SECTION V FLIGHT CONTROL SYSTEMS & AVIONICS

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SECTION V FLIGHT CONTROL SYSTEMS & AVIONICS

FLIGHT CONTROL SYSTEMS

The primary flight controls (ailerons, elevator, and rudder) are mechanically operated through the control columns, control wheels, and rudder pedals. The flaps and spoilers are hydraulically actuated and electronically controlled. Airplane trim systems (pitch, roll, and yaw) are electronically controlled.

AILERON

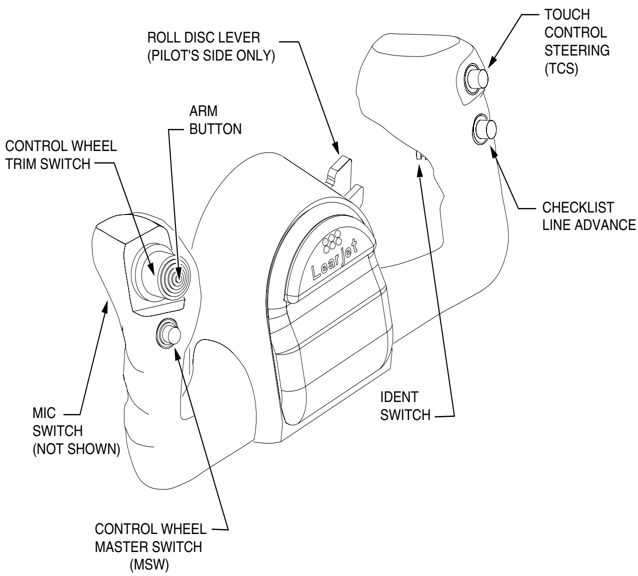
The aileron control system consists mainly of three control circuits, one in the fuselage area and one in each of the left and right wing area. In addition, a disconnect mechanism is incorporated into the pilot's control wheel which allows the disconnection of the aileron control system (in the event of a jam) and switching to spoileron system for roll control. The fuselage control circuit connects both pilot's and copilot's control wheels together, and each wing control circuit is connected to the aileron drive mechanism. The three control circuits are connected together via a common sector assembly. In normal operation, whether by an input from the autopilot or by manual input to one of the two control wheels, the two control circuits will move in unison to drive the two aileron panels. The aileron control system is considered the primary system for roll control and is interfaced with the spoileron system for roll augmentation.

ROLL DISCONNECT

If ailerons become jammed, the aileron control system can be disconnected and the spoileron system can be used for roll control. The pilot's control wheel is disconnected from the aileron cables and copilot's control wheel by the red lever labeled ROLL DISC located on the hub of the pilot's control wheel. This will also disconnect and prevent engagement of the autopilot. Safe flight can continue on spoilerons alone. For more information on roll disconnect, see Spoileron (ROLL DISCONNECT) system.

CONTROL WHEEL

Each flight station is equipped with a U-shaped control wheel. The pilot's control wheel is equipped with a disconnect assembly which employs a red lever labeled ROLL DISC located on the inboard side of the control wheel hub (Figure 5-1). Each control wheel contains the following switches: Control Wheel Trim, Control Wheel Master (MSW), MIC, IDENT, Touch Control Steering (TCS), and Checklist Line Advance.



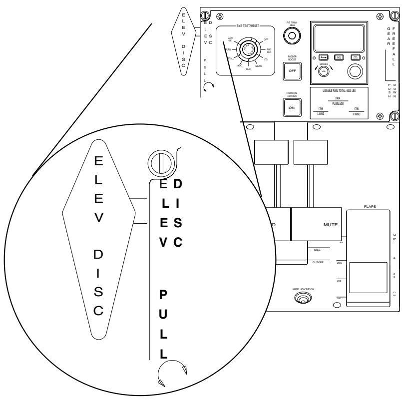
CONTROL WHEEL
Figure 5-1

ELEVATOR

Movement of the control columns is mechanically translated into elevator surface movement through levers, bellcranks, sectors, cables, and pushrods. The elevator control system consists of two parallel control circuits. The two control circuits are normally connected together via forward and aft disconnect assemblies in which either control column moves both the left and right elevator surfaces in union. A mechanical up/down spring is also used in the system to augment high and low speed trim ability of the airplane. If a jam occurs in either mechanical control circuit, an elevator disconnect feature is incorporated into the system.

ELEVATOR DISCONNECT

In the event of an elevator jam, the two cable circuits can be disconnected by pulling the red ELEV DISC T-handle (Figure 5-2) located at the left-forward edge of the pedestal. The airplane will then be controlled with the unjammed elevator. The forward and aft disconnect assemblies are dog clutch devices that are operated simultaneous by the handle being pulled. When the ELEV DISC T-handle is pulled, a cable connected to the handle shaft is pulled which disengages the forward clutch thereby disengaging the two control columns. Electrical switches sense the movement of the shaft and signal the aft disconnect to disengage.



ELEVATOR DISCONNECT
Figure 5-2

When the handle is pulled to full extension, it must be rotated 90°, either clockwise or counterclockwise, to lock it in the disconnect position. The elevator forward disconnect is a mechanical clutch mechanism located on the torque tube between the control columns. The clutch is held open when the ELEV DISC T-handle is pulled and locked in the extended position. This will disconnect and prevent engagement of the autopilot.

The elevator aft disconnect is an electro-mechanical device located in the top of the vertical stabilizer. When the ELEV DISC T-handle is pulled, a two-position linear actuator on the elevator aft disconnect assembly is energized to the extended position (disconnected position), separating operation of the two elevators. The linear actuator will remain in the extended position. When the elevator aft disconnect is actuated, the elevator disconnect sensor will send a signal to display a message on the Crew Alerting System (CAS). **Do not** reconnect. Obtain maintenance prior to next flight. Electrical power used by the elevator disconnect system is through the ELEV DISC circuit breaker located on the pilot's circuit breaker panel (FLIGHT group).

The following CAS illuminations are specific to the elevator disconnect:

CAS	Color	Description
ELEVATOR DISC	Amber	Elevator disconnect has split the elevator controls on the ground. Obtain maintenance prior to flight.
ELEVATOR DISC	White	Elevator disconnect has split the elevator controls during flight. **Do not** reconnect.

RUDDER

Directional control is provided by a dual closed-loop cable system with separate parallel paths in the engine area for rotor burst considerations. Rudder pedal movement is mechanically translated into rudder control surface movement through cables, pulleys, and bellcranks. There is an electrically driven rudder boost system to provide additional rudder control power in the event of an engine-out on takeoff.

RUDDER PEDAL ADJUSTMENT SWITCHES

The pilot's and copilot's rudder pedals are individually adjustable with a spring-loaded toggle switch to accommodate differences in crew size. The pilot operated toggle switch controls a linear actuator which provides forward and aft pedal adjustment. This toggle switch is labeled RUDDER PEDAL, and located on the lower outboard corner of the pilot's and copilot's switch panel. Each switch has three positions: FWD, Off, and AFT. Only the FWD and AFT positions are labeled. The rudder pedal adjustment is powered by 28-vdc supplied through two 3-amp circuit breakers, L RUD ADJUST and R RUD ADJUST, on the pilot's and copilot's circuit breaker panels (FLIGHTgroup).

RUDDER BOOST

The rudder boost system is provided to reduce rudder forces. Signals from force sensors in both sets of rudder pedal mechanisms are read by the ICs. The ICs then send rudder boost signals to the yaw damper servo. Rudder boost provides yaw servo torque proportional to rudder pedal force, when either the pilot's or copilot's rudder pedal force or the sum of their forces reaches 50 pounds. The rudder boost will override the yaw damper (if engaged) when this threshold is reached. When the force on the rudder pedals is released, the yaw damper will resume operation. The rudder boost system is armed when flap extension is greater than 3° and the RUD BOOST switch is selected to On. The RUD BOOST switch is located on the forward pedestal. When selected On, the switch is dark and when selected OFF, OFF will be displayed in the center of the switch. A white CAS illuminates when the switch is OFF or the system is disabled by the IC or the yaw force interface box. An amber CAS illuminates when the system is inoperative and not selected OFF.

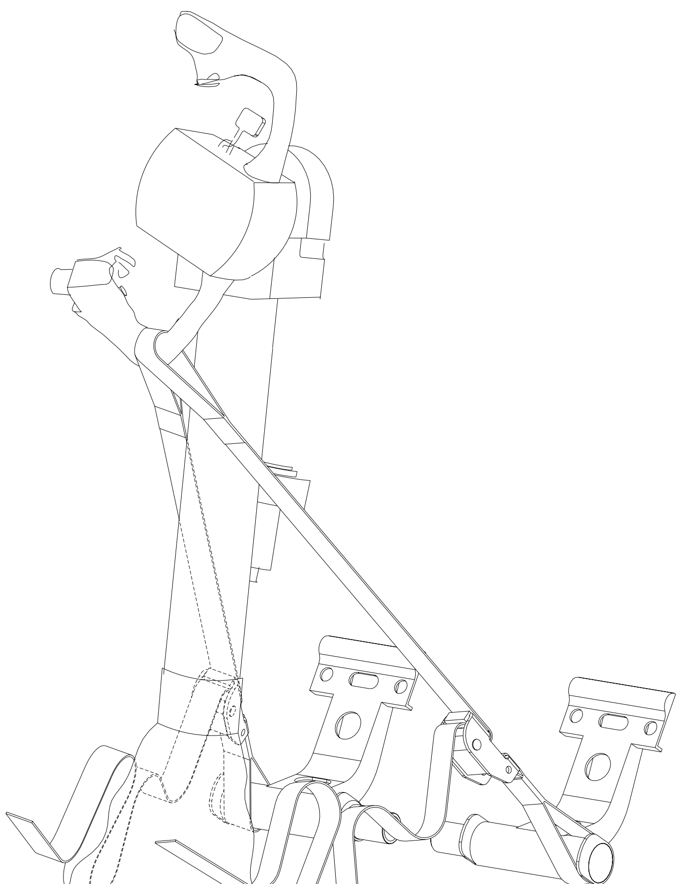
Dual, redundant power inputs are provided via the RUD FORCE circuit breaker on the right essential bus and the NOSE STEER COMPUTER circuit breaker on the left essential bus. If both these power sources should fail, the #2 IC-600 will disable rudder boost and provide appropriate annunciation.

The following CAS illuminations are specific to the rudder boost system:

CAS	Color	Description
RUD BOOST INOP	Amber	Rudder boost is inoperative and not selected OFF. Do not takeoff.
RUD BOOST INOP	White	Rudder boost is selected OFF. Do not takeoff.

CONTROLS GUST LOCK

A gust lock is provided to help prevent wind gust damage to moveable control surfaces. The gust lock is installed on the pilot's side only, with control wheel rotated counterclockwise until the bend in the handle aligns with the column, and the rudder pedals are centered. Loop straps around bottom heel, and draw both left and right pedal straps taut to seat control column against the primary stop. When installed, the gust lock secures the flight controls in the rudder centered, full aileron, and full down elevator position.



CONTROLS GUST LOCK
Figure 5-3

FLAPS

The airplane's single-slotted Fowler flaps are electronically controlled and operated by a hydraulic motor (flap power unit). Each flap panel, one on each wing, has three safe-life flap tracks and is driven by two screw jack actuators. A flexible drive shaft transfers power from the hydraulic motor to each flap actuator. The flap control lever is located on the center pedestal and is recessed to prevent inadvertent operation. The flap control lever has settings at 0° (up), 8°, 20° and 40° (down). To select a new flap position, the flap control lever is moved directly to the desired setting. Flap position is controlled by a microprocessor based controller (Flap Control Unit). The Flap Control Unit receives position command information and an arming signal from the flap control lever in the cockpit. It then provides the electrical arming and control signal to the arming solenoid valve located in the flap power unit and receives a feed-back signal from sensors mounted on the outboard actuator of each flap panel. When flaps are extended or retracted in flight, the configuration trim system automatically applies the appropriate amount of pitch trim to compensate for the pitching moment caused by flap repositioning.

The Flap Power Unit (FPU) is located under the center wing and contains the hydraulic motor, a servo control valve, an arming solenoid valve and a pressure switch. The servo valve responds to electrical signals from the flap control unit and meters hydraulic pressure to the extend or retract side of the bidirectional hydraulic motor. The arming solenoid valve must be energized open by the flap control unit before hydraulic pressure is available to the servo valve. The pressure switch, located upstream of the servo valve, monitors system pressure between the arming solenoid valve and the servo valve. If pressure is not available on flap selection, flaps will be inoperative, but if pressure is available without a selection command, the FPU will show a fault and the flaps will operate in a degraded mode, i. e. the flaps may deploy at a reduced speed when selected. These will cause the following CAS illuminations in order described above.

The following CAS illuminations are specific to the Flap System:

CAS	Color	Description
FLAPS FAIL	Amber	The flap system has failed and the flaps are inoperative.
FLAPS FAULT	Amber	The flap system is operating in a degraded mode.

Flexible drive shafts routed along the rear wing spar transmit the rotary motion of the flap power unit to the input shaft of each of the flap actuators. Two Rotary Variable Differential Transformers (RVDTs) mounted on the outboard side of each outboard flap actuator provide position information to the flap control unit and the flap position indicating unit. The flap actuators incorporate a screw jack and are attached to the rear spar. These actuators convert the rotary input motion into linear output motion through these screw jacks thus driving the flaps. Each actuator has overtravel end stops. Uncommanded retraction due to airloads, vibration, etc., is prevented by the screw jack design of the flap actuators. The flap control system operates on 28-vdc supplied through a 3- amp FLAP CTRL breaker located on the copilot's circuit breaker panel (FLIGHT group).

FLAP CONTROL LEVER

The FLAP control lever will operate in one of four positions (UP, 8°, 20°, and DN) with detents at the 8° and 20° positions. When retracting flaps, there is a gate at the 8° position; therefore, the lever must be pulled out slightly when raising the flaps above 8°. The lever is attached to dual RVDTs co-located with a flap lever detent switch within the throttle quadrant. These dual RVDTs transmit the selected position to a flap control unit. Moving the lever between positions actuates the flap lever detent switch and energizes a 75-second timing circuit within the flap control unit. This circuit allows the arming solenoid valve within the flap power unit to energize open for 75 seconds and then de-energizes. Normal flap extension from 0° to 40° will not exceed 10 seconds with engine-driven hydraulic pumps operating. However, this time will extend up to 60 seconds when using HYD XFLOW while lowering flaps from 0° to 20° in flight.

FLAP POSITION INDICATION

The flap position indicating unit has two separate and independent channels. Channel 1 provides left side equipment and Channel 2 provides right side equipment. Both channels are housed in a common chassis. Flap position is shown full time in a digital display on the Engine Indicating and Crew Alerting System (EICAS). The EICAS display is framed with a white box when the flaps are not in the selected position in flight, or on the ground with flaps not set for takeoff. The EICAS display turns red if power is advanced for takeoff and flaps are not properly set. The display turns amber if there is a fault or failure in the flap system. Flap selection and position are also displayed on the right side of the FLT (flight) system schematic page. The FLT system schematic display can be displayed on the EICAS or Multi-Function Display (MFD).

Selected flap position is indicated by a horizontal magenta line across the vertical scale. Actual left and right flap position is indicated by flap position pointers on each side of the vertical scale. When flaps have moved to their selected position, the pointers will overlay the magenta line. Flap position pointers turn red on the ground when power is advanced for takeoff and flaps are not properly set. Pointers turn amber when there is a fault or failure in the flap system. A digital indication of flap position is provided on the backup engine/systems page of the Radio Management Unit (RMU).

