes selected, the autopilot system will automatically select the roll and pitch hold mode of operation. The pitch hold mode is the flight director pitch hold mode. The autopilot can be manually controlled in the roll and pitch hold by the use of the turn knob and the vertical control switch. The autopilot also has a half bank mode and a soft ride mode, which modify the autopilot responses.

Heading Hold Mode (HDG)

Selection of the HDG mode brings the steering display into view and commands are provided to fly to the heading selected by the heading cursor on the EHSI (Figure 16-58). The heading mode is annunciated by the illumination of the HDG annunciator on the FCP and the EADI. The altitude preselect mode is automatically selected with the heading mode. The heading mode is also automatically selected when another lateral mode is selected and the selected lateral mode is armed. The steering and autopilot commands are limited to provide a maximum commanded bank angle of 25°.

The EFIS system DPU measures the difference between the current aircraft heading and the selected heading to develop the heading error signal. This heading error signal is sent to autopilot computer so that the computer can track the heading cursor.

Navigation Mode (NAV)

Cross-country VOR, localizer radio signals, or FMS deviation signals are flown using the NAV mode (Figures 16-59, 16-60, 16-61, 16-62, 16-63, and 16-64). The altitude preselect mode is automatically selected with the navigation mode.

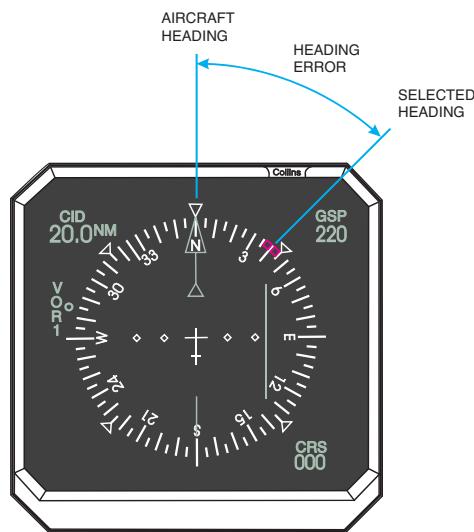


Figure 16-58 Flight Director Heading Mode

To use the NAV mode with VOR or localizer signals the following actions must be carried out:

- Tune the navigation receiver to the desired VOR or localizer station.
- Select VOR/LOC as the active course sensor on the EHSI.
- Set the course arrow to the desired radial.
- Set the heading cursor for the desired intercept angle.

The above actions set up the EHSI for correct NAV mode operation with VOR/LOC signals. The EFIS system DPU measures the difference between current aircraft heading and the selected course, and supplies a course datum signal to the autopilot computer. The DPU also provides the heading error signal to the autopilot computer. At this point, selecting the NAV mode puts the system into the



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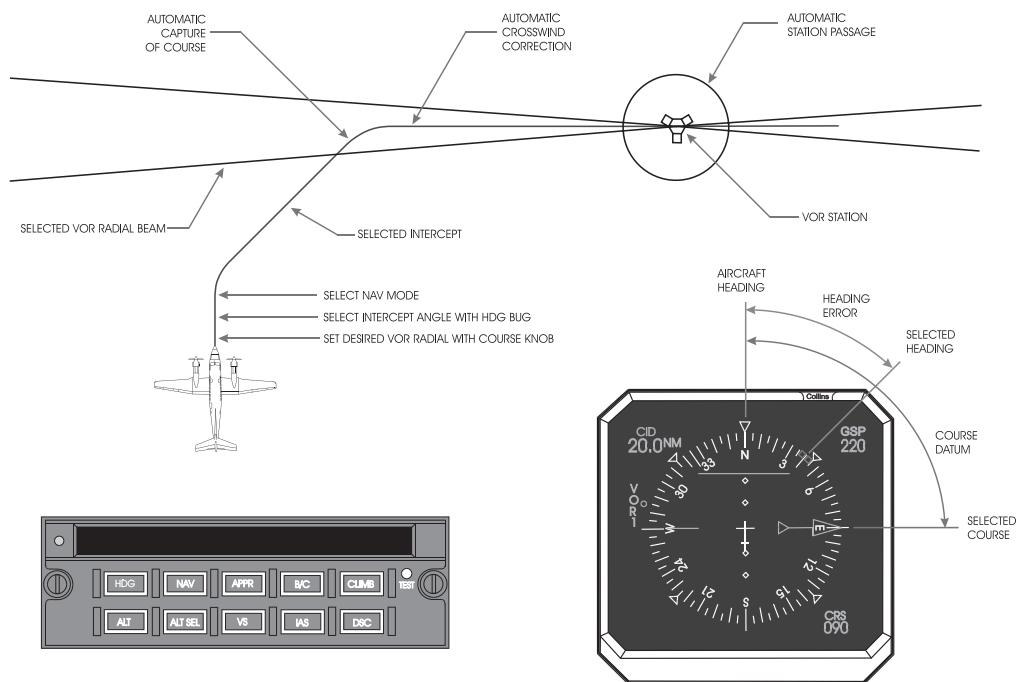


Figure 16-59 Flight Director NAV Mode VOR Before Heading Intercept

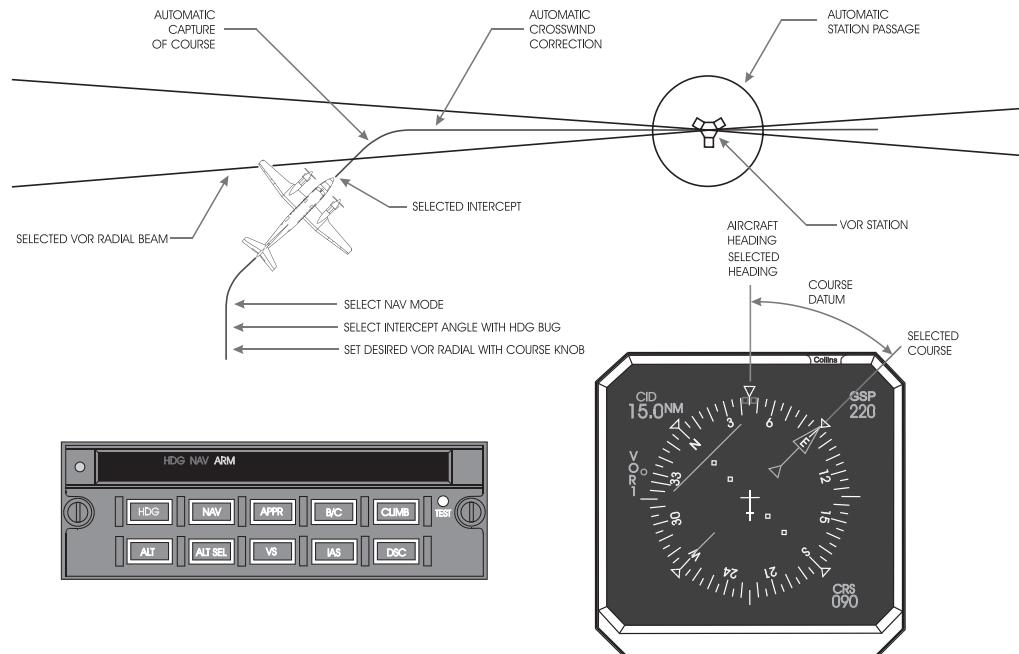


Figure 16-60 Flight Director NAV Mode VOR on Heading Intercept

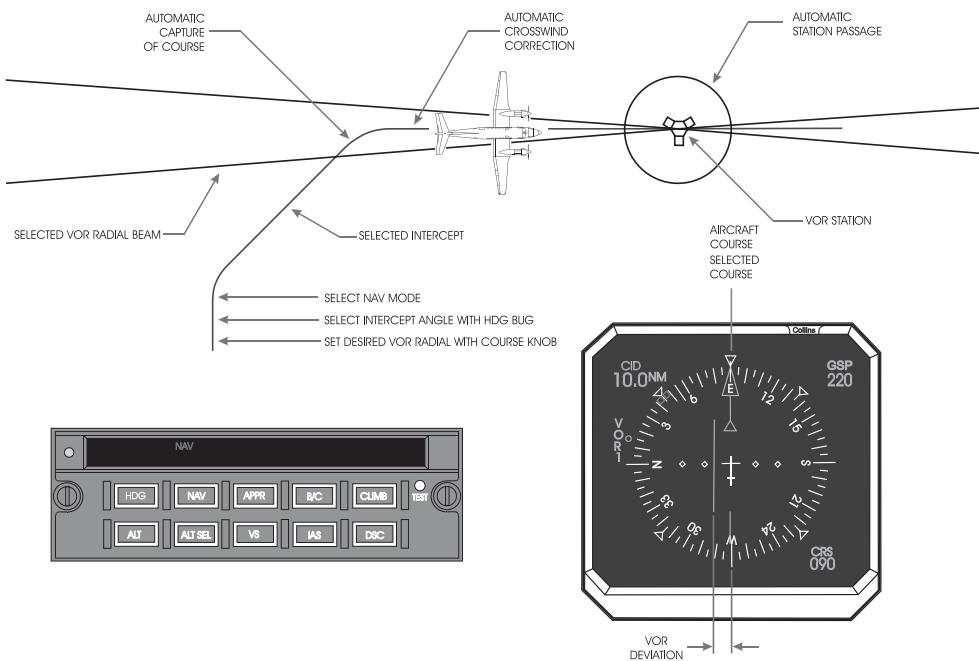


Figure 16-61 Flight Director NAV Mode VOR on Selected Radial

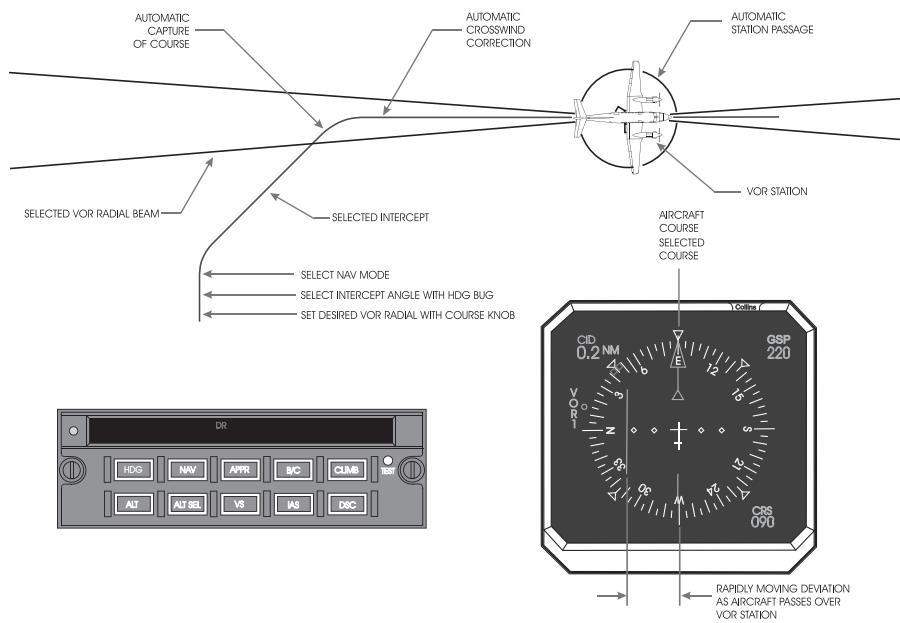


Figure 16-62 Flight Director NAV Mode VOR Passage Over Station

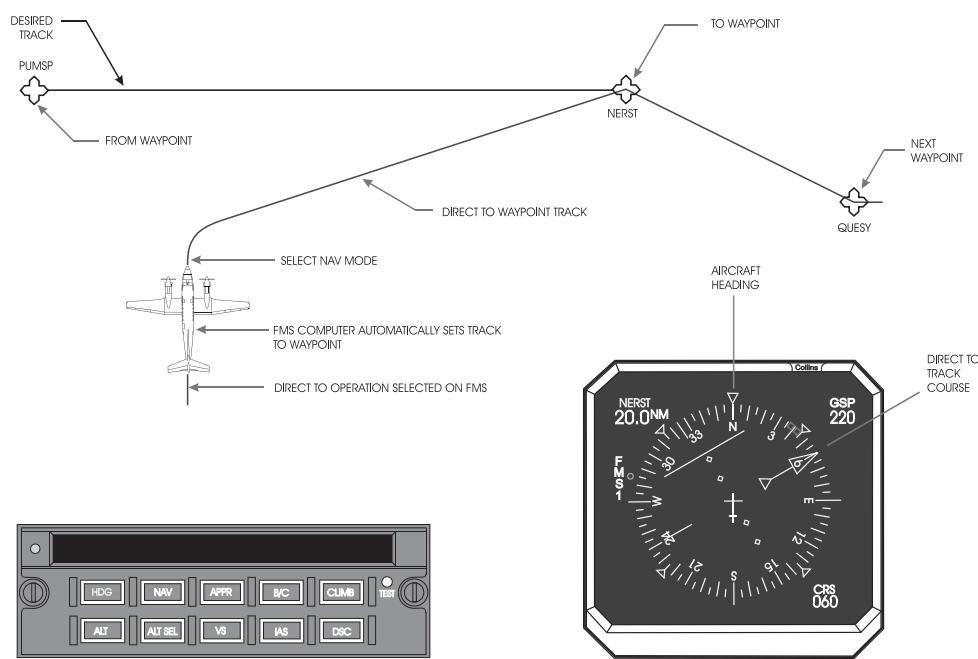


Figure 16-63 Flight Director NAV Mode FMS Before NAV Selected

NAV ARM submode. In the NAV ARM submode, the system annunciates HDG and NAV ARM and responds to inputs from the heading knob on the EHSI. The aircraft will fly to the selected heading, which will intercept the desired course or active track. The heading intercept angle can be up to 90°. When the flight path situation is such that the computer, using the deviation signals and course intercept angle, determines it is time to capture the radio course or GPS track, the system switches from NAV ARM to NAV. Steering commands are provided to capture and track the radio signal. After NAV capture using a VOR/LOC signal, the system responds to the lateral deviation signal as displayed on the EHSI to hold the aircraft on the selected course. The EFIS DPU supplies the deviation signals to the autopilot computer during VOR/LOC mode of operation. Automatic crosswind correction with crab angles up to 30° is provided. During VOR operation, command smoothing is provided to facilitate station passage and a dead reckoning (DR) annunciation is made to indicate when the system is operating in the station passage submode.

To use the NAV mode with the UNS FMS sensor the following actions must be carried out:

- Select FMS1 as the active course sensor.
- The course arrow will be automatically driven by the FMS to the next waypoint (FMS direct to operation).

The above actions set up the EHSI for correct NAV mode operation with VOR/LOC signals. When NAV is selected to autopilot, the system will automatically go to NAV capture and will track the roll steering signal from the FMS computer and turn the aircraft to fly direct to the active waypoint. The roll steering commands allow the UNS FMS system to directly steer the flight director command bars in roll to carry out turn anticipation, procedure turns and all procedural leg types, and other complex maneuvers that the FMS is capable of completing.

The FMS computer can also be set up to fly to a active track and intercept much like and VOR/LOC intercept.



Approach Mode (APPR)

The APPR mode is used anytime an approach to a runway is desired. There are four types of operations for carrying out the approach mode. These operations are described in the following paragraphs.

VOR/LOC Approach

The NAV receiver is tuned to a VOR frequency and VOR selected as the active sensor on the EHSI (Figures 16-65, 16-66, and 16-67). Then APPR mode is selected so the system configures itself in the NAV ARM submode for an all angle capture, similar to the NAV mode. The computer is set for a VOR approach situation and provides steering commands to acquire the VOR beam through the capture and track submodes. Also provided in this mode is a station passage submode that allows for course changes at the VOR station for a dogleg approach to published minimums. Glideslope operation is locked out and the system annunciations consist of HDG, APPR ARM, APPR, and DR (station passage) at the appropriate stages during the approach.

ILS Approach

The system is in the full ILS approach mode when the APPR mode is selected, the navigation receiver is tuned to a localizer frequency, LOC is selected as the active sensor on the EHSI, and the course arrow is set to the inbound course of the runway (Figures 16-65 and 16-66). The heading cursor is also set to provide an intercept angle to the inbound course.

The EFIS system measures the select course angle (course datum) and the heading angle (heading error) and provides these signals to the autopilot computer. Localizer operation in this mode consists of all angle capture and tracking of the localizer deviation similar to the NAV mode with appropriate changes to accommodate the localizer beam geometry.

When the system captures the localizer beam, the GS ARM annunciator illuminates to indicate that the system is ready to capture the glideslope beam. Glideslope capture can be accomplished from any vertical mode, from below the glideslope beam. Capture occurs prior to beam center to provide a smooth transition to the track mode, which tracks the glideslope deviation. Localizer

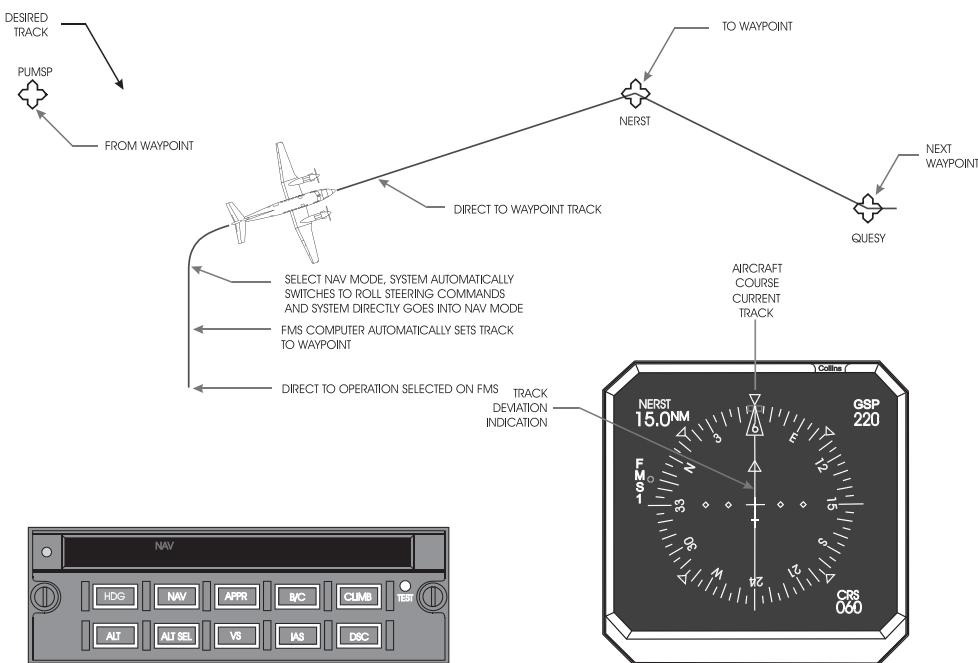


Figure 16-64 Flight Director NAV Mode FMS After NAV Selected

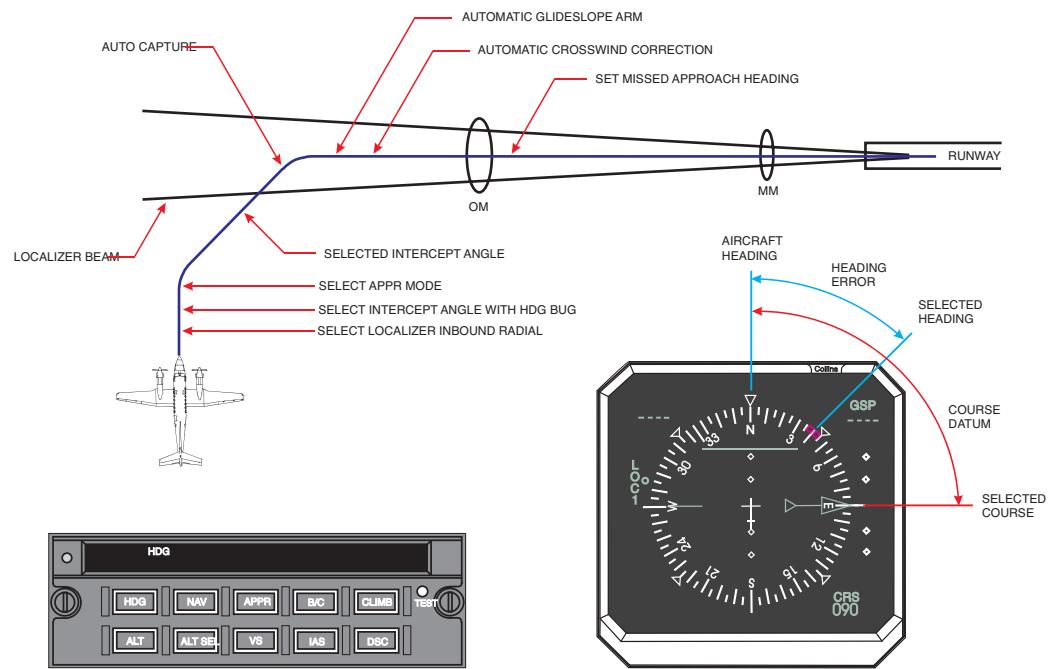


Figure 16-65 Flight Director APPR Mode LOC Before Heading Intercept

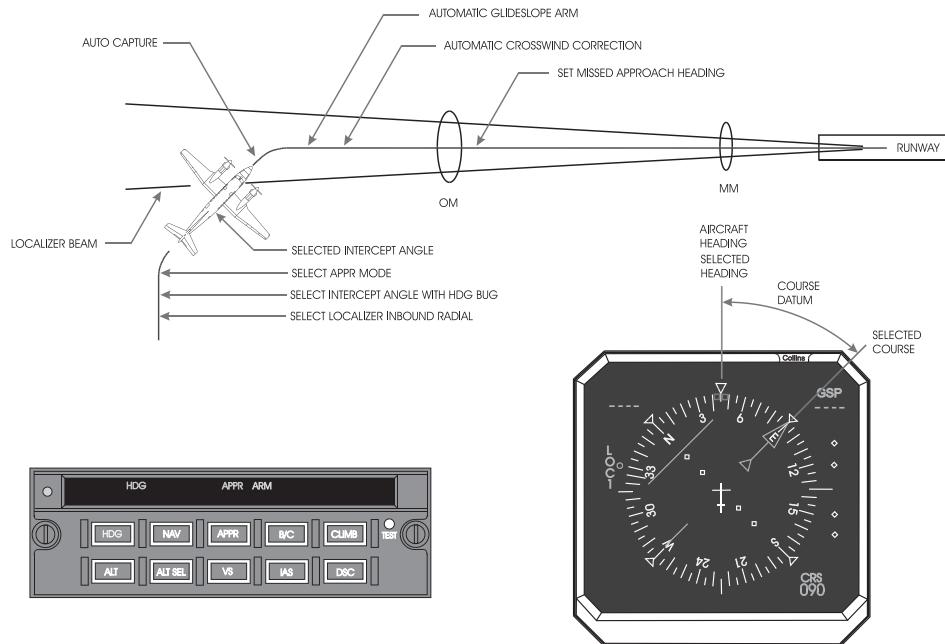


Figure 16-66 Flight Director APPR Mode LOC on Heading Intercept

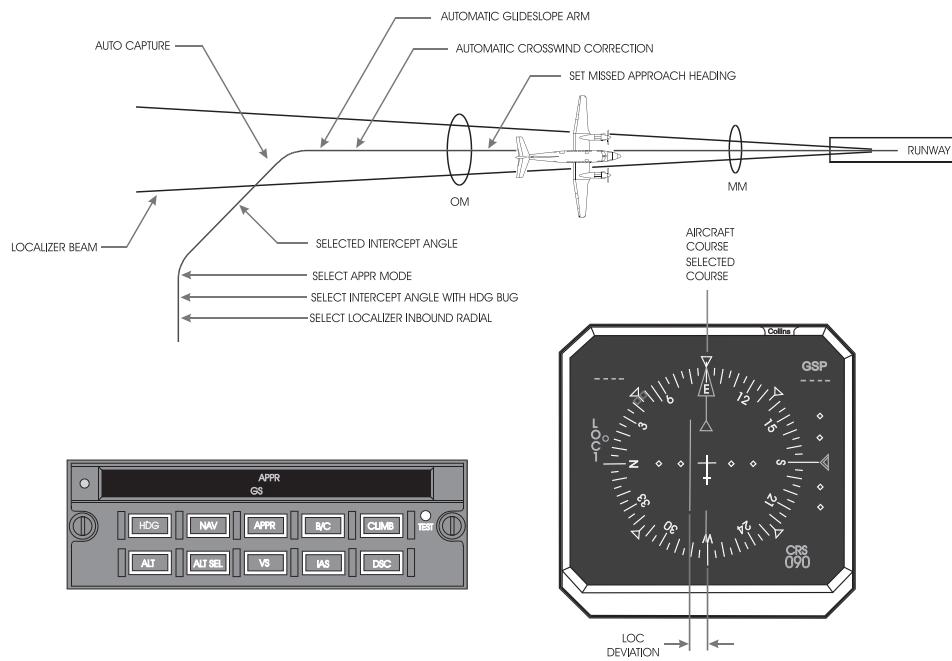
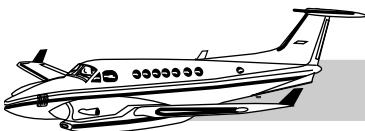


Figure 16-67 Flight Director APPR Mode LOC on Selected Course

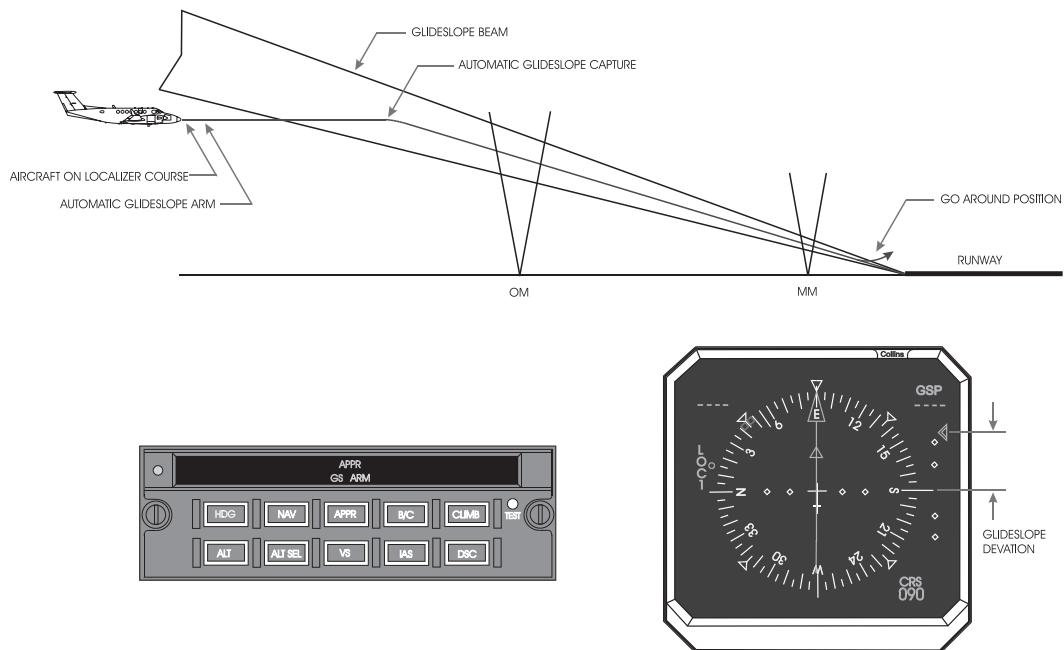


Figure 16-68 Flight Director APPR Mode GS Before Glideslope Capture

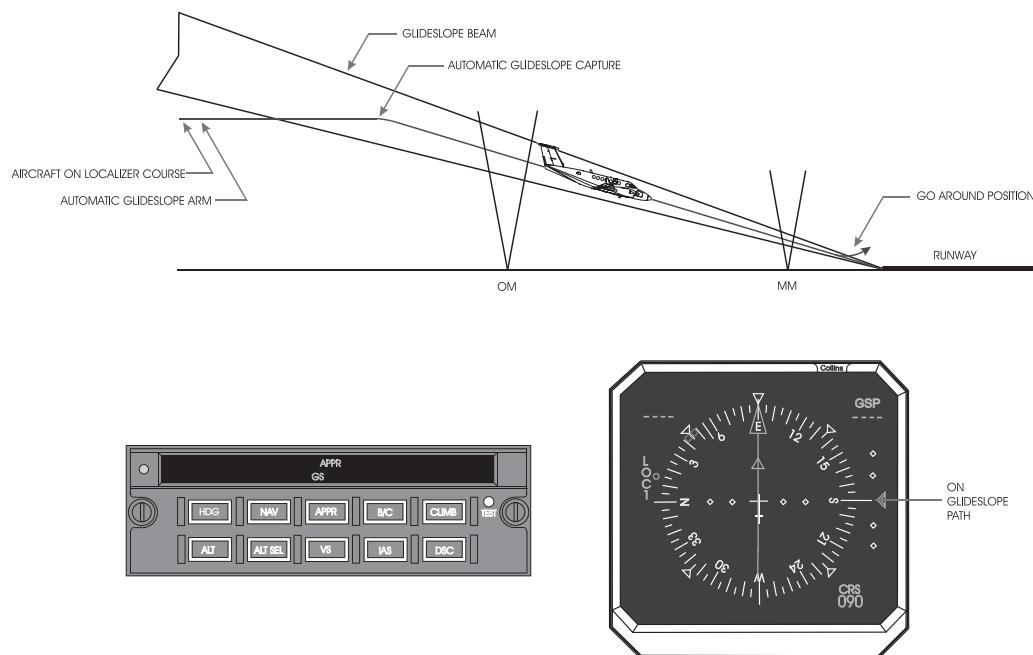


Figure 16-69 Flight Director APPR Mode GS After Glideslope Capture

deviation and glideslope deviation is supplied by the EFIS DPU to the autopilot computer.

