SECTION III
SYSTEMS DESCRIPTION
TABLE OF CONTENTS

SECTION III

SYSTEMS DESCRIPTION

TABLE OF CONTENTS (CONT'D)

SECTION III

SYSTEMS DESCRIPTION

TABLE OF CONTENTS (CONT'D)

SUBJECT *PAGE*

SECTION III

SYSTEMS DESCRIPTION

TABLE OF CONTENTS (CONT'D)

AIRFRAME

STRUCTURE

The Beech 1900D Airliner is a low-wing monoplane of metal construction. It has fully cantilevered wings, and a T-tail empennage.

SEATING ARRANGEMENTS

Seating for 19 passengers (18 for aircraft equipped with optional lavatory), plus crew, is available.

FLIGHT CONTROLS

CONTROL SURFACES

The airplane is equipped with conventional ailerons and rudder. It utilizes a T-tail horizontal stabilizer and elevator, mounted at the extreme top of the vertical stabilizer.

OPERATING MECHANISMS

The ailerons and elevators are operated by conventional control wheels interconnected by a T-bar. The rudder pedals are interconnected by linkage below the floor. These systems are connected to the control surfaces through push-rod and cable-and-bellcrank systems. An aileron-rudder interconnect is installed for coordination of aileron/rudder movement. Rudder, elevator, and aileron trim are adjustable with controls mounted on the center pedestal. Dual trim-tab push rods are provided for redundancy. A position indicator for each of the trim tabs is integrated with its respective control.

MANUAL ELEVATOR TRIM

Manual control of the elevator trim is accomplished with a hand wheel located on the left side of the pedestal. It is a conventional trim wheel which is rolled forward for nose-down trim, and aft for nose-up trim.

ELECTRIC ELEVATOR TRIM (IF INSTALLED)

The electric elevator trim system, if installed, is controlled by an ELEV TRIM (ON) - OFF switch located on the pedestal, a dual-element thumb switch on each control wheel, a trim-disconnect switch on each control wheel, and an ELEV TRIM circuit breaker in the FLIGHT group on the right side panel. Both elements of either dual-element thumb switch must be simultaneously moved forward to achieve nose-down trim, aft for nose-up trim; when released, they return to the center (OFF) position. Any activation of the trim system by the copilot's thumb switch can be overridden by the pilot's thumb switch. No one switch element should activate the system. Only the simultaneous movement of a pair of switch elements in the same direction should activate the system.

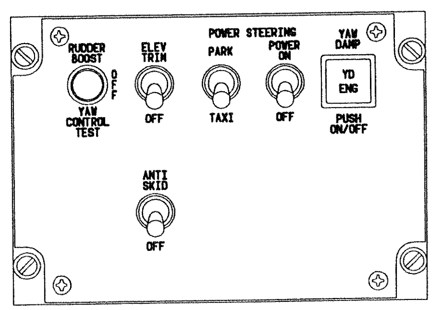
A bi-level, push-button, trim-disconnect switch is located inboard of the dual-element thumb switch on the outboard grip of each control wheel. The electric elevator trim system can be disconnected by depressing either of these switches. Depressing either trim-disconnect switch to the first of the two levels disconnects the autopilot (if installed), the yaw damp system, and interrupts the rudder boost. Depressing the switch to the second level disconnects the electric elevator trim system. The manual-trim control wheel can be used to change the trim anytime, whether or not the electric trim system is in the operative mode. The trim system must never be forced past the up or down stops during trim checks.

YAW DAMPER SYSTEM

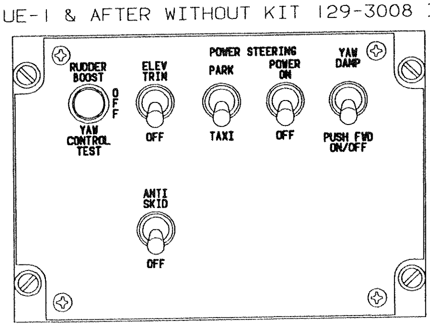
The yaw damp function senses changes in heading (from the compass system) and, utilizing an electric servo, drives the rudder control cables to deflect the rudder and stabilize the yaw axis of the airplane. Heading changes due to turns are sensed using information from the attitude gyro to allow for turn coordination.

If the airplane is equipped with an autopilot, the yaw damp system will be a part of the autopilot.

If an autopilot is not installed in the airplane, yaw damping is provided by an independent yaw damp system. The system is controlled by a combination push-button/annunciator switch placarded YAW DAMP - PUSH ON/OFF. When activated, the green YD ENG annunciator on the switch illuminates. On those airplanes modified by Raytheon Kit No. 129-3008, the push-button switch is replaced by a spring-loaded toggle switch placarded YAW DAMP - PUSH FWD - ON/OFF. The yaw damp system is engaged by pushing the switch forward, then releasing it. The yaw damp system is turned off using the same switch action. With either type of switch, a YD annunciator on each EADI illuminates when the system is turned on. In addition, a YAW annunciator illuminates on the pilot's or copilot's flight director control panel, depending on whether the left or right computer is selected.



(SERIALS UE-1 & AFTER WITHOUT KIT 129-3008 INSTALLED



(SERIALS UE-1 & AFTER WITH KIT 129-3008 INSTALLED)

YAW DAMP SWITCHES UE03C971287 C

RUDDER BOOST

The rudder boost is enabled by setting the pedestal mounted control switch, placarded RUDDER BOOST-OFF-YAW CONTROL TEST, to the RUDDER BOOST position. The system senses engine torque from both engines. When the difference in these torques exceeds a preset level, the electric servo is activated and deflects the rudder, which assists pilot effort. The servo contribution is proportional to the engine torque differential. Trimming of the rudder must be accomplished by the pilot. The rudder boost system is disabled if the RUDDER BOOST switch is OFF and interrupted when the DISC TRIM/YD switch is depressed to the first level.

An amber caution annunciator, RUD BOOST OFF, is provided on the caution/advisory annunciator panel to indicate the rudder boost control switch is in a position other than RUDDER BOOST (on).

If the airplane is equipped with a Raytheon Aircraft-installed autopilot, the autopilot will incorporate the rudder boost capabilities.

INSTRUMENT PANEL

The operation and use of the instruments, lights, switches, and controls located on the instrument panel is explained under the systems descriptions relating to the subject items.

ANNUNCIATOR SYSTEM

The annunciator system consists of a warning annunciator panel (with red readout) centrally located in the glareshield, and a caution/advisory annunciator panel (caution - amber; advisory - green and white) located on the center subpanel. Two red MASTER WARNING flashers 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 amber 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.

Whenever a fault condition covered by the annunciator system occurs, a signal is generated and the appropriate annunciator is illuminated.

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.

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.

Whenever an annunciator-covered fault occurs that requires the pilot's attention but not his immediate reaction, the appropriate amber caution annunciator in the caution/advisory panel illuminates, 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. Subsequently, when any caution annunciator illuminates, the MASTER CAUTION flashers will be activated again. An illuminated caution annunciator on the caution/advisory annunciator panel will remain illuminated until the fault condition is corrected, at which time it will extinguish. The MASTER CAUTION flashers will continue flashing until one of them is depressed.

The caution/advisory annunciator panel also contains the green and white advisory annunciators. There are no master flashers associated with these annunciators. An advisory annunciator can be extinguished only by changing the condition indicated on the illuminated lens.

The warning annunciators, caution annunciators, advisory annunciators, landing gear handle in-transit lights, and amber MASTER CAUTION flashers feature a bright and a dimming mode of illuminated intensity. The dimming mode will be selected automatically whenever all the following conditions are met: A generator is on the line, the OVERHEAD FLOOD LIGHTS switch is in the OFF position, the MASTER PANEL LIGHTS switch is in the ON position, the PILOT FLIGHT LIGHT switch is in the ON position, and the 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 flasher does not have a dimming mode.

The following annunciators are powered by the ANN POWER and ANN IND circuit breakers located on the right circuit breaker panel.

WARNING ANNUNCIATORS:

CABIN ALT HI, CAB DIF HI, LAVATORY SMOKE, L & R OIL PRES LO, L & R ENVIR FAIL, L & R AC BUS, CABIN DOOR, CARGO DOOR

CAUTION ANNUNCIATORS:

STALL HEAT, BATTERY CHARGE, PROP GND SOL, L & R FW VALVE, L & R COL TANK LOW, L & R ENG ICE FAIL, L & R BK DI OVHT, HYD FLUID LOW, ANTI SKID FAIL, L & R PITOT HEAT, XFR VALVE FAIL, PWR STEER FAIL, MAN STEER FAIL, AUTOFTHER OFF, L & R ENVIR OFF, FUEL TRANSFER

Illumination of the ANN PWR SOURCE annunciator indicates a partial or complete loss of power to one or more of the above annunciators.

WARNING ANNUNCIATOR PANEL ILLUSTRATION

L FUEL PRES LO	CABIN ALT HI	**LAVATORY SMOKE	CAB DIFF HI	R FUEL PRES LO
L OIL PRES LO	L ENVIR FAIL	CABIN DOOR	R ENVIR FAIL	R OIL PRES LO
- - - -	L AC BUS	CARGO DOOR	R AC BUS	- - - -
L BL AIR FAIL	*A/P TRIM FAIL	*ARM EMER LITES	*A/P FAIL	R BL AIR FAIL
- - - -	- - - -	- - - -	- - - -	- - - -

** Optional/If Installed*
*** Airplanes Modified by Raytheon Kit No. 129-5031-1.*

WARNING ANNUNCIATOR PANEL DESCRIPTION

NOMENCLATRURE	COLOR	CAUSE FOR ILLUMINATION
L FUEL PRES LO	RED	Fuel pressure low on left side.
CABIN ALT HI	RED	Cabin altitude exceeds 10,000 feet.
**LAVATORY SMOKE	RED	Smoke in the lavatory.
CAB DIFF HI	RED	Cabin pressure differential high.
R FUEL PRES LO	RED	Fuel pressure low on right side.
L OIL PRES LO	RED	Oil pressure failure on left engine.
L ENVIR FAIL	RED	Left environmental air duct over-temp or over-pressure.
CABIN DOOR	RED	Cabin door open or not secure.
R ENVIR FAIL	RED	Right environmental air duct over-temp or over-pressure.
R OIL PRES LO	RED	Oil pressure failure on right engine.
(BLANK)		
L AC BUS	RED	Left AC Bus has inoperative inverter.
CARGO DOOR	RED	Cargo door open or not secure.
R AC BUS	RED	Right AC Bus has inoperative inverter.
(BLANK)		
L BL AIR FAIL	RED	Melted or failed left bleed air failure warning line or system off.
*A/P TRIM FAIL	RED	Improper trim or no trim from autopilot trim command.
*ARM EMER LITES	RED	Emergency light controls disarmed.
*A/P FAIL	RED	A failure has occurred in the selected APC-65 computer.
R BL AIR FAIL	RED	Melted or failed right bleed air failure warning line or system off.

** Optional/If Installed*
*** Airplanes Modified by Raytheon Kit No. 129-5031-1.*

CAUTION/ADVISORY ANNUNCIATOR PANEL ILLUSTRATION

L DC GEN	L FUEL QTY	STALL HEAT	BATTERY CHARGE	**PROP GND SOL	R FUEL QTY	R DC GEN
L FW VALVE	L COL TANK LOW	L GEN TIE OPEN	BAT TIE OPEN	R GEN TIE OPEN	R COL TANK LOW	R FW VALVE
L ENG ICE FAIL	*L BK DI OVHT	HYD FLUID LOW	*ANTI SKID FAIL	ANN PWR SOURCE	*R BK DI OVHT	R ENG ICE FAIL
L FIRE LOOP	L PITOT HEAT	XFR VALVE FAIL	*PWR STEER FAIL	*MAN STEER FAIL	R PITOT HEAT	R FIRE LOOP
L NO AUX XFR	AUTOFTHER OFF	- - - -	*PITCH TRIM OFF	- - - -	AFX DISABLE	R NO AUX XFR
INBD WG DEICE	#YD/RB FAIL	- - - -	TAIL DEICE	- - - -	RUD BOOST OFF	OUTBD WG DEICE
L AUTOFEATHER	L IGNITION ON	- - - -	*PWR STEER ENGA	- - - -	R IGNITION ON	R AUTOFEATHER
L ENG ANTI-ICE	*L BK DEICE ON	- - - -	MAN TIES CLOSE	- - - -	*R BK DEICE ON	R ENG ANTI-ICE
L ENVIR OFF	*RDR PWR ON	FUEL TRANSFER	TAXI LIGHT	- - - -	EXTERNAL POWER	R ENVIR OFF

** Optional/If Installed*
*** Airplanes Modified by Raytheon Kit No. 129-9011-1.*
 # On Airplanes Without an Autopilot.

CAUTION/ADVISORY ANNUNCIATOR PANEL DESCRIPTION

NOMENCLATURE	COLOR	CAUSE FOR ILLUMINATION
L DC GEN	AMBER	Left generator off line.
L FUEL QTY	AMBER	Left fuel quantity below 324 pounds of usable fuel.
STALL HEAT	AMBER	Insufficient current to provide heat on stall warning transducer to prevent icing.
BATTERY CHARGE	AMBER	Excessive charge rate on battery.
**PROP GND SOL	AMBER	Ground: One or both ground idle stop solenoids is in the flight idle position. Flight: One or both ground idle stop solenoids is in the ground idle position.
R FUEL QTY	AMBER	Right fuel quantity below 324 pounds of usable fuel.
R DC GEN	AMBER	Right generator off line.
L FW VALVE	AMBER	Left fuel firewall valve has not reached its selected position.
L COL TANK LOW	AMBER	Left fuel system collector tank below 53 pounds of usable fuel (8 minutes cruise at 400 lbs/hr).
L GEN TIE OPEN	AMBER	Left Gen Bus isolated from center bus.
BAT TIE OPEN	AMBER	Battery isolated from generator buses.
R GEN TIE OPEN	AMBER	Right Gen Bus isolated from center bus.
R COL TANK LOW	AMBER	Right fuel system collector tank below 53 pounds of usable fuel (8 minutes cruise at 400 lbs/hr).
R FW VALVE	AMBER	Right fuel firewall valve has not reached its selected position.
L ENG ICE FAIL	AMBER	Left ice vane malfunction. Ice vane has not attained proper position.
*L BK DI OVHT	AMBER	Melted or failed left brake deice plumbing failure warning line.
HYD FLUID LOW	AMBER	Landing gear hydraulic fluid level low.
*ANTI SKID FAIL	AMBER	Electrical failure or low hydraulic oil pressure in the anti-skid brake system.
ANN PWR SOURCE	AMBER	Partial power loss to some annunciator lights.
*R BK DI OVHT	AMBER	Melted or failed right brake deice plumbing failure warning line.
R ENG ICE FAIL	AMBER	Right ice vane malfunction. Ice vane has not attained proper position.

CAUTION/ADVISORY ANNUNCIATOR PANEL DESCRIPTION (CONTINUED)

NOMENCLATURE	COLOR	CAUSE FOR ILLUMINATION
L FIRE LOOP	AMBER	Left engine fire detection sense loop is open.
L PITOT HEAT	AMBER	Insufficient current to provide heat on left pitot to prevent icing.
XFR VALVE FAIL	AMBER	Fuel cross transfer valve is not fully open or fully closed for 2 seconds or more.
*PWR STEER FAIL	AMBER	Electrical failure or low hydraulic oil pressure in power steering system.
*MAN STEER FAIL	AMBER	Nose gear will not free caster with power steering not engaged.
R PITOT HEAT	AMBER	Insufficient current to provide heat on right pitot to prevent icing.
R FIRE LOOP	AMBER	Right engine fire detector sense loop is open.
L NO AUX XFR	AMBER	No fuel transfer from left Aux to main tank.
AUTOFTHER OFF	AMBER	Autofeather system turned OFF with landing gear extended.
(BLANK)		
*PITCH TRIM OFF	AMBER	Electric trim de-energized by a trim disconnect switch on the control wheel with the system power switch on the pedestal turned ON.
(BLANK)		
AFX DISABLE	AMBER	Autofeather system is not capable of feathering the propellers.
R NO AUX XFR	AMBER	No fuel transfer from right Aux to main tank.
INBD WG DEICE	GREEN	Pressure in left and right inboard wing deice boots sufficient to deice.
#YD/RB FAIL	AMBER	A failure has occurred in the selected FYD-65 computer.
(BLANK)		
TAIL DEICE	GREEN	Pressure in tail deice boots sufficient to deice.
(BLANK)		
RUD BOOST OFF	AMBER	Rudder boost system is turned off.
OUTBD WG DEICE	GREEN	Pressure in left and right outboard wing deice boots sufficient to deice.
L AUTOFEATHER	GREEN	Left autofeather armed with power levers advanced above approximately 89-91% N_1.
L IGNITION ON	GREEN	Left engine ignitor powered.
(BLANK)		
PWR STEER ENGA	GREEN	Power steering operating.
(BLANK)		
R IGNITION ON	GREEN	Right engine ignitor powered.
R AUTOFEATHER	GREEN	Right autofeather armed with power levers advanced above approximately 89-91% N_1.
L ENG ANTI-ICE	GREEN	Left ice vane extended.
*L BK DEICE ON	GREEN	Left brake deice bleed air valve is in the open position.
(BLANK)		
MAN TIES CLOSE	GREEN	Manually closed generator bus ties.
(BLANK)		
*R BK DEICE ON	GREEN	Right brake deice bleed air valve is in the open position.
R ENG ANTI-ICE	GREEN	Right ice vane extended.
L ENVIR OFF	WHITE	Left environmental bleed air valves closed.
*RDR PWR ON	WHITE	Radar is selected to a position other than OFF (on ground).

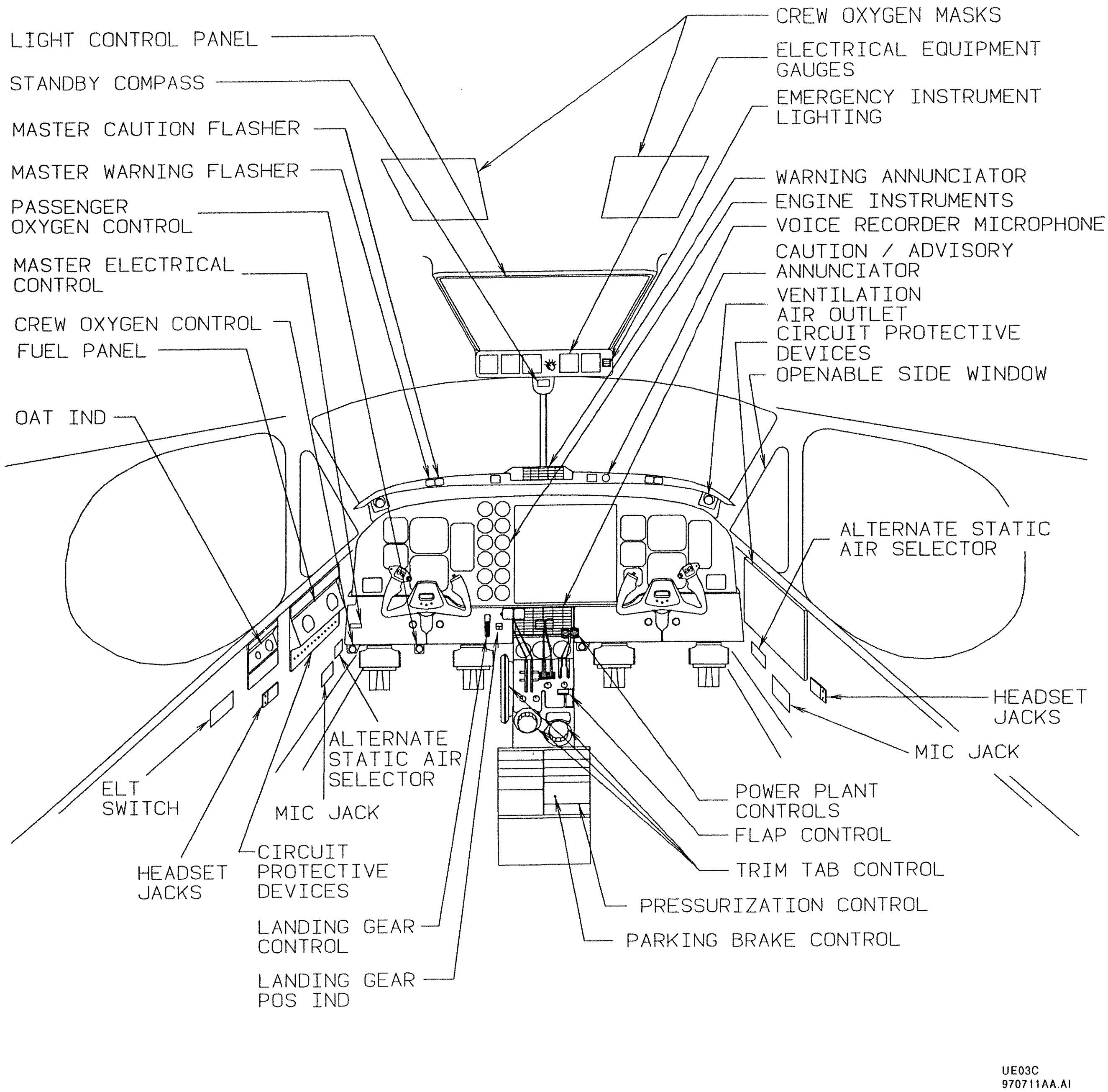
Raytheon Aircraft

CAUTION/ADVISORY ANNUNCIATOR PANEL DESCRIPTION (CONTINUED)

NOMENCLATURE	COLOR	CAUSE FOR ILLUMINATION
FUEL TRANSFER	WHITE	Fuel cross transfer valve is open.
TAXI LIGHT	WHITE	Taxi light on with landing gear up.
(BLANK)		
EXTERNAL POWER	WHITE	External power is plugged into aircraft.
R ENVIR OFF	WHITE	Right environmental bleed air valves closed.

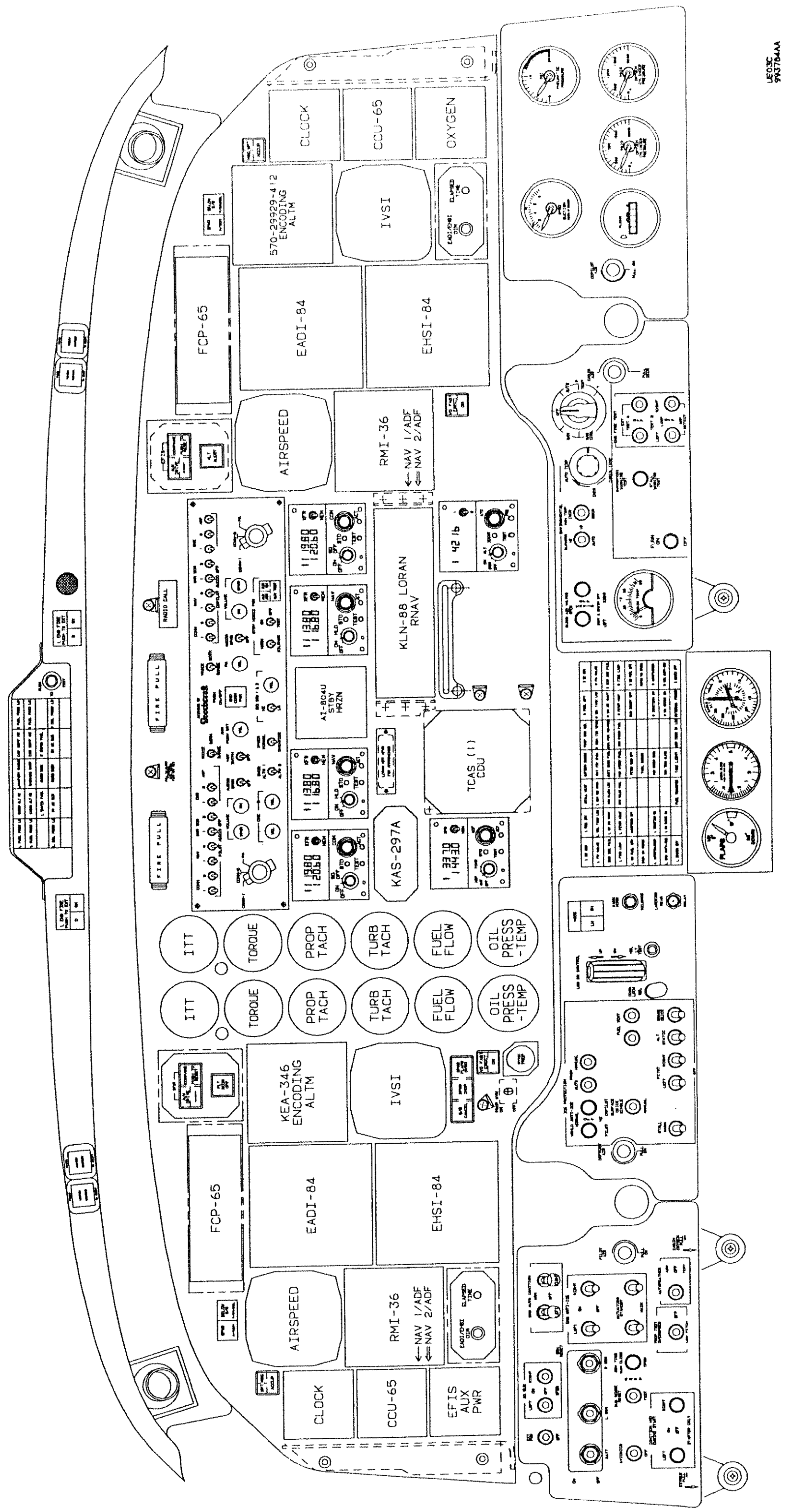
** Optional/If Installed*
*** Airplanes Modified by Raytheon Kit No. 129-9011-1.*
 # On Airplanes Without an Autopilot.