SPOILER SYSTEMS

Spoilers, one on the upper surface of each wing forward of the flaps, are provided for deceleration. The spoilers are electrically controlled and hydraulically operated. The spoilers are extended symmetrically for use as spoilers/speed brakes or asymmetrically for aileron augmentation. Each spoiler is hinged at four points and is extended or retracted with a single hydraulic actuator. The spoiler control lever, located on the left side of the throttle quadrant, is linked to two RVDTs. There are three labeled settings for the spoiler lever that correspond to detent positions: RET (retract), ARM (autospoilers), and EXT (full extension) - approximately 60° at slower airspeeds. The range between the ARM and EXT detents allow for variable spoiler positions in flight. There are also two unmarked detent positions between ARM and EXT which correspond to intermediate spoiler extension positions of approximately 15° and 30°. At high airspeeds the actuators cannot extend the spoilers fully; therefore, spoileron computer commands to the actuator servos are limited by airspeed inputs from the Air Data Computers (ADCs). At speeds below 175 knots, spoilers will extend to 60° when the spoiler control lever is placed to EXT; however, at higher speeds full extension is not possible.

NORMAL SPOILER MODE

The spoilers can be extended symmetrically on the ground or in flight by moving the spoiler lever aft of the ARM position. Placing the lever to any position aft of ARM while on the ground will cause full extension (60°) of the spoilers. Spoiler extension on the ground requires approximately 1 second and in flight, approximately 5 to 7 seconds. When the spoiler control lever is placed aft of the ARM position, the RVDTs will signal the spoileron computer. The computer, in turn, energizes torque motors on the servo valves to meter hydraulic pressure to the extend side of the actuators. The computer receives spoiler extension feedback from the RVDTs attached to the spoiler surfaces, and neutralizes the servo valves when the spoilers reach their selected position. In flight, the amount of spoiler extension will depend on spoiler control lever position and airspeed.

AUTOSPOILERS

Autospoiler mode is used to automatically extend spoilers on landing or in an aborted takeoff. When the SPOILER lever is set to ARM, the system will arm and CAS will illuminate. This will automatically extend spoilers when the main gear weight-on-wheels switch circuits indicate an "on ground" condition, thrust levers are in the IDLE position and the airplane has attained 60 knots ground speed. This mode fully extends spoilers at maximum rate (one second or less) when spoiler control lever is in the ARM position and autospoiler deploy logic is met. An autospoiler system is installed to automatically extend both spoilers in order to spoil lift after landing or during an aborted takeoff.

The following CAS illumination is specific to the autospoiler system:

CAS	Color	Description
AUTOSPLR ARMED	White	Autospoilers have been armed.

The main gear weight-on-wheels switch circuits are electronically latched in the "on ground" state once the initial weight-on-wheels signal is received. This prevents inadvertent spoiler retraction in the event the airplane should bounce during the ground roll. If either thrust lever is moved above IDLE while autospoilers are extended, the spoilers will immediately retract. Flap position has no effect on autospoiler operation and autospoilers are not operational when EXT or RET is selected.

The spoileron computer receives power from the L ESS BUS for operation and the spoiler indicating system receives power from the R ESS BUS. The circuits are protected by "SPLR CTRL" circuit breaker on the pilot's circuit breaker panel (FLIGHT group) and the "SPLR IND" circuit breaker on the copilot's circuit breaker panel (FLIGHT group).

When spoilers are extended or retracted in flight, depending on the mach number, the configuration trim system automatically applies the appropriate amount of pitch trim to compensate for the pitching moment caused by spoiler repositioning.

SPOILERON OPERATION

Spoilerons operate automatically on the ground and in flight to augment the ailerons whenever either control wheel is turned more than 5°. Rotation of either control wheel provides a roll input to the spoileron computer via dual RVDTs inside of the pilot's control wheel. The appropriate spoiler, left or right, extends to the commanded angle for the current conditions (Mach number, airspeed, AP engage and flap setting) while the other spoiler is commanded stowed. When in the mixed spoiler and spoileron mode, the spoiler command derived from the spoiler lever and spoileron command derived from the control wheel are added to form a composite position command for each spoiler panel. The spoiler command provides a bias position command common to both panels while the control wheel RVDTs generate a differential command. The control wheel inputs command the angular displacement that exists between the two spoilers regardless of the amount of spoilers command. Spoileron commands have priority over spoiler commands.

If the spoilers are extended, and the control wheel is turned right, the computer mix logic retracts the left spoiler first to give differential necessary for the roll commanded. If that is not enough differential for the roll commanded, the computer then extends the right spoiler as required.

SPOILERON (ROLL DISCONNECT MODE OF OPERATION)

Spoilerons provide automatic roll augmentation and backup roll control. The spoilerons are electrically controlled and hydraulically actuated. Artificial friction is introduced into the pilot control wheel upon disconnection from the mechanical aileron system to provide pilot feel and to preclude the control wheel from free-wheeling. If ailerons become jammed, the pilot's control wheel can be disconnected from the aileron control cables and the copilot's control wheel. Roll disconnect is activated with a red lever labeled (ROLL DISC) located on the hub of the pilot's control wheel. In addition to mechanically disconnecting the pilot's control wheel from the ailerons, activation of the ROLL DISC lever trips two disconnect switches within the control wheel hub. When the roll disconnect mode is activated within the spoileron computer, it outputs a signal for a CAS message to illuminate. When roll disconnect mode is activated, the autopilot will disengage.

The following CAS illuminations are specific to the spoileron computer:

CAS	Color	Description
ROLL DISC	Amber	Roll disconnect has occurred on the ground.
ROLL DISC	White	Roll disconnect has occurred in flight.

The roll disconnect mode provides roll control through RVDT signals from the pilot's control wheel to the spoileron computer. This mode is much the same as the normal spoileron mode but has a different gain curve relating to control wheel input and panel deflection begins at 1° movement instead of 5°. Spoileron operation is full time. Anytime either control wheel is turned more than 5°, there is a differential displacement of the spoiler surfaces to augment roll control. Spoilers can be operated in conjunction with the roll disconnect mode the same as they are with normal spoileron mode. The roll disconnect mode may be deselected in flight by returning the ROLL DISC lever to its normal position.

SPOILER INDICATIONS

Spoiler extension is indicated at the base of either the SUMRY or FLT system schematic page on either EICAS or MFD. On the FLT system schematic display, spoiler extension is presented as a digital display and as a vertical analog scale with dual points (one for each spoiler). The digital displays on the SUMRY and FLT pages only show spoiler extension commanded by autospoilers or with the spoiler lever. They do not reflect differential extension resulting from operation in spoileron mode. When the airplane is on the ground with spoilers extended, a white box will overlay the digital spoiler display. If power is advanced for takeoff with either or both spoilers extended, this digital display and box will turn red along with the pointers on the analog scale. In addition, a red CAS and "CONFIGURATION" voice message will activate. Spoilers should not be extended at the same time flaps are extended while in flight except as specified in the Airplane Flight Manual, or the following CAS message will be posted.

The following CAS illuminations are specific to the spoilers:

CAS	Color	Description
SPOILERS EXT	Red	The spoilers have moved from the stowed position, with aircraft on the ground, and either thrust lever is advanced to MCR or above.
SPOILERS EXT	Amber	The airplane is in flight and spoilers are extended with flaps extended more than 3°.
SPOILERS EXT	White	Spoilers are not fully retracted. Spoileron extension will not activate this CAS (flight and ground).

The analog scale and pointers are real time showing actual spoiler position for all conditions. When spoilers are extended as result of spoileron operation, pointers will indicate their differential on the analog scale. Digital spoiler indicators and analog scale pointers will turn amber when flaps are extended 3° or more with spoilers extended.

SPOILER MONITOR SYSTEM

The spoileron computer contains a monitor system to prevent electrical or mechanical faults from causing uncommanded extension or retraction of the spoilers. The spoileron computer uses electrical power from the L ESS BUS for operation and the spoiler indicating system uses power from the R ESS BUS. The circuits are protected by the SPLR CTRL and SPLR IND circuit breaker, respectively, located in the FLIGHT group on the pilot's and copilot's circuit breaker panels. If power to the spoileron computer is lost through the SPLR CTRL or SPLR IND circuit breaker, the spoilers will retract and be inoperative in all modes. The spoileron computer performs a self-test (BIT) at power-up. A test failure will trip the spoileron monitor. If the monitor detects a self-test failure or a fault during normal operation, hydraulic pressure is removed from the system by closing the spoiler shutoff valve. A hydraulic return is provided to blow the spoilers closed. During normal operation, the shutoff valve is held open by an electrical solenoid. A power failure will cause this valve to close. If the monitor does not stow the spoilers, the crew will initiate the stow with either Control Wheel Master Switch (MSW). When either MSW is held depressed, the spoiler shutoff valve is depowered closed and the spoilers will blow down, however, they may not fully retract.

A system malfunction will cause the spoileron monitor to trip and an amber CAS display. If the malfunction clears, the system may be reset using the "SPLRN RESET" position on the system test knob. If the monitor detects a jammed spoiler, the spoileron computer continues to operate using the spoiler that is not jammed and it applies a full retract input to the effected actuator for 5 to 7 seconds. This will also illuminate on the CAS.

The following CAS illuminations are specific to the spoileron monitor:

CAS	Color	Description
SPOILERS FAIL	Amber	A failure in the spoiler system is detected.
SPOILER JAM	Amber	The associated (L or R) spoiler is jammed.

PITCH TRIM

Pitch trim is provided by a moveable horizontal stabilizer. Operational structural redundancy has been incorporated by using primary and secondary sections that are independent. Primary and secondary each have electrically and mechanically independent motors (separated for rotor burst considerations), gear trains, and control inputs. Position sensors in each section of the actuator, geared directly off of the main drive screw, are monitored by both IC-600s. The computers compare the primary position sensors to the secondary position sensors in the actuator to annunciate to the pilot when the display position may not be accurate. The secondary section structure, construction, and operation is the same as the primary and both sections drive a common screwjack-type actuator to move the leading edge of the horizontal stabilizer up or down. The primary motor is actuated by manual primary pitch trim (control wheel trim switch), configuration trim, and Mach trim systems. The secondary motor is provided as a backup for primary trim and is operated by the secondary pitch trim and the autopilot.

The following CAS illuminations are specific to the pitch trim system:

CAS	Color	Description
PIT TRIM MISCMP	Red	Miscompare between the primary and secondary pitch trim on the ground and either thrust lever is advanced to MCR or above.
PIT TRIM MISCMP	White	Miscompare between the primary and secondary pitch trim in flight.

PITCH TRIM SELECTOR SWITCH

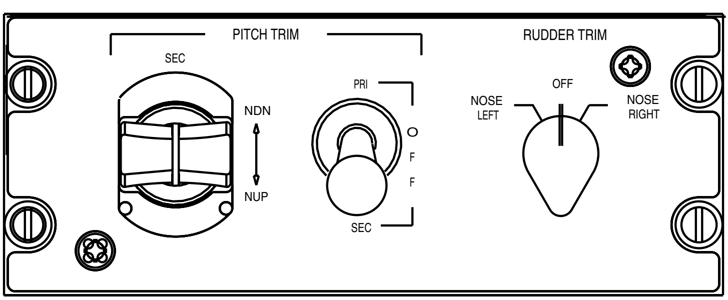
This switch, as shown in Figure 5-4, is located on the trim switch panel (pedestal) and is used to select which trim system will be used. The PRI position enables the primary trim switches in the control wheels, while the SEC position will enable the secondary trim switches in the panel. Selecting OFF or SEC will disable all #1 IC-600 trim functions. When set to OFF position, the power and ground circuits for the motor command functions in the actuator control box are disconnected and a CAS is posted.

The following CAS illumination is specific to the pitch trim system:

CAS	Color	Description
PITCH TRIM OFF	White	Pitch trim is selected to OFF.

PRIMARY PITCH TRIM

Each control wheel has a control wheel trim switch located on the outboard horn of each control wheel (Figure 5-1). Each switch is a four position (LWD, RWD, NOSEDOWN & NOSEUP) barrel switch with a momentary-action push button switch in the center of the barrel. This switch is used to input trim commands for pitch and roll and autopilot functions. Normally, the pitch trim control switch (Figure 5-4) is positioned to PRI. This position enables the control wheel trim switches and causes commands from either of these switches to be processed by the #1 IC. To complete the trim command circuit, the arming switch (button) on the top of the barrel must be depressed simultaneously with movement of the barrel. Trim commands from the pilot's control wheel trim switch will override commands from the copilot's. Primary trim speed is variable and is automatically controlled by the #1 IC based on indicated airspeed. The IC uses airspeed information from both ADCs to schedule trim speed and Mach trim. The #1 IC sends the primary trim commands to the primary trim actuator. The trim actuator and the IC both monitor the trim operation. The primary trim actuator performs a power-up circuit check. If the actuator detects a fault during the power-up check, a fault is posted on the CAS. Primary trim will still be available with the fault displayed, however, operation may be at a low trim rate and configuration trim and Mach trim may be inoperative depending on the malfunction. The primary trim actuator and #1 IC both monitor primary trim operations for a number of possible malfunctions including uncommanded trim and trim in the wrong direction. If either of these malfunctions is detected by the trim actuator, a fail is displayed on the CAS and primary trim is disabled.



TRIM SWITCH PANEL
Figure 5-4

Electrical power for primary pitch trim is provided by the L ESS BUS and is protected by the TRIM-PRI PITCH circuit breaker located on the pilot's circuit breaker panel (FLIGHT group). The dc electrical power to the #1 IC is also required for primary pitch trim, except for primary bypass trim. The power for #1 IC is provided by the L ESS BUS and protected by IC/SG 1 circuit breaker located on the pilot's circuit breaker panel (INSTRUMENT/INDICATIONS group).

Bypass Trim

Primary trim reverts to bypass as a result of a detected malfunction or #1 IC failure and cannot be selected by the crew. When in bypass trim a fault is displayed on CAS and control wheel trim switch commands go directly to the primary trim actuator, bypassing the IC circuits. The #1 IC trim functions (IC controller/monitored primary trim, configuration trim, and Mach trim) are all disabled in this case. When in bypass trim, the primary trim actuator operates at only two speeds (high or low). The speed depends on flap position. Dual flap position inputs are provided to the actuator electrical box to maintain redundancy. If the flap signals do not agree, the rate of trim function is limited to slow speed. Both flap signals must agree and must indicate flaps are greater than 3° for the actuator to operate at a high rate. When the flaps are up (<3°), primary bypass trim will be at a slow rate. The primary trim actuator continues to monitor for uncommanded trim and incorrect trim direction in the bypass trim mode. It also monitors for the correct trim speed based on flap position. If it detects a failure in any of these areas, primary trim is disabled and a fail CAS is displayed.

The following CAS illuminations are specific to the primary pitch trim:

CAS	Color	Description
PRI TRIM FAIL	Amber	The primary pitch trim system has failed.
PRI TRIM FAULT	White	The Integrated avionics Computer (IC) detects a fault in the primary pitch trim system.

SECONDARY PITCH TRIM

Secondary pitch trim is electrically independent of the primary trim, configuration trim, and Mach trim. In the event of primary trim failure, secondary pitch trim is available as a backup means of trimming the airplane in the pitch axis. The autopilot also uses the secondary trim actuator as a normal means of trimming in the pitch axis. The autopilot can use the secondary trim actuator with the trim selector in the PRI or SEC position.

The dual-segment SEC trim switch (Figure 5-4) is located on the center pedestal. Manual activation of secondary trim requires that the pitch trim selector be in the SEC position and that both segments of the spring-loaded SEC switch be moved at the same time. When SEC position is selected, a CAS is displayed.

