AIRFRAME

STRUCTURE

The BEECHCRAFT Super King Air B200/B200C is an all-metal, low-wing monoplane. It has fully cantilevered wings, and a T-tail empennage.

SEATING ARRANGEMENTS

The pilot and copilot seats are mounted in a separate forward compartment. Various configurations of passenger chairs and two- or four-place couch installations may be installed on the continuous tracks mounted on the cabin floor. One or two fold-up seats may be installed in the aft cabin area. The toilet is also equipped for use as a seat. Seating for up to 15 persons, including crew, is available. For additional information, refer to the "Cabin Arrangement Diagram" in Section VI, WEIGHT AND BALANCE/ EQUIPMENT LIST.

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 airplane is equipped with conventional dual controls for the pilot and copilot. 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. Rudder, elevator, and aileron trim are adjustable with controls mounted on the center pedestal. 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 handwheel 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

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 a PITCH TRIM circuit breaker in the FLIGHT group on the right side panel. The ELEV TRIM switch must be ON for the system to operate. 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. A before take-off check of both dual element thumb switches should be made by moving each of the four switch elements individually. No one switch element should activate the system; moving the two switch elements on either the pilot's or copilot's control wheel in opposite directions should not activate the system - only the simultaneous movement of a pair of switch elements in the same direction should activate the electric elevator-trim system.

A bi-level, push-button, momentary-on, 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. If an autopilot is installed, depressing either trim-disconnect switch to the first of the two levels disconnects the autopilot and the yaw damp system; depressing the switch to the second level disconnects the autopilot, the yaw damp system, and the electric elevator-trim system. If an autopilot is not installed, depressing the switch to the first level does not do anything, since the yaw damp system is controlled by a separate YAW DAMP switch on the pedestal; depressing the switch to the second level disconnects the electric elevator-trim system. A green annnciator on the caution/advisory annunciator panel, placarded ELEC TRIM OFF, alerts the pilot whenever the system has has been disabled with a trim-disconnect switch and the ELEV TRIM switch is ON. The system can be reset by cycling the ELEV TRIM switch on the pedestal from ON to OFF, then back to ON again. 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.

RUDDER BOOST

A rudder boost system is provided to aid the pilot in maintaining directional control in the event of an engine failure or a large variation of power between the engines. Incorporated into the rudder cable system are two pneumatic rudder-boosting servos that actuate the cables to provide rudder pressure to help compensate for asymmetrical thrust.

During operation, a differential pressure valve accepts bleed air pressure from each engine. When the pressure varies between the bleed air systems, the shuttle in the differential pressure valve moves toward the low pressure side. As the pressure difference reaches a preset tolerance, a switch on the low pressure side closes, activating the rudder boost system. The system is designed only to help compensate for asymmetrical thrust. Appropriate trimming is to be accomplished by the pilot. Moving either or both of the bleed air valve switches on the copilot's subpanel to the INSTR & ENVIR OFF position will disengage the rudder boost system.

The system is controlled by a toggle switch, placarded RUDDER BOOST - ON - OFF, located on the pedestal below the rudder trim wheel. The switch is to be turned ON before flight. A preflight check of the system can be performed during the run-up by retarding the power on one engine to idle and advancing power on the opposite engine until the power difference between the engines is great enough to close the switch that activates the rudder boost system. Movement of the appropriate rudder pedal (left engine idling, right rudder pedal moves forward) will be noted when the switch closes, indicating the system is functioning properly for low engine power on that side. Repeat the check with opposite power settings to check for movement of the opposite rudder pedal.

INSTRUMENT PANEL

The floating instrument panel design allows the flight instruments to be arranged in a group directly in front of the pilot and the copilot. Complete pilot and copilot flight instrumentation is installed, including dual navigation systems, two course indicators, dual gyro horizons, and dual turn and slip indicators.

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 - yellow; advisory - green) 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 yellow MASTER CAUTION flashers (located just inboard of the MASTER WARNING flashers), and a PRESS TO TEST button located immediately to the right of the warning annunciator panel.

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

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 yellow 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 to reset the circuit. 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 on 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 advisory annunciators. There are no master flashers associated with these annunciators, since they are only advisory in nature, indicating functional situations which do not demand the immediate attention or reaction of the pilot. An advisory annunciator can be extinguished only by correcting the condition indicated on the illuminated lens.

The warning annunciators, caution annunciators, advisory annunciators and yellow MASTER CAUTION flashers feature both a "bright" and a "dim" mode of illumination intensity. The "dim" mode wil be selected automatically whenever all of the following conditions are met: a generator is on the line; the OVERHEAD FLOOD LIGHTS are OFF; the PILOT FLIGHT LIGHTS are ON; 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 of these conditions are met, the "bright" mode will be selected automatically. On later airplanes, and earlier airplanes with modified annunciator circuitry, The MASTER WARNING flasher also features both a "bright" and "dim" mode of illumination.

The lamps in the annunciator system should be tested before every flight, and anytime the integrity of a lamp is in question. Depressing the PRESS TO TEST button, located to the right of the warning annunciator panel in the glareshield, illuminates all the annunciator lights, MASTER WARNING flashers, and MASTER CAUTION flashers. Any lamp that fails to illuminate when tested should be replaced (refer to LAMP REPLACEMENT GUIDE in Section VIII, HANDLING, SERVICING AND MAINTENANCE).

WARNING PANEL ILLUSTRATION

L ENG FIRE	INVERTER	CABIN DOOR	ALT WARN	R ENG FIRE
L FUEL PRESS	———	———	———	R FUEL PRESS
*L OIL PRESS	*L GEN OVHT	*A/P TRIM FAIL	*R GEN OVHT	*R OIL PRESS
L CHIP DETECT	L BL AIR FAIL	*A/P DISC	R BL AIR FAIL	R CHIP DETECT

** Optional Equipment*

WARNING PANEL DESCRIPTION

NOMENCLATURE	COLOR	CAUSE FOR ILLUMINATION
L ENG FIRE	*Red*	*Fire in left engine compartment*
INVERTER	*Red*	*The inverter selected is inoperative*
CABIN DOOR	*Red*	*Cabin/cargo door open or not secure*
ALT WARN	*Red*	*Cabin altitude exceeds 12,500 feet*
R ENG FIRE	*Red*	*Fire in right engine compartment*
L FUEL PRESS	*Red*	*Fuel pressure failure on left side*
R FUEL PRESS	*Red*	*Fuel pressure failure on right side*
**L OIL PRESS*	*Red*	*Low oil pressure left engine*
**L GEN OVHT*	*Red*	*Left generator temperature too high*
**A/P TRIM FAIL*	*Red*	*Improper trim or no trim from autopilot trim command*
**R GEN OVHT*	*Red*	*Right generator temperature too high*
**R OIL PRESS*	*Red*	*Low oil pressure right engine*
L CHIP DETECT	*Red*	*Contamination in left engine oil is detected*
L BL AIR FAIL	*Red*	*Melted or failed plastic left bleed air failure warning line*
**A/P DISC*	*Red*	*Autopilot is disconnected*
R BL AIR FAIL	*Red*	*Melted or failed plastic right bleed air failure warning line*
R CHIP DETECT	*Red*	*Contamination in right engine oil is detected*

** Optional Equipment*

CAUTION/ADVISORY PANEL ILLUSTRATION

L DC GEN	††HYD FLUID LOW	†*PROP SYNC ON	RVS NOT READY	———	R DC GEN
———	———	———	DUCT OVERTEMP	———	———
L ICE VANE	———	BATTERY CHARGE	EXT PWR	———	R ICE VANE
*L AUTOFEATHER	———	*ELEC TRIM OFF	AIR COND N_1 LOW	———	*R AUTOFEATHER
L ICE VANE EXT	*BRAKE DEICE ON	LDG/TAXI LIGHT	PASS OXY ON	———	R ICE VANE EXT
L IGNITION ON	L BL AIR OFF	———	FUEL CROSSFEED	R BL AIR OFF	R IGNITION ON

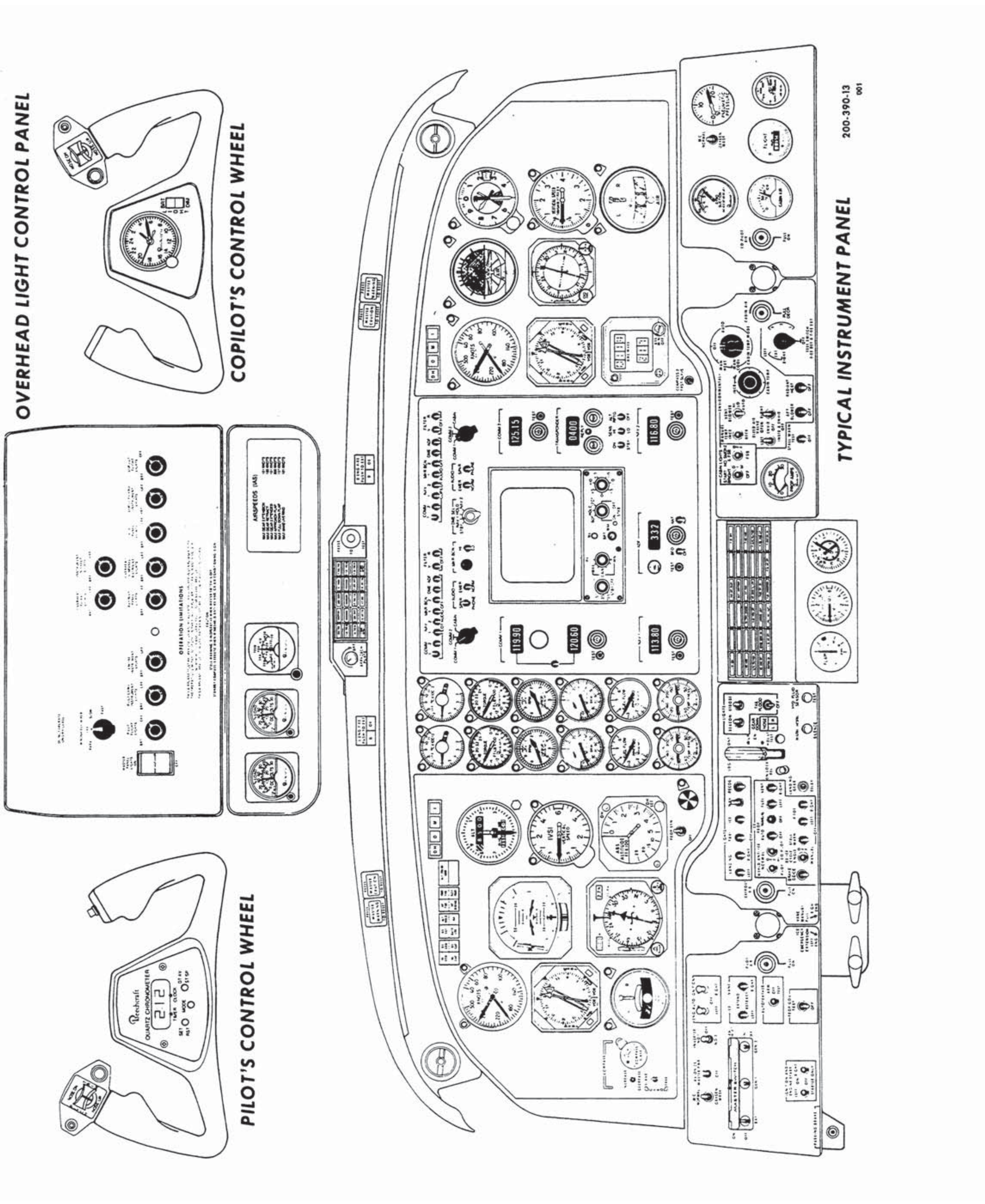
CAUTION/ADVISORY PANEL DESCRIPTION

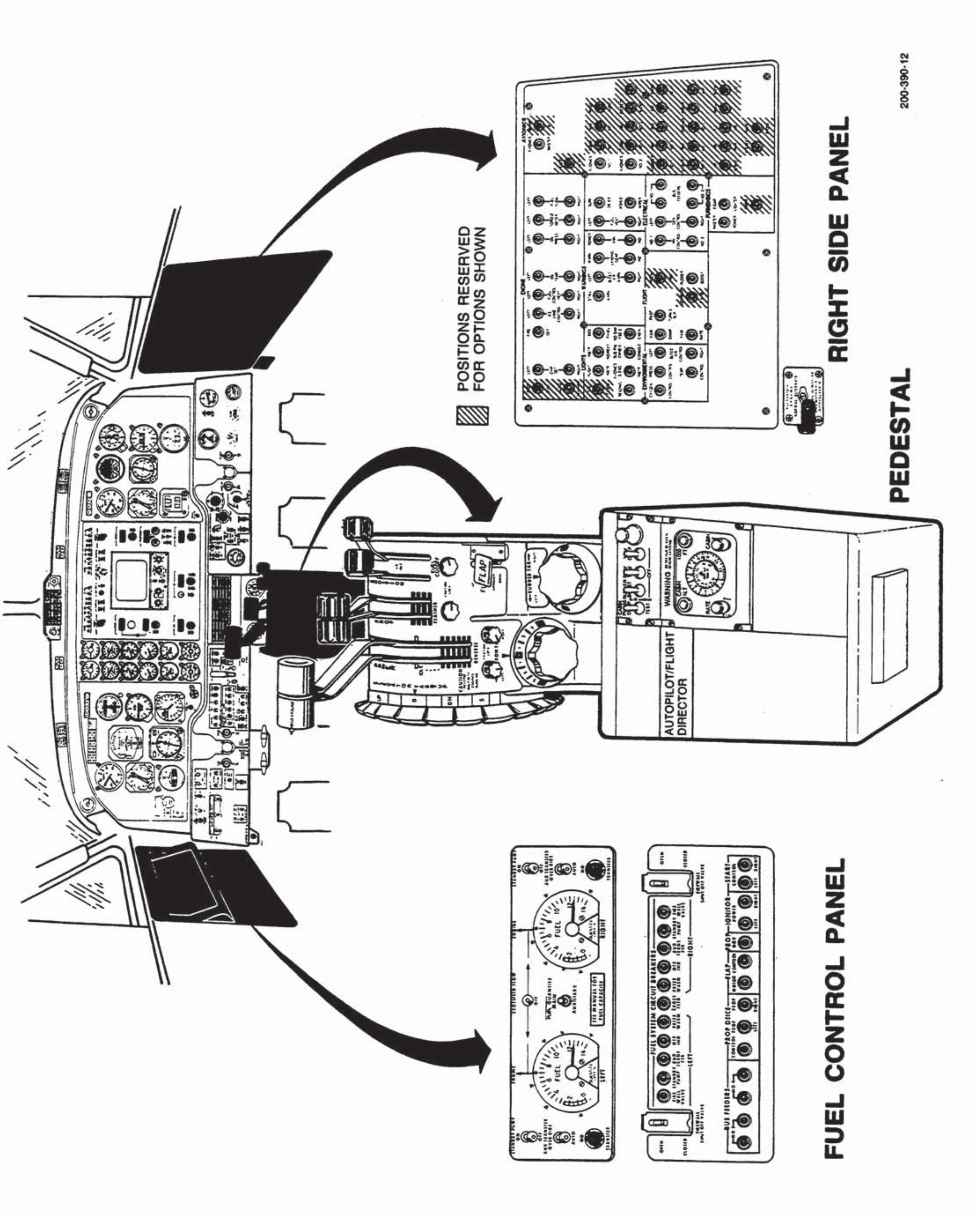
NOMENCLATURE	COLOR	CAUSE FOR ILLUMINATION
L DC GEN	*Yellow*	*Left generator off the line*
††HYD FLUID LOW	*Yellow*	*Hydraulic fluid in the landing gear system is low*
†PROP SYNC ON	*Yellow*	*Synchrophaser "ON" with landing gear extended*
RVS NOT READY	*Yellow*	*Propeller levers are not in the high rpm, low pitch position with landing gear extended*
R DC GEN	*Yellow*	*Right generator off the line*
DUCT OVERTEMP	*Yellow*	*Duct air too hot*
L ICE VANE	*Yellow*	*Left ice vane malfunction. Ice vane has not attained proper position*
BATTERY CHARGE	*Yellow*	*Excessive charge rate on battery*
EXT PWR	*Yellow*	*External power connector is plugged in*
R ICE VANE	*Yellow*	*Right ice vane malfunction. Ice vane has not attained proper position*
**L AUTOFEATHER*	*Green*	*Autofeather armed with power levers advanced above 90% N_1*
**ELEC TRIM OFF*	*Green*	*Electric trim de-energized by a trim disconnect switch on the control wheel with the system power switch on the pedestal turned on.*
AIR COND N_1 LOW	*Green*	*Right engine rpm too low for air conditioning load*
**R AUTOFEATHER*	*Green*	*Autofeather armed with power levers advanced above 90% N_1*
L ICE VANE EXT	*Green*	*Ice vane extended*
**BRAKE DEICE ON*	*Green*	*Brake deice system in operation*
LDG/TAXI LIGHT	*Green*	*Landing or taxi lights on with landing gear up*

**Optional Equipment*
†Not required when type II synchrophaser is used
†† On airplanes BB-1158, BB-1167, BB-1193 and after, BL-73 and after, and those prior airplanes incorporating Beech Kit P/N 101-8018-1

CAUTION/ADVISORY PANEL DESCRIPTION (Continued)

PASS OXY ON	*Green*	*Passenger oxygen system charged*
R ICE VANE EXT	*Green*	*Ice vane extended*
L IGNITION ON	*Green*	*Left starter/ignition switch is in the engine/ignition mode or left auto ignition system is armed and left engine torque is below approximately 410 ft-lbs*
L BL AIR OFF	*Green*	*Left environmental bleed air valve closed*
FUEL CROSSFEED	*Green*	*Crossfeed valve is open*
R BL AIR OFF	*Green*	*Right environmental bleed air valve is closed*
R IGNITION ON	*Green*	*Right starter/ignition switch is in the engine/ignition mode or right auto ignition system is armed and right engine torque is below approximately 410 ft-lbs*


OVERHEAD LIGHT CONTROL PANEL
COPILOT'S CONTROL WHEEL
TYPICAL INSTRUMENT PANEL
PILOT'S CONTROL WHEEL
200-390-13
001


POSITIONS RESERVED FOR OPTIONS SHOWN
RIGHT SIDE PANEL
PEDESTAL
AUTOPILOT/FLIGHT DIRECTOR
FUEL CONTROL PANEL
200-390-12

GROUND CONTROL

Direct linkage from the rudder pedals allows for nose wheel steering. When the rudder control is augmented by a main wheel brake, the nose wheel deflection can be considerably increased.

The minimum wing-tip turning radius, using partial braking action and differential engine power, is 39 feet 10 inches.

FLAPS

Two flaps are installed on each wing. Power is delivered from an electric motor to a gearbox mounted on the forward side of the rear spar. The gearbox drives four flexible drive-shafts which are connected to jackscrews, one of which operates each flap. The motor incorporates a dynamic braking system, through the use of two sets of motor windings. This feature helps prevent overtravel of the flaps. 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, from 0% (full up) to 100% (full down) is registered on an electric indicator on top of the pedestal. A side detent provides for quick selection of the APPROACH position (40% flaps). From the UP position to the APPROACH position, the flaps cannot be stopped in an intermediate position. Between APPROACH and DOWN, the flaps can be stopped anywhere by moving the handle to the DOWN position until the flaps reach the desired position, then moving the flap-switch handle back to APPROACH. The flaps can be raised to any position between DOWN and APPROACH by raising the handle to UP until the desired setting is reached, then returning the handle to APPROACH. Selecting the APPROACH position will stop flap travel anytime the flaps are deflected more than 40%.

The flap-motor power circuit is protected by a 20-ampere flap-motor circuit breaker placarded FLAP MOTOR, located on the left circuit breaker panel below the fuel control panel. A 5-ampere circuit breaker for the control circuit (placarded FLAP CONTROL) is also located on this panel.

Lowering the flaps will produce these results:

ATTITUDE - Nose UP

AIRSPEED - Reduced

STALL SPEED - Lowered

TRIM - Nose-Down Adjustment Required to Maintain Attitude

LANDING GEAR

MECHANICAL LANDING GEAR

A 28-volt motor and gear box, located on the forward side of the center-section main spar, extends and retracts the landing gear. The landing gear motor is controlled by the handle placarded LDG GEAR CONTROL - UP - DN on the pilot's right subpanel. The landing gear control handle must be pulled out of a detent before it can be moved from either the UP or the DN position. The motor incorporates a dynamic braking system controlled with "up" and "down" limit switches, which in conjunction with the landing gear locking mechanism prevents overtravel of the landing gear.

Torque shafts drive main gear actuators, and duplex chains drive the nose gear actuator. A spring-loaded friction-type overload clutch in the gearbox prevents damage to the structure and to the torque shafts in the event of a malfunction. A 150 amp limiter, located on the landing gear panel forward of the main spar under the center floorboard, protects the system from electrical overload.

The Beech air-oil type shock struts are filled with compressed air and hydraulic fluid. Linkage from the rudder pedals permits nose wheel steering when the nose gear is down. One spring-loaded link in the system absorbs some of the force applied to any of the interconnected rudder pedals until the nose wheel is rolling, at which time the resisting force is less and more pedal motion results in more nose wheel deflection. Since motion of the pedals is transmitted via cables and linkage to the rudder, rudder deflection occurs when force is applied to any of the rudder pedals. With the nose landing gear retracted, some of the force applied to any of the rudder pedals is absorbed by the spring-loaded link in the steering system so that there is no motion at the nose wheel, but rudder deflection still occurs. The nose wheel is self centering upon retraction.

When force on the rudder pedal is augmented by a main wheel braking action, the nose wheel deflection can be considerably increased.

A safety switch on the right main gear torque knee opens the control circuit when the strut is compressed. The safety switch also activates a solenoid-operated down-lock hook on the landing gear control handle located on the pilot's right subpanel. This mechanism prevents the landing gear control handle from being raised when the airplane is on the ground. The hook automatically unlocks when the airplane leaves the ground. In the event of a malfunction of the down-lock solenoid, the down lock can be released by pressing downward on the red down-lock release button. The release button is located just left of the landing gear control handle. 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 intermittently 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.