TYPICAL ILLUSTRATIONS

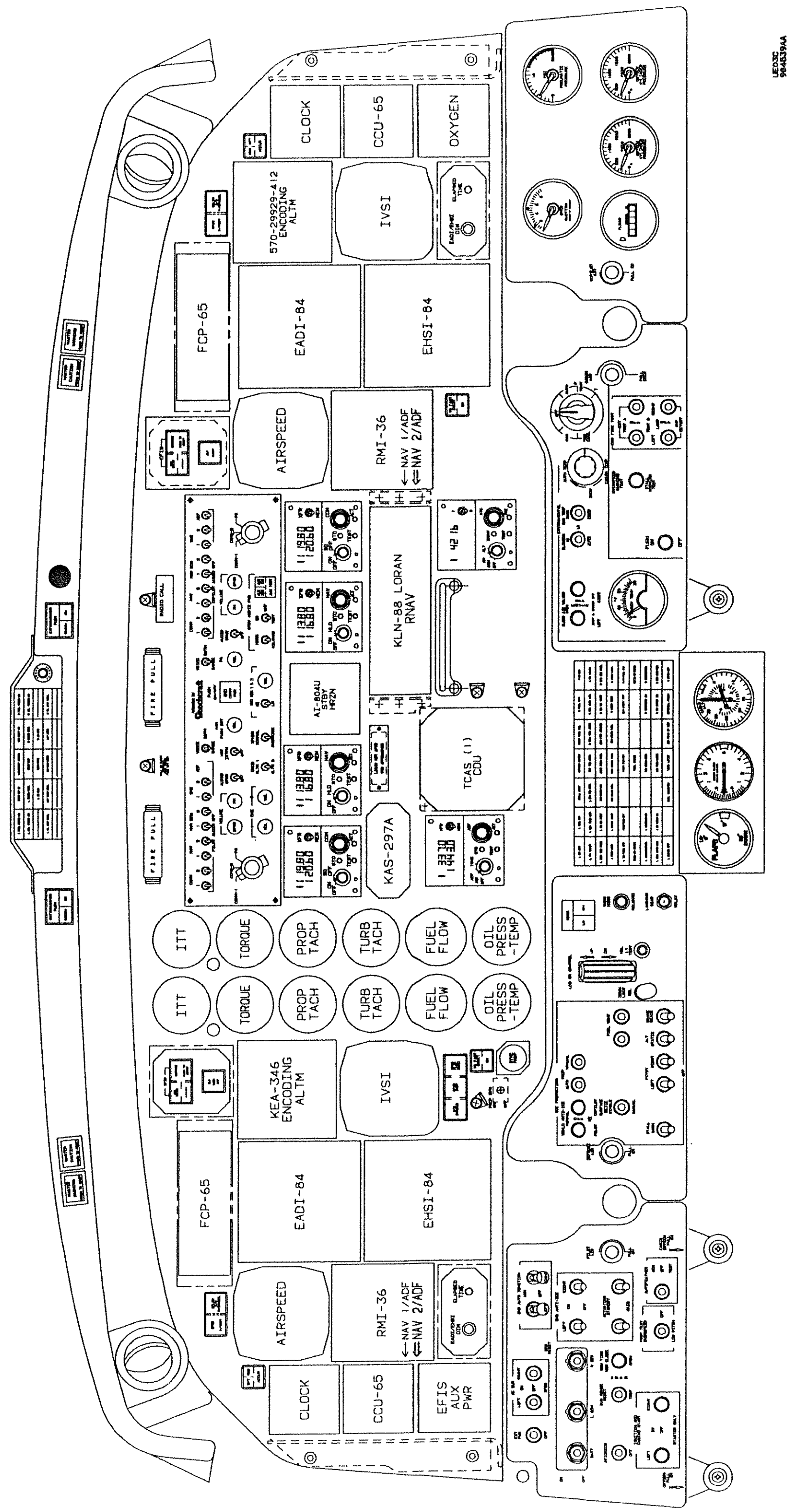


COCKPIT

Raytheon Aircraft

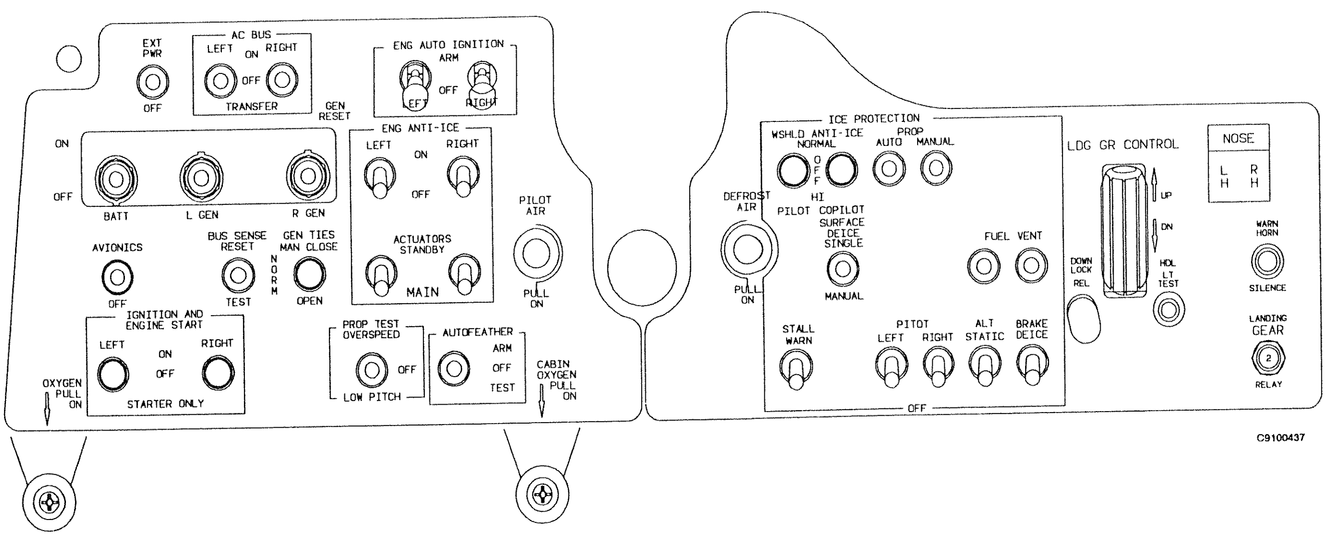


TYPICAL INSTRUMENT PANEL
(UE-1 THRU UE-92)

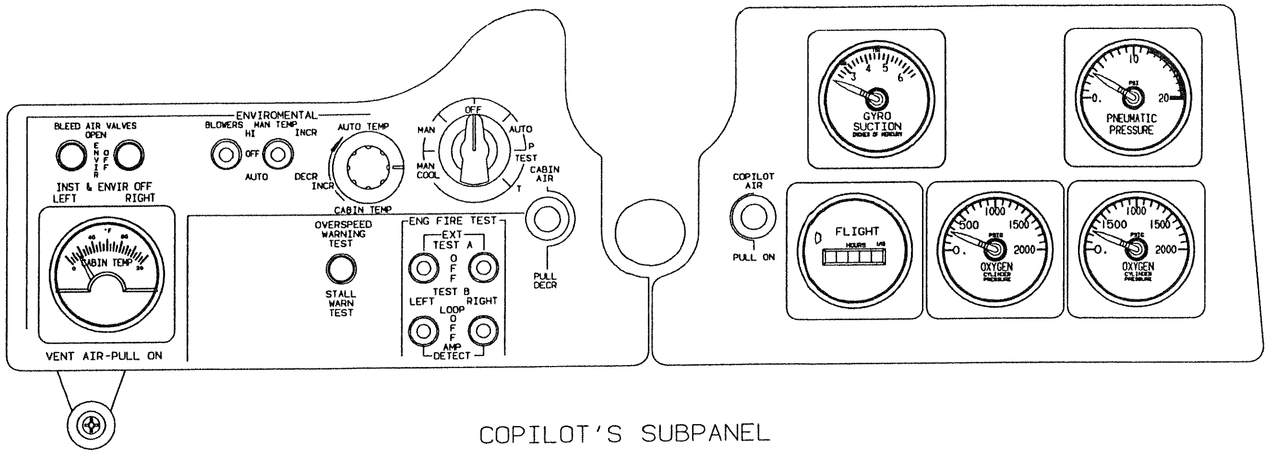


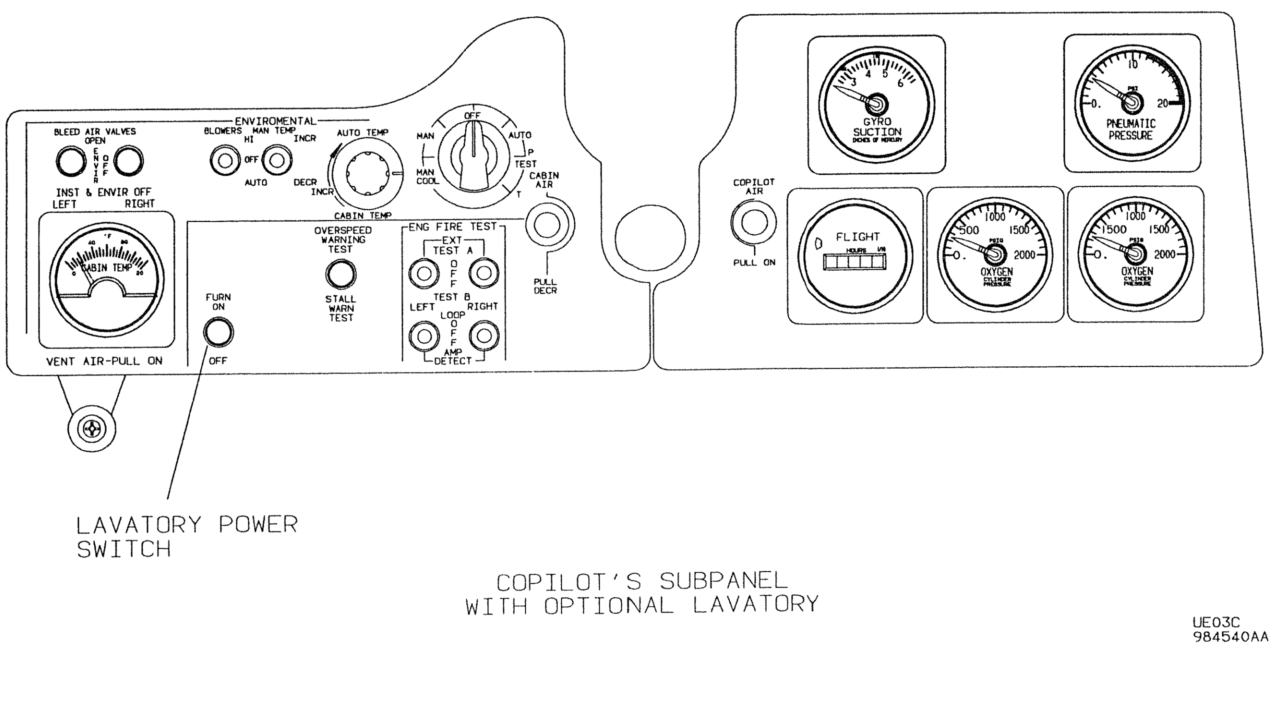
TYPICAL INSTRUMENT PANEL
(UE-93 AND AFTER)

Raytheon Aircraft

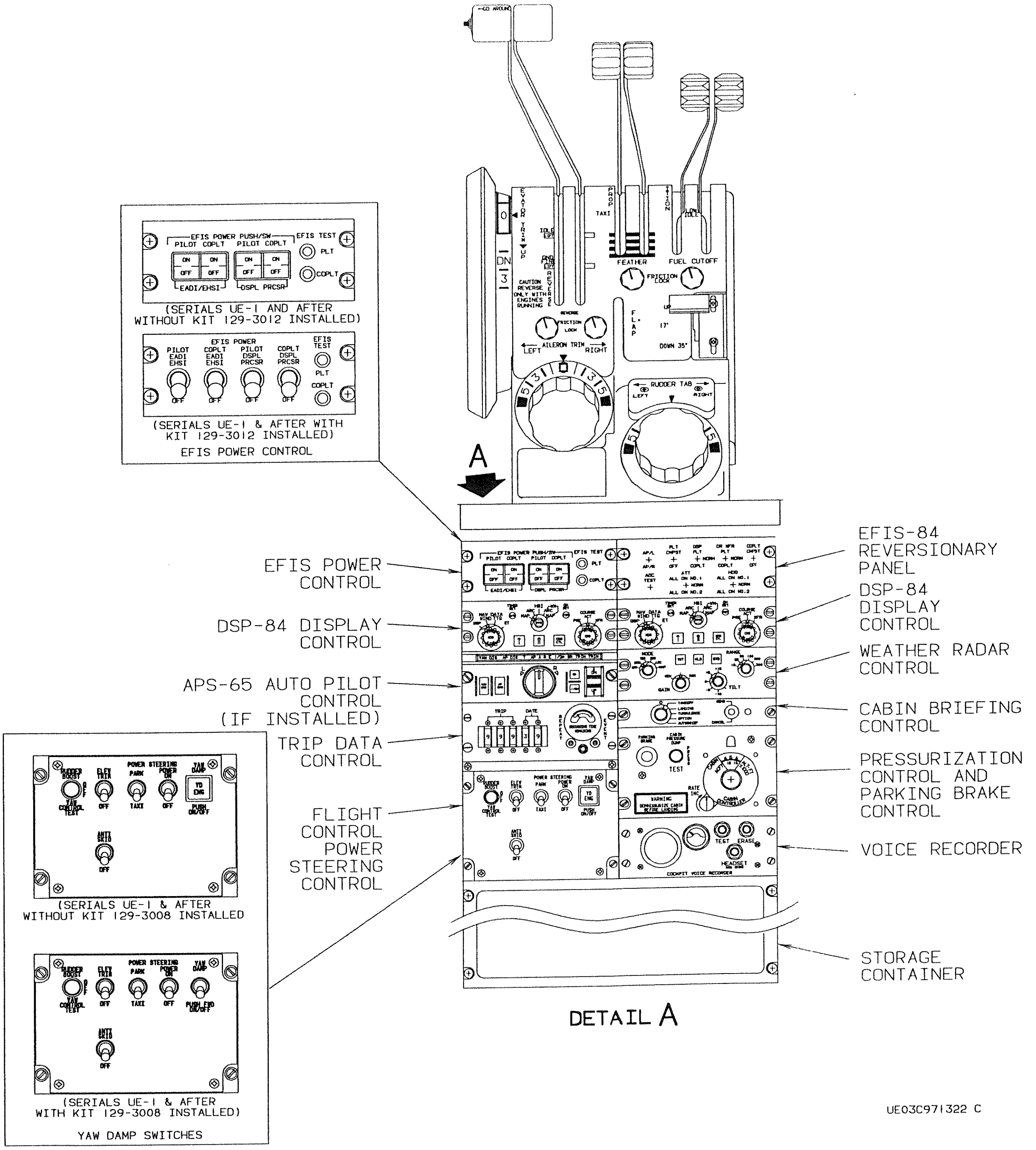


PILOT'S SUBPANEL

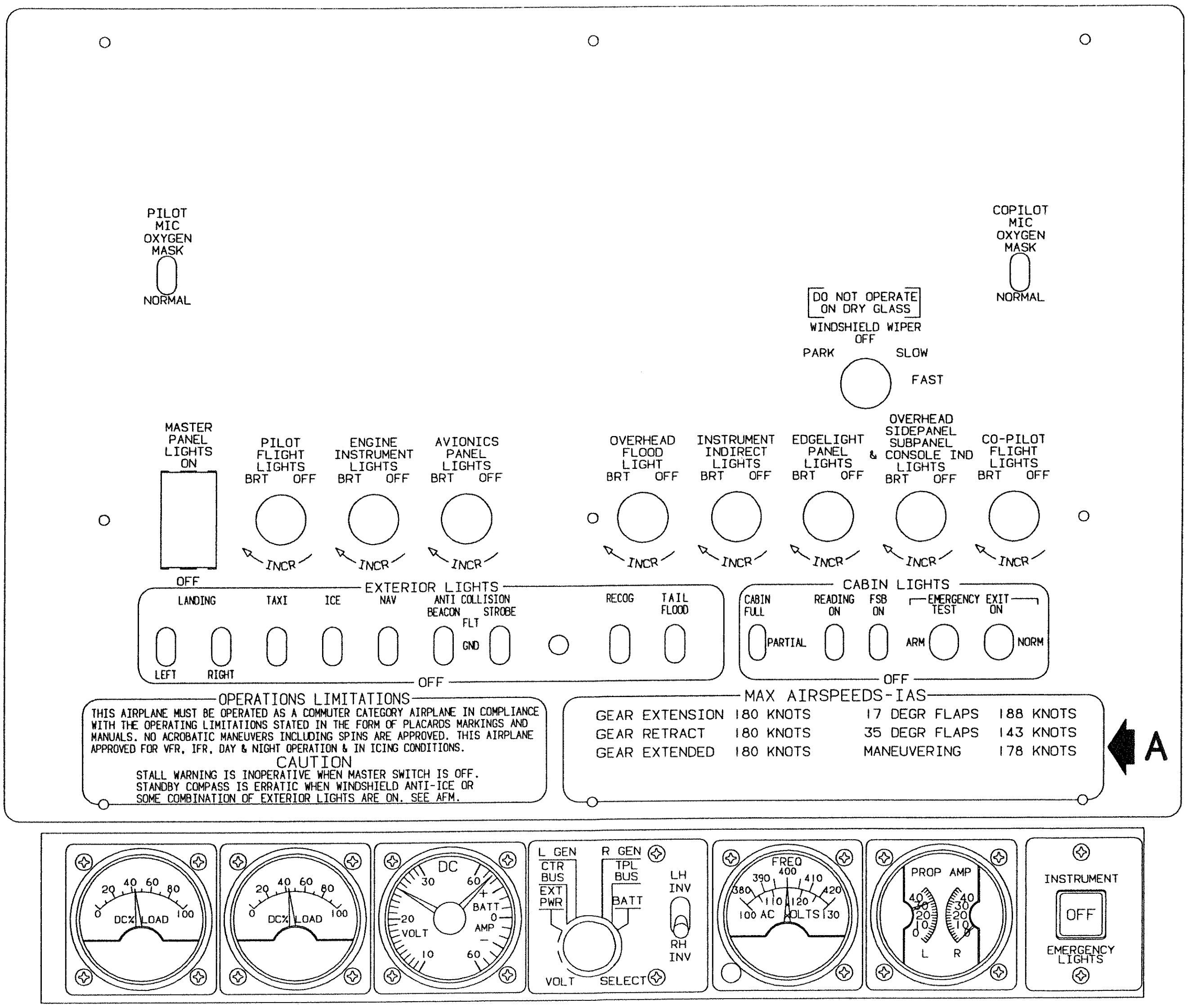




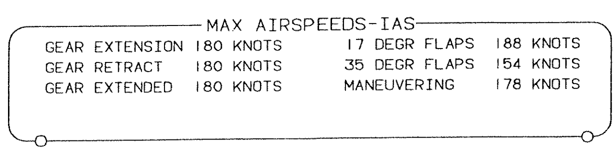
COPILOT'S SUBPANEL


(SERIALS UE-1 AND AFTER WITHOUT KIT 129-3012 INSTALLED)
(SERIALS UE-1 & AFTER WITH KIT 129-3012 INSTALLED)
EFIS POWER CONTROL
A
EFIS POWER CONTROL
DSP-84 DISPLAY CONTROL
APS-65 AUTO PILOT CONTROL (IF INSTALLED)
TRIP DATA CONTROL
FLIGHT CONTROL POWER STEERING CONTROL
EFIS-84 REVERSIONARY PANEL
DSP-84 DISPLAY CONTROL
WEATHER RADAR CONTROL
CABIN BRIEFING CONTROL
PRESSURIZATION CONTROL AND PARKING BRAKE CONTROL
VOICE RECORDER
STORAGE CONTAINER
(SERIALS UE-1 & AFTER WITHOUT KIT 129-3008 INSTALLED
(SERIALS UE-1 & AFTER WITH KIT 129-3008 INSTALLED)
YAW DAMP SWITCHES
DETAIL A
UE03C971322 C

PEDESTAL



(UE-2 THRU UE-78, NOT IN COMPLIANCE WITH S.B. 2512)

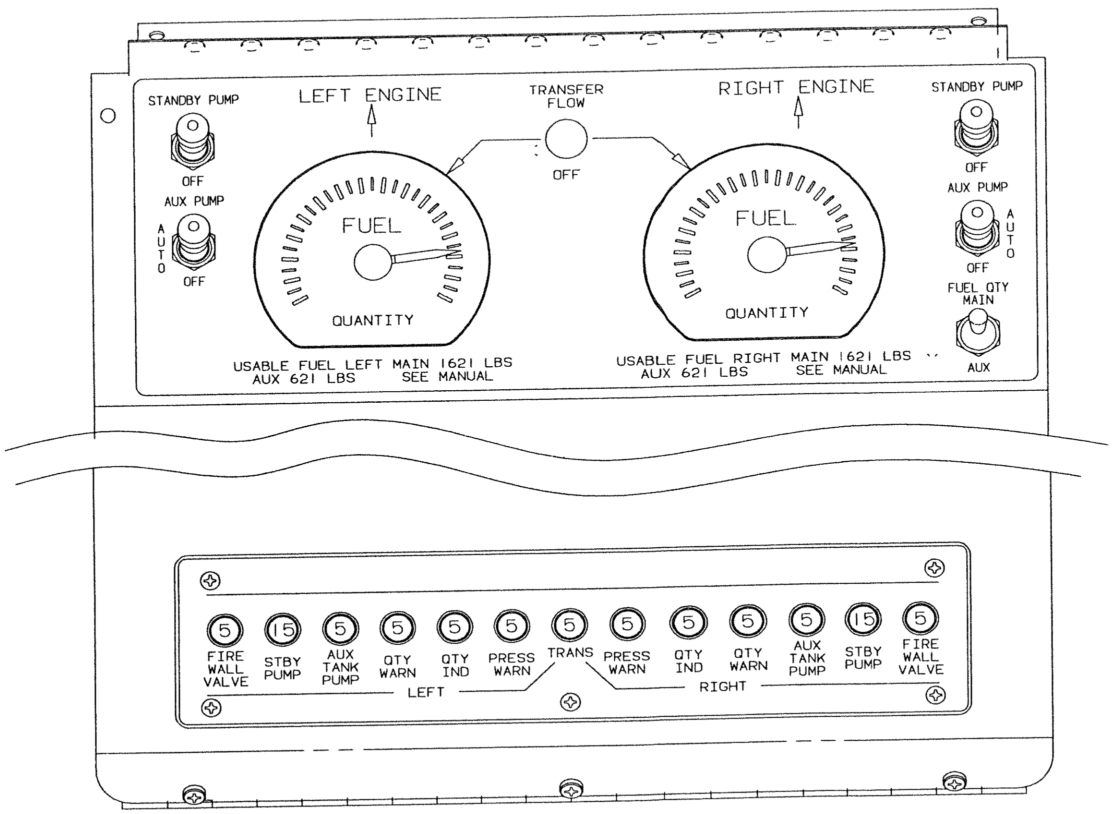


(UE-79 AND AFTER, AND EARLIER AIRPLANES IN COMPLIANCE WITH S.B. 2512)

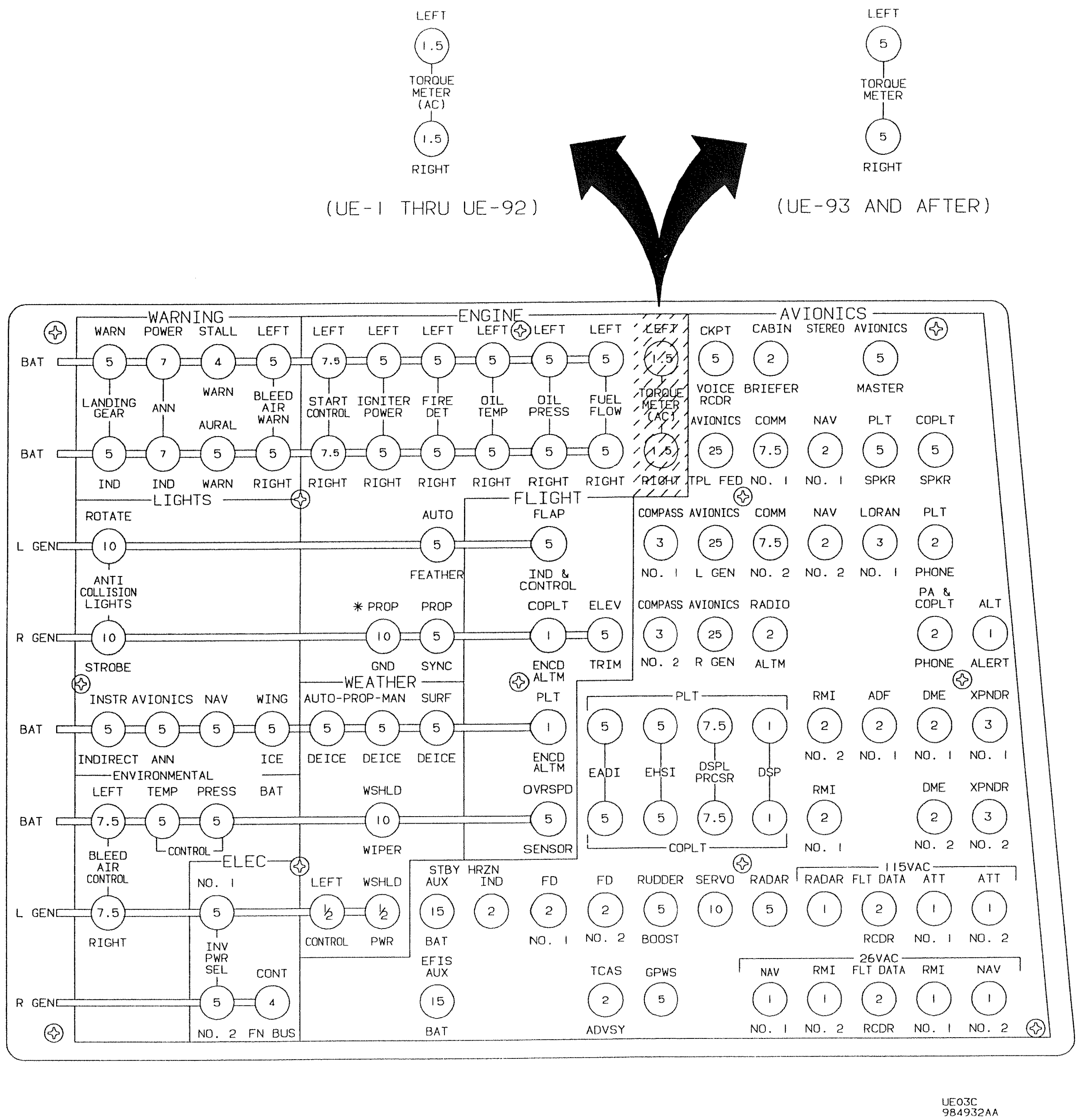
DETAIL A

UE03C
001789AA

OVERHEAD LIGHT CONTROL PANEL



FUEL CONTROL PANEL



* Airplanes modified by Raytheon Kit No. 129-9011-1.

RIGHT CIRCUIT BREAKER PANEL

GROUND CONTROL

Direct linkage from the rudder pedals allows for nose wheel steering or optional hydraulic actuated power steering to be installed.

The minimum wing-tip turning radius, using partial braking action and differential engine power, is 41 feet 2 inches (12.6 meters).

DECOTO POWER STEERING SYSTEM (IF INSTALLED)

The power steering system is electronically controlled and hydraulically actuated, with no mechanical connection between the rudder pedals and the nose gear. The system has three modes of operation: unpowered caster mode; powered TAXI mode (providing ± 15° of nose gear travel); and powered PARK mode (providing ± 55° of nose gear travel). The power steering system should not be used for any purpose other than taxiing or parking the airplane.

The system consists of a hydraulic actuator mounted atop the nose gear strut, a hydraulic pump and associated plumbing in the nose wheel well, an electronic amplifier with associated circuitry located under the co-pilot's seat, and a command potentiometer linked to the pilot's rudder pedal shaft under the floor. Hydraulic fluid is supplied from the brake reservoir in the nose avionics compartment.

The system is operated by turning the Power Steering Switch ON, selecting either the TAXI or PARK mode, using the two-position toggle switch placarded POWER STEERING - PARK - TAXI located on the pedestal, and then pressing the power steering switch on the left power lever. The system will remain on as long as either power lever is in the low-power position. If both power levers are advanced above approximately 89-91% N_1, the power steering system is disengaged, and the mode switch, if in PARK, will automatically move to the TAXI position.

Switching modes in either direction between TAXI and PARK should be accomplished only with the aircraft stopped or while moving at slow speed, since immediate movement of the nose wheel occurs whenever the rudder pedals are not centered (such as while commanding a turn, or taxiing straight in a crosswind, or on a sloping surface).

A green annunciator (PWR STEER ENGA) is provided on the caution/advisory annunciator panel to indicate when the system is engaged and operating.

Two amber annunciators on the caution/advisory panel are provided to caution the pilot that an abnormal power steering condition exists. They are PWR STEER FAIL and MANUAL STEER FAIL.

Steady illumination of the PWR STEER FAIL annunciator indicates system detection of an electrical fault or low hydraulic pressure. Detection of electrical faults will cause the system to automatically deactivate, return to the caster mode, and illuminate the PWR STEER FAIL annunciator. Low hydraulic pressure will also illuminate the PWR STEER FAIL annunciator; the system will continue to operate, although sluggishly. System deactivation and/or illumination of the PWR STEER FAIL annunciator will occur within two seconds of detection of any electrical fault or low hydraulic pressure.

If the system is overpowered by external loads or otherwise fails to respond to pilot commands, the system will disengage, and return to the caster mode. In this event, the PWR STEER ENGA annunciator will extinguish, and the PWR STEER FAIL annunciator will flash.

The PARK mode has a lesser degree of system fault protection than the TAXI mode. For this reason, use of the PARK mode shall be limited to slow speed maneuvering in gate and parking areas that require a tighter turning radius than available in the TAXI mode. The TAXI mode should be used at all other times.

An illuminated MAN STEER FAIL annunciator indicates that the nose gear has not returned to the caster mode after attempted disengagement of the power steering system. In the MAN STEER FAIL condition, the nose wheel will remain in the position existing when power was removed. In this circumstance, steering will only be available with power steering turned on.

FLAPS

Power is delivered from an electric motor to a gearbox, and then through four flexible driveshafts to jackscrews, one of which operates each flap. A safety mechanism is provided to disconnect power to the electric flap motor in the event of a malfunction which would cause any flap to be three to six degrees out of phase with the other flaps.

The flaps are operated by a sliding switch handle on the pedestal just below the condition levers. Flap travel is registered on an electric indicator on top of the pedestal. Detents provide for quick selection of UP, 17°, and 35° positions. The flaps cannot be stopped in an intermediate position.