At glideslope capture, the annunciation switches from GS ARM to GS and any other vertical modes are dropped. Annunciation during this mode consists of HDG, APPR ARM, APPR, GS ARM, and GS.

FMS Approach

This mode is similar to the ILS approach except the source signals are from the UNS Flight Management computer, which provides the lateral and vertical deviation signals to form a pseudo ILS (Figures 16-70, 16-71, 16-72, and 16-73).

When the FMS is the active sensor and detects that the aircraft is within the terminal area of the airport, it will automatically adjust the linear deviation scaling from enroute to approach

armed and then to approach active. On the EHSI, an annunciator SCALING will appear as the FMS is adjusting the deviation scale and then the FMS will annunciate APPR when the aircraft is on the final approach to the runway.

When the autopilot approach mode is selected, the autopilot will use the lateral deviation from the FMS computer to steer to the runway. The lateral deviation also arms the vertical mode which is annunciated VNAV ARM. At the appropriate point the system will capture the pseudo glideslope and track the vertical deviation signal to the runway.

The system annunciations consist of APPR ARM, APPR VNAV ARM and VNAV at the appropriate stages of the approach.

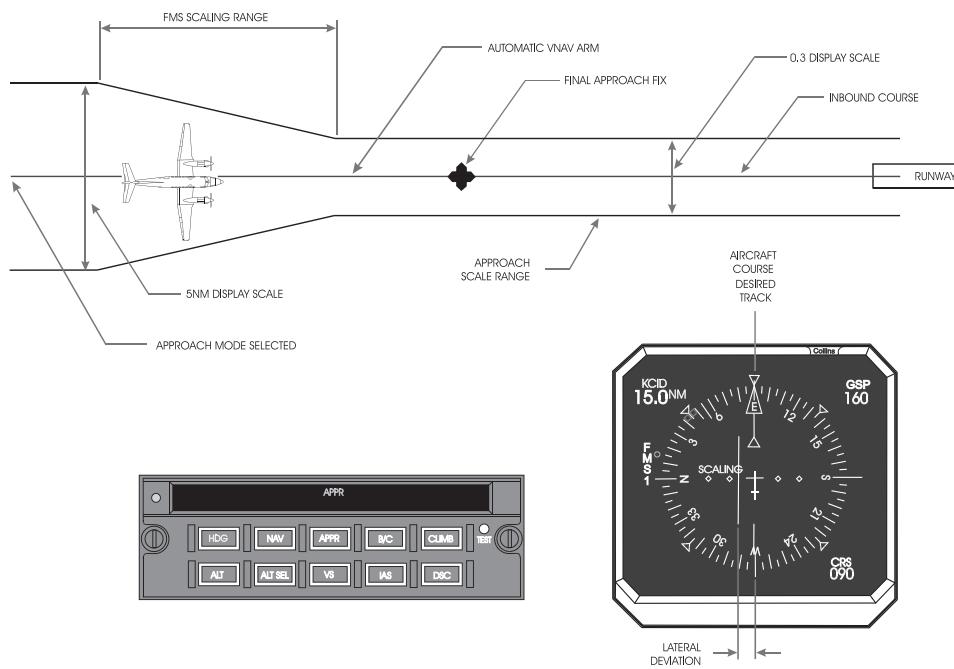


Figure 16-70 Flight Director FMS APPR Mode Lateral Mode Before Final Approach

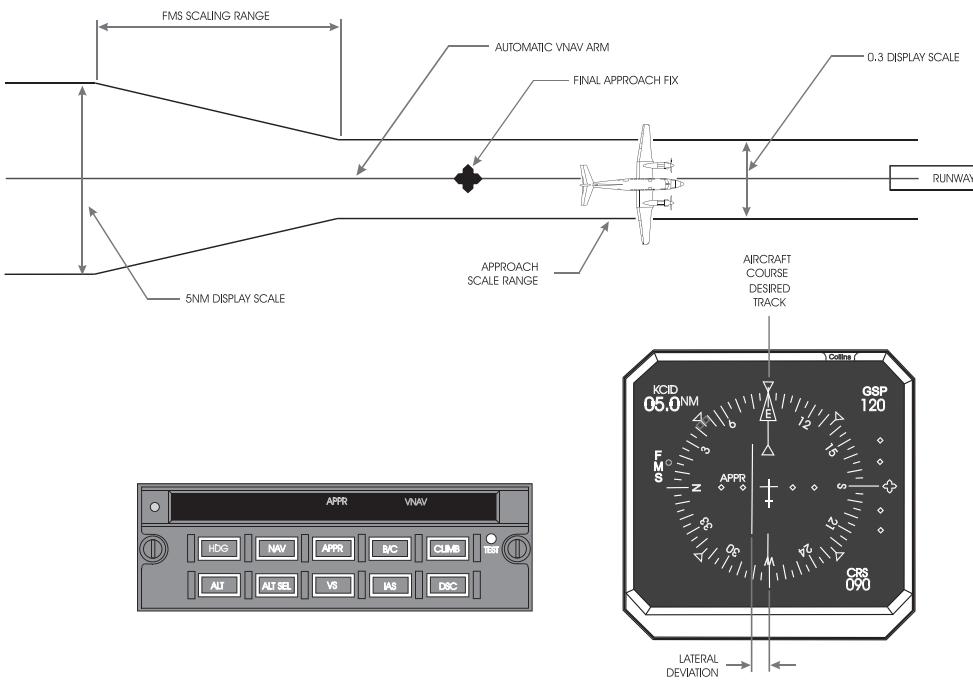


Figure 16-71 Flight Director FMS APPR Mode Lateral Mode on Final Approach



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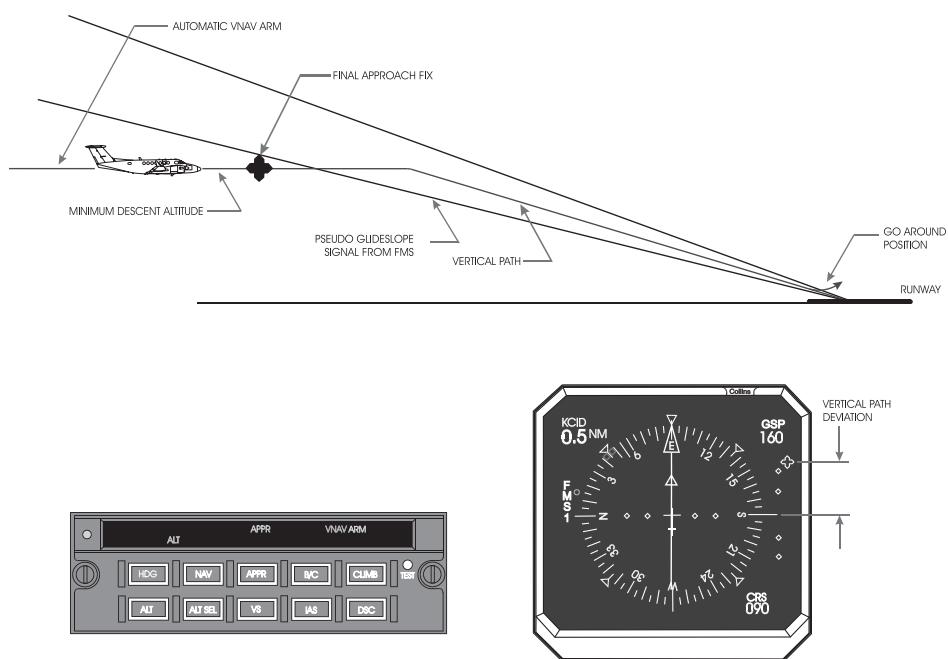


Figure 16-72 Flight Director FMS APPR Mode Before VNAV Capture

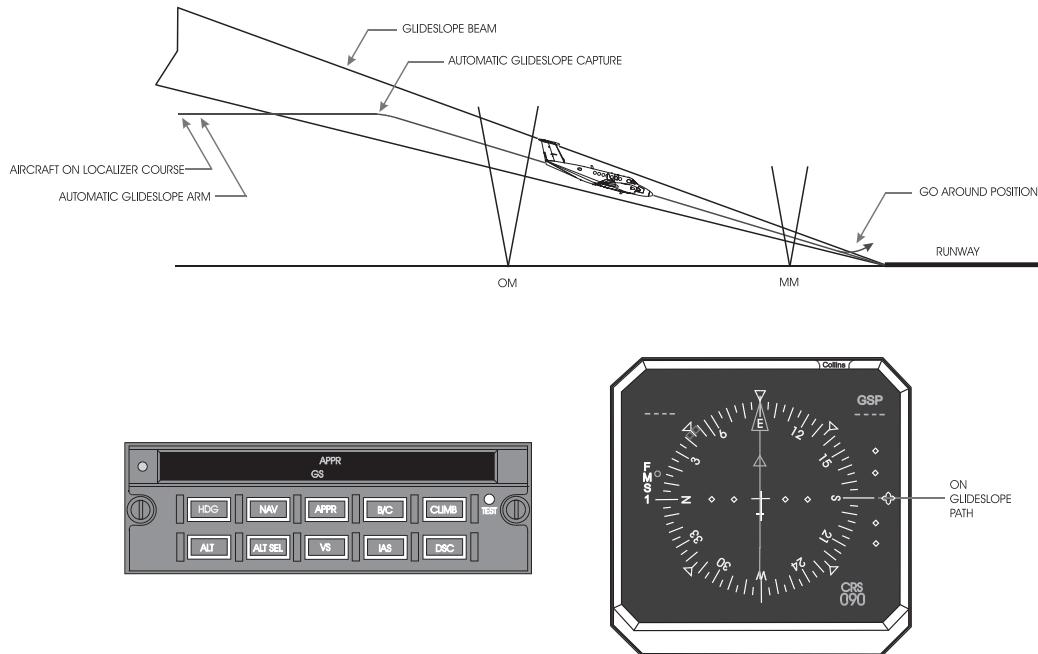


Figure 16-73 Flight Director FMS APPR Mode After VNAV Capture



Back Course Localizer Approach (B/C APPR)

Back course localizer approach is selected by pushing the B/C mode button. This mode is similar to the ILS approach mode except it provides commands to capture and track the localizer back course (Figures 16-74, 16-75, and 16-76). On the EHSI, LOC is selected as the active course sensor and the course arrow is selected to the inbound course to the runway. The EFIS DPU will detect a back course approach is being set up when a course arrow angle of over 105° from the current heading is selected and annunciates B/C. When B/C is annunciated on the EHSI, the deviation signals from the localizer will automatically be reversed to display the correct fly-to direction.

The EFIS system supplies the course datum signal and the heading error signal as in the ILS mode. In this configuration, the B/C annunciator is lit and the glideslope operation is locked out. Capture and track of the back localizer beam is the same as during the ILS approach except the course datum signal will be 180° instead of 0° as in an ILS approach when the system has captured LOC back course.

The system must be manually switched to the heading mode or the pilot must fly the aircraft manually as the aircraft approaches and flies over the localizer transmitter. Annunciation during this mode consists of B/C, HDG, APPR ARM, and APPR

Altitude Hold Mode (ALT)

The altitude hold mode provides commands to maintain the altitude at which the mode was selected (Figure 16-77). A lateral mode must be selected or the autopilot must be engaged before the ALT mode can be selected. The selected altitude can be changed in 25-feet increments by momentarily pushing the vertical control on the autopilot panel or by using a control wheel sync button and manually maneuvering the aircraft to a new altitude. The aircraft's vertical speed should be reduced to 500 feet per minute or less prior to selecting altitude hold for a smooth capture of the desired altitude. The ADS-65 provides the altitude hold signal.

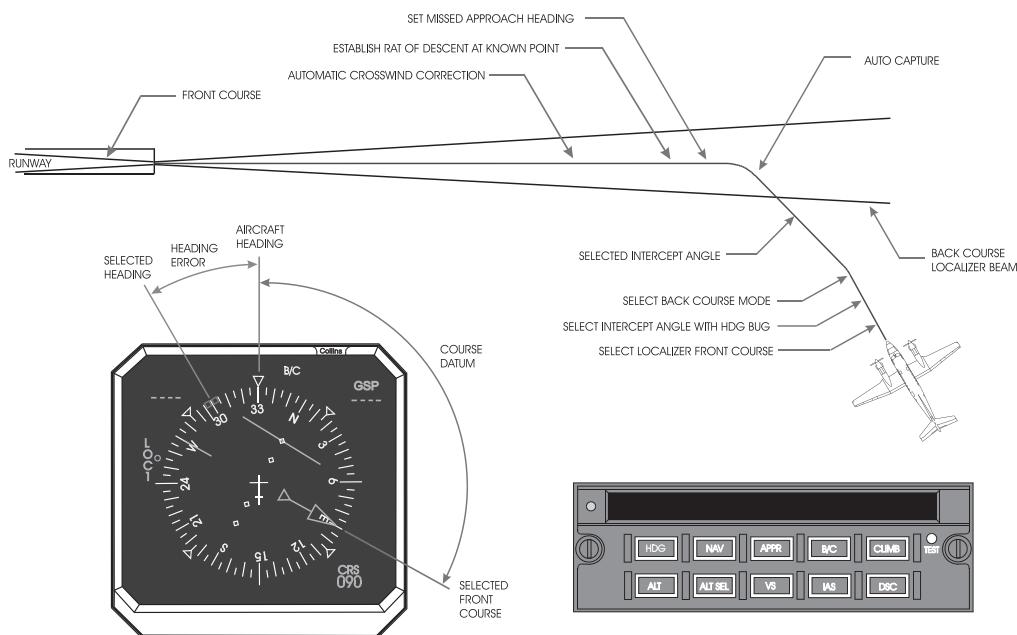


Figure 16-74 Flight Director Back Course Mode LOC Before Heading Intercept



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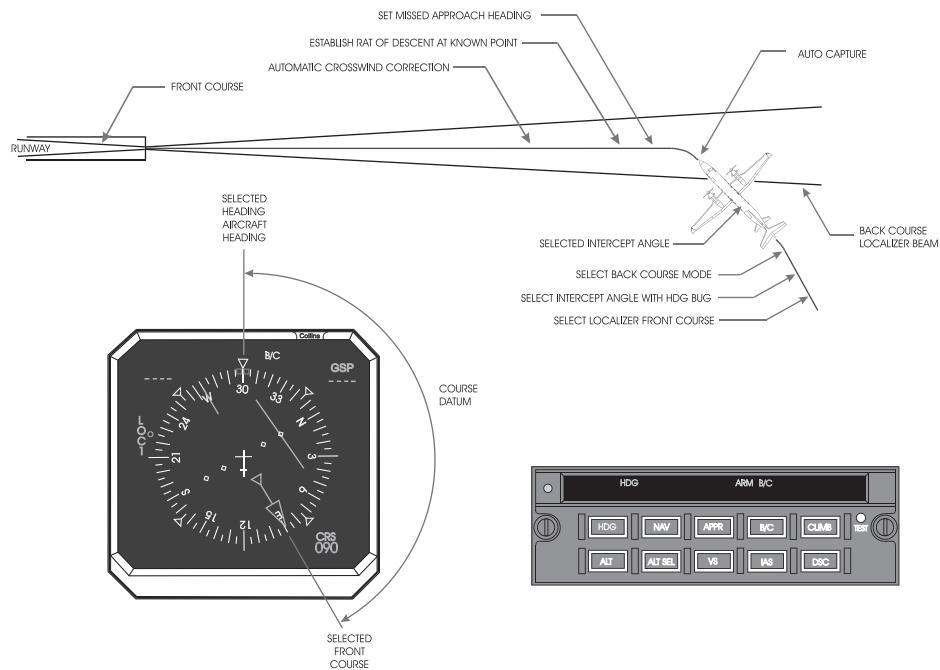


Figure 16-75 Flight Director Back Course Mode LOC On Heading Intercept

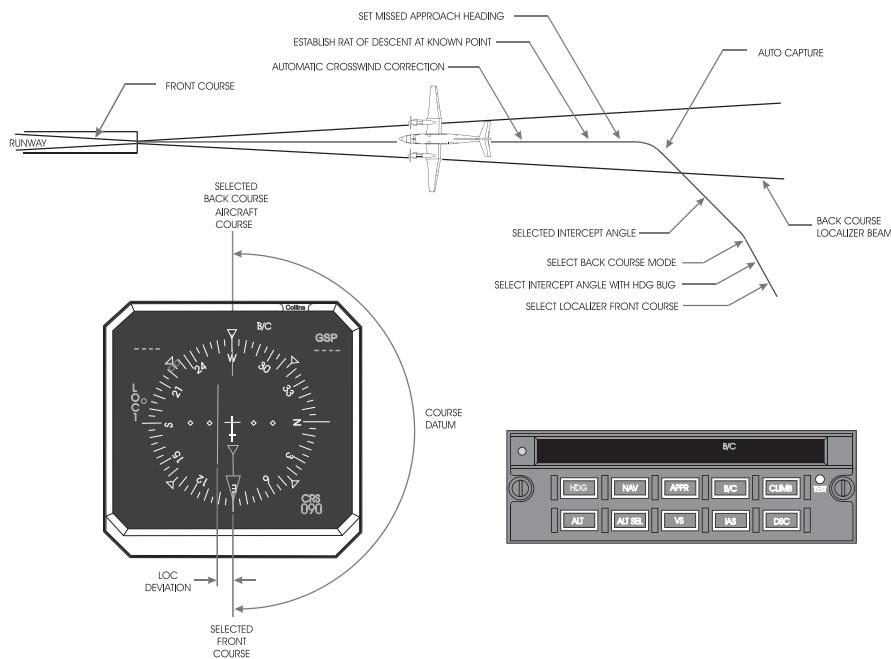


Figure 16-76 Flight Director Back Course Mode LOC On Selected Course

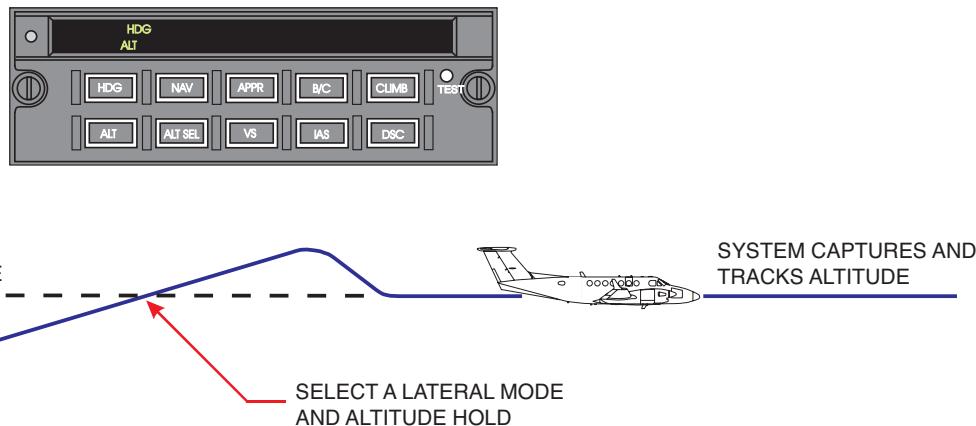


Figure 16-77 Altitude Hold Mode

Indicated Airspeed Mode (IAS)

The IAS mode is a hold mode that provides commands to maintain the indicated airspeed present at the time the mode is selected (Figure 16-78). A lateral mode must be selected or the autopilot must be engaged before the IAS mode can be selected. The vertical control can be used to change the IAS reference in 1-knot increments each time the control is activated. Also, using a control wheel sync and then changing the aircraft speed can be used to change the IAS reference. The ADS-65 provides the airspeed reference.

Vertical Speed Mode (VS)

The vertical speed mode (VS) is a hold mode that provides commands to maintain the vertical speed that was present at the time of mode selection (Figure 16-79). As with the other vertical modes, a lateral mode must be selected or the autopilot must be engaged before the VS mode can be selected. The VS mode reference can also be changed in 200 feet/minute increments by using the vertical control. Also using a control wheel sync and then changing the aircraft speed can be performed to change the IAS reference. The ADS-65 provides the VS mode reference.

Altitude Preselect Mode (ALT SEL)

The altitude preselect mode works in conjunction with the ADS-85 air data computer. The altitude preselect mode is automatically selected anytime a lateral flight director mode is selected or anytime the autopilot is engaged.

The desired altitude is set into the PRE-80C preselector which provides an altitude preselect error signal to the autopilot which is the difference between the current aircraft altitude and the selected altitude. The ALT SEL mode is automatically selected as described above or may be manually selected on the flight control panel. The ALT PRE select mode is also automatically reselected to a new altitude anytime the preselector knob is turned with the autopilot engaged or a lateral mode selected. When the knob is rotating the ALT ARM annunciator is flashing.

The altitude preselect mode provides commands to capture the desired altitude when it is approached. Another vertical mode is normally selected in conjunction with ALT SEL to establish a climb or descent to the preselected altitude. As the preselected altitude is approached, commands are provided to capture the altitude. After the altitude is captured, the system automatically switches to the altitude hold mode. The altitude error signal is provided from the PRE-80C to the ADC-85 air data computer which provides the signal on the autopilot computer.

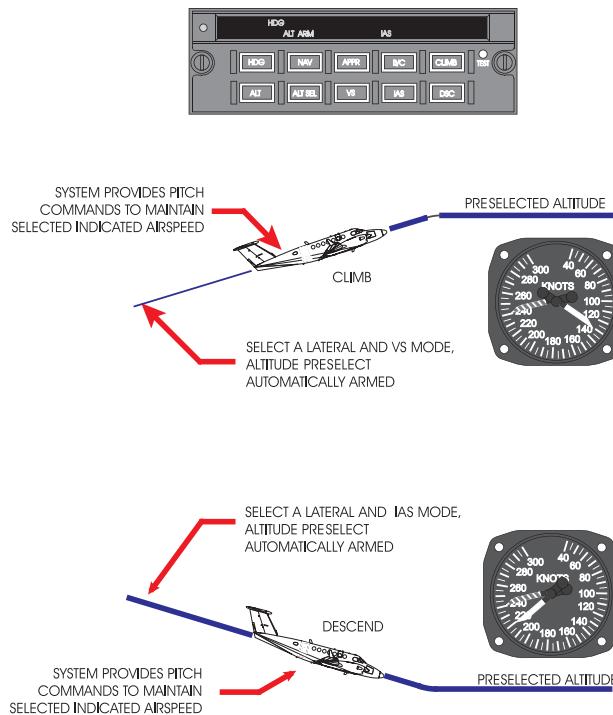


Figure 16-78 Indicated Airspeed Mode

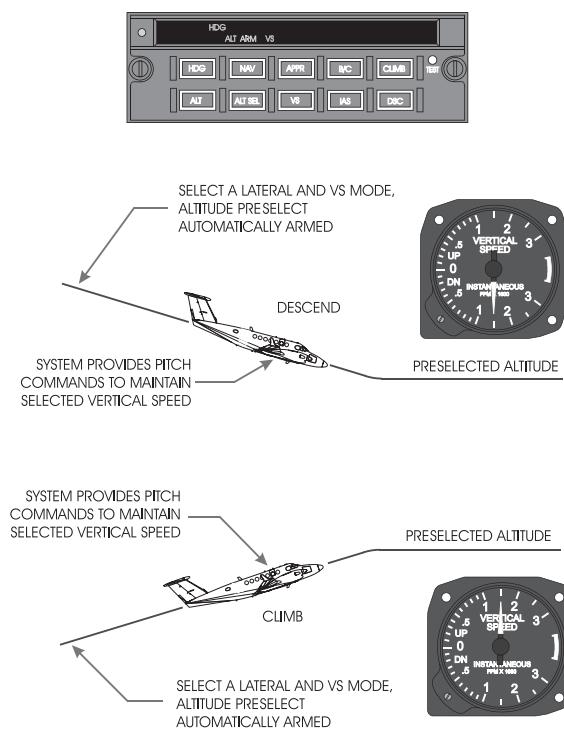


Figure 16-79 Vertical Speed Mode



Climb Mode (CLM)

The climb mode provides a smooth transition from the previously selected vertical mode to a preprogrammed vertical climb profile (Figure 16-80). The vertical profile is defined by a stored IAS command that is modified by the altitude of the aircraft. Selection of the mode automatically arms the altitude preselect mode, and commands a smooth pitch change to establish the preprogrammed IAS command. The CLM and ALT ARM annunciators illuminate when the mode is selected. The automatic selection of the altitude preselect mode can be canceled by pushing the ALT SEL button while the ALT ARM annunciator is illuminated. The system will hold the climb mode until another mode is selected or the preselected altitude is captured. The vertical control can be used to change the IAS reference in 1-knot increments each time the control is activated. However, the vertical control cannot be used to command a negative climb rate because the climb mode has a minimum climb rate of +50 feet/minute. If the climb mode is selected while the preselector is set at the present altitude, the system automatically cycles directly to the alti-

tude hold mode without climbing. The ADS-65 provides the IAS reference for use in this mode.