The secondary pitch trim actuator has a monitor function similar to the primary actuator. It performs a power-up check and if any faults are detected, a fault is displayed on the CAS, however, secondary trim operates normally. The secondary trim actuator also monitors for uncommanded trim, trim in wrong direction, and incorrect trim rate. If any of these malfunctions are detected, a fail annunciation is posted on CAS and the secondary actuator is disabled. The IC has no control or monitor functions for manual secondary trim.

The following CAS illuminations are specific to the secondary pitch trim:

CAS	Color	Description
SEC TRIM FAIL	Amber	Secondary pitch trim has failed.
SEC PITCH TRIM	White	Secondary pitch trim is selected by the crew.
SEC TRIM FAULT	White	A pitch trim actuator (secondary) fault is detected.

Electrical power for the secondary pitch trim system is provided from the R ESS BUS and is protected by the TRIM-SEC PITCH circuit breaker located on the copilot's circuit breaker panel (FLIGHT group).

AUTOPILOT PITCH TRIM

When the autopilot is engaged, it can drive the horizontal stabilizer trim to alleviate elevator servo loading. The autopilot pitch trim function is contained in the #2 IC autopilot processor. When elevator servo current exceeds a predetermined threshold for a given period of time, this is considered to be a steady state error and trim will run. As the trim runs, the horizontal stabilizer is re-positioned and the air load on the elevator primary servo is reduced. When this load falls below the threshold level, trim stops running.

Whenever the autopilot is engaged, #1 IC trim functions, which includes config/Mach trim, drop off-line. The autopilot commands pitch trim based on elevator servo current demand and airspeed. Autopilot pitch trim engagement is controlled by the autopilot engage logic. An autopilot engage signal is provided to the horizontal trim actuator. If the autopilot is disengaged as a result of a monitor trip, the aural tone alert will sound until the MSW switch is pushed. A red AP will also be displayed on the PFDs and flash for five seconds and then go steady. The #2 IC monitors for uncommanded trim, trim direction, and incorrect trim rate. If the actuator detects one of the above faults, a CAS is displayed.

The following CAS illumination is specific to the autopilot pitch trim:

CAS	Color	Description
AP ELEV MISTRIM	Amber	Autopilot elevator servo holding excessive torque.

TRIM-IN-MOTION INDICATION

A trim-in-motion potentiometer is installed on the secondary trim actuator. When the autopilot energizes the secondary trim actuator for more than 2 to 3 seconds, a series of audible clacker sounds is transmitted through the audio system. A built-in time delay allows trim operation for approximately 2 to 3 seconds before the clacker sounds which prevents a nuisance alarm on the clacker. For longer periods of continuous trim, the clacker will alert the crew. Unusual long periods of autopilot trimming may indicate trim runaway. There is no trim-in-motion clacker for any trim operation other than autopilot trim.

PITCH TRIM BIAS

The pitch trim bias system works in conjunction with the up/down spring assembly. Its function is to assist the pilot by providing added spring pressure against the elevator in the event the horizontal stabilizer is jammed in an out-of-trim position. Pitch trim bias is actuated by the crew using the three-position (spring loaded to the center position) PIT TRIM BIAS switch located at the front of the throttle quadrant. Power for the system is provided through the PIT TRIM BIAS circuit breaker on the copilot's circuit breaker panel (FLIGHT group).

The following CAS illuminations are specific to the pitch trim bias.

CAS	Color	Description
PIT TRIM BIAS	Red	Abnormal PIT TRIM BIAS configuration with the aircraft on the ground and either thrust lever is advanced to MCR or above.
PIT TRIM BIAS	White	The pitch trim bias system is moved from the normal position. PIT TRIM BIAS should only be used for jammed stabilizer conditions in flight.

CONFIGURATION TRIM

The configuration trim functions aid the pilot by providing automatic relief of control column loads via the #1 IC control of horizontal stabilizer position. The configuration trim system control and monitoring functions are provided by software contained in the #1 IC using inputs from spoiler lever position sensors. Through these interfaces, the configuration trim provides automatic pitch control for changes in airplane configuration. This mode is only functional when the trim selector switch is in the PRI position and the autopilot is not engaged. Trim commands from either control wheel trim switch will have priority over the configuration trim commands.

AILERON TRIM

The aileron trim system provides manual aileron trim tab control. The manual trim tab control system enables the pilot, with authority, and the copilot to eliminate out-of-trim forces which may be present in the aileron control circuit, preventing smooth operation of the control column. This enables the airplane to be flown without either pilot having to apply constant forces to the hand wheel to maintain the wing level. The aileron trim system is controlled by a control wheel trim switch mounted on the pilot and copilot's control wheels (Figure 5-1) and incorporates two switches, trim, and trim arm. To manually trim, the pilot or copilot must press and hold the ARM button while pushing the trim switch to the LWD or RWD position. The control wheel trim switch induce inputs into the roll trim control electrical system which translates commands to a rotary actuator mounted in the left aileron. The actuator moves the aileron trim tab through dual push rods to the command position. A trim tab position sensor is attached to the rotary actuator shaft and provides input to the Data Acquisition Units (DAUs) for display of aileron trim position on the cockpit Engine Indicating and Crew Alerting System (EICAS). Driving the actuator clockwise causes the trim tab to rise. This results in left aileron moving down and the right aileron moving up. This results in the airplane performing a Right Wing Down (RWD) movement. Conversely, driving the actuator counterclockwise causes the trim tab to lower. This causes the left aileron to move up and the right aileron to move down, resulting in the airplane performing a Left Wing Down (LWD) movement.

Aileron trim is powered from the L ESS BUS and is protected by TRIM-AIL 5-amp circuit breaker on the pilot's circuit breaker panel (FLIGHT group).

RUDDER TRIM

The rudder trim system provides manual rudder trim tab control. The manual trim control system enables the pilot and copilot to eliminate out-of trim forces which may be present in the rudder control circuit. This enables the airplane to be flown without either pilot having to apply a constant force to the rudder pedals.

Rudder trim changes are effected through an electronically driven rotary actuator mounted in the rudder and connected to the rudder trim tab with dual pushrods. The actuator is controlled manually by a double-pole, double-throw, center-off, momentary-action, rotary switch located on the trim switch panel (Figure 5-4) in the center pedestal. This switch is constructed in two sections with poles that are not mechanically linked. One pole of the switch is used to provide control of the rudder trim ARM circuit and is referred to as the ARM switch. The other pole of this switch is used to provide either nose left or nose right trim commands and is called the rudder trim switch. These poles are independent of each other except of the fact that they are both rotated by the same shaft. The failure of one pole will not affect the other. Since one pole provides ARM control and the other provides the trim command inputs, the failure of one pole will not result in a trim runaway. Setting and holding the switch to the NOSE LEFT or NOSE RIGHT position energizes the trim tab actuator, resulting in the rudder rotating either clockwise or counterclockwise. A trim tab position sensor is attached to the rotary actuator shaft and provides input via the #2 data acquisition unit for display of rudder trim position on the cockpit engine indicating and crew alerting system.

Electrical power for rudder trim is provided from the R ESS BUS and protected by TRIM-RUD 5-amp circuit breaker located on the copilot's circuit breaker panel (FLIGHT group). Rudder trim can be stopped by depressing and holding either control wheel master switch.

TRIM INDICATIONS

Pitch, aileron, and rudder trim indications are provided on the EICAS and the MFD. A digital display of pitch trim position (PIT TRIM) is always in view below the CAS window, on the right side of the EICAS. Pitch (PIT), aileron (AIL), and rudder (RUD) trim are digitally displayed on the SUMRY page. They are arranged in a vertical column labeled FLT on the right side of the SUMRY page. The SUMRY page is the power-up default display on the EICAS. The SUMRY page is displayed at the base of the MFD. Trim indications are correspondingly displayed on the left side of the FLT system schematic page.

The following CAS illuminations are specific to the trim indications:

CAS	Color	Description
TAKE OFF TRIM	Red	The aircraft is on the ground and either thrust lever is advanced to MCR or above, and aircraft trim (pitch, aileron, or rudder) is not set for takeoff.
TAKE OFF TRIM	White	The aircraft is on the ground and aircraft trim (pitch, aileron, or rudder) is not set for takeoff.

PITCH TRIM INDICATIONS

Pitch trim tab position is presented as both analog and digital display. The label PITCH, in cyan, is positioned above the pitch trim tab position digital readout. The range of pitch trim is from 0 to 10, and with 0 being maximum nose down trim and 10 being maximum nose up trim. The analog scale consists of a white vertical line with three horizontal tick marks on the right side. The labels NDN and NUP are displayed at the left top and bottom of the scale, respectively. The digits 0 and 10 are displayed at the right top and bottom of the scale, respectively. The analog scale has a white takeoff band located between 5.5 units and 8.7 units. There is a pointer which moves up and down the left side of the scale in accordance with the digital readout of the pitch trim tab position. If the pitch trim is not within the takeoff band, and the airplane is on the ground, the digital display of trim will have a white box around it and a message posted to CAS. If power is advanced for takeoff (MCR or greater) and pitch trim is not within the takeoff band, the "CONFIGURATION" voice warning will sound and the CAS message turns red along with the digits, pointer and box in the trim position display. Invalid data will replace the digits with amber dashes, and the pointer and box are removed.

AILERON TRIM INDICATIONS

Aileron trim tab position is presented as both analog and digital display. The label AILERON, in cyan, is positioned above the aileron trim tab position digital readout. The range of aileron trim position is from L12 to R12 and with L being left wing down, and R being right wing down. The analog scale consists of a white arc with three tic marks on the outside of the arc. The digits 10, in white, are displayed at the left and right ends of the scale, respectively. A white takeoff trim band is located on the outside of the scale between the values of +5 and -5. A pointer moves along the inside of the scale in accordance with the digital readout of the aileron trim tab position. If the aileron trim is not within the takeoff band while the airplane is on the ground the digital display will have a white box around it. CAS messages and alerting are the same as those described above in pitch trim.

RUDDER TRIM INDICATIONS

Rudder trim tab position is presented in both analog and digital display. The label RUDDER, in cyan, is positioned above the rudder trim tab position digital readout. The range of rudder trim position is from L12 to R12, and with L being nose left and R being nose right. The analog scale consists of a horizontal white bar with three tic marks on the top of the bar. The digits 10, in white, are displayed at the left and right ends of the scale, respectively. A white takeoff trim band is located on the top of the horizontal scale between +5 and -5. There is a pointer which moves along the bottom of the scale in accordance with the digital readout of the rudder trim tab position. If the rudder trim is not within the takeoff band while the airplane is on the ground the digital display will have a white box around it. CAS messages and crew alerting are the same as described in pitch trim. Pitch, aileron, and rudder trim indications are available on page 2 of the backup engine/system pages on the RMU.

MACH TRIM

Mach trim is a fully automatic system installed to increase longitudinal stability and counteract nose-down tendency at high Mach numbers. A circuit card in the #1 IC performs all the computational aspects for Mach trim and signals the primary trim actuator to apply trim as necessary. Airspeed information provided by the ADCs is used by the IC in computing the trim requirement.

The pitch trim selector (Figure 5-4), located on the center pedestal, must be in the PRI position for Mach trim to be functional and the autopilot must be disengaged for the Mach trim to become active. If the autopilot is engaged, it performs the pitch trim function using the secondary trim actuator and the Mach trim is in a passive mode. Mach trim automatically becomes active at 0.725 MI. Nose up trim will be applied as Mach increases and nose down as Mach decreases. When the horizontal stabilizer position changes, two Mach trim position sensors apply feedback signals to the IC. Mach trim is interrupted whenever the manual trim is activated. The system resynchronizes to function about the new horizontal stabilizer position when manual trim is released. If the IC detects a fault within the Mach trim system function, a fail is posted on the CAS and the overspeed cue on the airspeed indicator will also adjust to indicate a Mach limit of 0.76 to 0.78 MI.

The following CAS illuminations are specific to the Mach trim:

CAS	Color	Description
MACH TRIM FAIL	Amber	Mach trim function has failed and aircraft speed is greater than 0.76 to 0.78MI.
MACH TRIM FAIL	White	Mach trim function has failed and aircraft speed is equal to or less than 0.76 to 0.78MI.

STALL WARNING SYSTEM

The stall warning system, also referred to as the Angle-of-Attack (AOA) system, is installed to provide the crew with an indication of impending airplane stall. The stall warning system consists of two independent systems which use a dual channel computer.

Other system components include two AOA sensors, control column shaker motors and an interface with the PFDs. Left and right AOA indicators are available as an option. Each channel of the computer generates a reference signal to the corresponding stall vane and, in return, receives AOA information. The computer then processes this information with airspeed, altitude, flap setting, and weight-on-wheels inputs to determine the stall warning indications. The left and right stall warning systems are powered from the left and right essential buses respectively. The circuits are protected by the L STALL WARN and R STALL WARN circuit breakers located on the pilot's and copilot's circuit breaker panels (FLIGHT group).

STALL WARNING INDICATIONS

As the airplane approaches stall speed, stall warning indications are activated. The shaker speed will be above the stall speed at the most critical weight and Center of Gravity (CG). The stall warning computer sums inputs of AOA and altitude shift along with flap position from the flap position indication unit. Stall warning is biased for each flap setting. The stall warning system provides the following aural, tactile, and visual indications when the predetermined conditions have been reached:

(1) The left and right channels of the computer drive low-speed cues on the pilot's and copilot's PFDs respectively. The low-speed cue is a vertical red bar on the inside of the airspeed tape which rises from the bottom of the tape as the airplane AOA increases. The point at which the red bar reaches the airspeed pointer will coincide with the point at which other stall warning indications are activated.

(2) The left and right channels of the stall computer will activate the control column shaker motors.

(3) The non-cancelable voice message "STALL" will repeat until the AOA is decreased.

(4) The AOA indicators (if installed) will enter the red band on the indicator.

STALL WARNING OPERATION

The stall warning system is powered when the circuit breakers are in. The shakers, along with other visual and aural stall indications, are inhibited until the airplane is airborne. If installed, the AOA indicators will operate in both the air and ground modes. The stall warning system performs a power-up self-test (BIT) and monitors for a number of possible system faults. Detection of a fault appears on CAS.

The following CAS illumination is specific to the stall warning system:

CAS	Color	Description
STALLWARN FAIL	Amber	The associated (L or R) stall warning system has failed.

STALL VANE ANTI-ICE

The stall vanes are equipped with a 28-vdc heater to anti-ice the vane surfaces during icing conditions. The AOA vane heater of the angle-of-attack transmitter is monitored for open circuit when the vane heater power is applied. Detection of an open circuit will result in the appropriate CAS message as well as being logged into the stall computer as a fault for that flight. The vane heaters are controlled by the L and R PROBE anti-ice switches located on the anti-ice section of the center switch panel. Each vane heater is supplied power from the left and right main bus respectively and protected by the AOA 15-amp circuit breakers on the pilot's and copilot's circuit breaker panel (ANTI-ICE group).

The following CAS illumination is specific to the stall vane anti-ice system:

CAS	Color	Description
AOA HT FAIL	Amber	Associated (L or R) angle-of-attack vane heater has failed.

STALL SYSTEM TEST

A self-test mode is available when the weight-on-wheels signal indicates that the airplane is on the ground and no system failure is detected. When the system test switch is rotated to the STALL position and held down for approximately 7 to 10 seconds, the stall warning computer shall demonstrate that the stall warning system is fully operational by performing the following events in the order listed:

(1) L AOA HT FAIL message appears in CAS.