The flap motor power circuit is protected by a 20-ampere flap motor circuit breaker placarded FLAP MOTOR, located under the floor boards. A 5-ampere circuit breaker for the control circuit (placarded FLAP IND & CONTROL) is located on the Right Circuit Breaker Panel.

Lowering the flaps will produce these results:

ATTITUDE	Nose Up
STALL SPEED	Lowered
AIRSPEED	Reduced
TRIM	Nose-Down Adjustment Required to Maintain Altitude

A flap safety switch, located adjacent to the flap pairs in each wing, disables power to the flap motor when either flap in that pair is more that 3° to 6° out of phase with the adjacent flap. If this occurs, a flap override switch, located at the top of the avionic nose compartment, is provided to allow the flaps to be retracted electrically while the airplane is on the ground. Access to the switch is gained through the left avionic nose door.

LANDING GEAR

The nose and main landing gear assemblies are retracted and extended hydraulically.

The hydraulic landing gear is actuated by a switch placarded LDG GR CONTROL - UP - DN located on the pilot's right subpanel. The landing gear control handle must be pulled out of the detent before it can be moved from either the UP or the DN position. An overload protection circuit protects the system from electrical overload.

A safety switch on the right main gear opens the control circuit when the strut is compressed. This prevents the landing gear handle from being raised when the airplane is on the ground. The hook automatically disengages when the airplane leaves the ground and can be manually overridden by pressing down on the red downlock release button just left of the landing gear control handle in the event of a malfunction of the down-lock solenoid. The landing gear control handle should never be moved out of the DN detent while the airplane is on the ground. If it is, the landing gear warning horn will sound and the red gear-in-transit lights in the landing gear control handle will illuminate (provided the MASTER SWITCH is ON), warning the pilot to return the handle to the DN position.

In flight, as the landing gear moves to the full down position, the downlock switches are actuated and interrupt current to the pump motor. When the red in-transit light in the landing gear control handle extinguishes, the landing gear are in the fully retracted or extended position. Hydraulic system pressure performs the uplock function.

A caution light, placarded HYD FLUID LOW, in the caution/ advisory annunciator panel will illuminate whenever the hydraulic fluid is low in the power pack.

Visual indication of landing gear position is provided by individual green GEAR DOWN indicator lights NOSE - LH - RH on the pilot's right subpanel. The green lights may be checked by pressing the annunciator test switch.

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.

With the flap control handle positioned at UP, or at 17°, and either or both power levers retarded below a preset N_1, the warning horn will sound and the landing gear control handle lights will illuminate. The horn can be silenced by pressing the WARN HORN SILENCE button adjacent to the landing gear control handle; the lights in the landing gear control handle cannot be canceled. The landing gear warning system will be rearmed if either of the power levers is advanced sufficiently.

With the flaps beyond the 17° position, the warning horn and landing gear control handle lights will be activated regardless of the power settings, and neither can be canceled.

LANDING GEAR MANUAL EXTENSION

An alternate extension handle, placarded LANDING GEAR ALTERNATE EXTENSION, is located on the floor on the pilot's side of the pedestal. To engage the system, pull the LANDING GEAR RELAY circuit breaker, located to the right of the landing gear control handle and below the WARN HORN silence button on the pilot's right subpanel, and ensure that the landing gear control handle is in the DN position. Remove the alternate extension handle from the securing clip and pump up and down. While pumping, do not lower the handle below the level of the securing clip during the down stroke as this will allow accumulated hydraulic pressure to bleed off. Continue the pumping action until the three green gear-down annunciators are illuminated, then stow the handle in the securing clip. If one or more gear down annunciators do not illuminate, the alternate handle must not be stowed. Instead, leave it at the top of the up stroke. Continue to pump the handle when conditions permit until the gear is mechanically secured after landing. Refer to LANDING GEAR MANUAL EXTENSION in Section IIIA, ABNORMAL PROCEDURES of the AFM. If any of the following conditions exist, it is likely that an unsafe gear indication is due to an unsafe gear and is not a false indication.

- The inoperative gear down annunciator illuminates when tested.
- The red light in the handle is illuminated.
- The gear warning horn sounds when one or both power levers are retarded below a preset N_1.

After a practice manual extension of the landing gear, the gear may be retracted hydraulically. Refer to LANDING GEAR RETRACTION AFTER PRACTICE MANUAL EXTENSION in Section IV, NORMAL PROCEDURES of the AFM.

BRAKE SYSTEM

The parking brake control is located on the lower pedestal. After the pilot's brakes have been depressed to build up pressure in the brake lines, both valves can be closed simultaneously by pulling out the brake handle. This retains the pressure in the brake lines. The parking brake is released by depressing the pedals briefly to equalize the pressure on both sides of the valve, then pushing in the parking brake handle to open the valves.

HYDRO-AIRE MARK III ANTI-SKID SYSTEM (IF INSTALLED)

The anti-skid system provides a power brake function for normal braking applications and an anti-skid function for maximum braking performance. When the system is turned ON, brake pedal feel is much stiffer and master cylinder pressure is boosted, once it exceeds a preset level, to assist the pilot in braking effort. The anti-skid function controls brake pressure to provide maximum stopping performance while protecting the tires from undue scuffing or blowout. Both functions are available when the anti-skid switch is turned ON and the landing gear is down.

The anti-skid system is self-contained and completely independent of any other system except for electrical power. The system consists of a motor-driven pump, an accumulator, an electronic control computer, a power brake relay valve, and four wheel speed transducers. The pilot control equipment consists of one two-position anti-skid switch located on the console and an annunciator placarded ANTI-SKID FAIL to indicate a failure in the anti-skid system.

Brake control hydraulic pressure is directed to the power brake relay valve from the brake master cylinders. When the system is not activated, the pressure is directed to the brakes without any change. If the anti-skid system is turned on, the power brake or anti-skid mode is available.

The objective of the anti-skid system is to closely approach, but not reach, the brake pressure which would produce a skid. This pressure is not constant, and varies continuously during any braking process. In operation, the control computer continuously monitors wheel speed information that is transmitted by the transducers located on each wheel. The velocity signals from the inboard wheels are averaged and the signals from the outboard wheels are averaged. Both separately averaged signals are fed into two independent and identical skid control channels in the computer. When a skid is imminent, the computer signals the power brake relay valve which, in turn, adjusts brake pressure to obtain optimum braking effectiveness. When skid control is no longer required, the computer reverts to the monitoring mode and braking forces are totally controlled by the pilot.

The system incorporates test functions that continuously monitor the interfaces of the power brake relay valve and wheel speed transducers with the control computer. If electrical faults are detected, the annunciator light will inform the pilot that he does not have anti-skid protection. If the motor-driven pump should fail, the accumulator will provide sufficient fluid pressure for approximately ten brake applications, after which the power brake relay valve will revert the system back to master cylinder control, and the low pressure switch will cause the ANTI-SKID FAIL annunciator to illuminate.

Directional control is maintained with rudder input, nose wheel steering and, when required, differential braking. A combination of these steering techniques may be used.

During periods of medium to maximum braking effort, steering corrections made with conventional differential (or asymmetric) braking techniques may not produce the desired effect.

Conventional differential braking technique calls for increased pressure to the left brake(s) to turn left, or increased pressure to the right brake(s) to turn right.

There is a difference, however, in technique required for the anti-skid equipped airplane that the pilot must be familiar with. Simply stated, when the anti-skid system is actively modulating the brake pressure, the pilot must *reduce* metered pressure on the side *opposite* to which the turn or heading correction is desired; i.e., a right turn initiated by reduction of left brake pressure, or left turn initiated by reduction of right brake pressure. The amount of pressure reduction necessary will vary according to the existing runway friction level and numerous other factors.

Use of the conventional differential braking (steering) is valid only as long as the skid pressure level is not exceeded. Once the skid pressure level has been exceeded, and anti-skid activity starts, no amount of increased differential brake pressure will achieve the desired heading correction. The term "skid pressure level" is the brake pressure at which wheel skid activity occurs.

BRAKE DEICE SYSTEM (IF INSTALLED)

A switch on the pilot's subpanel, placarded BRAKE DEICE, controls the brake deice system. When this switch is activated, both solenoid control valves are opened and annunciators L BK DEICE ON and R BK DEICE ON, on the lower annunciator panel illuminate to advise the system is in operation.

High temperature bleed air from the engine is routed through a solenoid control valve in each main wheel well, through a flexible hose on the main gear strut, and to the distribution manifold around the brake assembly.

The brake deice system may be operated as required on a continuous basis with the landing gear extended, provided the appropriate LIMITATIONS are observed. To avoid excessive wheel well temperatures with the landing gear retracted, a timer is incorporated to automatically terminate system operation 8 to 12 minutes after the landing gear is retracted. The system indicator light should be monitored and the control switch selected OFF when the light extinguishes or if brake deice operation has not automatically terminated within 8 to 12 minutes. The landing gear must be extended before the timer will reset and permit subsequent system activation.

TIRES

The airplane is equipped with dual 22x6.75-10, 10-ply-rated, tubeless tires on each main gear.

The nose gear is equipped with a single 19.5x6.75-8, 10-ply-rated, tubeless tire.

BAGGAGE COMPARTMENT

WARNING

Unless authorized by applicable Department of Transportation Regulations, do not carry HAZARDOUS MATERIAL anywhere in the airplane.

FORWARD CABIN BAGGAGE COMPARTMENT

A seventeen cubic foot (0.5 cubic meters) baggage compartment with a structural capacity of 250 pounds (113 kilograms) is located opposite the forward door, aft of the crew compartment. A rod permits hanging up to 100 pounds (45 kilograms) of coats or garment bags. Small bags may also be stowed in the lower portion of this area.

AFT BAGGAGE COMPARTMENT

The aft baggage or cargo compartment is located at the extreme rear of the cabin and is separated from the passenger compartment by a solid bulkhead. It provides 175 cubic feet (5 cubic meters) of space with a total load capacity, depending on distribution and securing, of 1000 pounds (454 kilograms) forward of the nylon web, and 630 pounds (286 kilograms) between the nylon web and the aft bulkhead. A nylon web is provided for the restraining of loose items.

SEATS, SEATBELTS, AND SHOULDER HARNESSES

SEATS

COCKPIT

The pilot and copilot seats are adjustable fore and aft, as well as vertically. When the release lever under the front inboard corner of the seat is lifted, the seat can be moved forward or aft as required. When the release lever under the front outboard corner of the seat is lifted and no weight is on the seat, the seat will rise in half-inch increments to its highest position. When weight is on the seat and the lever is lifted, the seat will slowly move downward in half-inch increments until the lever is released, or until the seat reaches its lowest point of vertical travel. The armrests pivot at the aft end and can be raised to facilitate entry to and egress from the seats.

CABIN

Cabin seating includes 19 forward-facing seats (18 forward-facing seats on aircraft with optional lavatory installed) with removable seat pads and optional armrests. The two seats installed at F.S. 436 on aircraft equipped with an optional lavatory have an ash tray installed in each of the inboard arm rests.

NOTE

Refer to the FAA Approved Airplane Flight Manual Section VI, WEIGHT AND BALANCE/EQUIPMENT LIST for possible variations in seating arrangements.

SEATBELTS

Every seat in the airplane is equipped with a seatbelt.

SHOULDER HARNESSES

COCKPIT

The shoulder harness installations for the pilot and copilot seats consist of a Y-strap mounted to an inertia reel located in the lower seatback. One strap is worn over each shoulder and terminates with a fitting which inserts into a rotary buckle. The shoulder harness straps and inboard lap belt are released simultaneously by rotating the buckle release 1/8 of a turn in a clockwise direction.

DOORS AND EXITS

AIRSTAIR ENTRANCE DOOR

A nitrogen-filled snubber is installed on the airstair door to reduce the forces required to raise and lower the door. A gage on the snubber indicates its precharge pressure. The pressure should be 1350 - 1450 psi with the door open and 1000 - 1200 psi with the door closed. These pressures are valid for a cylinder temperature of 21°C (70°F).

CAUTION

If the airstair door pressure gage indicates less than 1000 psi, the restraining force required to lower the door will be increased. If pressure is zero, assistance may be required to lower the door normally and prevent it from free-falling.

CAUTION

No more than two people should be on the cabin 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.

To open the door, push and hold the release button adjacent to the door handle while rotating the door handle. As an additional safety measure, a differential-pressure-sensitive diaphragm is incorporated into the release button mechanism.

WARNING

Never attempt to unlock or even 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, the cabin should be depressurized (considering altitude first), and all occupants instructed to remain seated with their seatbelts fastened. After the airplane has made a full-stop landing, a crew member should check the security of the cabin door.

To close the door from outside the airplane, lift the free end of the airstair door and push it up against the door frame as far as possible. Next, rotate the handle clockwise; this will allow the airstair door to move into the closed position. Then rotate the handle counterclockwise as far as it will go. The release button should pop out, and the handle should be pointing aft. Check the security of the airstair door by attempting to rotate the handle clockwise without depressing the release button; the handle should not move.

To close the door from inside the airplane, grasp the aft handrail cable and pull the airstair door up against the door frame. Then grasp the handle and lift it as far as possible, continuing to pull inward on the door, thus allowing the door to move into the closed position. Next, push the handle down 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 lift the handle without depressing the release button; the handle should not move. Next, ensure the safety lock is in position around the diaphragm shaft when the handle is in the locked position. This area is observable by depressing 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 orange stripes on each of the rotary latches (4 on each side of the door) and ensure that each is aligned with the notch on the plate on the door frame. Perform the CABIN/CARGO DOOR ANNUNCIATOR CIRCUITRY CHECK in Section IV, NORMAL PROCEDURES of the AFM, prior to the first flight of the day. If any condition specified in this door-locking procedure is not met, DO NOT TAKE OFF.

CARGO DOOR

A large swing-up cargo door, hinged at the top, provides access for the loading of large items. The door locking mechanism is operated by rotating either the outside handle or the inside lever, both of which move simultaneously. To open the door from the inside, pull on the release ring next to the latch handle, then rotate the handle aft as far as it will go. Push out at the bottom of the door and release door pull-down cable. Once the door is opened a few feet, gas springs take over and raise the door to the fully open position. To open the door from outside the airplane, push and hold the release button adjacent to the door handle while rotating the handle clockwise. Pull out at the bottom of the door until the gas springs take over lifting it to the fully open position.

WARNING

Never attempt to unlock or check the security of the door in flight. If the CARGO DOOR annunciator illuminates in flight, or if the pilot has any reason to suspect that the door may not be securely locked, the cabin should be depressurized (considering altitude first), and all occupants instructed to remain seated with their seatbelts fastened. After the airplane has made a full-stop landing, a crew member should check the security of the door.

To close the cargo door from outside the airplane, pull down the free end of the cargo door and push it against the door frame as far as possible. Next rotate the handle counterclockwise as far as it will go. The release button should pop out and the handle should be pointed aft. Check the security of the cargo door by trying to rotate clockwise without pressing the release button; the handle should not move. Check alignment of orange stripes in the round inspection window located in the lower forward corner of cargo door.

To close the cargo door from inside the airplane, pull down the free end of the door and pull it against the door frame as far as possible. Next rotate the handle forward as far as it will go. The release mechanism will pop in. Check for security of the door by attempting to move the handle aft without pulling the release; the handle should not move. Check alignment of the orange stripes on each of the rotary latches (3 on each side of the door) and the orange alignment stripe in the round inspection window at the lower forward corner of the door.

EMERGENCY EXITS

Two emergency exit doors are located on the right side of the fuselage, at the leading and trailing edges of the wing. A third emergency exit door is located on the left side of the fuselage at the trailing edge of the wing. From the inside, the doors are released with a pull-down handle, placarded EXIT-PULL. From the outside, the door is released with a flush-mounted, pull-out handle. The nonhinged, plug-type door removes completely from the frame into the cabin when the latches are released.

The door can be locked so that it cannot be opened or removed from the outside using the lock pin supplied with the Loose Tools. The door should be locked only for security purposes when the airplane is parked. The lock pin should be removed and stowed prior to flight, allowing removal of the doors from the outside in the event of an emergency. A "Remove Before Flight" flag is attached to the lock pin so that it can be readily seen when it is installed. Removal of the door from the inside is not possible with the lock pin installed.

SUN VISORS

The sun visors are moved and held to the desired location through a combination of friction joints. The amount of friction is adjustable by tightening or loosening the screws at each joint. The visor blade may also be extended by loosening the tension knob located on the main rail. The visor is stowed by rotating the blade forward, up against the flat surface of the headliner.

CONTROL LOCKS

Install the control locks in the following sequence:

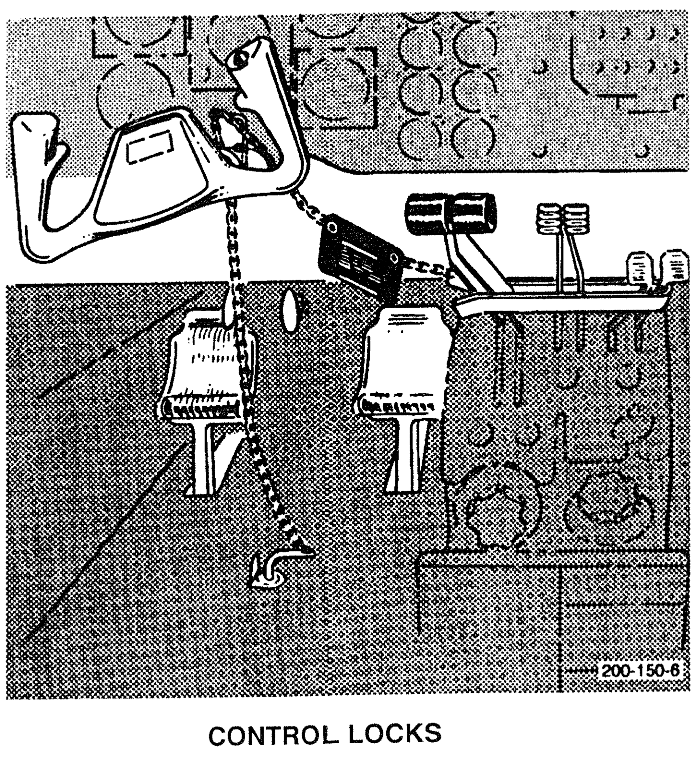
1. Position the U-clamp around the engine control levers.
2. Move the control column full forward and approximately 15° to the left to align the holes, then insert the L-shaped pin that is attached to the middle of the chain (approximately).
3. Insert the L-shaped pin (attached to the end of the chain) through the hole provided in the floor aft of the rudder pedals. The rudder pedals must be centered to align the hole in the rudder bellcrank with the hole in the floor. The pin is then inserted until the flange is resting against the floor. This will prevent any rudder movement.

WARNING

Before starting engines, remove the locks, reversing the above procedure.

CAUTION

Remove the control locks before towing the airplane. If towed with a tug while the rudder lock is installed, serious damage to the steering linkage can result.



CONTROL LOCKS

ENGINES

The Beech 1900D Airliner is powered by two Pratt & Whitney Canada, Inc. PT6A-67D turbo-propeller engines.

PROPULSION SYSTEM CONTROLS

The propulsion system is operated by three sets of controls; the power levers, propeller levers, and condition levers. The power levers serve to control engine power. The condition levers control the flow of fuel at the fuel control outlet and select fuel cut off, low idle and high idle functions. The propeller levers are operated conventionally and control the constant speed propellers through the primary governor.

POWER LEVERS

The power levers provide control of engine power from idle through take-off power by operation of the gas generator (N_1) governor in the fuel control unit. Increasing N_1 rpm results in increased engine power.

PROPELLER LEVERS

Each propeller lever repositions the pilot valve, which results in an increase or decrease of propeller rpm. For propeller feathering, each propeller lever lifts the pilot valve to a position which causes complete dumping of high pressure oil allowing the counterweights and feathering spring to change the pitch. Detents at the rear of lever travel prevent inadvertent movement into the feathering range. Operating range is 1400 to 1700 rpm.

CONDITION LEVERS

The condition levers have three positions; FUEL CUTOFF, LOW IDLE and HIGH IDLE. Each lever controls the idle cut off function of the fuel control unit and limits idle speed at 67% to 69% N_1 for low idle, and 70% to 72% N_1 for high idle.

PROPELLER GROUND FINE OPERATION

The propeller ground fine operation is used to provide optimum deceleration on the ground during landing by taking advantage of the maximum available propeller drag.

Ground fine operation is accomplished by a detent position for the power levers in the pedestal. The power levers must be retarded below the flight idle detent by raising over the detent and retarding the levers to the ground fine detent.

CAUTION

Power levers should not be moved to the ground fine position when the engines are not running as this will cause damage to the reversing tele-flex cable.

PROPELLER REVERSING

When the power levers are lifted over the IDLE detent, they control engine power through the ground fine range, and when lifted over the GND FINE position, they control engine power through the reverse range.

CAUTION

Propeller reversing on unimproved surfaces should be accomplished carefully to prevent propeller erosion from reversing airflow and, in dusty conditions, to prevent obscuring the operator's vision.

Condition levers, when set at HIGH IDLE, keep the engines operating at 70% to 72% N_1 for maximum reversing performance.

CAUTION

Power levers should not be moved into the reversing position when the engines are not running because the reversing system will be damaged.

FRICTION LOCKS

Four friction locks are located on the power quadrant of the pedestal, one for each power lever, one for both prop levers, and one for both condition levers.

ENGINE INSTRUMENTATION

UE-1 THRU UE-92

Engine instruments, located on the left of the center portion of the instrument panel, are grouped according to their function.

- The ITT indicators give an instantaneous reading of engine gas temperature between the compressor turbine and the power turbines. These instruments are self-powered and no circuit breaker is provided.
- The torquemeters give an indication of foot-pounds of torque being applied to the propeller. Power is supplied by either inverter and circuit breakers are located on the right circuit breaker panel.
- The N_2 (propeller) tachometers are read directly in revolutions per minute. Power is supplied by the triple fed bus. The circuit breakers for these instruments are not accessible to the pilot.
- The N_1 (gas generator) tachometers are read in percent of rpm, based on a figure of 37,468 rpm at 100%. Maximum continuous gas generator speed is limited to

39,000 rpm or 104% N_1. Power is supplied by the triple fed bus. The circuit breakers for these instruments are not accessible to the pilot.

- The fuel flow indicators read fuel flow in pounds-per-hour. Power is supplied by the triple fed bus. Circuit breakers for these instruments are located on the right circuit breaker panel.
- The oil temperature/oil pressure indicators display oil temperature (in Degrees Centigrade) and oil pressure (in PSI). Power is supplied by the triple fed bus. Circuit breakers for these instruments are located on the right circuit breaker panel.

UE-93 AND AFTER

Engine instruments, located on the left of the center portion of the instrument panel, are grouped according to their function. These instruments all operate on 28V DC power supplied by the triple fed bus.

- The ITT indicators give an instantaneous reading of engine gas temperature between the compressor turbine and the power turbines. The circuit breakers for these instruments are not accessible to the pilot.
- The torquemeters give an indication of foot-pounds of torque being applied to the propeller. The circuit breakers for these instruments are located on the right circuit breaker panel.
- The N_2 (propeller) tachometers are read directly in revolutions per minute. The circuit breakers for these instruments are not accessible to the pilot.
- The N_1 (gas generator) tachometers are read in percent of rpm, based on a figure of 37,468 rpm at 100%. Maximum continuous gas generator speed is limited to 39,000 rpm or 104% N_1. The circuit breakers for these instruments are not accessible to the pilot.
- The fuel flow indicators read fuel flow in pounds-per-hour. Circuit breakers for these instruments are located on the right circuit breaker panel.
- The oil temperature/oil pressure indicators display oil temperature (in Degrees Centigrade) and oil pressure (in PSI). Circuit breakers for these instruments are located on the right circuit breaker panel.

PROPELLER SYNCHROPHASER

The Propeller Synchrophaser system is an electronic system certified for all operations including takeoff and landing. The system automatically matches the rpm of both propellers and positions them at a preset phase relationship in order to reduce cabin noise.

Before engaging the system, manually set the RPM of each engine to within 10 RPM of each other. When the prop sync switch is turned on, engagement will automatically occur when the relative phase angle of the propellers is within 30° of the preset angle. When the system engages, both propeller speeds are increased by one-half the holding range of the system. To maintain synchronization, the system increases the RPM of the slower propeller and simultaneously reduces the RPM of the faster propeller. The system will never reduce RPM below that selected by the propeller control lever.

To change rpm with the system on, adjust both propeller controls by the same amount. If the synchrophaser is on but does not maintain synchronization, 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, the synchrophaser switch may be turned off, the propellers re-synchronized manually, and the synchrophaser turned back on.

ENGINE LUBRICATION SYSTEM

Engine oil, contained in an integral tank between the engine air intake and the accessory case, cools as well as lubricates the engine. An oil radiator located inside the lower nacelle, keeps the engine oil temperature within the operating limits.

The lubrication system capacity per engine is 3.9 U.S. gallons (14.8 liters). The oil tank capacity is 2.3 gallons (8.7 liters) with 5 quarts measured on the dipstick for adding purposes. Recommended oils are listed in Section V, HANDLING, SERVICE & MAINTENANCE.

STARTING AND IGNITION SYSTEM

Each engine is started by a three-position switch located on the pilot's left subpanel placarded IGNITION AND ENGINE START - LEFT - RIGHT - ON - OFF - STARTER ONLY. Moving the switch upward to the ON position activates both the starter and ignition, and the appropriate L or R IGNITION ON light on the annunciator panel will illuminate. When engine speed has accelerated through 50% N_1 or above on starting, the starter drive action is stopped by placing the switch in the center OFF position.

AUTO IGNITION

The auto ignition system provides automatic ignition to prevent engine loss due to combustion failure. This system is provided to ensure ignition during turbulence and penetration of icing or precipitation conditions.

INDUCTION AIR SYSTEM

The PT6A-67D is a reverse-airflow engine.

ICE PROTECTION

ENGINE AIR INLET

Engine exhaust heat is utilized for heating the engine air inlet lips. Hot exhaust is picked up by a scoop inside the left exhaust stack and plumbed downward to connect into the end of the inlet lip. Exhaust flows through the inside of the lip downward to the right side where it is exhausted out through the right exhaust stack. No shut-off or temperature indicator is necessary for this system.

ENGINE ANTI-ICE SYSTEM

An inertial separation system is built into each engine air duct to prevent moisture particles from entering the engine inlet plenum under icing conditions.

FIRE DETECTION SYSTEM

The fire detection system is designed to provide immediate warning in the event of fire in either engine compartment and consists of the following major components: Two fire zone cables installed in each nacelle and interconnected to form a continuous loop, a control circuit breaker placarded FIRE DET, a single control amplifier installed in the flight compartment on the forward pressure bulkhead and two toggle test switches installed in the copilot's left subpanel.

The fire zone cables are looped around the engine in such a manner as to monitor the most likely areas for fires to occur. The cables are composed of a center conductor surrounded by a thermal sensitive dielectric, all enclosed by a stainless steel sheath. The cable sheath is grounded while the center conductor is connected to the control amplifier.

Should a fire develop in the engine compartment, heating of the fire zone cable dielectric causes the resistance of the dielectric to drop drastically. The control amplifier senses this drop in resistance and at a preset point trips and illuminates the red lights in the appropriate FIRE PULL "T" handle. The FIRE PULL "T" handles are located in the upper center instrument panel below the glareshield.

The airplane is equipped with two toggle test switches placarded ENG FIRE TEST - DETECT, one for the LEFT system and one for the RIGHT system. The switches are three-position switches spring loaded to the center. The switch positions are placarded LOOP - OFF - AMP. When either toggle switch is placed in the LOOP position, the integrity of the appropriate fire zone cable is tested. A good test is indicated by the amber L and R FIRE LOOP annunciators being illuminated. When either toggle switch is placed in the AMP position, the integrity of the circuitry within the control amplifier is tested. A good test is indicated by the red lights in the appropriate FIRE PULL "T" handle being illuminated.