Descent Mode (DSC)

The descent mode provides a smooth transition from the previously selected vertical mode to a preprogrammed vertical speed descent (Figure 16-81). Selection of the mode automatically arms the altitude preselect mode, and commands a smooth pitch change to establish the preprogrammed vertical descent rate. The DSC and ALT ARM annunciators light when the mode is selected. The automatic selection of the altitude preselect mode can be canceled by pushing the ALT SEL button while the ALT ARM annunciator is illuminated.

After the DSC mode has been established, the vertical control on the APP-65A can be used to change the descent rate in 200-feet/minute increments. If the descent mode is selected while the preselector is set at the present altitude, the system automatically cycles directly to the altitude hold mode without descending. The ADS-65 provides the vertical speed reference for use in this mode.

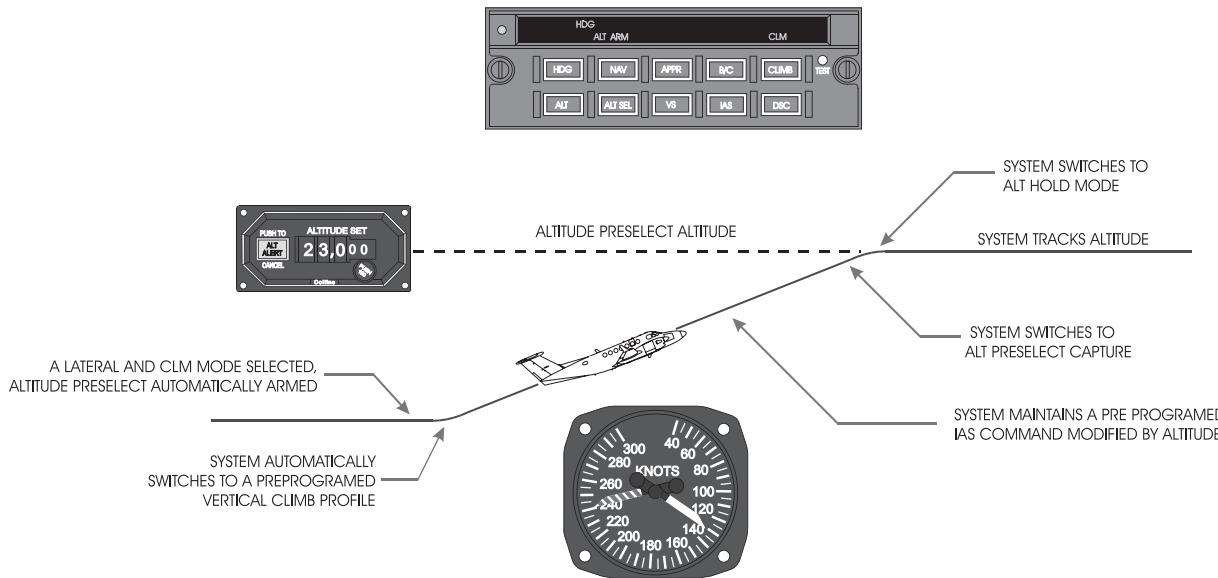


Figure 16-80 Climb Mode

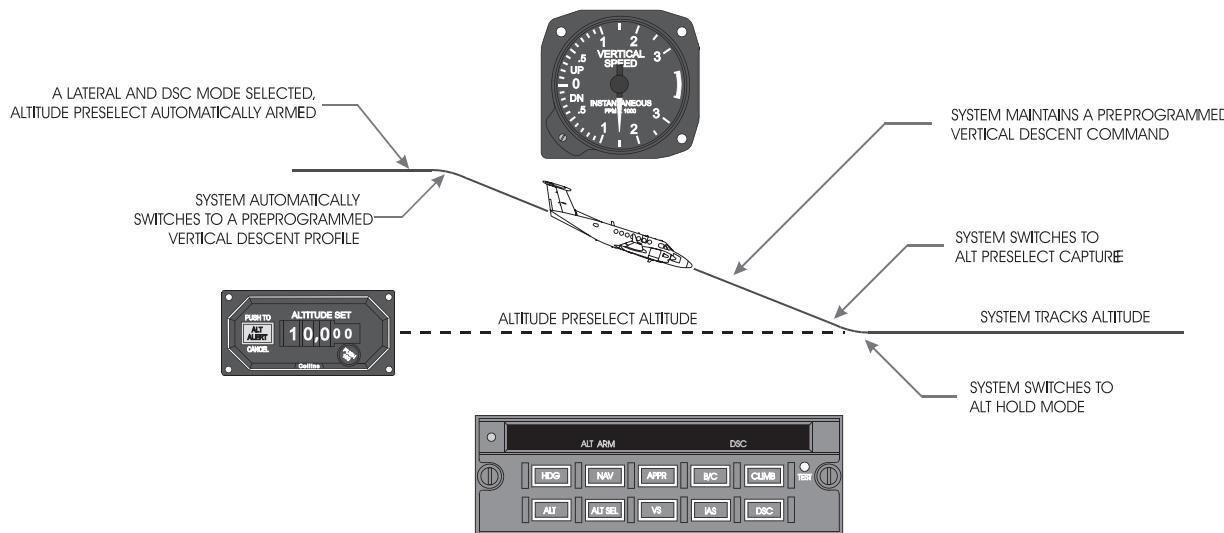


Figure 16-81 Descent Mode

Pitch Hold Mode

The pitch hold mode is selected when no vertical modes are selected and either the autopilot is engaged or a lateral mode (HDG, NAV or APPR) is selected. The system responds to the selected lateral mode and to the pitch attitude at the time the lateral mode was selected or the autopilot was engaged. The steering display is biased into view provided a lateral mode is selected. The pitch attitude reference can be changed with either the SYNC switch on the control wheel or the vertical control on the autopilot panel.

When the vertical control is pushed momentarily, it provides a 0.5° incremental pitch angle change. If the control is held for more than one second, it provides a $1^\circ/\text{second}$ slew rate until it is released. The pitch hold mode is not annunciated when this mode is active.

Go-Around Mode (GA)

The go-around mode is a wings-level, fixed pitch-up mode that is initiated by a pushbutton located on the power lever for the pilot and a control wheel switch for the copilot. Selection of the go-around mode disengages the autopilot.

The go-around mode can be selected from any lateral mode and is canceled by selection of a lateral mode. The system is internally synchronized to the aircraft pitch attitude while in the go-around mode and maintains the GA pitch attitude after a lateral mode is selected. If the SYNC button is pushed during the GA mode, the GA annunciator goes out, the system synchronizes to the aircraft pitch attitude, and the steering display stays in view even though no lateral modes are annunciated.

Autopilot Only Modes

Roll Hold Mode

The roll hold mode is the basic lateral operating mode and is selected when no other lateral modes are selected or when the turn knob on the APP-65A is moved out of the detent position. When the system is in the roll hold mode, the steering command bars on the EADI are biased out of view. Engaging the autopilot causes the system to hold the roll attitude present at the time of engagement provided the roll attitude is more than 5° from level.



The bank angle can be changed with either the turn knob on the autopilot panel or the PITCH SYNC & CWS button on the control wheel. Operation of the turn knob commands the autopilot to roll the aircraft smoothly to a roll attitude that is proportional to the displacement of the control. The roll attitude of the aircraft is limited at 30°. Activating the PITCH SYNC & CWS button disengages the aileron and elevator servos from the aircraft controls and permits the aircraft to be flown manually. The system will reengage and hold the roll angle present at the moment the PITCH SYNC & CWS button is released. Operation of the PITCH SYNC & CWS button while the turn knob is out of detent causes the turn knob to become inactive until the PITCH SYNC & CWS button has been released and the turn knob has been returned to the detent position.

When the turn knob is in the detent position or when the PITCH SYNC & CWS button is released with a roll angle less than 5°, the system switches to a wings level attitude hold submode, during which the aircraft is rolled to a wings level position.

Interlocks are provided to prevent the turn knob from taking command of the roll operation or using the PITCH SYNC & CWS button when the system is operating in the ILS approach mode.

Half-Bank Mode (1/2 Ø)

The half-bank mode can be selected on the APP-65A with any lateral mode except when the turn knob is used during the roll hold mode. When selected, this mode reduces the normal bank limit to approximately 12.5°. The mode is automatically dropped upon capture of the VOR or localizer signal during the NAV or APPR modes.

Soft Ride Mode

The soft ride mode, can be selected on the APP-65A and when selected, the soft ride mode changes the gain of various circuits in the computer to provide increased passenger comfort during turbulent conditions.

RUDDER BOOST SYSTEM

General

The rudder boost system senses engine torque from both engines through torque transmitters located on the forward left side of each engine. The torque transmitter output signal is sent to the APC-65J computer. The torque transmitters are powered from 28 VDC provided from the rudder boost CB and provide a proportional DC voltage output to engine torque (approximately 0 torque = 0 VDC, max torque = 5.5 VDC). The autopilot computer controls the operation of the rudder boost system. The rudder boost mode is independent of the autopilot and yaw engage functions and is fully automatic, when armed. This mode takes precedence over yaw damper operations whenever the computer detects a need for rudder boost and provides either right or left deflection of the rudder as required. When differences in engine torque between the engines exceeds a preset level, the autopilot computer activates the rudder servo in the aft fuselage section which deflects the rudder. This operation assists pilot effort in controlling the aircraft yawing. Servo contribution is proportional to engine torque differential. Trimming of the rudder must be accomplished manually by the pilot.

OPERATION

Activation of the rudder boost system is through a switch in the pedestal extension, placarded RUDDER BOOST—OFF—YAW CONTROL TEST. Switching to the RUDDER BOOST position arms the system. The rudder boost system is disabled if the RUDDER BOOST switch is OFF and is interrupted when the DISC TRIM/AP YD switch on the control wheel is depressed to the first level.

A yellow caution annunciation (KA-350 only), RUD BOOST OFF, is provided on the caution/advisory annunciation panel to indicate when the rudder boost control switch is in a position other than RUDDER BOOST (ON).



ALT-50A RADIO ALTIMETER SYSTEM

General

The ALT-50A radio altimeter system provides the pilot with a precision altitude indication during the approach phase of the flight. The system consists of a transceiver, and two antennas flush mounted on the belly of the aircraft. The radio altitude readout is displayed on the EFIS-85 EADI display and on the 339H-4 RAD ALT indicator (Figures 16-82 and 16-83).

The ALT-50A radio altimeter operates automatically when the aircraft is within 2,000 feet of the ground. It transmits RF energy to the ground from one antenna and receives the reflected energy back into another antenna. The amount of time the signal uses to transmit and to be received is converted into exact altitude over the terrain directly below the aircraft.

The ALT-50A system measures aircraft altitude up to 2,000 feet. In addition, the EFIS system and the RAD ALT indicator have adjustable decision height (DH) warnings made available on the respective displays which are tripped by the radio altimeter signal as the aircraft reaches the DH.

A DH output from the DPU and MPU is used to provide an aural tone warning at decision height. The indicator also generates a decision height output based on its index setting, which will illuminate an annunciator on the indicator and on a remote annunciator on the copilot's instrument panel.

To carry out the RAD ALT test, two test switches are available that can be used to place the ALT-50A into test mode. The RAD ALT test switch on the display control panel (DCP-85G) provides a digital message to the DPU and MPU, which will then provide a ground to the radio altimeter. The indicator has a test switch, which will also output a ground to the ALT-50A. Either test switch can be used to test the RAD ALT, but if the autopilot is engaged testing of the RAD ALT is inhibited by the use of the engage relay.

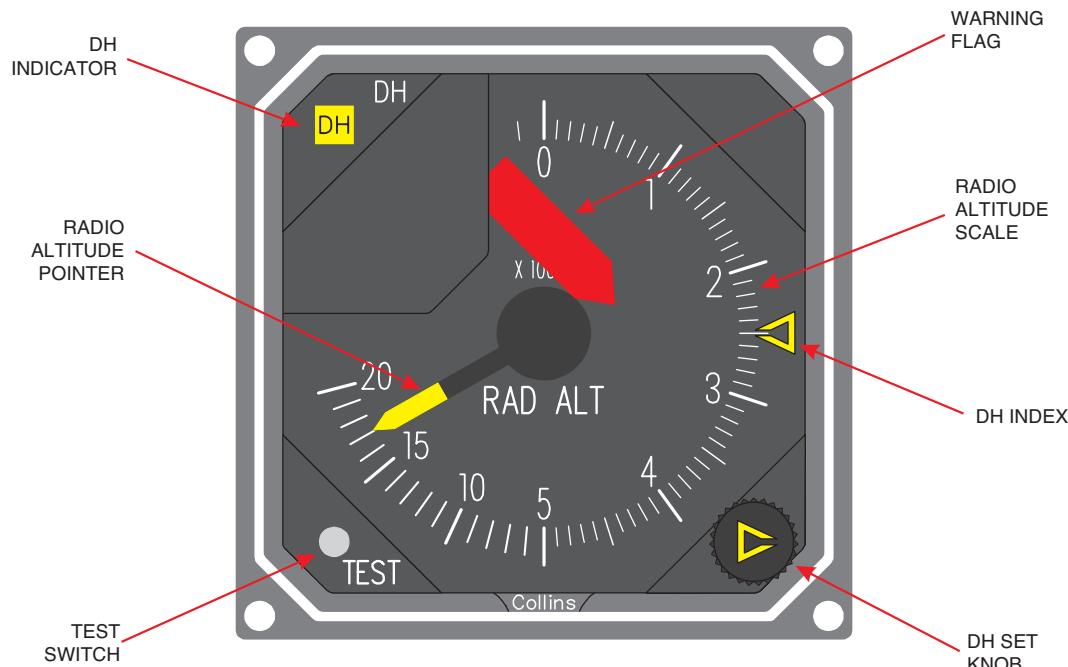
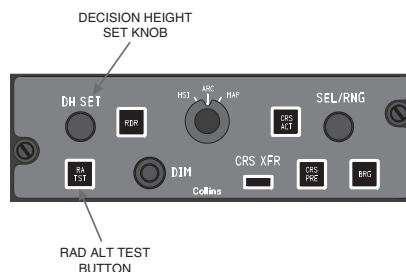


Figure 16-82 339H-4 Radio Altitude Indicator


Table 16-13 RADIO ALTITUDE INDICATOR, CONTROLS AND INDICATORS

CONTROL/INDICATOR	FUNCTION/DESCRIPTION
Pointer and dial scale	The pointer indicates aircraft height above the terrain during the approach phase of flight. The scale of the indicators displays altitude between -20 and 2000 feet.
Decision height (DH) index and DH adjustment knob	The DH knob selects the decision height, which is displayed by the decision height index.
DH indicator	The DH indicator lights when the aircraft descends to the selected decision height, as indicated by the DH index.
Warning flag	The warning flag indicates a loss of power whenever it comes into view during normal operation (excluding self test).
TEST switch	Pushing the TEST switch on the 339H-4 indicator initiates a functional test of the ALT-50A radio altimeter system (excluding antennas). When the ALT-50A system, excluding antennas, is functioning properly, the pointer indicates a self test altitude of 50 ± 5 feet.


PILOT'S EADI RADIO ALTITUDE DISPLAY

PILOT'S DISPLAY CONTROL PANEL (DCP-85G) RAD ALT CONTROLS
Figure 16-83 Radio Altitude Controls



Locations

The ALT-50A radio altimeter transceiver is located under the cabin floor for the B300 aircraft. The bottom-mounted antennas are located under the fuselage in the wing center section area.

Self Test

1. Apply power to the ALT-50A radio altimeter system and the EFIS system.
2. Adjust DH knob on the display control panel (DCP-85G) and the 339H-4 RAD ALT indicator to set the DH index to 50 feet.
3. Push and hold the RAD ALT TEST button on the display control panel (DCP-85G).
4. Observe the EADI RAD ALT display for the following indications:
 - The RAD ALT digital display indicates 50 ± 5 feet.
 - The DH annunciator appears flashing briefly before going steady.
 - The DH display flashes during the test.
5. For the 339H-4 indicator observe the following indications:
 - The RAD ALT pointer drives to 50 ± 5 feet.
 - The DH indicator lamp on the indicator illuminates.
 - The remote copilot panel DH annunciator illuminates.
6. The decision tone is heard through the speakers/headset.
7. Release the TEST button and observe EADI RAD ALT display. The RAD ALT displays should be zero foot nominal.

RADIO MAGNETIC INDICATOR SYSTEM

General

The airplane is equipped with two identical Collins RMI-30 radio magnetic indicators (Figure 16-84); one on the pilot's instrument panel, and one on the copilot's instrument panel. The RMI is an internally-lighted, panel-mounted, dual-needle indicator that displays the aircraft magnetic-heading information and bearing to either VOR or ADF stations. Each pointer is read against the compass card with a fixed lubber (index) line. Each pointer operates independently and can be switched to separate VOR or ADF receivers. The RMIs derive power from the aircraft's 28 VDC and 26 VAC electrical power systems. The 28 VDC is used to power the internal circuits of the RMI and the 26 VAC from the RMI fuse is used to provide a reference voltage for the compass heading signal and the NAV reference signal.

Compass information is displayed by the servoed compass card. A compass flag is used to show loss of power, compass invalid condition, and servo error. With the RMI operational, bearing information can be displayed from NAV system No. 1, or the ADF system No. 1 on the signal bar pointer and NAV system No. 2, or the ADF system No. 2 (if installed) on the double bar pointer. A choice of ADF or VOR heading information on the single bar pointer and double bar pointer is made by placing the single or double bar pointer button on the front of the indicator in the ADF or VOR position. The pointers will be stored in a horizontal 3 o'clock position when the NAV or ADF signal is invalid, or when the selected NAV or ADF receiver is turned OFF.

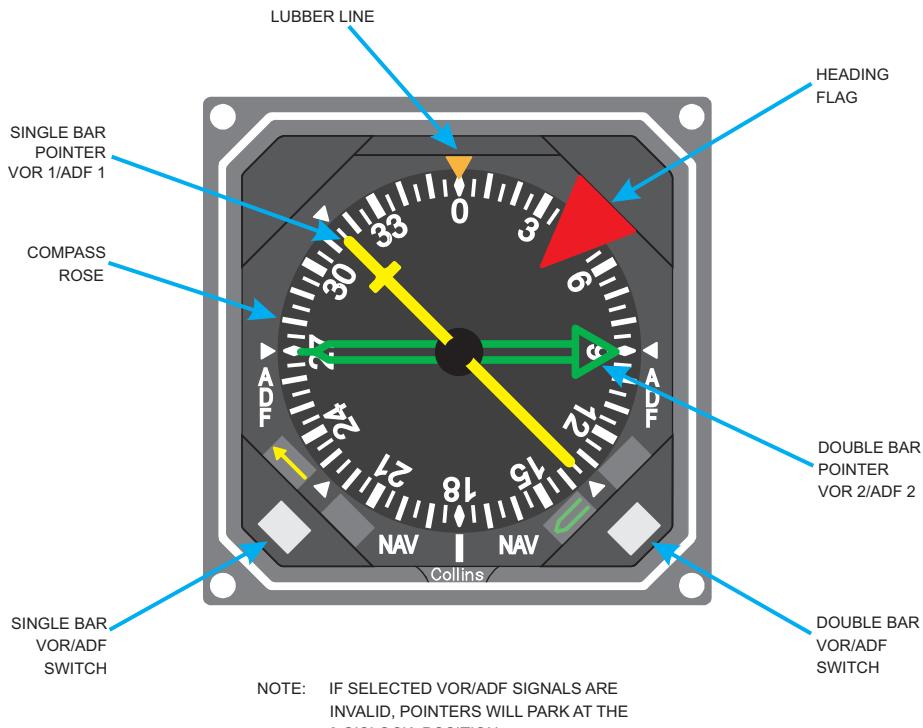
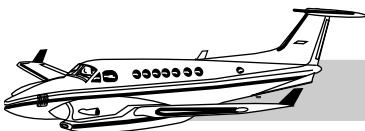


Figure 16-84 RMI-30 Radio Magnetic Indicator

ADS-85 AIR DATA SYSTEM

General

The Collins ADS-85 air data system on the B300 (350) aircraft provides altitude and airspeed information for the pilot, preselected altitude alerts and altitude preselect error to the autopilot.

ADC-85 AIR DATA COMPUTER

The ADC-85 air data computer senses pitot, static pressure and outside air temperature as its sensor inputs.

For the pilot's instruments, the ADC-85 provides digital altitude data to the ALI-80A altimeter. The baro-corrected altitude from the ALI-80A is sent to the PRE-80C alerter/preselector. The

PRE-80C sends data to the ADC-85, which consists of the selected altitude, the error altitude and the baro-corrected altitude.

For the EFIS system, the ADC-85 provides air data information to the EFIS system DPU and MPU, which drives the digital airspeed readout on the EADI and air data information, which may be displayed in the extended data pages on the MFD. The ADC-85 also provides ARINC 429 digital air data information to the flight management system as one of its sensor inputs.

For the autopilot system, the ADC-85 provides an analog altitude preselect error signal and a 28 VDC valid discrete. The valid discrete is removed if the air data system fails or the altitude alerter preselector knob is rotated.



Inputs required by the ADC-85 are pitot static pressure, 28 VDC for circuit operation, a strut switch position and a remote test switch located on the pedestal. In addition, outside air temperature is provided from an outside air temperature sensor. This outside air temperature probe is not heated for anti-icing since it has a natural ice shedding capability due to its shape.

An ADM-85 air data module is installed on top of the computer and contains flight characteristic data specific to the B300 aircraft. This air data module must stay with the aircraft and must be reinstalled in the replacement computer if it is ever replaced.

The air data computer is located in the left avionics bay.

ALI-80A BAROMETRIC ALTIMETER

The ALI-80A all electronic altimeter (Figure 16-85) on the instrument panel provides baro-corrected altitude to the pilot, pressure altitude to the Mode S transponder and baro-corrected altitude data to the altitude preselector. The altimeter requires 28 VDC for its operation, along with digital data from the ADC-85. The altitude display is a drum pointer type display along with the barometric setting display.

PRE-80C PRESELECTOR/ALERTER

The PRE-80C altitude preselector (Figure 16-85) is used to provide alert warnings in the form of annunciators and an alert tone when approaching or deviating from the selected altitude. The PRE-80C provides a preselect error signal to the autopilot to allow the aircraft to level off at the selected altitude. In climb mode, for example, the autopilot will level the airplane when the preselected altitude is reached. The preselector alerts the pilot when the selected altitude is within 1,000 feet. The light will automatically cancel when the selected altitude is within 200 feet. After the aircraft acquires the selected altitude, if the aircraft deviates more than 200 feet the light

will flash and the audio tone will turn on. A press of the ALT ALERT button cancels the flashing.

The PRE-80C inputs are baro-corrected altitude data from the ALI-80A altimeter and 28 VDC inputs for internal operation and lighting power from the annunciator dimming circuits. Outputs are provided to a remote ALT ALERT annunciator on the copilot's panel, to the EFIS display control panel to activate the ALT ALERT annunciator on the EADI and to the aural warning tone generator for the alert tone. A select knob is provided to set the selected altitude and displayed on a drum display.

ADS-85 AIR DATA SYSTEM SELF TEST

The ADS-85 Air Data System has built-in test capabilities that can be used to locate a faulty component in the system. Self Test is inhibited when the aircraft is airborne.

For the ALI-80A altimeter, a red warning flag comes into view when internal monitoring circuits detect a failure, or when data from the air data computer is lost. If able, the pointer will go to the nearest 250-foot mark. When the ALI-80A altimeter test switch on the face of the indicator is pressed and held, the warning flag comes into view in half a second and the altitude pointer goes to the nearest 750-foot mark. If these indications do not occur, the altimeter is faulty.

For the PRE-80C altitude preselector, a red warning flag comes into view when internal monitoring circuits detect a failure, or when data from the ALI-80A altimeter is lost. To test the PRE-80C altitude preselector, press the altitude alert button, the button should illuminate, and the flag should come into view. If these indications do not occur, the altitude preselector is faulty. The aural altitude alerting function is not tested.

Test of the air data system can also be initiated at the ADC computer or at the remote ADC test switch on the EFIS power panel. The computer must be powered up for at least 45 seconds for a valid test. To test the ADC-85 press the test

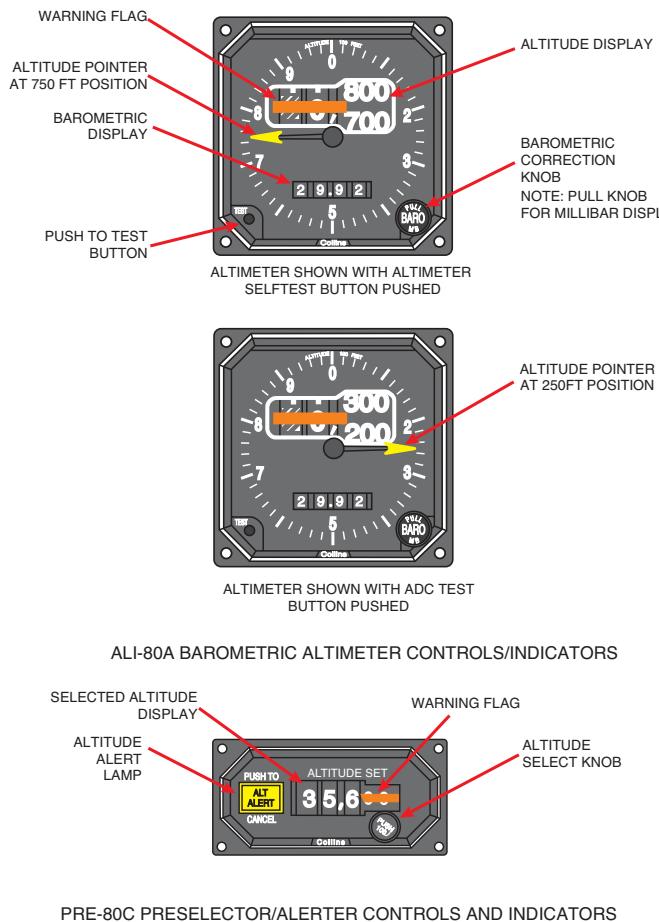
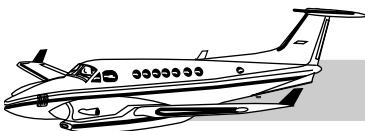


Figure 16-85 ADS-85 Instrument Displays

switch. The altimeter also goes into a loss-of-data mode. Now the altimeter parks at 250 feet instead of 750 feet, indicating the altimeter has, in fact, lost the data from the ADC-85.

STALL WARNING SYSTEM

The stall warning system provides a precise pre-stall warning to the pilot by activating a warning tone (horn) when specific lift coefficients are reached. The activation point of the warning tone is changed by the flap position.

The stall warning system consists of the following major components; the lift computer, the aural warning system, LH landing gear strut switch, a stall warning self test switch, a 5-amp circuit breaker (furnishing power to the system), a heat panel for the transducer, a stall warning heat sensor control relay and a 15-amp circuit breaker (providing power to the heat panel, flap position switches and the aural warning tone generator).