(2) Low Speed Awareness (LSA) bar will begin to sweep up the pilot side airspeed tape, the left (pilot's) column will shake when the LSA bar approximately reaches the indicated airspeed pointer, and the aural voice warning "STALL" will be repeated through the cockpit speakers and crew headphones.

(3) L AOA HT FAIL message extinguishes from the CAS window, the LSA bar scrolls down the airspeed tape, the left column stops shaking and the aural warning stops.

(4) R AOA HT FAIL message appears in CAS. (Note: Master caution tone may not sound when the R AOA HT FAIL is annunciated. If the master caution tone is not heard, then the STALL aural warning will be heard as called out in the next step below).

(5) The LSA bar will begin to sweep up the copilot side airspeed tape, the right (copilot's) column will shake when the LSA bar approximately reaches the indicated airspeed pointer, and the aural voice warning "STALL" will be repeated through the cockpit speakers and crew headphones (if the master caution tone was not heard in the previous step).

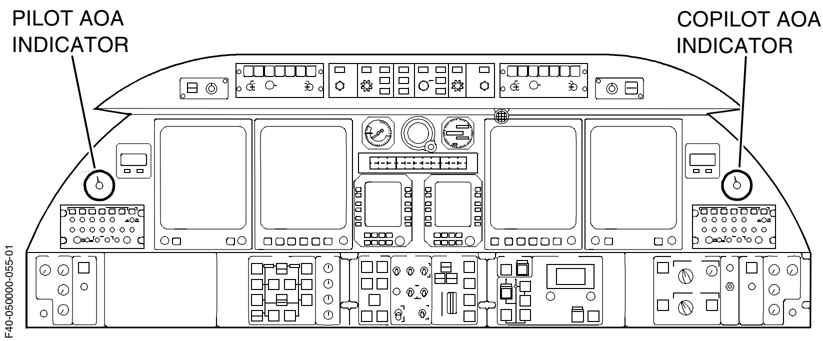
(6) R AOA HT FAIL message extinguishes from the CAS, the LSA bar scrolls down the airspeed tape, the right column stops shaking and the aural warning stops (if the aural warning was present in the previous step).

The left and right stall warning failure discretes are not checked during self-test. It was necessary to inhibit the output of the left and right failure discretes in order to permit display of the LSA bar on the PFDs during test.

ANGLE-OF-ATTACK INDICATORS (OPTIONAL)

The optional Angle-of-Attack (AOA) indicators system consists of two angle-of-attack indicators mounted in the instrument panel, one outboard of the pilot PFD and one outboard of the copilot PFD, Figure 5-5.

The angle-of-attack indicators display continuous angle-of-attack position to the flight crew. The AOA indicators are driven by the stall warning computer. The dual channel computer provides buffered outputs to the indicators for protection. The pilot AOA indicator receives data from the left channel of the stall warning computer and the copilot AOA indicator receives data from the right channel of the stall warning computer. The AOA indicator is adequately marked displaying .10, .20, .30, .40, .50, .60, .70, .80, .90, 1.0 with unnumbered marks half way between each. The beginning of the red band at .80 represents shaker activation and an imminent stall condition. The AOA indicators front plate markings are consistent with the stall warning information shown on the PFDs, a tape type presentation at the end of the airspeed tape. The AOA indicators are powered via the L STALL WARN and R STALL WARN circuit breakers located on the pilot's and copilot's circuit breaker panels (FLIGHT group).



INSTRUMENT PANEL LAYOUT AND AOA INDICATOR POSITION
Figure 5-5

AVIONICS

HONEYWELL PRIMUS 1000 AVIONICS SYSTEM

The Learjet 45 is equipped with a Honeywell Primus 1000 Avionics system. The primary component of the Primus 1000 system is the display flight guidance computer, or more simply, the IC-600. This computer, together with the appropriate controllers and sensors, comprises the Primus 1000 system. It consists of dual IC-600 (single autopilot is contained in the copilot's IC-600), dual air data computers, PRIMUS weather radar system and appropriate controllers. The radio sensor package is the Honeywell PRIMUS II integrated radio system.

ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS)

The PRIMUS 1000 EFIS System consists of four, 8 x 7 inch, Display Units (DUs) driven by two Symbol Generators (SGs) resident in the two IC-600s. The EFIS presents information to the crew in an uncluttered format, simplifying cockpit scan, and reducing pilot workload and fatigue. The flight instruments, engine instruments, system status, navigation, TCAS, RADAR, and electronic checklist are all displayed on these high resolution DUs. The EFIS is integrated with the Engine Indicating and Crew Alerting System (EICAS) and Crew Warning Panel (CWP) to provide the crew with not only flight monitoring indications but also with engine data, warning, cautionary and advisory alerts (visual and aural). Dual Primary Flight Displays (PFDs) combine attitude and HSI formats with airspeed, vertical speed and other essential information, such as resolution advisories for the optional TCAS system. A Multi-Function Display (MFD) offers a full spectrum of operational capabilities, from weather radar and mapping displays, to a custom programmable checklist. A digital audio control system and dual Radio Management Units (RMUs) support the communications and navigation functions.

The display information provided on EFIS is generated by two IC-600 computers located in the nose. Each of the IC-600s contains circuitry that performs the symbol generation function for the EFIS. Along with interfacing with the display units, the ICs receive data from the Data Acquisition Units (DAUs), Air Data Computers (ADCs), Attitude Heading Reference System (AHRS), navigation system, flight management system, autopilot and other various display controllers. The CAS monitors the IC-600 bus interconnect, the temperature of each IC-600, the IC cooling fans and Weight-On-Wheels (WOW). A CAS will also illuminate if communications between the left and right ICs are invalid.

The following CAS illuminations are specific to the IC-600:

CAS	Color	Description
IC 1-2 OVHT	Amber	#1 and/or #2 Integrated avionics Computer (IC) are/is overheated.
IC BUS FAIL	Amber	-The off-side IC has failed. or - IC bus invalid
IC1-2 FAN FAIL	White	#1 and/or #2 Integrated avionics Computer (IC) cooling fan has failed.
IC1-2 WOW INOP	White	The associated (#1 or #2) Integrated avionics Computer (IC) has tripped the weight-on-wheels validity monitor.

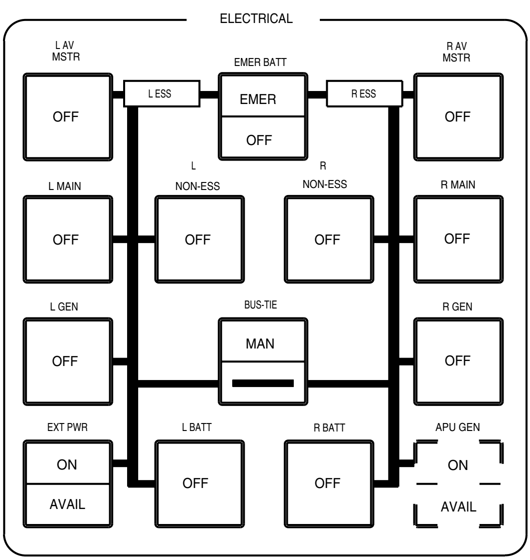
IC-600 POWER SOURCE

The #1 and #2 IC-600s are powered from the left and right essential buses respectively. The circuits are protected by the 7.5-amp IC/SG 1 and IC/SG 2 circuit breakers on the pilot's and copilot's circuit breaker panels (INSTRUMENT/INDICATIONS group).

AVIONICS MASTER SWITCHES

Left and right avionics master switches are located on the electrical control panel below DU 2 (Figure 5-6). When the alternate action avionics master switches are selected to On (OFF annunciator extinguished), contactors are closed that connect the left and right essential avionics buses and left and right main avionics buses to the respective generator buses.

The associated essential contactors and main bus contactors must be closed for the avionics buses to be powered. If the avionics master switches are on during ground start or for a starter assisted airstart, the essential avionics buses will continue to be powered, but the contactors for the main avionics buses will automatically open until the start is complete. The essential avionics buses must remain powered during an airstart since they power the critical flight display units. The emergency bus, essential buses and essential avionics buses are all powered by the emergency battery during a starter assisted start. The avionics equipment that must be on during a ground start is powered from the essential buses and the emergency battery bus.



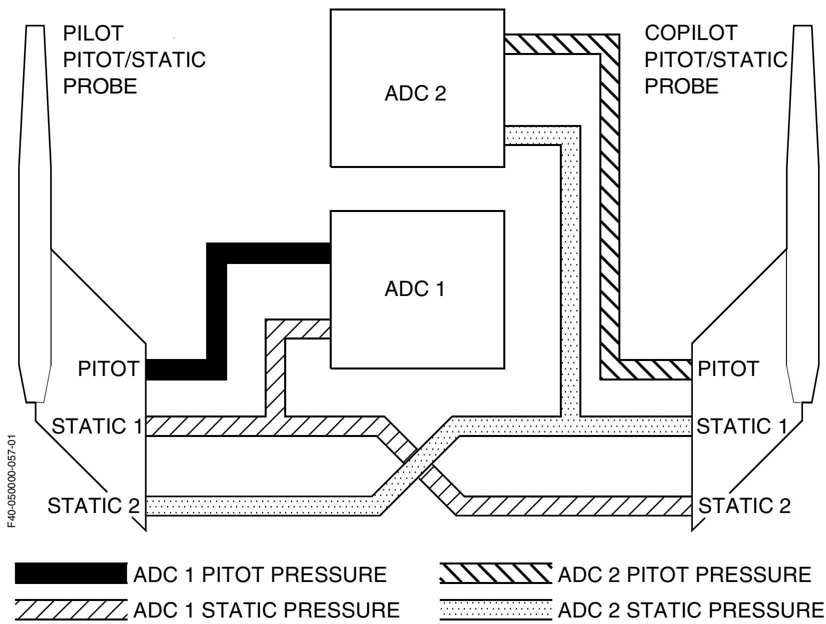
ELECTRICAL CONTROL PANEL
Figure 5-6

AIR DATA SYSTEM (ADS)

The air data system and air data instruments depend upon pitot pressure and static pressure sensing, as well as air temperature sensing. Air data is provided to the flight instruments and airplane systems by two Air Data Computers (ADCs) which receive pitot and static information from the main pitot static system. The ADCs receive total air temperature from a dual element temperature probe and barometric correction inputs via the BARO set knobs on the corresponding PFDs.

PITOT-STATIC SYSTEM

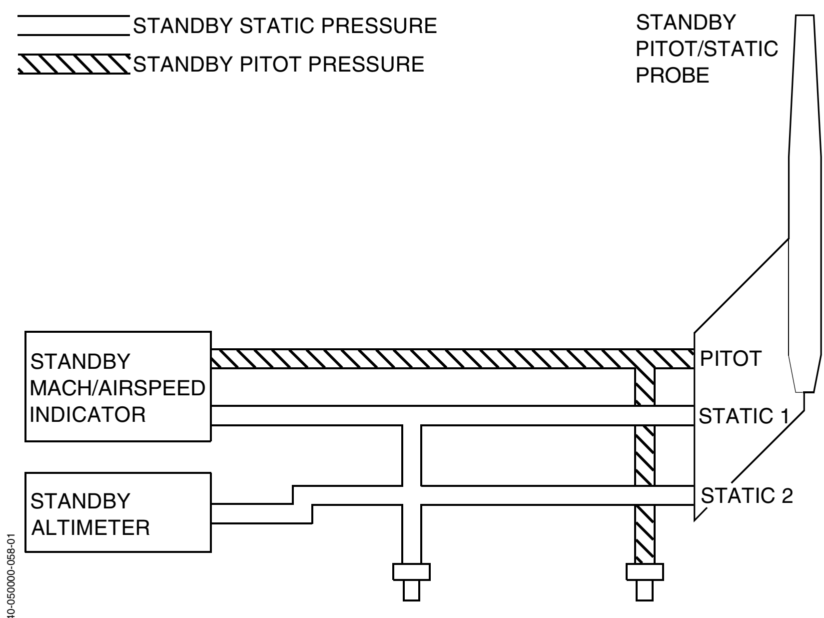
The primary pitot-static system consists of two pitot-static probes, located one on each side of the airplane's nose section. The pilot's pitot-static probe supplies the pilot's ADC with total pressure and the copilot's pitot-static probe supplies the copilot's ADC with total pressure. Each pitot-static probe has two isolated static ports. The pilot's ADC receives static pressure from coupled static ports off the pilot's and copilot's pitot-static probes (Figure 5-7). The copilot's ADC receives static pressure from separate coupled static ports off the pilot's and copilot's pitot-static probes which are isolated from the static ports used by the pilot's ADC.



PITOT-STATIC SYSTEM SCHEMATIC
Figure 5-7

Static 1 from the left probe connects with static 2 on the right probe to provide static pressure to the pilot's ADC. Static 1 from the right probe combines with static 2 on the left probe to provide static information to the copilot's ADC. This crossover arrangement reduces system errors (Figure 5-7).

A third pitot-static probe, mounted above the main probe on the right side of the airplane, provides total and static pressure inputs to the standby instrument group. Moisture drains are provided for the standby pitot-static lines. The two drains for the standby pitot-static system are flush mounted on the right side of the airplane just aft of the nose wheel door. The main pitot-static probes are physically located at the lowest point of the primary pitot-static system plumbing and therefore, do not require moisture drains. The pitot source on the standby probe provides total pressure to the standby Mach/airspeed indicator. There are two static sources on the standby probe, one provides static information to the standby altimeter and the other provides data to the standby Mach/airspeed indicator (Figure 5-8).



STANDBY PITOT-STATIC SYSTEM SCHEMATIC
Figure 5-8

AIR DATA COMPUTERS (ADCs)

The Learjet 45 utilizes two, independent micro air data computers as the primary source for air data. The is a self contained unit incorporating pressure sensing modules and all required processing and input/output functions in a single unit. Each computer is independent of the other and has independent circuit breakers.

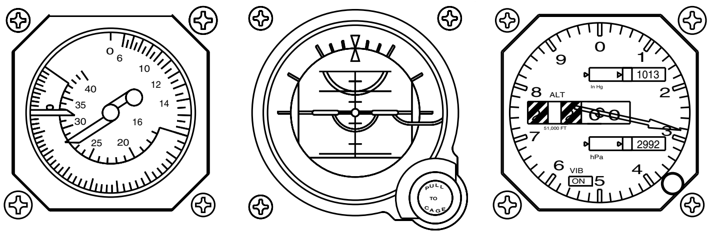
The air data system provides the required airplane airspeed, air temperature, altitude and vertical speed data for the Electronic Flight Instrument System (EFIS) displays, Attitude and Heading Reference System (AHRS), dual stall warning system, autopilot, transponders, spoileron computer, cabin pressurization, Digital Electronic Engine Control (DEEC) and landing gear warning system as required. The Air Data System (ADS) accepts static air pressure, total air pressure, total air temperature, various discrete signals and baro set inputs. The #1 and #2 ADCs receive power from the L and R ESS BUS respectively, through ADC 1 and ADC 2 circuit breakers. The circuit breakers are located on the pilot's and copilot's circuit breaker panels (INSTRUMENT/INDICATIONS group).

ADC REVERSION

Display of the ADS data on the EFIS display is controlled by the pilot or copilot via the ADC reversionary control switch (Figure 5-14). The reversionary control panel, located below the EICAS display (DU#2), incorporates an ADC reversion switch which has three positions, 1 - ADC NORM - 2. In the ADC NORM position, the IC-600s receive air data from their on-side ADC. In the "1 or 2" position both IC-600s receive air data from the selected source. If the switch is not in the NORM position, an annunciator of the selected source is displayed above and to the left of the ADI on both PFDs.