FIRE EXTINGUISHER SYSTEM

UE-1 THRU UE-92

The fire extinguisher control switches used to activate the system are located on the glareshield at each end of the warning annunciator panel. Their power is derived from the hot battery bus. Each push-to-actuate switch incorporates three indicator lenses. The red lens, placarded L (or) R ENG FIRE - PUSH TO EXT, indicates that the fire extinguisher system is armed. The amber lens, placarded D, indicates that the system has been discharged and the supply cylinder is empty. The green lens, placarded OK, is provided only for the test function. To discharge the cartridge, pull the FIRE PULL "T" handle which arms the system. Note that the red light in the push button illuminates. Raise the safety-wired clear plastic cover and press the face of the lens. This is a one-shot system and will be completely expended upon activation. The amber D annunciator will illuminate and remain illuminated until the pyrotechnic cartridge has been replaced. A gage, calibrated in psi, is provided on each supply cylinder for determining the level of charge. The gages should be checked during preflight.

The airplanes are equipped with two toggle test switches placarded ENG FIRE TEST - TEST A - TEST B, one for the LEFT system and one for the RIGHT system installed in the copilot's inboard subpanel. These switches are for the purpose of testing the circuitry of the fire extinguisher pyrotechnic cartridges. The switches are moved to the TEST A position while verifying the illumination of the appropriate amber D and the appropriate green OK annunciators on each fire extinguisher control switch on the glareshield. The switches are moved to the TEST B position while verifying the appropriate green OK annunciator on each fire extinguisher control switch on the glareshield. The toggle switches are spring loaded and will return automatically to the center OFF position.

UE-93 AND AFTER

The fire extinguisher control switches used to activate the system are located on the glareshield at each end of the warning annunciator panel. Their power is derived from the hot battery bus. Each push-to-actuate switch incorporates three indicator lenses. The upper lens, placarded EXTINGUISHER PUSH (in amber), indicates that the fire extinguisher system is armed. The lower left lens, placarded DISCH (in amber), indicates that the system has been discharged and the supply cylinder is empty. The lower right lens, placarded OK (in green), is provided only for the test function. To discharge the cartridge, pull the FIRE PULL "T" handle which arms the system. Note that the amber light in the push button illuminates. Raise the safety-wired clear plastic cover and press the face of the lens. This is a one-shot system and will be completely expended upon activation. The amber DISCH annunciator will illuminate and remain illuminated until the pyrotechnic cartridge has been replaced. A gage, calibrated in psi, is provided on each supply cylinder for determining the level of charge. The gages should be checked during preflight.

The airplanes are equipped with two toggle test switches placarded ENG FIRE TEST - TEST A - TEST B, one for the LEFT system and one for the RIGHT system installed in the copilot's left subpanel. These switches are for the purpose of testing the circuitry of the fire extinguisher pyrotechnic cartridges. The switches are moved to the TEST A position while verifying the illumination of the appropriate amber

DISCH and the appropriate green OK annunciators on each fire extinguisher control switch on the glareshield. The switches are moved to the TEST B position while verifying the appropriate green OK annunciator on each fire extinguisher control switch on the glareshield. The toggle switches are spring loaded and will return automatically to the center OFF position.

PROPELLER SYSTEM

DESCRIPTION

Each engine is equipped with a conventional four-blade, full-feathering, constant-speed, counter-weighted, reversing, variable-pitch propeller of composite construction mounted on the output shaft of the reduction gearbox. The propeller pitch and speed are controlled by engine oil pressure, through single-action, engine-driven propeller governors. Centrifugal counterweights, assisted by a feathering spring, move the blades toward the low rpm (high pitch) position and into the feathered position. Governor boosted engine oil pressure moves the propeller to the high rpm (low pitch) hydraulic stop and reverse position. The propellers have no low rpm (high pitch) stops; this allows the blades to feather after engine shutdown.

Propeller tie-down boots are provided for use on the moored airplane to prevent windmilling at zero oil pressure.

LOW PITCH STOPS

The propeller control systems are equipped with flight idle and ground idle low pitch stops. The flight idle low pitch stop is a mechanically actuated hydraulic stop. The ground idle low pitch stop, which produces a lesser blade angle than the flight idle low pitch stop, is an electrically actuated hydraulic stop controlled by the ground idle low pitch stop solenoid. This solenoid resets the governor beta valve to produce the desired blade angle. During ground operation, power is supplied to the solenoids from the PROP GND SOL circuit breaker through a relay controlled by the right squat switch. Power can also be supplied to the solenoids by lifting the power levers when they are positioned at flight idle. This action bypasses the right squat switch and ensures that the solenoids are activated during the landing roll for situations when the activation of the right squat switch may be delayed, such as in crosswind landings. Power is removed from the ground idle low pitch stop solenoid when the right squat switch is activated at liftoff, allowing the propeller low pitch stop to revert to the flight idle setting. If a failure occurs in the low pitch stop solenoid system, the PROP GND SOL annunciator, for airplanes modified by Raytheon Kit No. 129-9011-1, will illuminate. This annunciator monitors the position of the solenoid plungers during ground and flight conditions. Illumination of the annunciator indicates a failure in the system, which could include any one of the following:

Ground Operations	One or both solenoids are in the unpowered (flight) condition, either due to loss of power or a physical failure such as a sticking solenoid.
Flight Operations	One or both solenoids are in the powered (ground) condition, either due to a failure which allows power to the solenoid, or a physical failure such as a sticking solenoid.

In both cases there is an 8 ± 1-second delay from the time the failure occurs until the annunciator illuminates.

If the failure occurs in flight, the pitch of the associated propeller will continue to decrease from the flight idle stop to the ground idle stop when the propeller is no longer controlled by the governor, such as in the landing flare, causing an increase in disking drag and a yawing moment if only one propeller is affected. Power can normally be removed from the solenoids by pulling the PROP GND SOL circuit breaker on the copilot's circuit breaker panel. If the annunciator does not extinguish, a stuck solenoid is indicated, and a modified approach should be flown as described in Section IIIA, ABNORMAL PROCEDURES, in the AFM. This will ensure that the propellers do not reach the low pitch stop until the airplane has landed.

PROPELLER GOVERNORS

Two governors, a constant speed governor, and an overspeed governor, control the propeller rpm. The constant speed governor, mounted on top of the gear reduction housing, controls the propeller through its entire range. The propeller control lever controls the rpm of the propeller by means of this governor. If the constant speed governor should malfunction by requesting more than 1700 rpm, the overspeed governor cuts in at 1802 rpm and dumps oil from the propeller to keep the rpm from exceeding approximately 1802 rpm.

AUTO-FEATHER SYSTEM

The automatic feathering system provides a means of immediately dumping oil from the propeller servo to enable the feathering spring and counterweights to start the feathering action of the blades in the event of an engine failure.

MINIMIZING FOREIGN OBJECT DAMAGE (FOD)

Engines and propellers are more susceptible to foreign object damage (FOD) under certain conditions. The following operations should be avoided or minimized:

1. Engine/propeller operation at maximum power while the airplane is stationary.
2. Backing the airplane using propeller reverse.
3. Airplane operation in dust storms or sand storms.
4. Operation in icing conditions without extending the ice vanes.

5. Starting the engines in feather (except for external power starts).
6. Taxiing close to preceding airplanes. (Maintain maximum practical spacing.)

FUEL SYSTEM

The fuel system consists of two integral fuel tanks in each wing. A main tank extends from nacelle to wing tip and an auxiliary tank is located between nacelle and fuselage.

Each main tank holds a maximum of 240.5 gallons (910 liters) of usable fuel and is filled from a port located near the wing tip. A collector tank is contained within each main tank immediately outboard of the nacelle. Each collector tank is filled from the main tank by gravity feed and two jet transfer pumps which maintain the fuel level in the collector tank at normal flight attitudes.

Each auxiliary tank holds a maximum of 92.2 gallons (349 liters) of usable fuel and is filled from its own filler port located immediately inboard of the nacelle.

The main and auxiliary fuel systems are vented through a recessed ram vent coupled to a protruding ram vent on the underside of the wing tip. The recessed vent is naturally ice resistant while the protruding vent is heated to prevent icing.

An anti-siphon valve is installed in each filler port to prevent loss of fuel in the event of improper securing or loss of the filler cap.

FUEL PUMPS

An engine driven fuel pump (high pressure) is mounted on the accessory case in conjunction with the fuel control unit. A primary boost pump (low pressure) is also engine driven and is mounted on a drive pad at the aft accessory section of the engine.

An electrically driven standby pump (low pressure), located in the bottom of the collector tank sump, serves as a backup for the engine driven boost pump. This pump also provides cross transfer fuel flow from the side on which it is located to the collector tank in the opposite wing.

Each auxiliary fuel tank contains an electrically driven fuel pump which transfers fuel to the collector tank in the same wing.

FUEL DRAINS

During each preflight, the fuel sumps on the tanks, pumps and filters should be drained to check for fuel contamination. There are five drains in each wing. They are located as follows:

DRAINS	LOCATION
Auxiliary Tank (1)	Underside of wing, inboard of nacelle
Collector Tank (1)	Outboard side of nacelle
Main Tank (2)	Underside of wing, outboard of nacelle
Fuel Filter (1)	Underside of wing, outboard of nacelle

FUEL PURGE SYSTEM

Engine compressor discharge air (P3 air) pressurizes a small purge tank. During engine shutdown, fuel manifold pressure subsides allowing the engine fuel manifold poppet valve to open. The purge tank pressure forces fuel out of the engine fuel manifold lines, through the nozzles, and into the combustion chamber. As the fuel is burned, a momentary surge in (N_1) gas generator rpm should be observed. The entire operation is automatic and requires no input from the crew.

FUEL GAGING SYSTEM

The airplane is equipped with a capacitance type fuel quantity indication system. A maximum indication error of 3% full scale may be encountered in the system. The system is designed for the use of Jet A, Jet A-1, JP-5 and JP-8 aviation kerosene, and compensates for changes in fuel density due to temperature changes. If other fuels are used, the system will not indicate correctly. Refer to OTHER NORMAL PROCEDURES in Section IV, NORMAL PROCEDURES of the AFM for instructions when using Jet B, JP-4, or aviation gasoline.

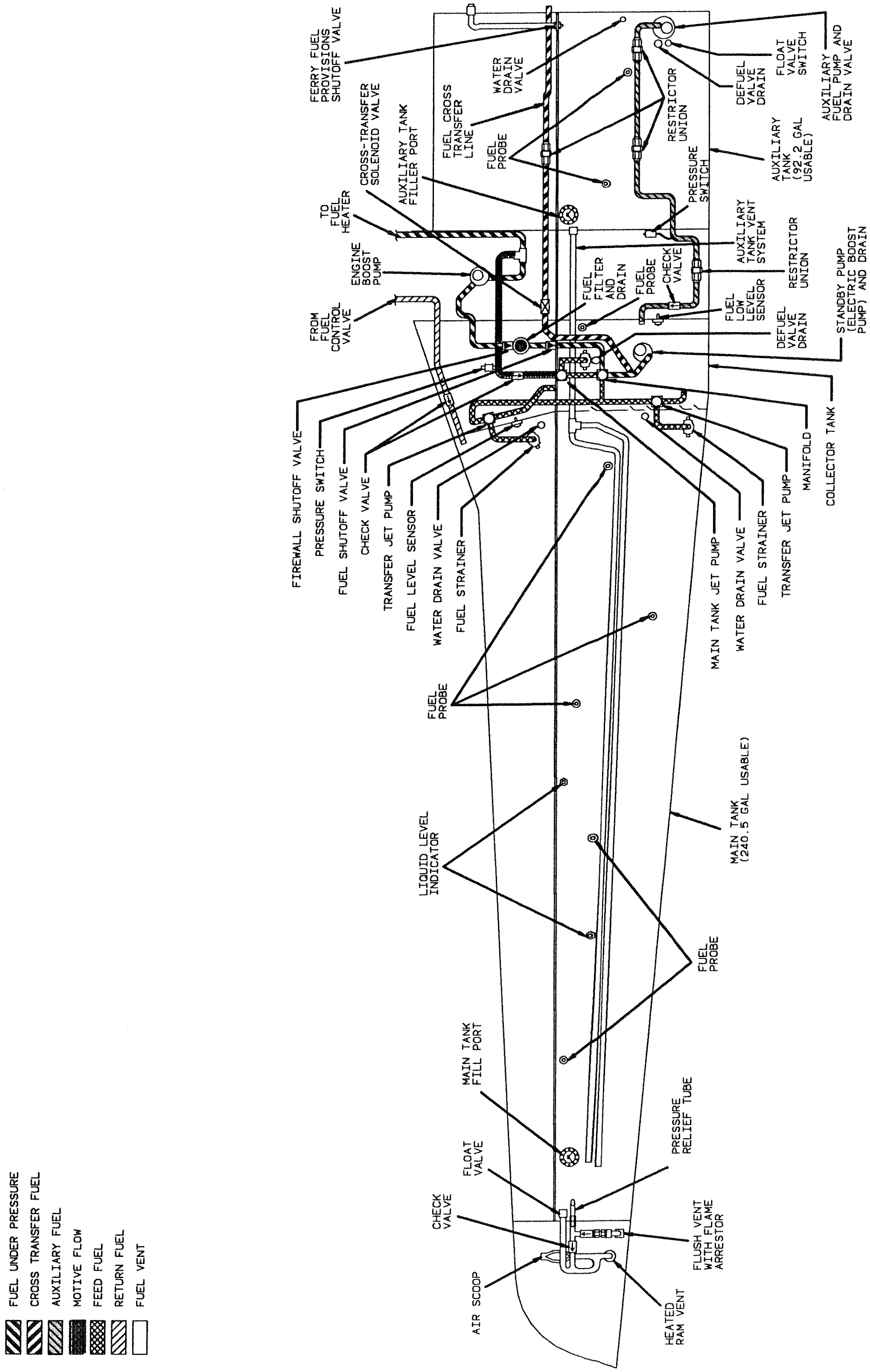
The left and right fuel quantity indicators on the fuel control panel indicate the amount of fuel remaining in their respective fuel tanks. Deflecting the spring-loaded Fuel Quantity switch on the fuel control panel to the AUX position will cause the indicators to indicate the fuel quantity in the auxiliary tanks. The indicators are marked in pounds.

Two visual fuel quantity gauges are located on the lower surface of each wing. The gages, when not submerged in fuel, are red with a black dot; when they are submerged they are totally black. The outboard probe, when red, indicates less than 1150 pounds of fuel, the inboard probe, when red, indicates less than 745 pounds of fuel. These gages are not certified and can not be used as an approved means of determining fuel quantity.

NORMAL FUEL SYSTEM OPERATION

During normal operation, fuel flow to each engine is provided by the engine driven fuel pumps (high pressure and boost) which draw fuel from the collector tank in the same wing. The collector tank draws from it's respective main tank unless fuel is being supplied from the auxiliary tank.

Any fuel contained in the auxiliary tanks is to be used prior to using fuel from the main tanks. This is accomplished by positioning the AUX PUMP switches on the fuel control panel to the AUTO position. This activates the electric transfer pump in each auxiliary tank and pumps fuel to the collector tank of the same wing. Fuel will continue to be transferred until the auxiliary tank is empty, at which time the pump will automatically shut off. In the event of a transfer system failure, it is permissible to temporarily operate the airplane with fuel in the auxiliary tanks providing fuel imbalance and fuel reserve requirements can be met.



FUEL SYSTEM SCHEMATIC

ENGINE DRIVEN PUMP FAILURE

Failure of the high pressure engine driven fuel pump results in an immediate flame out.

In the event of a primary boost pump failure, the respective red L or R FUEL PRES LO annunciator in the warning annunciator panel will illuminate. This light illuminates when pressure decreases below 5.0 ± 1 psi. The light will be extinguished by switching on the standby fuel pump on that side, thus increasing pressure above 10.0 psi. The standby fuel pump switches are located on the fuel control panel. Electrical power to operate the standby pumps is supplied from two independent sources. Both power sources are provided through the Center Bus and protected by 15 ampere circuit breakers located below the fuel control panel. Power is only available when the master switch is turned on.

CAUTION

Operation with the fuel pressure light on is limited to 10 hours between overhaul, or replacement, of the engine driven fuel pump.

AUXILIARY FUEL TRANSFER SYSTEM FAILURE

Malfunction of the auxiliary fuel transfer system (aux tank to collector tank) can be detected by illuminating the amber L or R NO AUX XFR annunciator in the caution/advisory annunciator panel.

If this occurs when auxiliary tank fuel has been depleted, failure of the auxiliary transfer pump's automatic shut-off feature is indicated. The pump can be manually switched off by placing the respective Aux Pump switch on the fuel control panel in the OFF position.

If the NO AUX XFR annunciator illuminates while there is fuel remaining in the tank, the respective Aux Pump switch should be switched to the ON position, thereby overriding the automatic shut-off feature of the auxiliary transfer pump. If this action restores the transfer of fuel, the NO AUX XFR annunciator will be extinguished until the auxiliary tank is depleted. Upon depletion of fuel the NO AUX XFR annunciator will again illuminate, at which time the respective Aux Pump switch should be placed in the OFF position.

If selecting Aux Pump ON does not restore the transfer of fuel (e.g., extinguish the NO AUX XFR annunciator), the fuel remaining in the auxiliary tank will not be available for flight. (Secondary jet pumps are not installed in the auxiliary tanks, nor is gravity feed available). In this event the respective aux transfer pump should be switched off and the flight should be re-planned as appropriate based on the non-availability of the remaining fuel in the affected auxiliary tank. Proper fuel management should be accomplished in order not to exceed the maximum allowable weight imbalance between left and right fuel systems.

CROSS TRANSFER

During single engine operation, it may become necessary to supply fuel to the operative engine from the fuel system on the opposite side. The simplified cross transfer system is placarded for fuel selection with a diagram on the upper fuel control panel. Place the STANDBY PUMP switches in the OFF position when cross transferring. A lever lock switch, placarded TRANSFER FLOW - OFF, is moved from the center OFF position to the left or to the right, depending on direction of flow. This opens the cross transfer valve, energizing the standby pump on the side from which cross transfer is desired. The amber XFR VALVE FAIL annunciator illuminates anytime the cross transfer valve does not match switch position. When the cross transfer mode is energized, a white FUEL TRANSFER annunciator on the caution/advisory panel will illuminate.

FIREWALL SHUTOFF

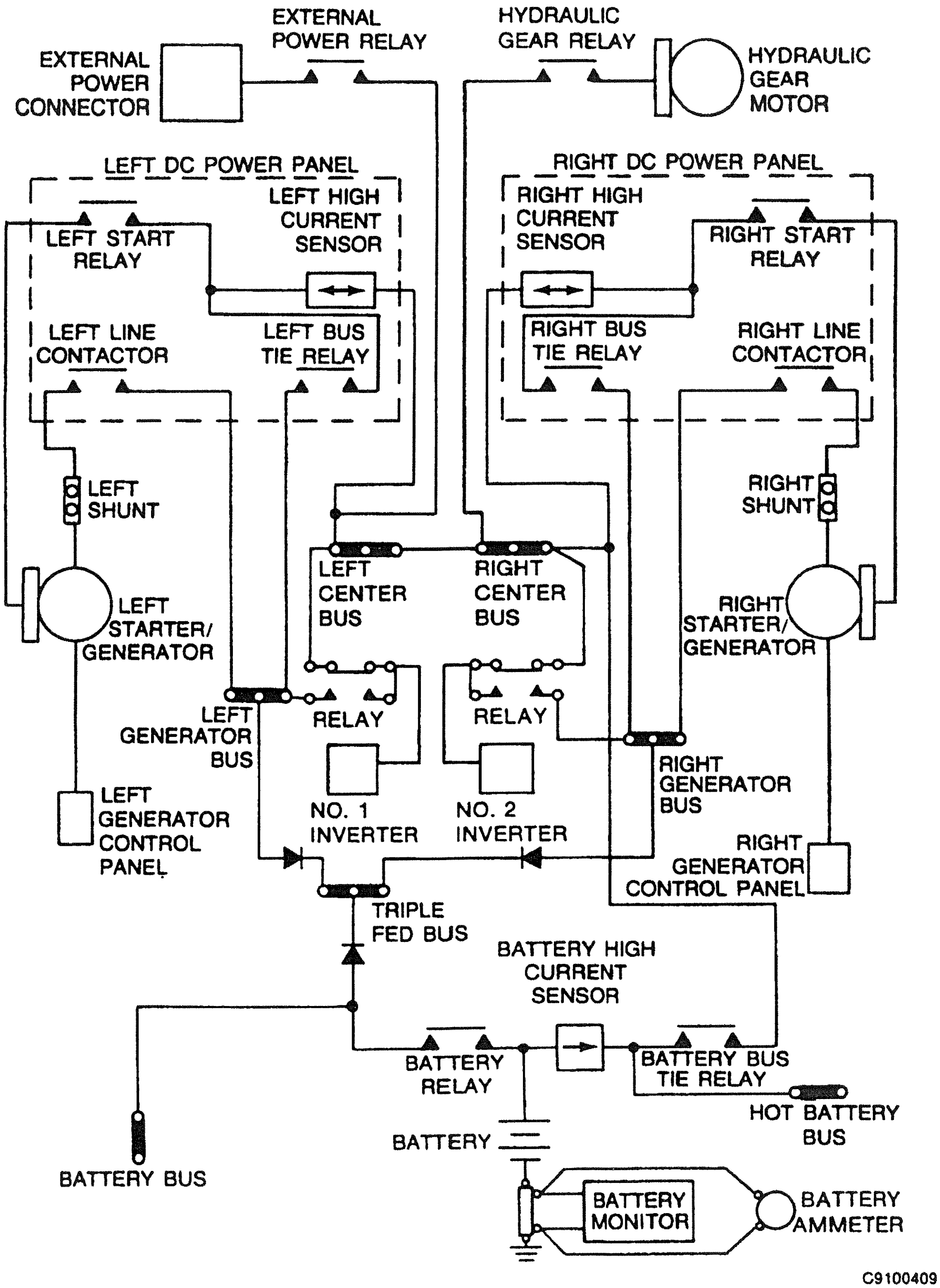
The system incorporates two firewall fuel valves, one on each engine. These valves, which are normally opened, are controlled by two FIRE PULL handles located on the upper center instrument panel. When the respective FIRE PULL handle is pulled, it not only closes the firewall shutoff valve, but also arms the respective fire extinguisher.

ELECTRICAL SYSTEM

The Beech 1900D Airliner electrical system is a negatively grounded 28-vdc (nominal) system. DC electrical power is provided by one 23-ampere-hour or optional 36-ampere-hour air-cooled battery.

A hot battery bus is provided for operation of equipment such as cockpit emergency lighting, cabin entry and threshold lighting circuits and fire extinguishing system.

The electrical system is made up of two generator buses, two center buses, a battery bus and a triple fed bus. The individual buses are powered by their respective sources which are the battery and the left and right generators. In normal operation, the buses are automatically tied into a single loop with all sources supplying power through individual protective devices. The triple fed bus is fed by means of a 60-amp current limiter and blocking diodes from each source. The generators supply power to their respective buses by means of a 300-amp shunt through a line contactor to the respective generator bus. The cross tie line between the two generators automatically ties the two systems together when either generator is brought on the line by means of two contactors controlled by the bus tie control unit.



POWER DISTRIBUTION SCHEMATIC

CIRCUIT BREAKER ELECTRICAL CONNECTION LOCATION

L GENERATOR BUS	R GENERATOR BUS
FLAP MOTOR	PITCH TRIM (OPT)
FLAP CONTROL & INDICATOR	PROP SYNCHROPHASER
NOSE WHEEL STEERING CONTROL (OPT)	SUBPANEL, OVHD & INSTR LIGHTS
AUTOFEATHER	CABIN INDIRECT (AISLE) LIGHTS - FULL
FLIGHT INSTRUMENT LIGHTS	READING LIGHTS
ENGINE & AVIONICS INSTR LIGHTS	EDGE LIGHT PANELS
R BLEED AIR CONTROL	RIGHT GEN BUS TIE POWER
VENT BLOWER CONTROL	L ENG ANTI-ICE CONTROL (STANDBY)
L FUEL VENT HEAT	STALL WARNING HEAT
BRAKE DEICE (OPT)	R ENG ANTI-ICE CONTROL (STANDBY)
L ENG ANTI-ICE CONTROL (MAIN)	R FUEL VENT HEAT
R ENG ANTI-ICE CONTROL (MAIN)	AIR CONDITIONER CLUTCH
L GEN BUS TIE POWER	ALTERNATE STATIC HEAT
L LANDING LIGHT	R LANDING LIGHTS
TAIL FLOOD LIGHT (OPT)	RECOGNITION LIGHTS
L ENGINE AUTOMATIC PROP DEICE	R ENGINE AUTOMATIC PROP DEICE
R FIREWALL VALVE	R WINDSHIELD ANTI-ICE
INVERTER NO. 1	L FIREWALL VALVE
POWER STEERING PUMP MOTOR (OPT)	INVERTER NO. 2
L GEN AVIONICS BUS	R GEN AVIONICS BUS
FWD VENT BLOWER	AFT VENT BLOWER
ANTI-SKID PUMP MOTOR (OPT)	PROP GOV TEST
L WINDSHIELD ANTI-ICE	R AUX FUEL TRANSFER PUMP
FURNISHING CONTROL (OPT)	MOD/FURNISHINGS POWER
L AUX FUEL TRANSFER PUMP	EMERGENCY EXIT LIGHTS
ANTI-SKID (OPT)	ANTI-COLLISION LIGHT - STROBE (OPT)
ANTI-COLLISION LIGHT - FLASHING	PA & COPILOT PHONE
PILOT PHONE	SECONDARY NAVIGATION LIGHT
GPWS (IF INSTALLED)	COPILOT'S EFIS
TCAS (IF INSTALLED)	COPILOT'S ENCODING ALTIMETER
PILOT'S EFIS	PROP GND SOL *

** Airplanes Modified by Raytheon Kit No. 129-9011-1.*

L GEN AVIONICS BUS:

CABIN BRIEFER

COMM NO. 2

DME NO. 1

ADF NO. 1

RADAR

EFIS AUX BATTERY

R GEN AVIONICS BUS:

NAV NO. 2

ALTITUDE ALERTER

RADIO ALTIMETER

COMPASS NO. 2

TRANSPONDER NO. 2

DME NO. 2

RMI NO.1

COPILOT'S DSP

STANDBY INSTRUMENTS AUX BATTERY (On Airplanes Modified by Raytheon Kit No. 129-9018-9)

STANDBY ATTITUDE INDICATOR AUX BATTERY (Serials UE-1 thru UE-249, UE-251 thru UE-258, and UE-260 thru UE-262, Unless Modified By Raytheon Kit No. 129-9018-9)

L CENTER BUS:

LANDING GEAR MOTOR

R FUEL BOOST PUMP

NO. 1 INVERTER POWER

L MANUAL PROP DEICE

R CENTER BUS:

CONDENSER BLOWER MOTOR

PNEUMATIC SURFACE DEICE

MANUAL PROP DEICE CONTROL

R MANUAL PROP DEICE

WINDSHIELD WIPER MOTOR

GENERATOR RESET

NAV LIGHTS (PRI)

ICE LIGHTS

TAXI LIGHTS

L FUEL BOOST PUMP

NO. 2 INVERTER POWER

GROUND WARNING LIGHT

CABIN INDIRECT LIGHTS - PARTIAL

L & R FIRE EXTINGUISHER MONITOR

HOT BATTERY BUS AND BATTERY BUS:

STEREO (OPT)

RNAV MEMORY (OPT)

L ENG FIRE EXTINGUISHER

R ENG FIRE EXTINGUISHER

L F.W. SHUTOFF VALVE

R F.W. SHUTOFF VALVE

EXTERNAL POWER OVER-VOLTAGE SENSOR & ADVISORY LIGHT

DOOR ENTRY LIGHTS

CARGO COMPARTMENT LIGHTS

COCKPIT EMER LIGHTS

CONTROL WHEEL CLOCK

R PITOT HEAT (BATTERY BUS)

CABIN LOADING LIGHTS

TRIPLE FED BUS:

L FUEL FLOW INDICATOR
L OIL TEMP INDICATOR
L OIL PRESS INDICATOR
L FIRE DETECTION
L IGNITION POWER
L START CONTROL
L BLEED AIR WARNING
STALL WARNING
ANNUNCIATOR POWER
LANDING GEAR WARNING HORN SILENCE
CABIN TEMP CONTROL
BATTERY BUS TIE POWER
AVIONICS MASTER CONTROL
AUTO PROP DEICE CONTROL
AVIONICS ANNUNCIATOR
TRIPLE FED AVIONICS BUS
LANDING GEAR CONTROL
L FUEL QUANTITY
L FUEL PRESSURE WARNING
L FUEL LOW LEVEL WARNING
L ITT INDICATOR (UE-93 & After)
L TORQUEMETER (UE-93 & After)
L PROPELLER TACHOMETER
L TURBINE TACHOMETER
RUDDER BOOST
PILOT SPEAKER
L GEN BUS TIE POWER & CONTROL
**LAVATORY SMOKE DETECTOR

R FUEL FLOW INDICATOR
R OIL TEMP INDICATOR
R OIL PRESS INDICATOR
R FIRE DETECTION
R IGNITION POWER
R START CONTROL
R BLEED AIR WARNING
AURAL WARNING
ANNUNCIATOR INDICATOR
LANDING GEAR POSITION CONTROL
L BLEED AIR CONTROL
CURRENT SENSE TEST
CABIN PRESS CONTROL
*PILOT'S ENCODING ALTIMETER
OVERSPEED SENSOR
L PITOT HEAT
FUEL TRANSFER VALVE
R FUEL QUANTITY
R FUEL PRESSURE WARNING
R FUEL LOW LEVEL WARNING
R ITT INDICATOR (UE-93 & After)
R TORQUEMETER (UE-93 & After)
R PROPELLER TACHOMETER
R TURBINE TACHOMETER
INSTRUMENT INDIRECT LIGHTS
COPILOT SPEAKER
COCKPIT VOICE RECORDER

**On Airplanes Equipped With a Flight Director, But No Autopilot.*
*** Airplanes Modified by Raytheon Kit No. 129-5031-1.*

TRIPLE FED AVIONICS BUS:

NAV NO. 1
TRANSPONDER NO. 1
RMI NO. 2
STANDBY COMPASS LIGHTING (NO CIRCUIT BREAKER)

COMM NO. 1
COMPASS NO. 1
PILOT'S DSP
*PILOT'S ENCODING ALTIMETER

**On Airplanes Equipped With an Autopilot.*

Three high current sensing devices control the three bus tie relays. The Left and Right Sensors are bi-directional while the Battery Sensor is uni-directional. These devices actuate when a current of 325 amperes or greater is supplied from a single source. Should current of this magnitude occur, the sensor affected will isolate the bus requiring the high current and allow the remaining power sources to continue to operate as a system.