LIFT TRANSDUCER

The lift transducer senses the angle of attack and provides a signal to the lift computer. The transducer has a sensor assembly powered by an excitation voltage of 10 VAC at 700 Hz, which is provided by the lift computer. A small vane protruding into the airstream responds to the change in lift coefficient of the wing with change in angle of attack, and transmits an AC signal output to the lift computer. The vane is protected from icing by an integral anti-icing heater and the lift transducer base also has an anti-icing heater built into the mounting face.

LIFT COMPUTER

The lift computer is a solid-state electronic unit incorporating power distribution, computation, monitoring, test and warning functions. The computer will process signals from the transducer, the flap position inputs and a crew compartment test switch. The lift computer will supply an output ground signal to activate the stall warning tone from the tone generator when the speed ratio decreases below the value indicated for flap position.

The lift computer incorporates adjustment potentiometers to adjust the stall warning activation point for flaps up, flaps approach and flaps down. The lift computer has a built-in self test system which when activated by the cockpit switch, will provide a 15 VDC test signal to bias the lift transducer vane and activate the stall warning output/horn.

The stall warning system is enabled only while the aircraft is in flight via a ground from the left strut switch. While the aircraft is on the ground the system is inhibited.

SURVEILLANCE SYSTEMS

WEATHER RADAR SYSTEMS

The B300 (350) aircraft are equipped with the WXR-840 color weather radar as standard, with the TWR-850 radar as an option. Operationally, there is no difference between the WXR-840 and TWR-850 apart from the turbulence detecting circuits and two extra mode switch positions for the turbulence mode of operation. Therefore the WXR-840 will be covered with the added features pointed out for the TWR-850.

WXR-840/TWR-850 General

The WXR-840/TWR-850 weather radar systems provide the crew with a display of radar-detectable precipitation within 60° on either side of the flight path. In addition to wet precipitation detection, the TWR-850 radar has a turbulence detection capability, which alerts the crew to the location of turbulent conditions. This feature depends on the liquid state of the turbulence. It is made possible by the use of an extremely stable transmitter, which allows detection of very small shifts in frequency indicating a relative velocity of the detected precipitation particles. The radars only transmit 24 watts of power, a much lower value than the earlier generation radars.

The system consists of two separate units; the RTA-842/RTA-852 receiver/transmitter/antenna and the WXP-840A/WXP-850A weather radar panel (Figures 16-86, 16-87, 16-88 and 16-89). The pilot's EFIS system, EHSI and MFD are used to display the radar presentation. The mechanical feature that distinguishes the RTA-842 unit from earlier generation weather radar designs is its one-piece construction, combining the receiver, transmitter, and antenna into a single unit. The forward portion of this unit is the 12-inch flat-plate antenna. Directly behind the antenna is the RF assembly, consisting of the transmitter and receiver. Mating the antenna and receiver-transmitter eliminates the need for a waveguide. The RTA (receiver/transmit-

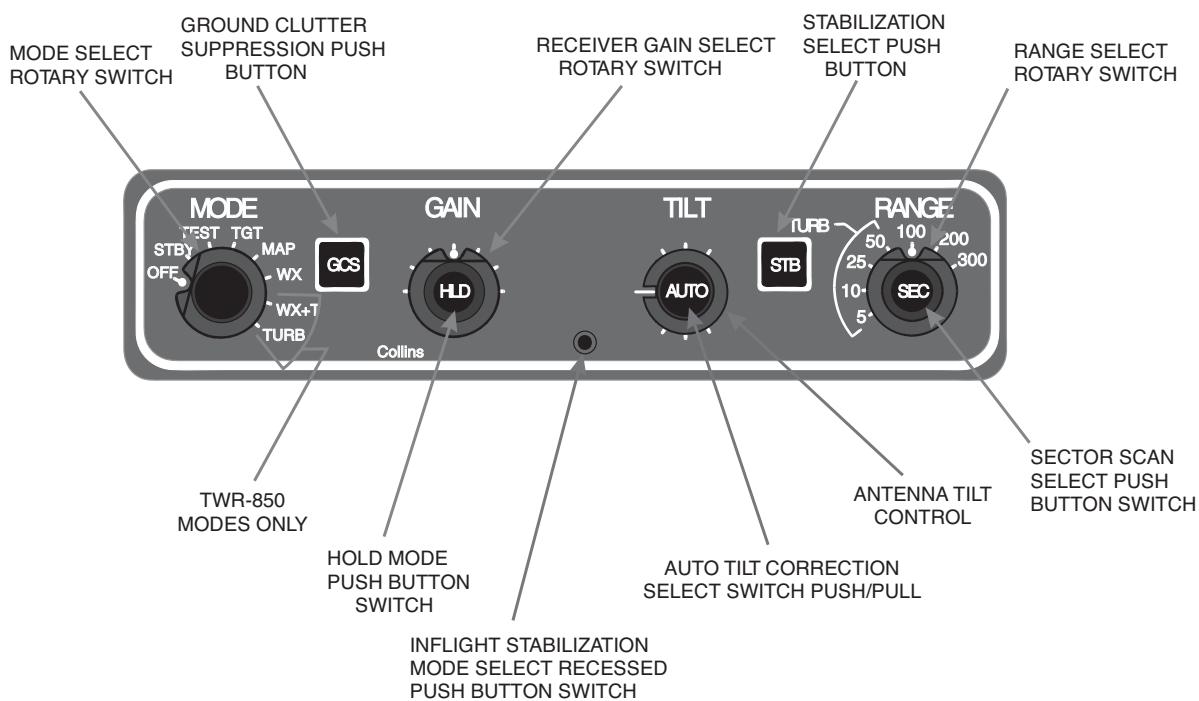


Figure 16-86 WXP-840A / WXP 850A Weather Radar Control Panel Controls

ter/antenna) assembly is mounted on the drive assembly and therefore swings from left to right as the system scans and points up or down for tilt. The drive assembly contains the motors and mechanical system for the scan and tilt functions and is attached to the base assembly. It serves as the mounting base and contains the power supplies and signal processing portion of the RTA-842 unit. The complete RTA-842 assembly is mounted on the forward bulkhead and enclosed by a radome. It is vital to proper performance of any radar system that the radome be at least 85 percent RF transparent.

The WXP-840A weather radar panel is mounted below the MFD and provides the operating controls for the radar system. The WXP-840A also supplies the final data processing function to format the video data as needed by the EFIS system.

Controls and Displays

The WXP-840B weather radar panel provides mode control, range selection, and other system operating controls for the WXR-840 weather radar system. Weather radar information from the WXP-840A is displayed on the pilot's EHSI and MFD when the radar display is selected.

The following paragraphs describe the WXP-840B operating controls.

Mode Switch Operating Modes

The MODE control (Figure 16-86) is a rotary switch used to select the various operating modes of the WXR-840 system. The MODE control switch positions and the corresponding modes are as follows:

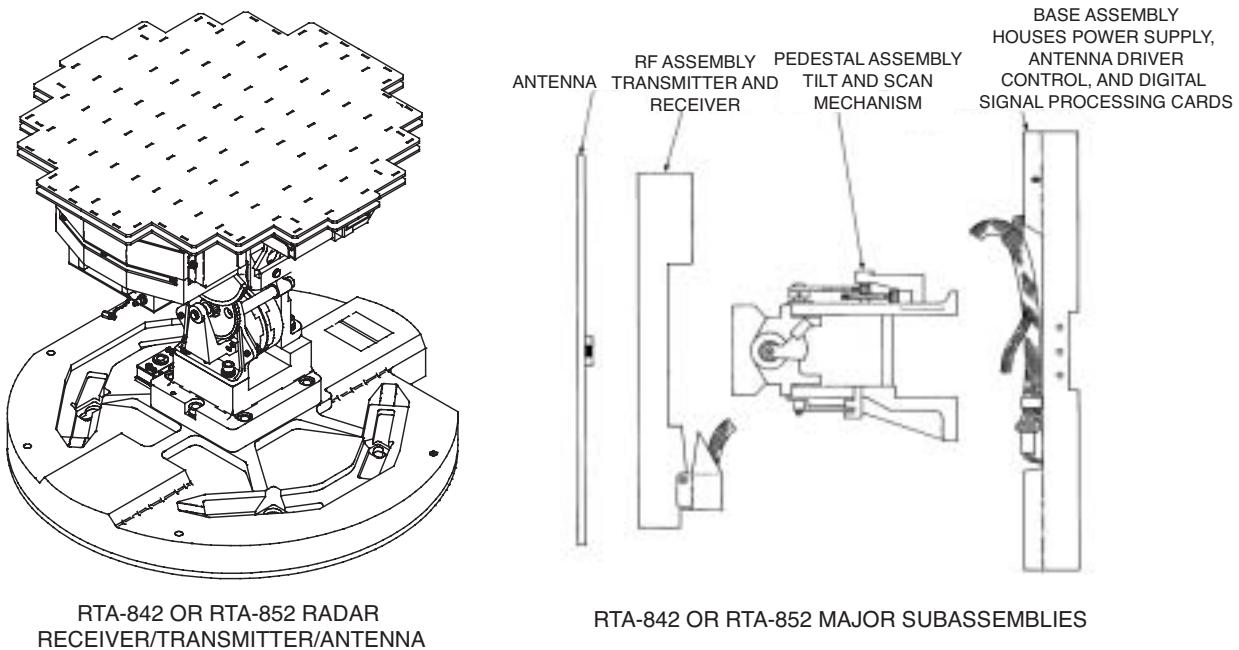


Figure 16-87 RTA-842/852 Radar Receiver/Transmitter/Antenna

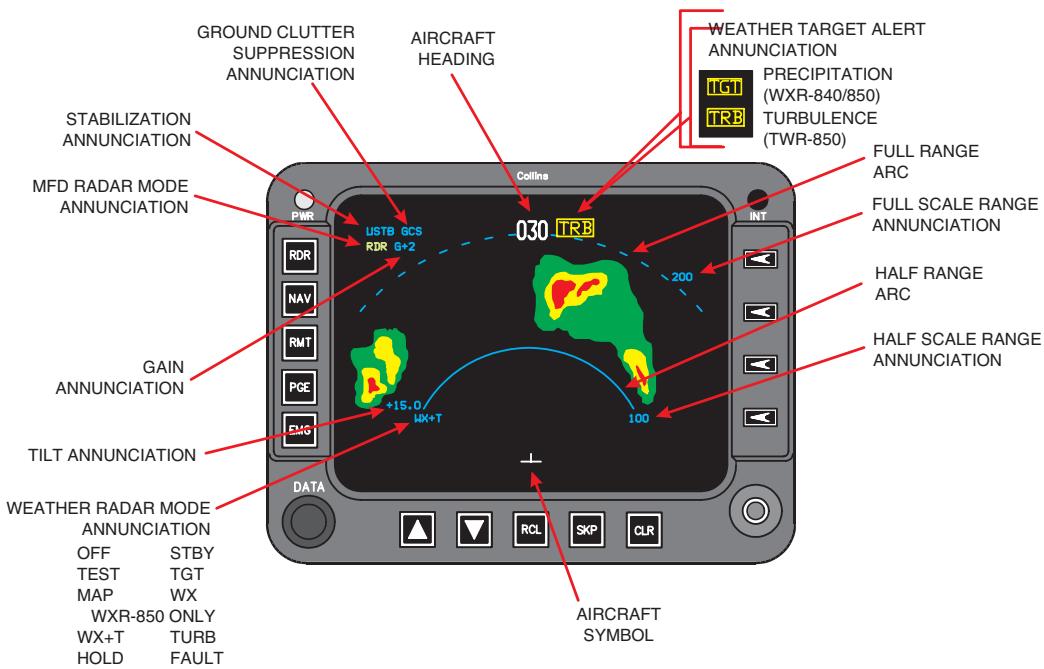


Figure 16-88 MFD Weather Radar Display



Figure 16-89 EHSI Weather Radar Display

OFF

OFF mode removes power from the RTA-842 receiver/transmitter/antenna and WXR-840A weather radar panel. OFF will be displayed on the EHSI radar display.

STBY (Standby)

In STBY mode, power is applied to the RTA-842 receiver/transmitter/antenna with a short (approximately three-second) initialization period. During the initialization process, the RTA drives its antenna to the boresight position (0° azimuth and 0° tilt). The antenna will remain in this position as long as STBY is selected. The weather display, radar transmitter and antenna scan drive circuits (after initialization) are inhibited whenever STBY is selected.

TEST

TEST mode initiates the radar self test function and causes the radar self test display to appear. TEST is annunciated on the EHSI and MFD

radar display (Figure 16-90). Any range can be selected for the self test function. The radar self test display consists of six colored arcs evenly spaced across the full sector of the display. Each arc verifies the system's ability to process and display the precipitation intensity level or other display data that is normally represented by that color. At the apex, occupying a range approximately equal to two of the color bands, are two arcs of black, which represent no detectable moisture. Progressing outward, the colored arcs are green, yellow, red, magenta, a ring that alternates between red and magenta, and, finally, an outer band of yellow. The antenna scans during test but the transmitter is inhibited.

NOTE

The test mode is not an end-to-end test of all weather radar functions, its primary function is to ensure that the EHSI can display the necessary colors for proper operation.

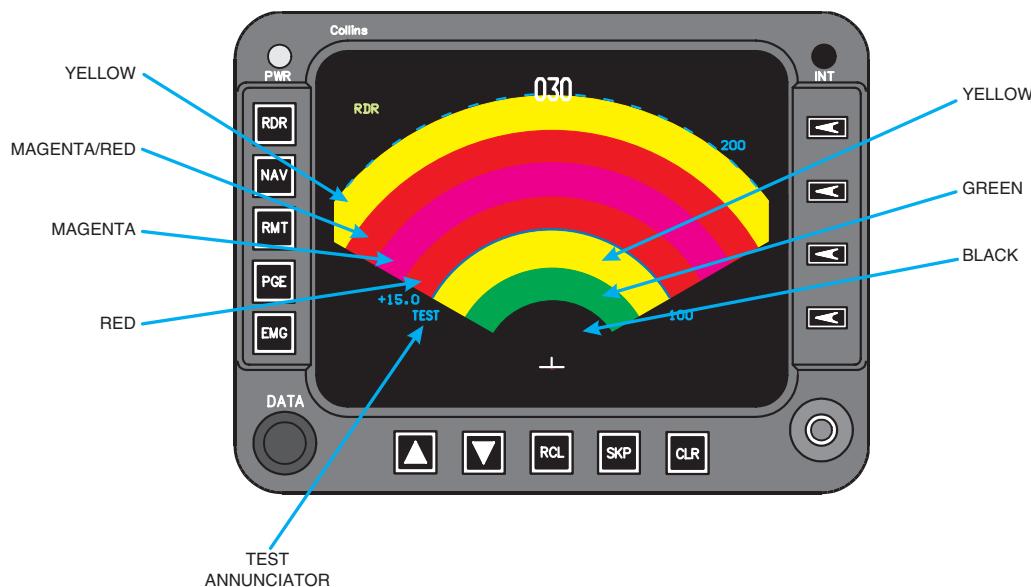


Figure 16-90 MFD Weather Radar Test Display

TGT (Target Alert)

TGT mode permits weather targets to be detected and annunciated without the targets themselves appearing on the EHSI display. This mode is normally used when the radar displays are not selected on the EHSI. When TGT mode is selected, the radar transmitter is energized, the antenna scans, TGT (mode) is annunciated and a box with "TGT" inscribed appears nonflashing to the right of the lubber line. The boxed TGT annunciator will flash when severe precipitation is detected within the target alert area, regardless of the range selected. This flashing annunciation indicates to the pilot that a severe precipitation rate has been detected.

The TWR-850 also has the ability to detect severe turbulence in the target mode. In this case the annunciator will be a flashing TRB in place of the TGT annunciator.

MAP

Selecting MAP mode, places the system in the terrain-mapping mode. Map mode automatically disables the PAC alert and GCS features. In MAP

mode, the ground targets appear in cyan, green, yellow, or magenta (least reflective to most reflective). The half range arc is green and a MAP annunciator will be displayed.

WX

Selecting WX mode allows the weather radar system to operate in the weather detection mode. Detectable weather is color-coded; detectable weather above the minimum display threshold appears as one of four colors; green, yellow, red, or magenta (least reflective to most reflective). The WX annunciator will be displayed.

With WX mode selected, the path attenuation correction (PAC) feature is automatically enabled. A PAC alert causes an arc to appear behind targets in certain conditions. When the arc appears, it means the cell in front of the arc is severe enough that it may possibly be shadowing (or attenuating) other cells beyond it. Therefore PAC is a warning that an unknown weather situation may exist beyond the targets, as seen on the display.



WX+T (Weather Plus Turbulence) (TWR-850)

WX+T mode is the same as WX mode except that, along with weather detection, precipitation-related turbulence is detected and displayed in the 5, 10, 25, and 50 nmi range positions only. In WX+T mode and with the range selected to one of the turbulence-related positions (5, 10, 25, or 50), the weather radar indicator will display detected turbulence along with the weather detection. Detected turbulence will be displayed in magenta and will consist of precipitation areas where horizontal wind velocity shifts in excess of 16.40 feet per second (5 metres per second) are detected. When selected WX+T is annunciated on the EHSI and MFD.

TURB (Turbulence Only) (TWR-850)

Selecting TURB mode removes all weather radar targets from the display except for areas of detectable precipitation-related turbulence that appear in magenta. The TURB position of the mode switch is spring-loaded and must be held in the TURB position for display. Releasing the MODE switch on the WXP returns it to the WX+T position and restores the full weather radar display.

When selected TURB is annunciated on the EHSI and MFD.

GAIN Control

The GAIN control is a seven-position switch for manual gain control of the radar receiver. When placed in the CAL position, the gain is preset to a value that allows the radar receiver to calibrate its operation. Each of the minus settings (-1, -2, -3) reduces receiver sensitivity below the CAL level by 6 dB for a total reduction of 18 dB when in the -3 position.

The selected GAIN is annunciated (G-1, G+3, etc.) on the EHSI and MFD.

TILT Control

The radar antenna is fully stabilized to compensate for aircraft pitch and roll attitude changes. The TILT control allows vertical aim adjustment

of the radar antenna from approximately 15° down to 15° up from horizontal. When stabilization (STB) is on, the combined TILT and stabilization control range is from 30° down to 30° up. The tilt angle is annunciated on the EHSI and MFD.

RANGE Control

The RANGE control is a rotary switch that is used to select the maximum display range of the WXR-840 system. A solid cyan half-range arc and a half-range annunciator are displayed on the EHSI and MFD radar display. The compass arc represents the full range arc.

GCS (Ground Clutter Suppression) Button

The ground clutter suppression GCS reduces the intensity of ground returns so that most disappear from the display, allowing better definition of precipitation targets. GCS will only operate in the WX or WX+T modes.

When selected, GCS is annunciated on the EHSI and MFD.

STB (Stabilization) Pushbutton

The stabilization (STB) pushbutton enables or disables the radar antenna pitch and roll stabilization circuits. This feature is disabled only when a failure in the horizontal and/or vertical reference system causes the radar system to become unusable. STB is normally enabled. If stabilization is not selected, USTB is annunciated on the MFD. If stabilization is selected, there is no stabilization-related annunciation. If there is an attitude fault, a flashing USTB is annunciated.

HLD (Hold) Button

Pressing the HLD button (located concentric with the GAIN control) interrupts updating of the weather display on the EHSI.

When in hold mode the selected mode annunciator alternates between HOLD and the selected mode (HOLD-WX-HOLD-WX, etc.).



AUTO (Autotilt) Switch

Autotilt is selected by pulling outward on the AUTO switch (located concentric with the TILT control). Autotilt is designed to reduce pilot workload by reducing the need to readjusted tilt manually for different altitudes or range selections. When autotilt is on and a different range is selected or the aircraft changes altitude, the RTA-842 automatically tilts the antenna to maintain the same relationship between beam deflection and display.

To determine the aircraft's AGL altitude, the radar beam dips into the ground at the end of every third antenna scan. The resulting return is used to calculate the aircraft AGL. This action does not affect the weather radar display.

Autotilt is annunciated with an "A" preceding the tilt angle (e.g., A+10.7, A-5.2, etc.) on the EHSI.

SEC (Sector) Button

The sector scan switch is located concentric with the RANGE control. Sector scan is selected by pressing in on the SEC switch. When SEC is selected, the antenna azimuth scan angle is reduced from 60° on either side of the aircraft's nose to 30°. With the antenna scan angle reduced by 50 percent, the weather radar display update rate is doubled.

SEC mode is annunciated by decreasing the range mark to the new reduced sector scan area.

Fault Monitor

The fault monitoring function built into the WXR-840 system has priority over any selected mode. The word FAULT is annunciated and alternates between FAULT and the selected mode annunciation if a fault is detected anywhere in the system. If the fault is in the form of an interrupted communication link between the RTA-842 and the WXP-840B, then the word FAULT appears nonflashing. Selecting STBY mode and then returning the mode control to any other operational selection can clear temporary fault conditions. If this method does not clear the fault, the radar system requires repair.

In Flight Stabilization Alignment

The radar system stabilization alignment can be adjusted in flight to introduce offsets into the radar system to compensate for mounting errors etc.

There is a recessed switch on the WXP-840B control panel to enter the alignment mode. When in the alignment mode, TRIM is annunciated on the EHSI radar display.

Procedures for the alignment can be found in the appropriate installation manual.

COCKPIT VOICE RECORDER (CVR)

GENERAL

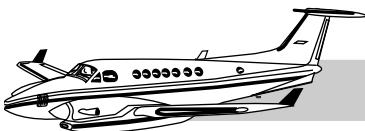
The airplane can be equipped with either a L3 Communications Fairchild Model A100S, FA2100 or a Universal Avionics CVR-30B cockpit voice recorder. The A100S CVR and the CVR-30B have a 30-minute recording capacity, while the FA2100 CVRs have either a 30-minute or a 2-hour recording capacity.

The CVR system consists of a cockpit voice recorder, a control unit, an area microphone, and an impact switch. The recorder and impact switch are located beneath the floorboards and forward of the aft cargo area. The control unit is located in the cockpit pedestal. The area microphone is located in the leading edge of the glareshield to the right of the fire extinguisher switch. Electrical power to the recorder is provided by the triple fed bus.

Input to the CVR comes from the following four sources:

Passenger Address Channel 1

Transmissions from the pilot or copilot over the passenger address system, along with the cabin chimes, will be recorded.



Copilot's Audio Channel 2

Transmissions heard by the copilot through the headset or speaker, or transmits over any of the microphones, will be recorded.

Pilot's Audio Channel 3

Transmissions heard by the pilot through the headset or speaker, or transmits over any of the microphones, will be recorded.

Area Mic Channel 4

Voices and other sounds in the cockpit will be recorded (stronger sounds may mask weaker ones).

The CVR is capable of reproducing the recorded audio data in one of the following formats:

A100S, CVR-30B and FA2100 30-Minute Recorders

Reproduces the last 30 minutes of audio recorded by the above sources in a high quality format on four separate channels.

FA2100 2-Hour Recorder

In addition to the 30 minutes of high quality audio noted for the 30-minute recorders, reproduces the last two hours of audio recorded by the above sources in a standard quality format on two channels, which contain the following material:

- A combination of the pilot, copilot, and passenger address audio data.
- Cockpit area microphone audio data.

COCKPIT VOICE RECORDER

A100S

The Model A100S accepts four channels of cockpit audio (pilot, copilot, PA and area microphones) and converts the audio signal to a

digital format for storage in a crash protected solid-state memory. The Model A100S consists of a chassis and front panel, three printed wiring assemblies (PWAs), and the flash crash survivable store unit (FCSSU), which contains the solid-state flash memory used as the recording medium. An underwater locator beacon (ULB), also referred to as an underwater locator device, is mounted on the front of the recorder. A headset jack (labeled Monitor) is provided on the front of the unit to monitor the combined audio for testing purposes and provides the same audio as the headset jack on the CVR control unit. An audio monitor adaptor test connector is provided under the outer cover of the CVR for readout of the recording with the use of special test adaptors.

FA2100 CVR

The FA2100 CVR simultaneously records four separate channels of cockpit audio, converts the audio to a digital format, and stores the data in a solid-state memory.

The FA2100 CVR consists of a chassis and front panel, three printed wiring assemblies aircraft interface PWA, audio compressor PWA, and acquisition processor PWA, and the crash survivable memory unit (CSMU). The CSMU contains the solid-state flash memory used as the recording medium. An underwater locator beacon (ULB), also referred to as an underwater locator device, is mounted horizontally on the front of the CSMU and is also used as the recorder's carrying handle. The ground support equipment (GSE) connector is located on the front of the FA2100 CVR. This connector provides the interface from the recorder to GSE for checkout of the recorder, or for readout of the recording with the use of special test adaptors.

CVR CONTROL UNIT (A100S AND FA2100 UNITS)

The control unit functions as the CVR cockpit user interface. The control unit contains operator controls/indications and a preamplifier printed wiring assembly (PWA) for interfacing with the cockpit area microphone and with the CVR. The control unit contains the following controls:



- Test Switch — Press and hold the test switch to carryout self test.
- Erase Switch — Press the erase switch to bulk erase the CVR recording in memory. CVR recording will only be erased when the aircraft is on the ground and after the switch has been held down for more than two seconds.
- Indicating Meter — During CVR test, the indicating meter will indicate the test tone level. The needle moving into the green section of the arc will indicate a satisfactory test.
- Headset Jack — The headset jack is provided to allow monitoring of the combined audio recording as it is being recorded. The audio input is processed and stored into memory and then read back from memory, processed and sent to the headset jack.

Previously recorded information cannot be played back from the CVR while it is installed in the aircraft.

Area Microphones (A100S and FA2100 Units)

The area microphone sends all the sounds in the flight compartment to the CVR control unit. The area microphone is a miniature electric condenser type microphone. The CVR control unit contains a preamplifier for the area microphone and provides 6 VDC bias current for microphone operation.

Impact Switch

The impact switch disconnects the power supply to the CVR. The impact switch operates if a force of 4g occurs on the aircraft in the fore-aft axis and left-right axis. If the impact switch is activated, 28 VDC power to the CVR is removed, used to latch a lockout relay and illuminate an indicator lamp on the impact switch. The impact switch will remain in this condition until the reset

button on the impact switch is pressed with power applied to the unit.

Self Test

Pressing the test switch should result in the following indications if the recorder is functioning properly:

A100S (30-Minute Recorder)

A 640Hz tone will be heard in the headset (if plugged into the control unit) and the needle will move into the green arc on the meter. The test tone will be sequentially switched through the four channels of the recorder in .5 second intervals. The tone will remain constant as long as the switch is pressed and the needle will remain in the green arc with a slight pulsing action as channels are switched. In the event of a channel failure, the meter will dropout of the green section of the meter scale.

To test the area microphone operation, speak with a normal voice six inches away from the area microphone while monitoring the audio on a headset plugged into the test jack. After a distinct delay, the spoken words should be heard without any significant distortion.

The erase switch may be used to erase the entire recording, and will only work when the landing gear is down and the weight of the airplane is on the landing gear. To prevent accidental erasures, a time delay circuit makes it necessary to hold the erase switch down for two seconds to start the erasure process. If a headset is plugged into the control unit, a proper erasure is indicated by a 400-Hz tone in the headset. The tone will last for approximately 14 seconds.