STANDBY INSTRUMENTS

The standby instrument group (Figure 5-9) includes a barometric altimeter, a airspeed/Mach indicator, an attitude indicator, mounted on the center instrument panel above the CWP and RMUs. The standby instruments have their own pitot-static probe to provide air data information. The instruments are of traditional mechanical design. If a fault occurs which causes one of the ADCs to output misleading information to the PFDs, the standby instruments act as a useful comparison to indicate which of the three displays is incorrect.



STANDBY INSTRUMENT GROUP
Figure 5-9

STANDBY ALTIMETER

The standby altimeter displays baro corrected altitude in a pointer/counter drum display. The dial graduations are marked every 20 feet. Above sea level the counter displays every 100 feet up to 55,000 feet of operational range. The indicator has dual barometric correction, from 27.9 to 31 inches of mercury and 946 to 1050 hectoPascals. Back lighting is provided by 5-vdc to illuminate the standby altimeter indicator at night.

STANDBY AIRSPEED/MACH INDICATOR

The standby airspeed/Mach indicator provides indicated airspeed by means of a pointer indicating against a 50- to 400-knot dial and a Mach sub-dial ranging from 0.3 to 1.0 Mach. Maximum allowable airspeed (Vmo) is indicated at 325 knots by a red radial mark on the airspeed dial. Maximum allowable Mach (Mmo) is indicated at 0.75 Mach by a red and white striped radial mark on the Mach sub-dial. Back lighting is provided by 5-vdc to illuminate the standby airspeed/Mach indicator at night.

STANDBY ATTITUDE INDICATOR

The standby attitude indicator provides a visual indication of the airplane flight attitude. It is located in the center of the standby instrument group (Figure 5-9) where it can be viewed easily by both pilots. It is powered from the emergency battery bus so that it will remain powered for at least one hour after the loss of airplane generator power. The standby attitude indicator will continue to provide an accurate display of aircraft attitude for a further nine minutes after the loss of all airplane power.

The indicator is an electrically-driven gyro whose vertical attitude is maintained by a mechanical erection system. The power warning flag is pulled from view after the gyro has spun up to valid operating speed and reappears if there is any interruption of source power or the unit is in caged mode.

Back lighting is provided by 5-vdc to illuminate the standby attitude indicator at night.

STANDBY COMPASS

The standby compass is located at the top of the windshield center post. It is a magnetic compass that does not require any electrical power to provide the crew with a continuous standby heading display. The only electrical input to the compass is 5-vdc to illuminate the compass at night.

ATTITUDE HEADING REFERENCE SYSTEM (AHRS)

Due to the design of the Honeywell Primus 1000 Avionics System, the various avionics systems are very integrated. The AHRS uses the PFDs as its primary display and the MFD in the event of PFD failure. The display units, display controllers and appropriate reversion switches are considered part of the electronic display system and are covered later in this section.

The Learjet 45 is equipped with either a dual Honeywell (AH-800) or dual LITEF (LCR-93) Attitude Heading Reference Units (AHRUs). Both units contain a memory module and are located in the aircraft's nose section.

Each system (#1 and #2 AHRS) incorporates a flux valve located in their respective wing tips. The AHRUs contain three Fiber Optic Gyros (FOGs) which sense angular rotation about the three principle axis (pitch, roll, and yaw) thus, computing the airplane's attitude and heading. When slaved to magnetic, the flux valves provide a magnetic heading reference. The memory module stores calibration data. This data is used to compensate AHRU inaccuracies caused from installation errors and local disturbances to the earth's magnetic field created by the aircraft's structure.

The AHRUs receive true airspeed (TAS) information from the on-side ADC. However, if a single ADC failure occurs, they will receive TAS from the operating ADC. True airspeed information is used to compute pitch and roll attitude. If TAS inputs to #1 AHRS or #2 AHRS are lost, a CAS will illuminate. Although system operation will be degraded, the AHRS still retains the same accuracy as a conventional spinning mass type gyro. AHRU's data output is received through their corresponding IC for attitude and heading displays on the PFDs/MFD. Attitude and/or heading information from the AHRS is used by the Flight Guidance System (FGS), Flight Management System (FMS), weather radar system, and the fuel quantity indicating system. In addition, AHRS #2 provides heading information through DAU #2 for the backup navigation display on the RMU.

The following CAS illumination is specific to the AHRUs:

CAS	Color	Description
AHRS 1-2 BASIC	White	Attitude Heading Reference System (AHRS 1 or 2) has reverted to basic mode due to a loss of true airspeed from both air data computers.

ATTITUDE AND HEADING COMPARISON MONITORS

The attitude and heading comparison monitors are functions within the IC-600s that compare the displayed data with the cross-side or secondary source data, depending on system reversionary status. Annunciations are provided to the crew if the attitude or heading on both sides differ.

The attitude comparison function is made of two monitors, the roll comparison monitor and the pitch comparison monitor. If the pitch data displayed on each side differ, the pitch comparison monitor trips and the PIT annunciation is displayed. The comparison threshold figure for the pitch monitor is 5°. Similarly, if the roll data on both sides differ, the ROL annunciation is displayed. The comparison threshold figure for the roll monitor is 6°. If both the roll and pitch comparison monitors trip, the ATT annunciation is displayed. If the heading comparison monitor trips, HDG is displayed. The normal comparison threshold figure for the heading monitor is 6°. However, if the displayed roll information is > 6°, the heading comparison threshold figure is increased to 12°.

All comparison monitor annunciations flash for 10 seconds on activation and then remain steady. These comparison monitors provide an extra safeguard to alert the cockpit crew in the event of any failures affecting the attitude or heading data displayed.

Other annunciations for attitude and heading which are displayed on the PFDs, not associated with the comparison monitors, are:

ATT FAIL and HDG FAIL. These red annunciations are displayed on the affected side's PFD whenever the heading or attitude display from that AHRS has failed. If an AHRS fails or both primary and auxiliary power supplies to an AHRS fail, both the red ATT FAIL and HDG FAIL annunciations are displayed.

ATT1/2 and DG1/2. These annunciations are displayed on the PFDs and indicate to the crew which AHRS is the source for the attitude and heading data on the display. If the onside AHRS is the source of display, the annunciation is white. If AHRS reversion has been performed, the cross-side PFD annunciation is amber. There are no crew actions required for these annunciations.

AHRU POWER SOURCE & COOLING

Each AHRU has a primary and a secondary dc electrical power source. The pilot's AHRU receives primary power from the left essential bus and a secondary or backup power from the right essential bus. The co-pilot's AHRU receives primary power from the right essential bus and secondary power from the left essential bus. Should either essential bus fail in flight, power to both AHRUs is uninterrupted. Separate circuit breakers for each system, primary and secondary, are provided in the INSTRUMENT/INDICATIONS group on the pilot's and copilot's circuit breaker panels. The AHRS #1 PRI and #2 SEC circuit breakers are located on the pilot's side, and the AHRS #2 PRI and #1 SEC circuit breakers are located on the copilot's side. The AHRUs are equipped with cooling fans which operate automatically to keep the AHRU within proper temperature limits. *On aircraft 45-002 thru 45-174 (Honeywell AH-800),* a CAS illuminates if the temperature exceeds predefined limits.

The following CAS illumination is specific to AHRU cooling:

CAS	Color	Description
AHRS 1-2 OVHT (Aircraft 45-002 thru 45-174)	Amber	Attitude Heading Reference System (AHRS 1 or 2) has reached an overheat condition.

AHRS REVERSION

Failure of an AHRS is apparent when the on-side horizon and pitch lines are removed from the ADI and a red ATT FAIL annunciator appears in the upper center of the ADI. The heading compass rose will display a HDG FAIL annunciator on the HSI. If either AHRS fail, the AHRS reversion switch on the reversionary control panel (Figure 5-14) will allow the pilot to select the remaining AHRS to provide attitude and heading information to both displays. The three-position switch is labeled 1 - AHRS NORM - 2.

ELECTRONIC DISPLAY SYSTEM (EDS)

Four electronic displays are used to provide the display formats for the Primary Flight Displays (PFDs), the Multi-Function Display (MFD) and the EICAS display in the electronic flight instrument system. The four display units are large format 8 x 7 inch, 16 color high resolution display tubes. The display units are identical and interchangeable, except for the bezel controllers attached to the front of the units. The bezel controllers for the outboard DUs are the same and the bezel controllers for the inboard DUs are the same. A display controller (two) provides the means for each pilot to control the display of the on-side PFD and to activate the EFIS test function. A display unit reversion panel, located above the PFDs, provides reversion control capability.

The display unit configuration powers up with the following displays:

DU#1 - Pilot's Primary Flight Display (PFD #1)
DU#2 - EICAS Display
DU#3 - Multi-Function Display (MFD)
DU#4 - Copilot's Primary Flight Display (PFD #2)

The above configuration can be changed using the EICAS reversion switch. This provides the ability to swap the DU #2 and DU #3 displays between EICAS and MFD as the pilots desire.

The display units require forced air circulation for cooling which is provided by two fans mounted on the rear of each DU. If a DU fan fails, a CAS will illuminate indicating DU 1, 2, 3, or DU 4 fail. If the temperature of the DU reaches approximately 120° F, a CAS will illuminate.

The following CAS illuminations are specific to the DUs:

CAS	Color	Description
DU 1-2 OVHT	Amber	#1 and/or #2 Display Unit (DU) is overheated.
DU 3-4 OVHT	Amber	#3 and/or #4 Display Unit (DU) is overheated.
DU1-2 FAN FAIL	White	#1 and/or #2 Display Unit (DU) cooling fan has failed.
DU3-4 FAN FAIL	White	#3 and/or #4 Display Unit (DU) cooling fan has failed.

PRIMARY FLIGHT DISPLAY (PFD)

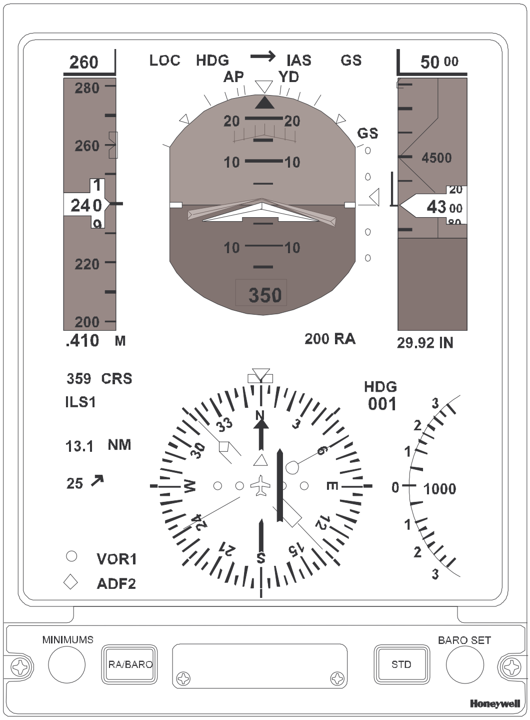
The PFD (DU #1 and DU #4) is a single display in which all of the required flight and navigation data is displayed for each pilot. The PFD (Figure 5-10) format is divided into two main sections. The top half displays an Attitude Director Indicator (ADI) with an airspeed tape to the left, and a barometric altitude tape to the right. A Horizontal Situation Indicator (HSI) is located on the lower half of the PFD. The HSI can be displayed in three different formats. The three options are full 360° compass rose (HSI), a 120° compass arc display (ARC), and a 120° map display (MAP). The MAP cannot be displayed on the PFD if the adjacent display (DU #2 or DU #3) is already displaying an MFD MAP format. Weather information can be displayed on the PFD ARC or MAP format. To the right of the HSI, a Vertical Speed Indicator (VSI) is displayed, and to the left, navigation information is annunciated.

Comparison monitors provide indications to the pilots that there is a difference between the data displayed on each PFD. This monitoring is a function within the IC-600s that compares what is being displayed on one side with either the cross-side displayed data or the secondary source data. Should data be out of tolerance between what is being reported from the source, and what is being sent to the display units, or if avionics related exceedances are detected, CAS will illuminate.

The following CAS illumination is specific to the PFDs:

CAS	Color	Description
PFD CHECK	Amber	The associated (L or R) Primary Flight Display (PFD) is displaying invalid data.

The brightness of each PFD is controlled by the DIM control on each respective display controller. Pilot's PFD (DU #1) receives 28-vdc power from the left essential avionics bus by a 15-amp circuit breaker DU 1 located in the INSTRUMENT/INDICATIONS group of the pilot's circuit breaker panel. Copilot's PFD (DU #4) receives 28-vdc from the right essential avionics bus by a 15-amp circuit breaker DU #4 located in the INSTRUMENT/INDICATIONS group of the copilot's circuit breaker panel.



PRIMARY FLIGHT DISPLAY AND BEZEL CONTROLLER
Figure 5-10

BEZEL CONTROLLERS

Many of the display control functions are controlled by the DU bezel controllers and by the menus displayed on the MFD and EICAS displays. The PFDs both use the BL-870 bezel controller located at the bottom of the PFDs. Each has two push buttons and two rotary knobs (Figure 5-10). The push buttons and rotary knobs have functions dedicated to decision height, minimum descent altitude and barometric correction.

The MFD and EICAS displays use the BL-871 bezel controllers which have six push buttons, menu keys, and a rotary knob for menu manipulation. The push buttons allow selection of functions displayed in the menus on the MFD. There are three functions that the MFD bezel push buttons provide - (1) selection of a submenu, (2) toggling the selection of a menu item and (3) selection of a variable parameter for setting. The MFD rotary knob is dedicated to the control of the map/plan range.

With weather radar selected for display, the MFD rotary knob will have no function. The EICAS bezel controller provides dedicated buttons for the displayed EICAS menu. These buttons toggle the selection of the EICAS system page displays. The EICAS rotary knob allows for scrolling of the CAS messages on the EICAS display.

MULTI-FUNCTION DISPLAY (MFD)

The MFD (normally DU #3) provides the flight crew with a means of displaying a variety of information. In its normal mode it can serve as a full time weather radar display superimposed on a 120° compass arc. There are two basic formats available on the MFD, a partial arc (Map) display, and a plan mode (North up). Like the PFD, the MFD may have flight plans composed of up to ten connected waypoints imposed on a compass card. True airspeed (TAS) provided from the ADC and ground speed (GSPD) provided from the FMS, are displayed on the MFD. Other information displayed full time include: FMS source, "TO" waypoint, distance to "TO" waypoint, time-to-go to "TO" waypoint, wind speed and direction, Static Air Temperature (SAT), and weather radar (WX) modes.

The MFD also provides a second source for access to EICAS systems pages as well as providing joystick functions. The MFD (DU #3) may serve as a backup for any other DU through pilot initiated reversionary modes.

Other information available on the MFD includes:

- TCAS mode (optional) — Controls the display of TCAS on the map presentation.
- MFD MENU — Activation of this key will enable the MFD SUB-MENU to appear.
- CHECKLIST PAGE — This key provides entry into the normal checklist procedure index page.
- SYSTEM PAGE — Selection of this key will access the systems sub-menu pages which are duplicates of the EICAS system pages.

28-vdc is provided from the right essential avionics bus by a 15-amp circuit breaker DU #3, located on the copilot's circuit breaker panel (INSTRUMENT/INDICATIONS group).