The high current sensing devices may be tested by means of a BUS SENSE - RESET - NORM - TEST switch located on the pilot's left subpanel. Adjacent to the RESET - NORM - TEST switch is a manual GEN TIES - MAN CLOSE - NORM - OPEN switch. The MAN CLOSE position is electrically held. Four annunciators, three amber and one green, located on the Caution/Advisory annunciator panel are associated with the sensing and bus tie system. All buses may be monitored by a voltmeter located in the overhead panel by selecting the desired bus with the voltmeter select switch located adjacent to the voltmeter.

Power from the generators to the bus system is supplied through line contactor relays, which are employed as reverse current devices by the generator control units. These devices prevent the generators from absorbing power from the bus when the generator voltage is less than the bus voltage. Each generator is controlled by a three-position switch on the pilot's left subpanel placarded GEN RESET - ON - OFF.

Power to each starter generator is provided from the battery through a starter relay. The start cycle for each engine is controlled by a three-position switch on the pilot's left subpanel placarded IGNITION AND ENGINE START - ON - OFF - LEFT - RIGHT - STARTER ONLY.

AC POWER

The two 115/26VAC 400 Hz inverters are controlled by the AC BUS LEFT and AC BUS RIGHT switches located on the pilot's left subpanel. Inadequate inverter power is indicated by the illumination of the R or L AC BUS annunciator. Should a dual generator failure occur, the DC input for the inverters is automatically transferred from the generator bus to the center bus and the equipment not requiring AC power automatically sheds.

EXTERNAL POWER

For ground operation, an external power socket, located under the aft portion of the left nacelle, is provided for connecting an auxiliary power unit. A relay in the external power circuit will close only if the external source polarity is correct. The EXT PWR (ON) - OFF switch, located on the pilot's left subpanel must be ON before the external power relay will close and allow external power to enter the airplane electrical system. The BATTery MASTER SWITCH(es) should also be ON, since the battery tends to absorb voltage transients when operating avionics equipment and during engine starts. Otherwise, the transients might damage solid state components in the airplane.

For starting, an external power source capable of supplying a minimum of 300 amperes continuous and 1000 amperes momentarily should be used. An advisory annunciator in the caution/advisory annunciator panel, EXTERNAL POWER, is provided to alert the operator when an external DC power plug is connected to the airplane.

LIGHTING SYSTEMS

COCKPIT

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 controls the pilot and copilot instrument lights, engine instrument lights, avionics panel lights, overhead flood lights, indirect instrument lights, edge lit panel lights, and overhead, side panel, subpanel, and console lighting. The instrument indirect lights in the glareshield and overhead map lights are individually controlled by separate rheostat switches.

The push-button FREE AIR TEMP switch, located on the left sidewall panel next to the outside air temperature gage, turns on and off the lights located near the gage. These lights are also controlled by a rheostat on the overhead light control panel.

CABIN

A three-position switch on the overhead panel, placarded CABIN LIGHTS - CABIN FULL - PARTIAL - OFF controls the cabin lights. A two-position switch directly to the right, placarded CABIN LIGHTS - READING - ON - OFF, provides power to the individual reading lights. With this switch in the ON position, the individual reading lights located along the top of the cabin may be turned ON or OFF by the passenger with a push button switch adjacent to each light.

Threshold lighting at the passenger entry way is provided by six lights; one above each of the 5 steps, and one on the interior partition just aft of the door. These lights are controlled by a two-position switch located on the side of the third step of the airstair door, and are connected to the hot battery bus.

The cargo compartment lights are controlled by a two-position switch located just inside the cargo door at floor level. The cargo compartment lights are connected to the hot battery bus.

EXTERIOR

RECOGNITION LIGHTS

The recognition lights are installed on each wing tip, forward of the strobe lights and inboard of the navigation lights. They are incorporated into the navigation light assembly and controlled by the RECOG toggle switch located on the Overhead Light Control Panel.

NAVIGATION LIGHTS

The dual navigation light installation consists of two red lights on the left wing tip, two green lights on the right wing tip, and two white lights located on the aft end of the vertical stabilizer fairing. The lights are controlled by the NAV toggle switch on the Overhead Light Control Panel. The navigation light system is actually two separate circuits, each circuit providing power to one of the dual navigation lights.

TAXI LIGHT

The taxi light installation consists of a single light mounted on the nose landing gear strut. It is controlled by the TAXI toggle switch on the Overhead Light Control Panel. If the TAXI switch is in the ON position when the landing gear is retracted, a white advisory panel light (TAXI LIGHT) will illuminate.

LANDING LIGHTS

The landing lights are installed in the leading edge of each wing and are controlled by the LEFT and RIGHT LANDING toggle switch on the Overhead Light Control Panel.

WING ICE LIGHTS

The wing ice lights are located on the outboard side of each engine nacelle and are controlled by the ICE toggle switch on the Overhead Light Control Panel.

TAIL FLOOD LIGHTS

The tail flood lights are located on the underside of the horizontal stabilizer. They are controlled by the TAIL FLOOD toggle switch on the Overhead Light Control Panel.

ANTI-COLLISION BEACON/STROBE LIGHTS

BEACON LIGHTS

The anti-collision beacon lights are installed on the underside of the fuselage and on the vertical stabilizer fairing. These lights are controlled by a 3-position toggle switch (OFF, GND, and FLIGHT) on the Overhead Light Control Panel. In the GND position, the lights are powered by the center battery bus. When in the FLIGHT position, they are powered by the left generator bus.

STROBE LIGHTS

The wing strobe lights are located on the outboard side of each winglet, and are controlled by the STROBE toggle switch located on the Overhead Light Control Panel.

ENVIRONMENTAL SYSTEM

The environmental system consists of the bleed air pressurization, heating and cooling systems, and their associated controls.

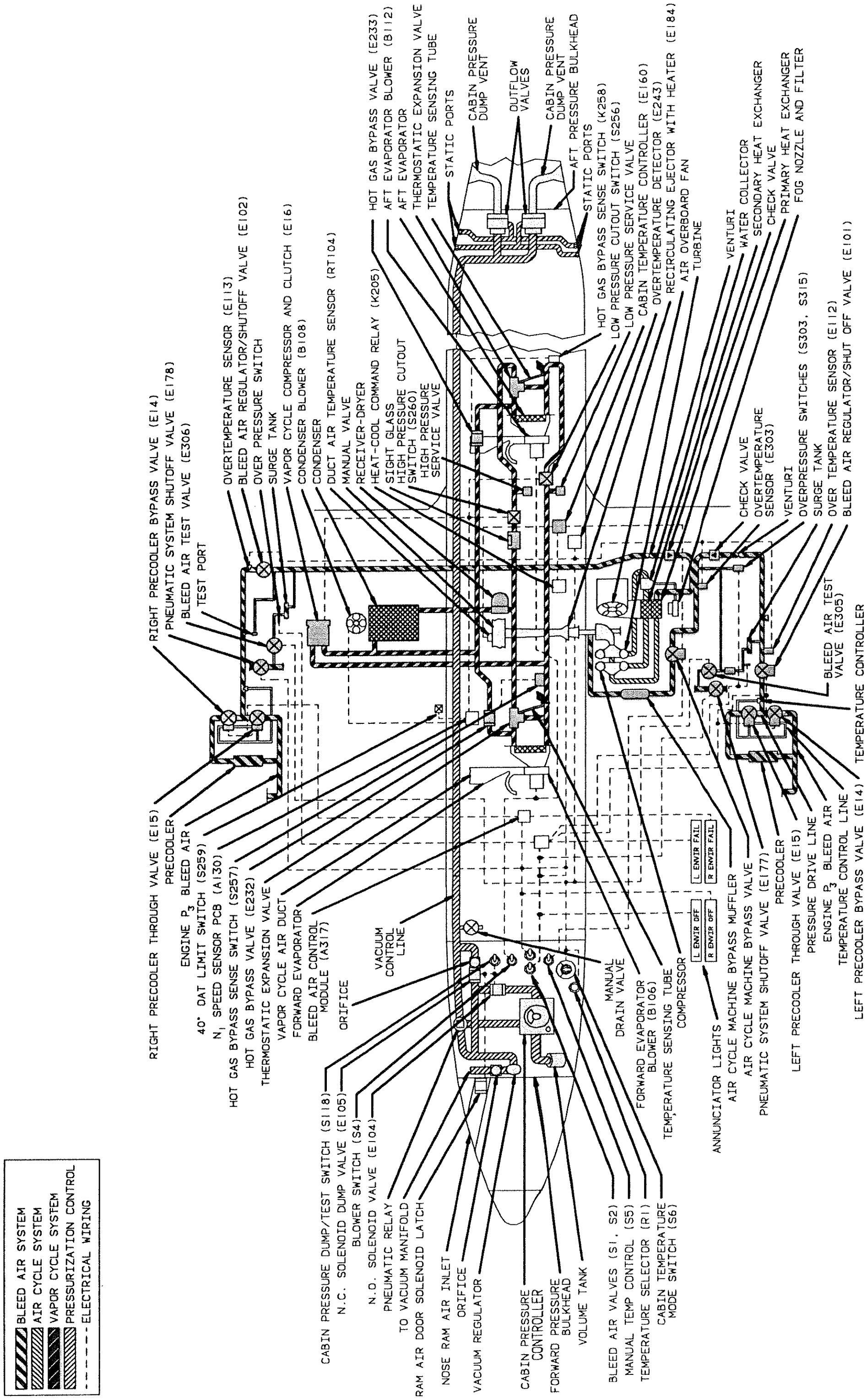
PRESSURIZATION SYSTEM

The pressurization system is designed to provide a pressure differential of 5.0 ± .1 psi. Bleed air from the compressor section of each engine is used to pressurize the cabin. A heat exchanger and two valves on each engine are used to precool the bleed air before it is ducted to the air cycle machine, or bypassed into the cabin. The environmental bleed air shutoff valves, controlled by switches placarded BLEED AIR VALVES - OPEN - ENVIR OFF - INSTR & ENVIR OFF - LEFT - RIGHT on the copilot's left subpanel, control bleed air flow and pressure.

When the environmental shutoff valves are open, conditioned air is delivered to the cabin through outlets in the lower sidewalls for pressurization, heating, and primary cooling from the air cycle machine. An adjustable cabin pressurization controller is mounted in the pedestal. It 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 is set to maintain. The inner scale (ACFT ALT) indicates 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 indicated value on each scale is read opposite the index mark at the forward (top) position of the dial. Both scales rotate together when the cabin altitude selector knob, placarded CABIN ALT is turned. The maximum cabin pressure altitude is selected by turning the cabin altitude selector knob until the desired setting on the CABIN ALT dial is aligned with the index mark. The maximum cabin altitude selected may be anywhere from SL to +10,000 feet. The rate at which the cabin pressure altitude changes from the current value to the selected value is controlled by rotating the rate control selector knob. The rate of change selected may be from approximately 200 to approximately 2000 feet per minute.

The actual cabin pressure altitude is continuously indicated by the cabin altimeter, which is mounted in the right side of the panel that is located between the caution/advisory annunciator panel and the pedestal. Immediately to the left of the cabin altimeter is the cabin vertical speed (CABIN CLIMB) indicator, which continuously indicates the rate at which the cabin pressure altitude is changing.

The cabin pressure switch, located forward of the pressurization controller on the pedestal, is placarded CABIN PRESSURE DUMP - PRESS - TEST. When this switch is in the DUMP position, the outflow valves are held open, so that the cabin will depressurize and/or remain unpressurized. When it is in the PRESS (center) position, the outflow valves are controlled by the pressurization controller. To facilitate testing of the pressurization system on the ground, the switch is held in the spring-loaded TEST (aft) position: this opens the landing gear safety switch circuit, de-energizing the dump solenoid and allowing the outflow valves to close. Prior to takeoff, the cabin altitude selector knob should be adjusted so that the ACFT ALT scale on the indicator dial indicates an



ENVIRONMENTAL SYSTEM SCHEMATIC

altitude approximately 1000 feet above the planned cruise pressure altitude (but not to exceed 25,000 feet pressure altitude), and the CABIN ALT scale indicates an altitude at least 500 feet above the take-off field pressure altitude. The rate control selector knob should be adjusted as desired; setting the index mark at the 12-o'clock position 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. As the airplane climbs, the cabin pressure altitude climbs at the selected rate of change until the cabin reaches the selected pressure altitude. The system then maintains cabin pressure altitude at the selected value. If the airplane climbs to an altitude higher than the value indexed on the ACFT ALT scale of the dial on the face of the controller, the cabin-to-ambient pressure differential will reach the pressure relief setting of the outflow valves. Either or both valves will then override the cabin pressurization controller in order to limit the cabin-to-ambient pressure differential to 5.0 ± .1 psi. If the cabin pressure altitude should reach a value of 9500 to 10,000 feet, a pressure-sensing switch mounted on the electrical panel under the cabin floor will close. This causes the CABIN ALT HI annunciator light to illuminate. During cruise operation, if the flight plan calls for an altitude change of 1000 feet or more, reselect the new altitude plus 1000 feet on the CABIN ALT dial.

If cabin differential pressure exceeds 5.25 ± .15 psi, a pressure switch located in the pedestal will close, illuminating the CAB DIFF HI annunciator.

During descent and in preparation for landing, the cabin altitude selector should be set to indicate a cabin altitude of approximately 500 feet above the landing field pressure altitude, and the rate control selector should be adjusted as required to provide a comfortable cabin-altitude rate of descent. 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. Then, as the airplane descends to and reaches the cabin pressure altitude, the pressurization controller modulates the outflow and safety valve poppets toward the open position, thereby equalizing the pressure inside and outside the pressure vessel. As 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 touch down.

UNPRESSURIZED VENTILATION

Ambient air can be supplied to the cabin through the floor vents when the airplane is depressurized. A manually controlled valve located in the nose ram air duct can be opened to allow air to enter the airplane through the ram air door solenoid valve and the manual valve when the cabin pressure control switch is set to DUMP. The control for the manual shutoff valve placarded VENT AIR - PULL ON is located under the copilot's left subpanel.

HEATING

Bleed air from the engines enters the cabin distribution ducts for heating through the air cycle machine bypass valve.

The air cycle machine bypass valve opens just when heating is required, and modulates the output of the air cycle machine. As the cabin temperature controller continues to request heat, the air cycle machine bypass valve opens fully. When the cabin temperature controller begins requesting cooling, operating current is shunted to the air cycle machine bypass valve, closing it.

The temperature controlled, heated air is routed through the outlets in the lower cabin sidewall, crew vents, and defroster.

COOLING

All cabin cooling is provided by the air cycle system and, when required, by the vapor cycle system. When heat loads are such that the air cycle system is producing maximum cooling, a signal is transmitted to the temperature control circuitry, causing the refrigerant compressor clutch to be engaged, thereby turning the compressor on and initiating the cooling cycle of the vapor cycle system.

AIR CYCLE SYSTEM

The air cycle machine utilizes engine bleed air to drive a compressor which compresses the bleed air to a point where the excess heat of compression can be removed through the use of heat exchangers. When the pressure is released through the air cycle machine turbine, the resulting air temperature is much lower than ambient air temperature.

VAPOR CYCLE SYSTEM

A condensing coil and blower assembly, located in the right forward inboard wing, removes heat from the high temperature, high pressure gaseous refrigerant being discharged from the compressor allowing the refrigerant to condense to the liquid state. The high pressure, low temperature liquid refrigerant passes through a metering device (thermostatic expansion valve) into the evaporator where the pressure is relieved and the refrigerant allowed to evaporate into the gaseous state producing a heat deficit in the gaseous refrigerant. Cabin air is circulated over the evaporator coil where heat is transferred from the cabin air to the gaseous refrigerant. The low pressure, low temperature refrigerant then returns to the compressor and the entire cycle is repeated.

Once the vapor cycle system is activated, it will remain in operation until such time as the air cycle machine bypass valve has opened fully, at which time a signal is transmitted to the heat side of the heat/cool relay, and vapor cycle cooling is terminated.

ENVIRONMENTAL CONTROLS

The ENVIRONMENTAL control section on the copilot's left subpanel provides for automatic or manual control of the system. This section contains all the major controls of the environmental function systems: bleed air valve switches; blower control switch; a manual temperature switch for control of the cabin-temperature control valves; a cabin-temperature-level control; and the environmental mode selector switch, for selecting automatic heating or cooling, manual heating or cooling, or off. Four additional manual controls on the pilot's and copilot's subpanels may be utilized for partial regulation of cockpit comfort. They are: pilot's air, defroster air, cabin air, and copilot's air control knobs. The fully out position of all these controls will provide the maximum airflow to the cockpit, and the fully in position will provide minimum airflow to the cockpit.

Cut-off valves, controlled by blade-type levers, are located under the pilot's and copilot's chairs for additional control of airflow in the cockpit.

For warm flights, such as short, low-altitude flights in summer, all the cabin eyeball outlets should be fully open for maximum cooling. For cold flights, such as high-altitude flights, night flights, and flights in cold weather, the eyeball outlets should all be closed for maximum heating in the cabin.

If the cabin temperature is comfortable but the cockpit temperature is not, the following procedures are suggested:

HEATING MODE

If the cockpit is too cold:

1. PILOT AIR, CO-PILOT AIR, and DEFROST AIR Knobs - PULLED FULLY OUT, or as required.
2. CABIN AIR Knob - PULLED OUT IN SMALL INCREMENTS (Allow 3 to 5 minutes after each adjustment for system to stabilize.)
3. Cockpit Overhead Eyeball Outlets - CLOSED, or as required.

If the cockpit is too hot:

4. PILOT AIR, CO-PILOT AIR, DEFROST AIR, and CABIN AIR Knobs - PUSHED FULLY IN, or as required.
5. Cockpit Overhead Eyeball Outlets - OPEN, or as required.

COOLING MODE

If the cockpit is too cold:

1. PILOT AIR, CO-PILOT AIR, and DEFROST AIR Knobs - PUSHED FULLY IN, or as required.
2. Cockpit Overhead Eyeball Outlets - CLOSED, or as required.
3. CABIN AIR - PUSHED FULLY IN, or as required.

If the cockpit is too hot:

4. PILOT AIR and CO-PILOT AIR Knobs - PULLED FULLY OUT, or as required.
5. CABIN AIR Knob - PULLED OUT IN SMALL INCREMENTS (Allow 3 to 5 minutes after each adjustment for system to stabilize.)
6. Cockpit Overhead Eyeball Outlets - OPEN, or as required.

AUTOMATIC TEMPERATURE CONTROL

The temperature of the cabin can be automatically controlled by presetting the CABIN TEMP selector to the desired temperature and setting the MODE CONTROL switch to AUTO. The cabin temperature controller then sends the appropriate heat or cool command to the air cycle machine bypass valve. A dual-element temperature sensor installed in the conditioned bleed air main duct completes the Wheatstone Bridge resistance circuitry of the cabin temperature controller. When the cabin temperature falls, the Wheatstone Bridge becomes unbalanced and current flows from the heat command output of the cabin temperature controller. The air cycle machine bypass valve receives the command and opens to allow more bleed air to enter the conditioned bleed air ducts. The bypass valve modulates the amount of bleed air flow in automatic response to the requirements indicated by the changing resistance of the duct temperature sensors and the temperature sensor located in the cabin temperature controller.

MANUAL TEMPERATURE CONTROL

The temperature of environmental air can be controlled manually by setting the MODE CONTROL switch to MAN and holding the MAN TEMP control switch to the INCR or the DECR positions. When the MAN TEMP switch is set to INCR or DECR, power is supplied through the TEMP CONTROL circuit breaker, the mode switch and the manual switch to the bypass valve. When the switch is set to INCR, the air cycle machine bypass valve opens, allowing air to flow through the floor outlets. The bypass valve closes, as required, to allow the air cycle system to cool the bleed air when the manual switch is set to DECR. If the manual temperature control switch is held in the DECR position, the air cycle machine bypass valve begins to close to direct air through the air cycle machine for cooling of the cabin.

With MAN COOL selected on the ENVIR MODE CONTROL switch, the vapor cycle system compressor clutch is powered independently of the air cycle machine bypass valve position, to provide the maximum rate of cooling to a heat-soaked cabin. Continuous operation in this position is not recommended, since it may cause ice formation in the air cycle machine, ejector and evaporators.

TEST FUNCTION

The environmental system incorporates a test function activated by the ENVIR MODE CONTROL switch.

Selection of the T TEST function simulates an over-temp condition, causing the sensing circuits to shut down the bleed air valves. The L & R ENVIR FAIL and L & R ENVIR OFF annunciators will illuminate, indicating correct system operation. Returning the ENVIR MODE CONTROL switch to the desired position will extinguish the L & R ENVIR FAIL and L & R ENVIR OFF annunciators.

The P TEST function performs a similar over-pressure test as part of routine maintenance inspections.

BLEED AIR CONTROL

Bleed air entering the cabin is controlled by the switches on the copilot's left subpanel, placarded BLEED AIR VALVES - OPEN - ENVIR OFF - INSTR & ENVIR OFF - LEFT - RIGHT. When the switch is in the OPEN position, the environmental flow control valves and the pneumatic instrument air valve are open. When the switch is in the ENVIR OFF position, the environmental flow control valves are closed and the pneumatic instrument air valve is open; in the INSTR & ENVIR OFF position, all are closed.

VENT BLOWER CONTROL

UE-2 THRU UE-6

The vent blowers are controlled by a three-position switch placarded BLOWERS - HIGH - LOW - AUTO and the MODE CONTROL switch, both located on the copilot's left subpanel. The blowers will operate in either the HIGH or LOW position when the MODE CONTROL switch is in any position. When the blower switch is set to AUTO, the vent blowers will operate at low speed when the mode switch is set to any position except OFF. Blower control is independent of all other components of the environmental system.

UE-7 AND AFTER

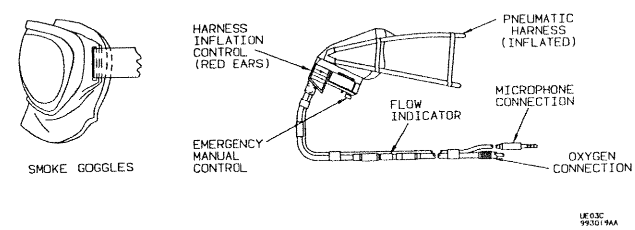
The vent blowers are controlled by a three-position switch placarded BLOWERS - HIGH - OFF - AUTO and the MODE CONTROL switch, both located on the copilot's left subpanel. The blowers will operate in the HIGH position when the MODE CONTROL switch is in any position. When the blower switch is set to AUTO, the vent blowers will operate at low speed when the mode switch is set to any position except OFF. Blower control is independent of all other components of the environmental system.

OXYGEN SYSTEM

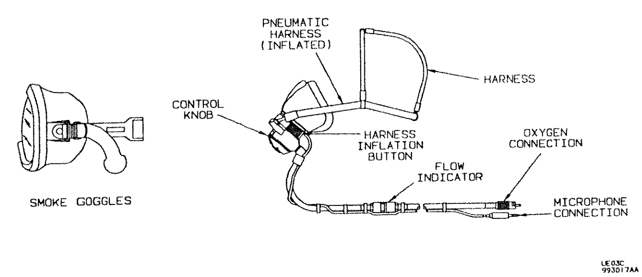
The Beech 1900D Airliner oxygen system is based on an adequate flow for a pressure altitude of 25,000 feet.

The crew oxygen system offers four types of masks. All are diluter demand with mask mounted regulator.

The Scott - EROS and the Puritan Bennett - Sweep On 2000 optional crew oxygen masks are available. Their function is identical, differing only in appearance. Both have three control selections (NORMAL, 100%, and EMERGENCY), and are retained on the face by an inflatable elastic type harness. They offer faster donning in addition to longer oxygen duration below 20,000 feet. These masks, in the NORMAL setting, provide adequate oxygen flow at all altitudes through 25,000 feet.

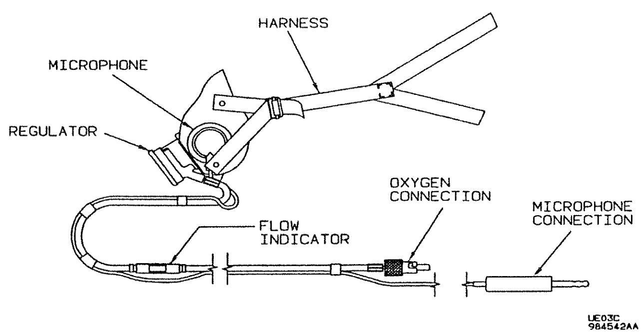


"SCOTT-EROS" OXYGEN MASK



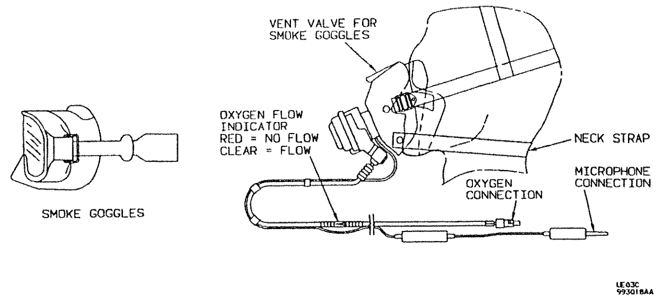
"PURITAN BENNETT - SWEEP ON 2000" OXYGEN MASK (UE-137 THRU UE-249, UE-251 THRU UE-258, UE-260 THRU UE-262)

The Scott-893 crew oxygen mask for UE-173 thru UE-249, UE-251 thru UE-258, and UE-260 thru UE-262 has a two-position selector (NORMAL and 100%) and is retained on the face by a strap type harness. It should be operated in the NORMAL position at cabin pressure altitudes up to and including 20,000 feet. To ensure adequate flow at cabin altitudes above 20,000 feet, it should be set to 100%. Consequently, this mask should be preset and stowed in the 100% position.



"SCOTT-893" OXYGEN MASK (UE-173 THRU UE-249, UE-251 THRU UE-258, UE-260 THRU UE-262)

The Scott-359 crew oxygen mask, for airplanes with Raytheon Kit No. 129-5032-1 installed is a diluter demand type with a mask vent valve. The vent valve is used in conjunction with smoke protection goggles. The mask has a diluter demand type regulator and is equipped with a dynamic microphone. To begin oxygen flow, place the regulator manual select switch to the desired operating position. An oxygen flow indicator on the mask hose indicates CLEAR with proper flow and RED with no flow. The manually operated push-pull vent valve is used in conjunction with the regulator EMERGENCY Pressure feature to divert a small flow of oxygen from the mask cavity into the smoke goggle cavity to vent smoke or fumes which may be present in the smoke goggle cavity. The smoke goggles are stowed above the crew members head on the forward cabin partition.