Visual indication of landing gear position is provided by individual green GEAR DOWN annunciators placarded NOSE - L - R on the pilot's right subpanel. The annunciators may be checked in flight by pressing the annunciator. Two red, parallel-wired indicator lights located in the control handle illuminate to show that the gear is in transit or not locked. They also illuminate when the landing gear warning horn is actuated. Absence of illumination indicates that the gear is up and locked or down and locked. The red control handle lights may be checked by pressing the HDL LT TEST button located to the right of the landing gear control handle.

LANDING GEAR WARNING SYSTEM (MECHANICAL 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. Various warning modes result, depending upon the position of the flaps.

With the flaps in the UP or APPROACH position and either or both power levers retarded below approximately 80% N_1, the warning horn will sound intermittently 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 cancelled. The landing gear warning system will be rearmed if the power lever(s) are advanced sufficiently.

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

MANUAL LANDING GEAR EXTENSION (MECHANICAL SYSTEM)

Manual landing gear extension is provided through a separate, chain-drive system. To engage the system, pull the LDG GEAR RELAY circuit breaker, located to the left of the landing gear control handle on the pilot's right subpanel, and ensure that the landing gear control handle is in the DN position. Pull up on the alternate engage handle (located on the floor) and turn it clockwise until it stops. This will electrically disconnect the motor from the system and lock the alternate drive system to the gear box. With the alternate drive locked in, the chain is driven by a continuous-action ratchet, which is activated by pumping the alternate extension handle located adjacent to the alternate engage handle. Stop pumping when all three green gear-down annunciators are illuminated. Further movement of the handle could damage the drive mechanism and prevent subsequent electrical gear retraction. Refer to LANDING GEAR MANUAL EXTENSION (MECHANICAL SYSTEM) in Section IIIA, ABNORMAL PROCEDURES. 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.

1. The inoperative gear down annunciator illuminates when tested.
2. The red light in the handle is illuminated.
3. 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 electrically. Refer to LANDING GEAR RETRACTION AFTER PRACTICE MANUAL EXTENSION (MECHANICAL SYSTEM) in Section IV, NORMAL PROCEDURES.

HYDRAULIC LANDING GEAR

The retractable tricycle landing gear is electrically controlled and hydraulically actuated. The system utilizes folding braces, called drag legs, that lock in place when the gear is fully extended. The nose gear actuator incorporates an internal mechanical down-lock to hold the gear in the fully extended position. The main gear incorporate mechanical locks on the drag leg and no locks on the actuators. The landing gear is held in the up-lock position by hydraulic pressure.

Hydraulic pressue to the system is supplied by a hydraulic power pack. A hydraulic reservoir located in the left center wing section provides hydraulic fluid to the power pack. The reservoir incorporates a dip stick to provide a visual check of fluid level.

Electrically actuated control valves control the flow of hydraulic fluid to the individual gear actuators. The control valves receive electrical power through the landing gear control handle.

Accidental retraction of the landing gear is prevented through safety switches located on the main landing gears.

LANDING GEAR EXTENSION AND RETRACTION (HYDRAULIC SYSTEM)

The nose and main landing gear assemblies are extended and retracted by a hydraulic power pack in conjunction with hydraulic actuators. The hydraulic power pack is located in the left center section, just forward of the main spar. One hydraulic actuator is located at each landing gear. The power pack consists of: a hydraulic pump, a 28VDC motor, a two section fluid reservoir, filter screens, gear selector valve, two solenoids, a fluid level sensor, and an up-lock pressure switch. For manual extension the system has a hand-lever-operated pump located on the floor between the crew seats. Hydraulic lines, one for normal extension, and one for retraction, routed from the power pack, and one for manual extension from the hand pump, are routed to the nose and main gear actuators. The normal extension lines and the manual extension lines are connected to the upper end of each hydraulic actuator. The hydraulic lines for retraction are fitted to the lower ends of the actuators. Hydraulic fluid under pressure generated by the power pack pump and contained in the accumulator acts on the piston faces of the actuators which are attached to folding drag braces resulting in the extension or retraction of the landing gear.

An internal mechanical lock in the nose gear actuator and the over-center action of the nose gear drag leg assembly lock the nose gear in the down position. Notched hook, lock link and lock link guide attachments fitted to each main gear upper drag leg provide positive down-lock action for the main gear.

Electrical overload to the system is prevented through the use of a 60 ampere circuit breaker located below the flooring near the hydraulic power pack.

The landing gear hydraulic power pack motor is controlled by the use of the landing gear control handle placarded LDG GEAR CONTROL - UP - DN located on the pilot's subpanel. The control handle must be pulled out of a detent before it can be moved from either the UP or DN position.

Safety switches, called squat switches, on the main gear torque knees open the control circuit when the strut is compressed. The squat switches must close to actuate a solenoid which moves a down-lock hook on the landing gear control handle to the released position. This mechanism prevents the landing gear control handle from being placed in the UP position when the airplane is on the ground. The hook automatically disengages when the airplane leaves the ground, and can be overridden by pressing down on the red down-lock release button located to the left of the landing gear control handle.

In flight, as the landing gear moves to the full down position, the down lock switches are actuated and interrupt current to the pump motor. When the red gear in-transit lights in the landing gear control handle extinguish and the three green GEAR DOWN annunciators illuminate, the landing gear is in the fully extended position.

Two gear select solenoids located on the valve body of the pump are energized through positioning of the landing gear control switch handle either to the UP or DN position. Once energized, the gear select valve is actuated, allowing hydraulic fluid to flow to the actuators.

Hydraulic system pressure performs the up-lock function, holding the landing gear in the retracted position. When the hydraulic pressure reaches 2775 ±55 psi, the up-lock pressure switch will cause the landing gear relay to open and interrupt the current to the pump motor. The same pressure switch will cause the pump to actuate, should the hydraulic pressure drop to approximately 2400 psi.

A caution annunciator, placarded HYD FLUID LOW, in the caution/advisory annunciator panel will illuminate (yellow) whenever the hydraulic fluid level in the hydraulic power pack is low. The annunciator is tested by pressing the HYD FLUID SENSOR TEST button located on the pilot's subpanel.

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 intermittently, 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.

Landing gear position is indicated by an assembly of three green annunciators. When illuminated, the annunciators indicate that the particular gear is down. Absence of illumination indicates that the gear is up.

Two red parallel-wired indicator lights, located in the landing gear control handle, illuminate to show that the gear is in transit or unlocked. The red lights in the handle also illuminate when the landing gear warning horn is actuated.

The red lights may be checked by pressing the HDL LT TEST button located adjacent to the landing gear control handle.

LANDING GEAR WARNING SYSTEM (HYDRAULIC SYSTEM)

The landing gear warning system is provided to warn the pilot that the landing gear is not down during specific flight regimes. Various warning modes result, depending upon the position of the flaps.

With the flaps in the UP or APPROACH position and either or both power levers retarded below approximately 80% N_1, the warning horn will sound intermittently 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 cancelled. The landing gear warning system will be rearmed if the power lever(s) are advanced sufficiently.

With the flaps beyond APPROACH position, the warning horn and landing gear switch handle lights will be activated regardless of the power settings, and neither can be cancelled.

MANUAL LANDING GEAR EXTENSION (HYDRAULIC SYSTEM)

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 left of the landing gear control handle 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. 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.

1. The inoperative gear down annunciator illuminates when tested.
2. The red light in the handle is illuminated.
3. 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.

BRAKE SYSTEM

The dual hydraulic brakes are operated by depressing the toe portion of either the pilot's or copilot's rudder pedals. The series system plumbing enables braking by either pilot or copilot.

Dual parking-brake valves are installed adjacent to the rudder pedals between the master cylinders of the pilot's rudder pedals and the wheel brakes. A control for the valves, placarded PARKING BRAKE, is located below the pilot's left subpanel. After the pilot's brake pedals have been depressed to build up pressure in the brake lines, both valves can be closed simultaneously by pulling out the parking 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 valve.

CAUTION

The parking brake should be left off and wheel chocks installed if the airplane is to be left unattended. Changes in the ambient temperature can cause the brakes to release or to exert excessive pressures.

TIRES

The airplane is normally equipped with dual 18x5.5 Type VII, 8-ply-rated, tubeless, rim-inflated tires on each main gear. For increased service life, 10-ply-rated tires of the same size may be installed.

Optionally, the airplane may be equipped with dual 22x6.75-10, 8-ply-rated, tubeless tires on each main gear. These tires provide higher flotation, and permit operation at approximately 2/3 the inflation pressure required for the standard 18x5.5 tires.

The nose gear is equipped with a 22x6.75-10, 8-ply-rated, tubeless tire.

BAGGAGE COMPARTMENT

The entire aft-cabin area (which is aft of the foyer) may be utilized as a baggage compartment. A nylon web is provided for the restraining of loose items. See "Dimensional and Loading Data" and "Cabin Arrangement Diagrams" in Section VI, WEIGHT AND BALANCE/EQUIPMENT LIST.

WARNING

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

Do not carry children in the baggage compartment unless secured in a seat.

CAUTION

Baggage and other objects should be secured by webs in order to prevent shifting in turbulent air.

Items stowed in the aft-cabin area are accessible in flight. The aft-cabin area can be closed off from the foyer by pulling the optional baggage compartment curtain across the opening and securing it with the snap fasteners provided. Alternately, a latching compartment door may be installed. The door is unlatched by rotating the latch handle clockwise, and latched by rotating the handle counterclockwise.

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

Various configurations of passenger chairs and 2- or 4-place couches may be installed on the continuous tracks which are mounted on the cabin floor. All passenger chairs are placarded either FRONT FACING ONLY or FRONT OR AFT FACING on the horizontal leg cross brace. Only chairs placarded FRONT OR AFT FACING may be installed facing aft. All aft-facing chairs (and all forward-facing chairs that are equipped with shoulder harnesses) are equipped with adjustable headrests.

WARNING

Before takeoff and landing, the headrest should be adjusted as required to provide support for the head and neck when the passenger leans against the seatback.

Some passenger chairs can be moved fore and aft, to suit legroom requirements of different passengers, by lifting a horizontal release lever that extends laterally under the front of adjustable seats. ("Front" is the direction opposite the seatback, regardless of whether the chair faces fore or aft.)

The seatbacks can be adjusted to any angle from fully upright to fully reclining, by depressing the release lever located on the side of the seat at the front inboard corner. When the lever is depressed and the passenger leans against the seatback, the seatback will slowly recline until the lever is released, or until the fully reclining position is attained. When no weight is placed against the seatback and the lever is depressed, the seatback will rise until the lever is released, or until the fully upright position is reached. The seatbacks of all occupied seats must be upright for takeoff and landing.

The passenger-chair seatback can also be folded flat over the seat cushion, after releasing the lock lever located on the side of the seat at the back inboard corner.

The optional lateral-tracking passenger chairs incorporate a flat, rectangular release lever underneath the front inboard corner of the seats. When this lever is lifted, the chairs can be adjusted fore and aft, as well as laterally. The seatback adjustments are the same as those on the standard passenger chairs. When occupied, these seats must be in the outboard position (i.e., against the cabin wall) for takeoff and landing.

Inboard armrests on passenger chairs - and both armrests on couches and lateral-tracking chairs - can be folded flush with the top of the seat cushions to facilitate entry to and egress from the seat. The armrests can be lowered by lifting the flat, rectangular release plate located under the front end of the armrest, then moving the armrest toward the front of the seat and downward. The armrest can be raised by pulling the armrest upward and toward the seatback until it locks into place.

The couches are not adjustable.

FOYER

Hinged seat-cushion halves mounted on top of the toilet form an extra passenger seat when the toilet is not is use.

AFT-CABIN AREA

One or two optional folding seats may be installed in the aft-cabin area. They are mounted on the cabin sidewall and swing inboard when unfolded. A latch mechanism on the leg locks the seats in place when they are unfolded. When this seating is not needed, the seat(s) may be folded against the cabin sidewall and held in place with retaining straps.

SEATBELTS

Every seat in the airplane is equipped with a seatbelt. The seatbelt can be lengthened by turning the male half of the buckle at a right angle to the belt, then pulling the male half in the direction away from the anchored end of the belt. The buckle is locked by sliding the male half into the female half of the buckle. The belt is then tightened by pulling the short end of the belt through the male half of the buckle until a snug fit is obtained. The buckle is released by lifting the large, hinged release lever on the female buckle half and pulling the male half of the buckle free. All occupants must wear seat belts during takeoff and landing.

SHOULDER HARNESSES

COCKPIT

The shoulder harness installations for the pilot and copilot seats consist of two straps each. Each strap is routed from the lower aft area of the seat, up the seatback, and through a retaining loop on top of the seatback. One strap is worn over each shoulder. Each strap terminates in a slotted bayonet-blade fastener which is aligned with one edge of the strap. When the two bayonet blades are placed together, the shoulder harness straps can be secured by sliding the male half of the seatbelt buckle through the bayonet slots and into the female half of the seatbelt buckle.

The shoulder harness straps proceed from inertia reels built into the crew chairs. Spring loading at the inertia reels keeps the shoulder harnesses snug, but allows the pilot and copilot all the freedom of movement normally required in flight. However, the inertia reels incorporate a locking device that will secure the harness straps in the event of sudden forward movement.

CABIN

The shoulder harness on passenger chairs consists of a single strap. It is routed through the top of the seatback and terminates in a triangular metal fastener. The strap is worn diagonally. It runs from the outboard shoulder to the inboard hip area, where it is secured by hooking the metal fastener around the securing stud on the male half of the seatbelt buckle.

The shoulder harness strap coils and uncoils from an inertia reel built into the passenger chair. Spring loading at the inertia reel keeps the shoulder harness strap snug, but allows considerable freedom of movement. However, the inertia reel incorporates a locking device that will secure the harness strap in the event of sudden forward movement. If the seat is equipped with a shoulder harness, it must be worn during takeoff and landing.

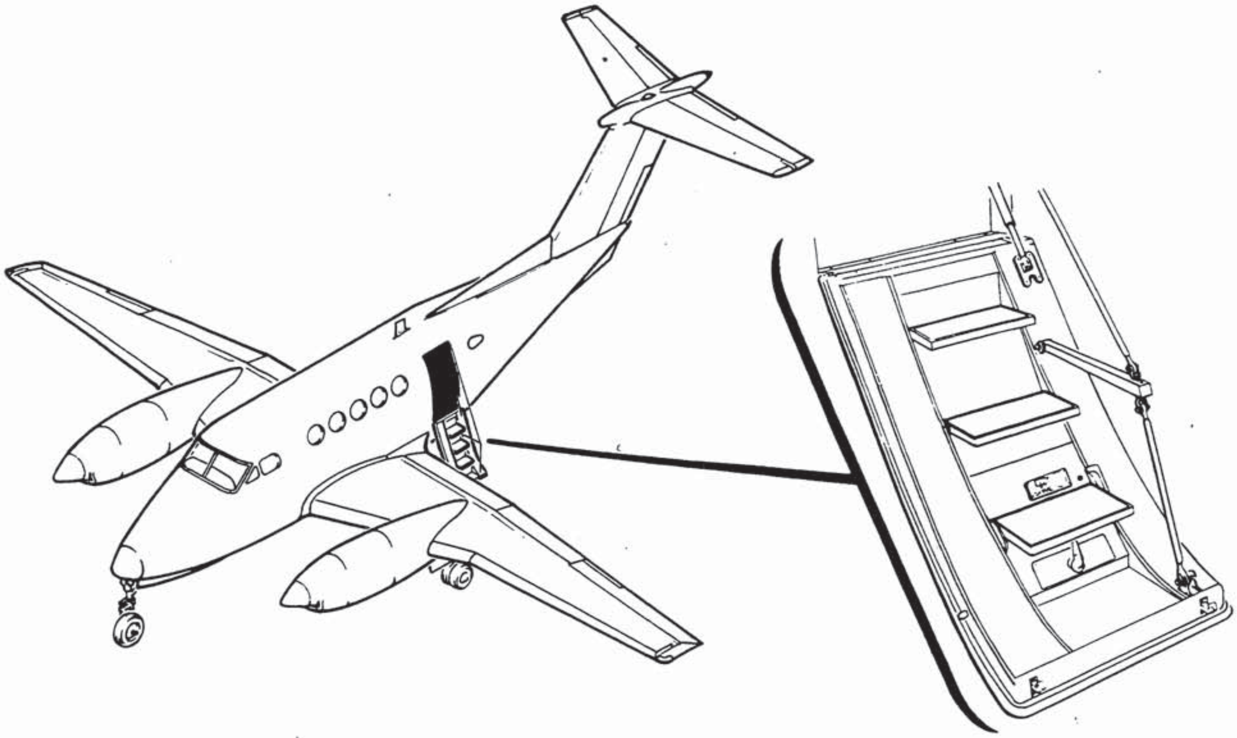
WARNING

Ensure the seatback is in the fully upright position and that the headrest is properly adjusted whenever the shoulder harness is used.

AFT-CABIN AREA

The shoulder harness for the aft-cabin area fold-up chairs is of a double-strap configuration. The middle portion of the strap is secured by a metal slip ring which is anchored to the aft pressure bulkhead. The two ends (which actually function as two separate straps) extend downward toward the seatbelt-buckle area. One end of the shoulder harness strap terminates in a slotted bayonet-blade fastener. The other end is attached to the upper edge of the shoulder harness adjuster. A short adjusting strap, which is also equipped with a slotted bayonet blade fastener, extends upward from the area of the seatbelt buckle and slides through the lower portion of the shoulder harness adjuster. A small, flexible adjusting tab is also attached to the lower edge of the adjuster.

One shoulder harness strap is worn over each shoulder. When the two bayonet blades are placed together, the shoulder harness straps can be secured by sliding the male half of the seatbelt buckle through the bayonet slots and into the female half of the seatbelt buckle. The shoulder harness strap can be lengthened by grasping the tab on the adjuster and pulling upward. The strap can be tightened by grasping the loose end of the adjusting strap and pulling it through the adjuster until the shoulder harness is snug.



B200-104-54

DOORS, WINDOWS, AND EXITS

AIRSTAIR ENTRANCE DOOR (B200)

The airstair entrance door (cabin door) is hinged at the bottom. It swings outward and downward when opened. A stairway is built onto the outboard side of the door. Two of the stairsteps fold flat against the door when the door is closed. A hydraulic damper ensures that the door will swing down slowly when it opens. While the door is open, it is supported by a plastic-encased cable, which also serves as a handrail. Additionally, this cable is utilized when closing the door from inside the airplane. An inflatable rubber seal is installed around the perimeter of the door, and seats against the door frame as the door is closed. When weight is off the landing gear, engine bleed air supplies pressure to inflate the door seal, which provides a positive pressure-vessel seal around the door. The outside door handle can be locked with a key, for security of the airplane on the ground.

CAUTION

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

AIRSTAIR DOOR

The door locking mechanism is operated by rotating either the outside or the inside door handle. The handles are linked together, so they move together. Two latch bolts at each side of the door, and two latch hooks at the top of the door, lock into the door frame to secure the airstair door.

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

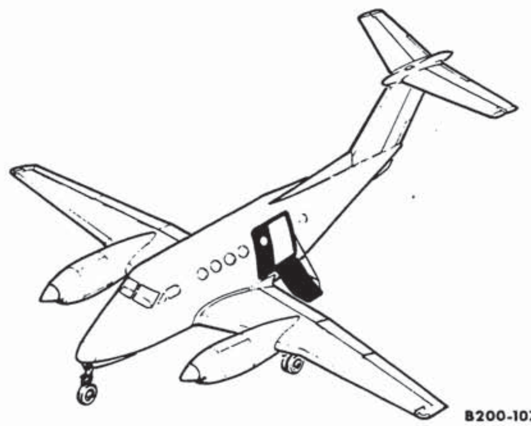
WARNING

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 whatever to suspect that the door may not be securely locked, the cabin should be depressurized (after first considering altitude), and all occupants instructed to remain seated with their seatbelts fastened. After the airplane has made a full-stop landing and the cabin has been depressurized, a crew member should check the security of the cabin door.

To close the door from outside the airplane, lift up the free end of the airstair door and push it up against the door frame as far as possible. Then grasp the handle with one hand and rotate it clockwise as far as it will go. The door will then 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 handrail cable and pull the airstair door up against the door frame. Then grasp the handle with one hand and rotate it counterclockwise as far as it will go, continuing to pull inward on the door. The door will then move into the closed position. Then turn the handle clockwise as far as it will go. The release button should pop out, and the handle should be pointing down. Check the security of the door by attempting to rotate the handle counterclockwise without depressing the release button; the handle should not move. Next, lift the folded stairstep that is just below the door handle. A placard adjacent to the round observation window advises the observer that the safety lock arm should be in position around the diaphragm shaft (plunger) when the handle is in the locked position. The placard also presents a diagram showing how the arm and shaft should be positioned. A red push-button switch near the window turns on a lamp inside the door, which illuminates the area observable through the window. If the arm is properly positioned around the shaft, proceed to check the indication in each of the visual inspection ports, one of which is located near each corner of the door. The green stripe painted on the latch bolt should be aligned with the black pointer in the visual inspection port. If any condition specified in this door-locking procedure is not met, DO NOT TAKE OFF.



AIRSTAIR ENTRANCE DOOR (B200C)

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

CAUTION

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

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

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

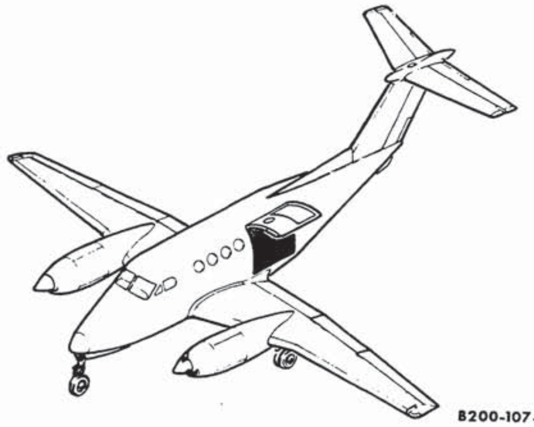
WARNING

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 whatever to suspect that the door may not be securely locked, the cabin should be depressurized (after first considering altitude), and all occupants instructed to remain seated with their seatbelts fastened. After the airplane has made a full-stop landing and the cabin has been depressurized, a crew member should check the security of the airstair door and the cargo door.