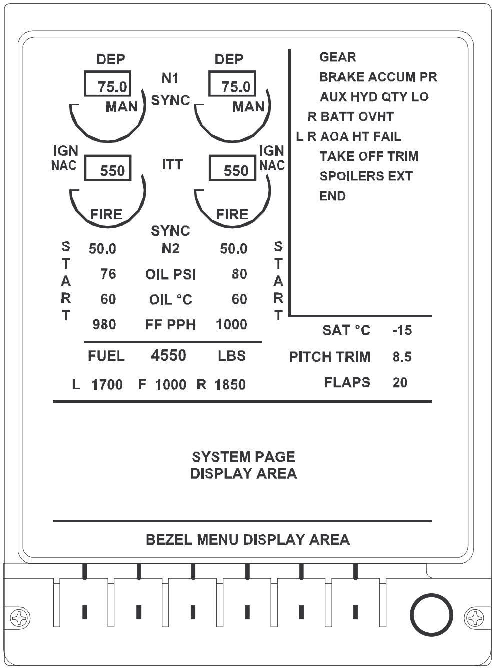
EICAS DISPLAY

The Engine Indicating and Crew Alerting System (EICAS) is an integrated digital computer/display system that replaces the majority of the traditional gauges and warning lights located throughout the cockpit. The EICAS display (Figure 5-11) is divided into four designated areas: engine instruments, CAS messages, system display pages and menu items. The EICAS also incorporates the Crew Warning Panel (CWP) which provides crew alerting by visual representation while the cockpit audio system provides the aural alerting. The Crew Alerting System (CAS) provides the crew with a visual attention getting means to alert them to a warning that requires immediate action, a caution alert that requires subsequent pilot or maintenance action, or an advisory indication that may require pilot or maintenance action at some point in time. Other airplane system parameters are displayed on the lower portion of the display via system pages and are selectable by the bezel controller at the bottom of the DU. Normally, the airplane system summary page (SUMRY on the menu) is in view, which provides brief status reports of all sub-systems. Menu selectable, a system schematic of airplane electrical, hydraulic, environmental control, flight control, and fuel systems can be individually selected for more detailed monitoring by the flight crew.

The following CAS illumination is specific to the EICAS display:

CAS	Color	Description
EICAS CHK	Amber	Available on MFD display only. EICAS wrap-around monitor.

28-vdc power is provided from the left essential bus by a 15-amp circuit breaker DU 2, located on the pilot's circuit breaker panel (INSTRUMENT/INDICATIONS group).



EICAS DISPLAY AND BL-871 BEZEL CONTROLLER
Figure 5-11

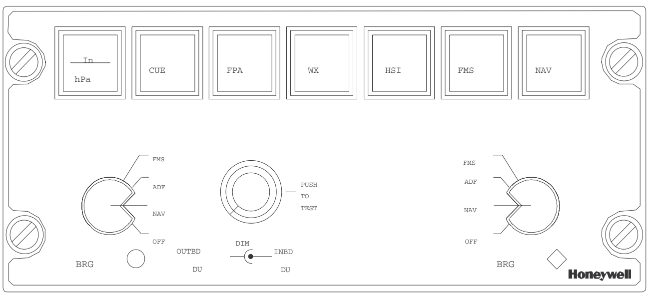
DISPLAY CONTROLLERS

The display controllers (two DC-550s), located on the glareshield, provide immediate access to and control of the objects on the PFDs. Each controller is configured with seven push buttons located on the front panel along with two rotary knobs used for reference selection for bearing source, two concentric knobs for DU dimming and a momentary push button (located inside concentric DU knob) used to initiate a system test (Figure 5-12).

The display controllers also provide a data acquisition function, collecting inputs from sources such as the bezel controllers, guidance controller, joystick, etc. The controllers pass these inputs to the corresponding IC-600 for processing.

The display controller buttons are as follows:

1. **IN/HPA** — Inches of mercury or hectopascals.
2. **CUE** — Selection of single cue or cross pointer command bars.
3. **FPA** — Controls selection and deselection of the flight path angle symbol and flight path acceleration display.
4. **WX** — Select or deselect weather radar display on the PFD.
5. **HSI** — Provides up to three different display options on the HSI.
6. **FMS** — Allows a navigation display of FMS information (alternately FMS 1 or FMS 2 if dual) to be selected for display on the PFD.
7. **NAV** — Alternately selects NAV 1 or NAV 2 as the source of NAV data on the HSI.



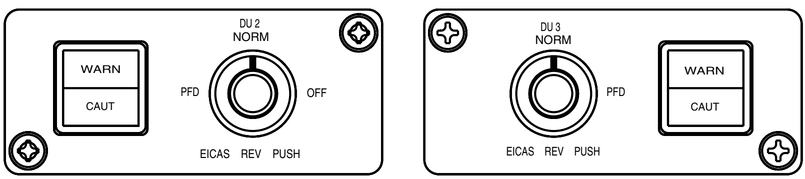
DISPLAY CONTROLLER
Figure 5-12

Each controller contains two rotary bearing source selector knobs that are used to assign the respective bearing pointers on the HSI or ARC displays to a particular navigation source.

Power for the display controllers, and the IC-600s, are from the left and right essential buses and are labeled IC/SG 1 and IC/SG 2 on the pilot's and copilot's circuit breaker panels in the INSTRUMENT/INDICATIONS group respectively.

DISPLAY UNIT REVERSION PANELS

A display unit reversion panel is located on the glareshield above the PFDs on each side of the cockpit. The panel on the pilot's side is for controlling the display on DU #2 and the panel on the copilot's side is for controlling the display on DU #3. The reversion selector knob on these panels plus the push function of the knobs allow the operators to switch the inboard DUs (DU #2 and DU #3) to display either PFD, MFD, or EICAS formats. With both reversion selector switches in NORM, an EICAS format is displayed on DU #2 and an MFD format on DU #3. Depressing the selector knob on either reversion panel flip-flops the DU #2 and DU #3 displays, reversing the MFD and EICAS display locations. Placing the reversion selector to the PFD position on either side causes the PFD format to move to the inboard display tube on that side and the outboard display to blank.



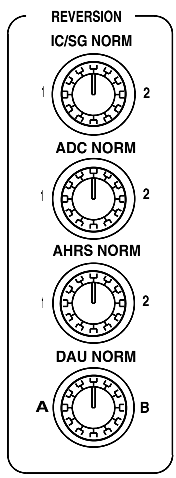
DISPLAY UNIT REVERSION PANELS
Figure 5-13

It is important to note that when selecting display unit reversion, the bezel controllers on DU #1 and DU #4 continue to work with the PFD display when it is transferred to an inboard display unit. The airplane master warning/caution lights are also located on the display unit reversion panels.

DATA ACQUISITION UNITS (DAUs)

There are two dual channel data acquisition units (DAUs) installed in the tailcone equipment area of the airplane. The DAUs receive engine and airplane systems sensor information and pass it, primarily, to the IC-600 computers. Both channels of DAU #1 provide left engine data and both channels of DAU #2 provide right engine data. For redundancy, both channels of each DAU independently convert on-side engine information to a common ARINC 429 data bus format and send it to both IC-600s. The IC-600s process the information and send it to the selected display unit (normally DU #2) for EICAS display. In addition to engine information, the DAUs also collect analog data from other airplane systems such as fuel, hydraulic and accumulator pressure, dc electrical power, flight control settings, cabin pressure settings/indications, and oxygen temperature/pressure.

A three-position DAU reversionary switch is provided on the reversionary control panel located below DU #2 (Figure 5-14). The switch positions are labeled A, DAU NORM, and B. With the switch in DAU NORM, both IC-600s use Channel A from the left DAU and Channel B from the right DAU for engine/systems displays. In the reversionary positions (A or B), each IC-600 uses only the selected channel from both DAUs.



REVERSIONARY CONTROL PANEL
Figure 5-14

If either channel of either DAU should fail, if either A or B reversion is selected, if an engine or system miscompare is detected, an appropriate CAS will illuminate.

The following CAS illuminations are specific to the DAUs:

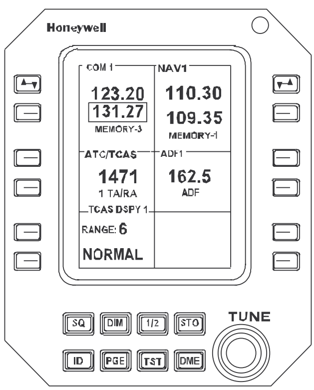
CAS	Color	Description
DAU 1A-1B FAIL	Amber	Channel A and/or B of the #1 Data Acquisition Unit (DAU) has failed.
DAU 2A-2B FAIL	Amber	Channel A and/or B of the #2 Data Acquisition Unit (DAU) has failed.
DAU A REV	White	Reversion of both Data Acquisition Units (DAUs) to Channel A is selected by the crew.
DAU B REV	White	Reversion of both Data Acquisition Units (DAUs) to Channel B is selected by the crew.
DAU ENG MISCMP	Amber	The associated (L or R) Data Acquisition Unit (DAU) has detected a miscompare between channel A and B involving an engine parameter (N_1,N_2,ITT).
DAU SYS MISCMP	Amber	The associated (L or R) Data Acquisition Unit (DAU) has detected a miscompare between channel A and B involving a system parameter (dc voltage, Emergency Bus voltage, dc amperage, Battery temperature, Main Hydraulic pressure, Brake Accumulator pressure, Oxygen temperature and pressure).
LBS/KGS CONFIG	Amber	The configuration of the integrated avionics computer is not compatible with that of the data acquisition unit (i.e., one is configured for pounds while the other for kilograms) on the ground.
LBS/KGS CONFIG	White	The configuration of the integrated avionics computer is not compatible with that of the data acquisition unit (i.e., one is configured for pounds while the other for kilograms) in flight.

The DAU circuit breakers are located in the INSTRUMENT/INDICATIONS group on each circuit breaker panel. On the left side is DAU 1 CH A and CH B, and on the right side is DAU 2 CH A and CH B. DAU 1 CH A and DAU 2 CH A are powered by the EMER BATT bus. DAU 1 CH B and DAU 2 CH B are powered by the L and R essential buses respectively.

RADIO MANAGEMENT UNITS (RMUs)

The two RM-855B Radio Management Units (RMUs) provide the central controlling functions for the entire basic radio system. Each RMU is a color, active matrix, Liquid Crystal Display (LCD) based unit. The primary function of each RMU is to select and control the frequencies and operational modes of each radio. Each RMU also provides access and storage for up to twelve pre-set channels for the VHF COM and VHF NAV functions. Cross-side operation, maintenance display, power on self-test and pilot activated self-test and optional FMS radio tuning features are also available on each RMU. Each RMU also provides backup engine and navigation display facilities in the event of EFIS/EICAS failure. Automatic presentation of engine data occurs on RMU #1 if neither IC-600 is providing EICAS data.

There are six line select keys on each side of the RMUs (Figure 5-15). The top key is referred to as a transfer button and has directional arrows on them. The remaining keys are referred to as line select keys. There are also eight function keys located at the bottom of the RMUs. The RMU main tuning page is divided into six dedicated windows. Each window groups the data associated with a particular function. The windows (COM, NAV, ATC/TCAS, ADF, and TCAS DSPY) each provide for control of both frequency and operational mode of the associated function. The RMU also has other display modes, called pages, which provide additional features and functions for the control of the radio system. The PGE function at the bottom of the RMU is used to access additional RMU pages. RMU display brightness is adjustable using the tuning knob after the DIM function key is depressed. To return the tuning knob to normal operation select any other key.



RADIO MANAGEMENT UNIT
Figure 5-15

RMUs each have a primary and a secondary power source. If the primary source is not available, the RMU will automatically switch to the secondary power source. RMU #1 primary is powered from the L essential bus, and RMU #1 secondary is powered from EMER BATT bus. RMU #2 primary is powered from R main avionics bus and the secondary from the R essential bus. RMU #1 and RMU #2 primary and secondary circuit breakers are located in the COMMUNICATIONS group on the pilot's and copilot's circuit breaker panels. RMU #2 primary is the only power source affected by the avionics master.

RMU CROSS-SIDE OPERATION

Should the pilot decide to tune the copilot's set of radios, he can push the 1/2 function key and transfer his entire RMU display and operation to the copilot's #2 system. Both RMU displays will be identical; however the pilot's RMU will show the function legends on the main tuning and memory pages in magenta to indicate that cross-control is being exercised. In addition to having access to the #2 system, the pilot still has the memory frequencies in their #1 RMU available for recall for use with the #2 system. Both pilots have this control transfer function available. It provides flexibility in crew coordinated tuning as well as a back up mode in the event one RMU becomes inoperative. The pilot may change any frequency or mode on the copilot's system using the pilot side RMU. Any changed frequency is annunciated in yellow on the copilot's RMU. The frequency will be white on the pilot's RMU. If the pilot should push the 1/2 button again, the pilot's side RMU will revert to the original display.

RMU BACKUP PAGES

Either RMU can provide two pages of backup engine and systems indication and one page for a backup navigation display. These backup displays can be selected on the PAGE MENU page of either RMU. The backup engine and systems pages can be selected by depressing the line select key adjacent to "ENGINE PG1" or "ENGINE PG2" on the PAGE MENU page. The backup navigation display is accessed by depressing the line select key adjacent to "NAVIGATION" on the PAGE MENU page.

The ENGINE PG1 and ENGINE PG2 contain information such as: ITT, O/P (oil pressure, left and right), FUEL, HYDM-B (hydraulic pressure), N_1, N_2, OIL ° C, FF PPH (fuel flow pounds per hour), VOLTS, EMER VOLTS, AMPS, OXY, SAT (oxygen quantity and static air temp), TRIM-PIT, AIL, and RUD (pitch, aileron and rudder trim).

The NAVIGATION page provides the following data, when valid data is available: NAV, ADF, CRS (selected course), DME (distance to tuned station), bearing pointers for VOR and ADF, TO/FROM indication, MARKER BEACONS, HEADING, lateral deviation (VOR and ILS), and vertical deviation (GS only). The navigation displays on both RMUs use AHRS#2 heading information and NAV information from NAV 1 and ADF 1.

VHF COM TUNING

Normal operation of the RMU is with the radio tuning page displayed. A section for COM is located in the upper left corner. The COM window displays two frequencies. The top line displays the active frequency of the COM, while the line below will display the preset frequency. When pressing the line select key (preset frequency) adjacent to the lower frequency, a yellow cursor encloses that frequency. This step is not always necessary since the cursor normally "parks" over the preset frequency box. Anytime the cursor has been moved to another area on the main radio tuning page, it will automatically return to the COM preset frequency after twenty seconds of inactivity on that page. When the yellow cursor is enclosing the preset frequency, that frequency can be changed by adjusting the tuning knobs. The preset frequency can then be changed (flip-flopped) with the active frequency by depressing the transfer key.

The storage function can be accomplished by pressing the STO (store) key located at the bottom of the RMU. When the STO key is depressed, the nomenclature below the preset COM frequency will change back to MEMORY, and the digit following MEMORY will indicate in which memory location the frequency is stored. With the main tuning page displayed, the rotary tuning knob can be used to scroll through the frequencies stored in memory. As each memory location (channel) is selected, the stored frequency will be shown on the COM preset line which can then be moved to the active position by depressing the transfer key.

A TX will appear at the top of the COM window when the associated radio is transmitting. Its purpose is to show that the transmitter is on and to alert the pilot in case of a stuck microphone key. If not attended to for approximately two minutes, a beep will sound on the audio and a MIC STK annunciation will appear at the top of the COM window until the mic button is released. Ten seconds after the MIC STK annunciation appears, the selected transmitter will automatically turn off. Depressing the SQ (squelch) function button at the bottom of the RMU causes the COM radio to open its squelch and allows any noise or signal present in the receiver to be heard. When selected, an SQ will appear at the top of the COM window. Pressing the button a second time closes the squelch.