"SCOTT-359" OXYGEN MASK (AIRPLANES WITH RAYTHEON KIT NO. 129-5032-1 INSTALLED)

The passenger oxygen system is equipped with an altitude compensated regulator, mounted on the left side of the airplane just aft of the forward pressure bulkhead. This regulator varies line pressure to the passenger masks, thus tailoring flow rates versus cabin altitude to maximize oxygen duration.

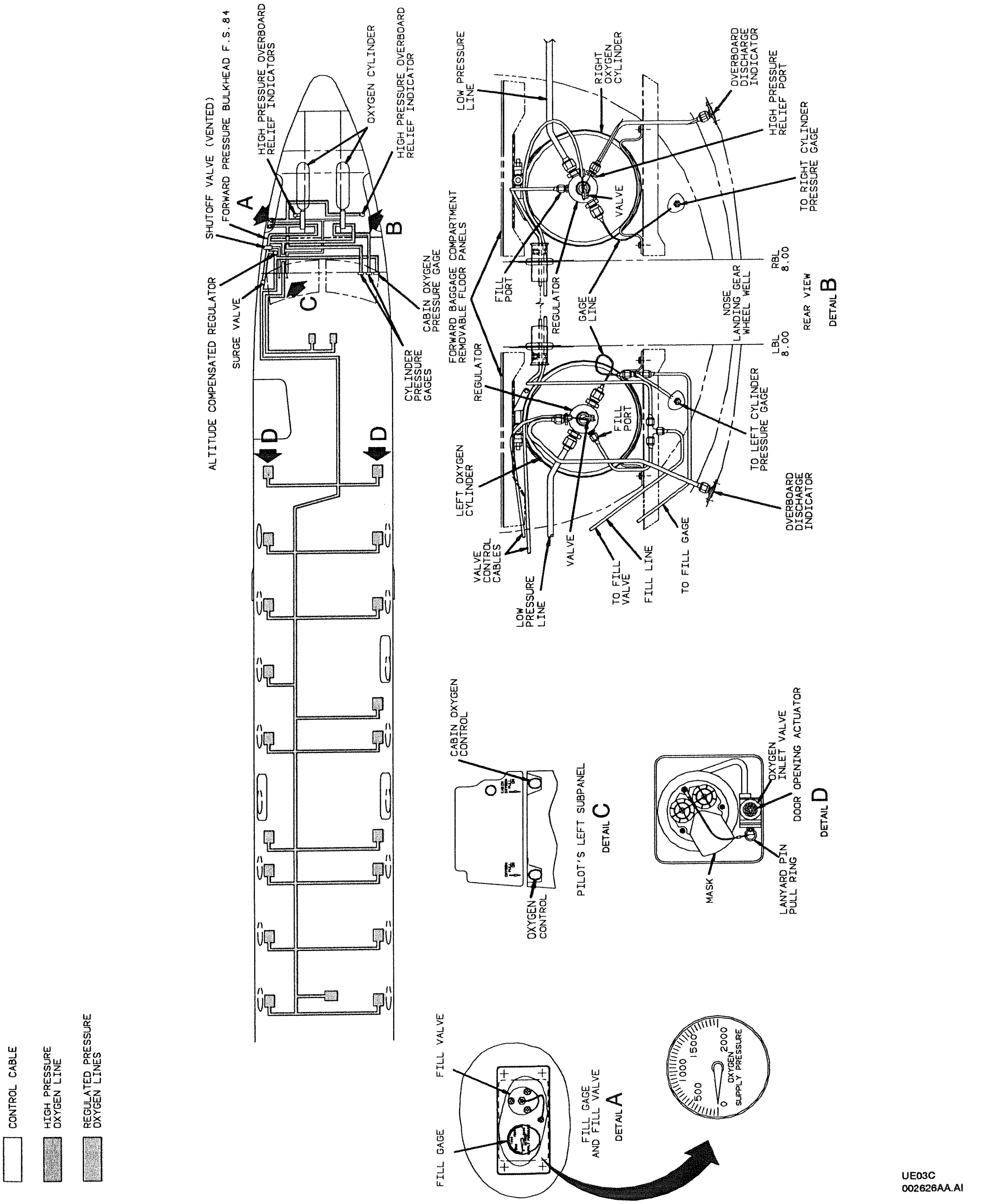
The oxygen system utilizes two 77-cubic foot cylinders, one mounted on either side of the nose under the floor of the nose compartment. Two cylinder pressure gages are mounted on the copilot's right subpanel. A gage, indicating pressure to the cabin masks is mounted at the right side of the instrument panel.

The cylinders are actuated simultaneously by a push-pull (OXYGEN - PULL ON) control knob located at the lower left corner of the pilot's left subpanel. Pressure from the cylinder is controlled by a regulator mounted on each cylinder, and an altitude compensated regulator mounted on the cabin side of the forward pressure bulkhead.

Two mask containers, located behind the overhead light control panel, provide ready access to crew masks. Access to the crew masks is achieved by removing them from their compartment. Oxygen is supplied upon donning the masks.

A push-pull (CABIN OXYGEN - PULL ON) control knob mounted below the right corner of the pilot's left subpanel governs oxygen flow to the passenger masks. When this control is pulled out, oxygen will be supplied to the oxygen masks, thus allowing a surge valve to momentarily permit high pressure to reach the passenger mask container assemblies located at the outboard side of the cabin above each window in the light assembly. The high pressure causes the container door to open, allowing access to the oxygen mask. In order to initiate oxygen flow from the masks, a lanyard valve pin must be pulled out. The pin must be reinserted to stop the flow of oxygen.

Raytheon Aircraft



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OXYGEN SYSTEM SCHEMATIC

PITOT AND STATIC SYSTEM

The normal static system provides two separate sources of static air - one for the pilot's flight instruments, and one for the copilot's. Each of the normal static air lines opens to the atmosphere through static air ports in the pitot/static masts. Cross plumbing between the two static sources is provided so that static pressure is available to both pilot's and copilot's instruments should either source become obstructed.

An alternate static air line is also provided for the pilot's and copilot's flight instruments. In the event of a failure of the pilot's or copilot's normal static air source, the alternate source should be selected by lifting the spring-clip retainer off the PILOT'S or COPILOT'S STATIC AIR SOURCE valve switch located on the left and right cockpit lower sidewalls respectively and moving the switch down to the ALTERNATE & DRAIN position. This will connect the alternate static air line to the respective flight instruments. The alternate line obtains static air on each side of the lower nose.

The pilot's and copilot's altimeter, vertical speed indicator, and airspeed indicator are connected to their respective static air sources. When the system is switched to the pilot's alternate air source, the altimeter and vertical speed indicator are affected as well as the airspeed indicator. See SCHEMATIC DIAGRAM of PITOT AND STATIC SYSTEM.

WARNING

Airspeed and altimeter indications change when the alternate static air source is in use. Refer to the Airspeed Calibration - Alternate System, and the Altimeter Correction - Alternate System graph in the FAA Approved Airplane Flight Manual Section V, PERFORMANCE, for operation when the alternate static air source is in use.

When the alternate static air source is not needed, ensure that the PILOT'S and COPILOT'S STATIC AIR SOURCE valve switch is held in the NORMAL position by the spring-clip retainer.

ENGINE BLEED AIR PNEUMATIC SYSTEM

High-pressure bleed air from each engine compressor, routed through the firewall shutoff valves and regulated at 18 psi, supplies pressure for the surface deice system, vacuum source, and the bleed air warning system. Vacuum for the pressurization and boots is derived from a bleed air ejector. One engine can supply sufficient bleed air for all these systems.

During single-engine operation, a check valve in the bleed air line from each engine prevents flow back through the line on the side of the inoperative engine. A suction gage calibrated in inches of mercury, located on the copilot's right subpanel, indicates instrument vacuum. To the right of the suction gage is a pneumatic pressure gage, calibrated in pounds per square inch, which indicates air pressure available to the deice distributor valve.

Refer to the Pneumatic Bleed Air and Surface Deice System Schematic.

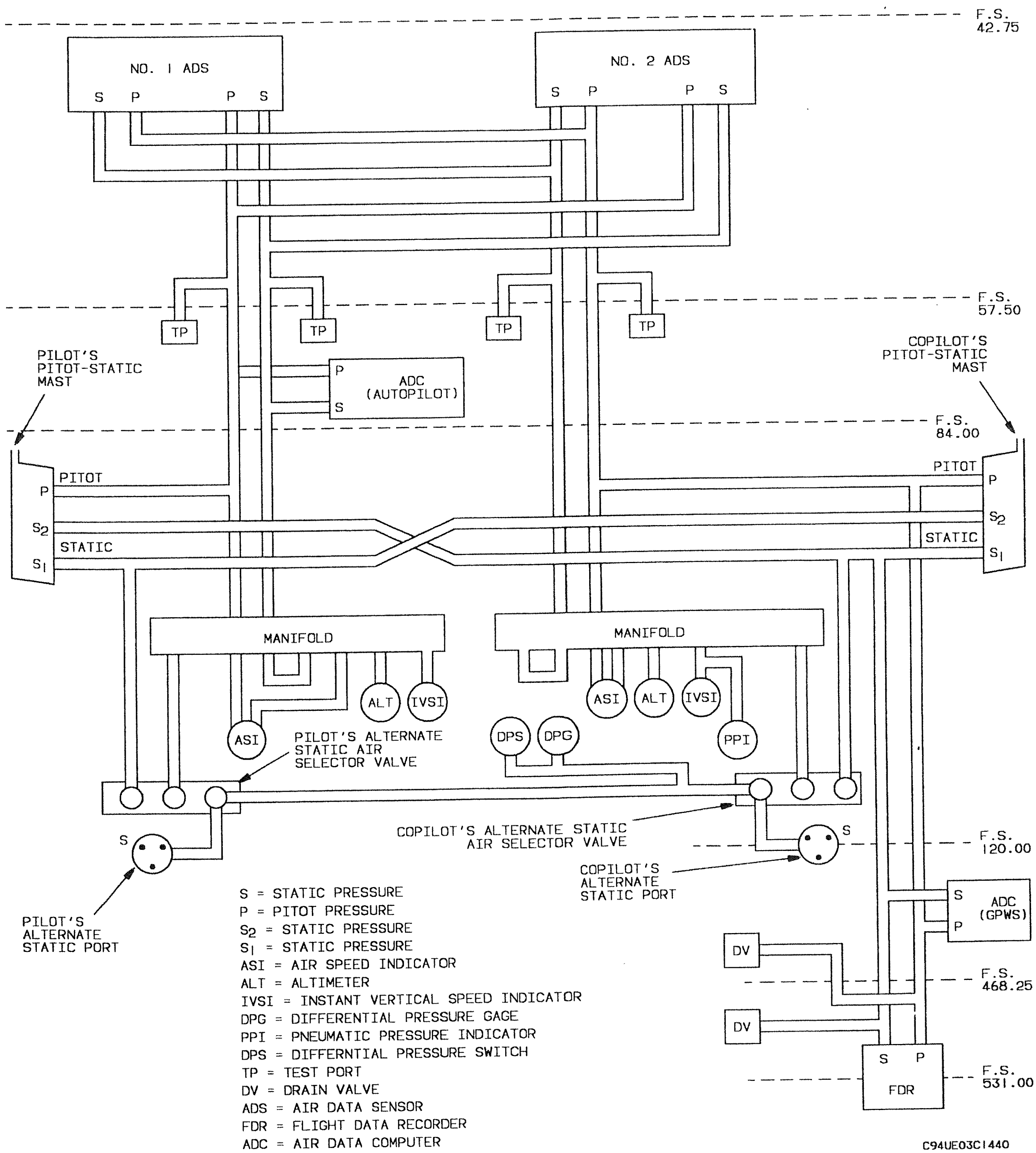
BLEED AIR WARNING SYSTEM

The bleed air lines from the engines to the cabin are shielded with insulation to protect other components from heat. The bleed air lines are accompanied in close proximity by plastic tubing from the firewall to the cabin. One end of the tubing is plugged off; the other end is connected to the regulated 18 psi pneumatic air source in the cabin, to supply the line with pressure. Excessive heat on the plastic tubing caused by a ruptured bleed air line, will cause the tubing to fail. Upon release of pressure in the tubing, a normally open switch in the line, located under the floor in the fuselage, will close, causing a circuit to be completed to the respective L or R BL AIR FAIL annunciator in the warning annunciator panel. When the indication of bleed air line failure becomes evident, the bleed air for that side should be turned off by placing the respective lever-lock BLEED AIR VALVE switch on the copilot's left subpanel in the INSTR & ENVIR - OFF position.

STALL WARNING SYSTEM

Angle of attack is sensed by aerodynamic pressure on the lift transducer vane located on the left wing leading edge. When a stall is imminent, the stall warning horn activates.

The system has preflight capability through the use of a switch placarded STALL WARN TEST on the copilot's left subpanel. Holding this switch in the TEST position activates the warning horn.



PITOT AND STATIC SYSTEM SCHEMATIC

ICE PROTECTION SYSTEMS

WINDSHIELD HEAT

Two levels of heat are provided for each windshield. When the WSHLD ANTI-ICE switches are in the NORMAL (up) position, heat is supplied to the major portion of the windshields. When they are in the HI (down) position, a higher level of heat is supplied to a smaller area of the windshields. Each switch must be lifted over a detent before it can be moved into the HI position. This lever-lock feature prevents inadvertent selection of the HI position when moving the switches from NORMAL to the OFF (center) position.

NOTE

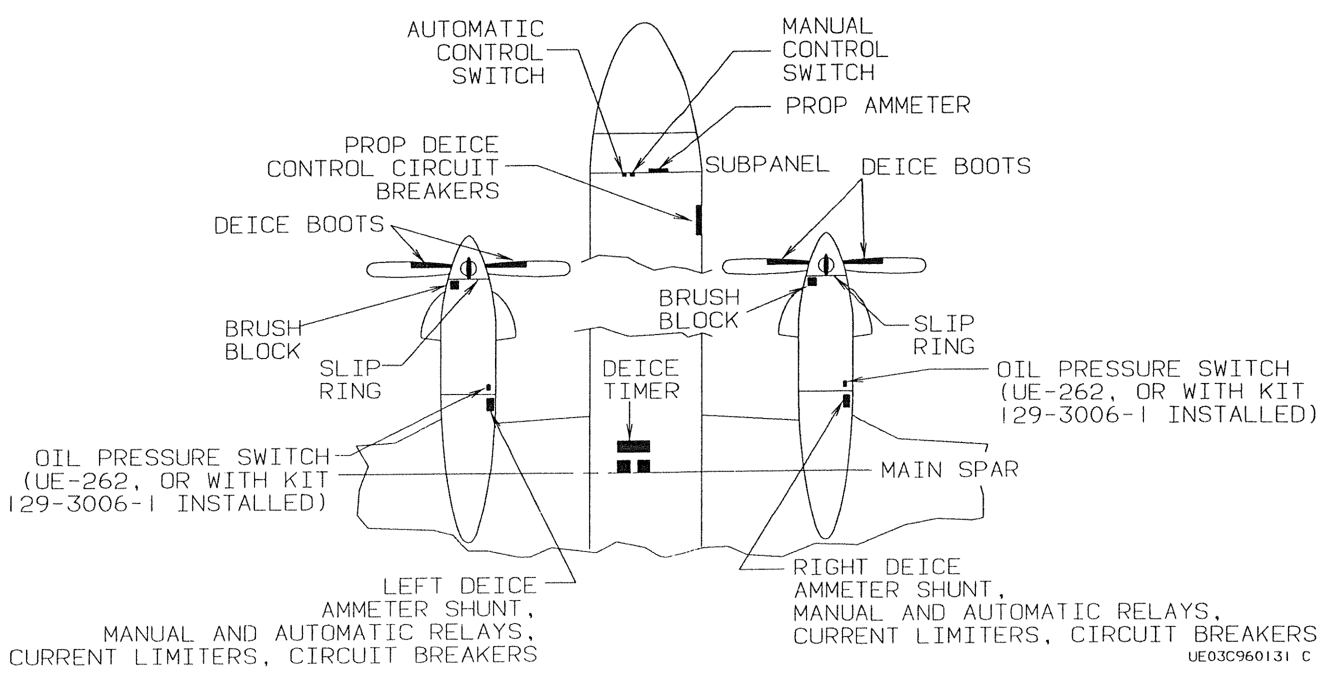
Erratic operation of the magnetic compass may occur while windshield heat is being used.

PROPELLER ELECTRIC DEICE SYSTEM

The propeller electric deice system includes electrically heated deice boots, slip rings and brush block assemblies, a timer for automatic operation, a dual-scale ammeter, two circuit breakers located on the right circuit breaker panel for protection of the automatic and manual propeller deice control circuits, circuit breakers and current limiters located in the nacelles for protection of the propeller deice boot wiring, and two switches located on the pilot's right subpanel for automatic and manual control of the system. On UE-262, or earlier airplanes with Raytheon Kit No. 129-3006-1 installed, the automatic mode of the propeller deice on each engine is inhibited by the respective oil pressure switch. If the oil pressure is low enough to cause the OIL PRES LO annunciator to illuminate, the automatic mode of the propeller deice system will be inoperative on that engine. In such cases, the manual mode will still be operational.

The switch, placarded PROP-AUTO, is provided to activate the system timer of the automatic system. With AUTO selected, power is supplied to the heating elements on one propeller for 90 seconds. The timer then switches power to the other propeller for 90 seconds, resulting in a complete cycle in approximately 3 minutes. The timer also includes a manual step feature which allows a quick check for proper timer operation. Momentarily selecting AUTO results in power being supplied to the heating elements of one propeller. When the deice switch is selected OFF and then back to AUTO within five seconds, power is supplied to the opposite propeller. The normal current to each propeller is 32 to 38 amps. For airplanes modified by Raytheon Kit No. 129-9024-1, -3 or -5, the normal current is 26 to 32 amps. This current may be observed on the dual-scale ammeter, placarded PROP AMPS, located on the overhead panel.

The switch, placarded PROP-MANUAL, is provided to activate the manual propeller deice system which is a backup to the automatic system. With the switch held in the MANUAL position, power is supplied to the heating elements of both propellers simultaneously. This switch is of the momentary type and must be held in position for approximately 90 seconds. As with the automatic system, current to the deicing boots can be monitored on the propeller deice ammeter.



PROPELLER ELECTRIC DEICE SYSTEM SCHEMATIC

Care should be used when operating the propeller deice system to prevent damage to the propeller or propeller deice boots. To avoid exceeding the temperature limit of the adhesives which attach the deice boots and the erosion shield to the propeller, observe the following:

1. Ensure system is OFF when propellers are static.
2. System operation is limited to one cycle per propeller at ambient temperatures above 10°C. (Refer to Section II, LIMITATIONS, of the AFM).

PITOT/STATIC MAST HEAT

Heating elements are installed in the pitot/static masts located on the nose. Each heating element is controlled by a circuit breaker switch placarded PITOT - LEFT - RIGHT, located on the pilot's right subpanel. Annunciators for both L PITOT HEAT and R PITOT HEAT are installed in the Caution/Advisory Panel. Illumination advises that insufficient current is being supplied to provide sufficient heat to the pitot heat mast for deicing. It is not advisable to operate the pitot heat system on the ground except for testing or for short intervals of time to remove ice or snow from the masts.

SURFACE DEICE SYSTEM

The surface deice system removes ice accumulations from the leading edges of the wings, stabilons, horizontal stabilizers and tailets. Ice removal is accomplished by alternately inflating and deflating the deice boots. The engine bleed air pneumatic system supplies pressure to inflate the boots. A venturi ejector creates vacuum to deflate the boots and hold them down while not in use. To assure operation of the system in the event of failure of one engine, a check valve is incorporated in the bleed air line from each engine to prevent loss of pressure through the compressor of the inoperative engine. Inflation and deflation phases are controlled by a distributor valve.

A three-position spring-loaded switch in the ICE PROTECTION group on the pilot's right subpanel placarded SURFACE DEICE - SINGLE - MANUAL - OFF, controls the deicing operation. When the SINGLE position is selected, the distributor valve opens to inflate the outboard wing boots. After an inflation period of approximately 6 seconds, an electronic timer switches the distributor to deflate the outboard wing boots, and opens to inflate the inboard wing boots, the horizontal stabilizer, stabilon and tailet boots. When these boots have inflated and deflated, the cycle is complete. Three annunciators, INBD WG DEICE, TAIL DEICE, and OUTBD WG DEICE, illuminate during this cycle to show proper operation of the deice system.

When the switch is held in the MANUAL position, all the boots will inflate simultaneously and remain inflated until the switch is released. The switch should be held until all three annunciators have illuminated. The switch will return to the OFF position when released. After the cycle, the boots will remain in the vacuum hold down condition until again actuated by the switch. Since all boots inflate at the same time, the pneumatic pressure may drop low enough to momentarily illuminate the BLEED AIR FAIL annunciator and trip the MASTER WARNING annunciator.

For most effective deicing operation, allow at least 1 to 1.5 inches (2.5 to 3.8 centimeters) 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 leading edge, thus making ice removal efforts ineffective.

STALL WARNING VANE HEAT

The lift transducer is equipped with anti-icing capability on both the mounting plate and the vane. The heat is controlled by a switch in the ICE PROTECTION group located on the pilot's right subpanel placarded STALL WARN - OFF. The level of heat is minimal for ground operation, but is automatically increased for flight operation through the left landing gear safety switch. A STALL HEAT annunciator in the Caution/Advisory panel illuminates if there is insufficient current to heat the vane and faceplate heaters.

WARNING

> The heating elements protect the lift transducer vane and face plate from ice. However, a buildup of ice on the wing may change or disrupt the airflow over the wing and prevent the system from accurately indicating an imminent stall. Remember that the stall speed increases whenever ice accumulates on any airplane.

FUEL HEAT

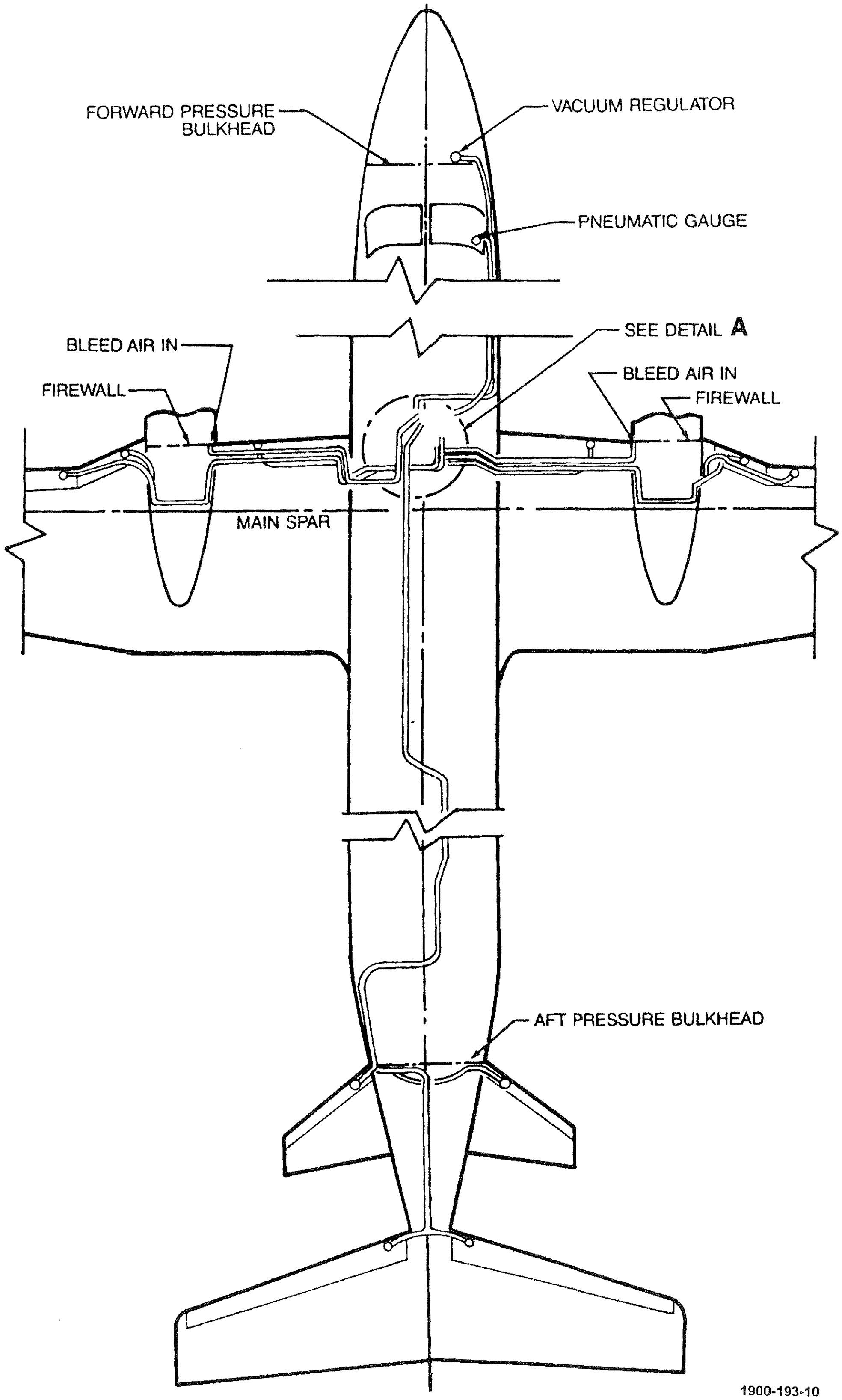
An oil-to-fuel heat exchanger, located on the engine accessory case, operates continuously and automatically to heat the fuel sufficiently to prevent ice from collecting in the fuel control unit.

PNEUMATIC SYSTEM HEAT (IF INSTALLED)

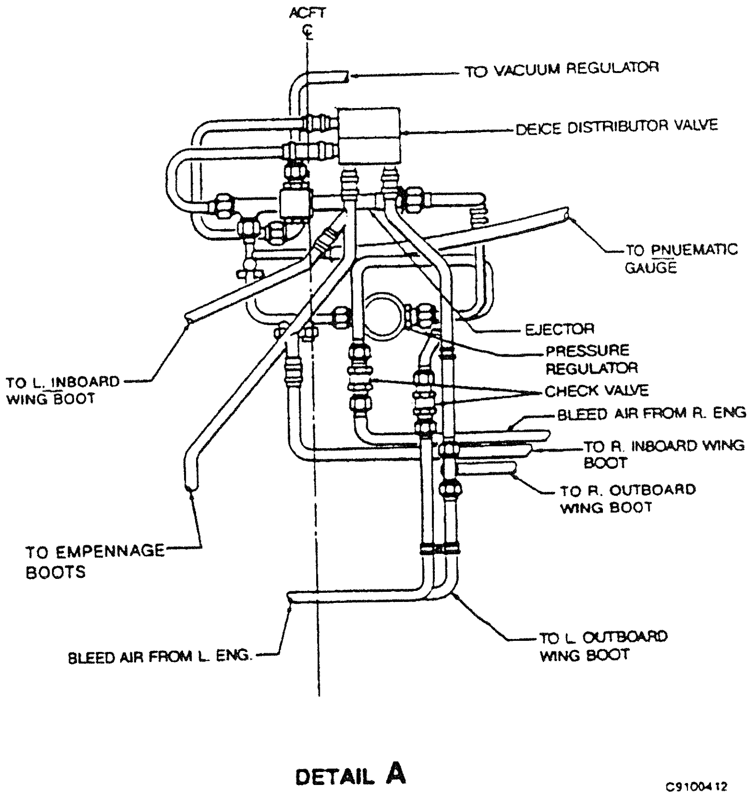
The pneumatic system electric heat installation provides electric heaters and additional thermal insulation on the deice pneumatic regulator and plumbing for the purpose of improving pneumatic system reliability when operating in cold and wet conditions.

System power is provided through a 10-amp circuit breaker tied to the triple fed bus.

The system is activated by a PNEU HEATER - OFF switch located in the ENVIRONMENTAL group on the copilot's left subpanel. Whenever operations at or below 40°F (4°C) are anticipated, the switch should be placed in the PNEU HEATER position. The system is protected from operation at excessive temperatures by a 300°F (150°C) thermostatic switch located on the pneumatic regulator.