To close the door from outside the airplane, lift up the free end of the airstair door and push it up against the door frame as far as possible. Then grasp the handle with one hand and rotate it clockwise as far as it will go. The door will then 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 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 handrail cable and pull the airstair door up against the door frame. Then grasp the handle with one hand and rotate it counterclockwise as far as it will go, continuing to pull inward on the door. The door will then move into the closed position. Then turn the handle clockwise as far as it will go. The release button should pop out, and the handle should be pointing down. Check the security of the door by attempting to rotate the handle counterclockwise without depressing the release button; the handle should not move. Next, lift the second folded airstep below the door handle. A placard adjacent to the round observation window advises the observer that the safety lock arm should be in position around the diaphragm shaft (plunger) when the handle is in the locked position. The placard also presents a diagram showing how the arm and shaft should be positioned. A red push-button switch near the window turns on a lamp inside the door, which illuminates the area observable through the window. If the arm is properly positioned around the shaft, proceed to check the orange stripe on each of the six rotary latches (three on each side of the airstair door) and ensure each is aligned with the notch in the plate on the door frame. Finally, turn the battery switch ON and check the warning annunciator panel in the cockpit; ensure that the red CABIN DOOR annunciator is extinguished. It will illuminate when the battery switch is ON and the airstair door is not closed and securely latched. With the BATtery switch OFF and the airstair door closed but not latched, the CABIN DOOR annunciator will illuminate. If any condition specified in this door-latching procedure is not met, DO NOT TAKE OFF.



B200-107-26

CARGO DOOR (B200C)

A large, swing-up cargo door, hinged at the top, provides access for the loading of large items. The cargo door latch system is operated by two handles: one in the upper aft area of the door, and the other in the lower forward area of the door. Two separate access covers must be opened in order to operate the two handles. In order to move the upper aft handle out of the latched position, depress the black release button in the handle and rotate the yellow handle upward as far as it will go. This movement is transmitted via cables to two hollow, crescent latches on the forward side and two on the aft side of the cargo door. The latches rotate to release latch posts mounted in the cargo door frame.

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

CAUTION

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

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

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

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

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

To latch the cargo door after it is closed, rotate the lower forward latch handle forward until the orange lock hook engages the stud on the handle. Check the security of this handle by attempting to move it aft without raising the lock hook; it should not move. Close the access cover. Next, check the observation window at the lower aft corner of the cargo door. Ensure that the orange stripe on the latch pin linkage is aligned with the orange pointer in the observation window.

Next, rotate the upper aft latch handle down until the black release button pops up. Check the security of this handle by attempting to pull it out and up without depressing the release button; it should not move. Close the access cover. Then, insure that the orange stripe on each of the four rotary latches (two on each side of the cargo door) is aligned with the notch in the plate on the door frame. Finally, check the caution annunciator panel in the cockpit and ensure that the amber CABIN DOOR annunciator is extinguished. With the battery switch ON, it will be illuminated if either the airstair door or the cargo door is open. With the battery switch OFF, it will be illuminated only if the airstair door is closed but not securely latched. Perform the "Cabin/Cargo Door Circuitry Check" in the NORMAL PROCEDURES Section prior to the first flight of the day. If any condition specified in this door-latching procedures is not met, DO NOT TAKE OFF.

EMERGENCY EXIT

The emergency exit door is located on the right side of the fuselage at the forward end of the passenger compartment. From the inside, the door is released with a pull-down handle, placarded EXIT-PULL. From the outside, the door is released with a flush-mounted, pull-out handle. The non-hinged, plug-type door removes completely from the frame into the cabin when the latches are released.

B200:

The door can be locked so that it cannot be removed or opened from the outside using the flush-mounted pull-out handle. The door is locked when the lock-lever (inside) is in the down or locked position. Locking the door is for security when the airplane is parked. The lock-lever should be in the up or unlocked position prior to flight, to allow removal of the door from the outside in the event of an emergency. Removal of the door from the inside is possible at all times using the EXIT-PULL handle, since this handle is not locked by the lock-lever. An exit lock placard is placed on the lock-lever so that it can be read when the lever is in the locked position.

B200C:

The door can be locked with a key from the inside, to prevent opening from the outside. The inside handle will unlatch the door, whether or not it is locked, by overriding the locking mechanism. The key lock should be unlocked prior to flight, to allow removal of the door from the outside in the event of an emergency. The keyhole is in the horizontal position when the door is locked. The key cannot be removed in this position.

A wiper-type disconnect for the air duct that supplies air to the eyeball outlet in the emergency exit door is located on the upper-aft edge of the door. As the door is removed, the duct is disconnected, since it is an integral part of the door.

Located on the lower-forward edge of the door is an electrical disconnect for the wiring that goes to the reading light and the fluorescent light in the emergency exit door. It will unplug as the door is being removed. Upon reinstalling the door, the electrical disconnect should be reconnected before moving the door into the closed position.

INTERIOR DOORS

Sliding doors are provided between the cockpit and cabin, and between the cabin and foyer. These doors provide privacy, and prevent the spilling of light from one compartment into another. The doors are closed by sliding the two partition-type door panels to the center of the aisle, where they are held together by a magnetic strip in the edge of each door.

CABIN WINDOWS

Each cabin window pane, which is composed of a sheet of polyvinyl butyral (PVB) laminated between two sheets of clear acrylic plastic, is stressed to withstand the cabin-to-ambient air pressure differential. It is then sealed into a window opening in the fuselage, and forms an integral part of the pressure vessel.

POLARIZED TYPE

Two dust panes are mounted inboard of the cabin window pane in each window frame. Each of these dust panes is composed of a film of polarizing material laminated between two sheets of acrylic plastic. The inboard dust pane rotates freely in the window frame and has a protruding thumb knob near the edge. Rotating the pane through an arc of 90° permits complete light regulation as desired. Rotation changes the relative alignment between the polarizing films, thus providing any degree of light transmission from full intensity to almost none.

WARNING

Do not look directly at the sun, even through polarized windows, because eye damage could result.

CAUTION

When the airplane is to be parked in areas exposed to intense sunlight, the polarized windows should be rotated to the clear position to prevent deterioration of the polarization coating. Sufficient ultraviolet protection is provided to prevent fading of the upholstery.

SHADE TYPE

A dust pane, which is a single sheet of tinted acrylic plastic, is mounted inboard of the cabin window pane in each window frame. An adjustable window shade is provided to control the amount of light admitted. The shade is adjusted by squeezing the two latch handles located on the lower center of the shade, then positioning the shade as desired. Detents in the shade tracks provide positive latching action at various positions.

SUNVISORS

OPERATING INSTRUCTIONS

TO OPERATE FROM STOWED POSITION:

Push straight back and pull down. Move along track to desired place and pivot out near windshield (or window), rotate knob clockwise to lock.

To change position:

Rotate knob counterclockwise to unlock, move to desired location and position, then relock knob by turning clockwise.

To stow:

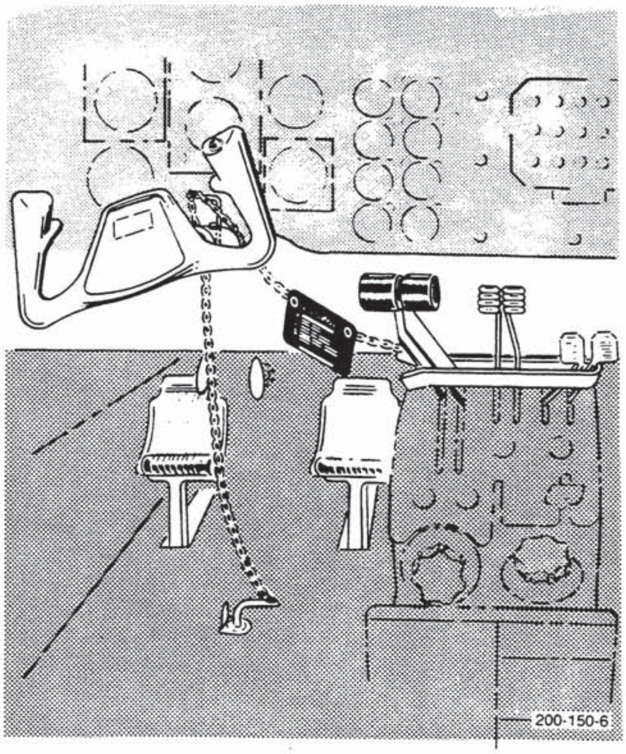
Rotate knob counterclockwise to unlock, move along track to aft end, pivot up against headliner to allow catch to retain sunvisor assembly.

CONTROL LOCKS

The control locks consist of a U-shaped clamp and two pins, all connected by a chain. The pins lock the primary flight controls; the U-shaped clamp fits around the engine control levers, serving to warn the pilot not to start the engines with the control locks installed. It is important that all the locks be installed and removed together, to preclude the possibility of attempting to taxi or fly the airplane with the engine control levers released, but with the pins still installed in the flight controls.

Install the control locks in the following sequence:

1. Position the U-clamp around the engine control levers.
2. Move the control column as necessary to align the holes, then insert the L-shaped pin that is attached to the middle of the chain (approx.). The control wheel position: full forward and rotated approximately 15° to the left.
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 control 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.

ENGINES

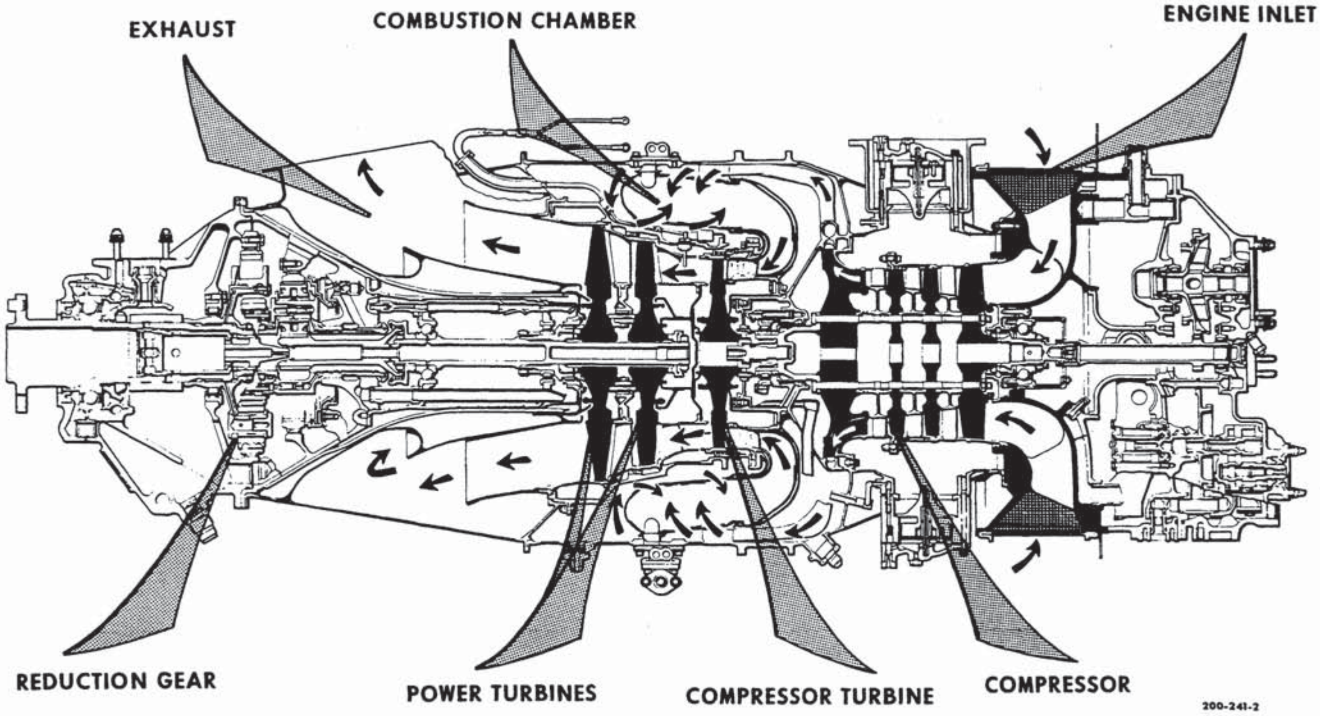
The BEECHCRAFT Super King Air B200/B200C is powered by two Pratt & Whitney Canada PT6A-42 turbopropeller engines, each rated at 850 SHP. Each engine has a three-stage axial flow, single-stage centrifugal-flow compressor, which is driven by a single-stage reaction turbine. The power turbine - a two-stage reaction turbine counter-rotating with the compressor turbine - drives the output shaft. Both the compressor turbine and the power turbine are located in the approximate center of the engine, with their shafts extending in opposite directions. Being a reverse flow engine, the ram air supply enters the lower portion of the nacelle and is drawn in through the aft protective screens. The air is then routed into the compressor. After it is compressed, it is forced into the annular combustion chamber, and mixed with fuel that sprayed in through 14 nozzles mounted around the gas generator case. A capacitance discharge ignition unit and two spark igniter plugs are used to start combustion. After combustion, the exhaust passes through the compressor turbine and two stages of power turbine and is routed through two exhaust ports near the front of the engine. A pneumatic fuel control system schedules fuel flow to maintain the power set by the gas generator power lever. Propeller speed within the governing range remains constant at any selected propeller control lever position through the action of a propeller governor, except in the beta range where the maximum propeller speed is controlled by the pneumatic section of the propeller governor.

The accessory drive at the aft end of the engine provides power to drive the fuel pumps, fuel control, the oil pumps, the refrigerant compressor (right engine), the starter/generator, and the tachometer transmitter. At this point, the speed of the drive (N_1) is the true speed of the compressor side of the engine, 37,500 rpm (which corresponds to 100% N_1). Maximum continuous speed of the engine is 38,100 rpm, which equals 101.5% N_1, with a transient overspeed of 38,500 rpm, which equals 102.6% N_1.

The reduction gearbox forward of the power turbine provides gearing for the propeller and drives the propeller tachometer tramsmitter, the propeller overspeed governor, and the propeller governor. The turbine speed on the power side of the engine is 30,000 rpm. After reduction gearing the propeller rpm is 2000.

Engine torque at the propeller shaft is indicated by a torquemeter located inside the first stage reduction gear housing. The torquemeter is a hydromechanical torque measuring device. It consists of: a ring gear and case (helical splines between ring gear and case), torquemeter cylinder, torquemeter piston, valve plunger and spring, differential pressure sensor and servo transmitter combination, and servo indicator calibrated to indicate ft-lbs.

Torque at the power turbine shaft and the resisting torque at the propeller shaft gears is converted from rotary motion by the helical splines to a translating motion at the piston face. A change in torquemeter oil pressure results from the piston translation. The valve plunger and spring maintains oil pressure proportional to engine torque. The differential pressure sensor uses a bellows system to sense differences between torquemeter oil pressure and a reference pressure. Bellows movement drives the transmitter servo. The electric signal from the transmitter drives the servo motor in the torquemeter indicator. Torque is indicated by indicator needle position on a calibrated dial.



Deceleration on the ground is achieved by bringing the propeller blades through the Beta range into a reversing pitch by utilizing the pitch change mechanism. The power levers must be retarded below the IDLE position by raising them over a detent. Reversing power is avalilable in direct proportion to the retarding of the levers in the reversing range.

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 cutoff, 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 operates a speeder spring inside the primary governor to reposition 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 1600 to 2000 rpm.

CONDITION LEVERS

The condition levers have three positions; FUEL CUT-OFF, LOW IDLE and HIGH IDLE. Each lever controls the fuel cutoff function of the fuel control unit and limits idle speed at 56% N_1 for low idle, and 70% N_1 for high idle.

PROPELLER REVERSING

When the power levers are lifted over the IDLE detent, they control engine power through the Beta and reverse ranges.

CAUTION

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

Condition levers, when set at HIGH IDLE, keep the engines operating at 70% N_1 high idle speed 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. When they are rotated counterclockwise, the propulsion system control levers can be moved freely. As the friction locks are rotated clockwise, the control levers progressively become more resistant to movement, so that they will not creep out of position.

ENGINE INSTRUMENTATION

Engine instruments, located to the left of the center portion of the instrument panel, are grouped according to their function. At the top, the ITT (Interstage Turbine Temperature) indicators and torquemeters are used to set take-off power. Climb and cruise power are established with the torquemeters and propeller tachometers while observing ITT limits. Gas generator (N_1) operation is monitored by the gas generator tachometers. The lower grouping consists of the fuel flow indicators and the oil pressure/temperature indicators.

The ITT indicator gives an instantaneous reading of engine gas temperature between the compressor turbine and the power turbines.

The torquemeters give an indication in foot-pounds of the torque being applied to the propeller.

The propeller tachometer is read directly in revolutions per minute. The N_1 or gas generator tachometer is read in percent of rpm, based on a figure of 37,500 rpm at 100%. Maximum continuous gas generator speed is limited to 38,100 rpm or 104.5% N_1.

Proper observation and interpretation of these instruments provide an indication of engine performance and condition.

A propeller synchroscope, located to the left of the oil pressure/temperature indicators, operates to give an indication of synchronization of the propellers. If the right propeller is operating at a higher rpm than the left, the face of the synchroscope, a black and white cross pattern, spins in a clockwise rotation. Left, or counterclockwise, rotation indicates a higher rpm of the left propeller. This instrument aids the pilot in obtaining complete synchronization of propellers.

PROPELLER SYNCHROPHASER (OPTIONAL)

TYPE I SYSTEM

The Propeller Synchrophaser system automatically matches the RPM of the right propeller (slave propeller) to that of the left propeller (master propeller) and maintains the blades of one propeller at a predetermined relative position with the blades of the other propeller. To prevent the right propeller from losing excessive rpm if the left propeller is feathered while the synchrophaser is on, the synchrophaser has a limited adjustment range from the manual governor setting. Normal governor operation is unchanged but the synchrophaser will continuously monitor propeller rpm and reset the governor as required. A magnetic pickup mounted in each propeller overspeed governor and adjacent to each propeller deice brush block transmits electric pulses to a transistorized control box installed forward of the pedestal.

The control box converts any pulse rate differences into correction commands, which are transmitted to a stepping type actuator motor mounted on the right engine cowl forward support ring. The motor then trims the right propeller governor through a flexible shaft and trimmer assembly to exactly match the left propeller. The trimmer, installed between the governor control arm and the control cable, screws in or out to adjust the governor while leaving the control lever setting constant. A toggle switch installed adjacent to the synchroscope turns the system on. With the switch off, the actuator automatically runs to the center of its range of travel before stopping to assure normal function when used again. To operate the system, synchronize the propellers in the normal manner and turn the synchrophaser on. The system is designed for in-flight operations and is placarded to be off for takeoff and landing. Therefore, with the system on and the landing gear extended, the caution flashers and a yellow light on the caution/advisory annunciator panel, PROP SYNCH ON, will illuminate.

The right propeller rpm and phase will automatically be adjusted to correspond to the left. To change rpm, adjust both propeller controls at the same time. This will keep the right governor setting within the limiting range of the left propeller. If the synchrophaser is on but is unable to adjust the right propeller to match the left, the actuator has reached the end of its travel. To re-center, turn the switch off, synchronize the propellers manually, and turn the switch back on.

TYPE II SYSTEM

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. A thermal element is used to regulate a bypass door which controls the volume of cooling air through the radiator. Engine oil also operates the propeller pitch change mechanism and the engine torquemeter system.

The lubrication system capacity per engine is 3.5 U.S. gallons. The oil tank capacity is 2.3 gallons with 5 quarts measured on the dipstick for adding purposes. Recommended oils and oil changing procedures are listed in the SERVICING Section.

MAGNETIC CHIP DETECTOR

A magnetic chip detector is installed in the bottom of each engine nose gearbox. This detector will activate a red annunciator, L CHIP DETECT or R CHIP DETECT, to alert the pilot of oil contamination indicating possible or pending engine failure.

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. Each switch may be moved downward to the STARTER ONLY position to motor the engine for the purpose of clearing it of fuel without the ignition circuit on. The switch is spring loaded and will return to the center position when released. Moving the switch upward to the ON position activates both the starter and ignition, and the appropriate green 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 takeoff, landing, turbulence, and penetration of icing or precipitation conditions. Arming the system prior to takeoff and turning the system off after landing is required to assure the system being armed in the required conditions. To arm the system, move the required ENG AUTO IGNITION switches, located on the pilot's subpanel, from OFF to ARM. If for any reason the engine torque falls below approximately 410 foot-pounds, the igniter will automatically energize and the IGNITION ON light on the caution/advisory annunciator panel will illuminate. For extended ground operation, the system should be turned off to prolong the life of the igniter units.

INDUCTION AIR SYSTEM

The PT6A-42 is a reverse-airflow engine. The compressor wheels draw ambient air into the engine through the induction air inlet at the lower front of the engine nacelle. As airspeed increases, ram air pressure rises, compressing the air inside the induction air duct. The air then flows into an annular inlet-air chamber located at the aft end of the engine compartment. It then passes through a protective screen and into the primary compressor impeller, where it is further compressed. Then the air is forced through a stator ring and successively through the second and third axial-flow compressor stages. It is finally compressed in the centrifugal-flow compressor stage, then discharged into the turbine plenum assembly. Air from the plenum enters the annular combustion chamber through a series of holes in the aft end of the combustion chamber, and mixes with fuel that is sprayed into the combustion chamber through 14 nozzles mounted around the gas generator case. The air-fuel mixture burns inside the combustion chamber, then the hot gasses expand forward out of the chamber and pass through the compressor turbine stage, both stages of the power turbine, and out to the atmosphere through two exhaust ports located at the side of each nacelle, near the front.

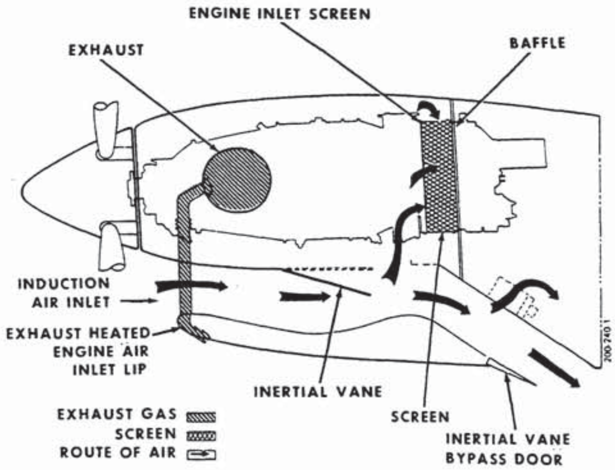
ICE PROTECTION

ENGINE AIR INLET

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

ICE VANES (INERTIAL SEPARATOR SYSTEM)

An inertial separation system is built into each engine air inlet to prevent moisture particles from entering the engine inlet plenum under icing conditions. A movable vane and a bypass door are lowered into the airstream when operating in visible moisture at +5°C or colder, by energizing electrical actuators with the switches, placarded ICE VANE - EXTEND - RETRACT - LEFT & RIGHT, located on the pilot's subpanel. The vane deflects the ram airstream slightly downward to introduce a sudden turn in the airstream to the engine, causing the moisture particles to continue on undeflected, because of their greater momentum, and to be discharged overboard.