FA2100 Series (30-Minute and 2-Hour Recorder)

A one- to two-second tone will be heard in the headset (if plugged into the control unit) when the test switch is pressed. The tone will then stop and the needle will move into the green arc on the meter and stay there as long as the switch is pressed.



To test the area microphone operation, speak with a normal voice six inches away from the area microphone while monitoring the audio on a headset plugged into the test jack. The spoken words should be heard without any delay or significant distortion.

The erase switch may be used to erase the entire recording, and will only work when the landing gear is down and the weight of the airplane is on the landing gear. To prevent accidental erasures, a time delay circuit makes it necessary to hold the erase switch down for two seconds to start the erasure process. If a headset is plugged into the control unit, a proper erasure is indicated by a 400-Hz tone in the headset that occurs when the erase switch is released. The tone will last for approximately 5 to 10 seconds.

CVR-30B Recorder

To test the system, put on the headset and plug into the control unit headphone jack.

Push the TEST button on the control unit. Hold for one second and release. This starts the CVR self test.

During self test, ensure the PASS and FAIL lights illuminate alternately, indicating that self test is in progress. Self Test takes approximately 15 seconds to perform. After 10 seconds of self test, cockpit conversation should be heard through the headset and visible on the SIGNAL LEVEL indicator. The test tone audio will be heard. No audio will be recorded during self test.

Upon completion of a successful self test, the PASS light will illuminate for approximately 10 seconds then go out. A failed self test will be indicated by an illuminated FAIL light (usually flashing). The flashing FAIL light indicates the nature of the failure. FAIL will flash a number of times, pause, and then repeat that same number of flashes. Refer to the maintenance manual for fail condition based on the number of FAIL flashes.

ARTEX 110-4 EMERGENCY LOCATOR TRANSMITTER (ELT)

GENERAL

The Artex 110-4 emergency locator transmitter (ELT) System will automatically activate during a crash and transmit a sweeping tone on 121.5 and 243.0 MHz to aid in the location of the crashed aircraft. The 110-4 ELT is designed to meet the requirements of TSO C91a. The system consists of the ELT transmitter, located in the aft fuselage area, an antenna mounted on the aft fuselage, and a remote switch with a yellow transmit light, located on the left cockpit sidewall next to the OAT gage.

ELT

The ELT 110-4 unit is a microprocessor controlled internal battery powered transmitter that can be activated by an internal G switch or by a switch on the ELT unit or a remote switch input.

The microprocessor controls all the ELT functions (ON/OFF, modulation, and RESET). The transmitter operates simultaneously on 121.5 and 243.0 MHz. The carrier is amplitude modulated with an audio frequency sweeping downward within the range of 1,600 to 300 Hz. The sweep repetition rate is between 2 and 4 Hz. The power output is between 100mW and 50mW over the 50-hour operating period.

The G-Switch or crash force activation sensor is designed to activate with a change of velocity of $3.5 \text{ fps} \pm 0.5 \text{ fps}$ in the fore aft axis.

The ON/OFF switch is located on the connector end of the ELT, it can be used to activate the ELT transmitter and reset the ELT to arm after activation. This switch cannot be positioned to prevent the automatic activation of the ELT transmitter. Also an ON light is located on the connector end of the ELT that indicates when the ELT transmitter is active.



The ELT battery pack consists of six 1.5 VDC "D" size Alkaline-Manganese Dioxide cells connected in series. Fresh battery pack voltage is 9.5 volts with a life of 26 months.

The ELT is housed in a high-impact fire-resistant Lexan plastic housing and installed in a mounting tray with an end cap to hold the two tray halves together.

ELT Remote Switch Panel

The remote switch panel consists of a lever lock switch labeled ARM/ON a yellow transmit light and an edge lit panel. The switch is lever locked in the ARM and the ON positions. This switch cannot be positioned to prevent the automatic activation of the ELT transmitter. The transmit light is wired to the aircraft hot battery bus, and the edge lit panel is controlled by the side panel rheostat located on the overhead panel.

Operation

The ELT transmitter can be activated by the G switch, the ON/OFF switch on the ELT, or by the remote cockpit switch.

When the ELT is removed from the aircraft, the G switch is disabled to prevent accidental activation of the ELT that could be caused by dropping, rough handling, or during shipping.

The ON/OFF switch on the ELT can be used to turn the ELT transmitter on by moving it to the ON position. If accidental activation occurs, reset the ELT by moving the switch to the ON position and then back to OFF. This switch can also be used to activate the ELT transmitter when it is removed from the aircraft and is in the OFF position for normal operation.

The remote ARM/ON switch can be used to turn the ELT transmitter on by moving it to the ON position and reset the ELT after accidental activation by moving it to the ON position and then back to ARM. Normal switch position is the ARM position.

Neither the ON/OFF on the ELT, or the remote ARM/ON cockpit switch, can be positioned to

prevent the G-switch activation of the ELT transmitter when installed in the aircraft.

When the ELT transmitter is active, the transmit light on the remote panel flashes to indicate ELT operation. Also the transmit light on the ELT will flash when the ELT is transmitting.

Installed ELT Transmitter Test

Perform the transmitter tests by activating the ELT and listening on 121.5 MHz.

1. Ensure that the main aircraft battery is installed and connected to the hot battery bus.
2. Tune a VHF receiver to 121.5 MHz.
3. Turn the ELT remote cockpit switch to "ON" position, wait for three sweeps on the receiver, which takes about one second, and then turn the switch back to "ARM."
4. To pass the test, you must hear the three sweeps and see the front panel light immediately begin to flash continuously.

During the first seconds of operation, the microprocessor in the ELT checks the "G-Switch" (automatic activation switch) latching circuit, pins 5 and 8 on the tray connector. If there is a problem, the processor will not immediately turn on the cockpit light during those first few sweeps, which indicates the unit is not working properly. For example, if the jumper between pins 5 and 8 was not installed or was open, the cockpit light would flash momentarily upon ELT activation then stay off for approximately three seconds before it begins to flash continuously.



ARTEX 110-406 EMERGENCY LOCATOR TRANSMITTER (ELT)

GENERAL

The ARTEX 110-406 emergency locator transmitter (ELT) will automatically activate during a crash and transmit a sweeping tone on 121.5, 243.0 MHz and a digital message on 406.025 MHz to aid in the location of the crashed aircraft. The ELT is a "Third generation ELT" designed to meet the requirements of TSO C126. The system consists of the ELT transmitter, located in the aft fuselage area, an antenna mounted on the aft fuselage, a buzzer located next to the ELT and a remote switch with a yellow transmit light, located on the left cockpit sidewall next to the OAT gage.

ELT

The ELT 110-4 unit is a microprocessor controlled internal battery-powered transmitter that can be activated by an internal G switch or by a switch on the ELT unit or a remote switch input.

The microprocessor controls all the ELT functions (ON/OFF, modulation, and RESET). The transmitter is divided into two modules 121.5/243.0 MHz module and the 406.025 MHz module. The 121.5/243.0 MHz module of the transmitter is nearly identical to that of the ELT 110-4 unit. An additional module is installed to give the ELT its 406.025 MHz transmission capabilities, with the top circuit board containing the microprocessor that controls the 406.025 MHz module.

The 121.5/243.0 MHz module operates simultaneously on 121.5 and 243.0 MHz. The carrier is amplitude modulated with an audio frequency sweeping downward within the range of 1,600 to 300 Hz. The sweep repetition rate is between 2 and 4 Hz. The power output is between 200mW and 100mW.

The 406.025 MHz module turns on and transmits every 50 seconds for 440 milliseconds (short message protocol). During that time an encoded digital message is sent to a satellite network that is in place to receive these messages. The information in the message sent to the satellite is programmed at the factory and contains the following information:

- Serial Number of the Transmitter
- Country Code Number (3-digit code Maritime Identification Number)
- I.D. Code (unique to each ELT)

The 406.025 MHz transmitter will operate for 24 hours and then shuts down automatically. The 121.5/243.0 MHz transmitter will continue to operate until the unit has exhausted the battery power, which typically will be at least 72 hours.

One of the advantages of the 406.025 MHz transmitter is that it will produce a much more accurate position, typically 1 to 2 kilometers as compared to 15 to 20 kilometers for 121.5/243.0 MHz transmitters. Once the ELT is activated and the 406.025 MHz signal is detected from the satellite and a position is calculated, the 121.5/243.0 MHz transmissions are used to home in on the crash site.

The decoded digital message information allows the search and rescue authorities to contact the owner/operator of the aircraft through information stored in a database. Information contained in the database that may be useful in the event of a crash is:

- Type of Aircraft
- Address of Owner
- Telephone Number of Owner
- Aircraft Registration Number
- Alternate Emergency Contact



Because aircraft communication radios are not capable of receiving 406.025 MHz transmissions, the only method of monitoring the ELT is the blinking cockpit light, the buzzer or the 121.5/243.0 MHz transmissions, which may be monitored on the aircraft communication transceiver. The buzzer is powered by the ELT battery, sounds at predetermined intervals and runs for shorter periods towards the end of the battery life. It is used to locate and operating ELT by the sound coming from the tail section of the aircraft.

The ELT battery pack consists of four “D” size Lithium Manganese Dioxide cells connected in series. To prevent the cells from being charged, diodes are connected across each cell and fuses are connected to the output. Fresh battery pack voltage is 12 volts with a life of five years.

The G switch or crash force activation sensor is designed to activate with a change of velocity of $3.5 \text{ fps} \pm 0.5 \text{ fps}$ in the fore aft axis.

The ON/OFF switch is located on the connector end of the ELT and can be used to activate the ELT transmitter and reset the ELT to arm after activation. This switch cannot be positioned to prevent the automatic activation of the ELT transmitter. Also an ON light is located on the connector end of the ELT that indicates when the ELT transmitter is active.

The ELT is housed in a high impact fire resistant polycarbonate plastic housing and installed in a mounting tray with an end cap to hold the two tray halves together.

ELT Remote Switch Panel

The remote switch panel consists of a lever lock switch labeled ARM/ON, a yellow transmit light, and an edge lit panel. The switch is lever locked in the ARM and the ON positions. This switch cannot be positioned to prevent the automatic activation of the ELT transmitter. The transmit light is wired to the aircraft hot battery bus, and the edge-lit panel is controlled by the side panel rheostat located on the overhead panel.

Operation

The ELT transmitter can be activated by the G switch, the ON/OFF switch on the ELT, or by the remote cockpit switch.

When the ELT is removed from the aircraft the G switch is disabled to prevent accidental activation of the ELT that could be caused by dropping, rough handling, or during shipping.

The ON/OFF switch on the ELT can be used to turn the ELT transmitter on by moving it to the ON position. After accidental activation, reset the ELT by moving it to the ON position and then back to OFF. This switch can also be used to activate the ELT transmitter when it is removed from the aircraft and is in the OFF position for normal operation.

The remote ARM/ON switch can be used to turn the ELT transmitter on by moving it to the ON position. After accidental activation, reset the ELT by moving it to the ON position and then back to ARM. Normal switch position is the ARM position.

Neither the OFF/ON on the ELT, or the remote ARM/ON cockpit switch, can be positioned to prevent the G switch activation of the ELT transmitter when installed in the aircraft.

When the ELT transmitter is active, the transmit light on the remote panel flashes to indicate ELT operation. Also the transmit light on the ELT will flash and the buzzer sounds when the ELT is transmitting.

Installed ELT Transmitter Test

Perform the transmitter tests by activating the ELT and listening on 121.5 MHz.

1. Ensure that the main aircraft battery is installed and connected to the hot battery bus.
2. Tune a VHF receiver to 121.5 MHz.
3. Turn the ELT remote cockpit switch to “ON” position, wait for three sweeps on the receiver, which takes about one second, and then turn the switch back to “ARM.”



4. To pass the test, you must hear the three sweeps and see the front panel light immediately begin to flash continuously.

During the ON to OFF transition, the microprocessor in the ELT checks the “G switch” (automatic activation switch) latching circuit, pins five and eight on the 12-pin connector at the ELT; the 406.025 MHz transmitter for proper RF output; that the OFF transition turned the 406.025 MHz transmitter off; and a battery check. If the ELT is working properly, the sequence following entry to the “ARMED” (OFF) condition will result in the panel LED staying illuminated for approximately one second, then extinguishing.

If a problem is detected, the LED provides a coded signal following the initial one-second pulse. The coded signal and related problem are as follows (in order of importance):

- a. One flash indicates a G-switch loop open failure.
- b. Three flashes indicates a 406.025 MHz transmitter problem (i.e., a bad or unconnected coax cable, an antenna problem, the battery not connected to the 406.025 MHz module, a programming error, or a problem with the ELT itself).
- c. Five flashes indicate there is no navigation data present. This is normal for the King Air installation as there is no ELT/NAV interface unit installed.
- d. Seven flashes indicate a battery problem (i.e., battery usage time over an hour).

There is a priority assigned to the problem reporting which is the same order as listed above. That is, if the G-switch circuit has a failure, there will be a single flash regardless of whether there is a 406.025 MHz transmitter problem. Once the G-switch problem is resolved, then three flashes would appear.

PITOT STATIC SYSTEM AND INSTRUMENTS

PITOT AND STATIC PRESSURE SYSTEMS (WITH APS-65 AUTOPILOT AND ADC-85 AIR DATA COMPUTER)

The pitot and static pressure systems provide a source of impact pressure and static air for operation of the pitot static instruments and associated avionics units.

The pitot portion of the system is composed of two mast-mounted pitot tubes, one on each side of the lower nose, electrical wiring for each pitot heater and tubing for instrument connection. The impact pressure entering the pitot tubes is transmitted to the dual airspeed indicators mounted on the instrument panel through separate tubing routed along each upper side of the nose compartment. The pitot pressure as it passes through the nose avionics bay is also used to provide impact pressure to the air data sensor ADS-65 from both the pilot's and copilot's pitot system and from the pilot's system only provides pitot pressure to the air data computer ADC-85. Since the pitot tube is the lowest point in pitot plumbing, the resultant natural drainage eliminates the need for drain valves. Two circuit breaker switches on the left inboard subpanel control the pitot tube heating elements.

The static portion of the system includes two unheated static ports on each side of the fuselage just forward of F.S. 402.75. Lines cross-connect the static ports for balance, to the instruments in the crew compartment and the associated avionics in the nose avionics bay. The static lines which form a pilot's and separate copilot's static system are routed from the static ports to the top center of the fuselage, then over to the right side of the fuselage. They are then routed forward in the fuselage sidewall beneath the windows, up under the instrument panel to the pilot's and



copilot's vertical speed indicator and airspeed indicator on the instrument panel. The copilot's altimeter is also connected to the copilot's static system. The pilot's altimeter is not connected to the static system as it is driven by electrical signals from the ADC-85. The static system lines are fed into the nose avionics bay to provide static pressure to the ADS-65 from both pilot's and copilot's systems, the ADC-85 receiving static pressure from the pilot's system only.

Static line drain valves are located behind an access trim panel located in the lower right hand crew compartment sidewall adjacent to the instrument panel (one for the copilot's system and two for the pilot's system before and after the alternate static selector valve). Additional drain valves can be found in the nose avionics bay (one for the pilot's static system on all aircraft and one for the copilot's static system on B300 aircraft only). The static lines should be drained anytime the airplane has been exposed to rain.

Should abnormal or erratic instrument readings indicate that the normal static source is restricted, an alternate air source may be utilized for operation of the pilot's static system only. The alternate system supplies static air from the interior of the aft fuselage. The alternate static line is routed through the aft pressure bulkhead forward along the right side of the fuselage to the pilot's static air selector valve. This selector valve is located below the right circuit breaker panel adjacent to the instrument panel. The pilot's static air selector valve is a mechanical toggle valve held in the normal position by a clip; the alternate air source is selected by raising the clip and moving the toggle from NORMAL to ALTERNATE. The pilot's instruments then function on the alternate air source.

The alternate static air is permanently connected to the following instruments that also require static pressure: the pneumatic pressure indicator, the cabin differential pressure indicator, and for the B300 (350) aircraft only the cabin differential pressure switch.

PITOT STATIC INSTRUMENTS

Airspeed Indicators

The pilot's and copilot's airspeed indicator shows the airspeed of the airplane continuously in knots per hour. The instrument has two pointers, one showing the airplane's indicated airspeed (IAS) and the other showing the maximum allowable airspeed (VMO). The range of indication is 40 to 300 knots. For the B300 aircraft movable airspeed reference markers ("bugs") are installed on the bezel allowing reference airspeeds to be set.

The airspeed indicator is rear-mounted with screws and receives impact air pressure and static air pressure for its operation. For the B300 aircraft, the indicator requires lighting power from the light dimmer circuits for the face light and in the case of the pilot's airspeed indicator 28 VDC for the overspeed warning circuit.

The B300 (350) aircraft pilot's airspeed indicator has an overspeed warning switch that sounds a warning tone through the audio system when the pitot air pressure exceeds the maximum speed for a given altitude and the IAS pointer has overlapped the VMO pointer. When the overspeed warning trips the indicator outputs 28 VDC to the aural warning tone generator to produce the overspeed warning tone. The overspeed warning circuit can be checked by holding "up" on the test switch located on the copilot's inboard subpanel to trip the indicator internal overspeed circuit.

Instantaneous Vertical Speed Indicator

The instantaneous vertical speed indicator (IVSI) indicates instantaneously the vertical velocity up or down of the aircraft. The indicator requires static air pressure for its operation and uses internal vertical acceleration pumps to provide the instantaneous response. For the B300 aircraft the indicator requires lighting power from the light dimmer circuits for the face light. The pointer can be manually adjusted to zero with a screwdriver. The indicator is rear-mounted with screws to the instrument panel.

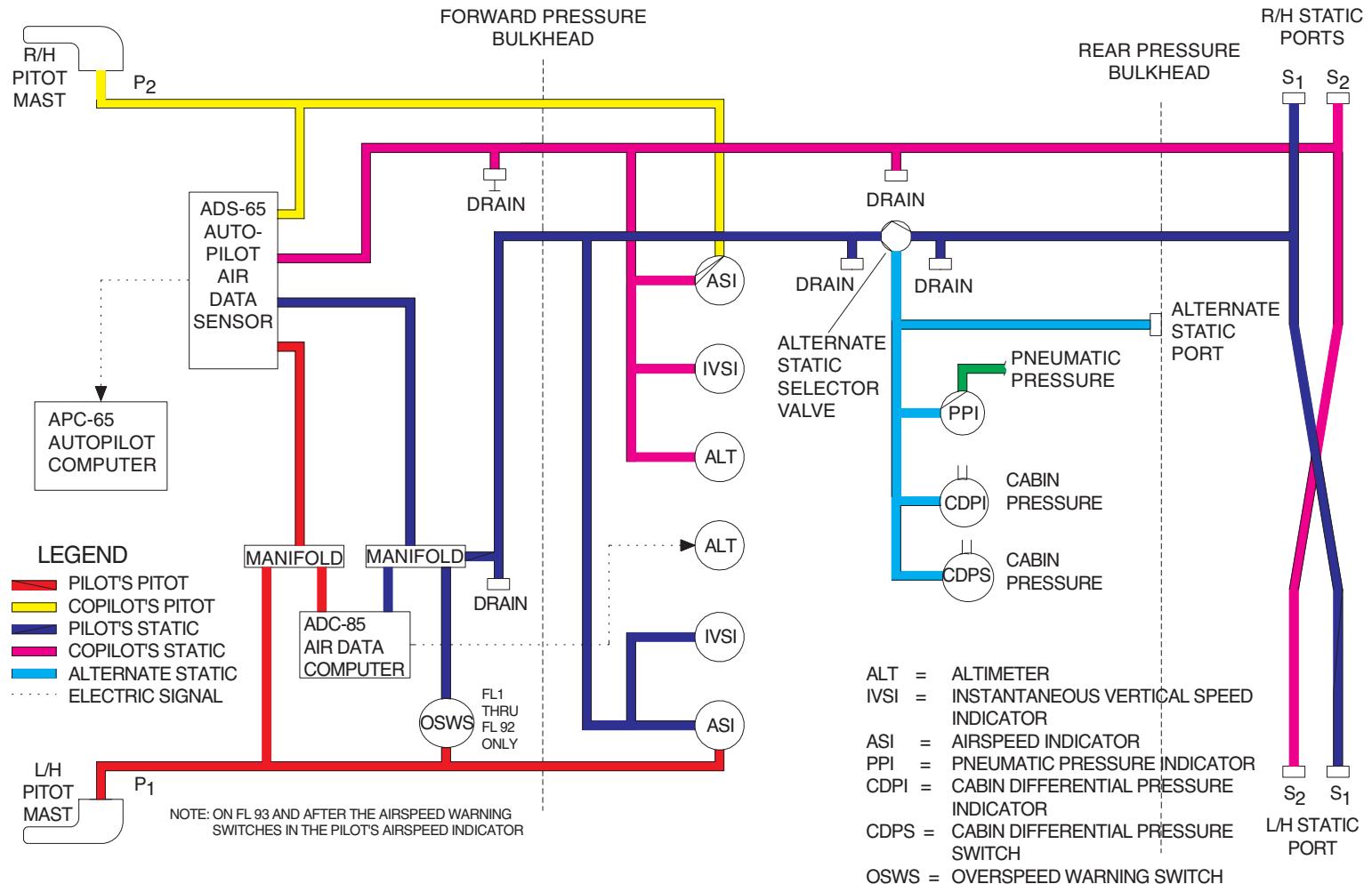


Figure 16-91 B300 Pitot Static System With APS-65 and ADC-85



Copilot's Encoding Altimeter (OPTIONAL)

An encoding altimeter is installed on the copilot's instrument panel. The altimeter is a standard mechanical direct reading instrument with a built in vibrator, which provides the copilot with a

baro-corrected altitude reading and encoded altitude signals to a transponder. The indicator requires static pressure for its operation, 28 VDC power for the encoder operation and vibrator operation. The indicator incorporates internal lighting, which receives power from the light dimmer circuits.

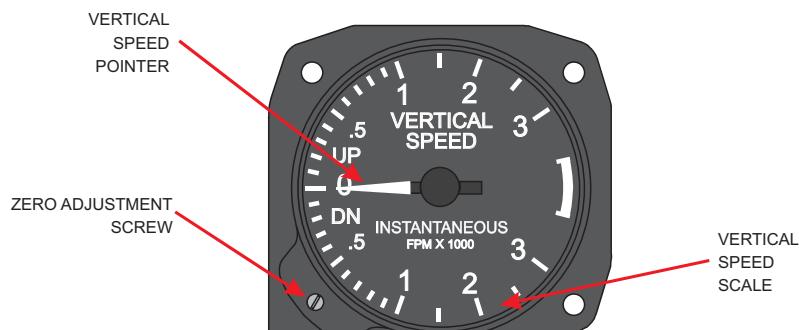
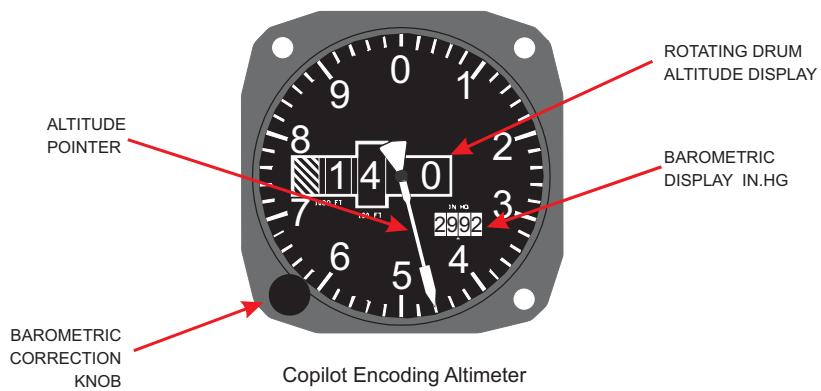
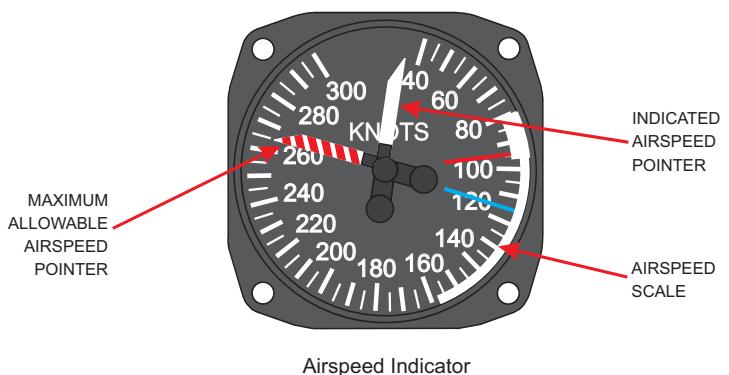


Figure 16-92 Pitot Static Flight Controls



Slaved Compass Systems

The most common compass system for the Super King Air is the Collins MCS-65; however, the Sperry C-14A-43 or the Collins MCS103 systems could be installed. As far as the pilot is concerned, all of these systems operate in a similar manner. They will be treated as one common system in the following discussion.

Occasionally, a Collins PN-101 system will be installed on the copilot's side. This system operates in a slightly different manner and will be discussed separately in this section.

MCS-65, MCS-103, and C-14A-43 Systems

From an operational standpoint, all three of these systems may be treated identically. All of these systems require 400 Hz electrical power from an inverter. In the unlikely event that both inverters fail, these systems would be inoperative. Each of them has the following components.

- Flux sensor (also called a flux gate or flux valve)—The function of this device is to sense the earth's magnetic field relative to the airplane and convert that information into an electrical signal which represents the airplane's magnetic heading.
- Slaving amplifier—The magnetic heading signal from the flux sensor is too weak to be used directly; therefore, it is amplified (made larger or stronger) by the slaving amplifier. The output signal is now strong enough to directly drive a torquer motor in the directional gyro and thus maintain the gyro rotor in alignment with magnetic north.
- Directional gyro—Once the gyro rotor is aligned with magneto north, it will have a natural tendency to stay there for a short period of time, due to a force called gyroscopic rigidity in space. This force will continue to keep the gyro mechanism in relatively good alignment as long as the gyro rotor continues to turn at its design speed. When the gyro drifts out of alignment (precesses), the condition will be sensed, and the magnetic heading reference information from the slaving amplifier will again drive the gyro rotor back into alignment with magnetic north, using the torquer motor previously described.

scopic rigidity in space. This force will continue to keep the gyro mechanism in relatively good alignment as long as the gyro rotor continues to turn at its design speed. When the gyro drifts out of alignment (precesses), the condition will be sensed, and the magnetic heading reference information from the slaving amplifier will again drive the gyro rotor back into alignment with magnetic north, using the torquer motor previously described.

- Horizontal situation indicator (HSI)—The gyro heading information (which should be the same as magnetic heading) is sent to a compass card on the HSI to display the magnetic heading to the pilot (Figure 16-94). This heading information is then sent from the HSI to the compass card on the opposite Radio Magnetic Indicator (RMI). In this way, gyro-stabilized, magnetic heading information is displayed in front of each pilot from two independent sources: the pilot's and the copilot's compass systems.
- Slaving meter—The slaving meter compares the sensed magnetic heading at the flux sensor (system input) to the displaced magnetic heading at the HSI (system output). The difference, if any, is displayed on the slaving meter by displacement of the slaving needle from the center position (which indicates synchronization or zero error). It is normal for this needle to deviate occasionally due to precession, however, it should always come back to center. If it is displaced to one side for more than approximately one minute, the gyro may be precessing excessively and/or the slaving system may not be doing its job. In any case, the accuracy of the compass system should be checked by cross-referencing the heading information from the opposite system and/or the magnetic compass.