SURFACE DEICE SYSTEM SCHEMATIC



SURFACE DEICE DISTRIBUTOR/REGULATOR ASSEMBLY

CABIN FEATURES

SIDE FACING EXTERNALLY SERVICED TOILET (IF INSTALLED)

The lavatory water system consists of a holding tank, an electric motor, timer, and a flush switch adjacent to the lavatory. Power for operation of the lavatory and associated systems is provided by 28 vdc power. Power is provided by the right generator bus through a FURN ON/OFF switch located on copilot's left subpanel and a TOILET circuit breaker. Pressing and then releasing the flush switch will automatically activate a flush motor and timer. The timer will permit the flush motor to operate for a pre-timed interval.

A lavatory holding tank is installed and has a capacity of 8.7 gallons. When servicing the holding tank, refill with 1.7 gallons. To avoid possible damage to pump motor, never operate pump when holding tank is empty.

For passenger safety a RETURN TO SEAT sign is installed in the lavatory. Control for this sign is from the cockpit by a two-position switch (OFF-FSB) located on the overhead light control panel. Illumination or extinguishing of lavatory lights is accomplished from the cockpit. There are two lights installed over the vanity which are controlled by a three-position switch (OFF-PARTIAL-FULL) located on the overhead light control panel. Positioning the switch to either PARTIAL or FULL will activate lavatory lighting. A reading light positioned above the lavatory is activated by a switch mounted adjacent to the light.

For proper servicing procedures refer to the *Beech 1900D Airliner Maintenance Manual*, Chapter 12.

CAUTION

When aircraft is parked where outside temperatures may drop below freezing, ensure that lavatory system is thoroughly drained/purged or removed.

LAVATORY SMOKE DETECTOR (IF INSTALLED)

On airplanes with Raytheon Kit No. 129-5031-1 installed, the airplane is equipped with a lavatory smoke detector. The system consists of a detector, circuit breaker and cockpit red warning annunciator placarded LAVATORY SMOKE.

The detector is 28 VDC powered from the triple fed bus through a circuit breaker located under the cabin floor. A green "power indicator" on the face of the detector is illuminated whenever power is on.

When smoke is sensed, the detector illuminates a small red "alarm indicator" on the face of the detector. It also signals the cockpit annunciator. As smoke clears, the detector will automatically reset. If for some reason, a warning should become undesirable, that warning output to the cockpit annunciator may be canceled by pressing the "interrupt switch". The small red "alarm indicator" may continue to be illuminated until smoke clears, at which time the detector resets.

Pressing the "self test switch" with a blunt probe or pencil point will momentarily check the detectors warning circuits and annunciator.

LAVATORY TRASH BIN FIRE EXTINGUISHER (IF INSTALLED)

With Raytheon Kit No. 129-5033-1 installed, the airplane is equipped with a lavatory trash bin fire extinguisher. This is a fully automatic, self contained system. It consists of a disposable, pressurized and hermetically sealed fire extinguisher with two discharge tubes that project into the lavatory trash bin. Each tube has an end cap that is sealed with a fusible alloy. As fire heats the end caps to approximately 175°F, the fusible alloy melts allowing the Halon 1300 extinguishing agent to flood the trash bin.

FIRE EXTINGUISHERS

A portable fire extinguisher is located on the forward side of the aft partition in the coat closet opposite the entry door. Another is installed beneath the copilot's seat.

WINDSHIELD WIPERS

The wipers have two speeds, one for light and one for heavy precipitation. After the control is moved to PARK to bring the wiper arms to their most inboard position, spring-loading returns the control to the OFF position.

CAUTION

Do not operate windshield wipers on dry glass.

CARGO RESTRAINT

Raytheon Aircraft offers an optional cargo restraint system. Any other restraint system used in this airplane must be approved by the FAA. Such approval is the sole responsibility of the owner/operator of the airplane.

AVIONICS SYSTEM

GENERAL

The following information describes the majority of the Avionic Systems supplied on the Beech 1900D airplane. Items not covered are those normally accepted as standards and require no information to operate. Even on these, equipment manuals are supplied with the airplane defining the methods of equipment operation. The system is designed such that the pilot and copilot have identical sets of equipment for flying the airplane. To accomplish this, Raytheon Aircraft has provided the airplane with dual Collins EFIS-84 Systems and dual Collins FDS-65 Flight Director Systems.

The avionic systems consist of power distribution, audio, standby EFIS power, compass, radar, standby gyro horizon, cabin briefer, electronic flight instrument system, weather radar, flight data recorder, and cockpit voice recorder systems as follows for a standard airplane configuration.

POWER DISTRIBUTION

AVIONICS POWER DISTRIBUTION

All avionics equipment may be turned on and off by the Avionics Master Switch. 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 Beech 1900D Airliner has multiple avionics buses to feed DC power to the various types of avionic equipment. To determine specifically what equipment is being fed from a specific bus or power source, refer to the wiring diagram entitled "Power Distribution Schematic" and the Circuit Breaker Electrical Connection Location tables located in this section.

During a normal engine starting sequence, as each generator is brought on line, the respective bus tie is closed. Therefore, assuming the avionics master switch is turned ON, all avionics systems will receive power from their respective buses under normal circumstances. Also, when running equipment checks on the ground with the external power switch ON and an APU connected, all avionics buses will be powered. In these instances, the bus ties are automatically closed.

An APU should be considered essential for running avionic equipment on the ground, since the avionic equipment and inverters would require over fifty amperes of current from the battery. This amount of current drain would deplete the battery in a short period of time.

It is desirable to have the avionic nose compartment doors removed to allow sufficient cross-ventilation and cooling of the equipment, particularly during practice sessions with the avionic equipment which exceed fifteen minutes in duration.

AC power is available from two 400 HZ inverters operated in parallel with the capability of any single inverter carrying the total AC load.

A standby EFIS power supply has been included to provide power to the pilot's EFIS system anytime the EFIS (Pilot only) power is reduced during engine starts in the air.

NOTE

Do not rely on the Standby EFIS Power System for sustained operation of the pilot's EADI and EHSI.

AUDIO SYSTEM

GENERAL

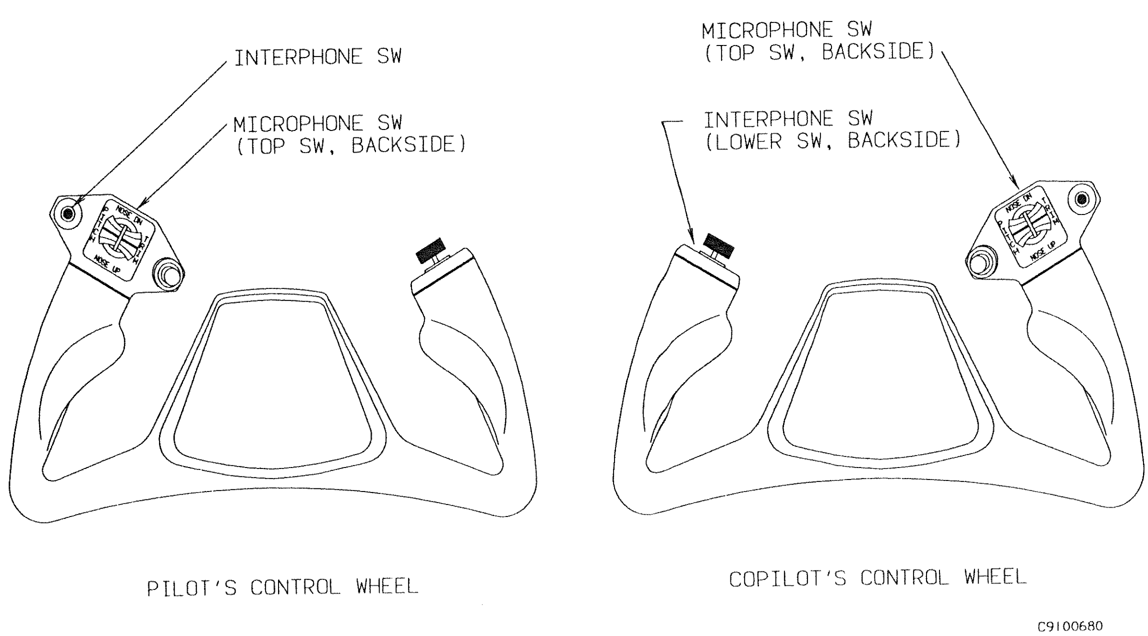
The audio system consists of independent audio control systems for the pilot and copilot. As part of the audio control system, push-to-talk (PTT) switches are provided on respective control wheels. In addition, mic and phone jacks are provided for use at each pilot's and copilot's station.

AUDIO CONTROL SYSTEM

The avionics installation has dual DB-437 Audio Systems which are totally independent of each other.

The following operating rules apply to the audio system. These rules will only be listed for the pilot's audio system. However, they apply equally to the copilot's audio system.

1. The speaker volume control regulates the speaker audio level.
2. The speaker switch (AUDIO SPKR) turns the cockpit speaker ON and OFF. With the speaker switch OFF, cockpit audio is inhibited from the speaker except for aural warning tones. On UE-262 and those airplanes modified by Raytheon Kit No. 129-3004-1, all cockpit audio, including aural warning tones, will be inhibited from the speaker when the speaker switch is in the OFF position. On these airplanes, the speaker switch must be left ON if headsets are not worn.
3. The phone volume control regulates the headphone audio level.
4. The headphones are operational at all times (as long as they are plugged into their jack).
5. The speaker and headphone audio channels are independent of each other and failure of one does not necessarily imply a failure of the other.
6. To select any audio source (e.g., Comm 1, ADF, etc.) turn ON the appropriate audio selector switch.
7. The switch placarded VOICE-BOTH-RANGE 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.
8. If the pilot's audio system has failed entirely, the pilot may still listen to audio through the copilot's speaker.
9. The communication transmit select switch, placarded COMM 1-COMM 2-PA, provides automatic selection of the corresponding receiver when a transmitter is selected. Speaker/phone volume levels may be set by the appropriate adjustment under the label volume.



PA function may also be selected. When selected, it provides corresponding cockpit microphone audio signal and general PTT to be routed to the passenger address system. The sidetone is generated in the corresponding headphone. PA audio levels may be adjusted by its appropriate control identified on the audio panel.

10. In addition, the audio level of each receiver may be adjusted for balance control by its appropriate placarded volume control potentiometer on the audio panel.
11. Hot mic provisions are provided by moving the selector switch placarded HOT INTPH/OFF to the HOT INTPH position. Interphone communications may be handled without using the PTT switch.

CONTROL WHEEL PUSH-TO-TALK (PTT) SWITCHES

Control wheel microphone (MIC) and interphone (INTPH) switches are located as shown.

GROUND COMMUNICATIONS/ELECTRIC POWER BUS

OPERATION

Operation of the Ground Communication Electric Power Bus is restricted to ground operation only. No authorization is extended to operate this system in any flight configuration.

The normal system operation is to turn off the battery and generator switches and push on the GND COMM PWR Switch. The annunciator will illuminate. System deactivation is accomplished by turning off the GND COMM PWR switch. The annunciator 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 DESCRIPTION

The ground communications electric power comes directly from the battery when selected by the pilot. Control of the system consists of a push on/push off solenoid-held annunciator switch located on the instrument panel. Circuit protection is provided by the RNAV MEMORY and STEREO fuses located near the battery in the wing. Activation of the system allows operation of the comm system connected to the STEREO fuse, and the audio system connected to the RNAV MEMORY fuse. Audio is provided to both the pilot and copilot headphones and speakers. Activation of the battery switch (or appropriate generator switch with engine running) will result in automatic disconnection of the ground communication bus from the comm system; however, the normal method for deactivation of the system is by pressing the GND COMM PWR switch.

STANDBY EFIS POWER SUPPLY (PILOT'S ONLY)

A standby EFIS power supply system is provided to prevent the pilot's EFIS displays from blanking during airstarts. The system consists of an EFIS auxiliary battery located in the nose avionics compartment, an EFIS AUX POWER control panel located on the pilots instrument panel, a 15-amp circuit breaker, placarded EFIS AUX BAT, located on the right circuit breaker panel, and a relay activated by the left squat switch which inhibits the ability of standby battery to power the pilot's EFIS on the ground. The relay is installed on UE-10 an after and those airplanes modified by Raytheon Kit No. 129-3000-1. The EFIS AUX POWER panel contains an ON - OFF-TEST switch, a HORN SILENCE button, and a cluster of annunciators which provide the following information to the pilot.

AUX ARM (Green)	Illuminates when the standby battery is selected on, the avionics switch is on, and the pilot's EFIS displays are being powered by the left generator bus.
AUX ON (Amber)	Illuminates when voltage to the pilot's EFIS has dropped below 18 VDC and operating power has switched to the standby battery power supply. A beeping warning horn will sound in conjunction with this annunciator.
AUX TEST (Green)	Illuminates when the ON-OFF-TEST switch is held to the TEST position. The test switch should not be held longer than 5 seconds, and released as soon as the AUX TEST annunciator illuminates. The annunciator may illuminate only momentarily, or as long as the switch is held to the TEST position. Either situation indicates the auxiliary battery has a sufficient charge. Refer to BEFORE ENGINE STARTING in Section IV, NORMAL PROCEDURES of the AFM.

The EFIS auxiliary battery is continually charged by the left generator bus through the left generator avionics bus.

A beeping warning horn is provided to alert the pilot that the standby battery is supplying power to the pilot's displays. This horn will activate in conjunction with the illumination of the AUX ON annunciator. The horn may be silenced by pressing the HORN SILENCE button. The horn and annunciator will activate during shutdown if the avionics switch is turned off before the auxiliary battery is turned off.

The standby EFIS power supply is intended for only short periods of use such as during the momentary drops in operating voltage that occur during airstarts. It should not be relied upon to maintain operation of the EFIS if the normal power supply should fail.

COMPASS SYSTEM

A dual independent compass system is installed on the Beech 1900D Airliner. The design of the system provides the pilot or copilot with compass information should any single gyro system fail. In such cases, system information lost may be obtained by using the reversionary switch.

The compass system installed uses two Collins MCS-65 systems. Control is provided through CCU-65 controls located on the pilot's and copilot's panels.

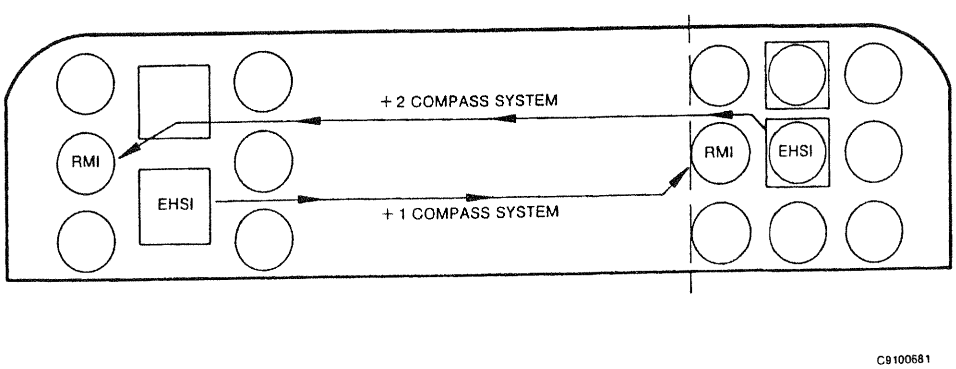
From an operational standpoint, this system requires 400 Hz electrical power from an inverter. Turning on the #1 or #2 inverter will provide power to the RMI system. In the unlikely event that both inverters fail, these systems would be inoperative. Each of them has the following major components.

1. 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.
2. Directional gyro: Once the gyro rotor is aligned with magnetic north, it will have a natural tendency to stay there, due to gyroscopic rigidity in space. This 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 (processes) the condition will be sensed, and the magnetic heading reference information from the slaving amplifier will drive the gyro rotor back into alignment with magnetic north.
3. Electronic Horizontal Situation Indicator (EHSI): The gyro heading information (which should be the same as magnetic heading) is sent to a compass card on the EHSI to display the magnetic heading to the pilot and copilot. This heading information is then sent from the EHSI 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.

COMPASS SYSTEM SLAVING

1. Slaving meter: The slaving meter on the CCU-65 Control compares the sensed magnetic heading at the flux sensor (system input) to the displayed magnetic heading at the EHSI (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.
2. Mode Switch: This is a push button switch used to select either the slaved mode (switch out) or the DG mode (switch pushed in) 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 DG 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. 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 EHSI to display the heading information from the directional gyro.
3. Slew Switches: These push button switches are spring loaded to the out position. When operating in the DG mode, the switches are used to manually change the directional gyro to the left or right, thus increasing or decreasing the displayed heading information. When operating in the slave mode, momentarily pushing either switch 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 processed excessively.



COMPASS SYSTEM CONNECTIONS

ELECTRONIC COPILOT ALTIMETER (IF INSTALLED) (ON AIRPLANES MODIFIED BY RAYTHEON KIT NO. 129-9018-9)

The altimeter is a solid-state unit with liquid crystal display. Barometric corrected altitude is displayed by a digital readout, plus a dial pointer display graduated in 20- and 100-foot increments. Barometric setting is displayed in both hecto-pascals and inches of mercury.

With lighting control turned on, dimming is controlled by the copilot instrument light rheostat. With lighting control turned off, dimming is controlled by a built-in light sensor. In the event of loss of power from the right generator bus, lighting control is provided by the light sensor.

An amber CODE flag, located in the upper part of the display, will be displayed with any fault that causes the altitude encoder output to be invalid. A white FAIL message replaces the digital altitude when the altimeter self-test detects a fault. This self-test is a continuous Built In Test (BIT) that monitors several altimeter parameters.

Power to the altimeter is provided from the right generator bus through the right generator avionics bus. The auxiliary battery is charged continually from the same bus and provides a minimum of 30 minutes power.

ENCODING ALTIMETER SELECTOR SWITCH

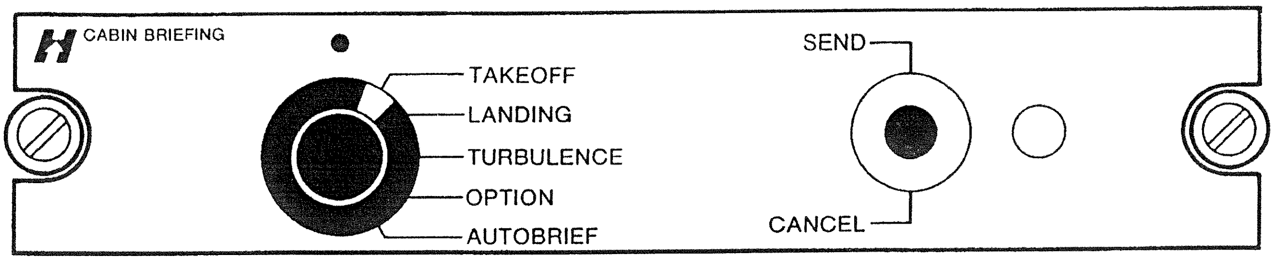
An encoding altimeter selector switch is located on the center avionics panel, and placarded ENCD ALTM 1/ALTM 2. This switch is used to select which altimeter, the pilot's (ALTM 1) or copilot's (ALTM 2), will provide encoding altimeter information to items such as the transponder, the traffic collision avoidance system (TCAS), the ground proximity warning system (GPWS), and the global positioning system (GPS).

RADAR SYSTEM

Various weather radar systems are available for installation in the Beech 1900D. The system consists of the transmitter/receiver unit located on the nose bulkhead and a control unit located in the pedestal. Radar data is interfaced such that it can be displayed on either the pilot's or copilot's EFIS-84 (EHSI) display. The operation of the radar is provided in the appropriate Operations Manual for the system installed and the integrated operation with the EFIS system is provided in the EFIS-84 (4-Tube) Operations Manual. These manuals are supplied with the airplane.

STANDBY ELECTRIC GYRO HORIZON SYSTEM

A standby electric gyro horizon system is provided as a back-up to the Pilot's EADI. The system consists of a two-inch standby horizon, an auxiliary battery located in the nose avionics compartment, two circuit breakers located on the right circuit breaker panel placarded STBY HRZN AUX BAT and STBY HRZN IND, and a STBY HORIZ PWR control panel located on the center avionics panel. This control panel is identical to the EFIS AUX POWER control panel provided for the standby EFIS power supply except that the warning horn provides a steady tone rather than a beeping tone. (See STANDBY EFIS POWER SUPPLY (PILOT'S ONLY) for a description of its operation.) The standby gyro is normally powered from the right generator bus through the right generator avionics bus. The auxiliary battery is charged continually from the same bus and provides a minimum of 30 minutes of standby gyro operation in the event power is lost from the right generator avionics bus. Refer to BEFORE ENGINE STARTING in Section IV, NORMAL PROCEDURES of the AFM for preflight test procedures. As with the EFIS auxiliary battery, if the standby gyro auxiliary battery is not turned off prior to turning the avionics switch off, the warning horn and annunciator will be activated.



CABIN BRIEFER

CABIN BRIEFER

The airplane is provided with a Heads Up Technologies Cabin Briefer HUCAB-( ). The briefer provides standard cabin briefings following the AFM requirements. In addition, the briefer can be supplied with custom briefings as desired by the operator. The system operates in the following modes; Standby (defined by the dot on the control panel), Takeoff, Landing, Turbulence, Option, Auto-brief, and Play/Cancel. The above modes provide the following when selected:

(.)	*System standby.*
TAKEOFF	*Before takeoff briefing.*
LANDING	*Before landing briefing.*
TURBULENCE	*Inflight turbulence briefing.*
OPTION	*Allows programming of one optional briefing.*
AUTO BRIEF	*(NOT FUNCTIONAL)*
SEND/CANCEL	*After selection is made, press SEND to begin briefing. Press CANCEL to stop a briefing at any time.*

The audio from the briefer is supplied to the cabin through the airplane Standard Paging System.

ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

GENERAL

The airplane is equipped with a Collins EFIS-84 (4-Tube) Electronic Flight Instrument System. In this system, attitude and navigation information is displayed on color crt's which serve the functions of EADI (Electronic ADI) and EHSI (Electronic HSI).

Additional information, such as weather radar, NAVAID/ waypoint locations, FCS mode annunciation, autopilot/yaw damper engage status, attitude comparitor warnings, decision height, and diagnostic messages may also be displayed.

The system gives a pilot the capability to display information available in the area of his central scan, by allowing him 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.

This manual includes descriptions of system components, typical operating procedures, and annunciations and warnings in the system.

The EFIS-84 Electronic Flight Instrument System uses input data from the following sources:

- VOR/localizer/glideslope
- Vertical gyro and compass system
- Radio altimeter
- Distance measuring system
- Flight control system
- Long range navigation system (if installed)
- Weather radar system
- Automatic direction finding system

The EFIS-84 system, in turn, uses these inputs to display V-bar or crosspointer steering commands and other navigational data for flight control purposes as well as information of an advisory nature.

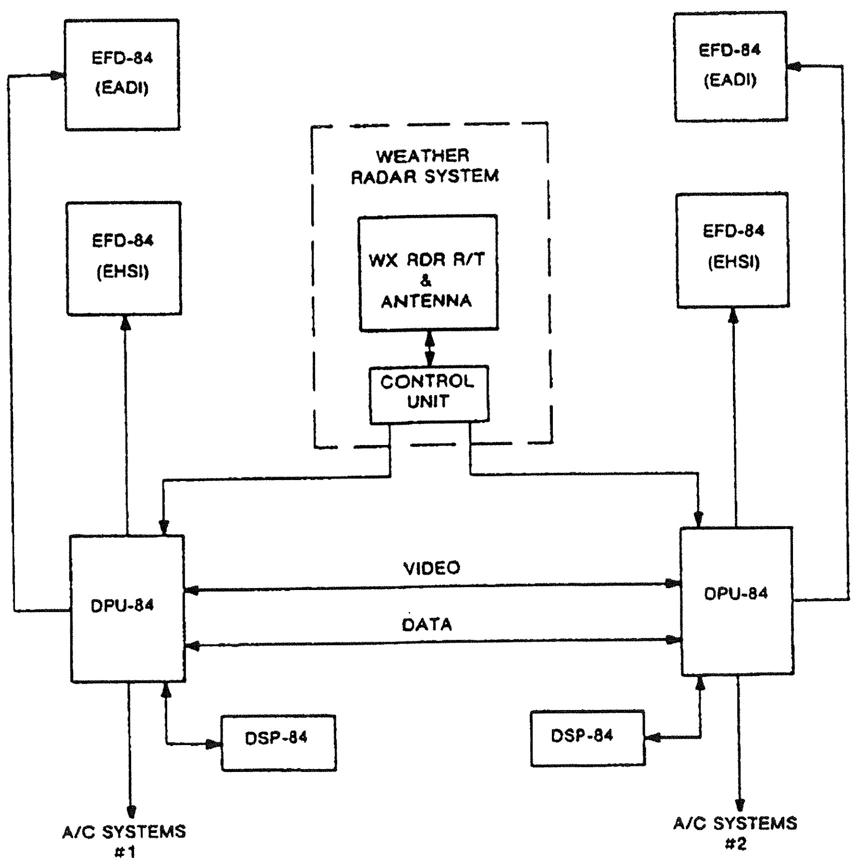
Extensive monitoring and comparitor circuits provide warning flags and other types of data flagging techniques to indicate possible equipment malfunctions.

NOTE

The EFIS-84 Electronic Flight Instrument System will provide normal operation if the voltage on the +27.5 VDC bus drops to +18 VDC. If the bus voltage drops below +18 VDC, the EADI's and EHSI's will blank. Normal operation is automatically restored when the voltage rises above +18.4 VDC. A copyright message appears on the EADI and EHSI for approximately 20 seconds each time the EFIS-84 is powered up.

The operation of the Collins EFIS-84 4-Tube System is provided in the Collins EFIS-84 (4-Tube) Pilot's Guide. The Pilot's Guide is furnished with the airplane and should be used to gain an understanding of the total system operation and capability.

The following illustration is a simplified functional block diagram of the EFIS-84 Electronic Flight Instrument System (4-tube version).



EFIS-84 ELECTRONIC FLIGHT INSTRUMENT SYSTEM (4-TUBE VERSION)

C9201875

SYSTEM DESCRIPTION

The Collins 4-Tube EFIS-84 Electronic Flight Instrument System (FIS) consists of four panel-mounted EFD-84 Electronic Flight Displays, two console-mounted DSP-84 Display Select Panels, two remote-mounted DPU-84 Display Processor Units, one pedestal-mounted EFIS Power Control Panel, and one pedestal-mounted EFIS Reversionary Select Panel. A weather radar system such as the Collins WXR-350 Weather Radar System is used with the EFIS-84, and detectable weather is displayed on the EHSI's. To assist in a clearer understanding of how these units are interconnected, illustrations of the EFIS Power Control Panel and EFIS Reversionary Panel are shown later.

EADI (ELECTRONIC ATTITUDE DIRECTOR INDICATOR)

The EADI is a multicolor crt that presents a display of aircraft attitude and flight control system steering commands, VOR, localizer, LNAV deviation, and glideslope deviation.

Flight control system mode annunciation, autopilot engage annunciation, attitude sensor annunciation, marker beacon annunciation, radio altitude, decision height set and annunciation, and comparitor warnings are also displayed.

Two fast erect switches are located on the instrument panel; one on the pilot's side to fast erect the pilot's vertical gyro, and one on the copilot's side to fast erect the copilot's vertical gyro. The switches are placarded VG FAST ERECT (white)/ ON (green). Pushing the switch will cause the ON annunciator to illuminate and the vertical gyro to enter the fast erect mode. The switch may then be released (ON annunciator extinguished) and the vertical gyro will continue in the fast erect mode at the maximum rate of 5°/minute until the system senses that it is within 1 1/2° of vertical. While the gyro is fast erecting, the EADI will be flagged and blanked; airplane attitude must be maintained using the standby attitude indicator or the opposite EADI. The airplane must be maintained in a level, unaccelerated flight condition while the fast erect mode is active. Illumination of the fast erect switches is controlled using the rheostat placarded ANN PUSH BRT located on the avionics control panel.

EHSI (ELECTRONIC HORIZONTAL SITUATION INDICATOR)

The EHSI is a multicolor crt that presents a plan view of the aircraft's horizontal navigation situation.