ENGINE ICE PROTECTION

While in the icing flight mode, the extended position of the vane and by-pass door is indicated by green annunciator lights, L ICE VANE EXT and R ICE VANE EXT.

In the non-ice-protection mode, the vane and by-pass door are retracted out of the airstream by placing the ice vane switches in the RETRACT position. The green annunciator lights will extinguish. Retraction should be accomplished at +15°C and above to assure adequate oil cooling. The vanes should be either extended or retracted; there are no intermediate positions.

If for any reason the vane does not attain the selected position within 15 seconds, a yellow L ICE VANE or R ICE VANE light illuminates on the caution/advisory panel. In this event, a mechanical backup system is provided, and is actuated by pulling the T-handles just below the pilot's subpanel placarded ICE VANE EMERGENCY - MANUAL EXTENSION - PULL - LEFT ENG - R ENG. Decrease airspeed to 160 knots or less to reduce forces for manual extension. Normal airspeed may then be resumed.

CAUTION

Once the manual override system has been engaged (i.e., anytime the manual ice vane T-handle has been pulled out), do not attempt to retract or extend the ice vanes electrically, even if the T-handle has been pushed back in, until the override linkage in the engine compartment has been properly reset on the ground. (See the maintenance manual for resetting procedure.)

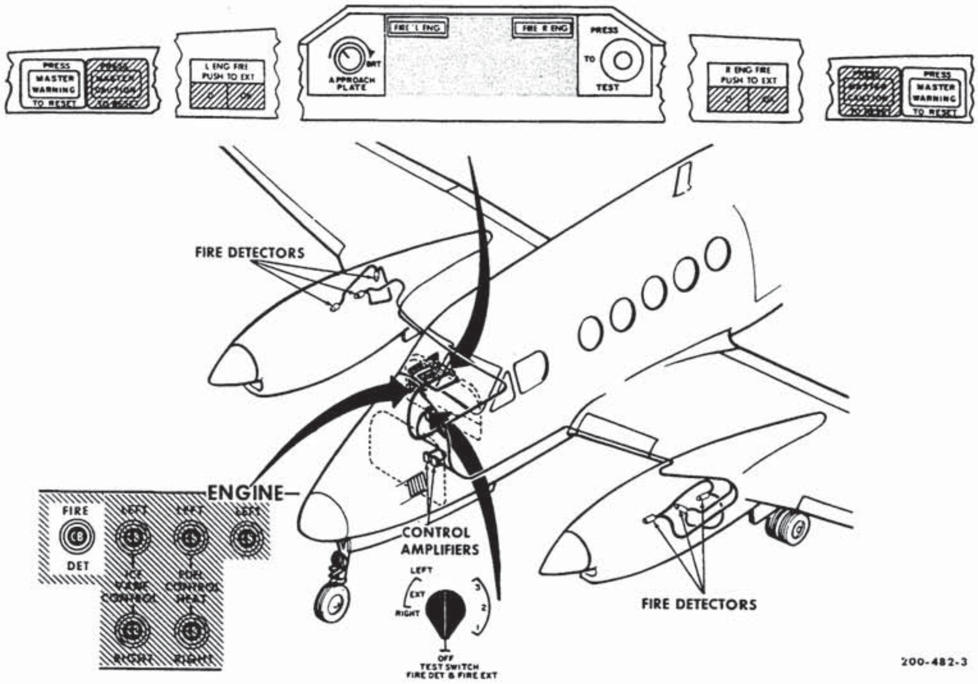
When the vane is successfully positioned with the manual system, the yellow annunciator lights will extinguish. The vane may also be retracted with the manual system. During manual system use, the electric motor switch position must match the manual handle position for a correct annunciator readout.

FUEL CONTROL

The engine fuel system consists of an engine-driven fuel pump, a fuel control unit (FCU), a flow divider and dump valve, dual fuel manifold, fourteen fuel nozzles, and two fuel drain valves. The fuel pump/fuel control unit assembly is mounted on the engine accessory case and is shaft-driven at a speed proportional to that of the compressor turbine. The fuel pump delivers fuel to the FCU. Engine power output is established by power lever position. The power lever is linked to the governor in the FCU which regulates fuel flow to the combustion section and thereby controls N_1 and power output. Increasing N_1 rpm results in increased engine power. System function depends upon the interaction of the fuel control unit governor and the propeller governor. The position of the fuel control unit metering valve is determined by differential pressures that vary proportionately with power required (as sensed by the fuel control unit) and propeller RPM.

The flow divider directs fuel from the metering valve to the primary and secondary fuel manifolds (or primary manifold only, depending on engine power requirements) and thence to the fuel nozzles. The flow divider also incorporates a dump valve that automatically drains residual fuel from both manifolds at engine shutdown. The fuel drain valves drain fuel from the combustion chamber at engine shutdown and at engine false starts. A fuel purge system has been installed in this case and on engine shutdown when fuel manifold pressure subsides, allows the dump poppett valve to open, the purge tank pressure forces fuel out of the engine fuel manifold lines through the nozzles and into the combustion chamber where the fuel is consumed. Constant fuel pressure is maintained by a fuel filter bypass valve and a pressure relief valve.

The oil-to-fuel heater mounted below the fuel pump on the accessory case is a heat exchanger that transfers heat from the engine lubricating oil to preheat the fuel. A fuel temperature-sensing oil bypass valve regulates the fuel temperature by either allowing oil to flow through the heater circuit, or bypass it to the engine oil tank.



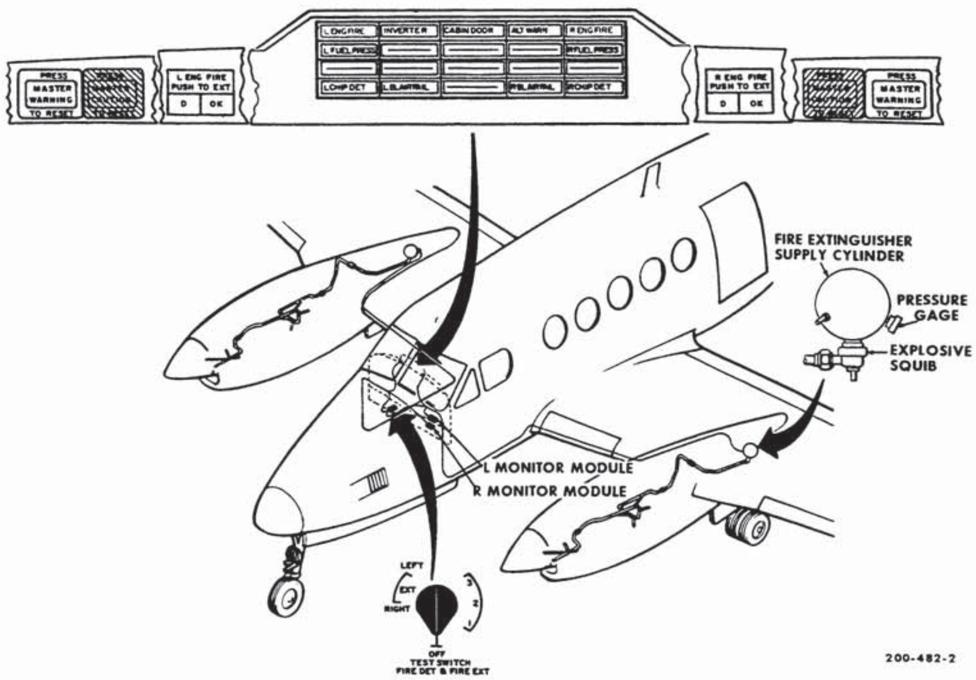
FIRE DETECTION SYSTEM SCHEMATIC

FIRE DETECTION SYSTEM

The fire detection system is designed to provide immediate warning in the event of fire in either engine compartment. The system consists of the following: three photoconductive cells for each engine; a control amplifier for each engine; two red warning lights on the warning annunciator panel, one placarded FIRE L ENG, the other FIRE R ENG; a test switch on the copilot's left subpanel; and a circuit breaker placarded FIRE DET on the right side panel. The six photoconductive-cell flame detectors are sensitive to infrared radiation. They are positioned in each engine compartment so as to receive both direct and reflected rays, thus monitoring the entire compartment with only three photocells.

Conductivity through the photocell varies in direct proportion to the intensity of the infrared radiation striking the cell. As conductivity increases, the amount of current from the electrical system flowing through the flame detector increases proportionally. To prevent stray light rays from signaling a false alarm, a relay in the control amplifier closes only when the signal strength reaches a preset alarm level. When the relay closes, the appropriate left or right warning annunciators illuminate. When the fire has been extinguished, the cell output voltage drops below the alarm level and the relay in the control amplifier opens. No manual resetting is required to reactivate the fire detection system.

The test switch on the copilot's left subpanel, placarded TEST SWITCH - FIRE DET & FIRE EXT, has six positions: OFF - RIGHT EXT - LEFT EXT - 3 - 2 - 1 (If the optional engine-fire-extinguisher system is not installed, the RIGHT EXT and LEFT EXT positions on the left side of the test switch will not be installed.) the three test positions for the fire detector system are located on the right side of the switch (3 - 2 - 1). When the switch is rotated from OFF (down) to any one of these three positions, the output voltage of a corresponding flame detector in each engine is increased to a level sufficient to signal the amplifier that a fire is present. The following should illuminate: the red pilot and copilot MASTER WARNING flashers, and, if the optional engine-fire-extinguisher system is installed, the red lenses placarded L ENG FIRE - PUSH TO EXT and R ENGI FIRE - PUSH TO EXT on the fire-extinguisher activation switches. The system may be tested anytime, either on the ground or in flight. The TEST SWITCH should be placed in all three positions, in order to verify that the circuitry for all six fire detectors is functional. Illumination failure of all the fire detection system annunciators when the TEST SWITCH is in any one of the three flame-detector-test positions indicates a malfunction in one or both of the two detector circuits (one in each engine) being tested by that particular position of the TEST SWITCH.



FIRE EXTINGUISHER SYSTEM SCHEMATIC

FIRE EXTINGUISHER SYSTEM

The optional engine-fire-extinguisher system incorporates a pyrotechnic cartridge inside the nacelle of each engine. When the activation valve is opened, the pressurized extinguishing agent is discharged through a plumbing network which terminates in strategically located spray nozzles.

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-activate switch incorporates three indicator lenses. The red lens, placarded L (or) R ENG FIRE - PUSH TO EXT, warns of the presence of fire in the engine. 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, 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 light will illuminate and remain illuminated, regardless of battery switch position, until the pyrotechnic cartridge has been replaced.

The fire-extinguisher-system test functions incorporated in the TEST SWITCH - FIRE DET & FIRE EXT test the circuitry of the fire extinguisher pyrotechnic cartridges. During preflight, the pilot should rotate the TEST SWITCH to each of the two positions (RIGHT EXT and LEFT EXT) and verify the illumination of the amber D light and the green OK light on each fire-extinguisher-activation switch on the glareshield.

A gage, calibrated in psi, is provided on each supply cylinder for determining the level of charge. The gages should be checked during preflight.

PROPELLER SYSTEM

DESCRIPTION

Each engine is equipped with a conventional three-blade, full-feathering, constant-speed, counter-weighted, reversing, variable-pitch propeller 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.

PRIMARY LOW PITCH STOP

Low pitch propeller position is determined by the primary low pitch stop which is a mechanically actuated hydraulic stop. Beta and reverse blade angles are controlled by the power levers in the Beta and reverse range.

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 operates the propeller by means of this governor. If the constant speed governor should malfunction by requesting more than 2000 rpm, the overspeed governor cuts in at 2080 rpm and dumps oil from the propeller to keep the rpm from exceeding approximately 2080 rpm. A solenoid actuated by the PROP GOV TEST switch located on the pilot's subpanel, is provided for resetting the overspeed governor to approximately 1830 to 1910 rpm for test purposes.

If the propeller sticks or moves too slowly during a transient condition causing the propeller governor to act too slowly to prevent an overspeed condition, the power turbine governor, contained within the constant speed governor housing, acts as a fuel topping governor. When the propeller reaches 2120 rpm, the fuel topping governor limits the fuel flow to the gas generator, reducing N_1 rpm, which in turn prevents the propeller rpm from exceeding approximately 2200 rpm. During operation in the reverse range, the fuel topping governor is reset to approximately 95% propeller rpm before the propeller reaches a negative pitch angle. This ensures that the engine power is limited to maintain a propeller rpm somewhat less than that of the constant speed governor setting. The constant speed governor therefore will always sense an underspeed condition and direct oil pressure to the propeller servo piston to permit operation in Beta and reverse ranges.

AUTOFEATHER 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. Although the system is armed by a switch on the pilot's subpanel, placarded AUTOFEATHER - ARM - OFF - TEST, the completion of the arming phase occurs when both power levers are advanced above 90% N_1 at which time both the right and left indicator lights on the caution/advisory annunciator panel indicate a fully armed system. The annunciator panel lights are green, placarded L AUTOFEATHER and R AUTOFEATHER. The system will remain inoperative as long as either power lever is retarded below 90% N_1 position. The system is designed for use only during takeoff and landing and should be turned off when establishing cruise climb. During takeoff or landing, if torquemeter oil pressure on either engine drops below a prescribed setting, the oil is dumped from the servo, the feathering spring starts the blades toward feather, and the autofeather system on the other engine is disarmed. Disarming of the autofeather portion of the operative engine is further indicated when the annunciator indicator light for that engine extinguishes.

FUEL SYSTEM

The fuel system consists of two separate systems connected by a valve-controlled crossfeed line. The fuel system for each engine is further divided into a main and auxiliary fuel system. The main system consists of a nacelle tank, two wing leading edge tanks, two box section bladder tanks, and an integral (wet cell) tank, all interconnected to flow into the nacelle tank by gravity. This system of tanks is filled from the filler located near the wing tip.

The auxiliary fuel system consists of a center section tank with its own filler opening, and an automatic fuel transfer system to transfer the fuel into the main fuel system.

When the auxiliary tanks are filled, they will be used first. During transfer of auxiliary fuel, which is automatically controlled, the nacelle tanks are maintained full. A swing check valve in the gravity feed line from the outboard wing prevents reverse fuel flow. Upon exhaustion of the auxiliary fuel, normal gravity transfer of the main wing fuel into the nacelle tanks will begin.

An anti-siphon valve is installed in each filler port which prevents loss of fuel or collapse of a fuel cell bladder in the event of improper securing or loss of the filler cap.

The two systems are vented through a recessed ram vent coupled to a protruding heated ram vent on the underside of the wing adjacent to the nacelle. One vent is recessed to prevent icing and the protruding vent, added as a backup, is heated to prevent icing.

All fuel is filtered with a firewall-mounted 20-micron filter. These filters incorporate an internal bypass which opens to permit uninterrupted fuel supply to the engine in the event of filter icing or blockage. In addition, a screen strainer is located at each tank outlet before the fuel reaches the boost and transfer pumps. The main engine driven fuel pump has an integral strainer to protect the pump.

A "differential pressure" fuel purge system is provided and is located in the aft compartment of each nacelle. The system purges the fuel that is left in the fuel manifolds at engine shutdown by forcing the fuel into the nozzles so that it is consumed in the combustion chamber.

FUEL PUMPS

The engine driven fuel pump (high pressure) is mounted on the accessory case in conjunction with the fuel control unit. Failure of this pump results in an immediate flameout. The primary boost pump (low pressure) is also engine driven and is mounted on a drive pad on the aft accessory section of the engine. This pump operates when the gas generator (N_1) is turning and provides sufficient fuel for start, takeoff, all flight conditions (except operation with hot aviation gasoline above 20,000 feet altitude) and operation with crossfeed.

In the event of a primary boost pump failure, the respective red FUEL PRESS light in the annunciator panel will illuminate. This light illuminates when pressure decreases below 10 ± 1 psi. The light will be extinguished by switching on the standby fuel pump on that side, thus increasing pressure above 11 ± 2 psi.

CAUTION

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

When using aviation gasoline during climbs above 20,000 feet, the first indication of insufficient fuel pressure will be an intermittent flicker of the FUEL PRESS lights. A wide fluctuation of the fuel flow indicator may also be noted. These conditions can be eliminated by turning on a standby pump.

An electrically driven standby boost pump (low pressure), located in the bottom of each nacelle tank, performs three functions; it is a backup pump for use in the event of a primary fuel boost pump failure, it is for use with hot aviation gasoline above 20,000 feet, and it is used during crossfeed operations. In the event of an inoperative standby pump, crossfeed can only be accomplished from the side of the operative pump.

SERIALS BB-734, BB-793, BB-829, BB-854 thru BB-870, BB-874 thru BB-891, BB-894, BB-896 thru BB-911, BB-913 thru BB-1095, and BB-1097; BL-37 thru BL-57:

Electrical power to operate the standby boost pumps is controlled by lever lock toggle switches, placarded STANDBY PUMP - ON - OFF, located on the fuel control panel and is supplied power from two independent sources. One source of power for either the right or the left standby pump is provided through the number 3 or number 4 feeder bus and is protected by a 10-ampere circuit breaker located on the fuel control panel. This power is only available when the master switch is turned on. Another source of power comes directly from the battery through the hot battery bus and is protected by dual 5-ampere fuses located in the right wing center section. The fuse panel may be serviced through an access door on the bottom side of the wing outboard of the battery. This power source makes power available for the pumps at all times, regardless of the battery master switch position. These circuits are protected by diodes to prevent the failure of one circuit from disabling the other circuit. During shutdown, make certain both standby pump switches are off to prevent battery discharge.

SERIALS BB-1096, BB-1098 and After; BL-58 and After:

Electrical power to operate the standby boost pumps is controlled by lever lock toggle switches, placarded STANDBY PUMP - ON - OFF, located on the fuel control panel. It is supplied power from the number 3 or number 4 feeder bus, and is protected by a 10-ampere circuit breaker located on the fuel control panel. This power is only available when the master switch is turned on. These circuits are protected by diodes to prevent the failure of one circuit from disabling the other circuit.

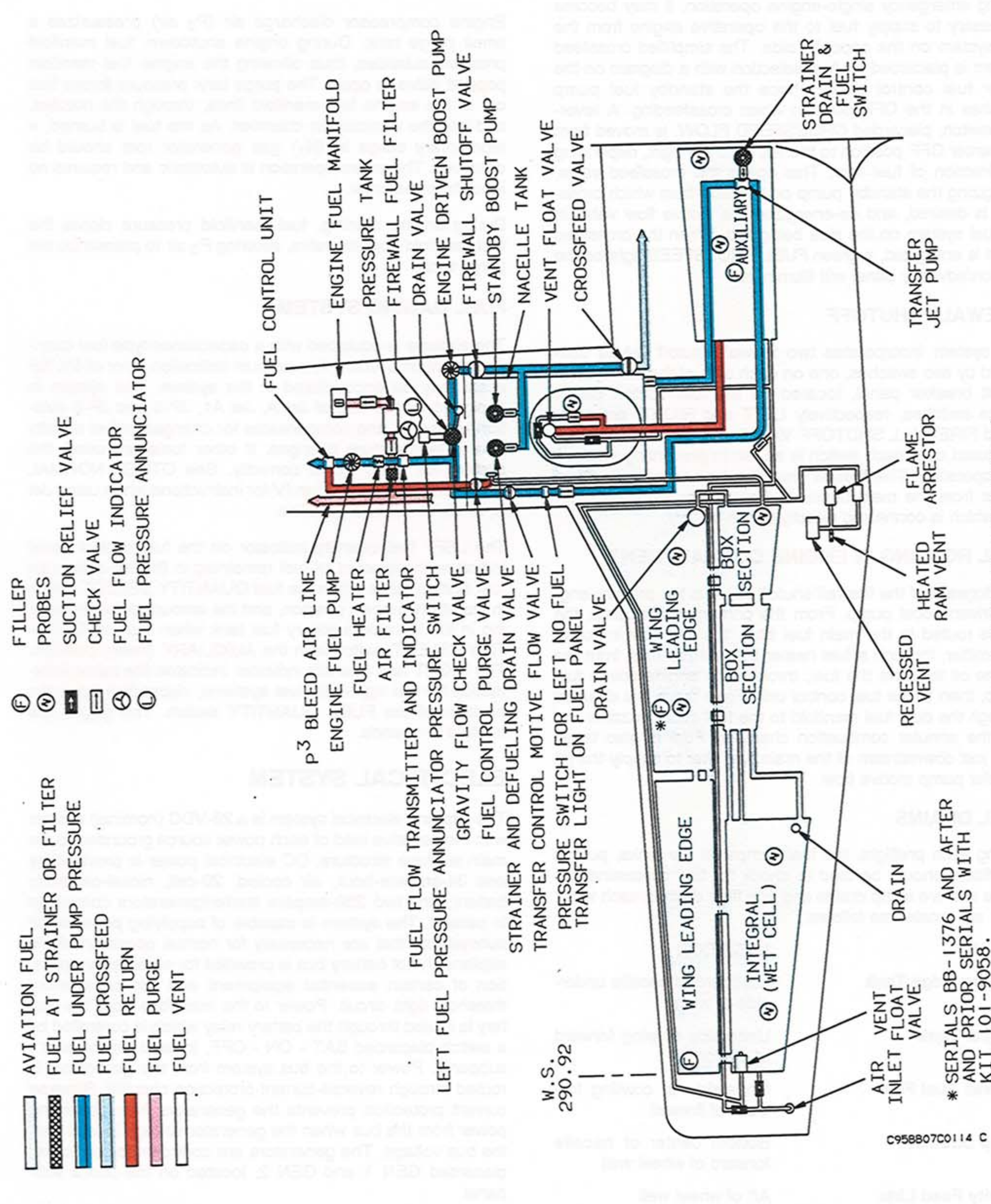
AUXILIARY FUEL TRANSFER SYSTEM

The auxiliary tank fuel transfer system automatically transfers the fuel from the auxiliary tank to the nacelle tank without pilot action. Motive flow to a jet pump mounted in the auxiliary tank sump is obtained from the engine fuel plumbing system downstream from the engine driven boost pump and and routed through the transfer control motive flow valve. The motive flow valve is energized to the open position by the control system to transfer auxiliary fuel to the nacelle tank to be consumed by the engine during the initial portion of the flight. When an engine is started, pressure at the engine driven boost pump closes a pressure switch which, after a 30 to 50 second time delay to avoid depletion of fuel pressure during starting, energizes the motive flow valve. When the auxiliary fuel is depleted, a low level float switch de-energizes the motive flow valve after a 30 to 60 second time delay provided to prevent cycling of the motive flow valve due to sloshing fuel.