FMS TUNING

The FMS interfaces with the RMUs for radio tuning. The FMS has a radio tuning page that can be used to control VHF COM, NAV, ADF and transponder codes. If it is suspected that the FMS is interfering with com/nav radio tuning, an FMS ENABLE/DISABLE selection on the RMU NAV memory page can be toggled with the adjacent line select key. The DISABLE selection will prevent tuning any of the radios through the FMS CDU.

NAV TUNING

The format of the NAV window (top right corner of the RMU) is identical to the COM window in that the top frequency is active and the bottom frequency is the preset frequency. Pressing the line select key alongside the NAV preselect window moves the cursor to that window. This connects the tuning knobs to the NAV preset frequency. By pressing the NAV transfer key (top right) the preset and active frequencies are exchanged. The preset frequency may be changed to a different frequency by using the tuning knobs or by pressing the line select key to bring up the next frequency from memory. Selection of stored frequencies can be accomplished by pressing the line select key by the NAV preset window until the tuning box encloses the memory mnemonic. Rotating the tuning knob scrolls through the stored frequencies displaying them in the preset area.

The memory functions and direct tuning operate the same as described under COM operation, except the NAV window has an added function called DME split tuning mode. Its operation is similar to the function called DME hold. Depressing the DME function key on the RMU allows the DME frequency to be tuned independent of the NAV frequency. Depressing the DME once causes the NAV window to split into two sections, the top one being the active VOR frequency and the lower one, now labeled "DME", containing the active DME frequency in VHF format. In this condition, the DME may be tuned directly by simply pressing the line select key to place the cursor box around the frequency and retuning using the tuning knobs. The DME digital station identifier will appear adjacent to the DME nomenclature on the top edge of the DME window. An amber H (hold) appears in the lower DME window. This indicates that the distance display (DME or TACAN) is not paired with the VOR/ILS navigation data. When the H is displayed on the RMU, it will also be displayed following the DME read-out on the PFD.

ADF TUNING

ADF operation is the same as COM and NAV tuning in that depressing the line select key beside the ADF frequency will place the cursor over the frequency to be changed. Rotating the small tuning knob slowly will advance the frequency in 0.5 kHz steps. This change will increase to 10 kHz steps when the large knob is used. The RMU has the capacity to store one ADF frequency in memory. This is done by selecting the desired frequency, then depressing the STO function key at the bottom of the RMU. To retrieve the stored frequency from memory, the frequency line select key must be depressed for two seconds. ADF modes are also controlled within the ADF window. Repetitively depressing the line select key adjacent to the ADF mode annunciator will step through the available ADF modes of operation. This can also be accomplished by placing the cursor over the mode annunciation, and using the tuning knobs to step up or down through the available modes.

The ADF operating modes are as follows:

(1) ANT (Antenna) — ADF audio signal only.

(2) ADF — ADF receives signal and calculates the relative bearing to the station.

(3) BFO (Beat Frequency Oscillator) — ADF adds a beat frequency oscillator to detect continuous wave (CW) signals.

(4) Voice — ADF has maximum audio clarity and fidelity, but no bearing information.

TRANSPONDER/TCAS TUNING

Transponder operation is similar to COM and NAV operation in that depressing a line select key beside the function desired will move the cursor to that location. Those aircraft without TCAS installed will have an ATC legend at the top of the transponder window, and those equipped with TCAS will have ATC/TCAS labeled above the window. Either transponder 1 or 2 can be selected for use and controlled by either RMU. A number 1 or 2 will appear in front of the transponder mode in the ATC window on both RMUs indicating which transponder has been selected. Transponder side selection is toggled by depressing the 1/2 key on either RMU with the cursor anywhere within the ATC window.

The transponder is switched from standby to an operating mode by depressing the line select key adjacent to the mode line. Once the cursor has been selected, the mode line select key acts as a toggle to switch the transponder between the standby mode and the active mode. Once the transponder is in the ALT ON mode, the mode of operation is changed using the tuning knobs. The active mode of operation can now be changed by rotating the concentric tuning knobs. Depressing the ID button of the RMU will initiate an approximate 18 second IDENT mode on the transponder. This will also illuminate an ID annunciation along the top edge of the transponder window. A reply annunciator is located in the upper right corner of the ATC window.

TCAS (OPTIONAL)

The Traffic Alert and Collision Avoidance System (TCAS) provides the crew with aural and visual indications of potentially dangerous flight paths relative to other aircraft in the vicinity. The system uses the transponder to interrogate other transponder-equipped aircraft and determine their bearing, range, and altitude, if the intruder has an altitude encoding transponder in operation. Advisories are issued to the crew via the airplane Primary Flight Displays (PFDs), audio system and Multi-Function Displays (MFDs) (traffic map). Two levels of TCAS are in use today, TCAS I and TCAS II. TCAS II is the same as TCAS I with the exception of providing Resolution Advisories (RAs) integrated with the vertical speed indicator on the PFDs and additional aural commands through the audio system. There is no RA display on TCAS I equipped aircraft. The TCAS system consists of a processor, two bearing antennas, and associated airplane wiring. System control is through the radio management units. Power for the TCAS system operation is 28-vdc supplied through the 5-amp TCAS circuit breaker located on the copilot's circuit breaker panel (INSTRUMENT/INDICATIONS group).

TCAS OPERATION

The Learjet 45 may be equipped with either the optional TCAS I or TCAS II system. The controls and displays are integrated with the Honeywell Primus 1000 system. Controls are through the RMUs and TCAS/annunciator displays are on the MFD and PFDs. On airplanes equipped with TCAS II, the Resolution Advisories (RA) are integrated with the vertical speed indicator display on the PFDs. The TCAS interrogates other aircraft transponders and analyzes the replies to determine range and bearing of the intruder. In addition, if the intruder's transponder is reporting altitude, the relative altitude is also determined. If the system predicts that safe boundaries may be violated, the system issues a Traffic Advisory (TA) which is displayed on the MFD. Should the TCAS II processor determine that a possible collision exists, it issues visual and audio advisories to the crew to initiate appropriate vertical avoidance maneuvers.

If an aircraft has a transponder, but does not have altitude reporting, the TCAS will depict it on the TA display, but without the altitude information tag, and without the capability of providing evasive commands. TCAS II is capable of generating a TA display of traffic from Mode A transponder-equipped aircraft, and it is also capable of generating RA signals to avoid Mode C-equipped aircraft. For similar Mode S-equipped aircraft, the airplane's TCAS II system coordinates evasive maneuvers for both aircraft. TCAS I can process transponder information from other aircraft equipped with Mode A, C, or S transponders, but does not receive altitude information to compute or coordinate a Resolution Advisory (RA). If the depicted traffic is reporting altitude and is climbing or descending at a rate of at least 500 feet per minute, a trend arrow is displayed beside the traffic symbol indicating that the aircraft is climbing or descending. If the intruder is not reporting altitude, the traffic symbol appears without an altitude tag or trend arrow. The RA displays are incorporated into the vertical speed indicator on the PFDs. Green FLY-TO zones and red NO-FLY zones are placed on the vertical speed arc by the TCAS for collision avoidance. The zones are not displayed on the arc until the TCAS detects an RA intruder and computes the collision avoidance data. Synthesized voice commands and announcements are issued by the TCAS over the airplane audio system.

SYSTEM CONTROLS AND DISPLAYS

Selection of the TA or TA/RA (TCAS II) modes is accomplished through the transponder window on either RMU main radio tuning page. After the cursor is placed over the transponder mode line, the desired mode is selected with the RMU tuning knob. The transponder selection options for TCAS I equipped aircraft will be STANDBY, ATC ON, ATC ALT and TA. Selections available with TCAS II include the same as TCAS I plus TA/RA. The selected TCAS mode will be annunciated in the top left corner of the TCAS display. The auto or manual mode can be selected on the ATC/TCAS CONTROL PAGE of the RMU. This page is accessed through the RMU PAGE MENU page. When AUTO is selected, traffic targets display only when a TA or RA target condition exists. When manual is selected, all traffic targets within the viewing airspace are displayed. In either the MAP or PLAN format display, the TCAS TA display is selected by pushing the TCAS menu key on the MFD Main Menu bezel controller. If the TCAS triggers an RA, and TCAS display is selected OFF, the main menu is activated on the MFD. This allows flight crew selection of TA displays with a single button push. This display is in addition to the resolution advisory on the VSI display on the PFD (TCAS II only).

TRAFFIC DISPLAY SYMBOLS

TCAS I will display three different traffic symbols and TCAS II four with the addition of Resolution Advisories (RA). The type of symbol selected by TCAS is based on the intruder's location and closing rate. The symbols change shape and color to represent increasing levels of urgency. The traffic symbols may also have an associated altitude tag which shows relative altitude in hundreds of feet, indicating whether the intruder is climbing, flying level or descending. A + sign and number above the symbol means the intruder is above your altitude. A - sign and number beneath indicates it is below your altitude. A trend arrow appears when the intruder's vertical rate is 500 feet per minute or greater. The symbology displayed on the PFDs and MFD is as follows:

(1) **NON-THREAT ADVISORY (OA) TRAFFIC —** An open cyan diamond indicates that an intruder's relative altitude is greater than ± 1200 feet, or its distance is beyond 6 nm range. It is not yet considered a threat.

(2) **PROXIMITY INTRUDER (PA) TRAFFIC —** A filled cyan diamond indicates that the intruding aircraft is within ± 1200 feet and within 6 nm range, but is still not considered a threat.

(3) **TRAFFIC ADVISORY (TA) TRAFFIC —** A symbol change to a filled amber circle indicates that the intruding aircraft is considered to be potentially hazardous. Depending on your own altitude TCAS II will display a TA when time to closest point of approach (CPA) is between 20 and 48 seconds. An advisory voice message "TRAFFIC, TRAFFIC" may be heard through the audio system.

(4) **RESOLUTION ADVISORY (RA) (TCAS II only) —** A solid red square indicates that the intruding aircraft is projected to be a collision threat. TCAS II calculates that the intruder has reached a point where a resolution advisory is necessary. The time to closest approach with the intruder is now between 15 and 35 seconds, depending on your altitude. The symbol appears with an audio warning and a vertical maneuver indication on the PFD VSI.

ENHANCED GROUND PROXIMITY WARNING SYSTEM (EGPWS)

EGPWS is shown on the MFD by pushing the EGPWS menu button on the MFD main menu. When TERRAIN is selected for display, the EGPWS sends the terrain data directly to the MFD display unit via the WX picture bus and replaces the WX display with terrain information.

If a potential terrain hazard is sensed by the EGPWS, terrain data automatically pops up on the MAP. This "pop-up" mode defaults to the 10 NM range. EGPWS annunciators are described in the table below and are displayed in the upper left corner of the MFD.

Annunciator	Description
TERR INHB or TERR INHIB (inhibit)	TERRAIN displays and aurals associated with terrain are inhibited (annunciation in white)
TERR FAIL	The TAWS is inoperative.
TERR TEST	EGPWS is in test mode.
TERR N/A	TERRAIN map not available.
TERR	TERRAIN map selected for display.

The terrain data is displayed above the airplane symbol on the MFD in green, yellow, and red colors that define the elevation of the terrain relative to the airplane's current altitude. Terrain that is more than 2000 feet below the airplane is not included in the display.

A moving marker scrolls across the bottom of the EGPWS display as an indication that the terrain is display is operational.

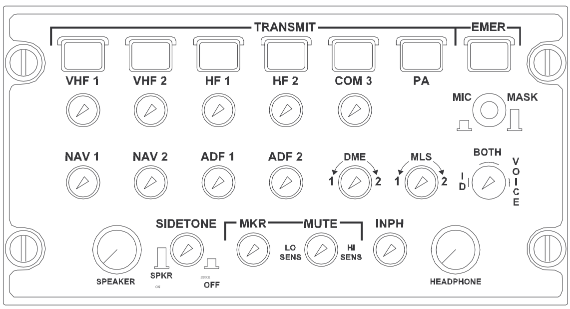
AUDIO CONTROL SYSTEM

The pilot's and copilot's digital audio control panels are located outboard of the PFDs on each side of the flight deck. Microphone transmit selector buttons are located in a row along the top edge of this panel (Figure 5-16). For night flying operations, the microphone selector buttons are annunciated with a lighted bar on the switch indicating the selected microphone. When these latching buttons are pushed, they connect the microphone (hand-held microphone, boom microphone, or oxygen mask microphone) to the selected radio. They simultaneously enable the audio associated with that radio, regardless of the setting on the audio on/off buttons located below these microphone transmit selector buttons. The microphones selector buttons are mechanically interlocked so that each new selection automatically deselects the previous selection. Depressing the PA button connects the on-side microphone to the passenger address amplifier. The audio level for the PA is automatically adjusted for conditions and cannot be adjusted by the crew. The pilot will use either a hand-held microphone or boom microphone for transmissions. Oxygen mask microphones are used when the MIC/MASK selector is in the extended (unlatched) position.

An EMER switch is located in the upper right corner of each audio control panel. When the EMER switch is depressed, the microphone and audio reception is connected directly to VHF 1 and NAV 1 and all functions of the audio control panel are bypassed except the headphone volume. In order to receive NAV 1 audio with EMER selected, the NAV AUDIO switch on the clearance delivery radio must be selected ON. When EMER is selected on the audio control panel and power is available to the control units, COM and NAV frequencies are set using either RMU or the clearance delivery radio. If EMER is selected and electrical power is still available to the audio panel, system warning audios will still be available through the cockpit speaker and audio will be routed to the cockpit voice recorder. Regardless of whether power is lost to the audio control panel, the EMER switch is operational, however, system warning audio and audio to the cockpit voice recorder are inoperative.

The audio source selector controls are located on the lower rows of the audio control panel. When these push-on/push-off switches are latched (in position) audio is turned off from that receiver. When unlatched (out position), the audio associated with that button is connected to the headphone and also to the speaker, if it is selected on. The audio level can be adjusted by rotating the button, counterclockwise to decrease, and clockwise to increase the volume.

One knob, labeled DME, controls the audio reception for both DME 1 and DME 2. When the DME knob is unlatched (out position) and the arrow on the knob is centered straight up, the audio level is at a minimum. Rotating the control knob in either direction toward 1 or 2 will increase the volume for that corresponding channel only. The audio level pointers on the knobs are displayed for night flight. There are separate controls for speaker volume and headphone volume which adjust the volume level for all audio buttons selected. The speaker push-on/push-off selector is combined with the sidetone knob. When the speaker switch is extended, it turns on audio to the on-side speaker. The speaker sidetone audio is controlled by the speaker SIDETONE volume control and the SPEAKER volume control for both on-side and off-side transmit conditions.



DIGITAL AUDIO CONTROL PANEL
Figure 5-16

The ID/BOTH/VOICE switch is located on the right side of the audio panel. In the ID position, the VOR and ADF audio is filtered to enhance the Morse Code identification and eliminate the voice signal. In the VOICE position, the ident audio is filtered to pass the voice content only and in the BOTH position, voice and ident signals will be heard simultaneously.

The controls for the marker beacon receiver are located at the bottom of the audio panel. They include the marker audio volume control (MKR), marker sensitivity control (LO SENS/HI SENS) and marker mute control (MUTE). The sensitivity is controlled by the rotation of the MUTE control. If either audio panel MKR sensitivity control is set LO, then both MKR receivers are set to LO, regardless of the position of the other audio panel controls. Either pilot can temporarily mute the marker beacon receiver by depressing the MUTE/HI/LO switch.

The INPH (interphone) volume control adjusts the on-side headset audio level when the interphone function is used. The interphone operates on a "hot microphone" basis. The interphone is not available over the cockpit speaker except when the oxygen mask audio is selected. The MIC/MASK control allows for microphone audio switching between the boom/hand-held microphone (MIC) and the oxygen mask microphone (MASK). When the switch is latched (depressed position), MIC is selected and when the switch is unlatched (out position), MASK is selected.