Information displayed includes magnetic heading, DG mode annunciation, selected heading, selected VOR and localizer deviation, navigation sensor annunciation, digital selected course/desired track readout, heading sensor annunciation, ILS and heading comparitor warnings, to/from information, back course localizer annunciation, distance to station/ waypoint, glideslope, ground speed, time-to-go, elapsed time, course information and sensor annunciation from a second navigation sensor, weather radar target alert, and two bearing pointers that can be driven by VOR, ADF, or LORAN sensors as selected on the display select panel. The EHSI can also be operated in an ARC (sector) format or a MAP (sector with map) format. Either of these two formats may be selected with or without weather radar information included on the display.

CAUTION

Operating the EFIS-84 Electronic Flight Instrument System at maximum brightness (dim controls fully clockwise) for extended periods of time may eventually result in a condition known as "imprinting" on the crt. Imprinting is evidenced by being able to see an image on the crt when the crt is turned off, or by being able to see an image other than the one desired. This last condition usually occurs when, for example, a crt that was used as an EADI is removed from service and reinstalled as an EHSI, or vice versa.

Proper management of the intensity controls minimizes the chances of imprinting and helps to maximize the service life of the crt. Collins recommends that the intensity controls not be left in the maximum bright (full clockwise) position unless cockpit conditions (such as direct sunlight, etc.,) dictate that maximum brightness be used. When those conditions requiring maximum brightness are no longer required, adjust the intensity controls to lower the brightness of the crt's to a comfortable viewing level.

There are other steps the pilot can take that will contribute greatly to extending the life of the crt's and minimize the chances of imprinting. For example, adjust the intensity controls for minimum brightness (fully counterclockwise) while at the terminal or waiting for passengers or during other periods of activity on the ground when the EFIS is not being used.

DSP (DISPLAY SELECT PANEL)

The display select panel provides EHSI format selection, navigation sensor selection, bearing pointer selection, weather radar selection, navigation data selection (ground speed, time-to-go, elapsed time), and the selection of course information from a second navigation sensor.

A DH SET Knob that sets the decision height shown on the EADI is also provided. The DH SET knob includes a TST (test) button for initiating radio altimeter test. The DSP also provides a TIMER SET knob for setting the count-up or count-down timer. The TIMER SET knob includes a S/S (start/stop) button for controlling the timer. Additionally, course, course direct to, heading, and heading sync are selected from the DSP.

DPU (DISPLAY PROCESSOR UNIT)

The display processor unit provides sensor input processing and switching, the necessary deflection and video signals, and power for the electronic flight displays. The DPU is capable of driving two electronic flight displays with different deflection and video.

EFIS POWER CONTROL PANEL

The EFIS Power Control Panel is located on the pedestal. The panel contains four EFIS power switches, which are combination push-button/annunciator switches, each incorporating a split annunciator. When each switch is pushed on, the green ON annunciator on the top half of the switch is illuminated. When each switch is pushed off, a yellow OFF annunciator on the bottom half of the switch is illuminated. On those airplanes modified by Raytheon Kit No. 129-3012, these switches are replaced by four lever-locked toggle switches. The toggle switches are pushed forward to turn the system on, and lifted and pulled aft to turn the system off. These switches contain no annunciators.) The four switches control electrical power to the pilot's and copilot's Electronic Flight Displays (EADI and EHSI), Display Select Panel (DSP), and Display Processor Units (DPU). When the switches are placed in the on position, the selected equipment is turned on, along with equipment cooling fans.

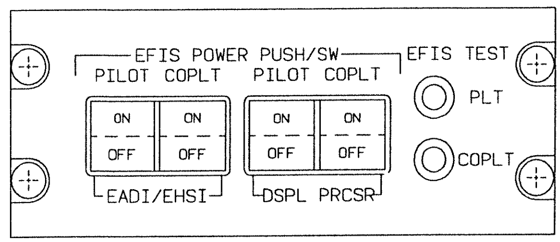
In addition to power control switches, the panel contains an EFIS Test Switch for the pilot and copilot system. The self-test sequence for either the pilot or copilot is initiated by actuating the EFIS TEST switch. When self-test is first initiated, an increment of 10 degrees is added to the current values of pitch and roll, and 20 degrees is added to or subtracted from heading. Pitch up and right roll are added to the pilot's side, and pitch down and left roll are added to the copilot's side. The word TEST, in red, appears on the pilot's EADI while the test is in process. The addition/subtraction of the differential 10- and 20-degree increments causes the comparitor logic in the DPU to give PIT and ROL (attitude) and HDG (heading) comparitor warning messages.

NOTE

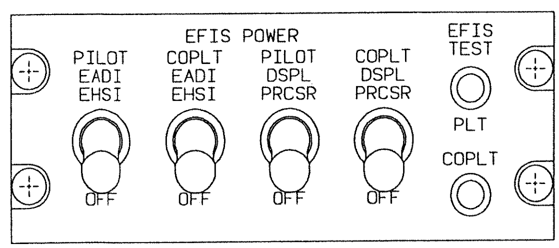
The EFIS self-test is the only indication the pilot has that the comparitors are operating properly and have not been disabled. For this reason, Collins recommends that the EFIS self-test be performed to check the comparitor operation as part of the airplane preflight. As an alternative, the comparitors should be checked (EFIS self-test performed) before the first flight of the day.

If the self-test switch is held for longer than 4 seconds, the tubes blank and all flags associated with the EADI and EHSI displays are brought into view. When the self-test switch is released, the displays return to normal operation.

The following is a view of the EFIS Power Control Panel:



(SERIALS UE-1 & AFTER WITHOUT KIT 129-3012 INSTALLED)



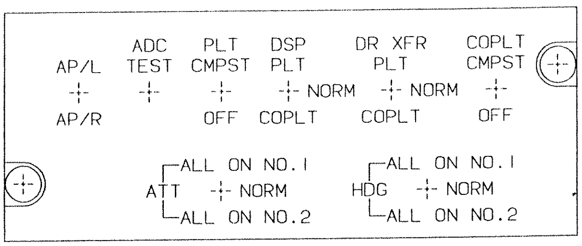
(SERIALS UE-1 & AFTER WITH KIT 129-3012 INSTALLED)

EFIS POWER CONTROL PANEL

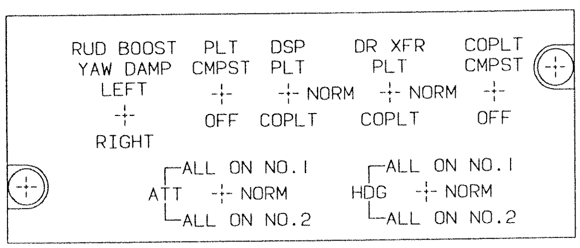
UE03C971286 C

EFIS REVERSIONARY SELECT PANEL

The EFIS Reversionary Select Panel is provided in two forms, one for the non-autopilot airplane configuration and one for the autopilot-equipped airplane. The following depicts each configuration:



AUTOPILOT REVERSIONARY SELECT PANEL



NON-AUTOPILOT REVERSIONARY SELECT PANEL

C9100810

The panels (with the exception of the AP/L, AP/R, and ADC Test Select Switch on the autopilot version versus the Rud Boost/Yaw Damp Left/Right Select Switch on the non-autopilot version) perform the same Reversionary Select functions.

On the autopilot Reversionary Panel, the AP/L and AP/R switch allows transfer of the active autopilot system to the pilot (L) or copilot (R) at the pilot's option. The ADC TEST switch allows testing of the Air Data Computer for proper system operation and annunciates faults.

On the non-autopilot Reversionary Panel, the RUD BOOST/ YAW DAMP L/R Switch, (if installed), allows transfer of the yaw damp/rudder boost active system from the pilot's Flight Director Computer (L) to the copilot's Flight Director Computer (R) at the pilot's option.

The remainder of the panel switches perform the following functions:

PLT CMPST or COPLT CMPST	Allows presentation of composite data on both the EADI and EHSI when the composite mode is selected.
DSP PLT/ NORM/ COPLT	Allows the pilot or copilot to use his own DSP or transfer operation to the pilot only or copilot only, as selected.
DR XFR PLT/ NORM/COPLT	Allows the pilot or copilot to use his own drive unit or transfer operation to the pilot's drive only, or copilot's drive only, as selected.
ATT ALL ON No. 1/NORM/ ALL ON No. 2	Allows the pilot or copilot to use his own ATT system or transfer both pilot's and copilot's systems to No. 1 or No. 2 ATT system only, as selected.
HDG ALL ON No. 1/NORM/ ALL ON No. 2	Allows the pilot or copilot to use his own HDG system or transfer both pilot's and copilot's systems to No. 1 or No. 2 HDG system only, as selected.

WEATHER RADAR PANEL

The weather radar panel provides the mode control, range selection, and other system operating controls to allow the display of weather radar information on the EHSI's. If a Collins WXR-350 Weather Radar System is installed, an annunciator is provided in the pedestal annunciator group which tells the pilot when the radar system is on. The annunciator (placarded RDR PWR ON) operates only when the airplane is on the ground.

To display detected precipitation, the weather radar system must be in a weather detection mode and the ARC/WX or MAP/WX mode must be selected on the applicable DSP (pilot's or copilot's).

BENDIX/KING KLN 88 MULTI-CHAIN LORAN NAVIGATION SYSTEM (IF INSTALLED)

The KLN 88 is a long-range, LORAN-C based navigation system which provides the pilot with present-position information and display guidance information based upon a pilot-programmed flight plan. The basic system consists of a panel-mounted unit which contains the LORAN-C sensor, the navigation computer, a CRT display, and all the controls required to operate the unit. It also houses the data base cartridge which plugs directly into the back of the unit.

The KLN 88 system has analog outputs which drive the left/ right deviation bar of the pilot's and copilot's EFIS-84 EHSI. In addition, the system is capable of being coupled to the Collins FCS-65 Autopilot and Flight Director. Outputs to remote annunciators to indicate the status of certain KLN 88 functions are also provided. The system uses its present-position information to determine crosstrack error, distance-to-waypoint, ground speed, track angle, time-to-waypoint, bearing-to-waypoint, and advisory VNAV guidance.

The internal data base of the KLN 88 contains a vast amount of information on airports, navaids, intersections, special use airspace, and other aeronautical information pertinent to navigation. A periodic update of the data base is made by replacing the obsolete cartridge with a current one. In addition to the published data base, up to 250 user-defined waypoints and up to 9 flight plans may be created and stored. The KLN 88 contains an internal lithium battery to retain the stored information.

FLIGHT DATA RECORDER (FDR)

The airplane is equipped with a LORAL Fairchild F1000 FDR.

The Digital Flight Recorder, located in the aft section of the airplane, converts analog data into digital data and records the information in its memory. The recorder will continuously record and retain the last 25 hours of flight data. It is powered by 115 volts AC.

A RECORDER FAULT light provides the pilot with an indication of system operation. The fault light should extinguish approximately 5 seconds after initial power application to the system. Reillumination of the fault light after 5 seconds indicates a possible problem in the recorder or incorrect input data to the recorder. There are no controls associated with the recorder and its operation is completely automatic.

41A remote trip and date encoder (if installed) allows input of trip number and date to the recorder as desired. The encoder also incorporates two push-button switches; the REPEAT button initiates a 15-minute coding cycle in the recorder, the EVENT button can be used to mark the recording data when desired. A light on the encoder panel illuminates during the coding cycle. The meter on the panel has no function for the recorder.

Pressing the REPEAT button (if installed) to initiate the 15-minute coding cycle does not interfere with the immediate recording of FDR data from the moment the button is pressed, or the correct identification of the date/flight number for which the data is recorded. No further action is required and the pilot may proceed without delay with the remainder of the check list.

The airplane is provided with a LORAL Fairchild manual covering the operation and maintenance of the recorder.

COCKPIT VOICE RECORDER (CVR)

The airplane is equipped with a L3 Communications Fairchild Model A100A, A100S or FA2100 cockpit voice recorder. The A100A and A100S CVR's have a 30-minute recording capacity, while the FA2100 CVR's 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 floor boards and forward of the aft cargo area. The control unit is located in the cockpit pedestal. The area microphone is located in the glareshield to the right of the fire extinguisher switch, or other strategic location. Electrical power to the recorder is provided by the triple fed bus.

Input to the CVR comes from the following four sources:

Cabin Briefer	What the pilot or copilot transmit over the cabin pager, or transmit over the automatic cabin briefer.
Pilot's Audio	Whatever the pilot hears in the headset or speaker, or transmits over any of the mic's, will be recorded.
Copilot's Audio	Whatever the copilot hears in the headset or speaker, or transmits over any of the mic's, will be recorded.
Area Mic	Voices and other sounds in the cockpit. (Stronger sounds may mask weaker ones.)

The CVR is capable of reproducing the recorded audio data in one of the following formats:

A100A, A100S, 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 2 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 cabin pager audio data. • Area Mic audio data.

The control unit contains the test switch, erase switch, indicating meter, and headset jack. The test switch and indicating meter provide a test of the recorder. The headset jack provides a means of monitoring all audio data that is being input to the CVR to ensure that the recorder is receiving the proper audio signals. Pressing the test switch should result in the following indications if the recorder is functioning properly:

A100A and A100S (30-Minute Recorder)	A 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 tone and needle position will remain constant as long as the switch is pressed.
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.

The erase switch may be used to erase the entire recording after a routine flight, 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 tone in the headset that occurs when the erase switch is released. The tone will last for approximately 5 seconds.

ALLIEDSIGNAL MK-VI GROUND PROXIMITY WARNING SYSTEM (IF INSTALLED)

The Mark-VI GPWS system includes a GPWS warning computer, a GPWS air data computer, cockpit annunciators and switches, and inputs from the airplane's radio altimeter, left landing gear downlock switch, copilot's pitot/static system, glideslope deviation signal, and cockpit flap control switch. The system is powered from the Left Generator bus and is protected by a 2-amp circuit breaker placarded GPWS located on the copilot's circuit breaker panel. This circuit breaker also protects the GPWS air data computer. Voice messages from the GPWS will not be heard over the cockpit speakers on UE-262, or on airplanes modified by Raytheon Kit No. 129-3004-1, unless the AUDIO SPKR switch is on.

The following switches and annunciators are located on the instrument panel:

- A combination Press-to-Test switch and red warning annunciator, placarded GPWS, located in front of each pilot.
- A combination Push-to-Cancel switch and amber caution annunciator, placarded BELOW G/S, located in front of each pilot.
- An amber caution annunciator, placarded G/S CANCLD, located below the pilot's IVSI.
- An amber caution annunciator, placarded GPWS INOP, located below the pilot's IVSI.
- A combination guarded switch and blue annunciator, placarded GPWS FLAP OVRD, located below the pilot's IVSI.

Illumination for the above annunciators is controlled by the ANN PUSH BRT rheostat located on the avionics panel.The GPWS and the BELOW G/S annunciators cannot be dimmed.

The MK-VI GPWS system will illuminate the red GPWS warning annunciator and provide the following voice messages if the airplane penetrates a predetermined envelope above the terrain. Refer to the MK-VI GPWS Pilot's Guide for specific details.

- "SINK RATE" repeated every 3 seconds followed by "PULL-UP" repeated continuously (Mode 1 - Excessive rates of descent in relation to AGL altitude.
- "TERRAIN, TERRAIN" heard once followed by "PULL-UP" heard continuously (Mode 2A - Terrain rising excessively fast underneath the airplane with the gear and flaps up.)
- "TERRAIN, TERRAIN" (Mode 2B - Terrain rising excessively fast underneath the airplane with the gear and flaps down.)
- "DON'T SINK" repeated every 3 seconds (Mode 3 - Excessive altitude lost after takeoff or missed approach.)
- "TOO LOW TERRAIN" or "TOO LOW GEAR" repeated every 3 seconds (Mode 4A - Insufficient terrain clearance during cruise.)
- "TOO LOW TERRAIN" or "TOO LOW FLAPS" repeated every 3 seconds (Mode 4B - Insufficient terrain clearance during approach with the landing gear down.)
- "TOO LOW TERRAIN" repeated every 3 seconds (Mode 4C - Insufficient terrain clearance during takeoff or missed approach due to continued descent or terrain rising faster than the airplane.)

The MK-VI GPWS system will illuminate the amber BELOW G/S annunciator (Mode 5) and provide a "GLIDESLOPE" voice message if the airplane descends below the glideslope sufficiently to produce a fly up command of 1.3 dots or more. The Mode 5 function is armed only when the following conditions are met:

- An ILS frequency is being received on NAV No. 1.
- The landing gear is down.
- The airplane is below 925 feet AGL.
- G/S CANCLD function is not selected.

Any of the following optional voice messages (Mode 6) may be installed. There are no annunciators associated with these messages.

- "FIVE HUNDRED" - Occurs when the airplane descends through 500 feet AGL.
- "MINIMUMS, MINIMUMS" - Occurs when the airplane descends through the decision height set by the pilot.

The GPWS annunciator/Press-to-Test switch is used to test the GPWS system prior to the first flight of the day. Either the pilot's or copilot's switch may be used to initiate the test function. If the GPWS INOP annunciator is illuminated on the ground after the avionics bus is powered, the test switch may be used to troubleshoot the system. With the switch activated, one or more of the following voice messages may be heard; "RADIO ALTITUDE FAULT", "GLIDESLOPE FAULT", or "BARO ALTITUDE FAULT". These messages indicate that the problem is a failure in a system providing input to the GPWS computer, or a broken wire between the system and the computer.

The BELOW G/S annunciator/Push-to-Cancel switch may be used to cancel the Mode 5 function either prior to an alert, if certain criteria are met, or during an alert. Push and hold the switch momentarily until the G/S CANCLD annunciator illuminates, then release the switch. Once the function has been canceled, it cannot be reinstated using the switch. The Mode 5 will reset itself automatically when any of the following occur.

- The airplane descends below 50 feet AGL.
- The pilot's NAV receiver is cycled to a VOR frequency, then back to an ILS frequency.
- The airplane climbs above 1900 feet AGL.

The Push-to-Cancel switch will not function unless all of the following criteria are met.

- The pilot's glideslope flag is pulled.
- The landing gear is down.
- The airplane is below 1900 feet AGL.

The guarded flap override switch, placarded GPWS FLAP OVRD, may be used to inhibit the Mode 4B "TOO LOW FLAPS" voice message and the GPWS annunciator when situations require a landing to be made with less than full flaps. Lift the guard and press and hold the switch momentarily until the blue annunciator illuminates, then release the switch. The switch will function only when the airplane is above 50 feet AGL. Once engaged, the Flap Override function may be turned off by pushing the switch again. It will automatically be reset to off when the airplane descends below 50 feet AGL. The FLAP OVRD switch may also be used to desensitize the GPWS system in the following modes.

- Mode 1 - Increases the rate of descent required to activate the alerts by 300 fpm.
- Mode 3 - Increases the loss of barometric altitude required to activate the alert from 10% to 20% of the radio altitude gained.

Refer to the Mark-VI Pilot's Information Guide for further information.

BFGOODRICH TRAFFIC ALERT AND COLLISION AVOIDANCE SYSTEM I (TCAS791) (IF INSTALLED)

WARNING

The TCAS791 does not provide protection from aircraft that do not have an operating transponder. The TCAS791 is only an aid in detecting other traffic and provides a means for the pilot to visually acquire and avoid aircraft which may impose a collision threat. Evasive maneuvers should not be made based solely on the TCAS display.

The TCAS791 system installed on the Beech 1900D includes a transmitter receiver computer (TRC), a control display unit (CDU), located in the lower center of the instrument panel, a directional antenna mounted on top of the fuselage above the cockpit, an L-band omnidirectional antenna mounted on the aft belly of the fuselage, and inputs from the pilot's or copilot's encoding altimeter, the No. 1 directional gyro, the radio altimeter, the left squat switch, and the left main gear downlock switch. The system is powered from the left generator bus and is protected by a 7.5 amp circuit breaker, placarded TCAS, located on the copilot's circuit breaker panel. The illumination of the control buttons on the CDU is controlled by the Avionics Panel Lights rheostat located on the overhead panel. Voice messages from the TCAS791 will not be heard over the cockpit speakers on UE-262, or on airplanes modified by Raytheon Kit No. 129-3004-1, unless the AUDIO SPKR switch is on.

The TCAS791 is an active-only system that interrogates the transponders of airplanes in the surrounding airspace. The system is capable of tracking up to 35 intruder aircraft simultaneously and displaying the eight most threatening ones on the CDU. The CDU displays traffic information in color coded symbols and text to indicate the intruder's range, bearing, relative altitude, and vertical speed direction if over 500 fpm. An aural traffic advisory, "TRAFFIC, TRAFFIC" is provided when the TCAS system predicts that an intruder airplane may present a collision threat. The system provides optimum performance only when the intruder airplanes are reporting their altitude.

The CDU is controlled using the following items:

- A power/brightness knob placarded DIM/OFF - Rotating the knob clockwise turns the system on and increases the brightness of the CDU.
- A range button, placarded RNG - Pressing the button selects maximum ranges of 5 nm, 10 nm, and 20 nm when airborne. The RNG button also provides the following functions during ground operations on those airplanes that have been modified by BFGoodrich Service Memo 104.

  Pressing the RNG button when in the Standby Mode switches the TCAS into the Above Display Mode and the 10 nm range.

  The RNG button may be used to select the 5 nm or 10 nm range when not in the Standby Mode.

  Pressing and holding the RNG button switches the TCAS back into the Standby Mode.
- A test button, placarded TEST - Pressing the test button initiates a self test of the system. The test function is available only when the airplane is on the ground. The system can be tested by either pressing and releasing the button, or by pressing and holding the button. If the button is held, the test screen will be shown until the button is released, unless a failure of the system is

detected. The test button also provides the following functions on those airplanes that have been modified by BFGoodrich Service Memo 104.

The Test function works only when the system is in the Standby Mode.

Pressing the test button repeatedly when not in the Standby Mode cycles the altitude display mode through the following sequence: Above (ABV), Normal (NRM), Below (BLW), Normal (NRM), Above (ABV), etc.

- The system must be tested prior to the first flight of the day in accordance with Section IV, NORMAL PROCEDURES of the AFM. If the test is satisfactory, a voice message "TCAS TEST PASSED" will be heard. If the test is unsatisfactory, a voice message "TCAS TEST FAILED" will be heard and the screen will display TCAS FAILED.

The CDU display will vary slightly depending on the level of software installed. Systems installed in those airplanes that have been modified by BFGoodrich Service Memo 104, differ from the original system as follows:

- After the pilot initiated self-test is accomplished, the display reverts to a STANDBY screen rather than a 5 nm screen. To switch out of the STANDBY mode, the RNG button must be pushed. This will cause the screen to switch to the above (ABV) display mode and the 10 nm range. The display can be switched back to the standby mode (only during ground operations) by pressing and holding the RNG button. The display will automatically switch to the standby mode 24 seconds after landing.
- Three altitude display modes are available; the (ABV) display mode, the normal (NRM) display mode, and the below (BLW) display mode.

Normal Display Mode (NRM)	Traffic detected within ± 2700 feet of your aircraft is displayed on the CDU. (This is the only mode available with the original software.)
Above Display Mode (ABV)	Traffic detected within +9000 feet and -2700 feet of your aircraft is displayed on the CDU.
Below Display Mode (BLW)	Traffic detected within +2700 feet and -9000 feet of your aircraft is displayed on the CDU.

- Display modes can be selected on the ground or in the air by repeatedly pressing the TEST button until the desired mode is displayed.
- A TCAS FAILED/Barometric Input screen is available. This screen will be displayed any time the barometric input from the airplane's encoding altimeter is lost. If such a failure is intermittent, the TCAS791 will automatically return to normal operation when the barometric altitude input is restored.
- The RNG button may be used to select the 5 or 10 nm range settings during ground operations. The RNG button is inhibited on the ground on those units with the original software installed, and only the 5 nm range is available.

Maintenance codes (Example: "Maint. Code: Heading") can appear at the bottom of the screen during ground operations only. The TCAS791 can be used when a maintenance code is displayed, but should be repaired as soon as possible.

The TCAS791 can display the following symbols for traffic during ground operations, when not in Standby, and during flight:

- A solid yellow circle (Traffic Advisory): Traffic within 30 seconds (20 seconds for non-altitude reporting airplanes) of their closest point of approach (CPA) and all traffic within.55 nm and ± 800 feet. Below 2000 feet AGL this criteria is reduced to 20 seconds (15 seconds for non-altitude reporting airplanes) of their closest point of approach and within 0.2 nm and ± 600 feet vertically. This traffic advisory is accompanied by a voice message of "TRAFFIC, TRAFFIC". The voice message is transmitted over the cockpit speakers and headphones at a preset level which is not adjustable in the cockpit. (On UE-262, the voice message will not be heard over the speakers unless the AUDIO SPKR switch is on.) The voice message is inhibited when the airplane is below 400 feet AGL or when a GPWS voice message occurs.
- A solid yellow semi-circle (Out of Range Traffic Advisory): Traffic which meets the Traffic Advisory criteria but is beyond the displayed range. This will be accompanied by a voice message of "TRAFFIC, TRAFFIC."
- A solid white diamond (Other Traffic): Traffic that does not generate a Traffic Advisory and is within 4 nm and ± 1200 feet of your airplane. Non-altitude reporting airplanes are considered to be at your altitude.
- An open white diamond (Other Traffic): Traffic that is within the selected range and altitude mode (if available) that is not generating a Proximity Advisory or a Traffic Advisory. Non-altitude reporting airplanes are considered to be at your altitude.
- No Bearing Traffic Advisory (Example: "TA 1.1-06 ↑" shown in yellow): Traffic which meets the Traffic Advisory criteria, but for which no bearing is available. The traffic advisory shows the intruders range in nm, a data tag, and a vertical trend arrow. This will be accompanied by a voice message of "TRAFFIC, TRAFFIC."
- No Bearing, No Altitude Traffic Advisory (Example: "TA 2.6" shown in yellow): Traffic which meets the Traffic Advisory criteria, but for which no bearing or altitude is available. The traffic advisory shows the intruders range in nm. This will be accompanied by a voice message of "TRAFFIC, TRAFFIC."

The following symbols are used in conjunction with the traffic symbols:

- Vertical Trend Arrow: A vertical trend arrow will be shown to the right of the traffic symbol to indicate that it is descending ↓ or ascending ↑ at a rate greater than 500 fpm. No arrow is shown for non-altitude reporting airplanes.
- Data Tag (Example: + 18): A two digit number representing the relative altitude, in hundreds of feet, of the intruder airplane. A positive number is displayed above the traffic symbol and a negative number is displayed below the traffic symbol. If the intruder is at the same altitude, 00 will be displayed above the traffic symbol if the intruder closed from above, or below the traffic symbol if the intruder closed from below.

Refer to the BFGoodrich TCAS791 Pilot's Guide for further information.

ARTEX 110-4-002 EMERGENCY LOCATOR TRANSMITTER (ELT) WITH REMOTE SWITCH (IF INSTALLED)

The Artex 110-4-002 Emergency Locator Transmitter (ELT) System 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. The switch is lever locked in the ARM and the ON positions. Neither this switch, nor the switch on the ELT transmitter, can be positioned to prevent the automatic activation of the ELT transmitter. The System is independent from other airplane systems except for the transmit light, which is hot-wired to the airplane battery, and the edge lit panel which is controlled by the Side Panel rheostat located on the overhead panel.

The ELT will automatically activate during a crash and transmit a sweeping tone on 121.5 and 243.0 Mhz. This activation is independent of the remote switch setting or availability of aircraft power. The remote switch is installed to perform the following functions:

- Test the ELT.
- Deactivate the ELT if it has been inadvertently activated. by the "G" switch.
- Activate the ELT during an in-flight emergency if an off-airport landing is anticipated.
- Activate the ELT after an off-airport landing, if the impact did not automatically activate it.

The ELT should be tested every three months. The test consists of turning the unit on and then resetting it using the following procedures.

- Tests should be conducted between the times of on-the-hour until 5 minutes after the hour.
- Notify any nearby control towers.
- Provide power to an aircraft radio and tune it to 121.5 Mhz.
- Place the ELT remote switch to ON. Wait for at least 3 sweeping tones on the aircraft radio, which will take about 1 second, then return the switch to ARM.
- The test is successful if the sweeping tones are heard and the yellow transmit light next to the switch blinks immediately. If there is a delay in the illumination of the transmit light, the system is not working properly.

If the ELT should be inadvertently activated by the "G" switch, the transmit light next to the switch will blink. The ELT can be deactivated by momentarily placing the remote switch to ON and then back to ARM.

Raytheon Aircraft

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