In the event of a failure of the motive flow valve or the associated control circuitry, the loss of motive flow pressure when there is still fuel remaining in the auxiliary fuel tank is sensed by a pressure switch and float switch, respectively, which illuminates a light placarded NO TRANSFER on the fuel control panel. During engine start, the pilot should note that the NO TRANSFER lights extinguish 30 to 50 seconds after engine start. A manual override is incorporated as a backup for the automatic transfer system. This is initiated by placing the AUX TRANSFER switch, located in the fuel control panel to the OVERRIDE position.

USE OF AVIATION GASOLINE

If aviation gasoline must be used as an emergency fuel, it will be necessary to determine how many hours the airplane is operated on gasoline. Since the gasoline is being mixed with the regular fuel, it is expedient to record the number of gallons of gasoline taken aboard for each engine. Each engine is permitted 150 hours of operation on aviation gasoline between overhauls. This means that if one engine has an average fuel consumption of 50 gallons per hour, for example, it is allowed 7500 gallons of aviation gasoline between overhauls. (Two engines; 15,000 gallons between overhauls.)



FUEL SYSTEM SCHEMATIC

CROSSFEED

During emergency single-engine operation, it may become necessary to supply fuel to the operative engine from the fuel system on the opposite side. The simplified crossfeed system is placarded for fuel selection with a diagram on the upper fuel control panel. Place the standby fuel pump switches in the OFF position when crossfeeding. A lever-lock switch, placarded CROSSFEED FLOW, is moved from the center OFF position to the left or to the right, depending on direction of fuel flow. This opens the crossfeed valve, energizing the standby pump on the side from which crossfeed is desired, and de-energizes the motive flow valve in the fuel system on the side being fed. When the crossfeed mode is energized, a green FUEL CROSSFEED light on the caution/advisory panel will illuminate.

FIREWALL SHUTOFF

The system incorporates two firewall shutoff valves controlled by two switches, one on each side of the fuel system circuit breaker panel, located on the fuel control panel. These switches, respectively LEFT and RIGHT, are placarded FIREWALL SHUTOFF VALVE - OPEN - CLOSED. A red guard over each switch is an aid in preventing inadvertent operation. The firewall shutoff valves receive electrical power from the main buses and also from the hot battery bus which is connected directly to the battery.

FUEL ROUTING IN ENGINE COMPARTMENT

Just forward of the firewall shutoff valve is the primary engine driven boost pump. From the primary boost pump, the fuel is routed to the main fuel filter, the fuel flow indicator transmitter, through a fuel heater that utilizes heat from the engine oil to warm the fuel, through the engine driven fuel pump, then to the fuel control unit. From there it is directed through the dual fuel manifold to the fuel outlet nozzles and into the annular combustion chamber. Fuel is also taken from just downstream of the main fuel filter to supply the jet transfer pump motive flow.

FUEL DRAINS

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

DRAINS	LOCATION
Leading Edge Tank	Outboard of nacelle underside of wing
Integral Tank	Underside of wing forward of aileron
Firewall Fuel Filter	Underside of cowling forward of firewall
Sump Strainer	Bottom center of nacelle forward of wheel well
Gravity Feed Line	Aft of wheel well
Auxiliary Tank	At wing root just forward of the flap

FUEL PURGE SYSTEM

Engine compressor discharge air (P_3 air) pressurizes a small purge tank. During engine shutdown, fuel manifold pressure subsides, thus allowing the engine fuel manifold poppett 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.

During engine starting, fuel manifold pressure closes the fuel manifold poppett valve, allowing P_3 air to pressurize the purge tank.

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 A1, 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. See OTHER NORMAL PROCEDURES in Section IV for instructions when using Jet B, JP-4, or aviation gasoline.

The LEFT fuel quantity indicator on the fuel control panel indicates the amount of fuel remaining in the left-side main fuel system tanks when the fuel QUANTITY SELECT switch in the MAIN (upper) position, and the amount of fuel remaining in the left-side auxiliary fuel tank when the fuel QUANTITY SELECT switch is in the AUXILIARY (lower) position. The RIGHT fuel quantity indicator indicates the same information for the right-side fuel systems, depending upon the position of the FUEL QUANTITY switch. The gages are marked in pounds.

ELECTRICAL SYSTEM

The airplane electrical system is a 28-VDC (nominal) system with the negative lead of each power source grounded to the main airplane structure. DC electrical power is provided by one 34-ampere-hour, air cooled, 20-cell, nickel-cadmium battery, and two 250-ampere starter/generators connected in parallel. The system is capable of supplying power to all subsystems that are necessary for normal operation of the airplane. A hot battery bus is provided for emergency operation of certain essential equipment and the cabin entry threshold light circuit. Power to the main bus from the battery is routed through the battery relay which is controlled by a switch placarded BAT - ON - OFF, located on the pilot's subpanel. Power to the bus system from the generators is routed through reverse-current-protection circuitry. Reverse current protection prevents the generators from absorbing power from the bus when the generator voltage is less than the bus voltage. The generators are controlled by switches, placarded GEN 1 and GEN 2, located on the pilot's subpanel.

NOTE

In order to turn the generator ON, the generator control switch must first be held upward in the spring-loaded RESET position for a minimum of one second, then released to the ON position.

Starter power to each individual starter/generator is provided from the main bus through a starter relay. The start cycle is controlled by a three-position switch for each engine, placarded IGNITION AND ENGINE START - ON - OFF - STARTER ONLY, on the pilot's subpanel. The starter/generator drives the compressor section of the engine through the accessory gearing. The starter/generator initially draws approximately 1100 amperes, then drops rapidly to about 300 amperes as the engine reaches 20% N_1.

Power is supplied from three sources: the battery, the right generator, and the left generator. The generator buses are interconnected by two 325-ampere current limiters. The entire bus system operates as a single bus, with power being supplied by the battery and both generators. There are four dual-fed sub-buses. Each sub-bus is supplied power from either generator main bus through a 60-amp limiter, a 70-amp diode, and a 50-amp circuit breaker. All electrical loads are divided among these buses except as noted on the accompanying Power Distribution Schematic. The equipment on the buses is arranged so that all items with duplicate functions (such as right and left landing lights) are connected to different buses. Among the loads on the generator buses are the number 1 and number 2 inverters. Through relay circuitry, the INVERTER selector switch activates the selected inverter, which provides 400-hertz, 115-volt, alternating current to the avionics equipment,and 400-hertz, 26 VAC to the torquemeters. The volt/frequency meter indicates the voltage and frequency of the alternating current being supplied to the avionics equipment.

The generators are controlled by individual voltage regulators which maintain a constant voltage during variations in engine speed and electrical load requirements. The generators are connected to the voltage regulating circuits by means of control switches located on the pilot's left subpanel. The voltage regulating circuit will automatically disable or enable a generator's output to the bus. The load on each generator is indicated by the respective left and right volt/loadmeter located in the overhead panel.

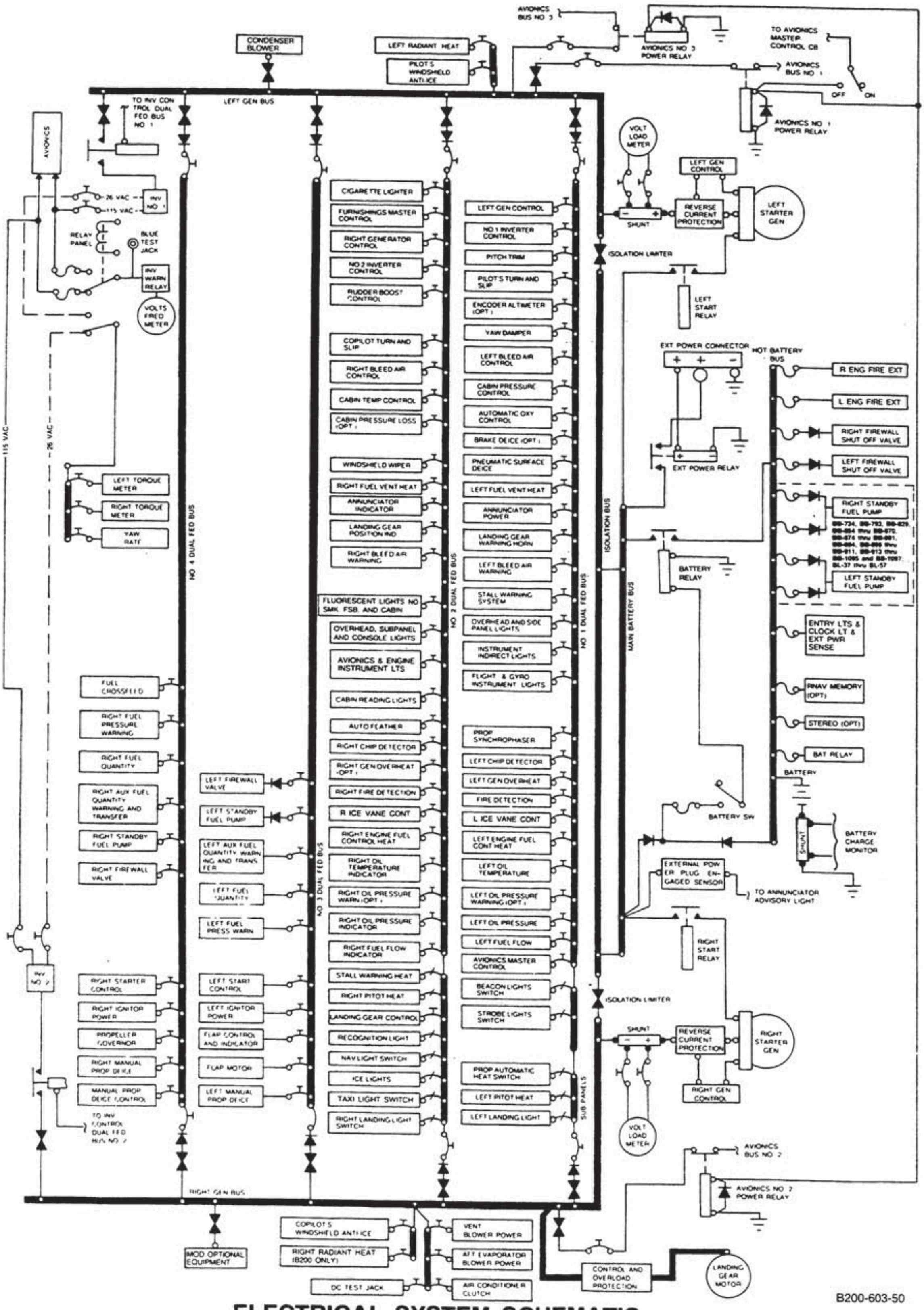
Overheating of the nickel-cadmium battery will cause the battery charge current to increase. Therefore, a yellow BATTERY CHARGE caution annunciator light is provided in the caution/advisory annunciator panel to alert the pilot of the possibility of battery overheating. A Battery Charge Current Detector will cause illumination of the yellow BATTERY CHARGE annunciator whenever the battery charge current is above 7 amps. Thus, the BATTERY CHARGE annunciator may occasionally illuminate for short intervals when heavy loads switch off. Following a battery-powered engine start, the battery recharge current is very high and causes illumination of the BATTERY CHARGE annunciator, thus providing an automatic self-test of the detector and the battery. As the battery approaches a full charge and the charge current decreases below 7 amps, the annunciator will extinguish. This will normally occur within a few minutes after an engine start, but may require a longer time if the battery has a low state of charge, low charge voltage per cell (20-cell battery), or low battery temperature. This system is designed for continuous monitoring of the battery condition.

Illumination of the BATTERY CHARGE annunciator in flight cautions the pilot that conditions may exist that may eventually damage the battery. The operator should check the battery charge current with the loadmeter. This is accomplished by turning off one generator and noting the load on the remaining generator. Turn off the battery and note the loadmeter change. If the change is greater than .025, the battery should be left off the bus and should be inspected after landing. If the annunciator remains on after the battery switch is moved to the OFF position, a malfunction is indicated in either the battery system or charge current detector, in which case the airplane should be landed as soon as practicable. The battery switch should be turned ON for landing in order to avoid electrical transients caused by power fluctuations.

EXTERNAL POWER

For ground operation, an external power socket, located under the right wing outboard of the nacelle, is provided for connecting an auxiliary power unit. A relay in the external power circuit will close only of the external source polarity is correct. The battery switch *must* be on before the external power relay will close and allow external power to enter the airplane electrical system. The battery will also tend to absorb voltage transients when operating avionics equipment and during engine starts. Otherwise, the transients might damage the many solid state components in the airplane.

For starting, an external power source capable of supplying up to 1000 amperes (300 amperes maximum continuous) should be used. A caution light on the caution/advisory annunciator panel, EXT PWR, is provided to alert the operator when an external DC power plug is connected to the airplane.



ELECTRICAL SYSTEM SCHEMATIC

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 overhead light control panel lights,fuel control panel lights, engine instrument lights, radio panel lights, subpanel and console lights, pilot and copilot instrument lights, and gyro instrument lights. 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 gage, turns ON and OFF the lights near the outside air temperature gage.

CABIN

A three-position switch on the copilot's subpanel, placarded CABIN LIGHTS - START BRIGHT - DIM - OFF, controls the fluorescent cabin lights. The switch to the right of the interior light switch activates the cabin NO SMOKING/FASTEN SEAT BELT signs and accompanying chimes. This three-position switch is placarded CABIN LIGHTS - NO SMOKE & FSB - FSB - OFF.

The baggage-area light is controlled by a two-position switch just inside the airstair door aft of the door frame and is connected to the hot battery bus.

A threshold light is located forward of the airstair door at floor level, and an aisle light is located at floor level aft of the spar cover. A switch adjacent to the threshold light turns both these lights on and off. The switch also turns the exterior entry light on and off. When the airstair door is closed, all the lights controlled by the threshold light switch will extinguish.

When the master switch is on, the individual reading lights along the top of the cabin may be turned on or off by the passengers with a push-button switch adjacent to each light.

EXTERIOR

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

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

A flush-mounted floodlight forward of the flaps in the bottom of the left wing may be installed. This entry light provides illumination of the area around the airstair door, to provide passenger convenience at night. It is controlled by the threshold light switch just inside the door on the forward door frame, and will extinguish automatically whenever the cabin door is closed.

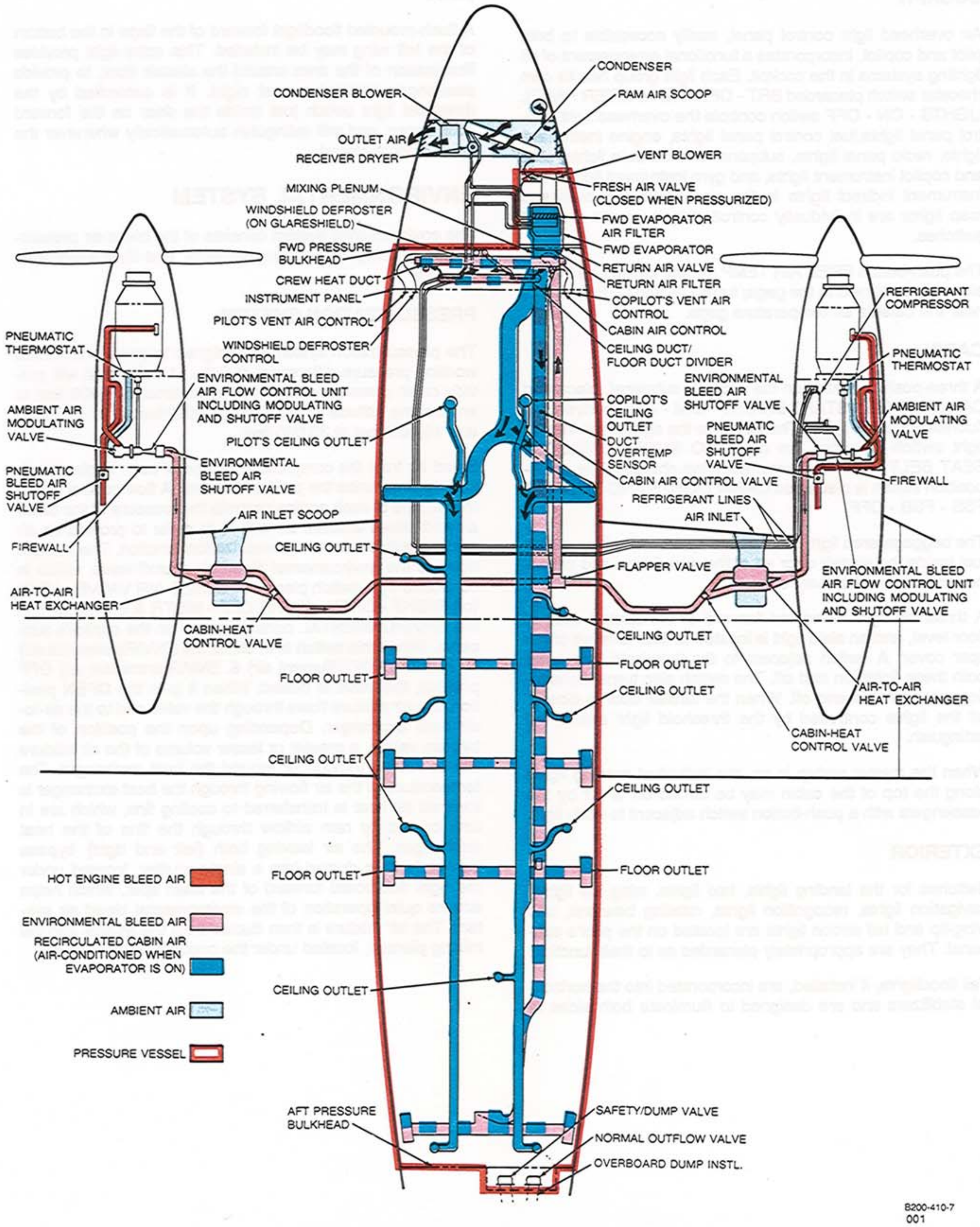
ENVIRONMENTAL SYSTEM

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 normal working pressure differential of 6.5 ± .1 psi, which will provide cabin pressure altitudes of approximately; 2800 feet at an airplane altitude of 20,000 feet, 8600 feet at 31,000 feet, and 10,400 feet at 35,000 feet.

Bleed air from the compressor section of each engine is utilized to pressurize the pressure vessel. A flow control unit in the nacelle of each engine controls the pressure of the bleed air and mixes ambient air with it, in order to provide an air mixture suitable for the pressurization function. The mixture flows to the environmental bleed air shutoff valve, which is controlled by a switch placarded BLEED AIR VALVE - LEFT (or) RIGHT - OPEN - ENVIR OFF - INSTR & ENVIR OFF in the ENVIRONMENTAL controls group on the copilot's subpanel. When this switch is in either the ENVIR(onmental air) OFF or the INSTR(ument air) & ENVIR(onmental air) OFF position, the valve is closed. When it is in the OPEN position, the air mixture flows through the valve and to the air-to-air heat exchanger. Depending upon the position of the bypass valves, a greater or lesser volume of the air mixture will be routed through or around the heat exchanger. The temperature of the air flowing through the heat exchanger is lowered as heat is transferred to cooling fins, which are in turn cooled by ram airflow through the fins of the heat exchanger. The air leaving both (left and right) bypass valves, is then ducted into a single muffler, located under the right floorboard forward of the main spar, which helps ensure quiet operation of the environmental bleed air system. The air mixture is then ducted from the muffler into the mixing plenum, located under the copilot's floorboard.



B200 ENVIRONMENTAL SYSTEM SCHEMATIC

A partition divides the mixing plenum into two sections. One section supplies the floor-outlet duct, and the other supplies the ceiling-outlet duct. Both sections receive recirculated cabin air from the forward vent blower. This air passes through the forward evaporator, so it will be cooled if the air conditioner is operating. Even in the event that the forward vent blower becomes inoperative, some air will still be circulated, due to a special nozzle in the discharge side of the mixing plenum.

The environmental bleed air duct is routed into the floor-duct section of the mixing plenum, then curves back to discharge the environmental bleed air toward the aft end of the floor-duct section of the mixing plenum. Forward of the discharge end of the environmental bleed air duct, warm air is tapped off and ducted up through the top of the mixing plenum and into the crew heat duct, which also receives recirculated cabin air from the mixing plenum. A valve on the forward side of the crew heat duct allows air to be tapped off for delivery to the windshield defroster when the DEFROST AIR knob on the pilot's left subpanel is pulled out. Eyeball outlets at the left and right sides of the glareshield are also supplied air from the defrost air tap.

The air from the environmental bleed air duct is mixed with recirculated cabin air (which may or may not be air conditioned) in the mixing plenum, then routed into the floor-outlet duct. This pressurized air is then introduced into the cabin through the floor registers. Finally, the air flows out of the pressure vessel through the outflow valve, located on the aft pressure bulkhead. A silencer on the outflow and safety/dump valves ensures quiet operation.

The mixture from both flow control units is delivered to the pressure vessel at a rate which can vary from about 8 to 16 pounds per minute, depending upon ambient temperature and pressure altitude. Pressure within the cabin and the rate of cabin-pressure changes are regulated by pneumatic modulation of the outflow valve, which controls the rate at which air can escape from the pressure vessel.