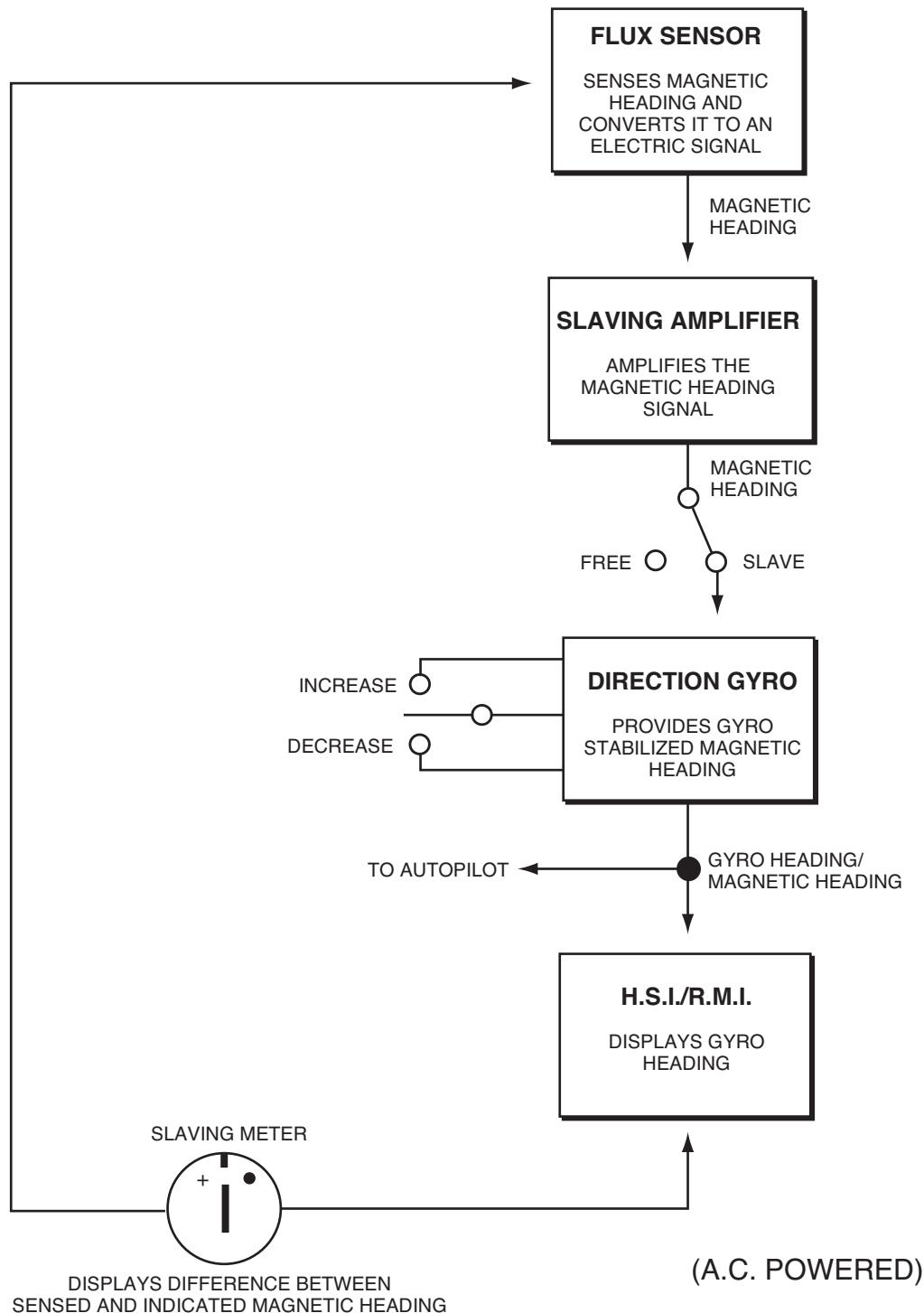


Figure 16-93 Compass System Connections



- SLAVE/FREE switch—This lever-locking switch is used to select either the slaved or the free mode of operation for the compass system. This switch should normally remain in the slaved mode of operation. In this mode, when power is initially applied to the system, it will automatically “slave” itself to the correct magnetic heading and remain there throughout the flight, correcting for precession as necessary. The free mode of operation is generally reserved for occasions when the slaved (automatic) mode of operation has failed and the pilot wishes to revert to a directional gyro mode of operation. This mode may also be used for flight in polar regions where extreme levels of magnetic variation exist. In this mode of operation, the flux sensor and the slaving amplifier are disconnected from the rest of the system. The result is that the pilot now has a directional gyro (which will precess and must be corrected manually using the increase/ decrease switch) which uses the HSI to display the heading information from the directional gyro.
- Increase/decrease switch—This is a toggle switch which is spring loaded to the center (OFF) position. The switch may be used when in the free mode of operation

to manually change the directional gyro to the left or right, thus increasing or decreasing the displayed heading information. When in the slaved mode of operation, momentarily holding this switch in either position causes the system to “reset” itself to the fast-slave mode of operation, thereby correcting any displayed error at a rapid rate. This could be helpful if for any reason the gyro had tumbled or precessed excessively.

Collins PN-101 System

This compass system (Figure 16-95) is occasionally installed on the copilot's side. It has the advantage of being directly powered by the 28-volt DC electrical system. If both inverters fail, the system would continue to operate. However, the PN-101 system does not have a manual back-up mode of operation (FREE) if the slaving system (flux sensor and/or slaving amplifier) fails. The PN-101 system does have a fast-slave switch which may be momentarily held in the UP position to initiate the fast-slaving sequence (see fast-slaving explanation under increase/decrease switch above). Except for the differences mentioned here, the basic operation of this system is virtually identical to that of the Collins MCS-65, MCS-103, and the Sperry C-14A-43 systems described previously.

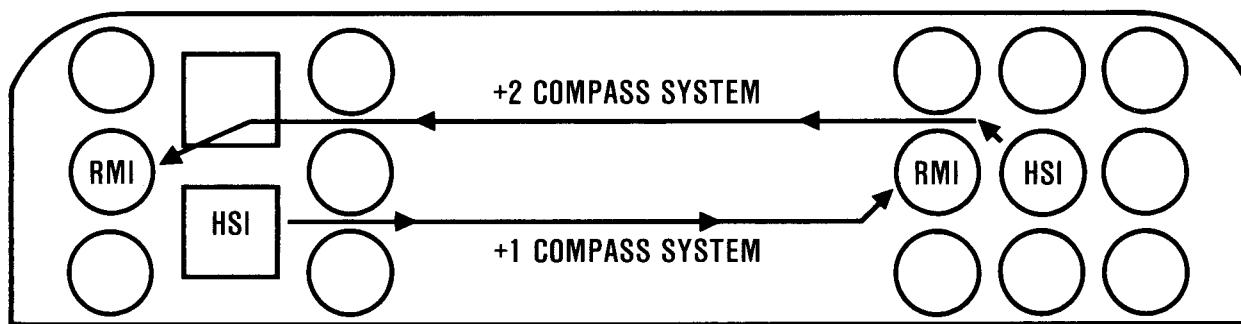


Figure 16-94 Compass System Connections



INSTALLED AVIONICS

KING AIR AVIONICS

The following is a listing of typical Avionics equipment found in the current production B300 (350) aircraft.

Pilot's Displays

3" Direct Reading Indicated Airspeed Indicator

3" Direct Reading Instantaneous Vertical Speed Indicator

Collins ALI-80A Counter Pointer Encoding Altimeter with PRE-80C Altitude Alerter and Preselector (Part of ADS-85 Air Data System)

2" Electric Turn and Slip Indicator

Collins RMI-30 Radio Magnetic Indicator

Collins IND-42A Distance Measuring Equipment Indicator. (Part of DME-42 DME system)

Collins CCU-65 Compass Compensator Unit (Part of MCS-65 Compass System)

Collins EFIS-85B (14) 3-tube Dual Processor System, which consists of the following items:

- DPU-85N Display Processor Unit (Located in the Nose)
- MPU-85N Multifunction Processor Unit (Located in the Nose)
- EADI-85 (EFD-85) Electronic Attitude Display Indicator

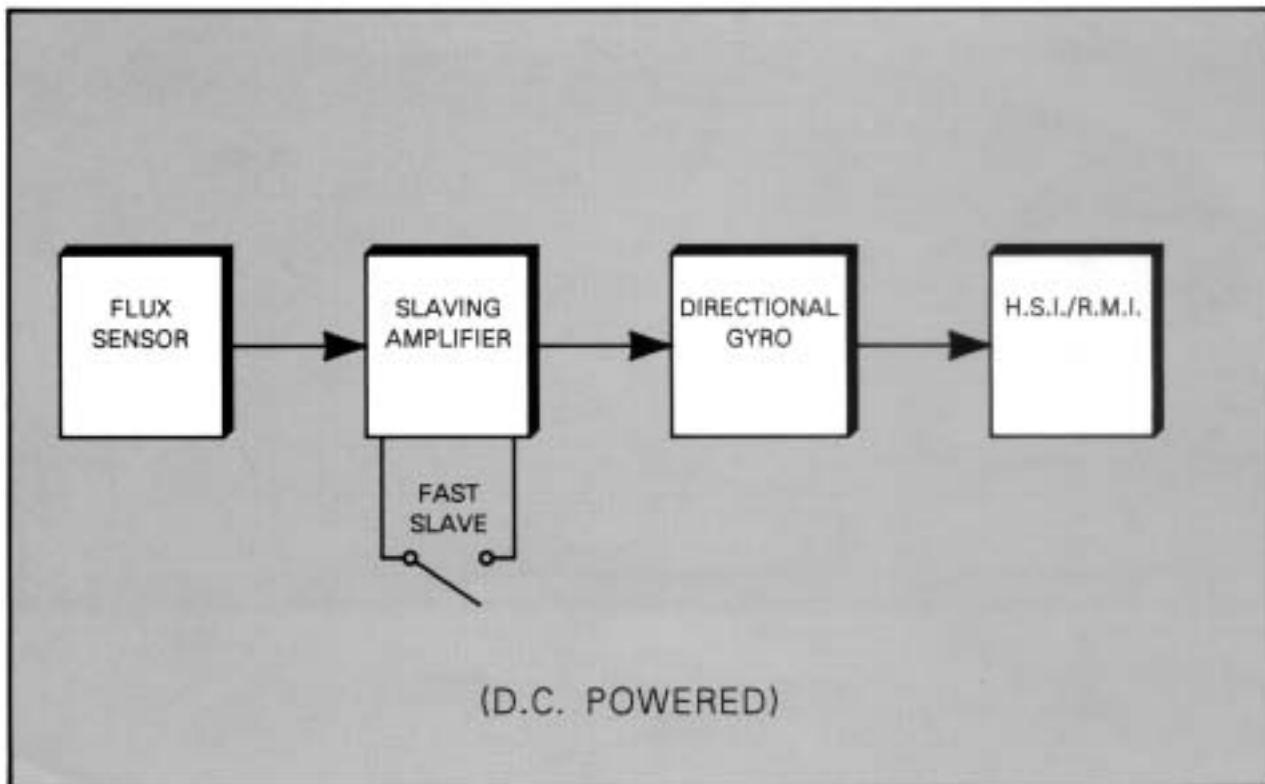
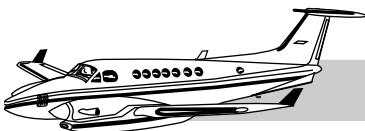


Figure 16-95 Compass System Connections



- EHSI-85 (EFD-85) Electronic Horizontal Situation Indicator
- MFD-85B Multifunction Display
- DCP-85G Display Control Pane (Located on Center Pedestal)
- CHP-85B Course Heading Panel (Located on Center Pedestal)

Copilot's Displays

3" Direct Reading Indicated Airspeed indicator

3" Direct Reading Instantaneous Vertical Speed Indicator

Aerosonic Direct Reading Counter Pointer Encoding Altimeter

3" Air Driven Horizon Indicator

2" Electric Turn and Slip indicator

Collins RMI-30 Radio Magnetic Indicator

Collins IND-42C Distance Measuring Equipment Indicator. (Part of ME-42 DME System)

Collins CCU-65 Compass Compensator Unit. (Part of MCS-65 Compass System)

Collins ALI-55 Radio Altimeter Indicator (Part of Radio Altitude System)

Collins EHSI-74A Electronic Horizontal Situation Indicator, which consists of the following items:

- HPU-74A HSI Processor Unit
- EFD-74 Electronic Flight Display (Located in the nose)
- HCP-74 HSI Control Unit (Located on Center Pedestal)

Other Avionics Systems

Autopilot/Flight director Collins APS-65 Autopilot system

Air Data System Collins ADS-85 Air Data System

Attitude Sensor Collins 332D-11T Vertical Gyro (Located in the nose)

Compass No. 1 Collins MCS-65 Compass System

Compass No. 2 Collins MCS-65 Compass System

Weather Radar

Weather radar system can be one of the following:

- Collins WXR-840 Color Weather Radar with a 12" antenna (Used with the 3-display EFIS-85 system, displayed on MFD and EHSI)
- Collins TWR-850 Turbulence Weather Detection Radar with a 12" antenna (Used with the 3-Display EFIS-85 system, displayed on MFD and EHSI)

Collins Pro Line II Radios

COMM 1 Collins VHF-22A Transceiver with CTL-22 Control and Antenna

COMM 2 Collins VHF-22A Transceiver with CTL-22 Control and Antenna

NAV 1 Collins VIR-32 VOR / LOC / GLS / MKR Receiver with CTL-32 Control, Connected To NAV and Glideslope Antenna

NAV 2 Collins VIR-32 VOR / LOC / GLS / MKR Receiver with CTL-32 Control, Connected To NAV and Glideslope Antenna

DME 1 Collins DME-42 With IND-42A Indicator And Antenna controlled by NAV 1 CTL-32

DME 2 Collins DME-42 With IND-42C Indicator And Antenna controlled by NAV 2 CTL-32



ADF Collins ADF-60A with CTL-62 Control and ANT-60A Antenna

Transponders Dual Collins TDR-94 with Single CTL-92 Control and Dual Antennas

Audio System

Audio Dual DB Systems Model 438 Audio Control Amp with Omni Range Filters, ADF Voice / Range Filter, Dual Audio Control Switches and Auto COMM Switches

Cabin Paging DB Systems Model 240 Passenger Address Amp with Four (Five B300 aircraft) Cabin Speakers

DB Systems Model 600 Aural Warning Tone Generator

CVR

The CVR can be one of the following systems:

- L3 Communications A100S Cockpit Voice Recorder
- L3 Communications FA2100 Cockpit Voice Recorder
- Universal Navigation Corporation CVR-30B Cockpit Voice Recorder

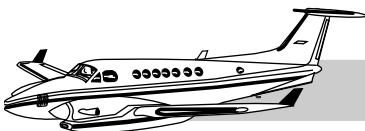
ELT

The ELT can be one of the following:

- Artex ELT-110-4 Two Frequency Emergency Locator Transmitter,
- Artex ELT-110-406 Three Frequency Emergency Locator Transmitter

Table 16-14 B300 (350) AVIONICS ACRONYMS GLOSSARY

1/2 Ø	Half Bank
ACT	Active
ADC	Air Data Computer
ADF	Automatic Direction Finder
ADM	Air Data Module (GPWS System)
ADS-65	Air Data Sensor (Autopilot System)
ADS-85	Air Data System
AH	Artificial Horizon
AI	Attitude Indicator
ALI	Altimeter (Collins)
ALT	Altimeter
ALT-50A	Radio Altimeter (Collins)
AP	Autopilot
APC	Autopilot Computer
APP	Autopilot Panel
APPR	Approach
APS	Autopilot System

**Table 16-14 B300 (350) AVIONICS ACRONYMS GLOSSARY**

ARINC	Aeronautical Radio Incorporated
ASI	Air Speed Indicator
ATT	Attitude
B/C	Back Course
CCU	Control and Compensation Unit (Compass System)
CHP	Course Heading Panel
CLM	Climb
COMM	Communications (VHF Radio)
CRS	Course
CSDB	Commercial Standard Data Bus
CTL	Control (Collins Radio Systems)
CVR	Cockpit Voice Recorder
CWS	Control Wheel Steering
DCP	Display Control Panel
DFDR	Digital Flight Data Recorder
DG	Directional Gyro
DGS	Directional Gyro System (Compass System)
DH	Decision Height
DIAG	Diagnostic
DIS	Disengage
DME	Distance Measuring Equipment
DPG	Differential Pressure Gage
DPS	Differential Pressure Switch
DPU	Display Processor Unit
DR	Dead Reckoning
DSC	Descend
DTU	Data Transfer Unit
DV	Drain Valve
EADI	Electronic Attitude Director Indicator
EFD	Electronic Flight Display
EHSI	Electronic Horizontal Situation Indicator



Table 16-14 B300 (350) AVIONICS ACRONYMS GLOSSARY (Cont)

ELT	Emergency Locator transmitter
ET	Elapsed Time
FCP	Flight Control panel
FCS	Flight Control System
FD	Flight Director
FDR	Flight Data Recorder
FDU	Flux Detector Unit (Compass System)
FMS	Flight Management System
FPDCU	Flat Panel Display Control Unit
G/S	Glideslope
GA	Go Around
GPS	Global Positioning System
GPWS	Ground Proximity Warning System
GS	Glideslope
GSP	Ground Speed
HCP	HSI Control Panel
HDG	Heading
HF	High Frequency (Radio)
HLD	Hold
HPU	HSI Processor Unit
IAS	Indicated Airspeed
INTPH	Interphone (Audio Between Pilots)
IVSI	Instantaneous Vertical Speed Indicator
KHF	King HF System
LCD	Liquid Crystal Display
LIN	Linear Deviation
LOC	Localizer
MB	Marker Beacon
MEM	Memory
MFD	Multifunction Display
MIC	Microphone

**Table 16-14 B300 (350) AVIONICS ACRONYMS GLOSSARY (Cont)**

MM	Middle Marker
MPU	Multifunction Processor Unit
MSG	Message (GPS)
NAC	Normal Accelerometer (Vertical) (Autopilot)
NAV	Navigation (Radio Receiver)
NCU	Navigation Computer Unit
NM	Nautical Mile
OAT	Outside Air Temperature
OM	Outer marker
P	Pitot Pressure
PA	Passenger Address
PH	Phone (Headphones, Headsets)
PPI	Pneumatic Pressure Indicator
PRE	Altitude Alerter Preselector (Collins)
PRE	Preset
RA	Radio Altitude
RAD ALT	Radio Altitude
RMI	Radio Magnetic Indicator
RNG	Range
RS 323	Recommended Standard (PC computer Data Bus)
RS 422	Recommended Standard (PC computer Data Bus)
RTA	Weather Radar Receiver/Transmitter/Antenna
S	Static Pressure
S1	Pilot's Static Pressure
S2	Copilot's Static Pressure
SMT	Servo Mount (Autopilot)
SR	Soft Ride
SSFDR	Solid State Flight Data Recorder
SSS	Skid Slip Sensor (Autopilot)
STB	Stabilization
STO	Store

**Table 16-14 B300 (350) AVIONICS ACRONYMS GLOSSARY (Cont)**

SVO	Servo (Autopilot)
TCAS	Traffic Alert Collision Avoidance System
TCAS I	Baseline TCAS System Provides Warnings Only
TCAS II	Provide Warnings and Collision Resolution Information
TDR	Transponder (Collins)
TGT	Target
TP	Test Point
TTG	Time To Go
TURB	Turbulence
TWR	Turbulence Weather Radar
TX	Transmit
ULB	Underwater Locating Beacon
UNS	Universal Navigation System
VG	Vertical Gyro
VIR	Navigation Receiver VOR, LOC, GS, MB (Collins)
VMO	Maximum Allowable Airspeed Indicator (Barber Pole)
VOR	VHF Omnidirectional Radio Range
VS	Vertical Speed
WXP	Weather Radar Control Panel
WXR	Weather radar
XDTA	Cross-side Data
XFR	Transfer
YD	Yaw Damper
YRS	Yaw Rate Sensor (Autopilot)



CHAPTER 17

MISCELLANEOUS SYSTEMS

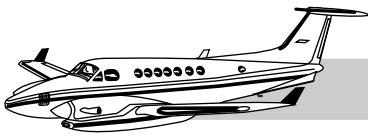
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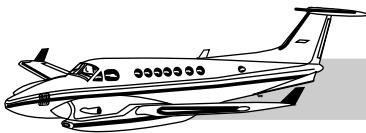
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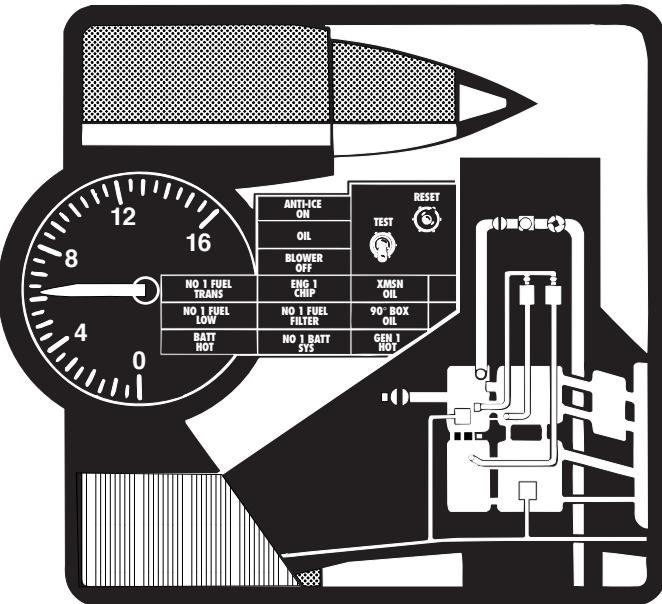
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CHAPTER 17

MISCELLANEOUS SYSTEMS



INTRODUCTION

Pilot and passenger comfort and safety are of prime importance in operating this airplane. The task is to teach flight crewmembers to use the oxygen system safely and effectively when required, and within the requirements of applicable FARs.

DESCRIPTION

This chapter presents a description and discussion of the Oxygen System, including general description, principle of operation, controls, and procedures. Use of the oxygen duration chart involves working simulated problems under various flight conditions.

FAR requirements for crew and passenger needs are part of the discussion, as well as the types and availability of oxygen masks. Servicing procedures referenced in the *Pilot's Operating Handbook* are also included.



OXYGEN SYSTEM

Current FARs require that any time an aircraft flies above 25,000 feet, oxygen must be immediately available to the crew and passengers. The Super King Air 300 and 350 oxygen system complies with this requirement.

The system consists of an oxygen bottle mounted in the aircraft tail section, crew masks which stow in compartments in the cockpit overhead, passenger oxygen masks which stow in compartments in the cabin ceiling, and a first-aid mask which stows in a compartment over the cabin door. The pilot has two push-pull controls in the cockpit. One which arms the system is on the left side of the center pedestal labeled PULL ON-SYSTEM READY and a PASSENGER MANUAL DROP OUT control handle is located on the right side of the center pedestal (Figure 17-1).

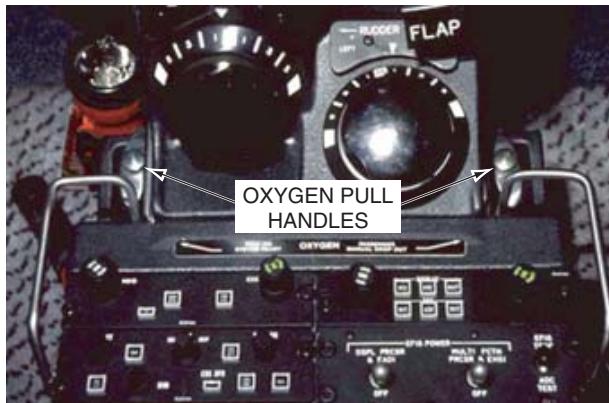


Figure 17-1 Oxygen System Control Handle

The PULL ON-SYSTEM READY handle operates a cable which opens and closes the shutoff valve located at the oxygen supply cylinder in the aft, unpressurized area of the fuselage. When this handle is pushed in, oxygen is unavailable in the airplane. It should be pulled out prior to engine starting to ensure that oxygen will be immediately available any time it is needed.

When this handle is pulled out, the primary oxygen supply line is charged with oxygen, provided the oxygen cylinder is not empty. (Check the

oxygen supply pressure gage on the copilot's right subpanel, and verify that sufficient oxygen is available for the flight.) The primary oxygen supply line delivers oxygen to the two crew masks in the cockpit, to the first aid outlet, and to the passenger system shutoff valve (Figure 17-2).

On the 350, an oxygen pressure switch is located at the passenger system shutoff valve. This switch senses pressure in the primary oxygen supply line. When the system is not armed and there is no pressure in the primary oxygen supply line, the switch will illuminate the OXY NOT ARMED annunciator. If sufficient pressure is in the line, the switch will extinguish the OXY NOT ARMED annunciator.

The Super King Air 300 and 350 auto-deployment system provides an adequate oxygen flow up to a cabin pressure altitude of 35,000 feet. The normal mask flow rate is 3.9 liters per minute (liters per minute—normal temperature/pressure differential). The diluter-demand masks (Figure 17-3) used by the flight crew use twice the normal amount when used in 100% or EMERG. Before each flight, check that the crew masks are in the 100% mode.

OXYGEN SYSTEM CONTROLS

A shutoff valve regulator, located on the cylinder, is actuated by the PULL ON-SYSTEM READY push-pull handle located on the left side of the center pedestal in the cockpit. The regulator is a constant-flow type which supplies low-pressure oxygen through aluminum plumbing to the outlets. The control handle should be pulled out (to arm the system) prior to engine start.

Electrical power is required for the automatic sensor to deploy the passenger mask system. It is powered through the OXY CONTROL circuit breaker on the copilot's circuit-breaker panel (Figure 17-4). As cabin altitude climbs above approximately 12,500 feet, the automatic sensor opens the passenger oxygen system shutoff valve. When the cabin descends below 12,500 feet or electrical power is removed from the circuit, the shutoff valve will close, thereby terminating oxygen flow to the passenger masks.