A vacuum-operated safety valve is mounted adjacent to the outflow valve on the aft pressure bulkhead. It is designed to serve three functions: to provide pressure relief in the event of malfunction of the normal outflow valve; to allow depressurization of the pressure vessel whenever the cabin pressure switch is moved into the DUMP position; and to keep the pressure vessel unpressurized while the airplane is on the ground with the left landing-gear safety switch compressed. A negative-pressure relief function is also incorporated into both the outflow and the safety valves. This prevents outside atmospheric pressure's exceeding cabin pressure by more than 0.1 psi during rapid descents, even if bleed air inflow ceases.

On airplanes BB-1284 and after, both the outflow valve and the safety valve are vented overboard, to preclude moisture build-up in the aft fuselage.

When the BLEED AIR VALVE switches on the copilot's subpanel are OPEN (up), the air mixture from the flow control units enters the pressure vessel. While the airplane is on the ground, a left landing gear safety-switch-activated solenoid valve in each flow control unit keeps the ambient-air intake port closed, allowing only bleed air to be delivered into the pressure vessel. At lift-off, the safety valve closes and the ambient air shutoff solenoid valve in the left flow control unit opens; approximately 6 seconds later, the solenoid in the right flow control unit opens. Consequently, by increasing the volume of airflow into the pressure vessel in stages, excessive pressure bumps during takeoff are avoided.

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 -1000 to +15,000 feet MSL. The rate control selector knob is placarded RATE - MIN - MAX. The rate at which the cabin pressure altitude changes from the current value to the selected value is controlled by rotating the rate control selector knob. The rate of change selected may be from approximately 50-300 to approximately 2000 ±500 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.

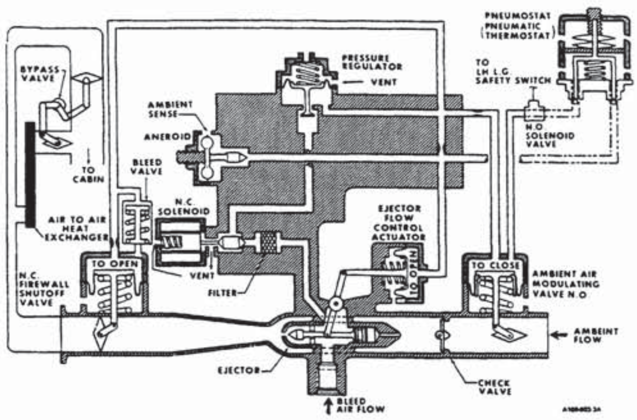
The cabin pressure switch, located forward of the pressurization controller on the pedestal, is placarded CABIN PRESS - DUMP - PRESS - TEST. When this switch is in the DUMP (forward) position, the safety valve is held open, so that the cabin will depressurize and/or remain unpressurized. When it is in the PRESS (center) position, the safety valve is normally closed in flight, and the outflow valve is controlled by the pressurization controller, so that the cabin will pressurize. When the switch is held in the spring-loaded TEST (aft) position, the safety valve is held closed, bypassing the landing-gear safety switch, to facilitate testing of the pressurization system on the ground.

Prior to takeoff, the cabin altitude selector knob should be adjusted so that the ACFT ALT scale on the indicator dial indicates an altitude approximately 1000 feet above the planned cruise pressure altitude, and the CABIN ALT scale indicates an altitude at least 500 feet above the takeoff field pressure altitude. The rate 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 valve and safety valve. Either or both valves will then override the cabin pressurization controller in order to limit the cabin-to-ambient pressure differential to 6.5 ± .1 psi. If the cabin pressure altitude should reach a value of 12,500 feet, a pressure-sensing switch mounted on the overhead light control panel will close. This causes the ALT WARN annunciator light to illuminate, warning the pilot of operation requiring the use of oxygen. If an auto-deployment oxygen system is installed, a pressure-sensing switch mounted on the cabin sidewall forward of the emergency exit also closes, signalling the passenger oxygen masks to drop out. 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.

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 negative-pressure relief function modulates the outflow and the safety valve poppets toward the fully open position, thereby equalizing the pressure inside and outside the pressure vessel. As the airplane continues to descend below the pre-selected cabin pressure altitude, the cabin will be unpressurized and will follow the airplane rate of descent to touchdown.

FLOW CONTROL UNIT (THRU BB-1179, BL-69)

Each flow control unit consists of an ejector and an integral bleed air modulating valve, firewall shutoff valve, ambient air modulating valve, and a check valve that prevents the bleed air from escaping through the ambient air intake. The flow of bleed air through the flow control unit is controlled as a function of atmospheric pressure and temperature. Ambient air flow is controlled as a function of temperature only.



NOTE: Relative size of components intentionally exaggerated

LH LG - left hand Landing Gear

N.C. - normally closed

N.O. - normally open

BLEED AIR FLOW CONTROL UNIT (Thru BB-1179, BL-69)

When the BLEED AIR VALVE switches on the copilot's subpanel are OPEN, an electronic solenoid valve on each flow control unit opens to allow the bleed air into the unit. As the bleed air enters the flow control unit, it passes through a filter before going to the reference pressure regulator. The regulator will reduce the pressure to a constant value (18 to 20 psi). This reference pressure is then directed to the various components within the flow control unit that regulate the output to the cabin. One reference pressure line is routed to the firewall shutoff valve located downstream of the ejector. An orifice is placed in the line immediately before the shutoff valve to provide a controlled opening rate. At the same time, the reference pressure is directed to the ambient air modulating valve located upstream of the ejector. A pneumatic thermostat with a variable orifice is connected to the modulating valve. The pneumatic thermostat is located on the lower aft side of the fireseal forward of the firewall. The bimetallic sensing discs of the thermostat are inserted into the cowling intake. These discs sense ambient temperature and regulate the size of the thermostat orifices. Warm air will open the orifice; cold air will restrict it, until, at -30°F, the orifice will completely close. When the variable orifice is closed, the pressure buildup will cause the modulating valve to close off the ambient air source. An electric solenoid valve located in the line to the pneumatic thermostat is wired to the left landing gear safety switch. When the airplane is on the ground, the solenoid valve is closed, thereby directing the pressure to the modulating valve, causing it to shut off the ambient air source. The exclusion of ambient air allows faster cabin warmup during cold weather operation.

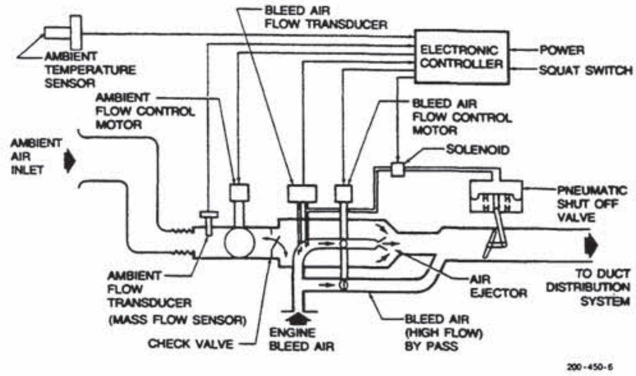
An electric circuit containing a time-delay relay is wired to the above-mentioned solenoid valves to allow the left valve to operate approximately 6 seconds before the right valve. This precludes the simultaneous opening of the shutoff valves, which would result in a sudden pressure surge into the cabin. A check valve, located downstream from the modulating valve, prevents the loss of bleed air through the ambient air intake. At the same time the above operations have been taking place in the control unit, reference pressure is directed to the ejector flow control actuator. This actuator is connected to another variable orifice of the pneumatic thermostat and avariable orifice controlled by an isobaric aneroid. The thermostat orifice is restricted by decreasing ambient temperature, and the isobaric aneroid orifice is restricted by decreasing ambient pressure. The restriction of either orifice will cause a pressure buildup on the ejector flow control actuator, permitting more bleed air to enter the ejector.

FLOW CONTROL UNIT (BB-1180 AND AFTER, BL-70 AND AFTER, AND EARLIER AIRPLANES IN COMPLIANCE WITH BEECH SERVICE BULLETIN 2002).

To meet cabin pressurization and heating requirements, bleed air is routed from the compressor section of each engine to electronic flow control valves mounted on the firewalls. The flow control unit controls the flow of ambient and bleed air. Each flow control unit consists of an ambient temperature sensor, an electronic controller and a pneumatic control valve. The control valve consists of an air ejector, bleed air flow modulating valve with bypass, bleed air valve switch, ambient air flow modulating valve, firewall shut-off valve, a check valve that prevents the bleed air from escaping through the ambient air intake, solenoid valve, bleed air flow transducer, and an ambient air flow transducer.

The ambient air flow modulating valve is actuated by a safety switch located on the left-hand landing gear. When weight is on the landing gear, the ambient air valve is closed off, preventing entry of contaminants into the environmental system during ground operations. After the engine is started and the flow control unit energized, the bleed air modulating valve will close and actuate the bleed air shaft microswitch. When the microswitch is actuated, the electronic controller signals the solenoid to open, this enables the pneumatic line to the firewall shutoff valve to be pressurized and to open the valve. The bleed air shaft continues to open until the desired bleed air flow rate to the cabin is attained. The correct flow rate is determined by the bleed air flow transducer, the ambient air temperature sensor, and the electronic controller.

As the airplane climbs to higher altitudes or enters a cooler environment, the ambient air flow is gradually reduced and the bleed air flow is increased to maintain cabin heat. At approximately 50°F ambient temperature, the ambient air valve is completely closed and the bleed air valve bypass section is opened to allow more bleed air flow around the air ejector.



BLEED AIR FLOW CONTROL UNIT (BB-1180 and After, BL-70 and After, and Earlier Airplanes in Compliance With Beech Service Bulletin 2002)

The bleed air mixture passes from the flow control valves to heat exchangers located in each side of the center section. The heat exchangers remove any excess heat from the bleed air mixture by passing it through an air cooled core. The cooling air from the heat exchanger core is ducted from the leading edge of the center section through the heat exchanger core and exhausted through the louvered plates on the lower side of the center section. Should full utilization of the bleed air be required for the cabin environment, bypass valves provide an alternate path around the heat exchangers. At the juncture of the LH and RH bleed air ducts is a check valve that assures continued cabin pressurization and heat, even though one engine is shut down. Forward of the check valve is the muffler which quiets the movement of air in the system. From the muffler, the bleed air mixture moves to the mixing plenum where it is mixed with recirculated cabin air. If additional cooling of the bleed air mixture is desired, cool air from the air conditioner may be injected into the mixing plenum.

On takeoff, the safety switch immediately reopens the LH ambient air valve. A time delay module integrated into the ambient air circuitry delays opening of the RH valve for approximately seven seconds after opening the LH valve. This delay prevents cabin pressurization surges.

Three-position toggles switches on the RH subpanel
f the flow control valves. When these switches are in the up position (Bleed Air Valves OPEN), the flow control valves provide the required cabin heat and pressurization. With the

switches in the center or neutral position (ENVIR OFF), the flow controls shut off all environmental bleed air. In this position the pneumatic bleed air, which tees off the bleed air lines forward of the flow control valves, is still in operation. When the switches are moved to the down or keying position (INST & ENVIR OFF), all bleed air to the cabin is shut off at the firewalls.

The rate of ambient air flow through the flow control valve is in direct proportion to the temperature of the air in which the airplane is operating. At high atmospheric temperatures, the amount of bleed air used is minimized by mixing ambient air inside the control valve through an air ejector. When atmospheric temperatures drop, the ambient air is reduced or shut off. An ambient air temperature sensor (thermistor), adjacent to the ambient air inlet, sends temperature signals to the electronic controller enabling the controller to determine the correct mix of ambient and bleed air for the cabin environment.

If bleed air pressure or temperature become excessive, the electronic controller will automatically shut off the environmental air at the affected valve.

UNPRESSURIZED VENTILATION

Fresh-air ventilation is provided from two sources. One source, which is available during both the pressurized and the unpressurized mode, is the bleed air heating system. This air mixes with recirculated cabin air and enters the cabin through the floor registers. The volume of air from the floor registers is regulated by using the CABIN AIR control knob located on the copilot's subpanel.

The second source of fresh air, which is available during the unpressurized mode only, is ambient air obtained (through a check valve) from the condensor section in the nose of the airplane. During pressurized operation, cabin pressure forces the check valve closed. During the unpressurized mode, a spring holds the check valve open, so that the forward blower can draw this air into the cabin. The ambient air then mixes with recirculated cabin air, goes through the forward blower, through the forward evaporator, (if it is operating, the air will be cooled), into the mixing plenum, into both the ceiling-outlet and the floor-outlet duct, and into the cabin through all the ceiling and floor outlets. Air ducted to each individual ceiling eyeball outlet can be directionally controlled by moving the eyeball in the socket. Volume is regulated by twisting the outlet to open or close the damper.

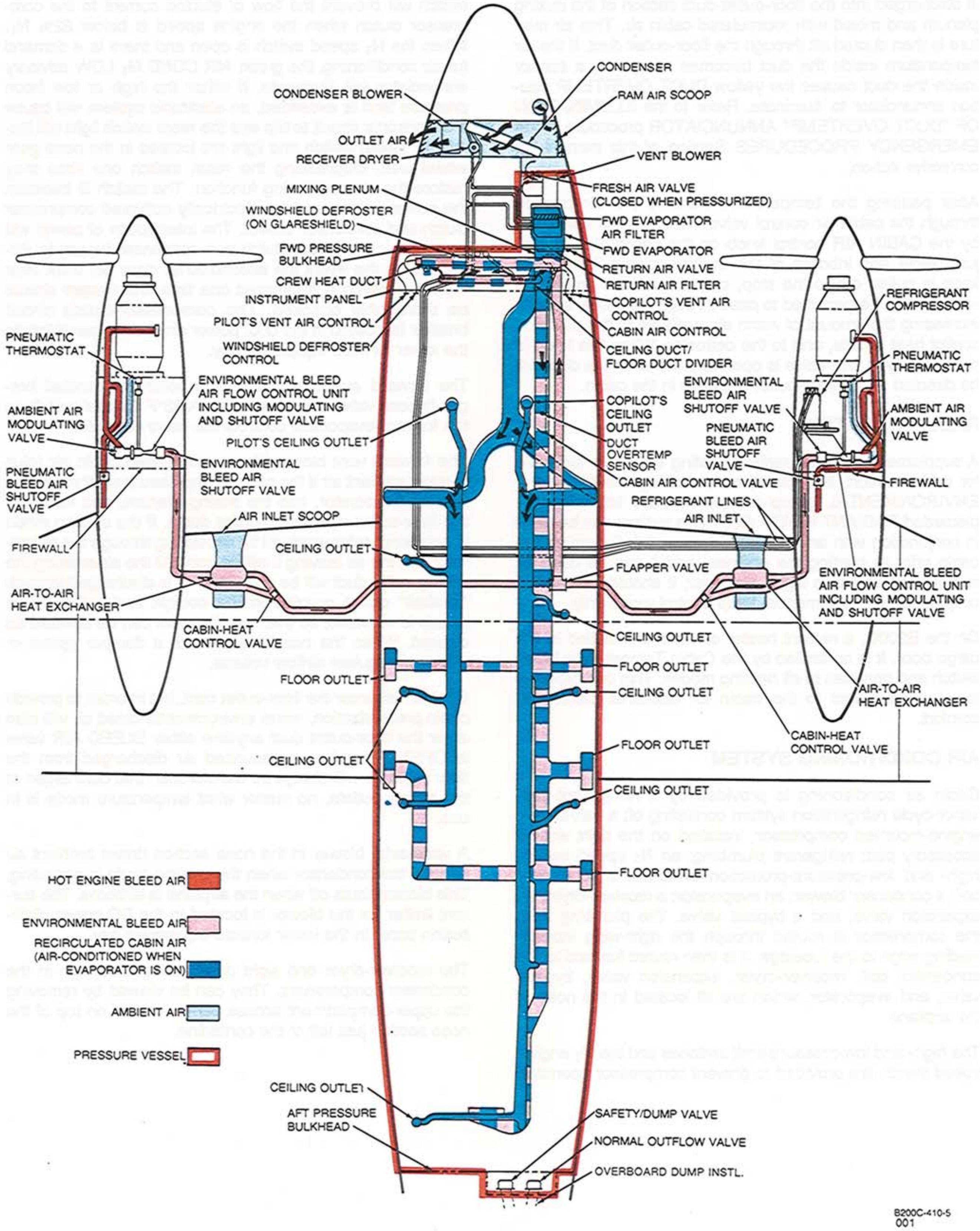
HEATING

When air is compressed, its temperature is increased. Therefore, the bleed air extracted from the compressor section of each engine for pressurization purposes is hot. This heat is utilized to warm the cabin.

When the left landing gear safety switch is in the on-the-ground position, the ambient air valve in each flow control unit is closed. Consequently, only bleed air is delivered to the environmental bleed air duct when the airplane is on the ground. In flight, the ambient air valve is open, and ambient air is mixed with the engine bleed air in the flow control unit. This environmental bleed air mixture is then routed into the cabin.

If the environmental bleed air mixture is too warm for cabin comfort, the bypass valve routes some or all of it through the air-to-air heat exchanger, located in the wing center section. The position of the damper in the cabin-heat control valve is determined by positioning of the controls in the ENVIRONMENTAL group on the copilot's subpanel. An air intake on the leading edge of the inboard wing brings ram air into the heat exchanger to cool the bleed air. After leaving the heat exchanger, the ram air is ducted overboard through louvers on the underside of the wing.

After the bleed air passes through or around the air-to-air heat exchanger, it is ducted to the mixing plenum. Some of the environmental bleed air is tapped off and delivered to the pilot/copilot heat duct, which is located below the instrument panel. An outlet at each end of this duct is provided to deliver warm air to the pilot and copilot. A mechanically controlled damper in each outlet permits the volume of airflow to be regulated. The pilot's damper is controlled by the PILOT AIR knob, located on the pilot's subpanel just below the outboard of the control column. The copilot's damper is controlled by the CO-PILOT AIR knob, located on the copilot's subpanel just below and outboard of the control column. The DEFROST AIR control knob is located on the pilot's subpanel just below and inboard of the control column. This knob controls a valve at the forward side of the pilot/copilot heat duct which admits air to two ducts that deliver the warm air to the defroster, located just below the windshields in the top of the glareshield. An air plenum built into the glareshield feeds air to "eyeball" outlets on the left and right sides. Defrost air is the air source for the "eyeball" outlets; thus the use of the DEFROST AIR control knob also controls air to the eyeball outlets.



B200C ENVIRONMENTAL SYSTEM SCHEMATIC

The remainder of the air in the environmental bleed air duct is discharged into the floor-outlet-duct section of the mixing plenum and mixed with recirculated cabin air. This air mixture is then ducted aft through the floor-outlet duct. If the air temperature inside the duct becomes excessive, a sensor inside the duct causes the yellow DUCT OVERTEMP caution annunciator to illuminate. Refer to the ILLUMINATION OF "DUCT OVERTEMP" ANNUNCIATOR procedure in the EMERGENCY PROCEDURES Section of this manual for corrective action.

After passing the temperature sensor, the air passes through the cabin air control valve. This valve is controlled by the CABIN AIR control knob on the copilot's subpanel, just below and inboard of the control column. When this knob is pulled out to the stop, only a minimum amount of warm air will be permitted to pass through the valve, thereby increasing the amount of warm air available to the pilot and copilot heat outlets, and to the defroster. When this knob is pushed fully in, the valve is open and the air in the duct will be directed to the floor-outlet registers in the cabin.

RADIANT HEATING

A supplemental electric radiant heating system is available for cabin comfort. It is turned on and off by a switch in the ENVIRONMENTAL group on the copilot's left subpanel placarded RADIANT HEAT - OFF. This system can be used in conjunction with an auxiliary power unit for warming the cabin prior to starting the engines, and it can be used as supplemental heat in flight. However, it should be used in conjunction with the manual temp control mode only.

On the B200C, a radiant heater element is installed in the cargo door. It is controlled by the Cabin Temperature Mode switch and operates in all heating modes. This unit provides supplemental heat to the cabin for additonal passenger comfort.

AIR CONDITIONING SYSTEM

Cabin air conditioning is provided by a refrigerant-gas-vapor-cycle refrigeration system consisting of: a belt-driven, engine-mounted compressor, installed on the right engine accessory pad; refrigerant plumbing; an N_1 speed switch; high- and low-pressure-protection switches; a condensor coil; a condensor blower; an evaporator; a receiver-dryer; an expansion valve; and a bypass valve. The plumbing from the compressor is routed through the right-wing inboard leading edge to the fuselage. It is then routed forward to the condensor coil, receiver-dryer, expansion valve, bypass valve, and evaporator, which are all located in the nose of the airplane.

The high- and low-pressure-limit switches and the N_1 engine speed switch are provided to prevent compressor operation outside of established limitation parameters. The N_1 speed switch will prevent the flow of electric current to the compressor clutch when the engine speed is below 62% N_1. When the N_1 speed switch is open and there is a demand for air conditioning, the green AIR COND N_1 LOW advisory annunciator will illuminate. If either the high or low freon pressure limit is exceeded, an electronic system will cause the electronic circuit to trip and the reset switch light will illuminate. Reset switch and light are located in the nose gear wheel well. Depressing the reset switch one time may restore the airconditioning function. The switch is between the power source and the electrically activated compressor clutch and condensor blower. The interruption of power will cause the compressor clutch and condenser blower to disengage. In the event the airconditioner does not work after the reset switch is depressed one time, the system should be thoroughly checked. The compressor-clutch circuit breaker is located in the D.C. power distribution panel inside the lower forward equipment bay.