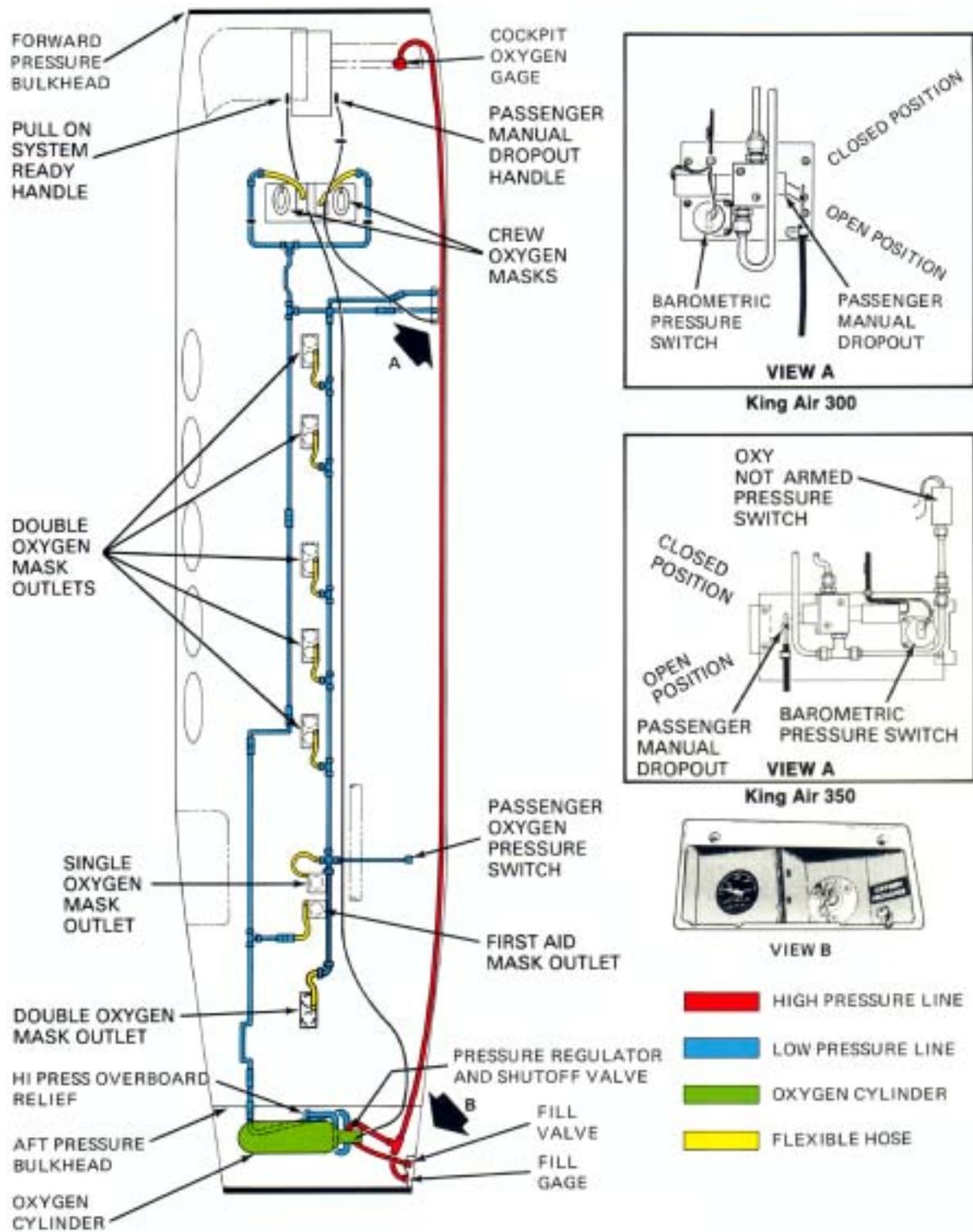


Figure 17-2 Oxygen System Schematic



Figure 17-3 Crew Oxygen Mask

On the right side of the pedestal is the PASSENGER MANUAL DROP OUT (override) push-pull handle. The manual drop-out handle will open the passenger oxygen shutoff valve, which will pressurize the passenger oxygen system whenever the oxygen system is armed. Electrical power is not required for the manual drop-out system to operate.

DILUTER-DEMAND CREW OXYGEN MASKS

The crew is provided with diluter-demand, quick-donning oxygen masks. These masks are stowed in the cockpit overhead, between the pilot's and copilot's seats. Since they are always plugged in and conveniently stowed, they can be

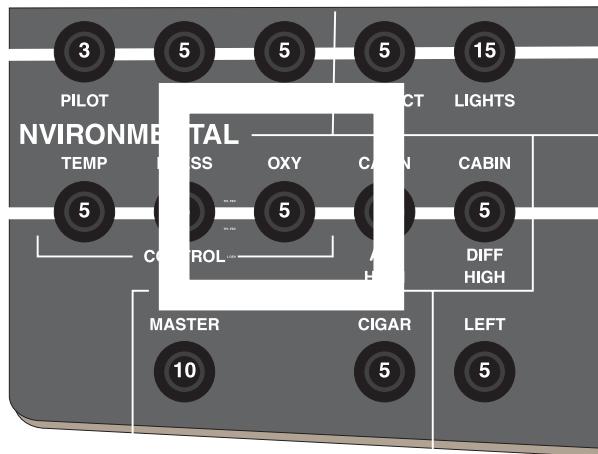


Figure 17-4 OXY Circuit Breaker

donned easily and quickly with one hand. Squeezing the red finger grip control switch on the face of the mask will inflate the elastic straps with oxygen pressure.

Fit the inflated straps over the head, and hold the mask over the nose and mouth as the grip is released. The elastic straps will deflate and hold the mask tightly against the face. The diluter-demand crew masks deliver oxygen to the user only upon inhalation. Consequently, there is no loss of oxygen when the masks are plugged in and the PULL ON-SYSTEM READY handle is pulled out, even though oxygen is immediately available upon demand.

A small switch on each diluter-demand oxygen mask permits the selection of two modes of operation: NORMAL and 100%. In the NORMAL position, air from the cockpit is mixed with the oxygen supplied through the mask. This reduces the rate of depletion of the oxygen supply, and it is more comfortable to use than 100% aviator's breathing oxygen. However, in the event of smoke or fumes in the cockpit, the 100% position should be used to prevent the breathing of contaminated air. For this reason, the selector levers should be left in the 100% position when the masks are not in use so that the masks are always ready for emergency use.

A red pushbutton/knob on the crew mask can be pushed to test the mask. Oxygen will flow from the mask when the system is on and the button is



depressed. For certain emergencies, the crew masks must be selected to EMERG. Turn this red knob to the EMERG position, and a small constant, positive flow of oxygen will be supplied to the mask. In EMERG, the crew mask is considered to be a constant-flow type until a small amount of pressure builds in the face mask.

FIRST-AID OXYGEN MASK

Any time the primary oxygen supply line is charged, oxygen can be obtained from the first-aid mask (located near the entrance door in an overhead compartment) by manually opening the overhead access door (placarded FIRST AID OXYGEN-PULL) and opening the ON/OFF valve inside the box (Figure 17-5). The PULL ON-SYSTEM READY handle in the cockpit must be pulled out before oxygen will flow from the first-aid oxygen mask.



Figure 17-5 First-Aid Mask Access Panel

PASSENGER OXYGEN AUTODEPLOYMENT SYSTEM

The passenger oxygen system is of the constant-flow type. With electrical power available, any time the cabin pressure altitude exceeds approximately 12,500 feet, a barometric-pressure switch automatically energizes a solenoid, which opens the passenger oxygen system shutoff valve. The pilot can open the valve manually any time by pulling the PASSENGER MANUAL DROP OUT handle out. Once the passenger oxygen system shutoff valve has been opened (either automatically or manually), and the primary oxygen system line has been charged (if the oxygen supply bottle contains oxygen and the PULL ON-SYSTEM READY handle in the cockpit is pulled out), oxygen will flow into the passenger oxygen supply line.

When oxygen flows into the passenger oxygen system supply line, a pressure-sensitive switch in the line closes a circuit to illuminate the green PASS OXY ON annunciator (Figures 17-6 and 17-7). This switch also causes the cabin indirect lights to illuminate bright, regardless of the CABIN LIGHTS BRIGHT-DIM-OFF switch position located on the overhead panel. In addition, the NO SMOKING and FASTEN SEAT BELT signs illuminate and the chime sounds.

The pressure of the oxygen in the passenger oxygen system supply line automatically extends a plunger against each of the passenger mask dispenser doors, forcing the doors open. The masks then drop down about 9 inches below the dispensers (Figure 17-8). The lanyard valve pin at the top of the mask hose must be pulled out in order for oxygen to flow to the masks. The pin is connected to the oxygen mask via a flexible cord; when the oxygen mask is pulled down for use, the cord pulls the pin out of the lanyard valve. The lanyard valve pin must be manually reinserted into the valve in order to stop the flow of oxygen when the mask is no longer needed.

Passenger oxygen can be shut off and the remaining oxygen isolated to the crew and first-aid outlets by pulling the OXY CONTROL circuit breaker in the ENVIRONMENTAL group on the copilot's circuit-breaker panel, provided the



L DC GEN	L GEN TIE OPEN	HYD FLUID LOW	RVS NOT READY	R GEN TIE OPEN	R DC GEN
L CHIP DETECT	L NO FUEL XFR	BAT TIE OPEN	DUCT OVERTEMP	R NO FUEL XFR	R CHIP DETECT
L ENG ICE FAIL	L FUEL QTY	BATTERY CHARGE	EXT PWR	R FUEL QTY	R ENG ICE FAIL
L BL AIR OFF	AUTOFILTER OFF	PITCH TRIM OFF	OXY NOT ARMED	RUD BOOST OFF	R BL AIR OFF
L IGNITION ON	L ENG ANTI-ICE	FUEL CROSSFEED	_____	R ENG ANTI-ICE	R IGNITION ON
WING DEICE	L BK DEICE ON	MAN TIES CLOSE	ELEC HEAT ON	R BK DEICE ON	TAIL DEICE
L PROP PITCH	CABIN ALTITUDE	LDG/TAXI LIGHT	PASS OXY ON	AIR COND N ₁ LOW	R PROP PITCH

Figure 17-6 King Air 350 Caution and Advisory Annunciators Prior to FL 121

PASSENGER MANUAL DROP OUT handle is pushed in (Figure 17-1).

OXYGEN SUPPLY CYLINDER

Oxygen for unpressurized, high-altitude flight is supplied by a cylinder mounted behind the aft pressure bulkhead (Figure 17-9). The cylinder is filled

by a valve accessible through an access door on the right side of the aft fuselage. The high-pressure system has two pressure gages, one located on the copilot's right subpanel in the cockpit for preflight and in-flight use (Figure 17-10), and one adjacent to the filler valve for checking system pressure while servicing (Figure 17-11). The cylinder is available in four different capacities on the King Air 300: 22

L DC GEN	L GEN TIE OPEN	HYD FLUID LOW	RVS NOT READY	R GEN TIE OPEN	R DC GEN
L CHIP DETECT	L NO FUEL XFR	BAT TIE OPEN	DUCT OVERTEMP	R NO FUEL XFR	R CHIP DETECT
L ENG ICE FAIL	L FUEL QTY	_____	EXT PWR	R FUEL QTY	R ENG ICE FAIL
L BL AIR OFF	AUTOFILTER OFF	PITCH TRIM OFF	OXY NOT ARMED	RUD BOOST OFF	R BL AIR OFF
_____	_____	_____	PROP GND SOL	_____	_____
L IGNITION ON	L ENG ANTI-ICE	FUEL CROSSFEED	_____	R ENG ANTI-ICE	R IGNITION ON
WING DEICE	L BK DEICE ON	MAN TIES CLOSE	ELEC HEAT ON	R BK DEICE ON	TAIL DEICE
L PROP PITCH	CABIN ALTITUDE	LDG/TAXI LIGHT	PASS OXYGEN ON	AIR COND N ₁ LOW	R PROP PITCH

Figure 17-7 King Air 350 Caution, Advisory, and Status Annunciators FL289 and After



cubic feet, 50 cubic feet, 77 cubic feet, or 115 cubic feet. On the King Air 350, only 50, 77, or 115 cubic foot cylinders are available.

WARNING

To prevent overheating, fill the oxygen system slowly by adjusting the recharging rate with the pressure regulating valve on the recharging unit. All oxygen cylinders should be filled to 1,800 psi at a temperature of 21°C. This pressure may be increased an additional 3.5 psi for each degree above 21°C; similarly, for each degree below 21°C, reduce the pressure for the cylinder by 3.5 psi. When the oxygen system is properly charged, disconnect the filler hose from the filler valve, and replace the protective cap. When servicing, purging, or replacing the oxygen cylinder, if it becomes necessary to disconnect a fitting, the threads of the fitting should be wrapped with MIL-T-27730 anti-seize tape prior to being reconnected to the system.

OXYGEN DURATION

It is a preflight requirement to check the oxygen available, considering the number of crew and passengers, to assure that it is sufficient for descent to 12,500 feet, or until loss of pressure in the airplane can be corrected and cabin altitude pressure restored. Full oxygen system pressure is $1,800 \pm 50$ psi at 21°C.

In order to determine if sufficient oxygen is available, first read the oxygen pressure gage and note the pressure. Use the Oxygen Available with a Partially Full Bottle graph to find the percent of usable capacity. To obtain the duration in minutes of the supply, obtain the duration for a full bottle from the Oxygen Duration chart (Figure 17-12), considering the number of persons aboard. Multiply the full bottle duration by the percent of usable capacity to obtain the available oxygen duration in minutes.



Figure 17-8 Passenger Oxygen Mask

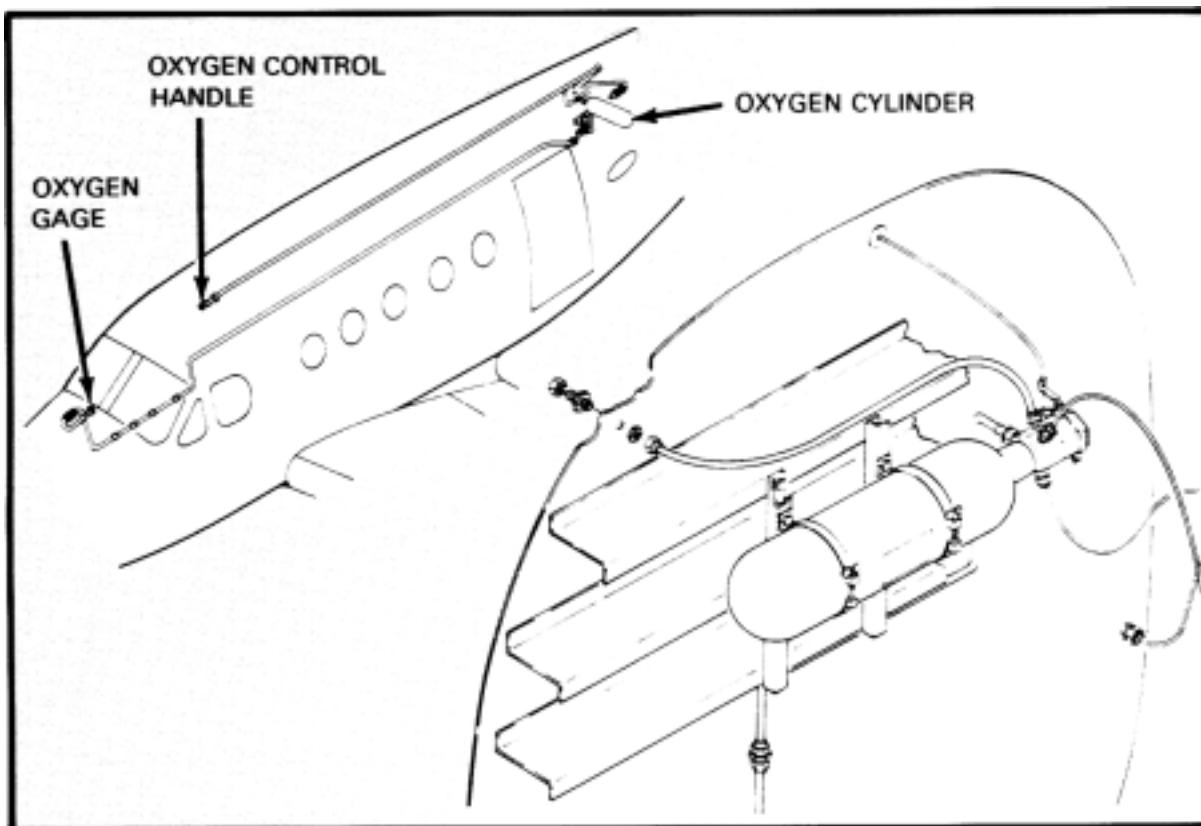


Figure 17-9 Oxygen Cylinder Installation

Oxygen Duration Computation

In this computation, oxygen duration is computed for an autodeployed mask, rated 3.9 Liters Per Minute (LPM-NTPD), color-coded orange and white, and approved for pressure altitudes up to 35,000 feet. The table used to compute the duration is also used for the quick-donning, diluter-demand crew oxygen masks. For all oxygen duration computations, count each diluter-demand crew mask in use in the 100% mode as two people. For example, to compute oxygen duration for four passengers and two crewmembers in the airplane, consider it as eight people using oxygen.

To compute the duration in minutes of the available oxygen for eight people, assume the pressure gage shows 1,500 pounds at 32°F. Enter the Oxygen Available with a Partially Full Bottle graph (Figure 17-13) at 1,500 pounds, and read

across to the 32°F intersection, then down to read 85% of usable capacity. To compute the duration available, enter the Oxygen Duration chart at the “eight people using” column, and read down to find “61 minutes available” for a 77-cubic-foot supply bottle. Now take 85% of 61, and find the current oxygen duration available of approximately 51.85 minutes.

PURGING AND SERVICING THE OXYGEN SYSTEM

The oxygen system is serviced by a filler valve accessible by removing an access plate on the right side of the aft fuselage (Figure 17-11). The system has two pressure gages, one located on the copilot's right subpanel in the cockpit for preflight and in-flight use, and one adjacent to the filler valve for checking system pressure during filling.



Figure 17-10 Oxygen Pressure Gage



Figure 17-11 Oxygen Fill Valve and Gage



STATED CYLINDER SIZE (CU FT)	**NUMBER OF PEOPLE USING								
	1	2	3	4	5	6	7	8	9
DURATION IN MINUTES									
22	144	72	48	36	28	24	20	18	16
50	317	158	105	79	63	52	45	39	35
77	488	244	162	122	97	81	69	61	54
115	732	366	244	183	146	122	104	91	81

Figure 17-12 Oxygen Duration Chart

Refer to the *POH* and *Maintenance Manual* prior to purging or servicing the oxygen system. Access to the oxygen system pressure indicator and filler valve may be gained through an access door located on the right side of the aft fuselage. To recharge the oxygen system, remove the protective cap from the filler valve, and attach the hose from an oxygen recharging unit. Make sure that both the airplane oxygen system and the servicing equipment are properly grounded before servicing the system.

Offensive odors may be removed from the oxygen system by purging. The system should also be purged any time system pressure drops below 50 psi or the lines are left open. Purging is accomplished by connecting a recharging unit into the system and permitting oxygen to flow through the lines and outlets until any offensive odors have been carried away.

If any offensive odor still lingers, continue purging the system for an additional hour. If such odors still remain, replace the supply cylinder. After the system has been adequately purged, return the cylinder valve to its normal operation position, and service the system.

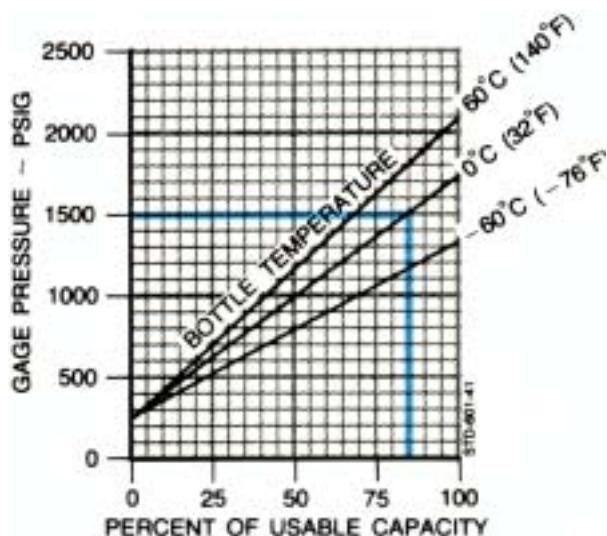


Figure 17-13 Percent of Usable Oxygen Capacity

WARNING

The following precautions should be observed when purging or servicing the oxygen system:

1. Avoid any operation that could create sparks. Keep all burning cigarettes or fire away from the vicinity of the airplane when the outlets are in use.
2. Inspect the filler connection for cleanliness before attaching it to the filler valve.
3. Make sure that your hands, tools, and clothing are clean, particularly of



- grease or oil stains, for these contaminants are extremely dangerous in the vicinity of oxygen.
4. As a further precaution against fire, open and close all oxygen valves slowly during filling.

CAUTION

Use only Aviators Breathing Oxygen (MIL-0-27210) for servicing the oxygen system. Do not use oxygen intended for medical purposes or such industrial uses as welding. Such oxygen may contain excessive moisture that could freeze in the valves and lines of the oxygen system.

OXYGEN CYLINDER RETESTING

Oxygen cylinders used in the airplane are of two types. Lightweight cylinders, marked "3HT" on the side plate, must be hydrostatically tested every three years and the test date marked on the cylinder. This bottle has a service life of 4,380 pressurizations or 24 years, whichever occurs first, and then must be discarded. Regular weight cylinders, stamped "3A" or "3AA," must be hydrostatically tested every five years and stamped with the retest date. Service life on these cylinders is not limited. Refer to the *Maintenance Manual* for more details.

TIME OF USEFUL CONSCIOUSNESS

In the event of decompression at altitude, the body's primary need is for oxygen to prevent hypoxia. Hypoxia is a lack of the oxygen needed to keep the brain and other body tissues functioning properly. The early symptoms of hypoxia, such as an increased sense of well-being, quickly give way to slow reactions, impaired thinking ability, unusual fatigue, and a dull headache. Therefore, the crew must act quickly to don oxy-

gen masks and provide oxygen to the passengers before the onset of hypoxia.

The Time of Useful Consciousness table shows the average time of useful consciousness available at various altitudes. This is the time from the onset of hypoxia until loss of effective performance. Individual reactions may differ from those shown in the table. Using the **EMERGENCY DESCENT** procedure in the Emergency Procedures section of the *POH* can minimize the effects of hypoxia.

NOTE

Table 17-1 sets forth the average time of useful consciousness (time from onset of hypoxia until loss of effective performance) at various altitudes.

Table 17-1 TIME OF USEFUL CONSCIOUSNESS

ALTITUDE	TIME
35,000 feet	1/2 to 1 minute
30,000 feet	1 to 2 minutes
28,000 feet	2 1/2 to 3 minutes
25,000 feet	3 to 5 minutes
22,000 feet	5 to 10 minutes
12 to 18,000 feet	30 minutes or more

PHYSIOLOGICAL TRAINING

What Is It?

Physiological training is a program directed toward understanding and surviving in the flight environment. It covers the problems of both high and low altitudes and recommends procedures to prevent or minimize the human-factor errors which occur in flight.



Who Needs It?

The course is primarily of benefit to pilots. It is also recommended for other aircrew personnel, Air Traffic Controllers, Aviation Medical Examiners, and other personnel from the national aviation system.

Where Can You Get It?

A resident physiological training course at the FAA's Aeronautical Center in Oklahoma City is devoted entirely to problems in civil aviation. Many military installations, and the National Aeronautics and Space Administration (NASA) in Houston, Texas, conduct a resident program for non-government personnel.

How Long Is the Course?

The course takes one full day.

What Is Contained in the Course?

Many topics are covered. They include the environment to which the flyer is exposed, physiological functions of the body at ground level, and alteration of some of these functions by changes in the environment. The higher one flies, the more critical becomes the need for supplemental oxygen. This need is discussed so that the trainee will understand why a pilot cannot fly safely at altitudes in excess of 12,500 feet for a prolonged period without some aid, either supplemental oxygen or a pressurized aircraft. Both oxygen equipment and pressurization are discussed.

When humans are confronted with certain stressful situations, there is a tendency to breathe too rapidly. This topic (hyperventilation) and methods of control are discussed. Ear pain on descent and other problems with body gases and procedures to prevent or minimize gas problems are explained. Alcohol, tobacco, and drugs are also discussed as they apply to flying. Pilot vertigo is discussed and demonstrated so that the trainee will understand why

a non-current instrument pilot should never attempt to fly in clouds and other weather situations where visibility is reduced.

Resident courses include an altitude chamber flight (Figure 17-14), where the trainees experience individual symptoms of oxygen deficiency as well as decompression.

This flight will demonstrate that:

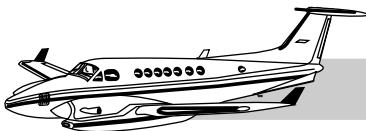
- Proper oxygen equipment and its use will protect an individual from oxygen deficiency.
- An individual can experience and recognize symptoms that will be the same as those found in actual flight and therefore take the necessary action to prevent loss of judgment and consciousness.
- Decompression is not dangerous, provided proper supervision is present and proper actions are planned and taken when necessary.

What Are the Prerequisites for Training?

Personnel must have a valid FAA Medical Certificate. A fee of twenty dollars is required. The applicant must be eighteen years of age or older.



Figure 17-14 FAA Altitude Chamber



How Do You Apply for Training?

For more information about FAA Aeromedical training programs contact:

FAA Civil Aerospace Medical Institute
Aeromedical Education Division, AAM-400
P.O. Box 25082
Oklahoma City, OK 73125
Phone: (405) 954-4837
Fax: (405) 954-2305
E-Mail: rogers_v_shaw@mmacmail.jccbi.gov

How Can You Get Further Information?

Write to the Airman Education Section at the above address, or phone (405) 680-4837.



APPENDIX SYMBOLS, ABBREVIATIONS, AND TERMINOLOGY

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APPENDIX

SYMBOLS, ABBREVIATIONS, AND TERMINOLOGY

AIRSPEED

CAS—Calibrated Airspeed is the indicated airspeed of an airplane corrected for position and instrument error. Calibrated airspeed is equal to true airspeed in standard atmosphere at sea level.

GS—Groundspeed is the speed of an airplane relative to the ground.

IAS—Indicated Airspeed is the speed of an airplane as shown on the airspeed indicator.

KCAS—Calibrated Airspeed expressed in Knots.

KIAS—Indicated Airspeed expressed in Knots.

M—Mach number is the ratio of true airspeed to the speed of sound.

TAS—True Airspeed is the airspeed of an airplane relative to undisturbed air, which is the CAS corrected for altitude, temperature, and compressibility.

V₁—Takeoff Decision Speed.

V₂—Takeoff Safety Speed is the speed at 35 feet AGL, assuming an engine failure at V₁.

V₃₅—Takeoff Safety Speed at 35 feet AGL with both engines operating.

V_A—Maneuvering Speed is the maximum speed at which application of full available aerodynamic control will not overstress the airplane.

V_F—Design Flap Speed is the highest speed permissible at which wing flaps may be actuated.

V_{FE}—Maximum Flap Extended Speed is the highest speed permissible with wing flaps in a prescribed extended position.

V_{LE}—Maximum Landing Gear Extended Speed is the maximum speed at which an airplane can be safely flown with the landing gear extended.

V_{LO}—Maximum Landing Gear Operating Speed is the maximum speed at which the landing gear can be safely extended or retracted.

V_{MCA}—Air Minimum Control Speed is the minimum flight speed at which the airplane is directionally controllable, as determined in accordance with Federal Aviation Regulations. The airplane certification conditions include: one engine becoming inoperative with autofeather armed, a 5° bank toward the operative engine, takeoff power on the operative engine, rudder boost on, landing gear up, flaps in the takeoff position, and most rearward CG. For some conditions of weight and altitude, stall can be encountered at speeds above V_{MCA}, as established by the certification procedure described above, in which event stall speed must be regarded as the limit of effective directional control.