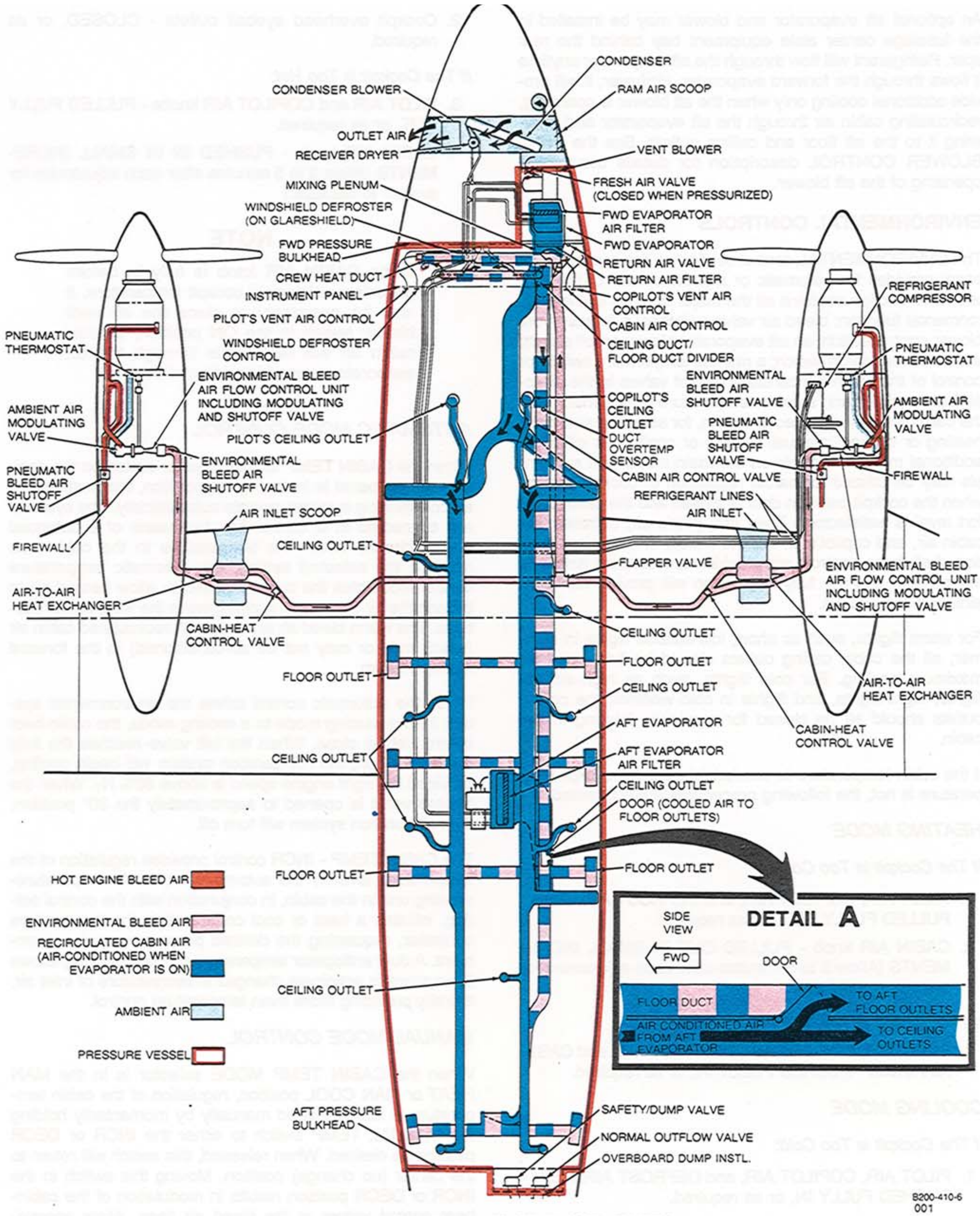
The forward evaporator utilizes a solenoid-operated hot-gas-bypass valve to prevent icing. A 33°F thermal switch on the forward evaporator controls the valve solenoid.

The forward vent blower blows recirculated cabin air (plus outside ambient air if the cabin is unpressurized) through the forward evaporator, into the mixing plenum, and into both the floor-outlet and ceiling-outlet ducts. If the cooling mode is operating, refrigerant will be circulating through the evaporator and the air leaving it will be cool. All the air entering the ceiling-outlet duct will be cool. This air is discharged through "eyeball" outlet nozzles in the cockpit and cabin. Each nozzle is movable, so that the airstream can be directed as desired. When the nozzle is twisted, a damper opens or closes to regulate airflow volume.

Cool air will enter the floor-outlet duct, but in order to provide cabin pressurization, warm environmental bleed air will also enter the floor-outlet duct anytime either BLEED AIR valve is OPEN. Therefore, pressurized air discharged from the floor registers will always be warmer than that discharged at the ceiling outlets, no matter what temperature mode is in use.

A vane-axial blower in the nose section draws ambient air through the condensor when the cooling mode is operating. This blower shuts off when the airplane is airborne. The current limiter for the blower is located in the DC power distribution panel in the lower forward equipment bay.

The receiver-dryer and sight gage are located high in the condenser compartment. They can be viewed by removing the upper-compartment access panel, located on top of the nose section just left of the centerline.



ENVIRONMENTAL SYSTEM SCHEMATIC
(Optional Aft Evaporator System Shown)

An optional aft evaporator and blower may be installed in the fuselage center aisle equipment bay behind the rear spar. Refrigerant will flow through the aft evaporator anytime it flows through the forward evaporator. However, it will provide additional cooling only when the aft blower is operating, recirculating cabin air through the aft evaporator and delivering it to the aft floor and ceiling outlets. See the VENT BLOWER CONTROL description for details concerning operating of the aft blower.

ENVIRONMENTAL CONTROLS

The ENVIRONMENTAL control section on the copilot's subpanel provides for automatic or manual control of the system. This section contains all the major controls of the environmental function: bleed air valve switches; a forward vent blower control switch; an aft evaporator blower on/off switch; a radiant heat on/off switch; a manual temperature switch for control of the cabin temperature control valves in the air-to-air heat exchangers; a cabin-temperature-level control; and the cabin temp mode selector switch, for selecting automatic heating or cooling, manual heating or cooling, or off. Four additional manual controls on the main instrument subpanels may be utilized for partial regulation of cockpit comfort when the cockpit partition door is closed and the cabin comfort level is satisfactory. 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.

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

If The Cockpit is Too Hot:

3. PILOT AIR, COPILOT AIR, DEFROST AIR, and CABIN AIR knobs - PUSHED FULLY IN, or as required.

COOLING MODE

If The Cockpit is Too Cold:

1. PILOT AIR, COPILOT AIR, and DEFROST AIR knobs - PUSHED FULLY IN, or as required.
2. Cockpit overhead eyeball outlets - CLOSED, or as required.

If The Cockpit is Too Hot:

3. PILOT AIR and COPILOT AIR knobs - PULLED FULLY OUT, or as required.
4. CABIN AIR knob - PUSHED IN IN SMALL INCREMENTS (Allow 3 to 5 minutes after each adjustment for system to stabilize.)

NOTE

If the CABIN AIR knob is fully in before obtaining satisfactory cockpit temperature, it may be necessary to place the aft vent blower switch in the ON position, so that cabin air will recirculate through the aft evaporator to provide additional cooling.

AUTOMATIC MODE CONTROL

When the CABIN TEMP MODE selector switch on the copilot's left subpanel is in the AUTO position, the heating and air conditioning systems operate automatically. The systems are connected to a control box by means of a balanced bridge circuit. When the temperature in the cabin has reached the selected setting, the automatic temperature control modulates the bypass valves to allow heated air to bypass the air-to-air heat exchangers in the wing center sections. The warm bleed air is mixed with recirculated cabin air (which may or may not be air-conditioned) in the forward mixing plenum.

When the automatic control drives the environmental system from a heating mode to a cooling mode, the cabin-heat control valves close. When the left valve reaches the fully closed position, the refrigeration system will begin cooling, provided the right engine speed is above 60% N_1. When the bypass valve is opened to approximately the 30° position, the refrigeration system will turn off.

The CABIN TEMP - INCR control provides regulation of the temperature level in the automatic mode. A temperature-sensing unit in the cabin, in conjunction with the control setting, initiates a heat or cool command to the temperature controller, requesting the desired pressure-vessel environment. A duct anticipator temperature probe (duct stat) allows the system to anticipate changes in temperature of inlet air, thereby providing more even temperature control.

MANUAL MODE CONTROL

When the CABIN TEMP MODE selector is in the MAN HEAT or MAN COOL position, regulation of the cabin temperature is accomplished manually by momentarily holding the MANUAL TEMP switch to either the INCR or DECR position as desired. When released, this switch will return to the center (no change) position. Moving this switch to the INCR or DECR position results in modulation of the cabin-heat control valves in the bleed air lines. Allow approximately 30 seconds per valve (1 minute total time) for the valves to move to the fully open or fully closed position. Only one valve at a time moves. Movement of these valves varies the amount of bleed air routed through the air-to-air heat exchanger. Consequently, the temperature of the incoming bleed air will vary. This bleed air mixes with recirculated cabin air (which will be air-conditioned if the refrigeration system is operating) in the mixing plenum, and is then ducted to the floor registers. As a result, the cabin temperature will vary according to the position of the cabin-heat control valves, whether or not the air conditioner is operating.

When the CABIN TEMP MODE selector is in the MAN COOL position, the air conditioner system will operate, provided the speed of the right engine is above 60% N_1.

NOTE

The air conditioner compressor will not operate unless the cabin-heat control valves are closed. To ensure that the valves are closed, hold the MANUAL TEMPerature switch in the DECRease position for one minute.

BLEED AIR CONTROL

Bleed air entering the cabin is controlled by the switches placarded BLEED AIR VALVE - LEFT - OPEN - RIGHT - ENVIR OFF - INSTR & ENVIR OFF. When the switch is in the OPEN position, the environmental flow control unit and the pneumatic instrument air valve are open. When the switch is in the ENVIR OFF position, the environmental flow control unit is closed and the pneumatic instrument air valve is open; in the INSTR & ENVIR OFF position, both are closed. For maximum cooling on the ground, turn the bleed air valve switches to the ENVIR OFF position.

VENT BLOWER CONTROL

The forward vent blower is controlled by a switch in the ENVIRONMENTAL group placarded VENT BLOWER - HI - LO - AUTO. When this switch is in the AUTO position, the forward vent blower will operate at low speed if the CABIN TEMP MODE selector switch is in any position other than OFF (i.e., MANual COOL, MANual HEAT, or AUTOmatic).

When the VENT BLOWER switch is in the AUTO position and the CABIN TEMP MODE selector switch is in the OFF position, the blower will not operate. Anytime the VENT BLOWER switch is in the LO position, the forward vent blower will operate at low speed, even if the CABIN TEMP MODE selector switch is OFF. Anytime the VENT BLOWER switch is in the HI position, the forward vent blower will operate at high speed, regardless of the position of the CABIN TEMP MODE selector switch (i.e., MAN COOL, MAN HEAT, OFF, or AUTO).

If the optional aft evaporator unit is installed in the air conditioning system, an aft blower is also installed under the floor.

The aft blower draws in cabin air, blows it across the aft evaporator, and to the aft floor and ceiling outlets. This blower operates at high speed only. It is controlled by a switch placarded AFT BLOWER - OFF, located in the ENVIRONMENTAL group on the copilot's subpanel. The blower is independent of any other control. The aft blower is intended for use only when maximum cabin cooling (air conditioning) is desired. If the blower should be turned on during a heating mode of operation, the door between the aft-blower duct and the warm-air (floor-outlet) duct will open. This will stop the flow of heated air to the aft floor registers, and deliver recirculated cabin air (which is not cooled, since refrigerant is not flowing through the aft evaporator) to the aft floor registers and ceiling outlets.

Both blower circuit breakers are located in the DC power distribution panel in the lower forward equipment bay.

OXYGEN SYSTEM

The Super King Air B200/B200C oxygen system is based on an adequate oxygen flow for a pressure altitude of 35,000 feet. The masks and Oxygen Duration Chart, Section IV, NORMAL PROCEDURES, are based on a flow rate of 3.9 Liters Per Minute (LPM-NTPD). The only exception is the diluter-demand crew mask. For oxygen duration computation, each diluter-demand mask being used is counted as two masks at 3.9 LPM-NTPD. At cabin altitudes above 20,000 feet, SELECT 100% MODE.

A push/pull handle (PULL ON - SYStem READY), located aft of the overhead light control panel, is used in conjunction with the automatically deployed passenger oxygen system. This handle operates a cable which opens and closes the shut-off valve located at the oxygen supply bottle in the aft, unpressurized area of the fuselage. When this handle is pushed in, no oxygen supply is available anywhere in the airplane. It should be pulled out prior to engine starting to ensure that oxygen will be immediately available anytime it is needed. When this handle is pulled out, the primary oxygen supply line is charged with oxygen, provided the oxygen supply bottle is not empty (Check the oxygen supply pressure gage on the right subpanel and verify that sufficient oxygen is available for the flight). The primary oxygen supply line delivers oxygen to the two crew oxygen outlets in the cockpit, to the first aid oxygen outlet in the toilet area, and to the passenger oxygen system shutoff valve.

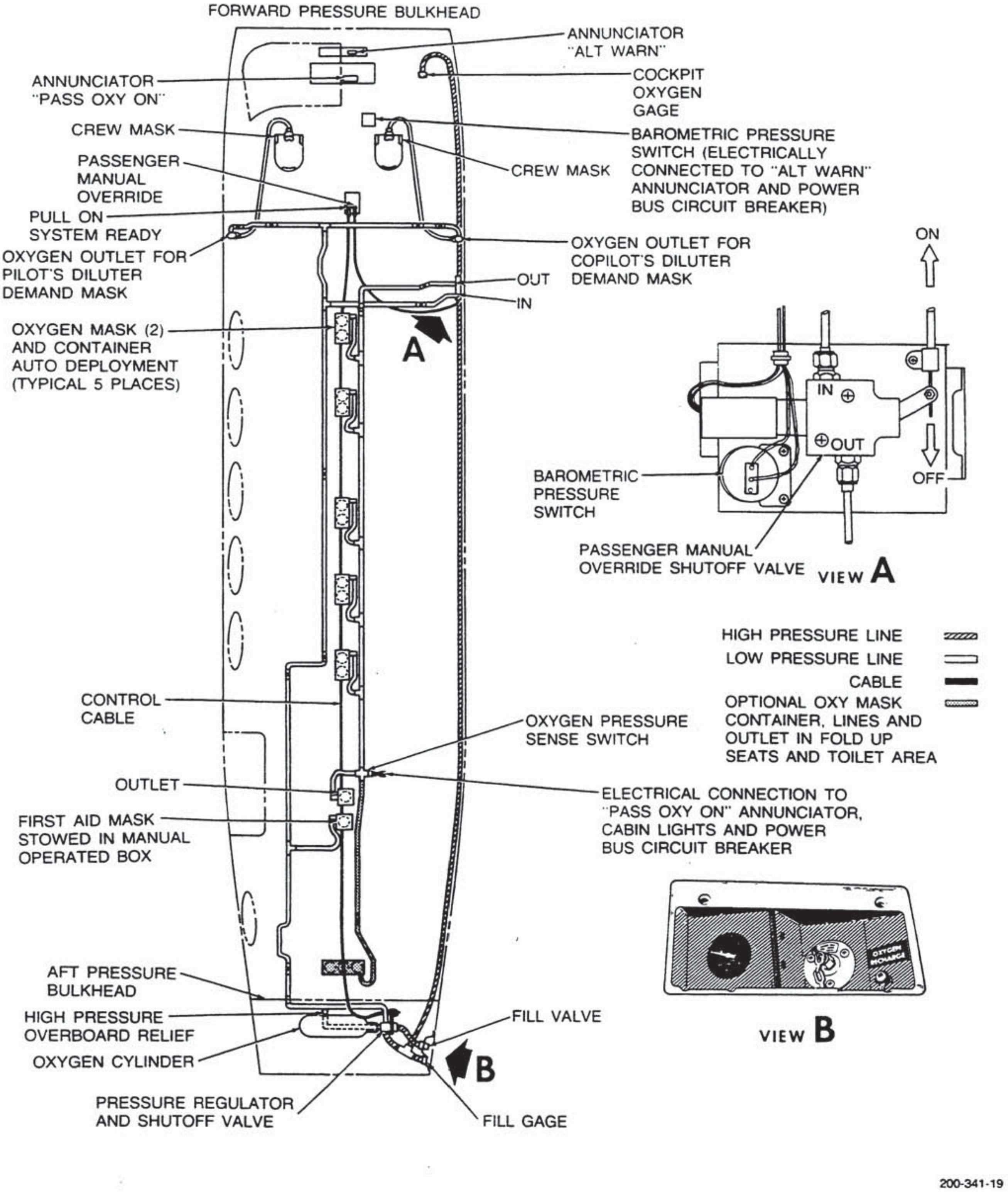
The crew is provided with diluter-demand, quick-donning oxygen masks. These masks hang on the aft cockpit partition behind and outboard of the pilot and copilot seats. They are held in the armed position by spring-tension clips, and can be donned immediately with one hand. The diluter-demand crew masks deliver oxygen to the user only upon inhalation. Consequently, there is no loss of oxygen when the masks are plugged in and the PULL - ON - SYStem READY handle is pulled out, even though oxygen is immediately available upon demand.

A small lever on each diluter-demand oxygen mask permits the selection of two modes of operation: NORMAL and 100%. In the NORMAL position, air from the cockpit is mixed with the oxygen supplied through the mask. This reduces the rate of depletion of the oxygen supply, and it is more comfortable to use than 100% aviators breathing oxygen. However, in the event of smoke or fumes in the cockpit, the 100% position should be used to prevent the breathing of contaminated air. For this reason, the selector lever should be left in the 100% position when the masks are not in use.

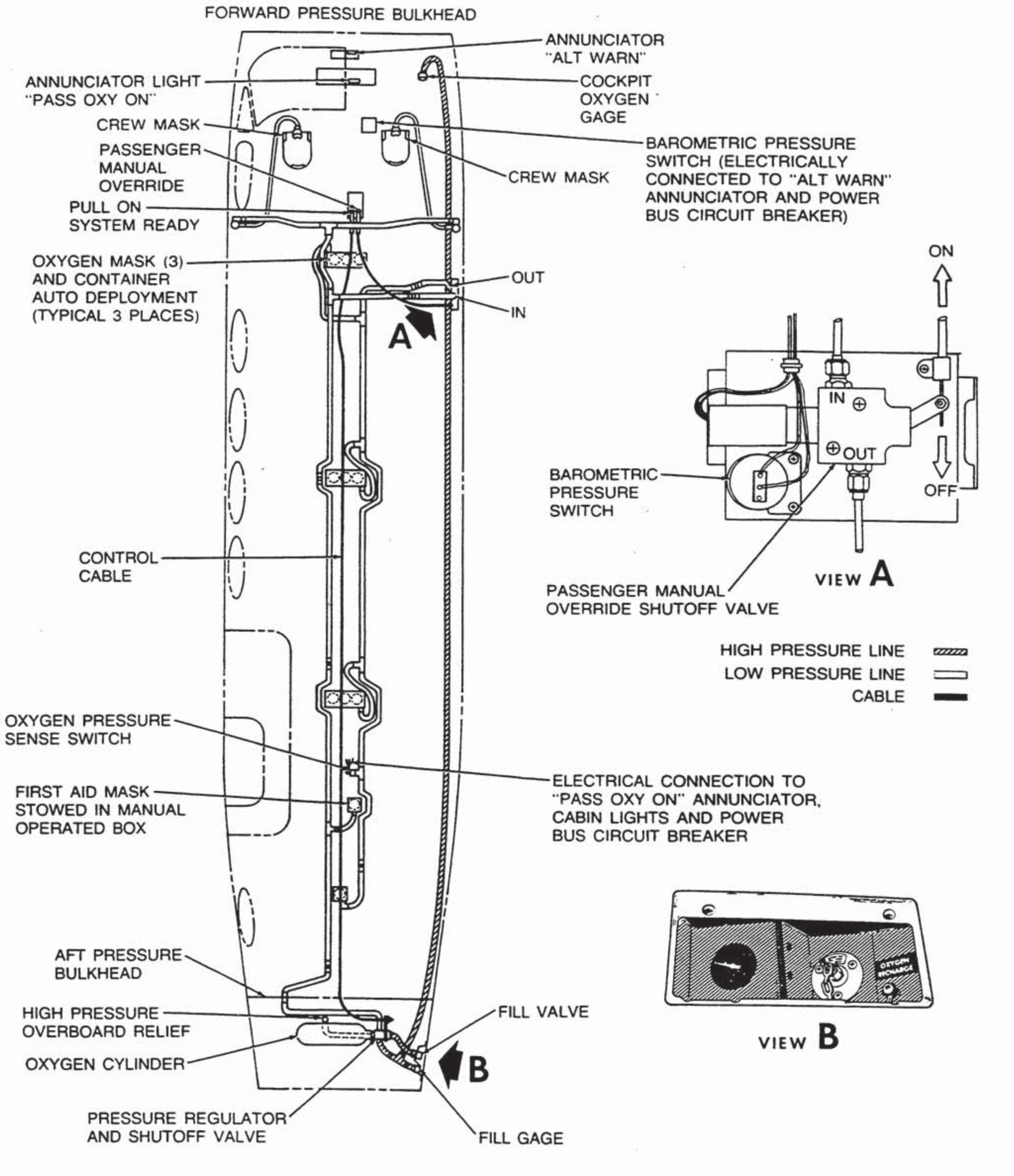
Anytime the primary oxygen supply line is charged, oxygen can be obtained from the first aid oxygen mask located in the toilet area, by manually opening the overhead access door (placarded FIRST AID OXYGEN - PULL) and opening the ON-OFF valve inside the box. A placard (NOTE: CREW SYStem MUST BE ON) reminds the user that the PULL ON - SYStem READY handle in the cockpit must be pulled out before oxygen will flow from the first aid oxygen mask.

The passenger oxygen system is of the constant flow type. Anytime the cabin pressure altitude exceeds approximately 12,500 feet, a barometric-pressure switch automatically energizes a solenoid which opens the passenger oxygen system shut-off valve. The pilot can open the valve manually anytime by pulling out the PASSENGER MANUAL OverRIDE handle, located aft of the overhead light control panel. Once the passenger oxygen system shut-off valve has been opened (either automatically or manually), oxygen will flow into the passenger oxygen supply line, if the primary oxygen system line has been charged (i.e., if the oxygen supply bottle contains oxygen and the PULL ON - SYStem READY handle in the cockpit is pulled out). When oxygen flows into the passenger oxygen system supply line, a pressure-sensitive switch in the line closes a circuit to illuminate the green PASS OXYGEN ON annunciator on the cautionary/advisory annunciator panel. This switch will also cause the cabin lights (all fluorescent lights, the foyer light and the center baggage compartment light) to illuminate in the full bright mode, regardless of the position of the interior lights switch placarded CABIN LIGHTS - START BRIGHT - DIM - OFF located on the copilot's left subpanel.