V_{MCG}—Ground Minimum Control Speed.

V_{MO/M_{MO}}—Maximum Operating Limit Speed is the speed limit that may not be deliberately exceeded in normal flight operation. V is expressed in knots and M in Mach number.

V_R—Rotation Speed.

V_{REF}—Reference Landing Approach Speed with the landing gear and flaps down.



V_S—Stalling Speed or the minimum steady flight speed at which the airplane is controllable.

V_{SO}—Stalling Speed or the minimum steady flight speed at which the airplane is controllable in the landing configuration.

V_{SSE}—Intentional One-Engine-Inoperative Speed is a speed above both V_{MCA} and stall speed, selected to provide a margin of lateral and directional control when one engine is suddenly rendered inoperative. Intentional failing of one engine below this speed is not recommended.

V_X—Best Angle-Of-Climb Speed is the airspeed which delivers the greatest gain of altitude in the shortest possible horizontal distance.

V_Y—Best Rate-Of-Climb Speed is the airspeed which delivers the greatest gain in altitude in the shortest possible time.

METEOROLOGICAL

Altimeter Setting—Barometric pressure corrected to sea level.

Indicated Pressure Altitude—The number actually read from an altimeter when the barometric subscale has been set to 29.92 inches of mercury (1013.2 millibars).

IOAT—Indicated Outside Air Temperature is the temperature value read from an indicator.

ISA—International Standard Atmosphere in which:

- Air is a dry, perfect gas.
- Temperature at sea level is 59°Fahrenheit (15°Celsius).
- Pressure at sea level is 29.92 inches of mercury (1013.2 millibars).
- Temperature gradient from sea level to the altitude at which the temperature is -69.7°F (-56.5°C), is $-0.003566^{\circ}\text{F}$ (-0.00198°C) per foot, and is zero above the altitude.

OAT—Outside Air Temperature is the free air static temperature, obtained either from the temperature indicator (IOAT) adjusted for compressibility effects or from ground meteorological sources.

Pressure Altitude—Altitude measured from standard sea level pressure (29.92 inches Hg) by a pressure (barometric) altimeter. It is the indicated pressure altitude corrected for position and instrument error. In this manual, altimeter instrument errors are assumed to be zero. Position errors may be obtained from the altimeter correction graphs.

Station Pressure—Actual atmospheric pressure at field elevation.

Temperature Compressibility Effects—An error in the indication of temperature caused by airflow over the temperature probe. The error varies depending on altitude and airspeed.

Wind—The Wind velocities recorded as variables on the charts of this manual are to be understood as the headwind or tailwind components of the reported winds.

POWER

Beta Range—The region of the power lever control which is aft of the idle gate.

Cruise Climb—Cruise Climb is the maximum power approved for normal climb. These powers are torque, N₁, or temperature (ITT) limited.

High Idle—High Idle is obtained by placing the condition lever in HIGH IDLE position, which sets minimum engine speed to approximately 70% of maximum N₁ rpm.

Low Idle—Low Idle is obtained by placing the condition lever in LOW IDLE position, which sets minimum engine speed to approximately 62% of maximum N₁ rpm.

Maximum Continuous Power—Maximum Continuous Power is the highest power rating not limited by time. Use of this rating is at the discretion of the pilot.



Maximum Cruise Power—Maximum Cruise Power is the highest power rating for cruise and is not time limited.

Ground Fine Range—The Ground Fine Range may be selected by moving the power levers aft of the idle gate to, but not over, the ground fine gate. Propeller ground fine operation is used to provide deceleration on the ground during landing and accelerate-stop conditions by taking advantage of maximum available propeller drag without creating negative thrust.

Reverse—Reverse thrust is obtained by lifting the power levers and moving them aft of the ground fine gate.

SHP—Shaft Horsepower.

Static or Minimum Takeoff Power—Static or Minimum Takeoff Power is the minimum power which must be available for takeoff without exceeding the engine limitations.

Takeoff Power—Takeoff Power is the maximum power rating. Takeoff power is the same as maximum continuous power on the PT6A-60A.

CONTROL AND INSTRUMENT

Condition Lever (Fuel Cutoff Lever)—The fuel cutoff lever actuates a valve in the fuel control unit which controls the flow of fuel at the fuel control outlet and regulates the idle range from low to high idle.

ITT (Interstage Turbine Temperature)—Ten probes, wired in parallel, sense the temperature between the compressor and power turbines, and send the reading to the ITT indicator in degrees centigrade x 100.

N₁ Tachometer (Gas Generator RPM)—The N₁ tachometer registers the rpm of the gas generator in percent, with 100% representing a gas generator speed of 37,500 rpm.

Power Lever (Gas Generator N₁ rpm)—The power lever serves to modulate engine power from full reverse thrust to takeoff. The position for idle represents the lowest recommended level of power for flight operation.

Propeller Control Lever (N_P rpm)—The propeller control is used to control the rpm setting of the propeller governor. Movement of the lever results in an increase or decrease in propeller rpm. Propeller feathering is the result of lever movement beyond the detents at the low rpm (high pitch) end of the lever travel.

Propeller Governor—The Propeller Governor senses changes in rpm and hydraulically changes propeller blade angle to compensate for the changes in rpm. Constant propeller rpm is thereby maintained at the selected rpm setting.

Torquemeter—The Torquemeter System indicates the propeller shaft output torque. Differential pressure from the mechanism within the reduction gearcase causes a bellows and servo system to indicate torque on a meter. Instrument readout is in percent of maximum torque available.

GRAPH AND TABULAR

Accelerate-Go—Accelerate-Go is the distance to accelerate to takeoff decision speed (V₁), experience an engine failure, continue accelerating to liftoff, then climb and accelerate in order to achieve takeoff safety speed (V₂) at 35 feet above the runway.

Accelerate-Stop—Accelerate-Stop is the distance to accelerate to takeoff decision speed (V₁) and then bring the airplane to a stop.

AGL—Above Ground Level.

Best Angle-Of-Climb—The Best Angle-Of-Climb delivers the greatest gain of altitude in the shortest possible horizontal distance with gear and flaps up.



Best Rate-Of-Climb—The Best Rate-Of-Climb delivers the greatest gain of altitude in the shortest possible time with gear and flaps up.

Clearway—A Clearway is an area beyond the airport runway not less than 500 feet wide, centrally located about the extended centerline of the runway, and under the control of the airport authorities. The clearway is expressed in terms of a clear plane, extending from the end of the runway with an upward slope not exceeding 1.25%, above which no object nor any terrain protrudes. However, threshold lights may protrude above the plane if their height above the end of the runway is 26 inches or less and if they are located to each side of the runway.

Climb Gradient—Climb Gradient is the ratio of the change in height during a portion of a climb to the horizontal distance traversed in the same time interval.

Demonstrated Crosswind—Demonstrated Crosswind is the demonstrated crosswind component for which adequate control of the airplane during takeoff and landing was actually demonstrated during certification; however, this is not considered a limitation.

MEA—Minimum Enroute Altitude.

Net Gradient Of Climb—Net Gradient Of Climb is the gradient of climb with the flaps in the takeoff position and the landing gear retracted. “Net” indicates that the actual gradients of climb have been reduced by 8 to 10% to allow for turbulence and pilot technique. The net gradient of climb graphs are constructed so that the value(s) obtained using the airport pressure altitude and outside air temperature will be the gradient approximately 700 feet above the runway.

Route Segment—Route Segment is a part of a route. Each end of that part is identified by a:

- Geographic location, or
- Point at which a definite radio fix can be established

Takeoff Field Length—The minimum runway length required for departure. This distance is the longest of the:

- Distance to accelerate and recognize an engine failure at V_1 , accelerate to and rotate at V_R , then climb and accelerate in order to achieve V_2 at 35 feet above the runway, or
- Distance to accelerate and recognize an engine failure at V_1 and bring the airplane to a stop, or
- All-engine-operating distance to accelerate to and rotate at V_R , then climb and accelerate in order to achieve V_{35} at 35 feet above the runway, increased by 15%.

Takeoff Flight Path—Takeoff Flight Path is the minimum gradient of climb required to clear obstacles in excess of 35 feet, measured horizontally from reference zero and vertically at the altitude above the runway. Reference zero is the point where the airplane has reached 35 feet above the runway, as determined from the accelerate-go graphs.

WEIGHT AND BALANCE

Approved Loading Envelope—Those combinations of airplane weight and center of gravity which define the limits beyond which loading is not approved.

Arm—Arm is the distance from the center of gravity of an object to a line about which moments are to be computed.

Basic Empty Weight—Basic Empty Weight is the weight of an empty airplane, including full engine oil and unusable fuel. This equals empty weight plus the weight of unusable fuel, and the weight of all the engine oil required to fill the lines and tanks. Basic empty weight is the basic configuration from which loading data is determined.



Center Of Gravity—Center Of Gravity is the point at which the weight of an object may be considered concentrated for weight and balance purposes.

CG Limits—CG Limits are the extreme center-of-gravity locations within which the airplane must be operated at a given weight.

Datum—Datum is a vertical plane perpendicular to the airplane's longitudinal axis from which fore and aft (usually aft) measurements are made for weight and balance purposes.

Empty Weight—Empty Weight is the weight of an empty airplane before any oil or fuel has been added. This includes all permanently installed equipment, fixed ballast, full hydraulic fluid, full chemical toilet fluid, and all other operating fluids full, except that the engines, tanks, and lines do not contain any engine oil or fuel.

Engine Oil—That portion of the engine oil which can be drained from the engine.

Jack Point—Jack Points are points on the airplane identified by the manufacturer as suitable for supporting the airplane for weighing or other purposes.

Landing Weight—Landing Weight is the weight of the airplane at landing touchdown.

Leveling Points—Leveling Points are those points which are used during the weighing process to level the airplane.

Maximum Weight—Maximum Weight is the greatest weight allowed by design, structural, performance, or other limitations.

Maximum Zero Fuel Weight—Any weight above the value given must be loaded as fuel.

Moment—Moment is a measure of the rotational tendency of a weight, about a specified line, mathematically equal to the product of the weight and the arm. This is also referred to as torque.

Payload—Payload is the weight of occupants, cargo, and baggage.

PPH—Pounds Per Hour.

Ramp Weight—Ramp Weight is the airplane weight at engine start, assuming all loading is completed.

Station—Station is the longitudinal distance from some point to the zero datum or zero fuselage station.

Takeoff Weight—Takeoff Weight is the weight of the airplane at liftoff from the runway.

Tare—Tare is the apparent weight of any items (wheel chocks, jack stands, etc.) used on the scales but which are not a part of the airplane weight.

Unusable Fuel—Unusable Fuel is the fuel remaining after consumption of usable fuel.

Usable Fuel—Usable Fuel is that portion of the total fuel which is available for consumption as determined in accordance with applicable regulatory standards.

Useful Load—Useful Load is the difference between the airplane ramp weight and the basic empty weight.

Zero Fuel Weight—Zero Fuel Weight is the airplane ramp weight minus the weight of fuel on board.

AVIONICS

ADF Mode—A Mode of Automatic Direction Finder operation allowing the ADF needle to point to the station.

NOTE

In this mode of operation, on many receivers the audio fidelity is severely limited.



Air Data Computer—An electronic system primarily designed to gather information for an autopilot flight director system with outputs relating to pitot and static data. Possible information from this system includes: pressure altitude, indicated airspeed, total air temperature, static air temperature, and other information related to autopilot operation.

Altitude Alert Light—An amber light associated with an altitude alerter system. This light will be illuminated prior to intercepting a preselected altitude, or if for any reason the aircraft strays beyond a preset limit from the selected altitude once the aircraft has intercepted the altitude.

Altitude Preselector—An autopilot flight director subsystem that allows a pilot to preselect the altitude to which he desires to climb or descend. The controlling mechanism for an altitude preselect system is normally combined with the same device which controls the altitude alerter system.

Amplifier—A basic type of electronic device that seeks to make an electrical signal greater in strength. A public address system, for instance, is a type of amplifier. Amplifying devices are typically tubes or transistors.

Analog—A type of electronic circuitry that is characterized by smooth, continuous operation rather than discrete steps, as would be observed with digitally operated equipment.

Angle-Of-Attack (AOA) Indicator—A supplemental flight instrumentation system that attempts to read out to the pilot the angle-of-attack or deck angle information. Several variations of this system are available.

Angular Deviation—A means of showing displacement from a selected course either to or from a VOR station, TACAN, or NDB, showing displacement from the desired course in terms of angle. This is commonly used with the VOR system having a normal course width of 10° on each side of the course.

Annunciator—An indicator light with a message. An annunciator makes an announcement as to the specific status of a system or subsystem.

ANT (Antenna) Mode—This mode of ADF operation allows improved audio fidelity in order to listen to the music or voice programs of an AM broadcast station. However, the ADF needle operation is defeated in this mode of operation.

Area Navigation System (RNAV)—A system of direct point-to-point navigation having four further subclassifications:

- **Course Line Computer**—A computer, utilizing information from VOR and colocated DME stations, that allows the operator to change the location of the VOR station from its physical position to wherever the operator wants.
- **Inertial Navigation System**—See related definition.
- **GPS System**—See related definition.
- **Loran System**—Operationally similar to an GPS system.

Asymptotic—A design characteristic of an autopilot or flight director system. The function of this characteristic is to allow the autopilot to attempt to intercept a given course or altitude without overshooting. This is done by continuously reducing the intercept angle as the aircraft approaches the selected ground track or altitude.

Attenuation—The process of electrically reducing the size of a radio or audio signal (i.e., to turn down or make smaller.)

Attitude Director Indicator (ADI) (Flight Director Indicator)—This instrument combines the basic functions of an attitude indicator with the steering commands received from the flight director system.

Attitude Indicator (Artificial Horizon)—A gyroscopically controlled instrument used to display the aircraft's pitch and roll attitude relative to the earth's surface. The gyro used to display this information may be contained within the case of the displayed instrument, or it may receive its information from a remotely located attitude gyro.



Audio Filters—An electronic means of removing a portion of the audio which the pilot does not desire to listen to. The pilot may choose to remove either the voice portion or the Morse code identifier of a VOR or an ADF system.

Audio Selector Switches—The system of switches which allows one or several audio systems to be “piped-in” to the speaker or headphones of an aircraft.

Audio System—The electronic system that serves as a switchboard and amplification system for the varied receivers that require the audio to be funneled to the speaker or headphones.

Autopilot/Flight Director Modes:

- **Vertical Modes**—Vertical Modes control changes in the pitch attitude of the aircraft using the elevator servo. Examples of vertical modes are: altitude hold, altitude preselect, indicated airspeed hold, and vertical speed hold.
- **Lateral Modes**—Lateral Modes control autopilot operation by controlling the aileron and rudder servos. Examples of lateral modes are: heading hold, navigation modes and submodes (e.g., en route nav tracking, approach, backcourse, etc.).

Autopilot/Flight Director Submode—Generally speaking, this concept represents two subclassifications of operation within a given mode. Namely:

- **Arm**—The process of activating a system or preparing it to operate at a future time. For instance, if you push the Nav button to track a specific radial from a VOR station, but the CDI needle is displaced full-scale to the left or right at the moment you push the Nav button, then the autopilot flight director system will initially be activated in the nav-arm mode while the aircraft continues to intercept the selected radial.

- **Capture**—A submode allowing the autopilot flight director system to track a specified lateral or vertical reference (e.g., altitude or glide slope as a vertical mode; VOR or localizer course as a lateral mode).

Avionics Master Circuit Breaker—The circuit breaker that supplies power to the avionics master switch in Beechcraft factory-installed avionics packages on Baron, Bonanza, Duke, and King Air installations. This circuit breaker serves as a backup means of activating the avionics system should the avionics master switch fail for any reason.

Avionics Master Switch—A central on/off power switch for the entire avionics package in an aircraft. This switch conveniently allows the pilot to turn on the entire avionics package by turning on only one switch.

Beat Frequency Oscillator (BFO)—A device used on an ADF receiver that generates a tone allowing the pilot to identify the Morse code being transmitted by some nondirectional beacons.

NOTE

This type of transmitter is not employed in the United States.

Carrier—That portion of the transmitted radio energy which “carries” the useful information (i.e., modulation).

Compass System Slaving—The process of automatically aligning the directional gyro in a compass system with the earth’s magnetic field to display the aircraft’s magnetic heading. When the compass system is initially powered, slaving occurs at a fast rate to quickly align the compass system with magnetic north. Once the fast-slaving rate is accomplished, the system automatically goes into a slow-slaving rate for continuous operation. It will correct for precession errors of the compass system up to a maximum error of about 3° per minute.



Concentric—Two or more knobs mounted on one common system of shafts having the same axis. For example, most frequency selector knobs used in all avionic systems employ concentric knobs in the interest of conserving panel space.

Course Deviation Indicator (CDI)—An indicator used with a VOR/localizer receiver that shows only left/right deviation and to/from information. This instrument has a knob called an OBS knob, meaning “omni bearing selector,” which allows the pilot to choose the course to or from a VOR station.

Course Knob—The name applied to the omni bearing selector on an HSI type of instrument. The course knob is attached to the resolver and a course pointer on the HSI indicator.

Course Width—Displacement left or right of the desired course:

- **Angular**—Degrees left or right of the desired course. Course width using the VOR system is 10° on each side of the desired course.
- **Linear**—In the “en route mode” most course line computer RNAVs have a course width of ± 5 nautical miles. In the approach mode most course line computer RNAVs have a course width of $\pm 1 \frac{1}{4}$ nautical miles on each side of the centerline.

Cross Sidetone—Sending sidetone audio across the cockpit from one side to the other; for example, this allows the pilot to hear what the copilot is saying on the transmitter. Cross sidetone may be heard through either the phones or the speaker.

Digital—A type of electronic circuitry technology that operates in specific steps, as opposed to the smooth, sweeping type of operation employed in analog.

Double-Cue Flight Director System (Cross Pointer)—A command presentation system using one vertical bar to indicate commanded

roll-attitude instructions and one horizontal bar to indicate commanded pitch instructions.

Electronic Flight Instrument System (EFIS)—A type of flight instrumentation system employing cathode ray tubes (television screens) to display information.

Emergency/Normal Switch—In the event of the failure of the audio system, this switch (when placed in the emergency position) allows audio from the aircraft receivers to be “piped” directly to the headphones.

Fast Erect—A mode of operation whereby an attitude indicator may be quickly realigned with the earth’s horizon if for any reason the gyro has precessed or tumbled.

Flux Valve (Flux Gate)—A component of a slaved compass system that senses the earth’s magnetic field and converts this information into an electrical signal representing magnetic north.

Free Operation—A mode of operation for a slaved compass system whereby the directional gyro is disconnected from the slaving system. Normally this would be used when the slaving system fails or for operation in the polar regions where the earth’s magnetic field will not permit normal slaved operation. The concept here is that the directional gyro is free of its master, magnetic north.

Gain—The relative amount of amplification of a radio receiver. A gain control is commonly used on a radar indicator to control the relative amount of amplification of the received radar echo. This allows the pilot to optimize the information displayed, especially when the radar is used for terrain mapping purposes.

Go-Around Mode—An autopilot flight director mode intended to be used during a missed approach. This mode will command a pitch-up attitude appropriate for a climbout with an associated wings-level command. The autopilot may or may not remain engaged during the go-around mode, depending upon the type of autopilot installed in a specific aircraft.



(GPS) Global Positioning System (Satellite Navigation)—A world-wide navigation system that allows direct great circle flight from one point to another. The system utilizes a constellation of satellites for navigation information.

Gyro Erection—The process of an attitude gyro becoming aligned with the earth's horizon or, viewed in another way, aligned with true vertical. This happens automatically when the system first receives power.

Half Bank—An autopilot mode of operation whereby the bank angle is limited during turns in such a way that the aircraft will only bank approximately half as much as normal. This is designed to give the passengers the perception of a smoother ride with no steep banks.

Heading Bug—An adjustable marker used on a heading indicator to direct an autopilot and/or flight director system according to the magnetic heading the pilot desires to fly. Also, this device may be used simply as a reminder to the pilot of what heading he is to fly when not using the autopilot flight director system.

Heading Indicator (Directional Gyro)—A gyroscopically controlled instrument used to display an aircraft's heading relative to magnetic north. The compass card of this indicator may be receiving the information which it displays from a remotely located gyro and an associated slaved compass system.

Hertz—The unit of measure used to describe the number of cycles of alternating current per second.

Horizontal Situation Indicator (HSI)—This instrument, alternately called a CDI by some manufacturers, displays heading information from a compass system, left/right and to/from information from a VOR/localizer receiver, and deviation above and below a glide slope from a glide-slope receiver. The pilot's workload is reduced by integrating these displays onto one indicator.

Inertial Navigation System—This system allows direct point-to-point navigation via a great circle route. This system is completely self-sufficient, utilizing a group of gyros and accelerometers to sense movement along the earth's surface.

Integrated Autopilot/Flight Director System—A system utilizing both autopilot and flight direction information to respond to selected modes.

Interrogation—In the secondary surveillance radar system the ground-based radar unit is said to "interrogate" the transponders of all aircraft flying within reception range of that radar. Once a transponder has been interrogated, it should reply to the ground radar unit by sending a brief transmission of radio energy. For general aviation aircraft a transponder may be interrogated in both modes. A mode supplies azimuth and distance information, and altitude information is provided through mode C.

Keying—The process of turning on the transmitter by means of the push-to-talk button located on the microphone or the control wheel.

Latitude—The angular displacement of a geographic location north or south of the equator. This is normally expressed in terms of degrees, minutes, and tenths of minutes.

Linear Deviation—A means of showing lateral displacement from the desired navigational course calibrated in miles. Linear deviation allows for parallel course boundaries whether far away from or near a station.

Longitude—The angular displacement of a geographic location east or west of the prime meridian located in Greenwich, England. This is normally expressed in terms of degrees, minutes, and tenths of minutes.

Magnetic Bearing—The direction of a nondirectional beacon (NDB) or VOR station relative to magnetic north.

Meter Movement—An application of an ammeter used in any instrumentation system to show deviation such as left/right, to/from, slaving indicator, etc.



Mode—One of several operating conditions of a system. For instance, most airborne weather radars have both weather mapping and terrain mapping modes of operation.

Mode A—That portion of the transponder reply which transmits azimuth and distance information for the radar controller.

Mode C—The portion of a transponder reply containing the pressure altitude of an aircraft as provided by an encoding altimeter.

Modulation—The addition of useful information to the carrier wave that is emitted from a transmitter; for example, talking into the microphone or the transmission of the Morse code identification from a VOR station.

Muting—The silencing of incoming receiver audio while one is transmitting.

Nonintegrated AP/FD System—Two separate flight control systems, each using its own computer. Information coming from these two systems may or may not agree at any given time.

Parallax Error—A problem that can cause inaccurate interpretations of an instrument reading. It is caused by the user's viewing angle not being directly in line with the instrument.

Parallax Error Adjustment—An adjustment of some single-cue flight director systems which allows the command bars to be adjusted up or down in order to "nestle" just above the aircraft symbol.

Parked Or Stowed ADF Needle—The process of placing the ADF needle at the 3 o'clock relative bearing location to indicate that the ADF unit is in the antenna mode and that the needle is not operating.

Radio Magnetic Indicator (RMI)—The combined display of magnetic heading from a compass system and relative bearing to a nondirectional beacon or VOR, which results in displaying the product, called "magnetic bearing," to or from the station.

Range Filter—An audio filter designed to remove the Morse code identification from a radio transmission. Actually, range is something of a misnomer as pilots know it today—you should think of this as being an "ident" filter.

Relative Bearing—The direction of a nondirectional beacon relative to the longitudinal axis of the aircraft.

Relay—An application of an electromagnet to perform switching duties. A relay may be used to switch large quantities of current. A multiple pole relay will allow a single pole switch to switch many circuits from a remote location.

Remote Mounted Avionics—Avionic equipment which is not fully self-contained and mounted on the instrument panel. Typically, the "black boxes" for these systems are located in the forward avionics compartment, forward of the front pressure bulkhead, or aft of the rear pressure bulkhead.

Resolver—The electronic device to which the course knob or OBS knob is attached. This device communicates the desired course, which the pilot selects, to the VOR receiver.

Servo System—Using an electric motor in any one of several applications to reduce pilot workload or allow automatic operation of some systems; for example: autopilot servos, electric elevator trim servos, servoed altimeters, compass systems, etc.

Sidetone—The ability to hear oneself talk while transmitting. The sidetone may be heard through either the headphones or the speaker. Additionally, sidetone may be considered as a means of verifying normal transmitter and receiver operation. If the receiver and transmitter are working properly, the sidetone will sound "normal." If either the transmitter or receiver is malfunctioning, the sidetone will sound weak or garbled.

Single-Cue Flight Director System (V-Bar System)—A command display system using a pair of bars which work in unison to display the commanded attitude to the pilot.



Slant/Range Correction—A means of correcting for the inherent error in raw slant/range data which will result in a true lateral distance from the aircraft to the DME station. Many of the more sophisticated RNAV computers provide slant/range correction.

Slant/Range Distance—Conventional, uncorrected DME distance to the station.

Slaved Compass System—A directional gyro system that is automatically synchronized to the magnetic heading of the aircraft. The concept of this system is that magnetic north is the master; therefore, the compass system is its slave.

Slaved Operation—The normal mode of a slaved compass system whereby the directional gyro automatically remains synchronized to magnetic north. This type of operation continually compensates for gyro precession and other compass system errors. The concept is that the compass system is a slave to magnetic north.

Slaving Amplifier—An amplifier which takes the weak signal representing magnetic north, coming from the flux valve, and boosts that signal to a usable level in order to drive the directional gyro to the proper magnetic heading.

Slaving Indicator—A meter used in some slaved compass systems that displays the difference between sensed magnetic heading and displayed magnetic heading. If the needle on this indicator is centered, there is no error between sensed and indicated magnetic heading. If the needle is off to the left or right, a small amount of error is indicated. Normal operation of the compass system causes the needle to sway to the left and right because of gyro precession and other factors.

Soft Ride—A mode for an autopilot whereby the responsiveness of the autopilot to rough air is altered in such a way that the ride is perceived to be much smoother than it is.

Squelch—A silencing circuit employed in communication receivers that allows undesirable background noise to be omitted. Only a strong incoming signal from a transmitter will be heard.

Transponder Code—A specific four-digit code that may be selected by the pilot on his transponder to identify his specific aircraft.

NOTE

A common misconception is that this control changes the transponder reply frequency. The transponder always operates on the same frequency.

Voice Filter—An audio filter designed to remove the voice portion of a received radio transmission.

Voice Terrain Advisories—Voice callouts of pertinent altitude-above-ground information. The information announced will be determined by the type of system installed.

Waypoint—The geographic location of navigational fix used in area navigation. This may be used in either a GPS system or a VOR/DME system utilizing a course line computer.

Waypoint Address—The radial and distance of a waypoint from a VORTAC.

Waypoint Coordinates—The latitude and longitude of the waypoint used with a GPS system.

Yaw Damper—A system connected to the rudder servo that seeks to dampen or reduce oscillations of the aircraft about the yaw axis. The yaw damper system significantly reduces the level of motion sickness experienced by passengers. This system should be engaged soon after takeoff and under normal operations should remain engaged until just prior to landing.