The pressure of the oxygen in the passenger oxygen system supply line then automatically extends a plunger against each of the passenger oxygen mask dispenser doors, forcing the doors open. The oxygen masks then drop down about 9 inches below the dispensers. The lanyard valve pin at the top of the oxygen mask hose must be pulled out in order for oxygen to flow from the mask. The pin is connected to the oxygen mask via a flexible cord; when the oxygen mask is pulled down for use, the cord pulls the pin out of the lanyard valve. The lanyard valve pin must be manually reinserted into the valve in order to stop the flow of oxygen when the mask is no longer needed. The passenger oxygen can be shut off and the remaining oxygen isolated to the crew and first aid outlets by pulling the OXYGEN CONTROL circuit breaker in the ENVIRONMENTAL group on the right side panel, providing the PASSENGER MANUAL O'RIDE handle is pushed in to the OFF position.



B200 OXYGEN SYSTEM SCHEMATIC



B200C OXYGEN SYSTEM SCHEMATIC

200-341-18

PITOT AND STATIC SYSTEM

The pitot and static system provides a source of impact air and static air for operation of the flight instruments. A heated pitot mast is located to each side of the lower portion of the nose. Tubing from the left pitot mast is connected to the pilot's airspeed indicator, and tubing from the right pitot mast is connected to the copilot's airspeed indicator.

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 two static air ports - one on each side of the fuselage.

An alternate static air line is also provided for the pilot's flight instruments. In the event of a failure of the pilot's normal static air source (e.g., if ice accumulations should obstruct the static air ports), the alternate source should be selected by lifting the spring-clip retainer off the PILOT'S STATIC AIR SOURCE valve handle, located on the right side panel, and moving the handle aft to the ALTERNATE position. This will connect the alternate static air line to the pilot's flight instruments. The alternate line obtains static air just aft of the rear pressure bulkhead, from inside the unpressurized area of the fuselage.

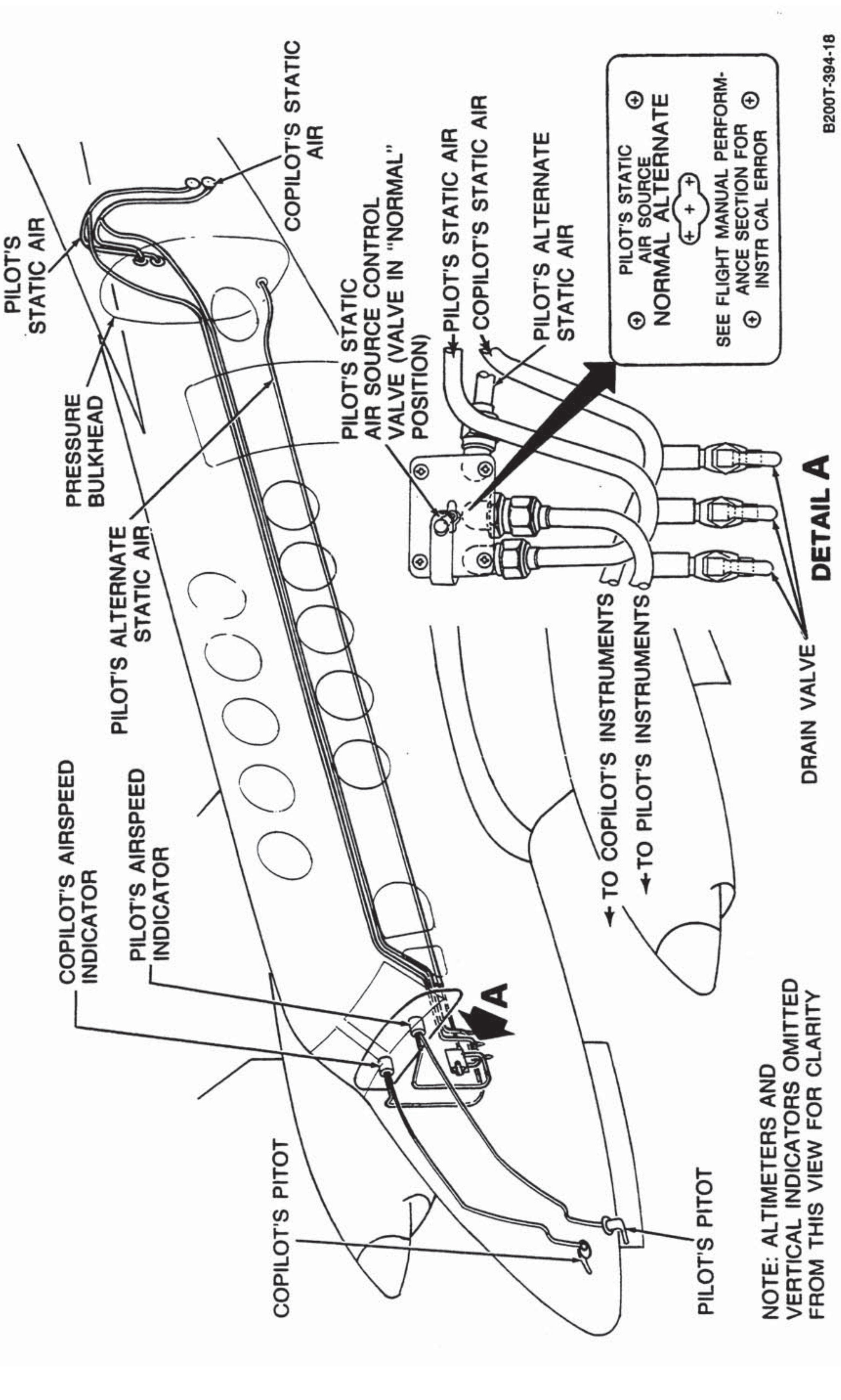
The pilot's altimeter, vertical speed indicator, and airspeed indicator are connected to the pilot's static air source. When the system is switched to the pilot's alternate air source, the pilot's altimeter and vertical speed indicator are affected as well as the pilot's airspeed indicator. The copilot's airspeed indicator, altimeter, and vertical speed indicator are all on the copilot's static air source and cannot be switched to alternate source. See SCHEMATIC DIAGRAM of PITOT AND STATIC SYSTEM.

WARNING

The pilot's 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 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 STATIC AIR SOURCE valve handle is held in the forward (NORMAL) position by the spring-clip retainer.

Three petcocks are provided to facilitate draining moisture from the static air lines. They are located behind an access cover below the circuit breakers on the right side panel. The drain valves should be opened to release any trapped moisture at each 150-hour inspection, and after exposure to visible moisture on the ground. They must be closed after draining.



PITOT AND STATIC SYSTEM SCHEMATIC

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 surface deice system and vacuum source. Vacuum for the flight instruments 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 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. Heat is also dissipated in the air-to-air heat exchanger in the center wing section. The bleed air lines are accompanied in close proximity by plastic tubing from the engines to the cabin. One end of the tubing is plugged off; the other end is connected to a bleed 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 copilot's floor in the fuselage, will close, causing a circuit to be completed to the respective BL AIR FAIL light 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 subpanel in the INSTR & ENVIR - OFF position.

AUTOMATIC DEVICES IN THE CONTROL SYSTEM

YAW DAMP

A yaw damp system is provided to aid the pilot in maintaining directional control, and to increase ride comfort. The system may be used at any altitude, and is required for flight above 17,000 feet. It should be deactivated for takeoff and landing.

If the system is equipped with an autopilot, the yaw damp system will be a part of the autopilot. Operating instructions for this system will be contained in the appropriate Airplane Flight Manual Supplement.

If an autopilot is not installed in the airplane, yaw damping is provided by an independent yaw damp system. The components include a yaw sensor, amplifier, and control valve. Regulated air pressure from the control valve is directed to the same pneumatic servos used for the rudder boost system. The system (on airplanes without an autopilot) is controlled by a YAW DAMP switch adjacent to the RUDDER BOOST switch on the pedestal. In the event the YAW DAMP switch is inadvertently left ON during takeoff or landing, the circuit for the yaw damping system will be interrupted by the left landing gear safety switch while the airplane is on the ground, rendering it inoperative.

STALL WARNING SYSTEM

The stall warning system consists of a transducer, a lift computer, a warning horn, and a test switch. 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 output of the transducer activates a stall warning horn.

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

ICE PROTECTION SYSTEMS

WINDSHIELD HEAT

Windshield heat switches are located on the pilot's subpanel (inboard) and are placarded ICE - WSHLD ANTI-ICE - NORMAL - OFF - HI - PILOT - COPILOT.

Two levels of heat are provided. When the 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.

Controllers with temperature-sensing units provide for proper heat at the windshield surfaces. Five-ampere circuit breakers, located on a panel on the forward pressure bulkhead, protect the control circuits. The power circuit of each system is protected by a 50-ampere circuit breaker located in the power distribution panel under the floor forward of the main spar.

NOTE

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

PROPELLER ELECTRIC DEICE SYSTEM

PRIOR TO BB-829, AND PRIOR TO BL-37:

The propeller electric deice system includes: an electrically heated boot with two elements, (inner and outer) for each propeller blade, brush assemblies, slip rings, an ammeter, a timer for automatic operation, and a circuit for manual control for backup.

A 20-ampere circuit breaker switch on the pilot's subpanel, placarded PROP - AUTO - OFF, is provided to activate the automatic system. A deice ammeter on the right subpanel registers the amount of current (14 to 18 amperes) passing through the system being used. During AUTO operation, power to the timer will be cut off if the current rises above the circuit breaker switch rating. Current flows from the timer to the brush assembly and then to the slip rings installed on the spinner backing plate. The slip rings carry the current to the deice boots on the propeller blades. Heat from the boots reduces the grip of the ice, which is then thrown off by centrifugal force aided by the air blast over the propeller surfaces. Power to the two heating elements on each blade, the inner and outer element, is cycled by the timer in the following sequence: right propeller outer elements, right propeller inner elements, left propeller outer elements, and left propeller inner elements. Loss of one heating element on one side does not mean that the entire system will be inoperative.

Proper operation can be checked by noting the correct level of current usage on the ammeter. An intermittent flicker of the needle approximately every 30 seconds indicates switching to the next group of heating elements by the timer.

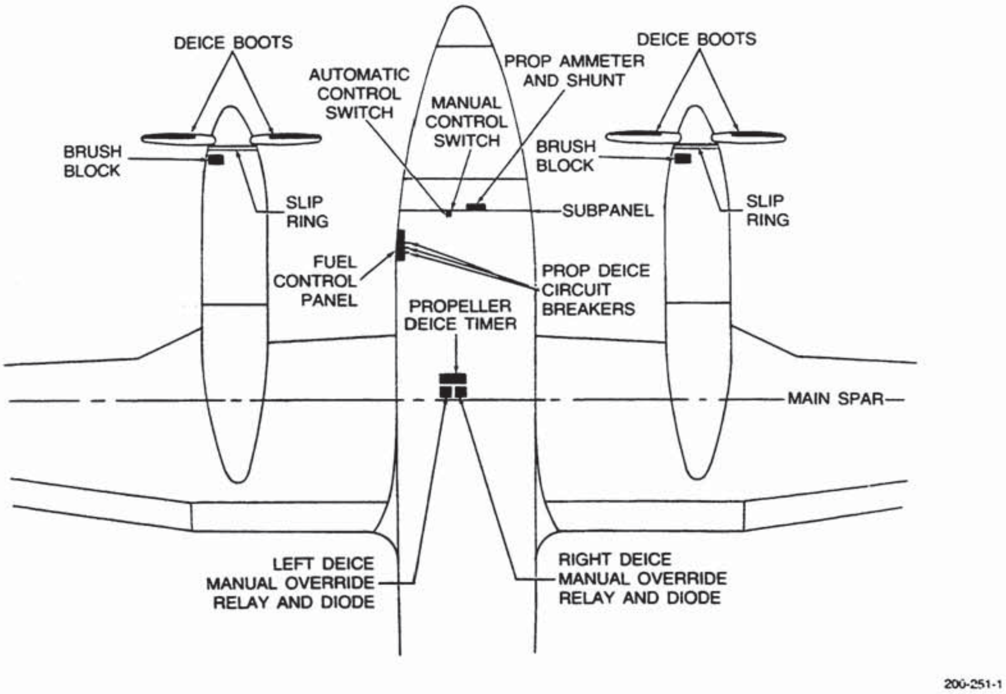
The manual prop deice system is provided as a backup to the automatic system. A control switch located on the left subpanel, placarded PROP - INNER - OUTER, controls the manual override relays. When the switch is in the OUTER position, the automatic timer is overridden and power is supplied to the outer elements of both propellers simultaneously. The switch is of the momentary type and must be held in position until the ice has been dislodged from the propeller surface. After deicing with the outer elements, the switch must be held in the INNER position to perform the same function for the inner elements of both propellers. The loadmeters will indicate approximately a .05 increase of load per meter when manual prop deice is operating. The prop deice ammeter will not indicate any load in the manual mode of operation.

BB-829 AND AFTER, BL-37 AND AFTER:

The propeller electric deice system includes: electrically heated deice boots, slip rings and brush block assemblies, a timer for automatic operation, an ammeter, three circuit breakers located on the fuel control panel for left and right propeller and control circuit protection, and two switches located on the pilot's subpanel for automatic or manual control of the system.

A circuit breaker switch located on the pilot's subpanel, placarded PROP - AUTO - OFF, is provided to activate the automatic system. Upon placing the switch to the AUTO position, the timer diverts power through the brush block and slip ring to all heating elements on one propeller. Subsequently, the timer then diverts power to all heating elements on the other propeller for the same length of time. This cycle will continue as long as the switch is in the AUTO position. The system utilizes a metal foil type single heating element energized by DC voltage. The timer switches every 90 seconds, resulting in a complete cycle in approximately 3 minutes.

A manual prop deice system is provided as a backup to the automatic system. A control switch located on the left subpanel, placarded PROP - MANUAL - OFF, controls the manual override relay. Upon placing the switch is in the MANUAL position, the automatic timer is overridden and power is then supplied to the heating elements of both propellers simultaneously. This switch is of the momentary type and must be held in position for aproximately 90 seconds to dislodge from the propeller surface. Repeat this procedure as required to avoid significant buildup of ice which will result in loss of performance, vibration, and impingement upon the fuselage. The prop deice ammeter will not indicate a load while the propeller deice system is being utilized in the manual mode. However, the loadmeters will indicate an approximate .05 increase of load per meter while the manual prop deice system is operating.



PROPELLER ELECTRIC DEICE SYSTEM SCHEMATIC

PITOT MAST HEAT

Heating elements are installed in the pitot masts located on the nose. Each heating element is controlled by an individual circuit breaker switch placarded PITOT - LEFT - RIGHT, located on the pilot's subpanel. 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 mast.

SURFACE DEICE SYSTEM

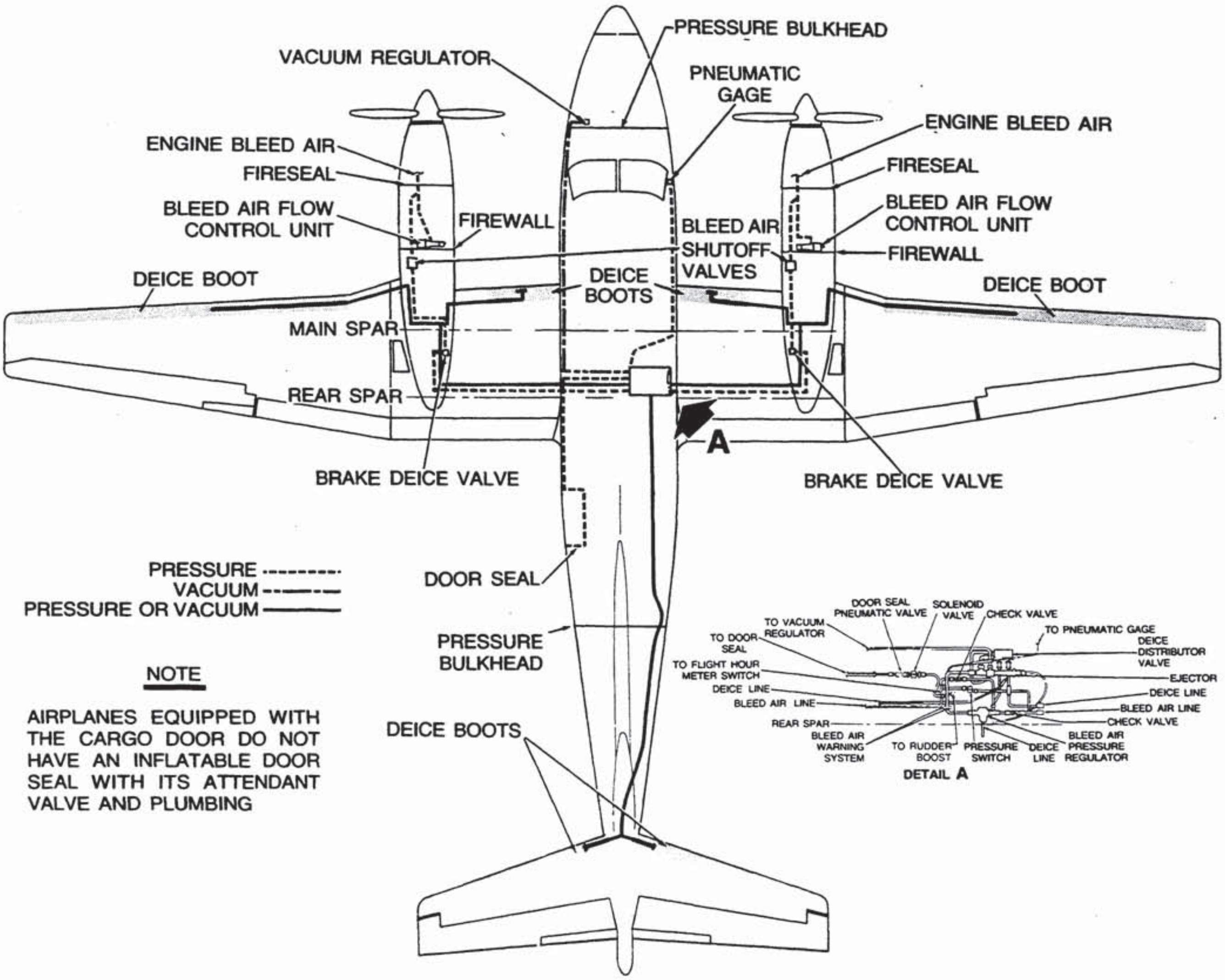
The surface deice system removes ice accumulations from the leading edges of the wings and horizontal stabilizers. Ice removal is accomplished by alternately inflating and deflating the deice boots. Pressure-regulated bleed air from the engines supplies pressure to inflate the boots. A venturi ejector, operated by bleed air, 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.

CAUTION

Operation of the surface deice system in ambient temperatures below -40°C can cause permanent damage to the deice boots.

A three-position switch in the ICE group on the pilot's subpanel identified: DEICE CYCLE - SINGLE - OFF - MANUAL, controls the deicing operation. The switch is spring-loaded to return to the OFF position from SINGLE or MANUAL. When the SINGLE position is selected, the distributor valve opens to inflate the wing boots. After an inflation period of approximately 6 seconds, an electronic timer switches the distributor to deflate the wing boots, and a 4-second inflation begins in the horizontal stabilizer boots. When these boots have inflated and deflated, the cycle is complete.

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 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.



PNEUMATIC BLEED AIR SYSTEM AND SURFACE DEICE SYSTEM SCHEMATIC

For most effective deicing operation, allow at least 1/2 inch of ice to form before attempting ice removal. Very thin ice may crack and cling to the boots instead of shedding. Subsequent cyclings of the boots will then have a tendency to build up a shell of ice outside the contour of the 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 group located on the pilot's right subpanel identified: STALL WARN. The level of heat is minimal for ground operation, but is automatically increased for flight operation through the left landing gear safety switch.

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 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 when the procedure under "APPROVED FUEL ADDITIVE" located in LIMITATIONS Section is complied with.

Each pneumatic fuel control line is protected against ice by an electrically heated jacket. Power is supplied to each fuel control air line jacket heater by two switches actuated by moving the condition levers in the pedestal out of the fuel cutoff range. Fuel control heat is automatically turned on for all flight operations.

COMFORT FEATURES

TOILET

The side facing toilet is installed in the foyer and faces the airstair door. The foyer can be closed off from the cabin by sliding the two partition-type door panels to the center of the fuselage, where they are held closed by magnetic strips. A forward facing toilet, when installed, is located in the aft cargo area and is enclosed by the cargo partition. The toilet may be either the chemical type or the electrically flushing type. In either case, the two hinged lid half-sections must be raised to gain access to the toilet. A toilet tissue dispenser is contained in a slide-out compartment on the forward side of the toilet compartment.

CAUTION

If a Monogram electrically flushing toilet is installed, the sliding knife valve should be open at all times, except when actually servicing the unit. The cabinet below the toilet must be opened in order to gain access to the knife valve actuator handle.

RELIEF TUBES

A relief tube is contained in a special tilt-out compartment at the aft side of the toilet cabinet. A relief tube may also be installed in the cockpit and stowed under the pilot or copilot chair. The hose on the cockpit relief tube is of sufficient length to permit use by either pilot or copilot.

A valve lever is located on the side of the relief tube horn. This valve lever must be depressed at all times while the relief tube is in use. Each tube drains into the atmosphere through its own special drain port, which protrudes from the bottom of the fuselage. Each drain port atomizes the discharge to keep it away from the skin of the airplane.

NOTE

The relief tubes are for use during flight only.

CABIN FEATURES

FIRE EXTINGUISHERS

An optional portable fire extinguisher may be installed on the floor on the left side of the airplane forward of the airstair entrance door, just aft of the rearmost seat. Another one may also be installed underneath the copilot's seat.

WINDSHIELD WIPERS

The dual windshield wiper installation consists of an electric motor, arm and wiper assemblies, drive shafts, and converters, all located forward of the instrument panel. The system also includes a control switch, located in the upper left corner of the overhead light control panel, and a circuit breaker located on the right circuit breaker panel. The control knob, placarded WINDSHIELD WIPER - PARK - OFF - SLOW - FAST, controls the wipers. The wipers have two speeds, one for light and one for heavy precipitation. After the control is turned to PARK to bring the wiper arms to their most inboard position, spring-loading returns the control to the OFF position.

Windshield wipers may be used during either ground or flight operations.

CAUTION

Do not operate windshield wipers on dry glass.

CARGO RESTRAINT (B200C)

Beech Aircraft Corporation offers an FAA approved cargo restraint system as Kit No. 101-5040. 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.