The MASK intercom feature provides interphone audio to the on-side cockpit speaker. Audio is available regardless of the SPKR ON/OFF button position. Selecting INPH allows adjustable volume control of the off-side MASK intercom on the speaker.

Warning system audio signals are input to the audio panel for dissemination to the flight crew over the headphones and speaker. The audio output from the headphone, speaker, and microphone are recorded by the Cockpit Voice Recorder (CVR). The CVR microphone is the input for the AGC circuit and if the CVR microphone becomes disabled, or the CVR circuit breaker is pulled, then the aural warnings will be at the fixed HIGH volume level. If the Crew Warning Panel (CWP) has detected a fault in any one of the audio output channels, or in the Automatic Gain Control (AGC) input, a CAS annunciation will be posted.

The following CAS illuminations are specific to the Crew Warning Panel audio:

CAS	Color	Description
WARN AUDIO	Amber	The audio function of the Crew Warning Panel (CWP) has failed.
WARN AUDIO	White	• A Crew Warning Panel (CWP) audio output channel fault is detected. or • A problem exists with the Automatic Gain Control (AGC).

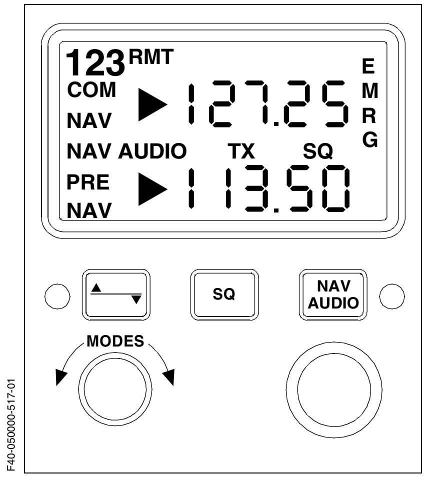
The pilot's audio panel receives 28-vdc from the left essential bus and is protected by a 3-amp circuit breaker labeled AUDIO 1/CLR DLY on the pilot's circuit breaker panel (AVIONICS [COMMUNICATIONS] group). The copilot's audio panel receives 28-vdc from the right essential bus and is protected by a 3-amp circuit breaker labeled AUDIO 2 on the copilot's circuit breaker panel (AVIONICS [COMMUNICATIONS] group). The passenger address amplifier receives power from the left essential bus and is protected by the CABIN PA 5-amp circuit breaker on the pilot's circuit breaker panel (AVIONICS [COMMUNICATIONS] group).

CLEARANCE DELIVERY RADIO (CDR)

The Clearance Delivery Radio (CDR), Figure 5-17, is located on the right, front corner of the center pedestal. The CDR provides an alternative capability for tuning the VHF COM 1 transceiver and the VHF NAV 1 radio. The CDR will tune the VHF COM radio prior to applying electrical power to the airplane. The CDR control head is normally powered by the left essential bus through the AUDIO 1 circuit breaker; however, it and other communication related equipment can be powered from the right forward hot bus, prior to applying electrical power to the aircraft.

With airplane batteries OFF, depressing the momentary action RADIO CTL HOT BUS switch on the center pedestal applies power from the right hot bus to the left audio control panel, CDR control panel, COM section of the integrated communications unit, and NAV section of the integrated navigation unit. The display on the CDR is liquid crystal type with white letters on a black background. The push button, display, and control identifier legend are on a black background displayed with electroluminescent lighting.

In the emergency mode, RMU and FMS tuning capabilities are inhibited and the COM and NAV units are tuned exclusively by the CDR. An AUX ON caption replaces the NB or WB annunciators on the RMUs to indicate that tuning through the RMUs is inhibited. Tuning through the CDR is no different when EMRG is selected, but the CDR does not look at the radio bus data to check the echoed frequency.



CLEARANCE DELIVERY RADIO
Figure 5-17

The CDR controls are as follows:

- **Transfer Key —** Alternately selects either the COM (top) or NAV (bottom) frequency to be connected to the tuning knobs.
- **Tuning Knobs —** Used to change the frequency indicated by the tuning cursor.
- **Normal/Emergency Mode Switch —** This rotary knob provides alternate selection of Normal and Emergency modes.
- **NAV AUDIO On/Off Switch —** This alternate action push button switch is used to toggle NAV audio ON or OFF when in the EMER audio mode on the audio control panel.
- **Squelch (SQ) Switch —** Used to toggle COM squelch on or off.

FLIGHT GUIDANCE CONTROL SYSTEM

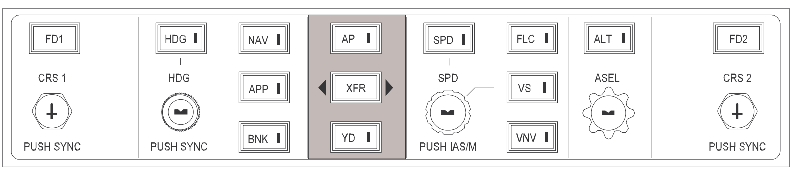
The Primus 1000 system includes an autopilot, yaw damper and dual synchronized flight directors. These are all co-located in two IC-600 display flight guidance computers located in the nose avionics bay. Each IC-600 houses a flight director; however, only the copilot IC-600 is connected to the pitch, roll and yaw servos for the autopilot/yaw damper and rudder boost functions. A flight guidance controller, located in the center of the glareshield, provides the means of engaging the autopilot/yaw damper and controlling both Flight Director (FD) systems. It also contains a transfer (XFR) switch that allows the crew to select either the left or right flight director as master and for autopilot coupling. The autopilot is a single channel with a fail passive design. The monitor system provides for automatic disconnect in the event of a malfunction in autopilot, yaw damper, or rudder boost. All automatic disconnects, which result from monitor trips, will be stored in a non-volatile memory for later recall by technicians.

FLIGHT DIRECTOR

The flight director system, utilizing two separate IC-600 computers, provides dual flight director computations, either of which can be coupled to the autopilot. Only one flight director can be coupled to the autopilot at a time. In this case the coupled flight director is classified as the master flight director, and the other flight director is classified as the slave flight director. The flight director can couple to Short Range Navigation (SRN) units (e.g. VOR, ILS), dependent upon which SRN is being displayed as a NAV data source on the Primary Flight Display (PFD). The flight director will use the displayed SRN data on the associated PFD for command-bar control computations. The flight director can be coupled to optional Long-Range Navigation (LRN) units (e.g. flight management system) if it is installed and selected as the NAV data source and displayed on the associated PFD. The flight director utilizes the lateral and/or vertical steering commands from the LRN in the command control computations. Each flight director uses the displayed on-side Air Data Computer (ADC) data for all vertical modes and gain programing. The flight director modes are synchronized in a manner that allows selection of the modes to be accomplished by the single set of FD mode select buttons on the flight guidance controller (FGC), see Figure 5-18. The FD mode select buttons on the FGC panel are momentary action and each has a vertical green bar that illuminates whenever the mode is selected.

FLIGHT GUIDANCE CONTROLLER (FGC)

The Flight Guidance Controller (Figure 5-18) is the prime controller for the flight director and the autopilot/yaw damper. Located on the center glareshield, the FGC provides the means, via push button switches, for flight director mode selection, couple status and autopilot/yaw damper engage selection. Flight director modes engage status is indicated to the crew via a green light on the right edge of each mode switch, which is illuminated when the mode is active and extinguished when the mode is inactive or dropped. The controller also has several rotary controls enabling selection of IAS, MACH, and VS targets, altitude, course and heading. All push button selections are signaled to both IC-600s via a discrete output from the FGC. The IC-600s then provide the drive to illuminate the appropriate light on the FGC.



FLIGHT GUIDANCE CONTROLLER
Figure 5-18

The FGC annunciations and controls are as follows;

FD 1/2 buttons — The flight director buttons (FD 1 and FD 2) are located on the upper left/right corners of the flight guidance controller. Depressing these buttons alone will not bring the FD command bars into view. Any FD mode selection causes the FD command bars to appear on both PFDs. When the FD command bars are in view on both PFDs and the autopilot is not engaged, depressing the master side FD button will disengage all FD modes and remove the command bars from both sides. Pressing the slave side FD button will remove the command bars from the PFD on that side acting as a flight director clear function.With the autopilot engaged, the FD command bars will be in view at all times on the coupled side and cannot be removed from the PFD. The opposite side FD command bars can be removed from view by depressing the appropriate FD 1 or FD 2 button.

Course set knobs — A course set knob (CRS 1 and CRS 2) is located at each end of the FGC. These knobs are used to individually set the courses on the left and right PFD HSI displays. They are used primarily to set the course for a VOR radial or LOC course. The course knobs have a push button in the center to synchronize the display to the aircraft's "direct-to" course.

Heading set knobs — Heading is selected via a rotary knob, with a "Heading Bug" symbol on the face of the knob. The heading knob controls the heading bug and digital display on both PFDs and the bug on the MFD MAP display. Depressing the HDG knob will synchronize the HDG bug on all display units to the aircraft's current heading.

HDG (heading) button — Depressing the HDG button engages the heading mode and displays a green HDG annunciation on the PFDs. The flight director command bars will command a turn in the direction the heading bug was moved to achieve the set heading. Heading select is used to maintain a magnetic heading. The heading bug is positioned to the desired heading on the HSI using the HDG knob on the FGC. The heading select mode is canceled when any armed lateral mode captures or if GA is selected.

NAV (navigation) button — Pressing the NAV button alternately selects and deselects the navigation mode. The NAV mode is normally used to intercept route segments identified with VOR radials and to intercept and fly desired FMS tracks (SIDs, routes, holding and STARS).

APP (approach) button — The intended function of the APP mode is that APP be used for all approaches, regardless of nav source or whether a vertical mode is also associated with the approach. The APP mode is normally used to select lateral and vertical steering for ILS and FMS. The VOR approach mode is selected by pressing the APP mode button with the navigation receiver tuned to a VOR frequency and selected as the active nav source. Pressing the APP button arms both localizer and glideslope modes when the navigation receiver is tuned to an ILS frequency and ILS is selected as the active navigation source. Selection of APP mode when the nav source is FMS engages the FMS lateral mode the same as described for NAV and also arms VNAV for approach.

BNK (bank) button — Pressing this button alternately selects or deselects a reduced maximum bank angle of 14° (for all lateral modes, except roll) on both FDs. When selected, a green low bank arc appears on the top of the ADIs and BNK is annunciated on the PFD.

AP (autopilot) button — Depressing this button engages the autopilot. Depressing a second time disengages the autopilot.

XFR (transfer) button — Located in the center section of the FGC. The XFR button is used to select the desired flight director (left or right) to command the autopilot.

YD (yaw damper) button — Depressing this button engages yaw damper. The YD can be engaged independent of the AP, but the autopilot system will not engage, or remain engaged, without the YD.

SPD (speed) knob — The rotary SPD knob is used to change the IAS/Mach speed reference (SPD mode) and the vertical speed reference (VS mode). The speed knob changes the bug airspeed at any time, as long as VS is not selected. When VS mode is engaged, rotation of the SPD knob changes the digital vertical speed reference and the vertical speed bug position. The integral PUSH IAS/M button within the SPD knob is used to toggle the airspeed tape between IAS and Mach. The master flight director computes the airspeed reference, and the slave flight director synchronizes to this reference.

SPD (speed) button — Depressing the SPD button engages the speed hold mode (IAS or Mach) on both FDs. The speed select mode is used to fly to a selected airspeed or Mach number, and to provide limited overspeed/underspeed protection during climbs and descents. When speed select mode is active, a green IAS or Mach annunciation is displayed in the captured vertical mode field on the PFDs.

FLC (Flight Level Change) button — Depressing the FLC button once engages the normal climb/descent profile on both PFDs. Depressing it a second time selects the high speed climb/descent schedule. A third depression deselects the mode. The FD chooses between the climb and descent schedule based upon the aircraft's present altitude and preselected target altitude. The FD annunciation on the PFD is FLC for the normal profile and FLCH for the high speed profile.

VS (Vertical Speed) button — Depressing the VS button engages the vertical speed hold mode on both FDs. When VS is selected, the speed bug disappears and reference goes to dashes.The FD commands pitch changes to hold the vertical speed that existed at the time of engagement. Once engaged, the vertical speed bug positions on the inner side of the vertical speed scale and a digital readout appears above the vertical speed indicator.

VNV (Vertical Navigation) button — Depressing the VNV button arms, then captures the FMS pitch steering commands of the FDs if FMS is selected as the NAV source, the FMS is programmed for a vertical navigation profile, the altitude preselector is set below existing altitude and the aircraft is within the TOD (top of descent) window. When the VNAV mode is armed, a white VNAV is annunciated on the PFDs in the FD vertical mode annunciation field and will turn green upon capture.

ASEL (Altitude Select) knob — The preselected altitude is set via the ASEL rotary knob on the FGC. The altitude preselect mode provides a means for FD/AP to climb or descend to a preselected altitude and then level off and maintain the preselected altitude. The ASEL knob is used to set the altitude preselect function, and also provides the altitude reference for the altitude alerter function.

ALT (Altitude Hold) button — Altitude hold may be engaged by depressing the ALT button on the FGC. When ALT is engaged, the FD commands pitch to hold the existing altitude at the time the ALT button was depressed, or at the ASEL reference altitude if ALT automatically engages.

AUTOPILOT/YAW DAMPER

The autopilot is a single-channel autopilot which may be coupled to either flight director. The autopilot function is contained within the #2 IC-600 located in the nose. The IC-600 will fly the aircraft based on the selected control mode and guidance inputs from the coupled flight director, via servo control of the elevator, aileron and rudder. When engaged, the autopilot will also automatically command trim changes as required to alleviate the aerodynamic loading on the elevator (pitch trim), and will allow control surface commands to be entered via the control wheel trim switches for the ailerons and elevator. There is also a Touch Control Steering (TCS) feature which allows the cockpit crew to manually maneuver the aircraft with the autopilot engaged when the TCS switch is pressed. The autopilot provides aircraft control in response to pitch and roll steering commands from either flight director.

The yaw damper software computes servo commands based on sensor input data only. the yaw damper control software provides yaw rate damping that holds rudder force to zero. The servo position reference is synchronized to zero at engagement and is constantly washed out to ensure that the steady-state rudder forces are zero. The yaw damper can be engaged independent of the autopilot but the autopilot cannot be engaged without the yaw damper.

AP/YD Annunciation

At the very top, center section of the PFDs is an area dedicated to flight director and autopilot annunciations. A horizontal arrow appears at the top center of each PFD between the flight director vertical and lateral mode annunciators. This arrow points left or right, as selected on the FGC XFR switch, to indicate which flight director the autopilot will couple to when engaged. It also indicates which flight director has priority. Just below the arrow is a line reserved for autopilot and yaw damper annunciation. A green AP and YD appear in this area when the autopilot and/or the yaw damper is /are engaged.

Control Wheel Trim Switch

Manual primary pitch trim or roll trim commands are initiated by pressing and holding of the arm switch and actuation of the control wheel trim switch. Pressing and holding of the arm switch with control wheel trim switch input will result in immediate autopilot disengagement.

The yaw damper and flight director modes are not affected by manual pitch or roll trim commands. Actuation of the control wheel trim switch without pressing the arm switch allows manual autopilot commands. During autopilot engagement, basic attitude commands (pitch and roll) can be entered through either the pilot's or copilot's control wheel trim switch (dependent upon which flight director is coupled to the autopilot). With the autopilot engaged, activation of the control wheel trim switch (without arm switch depressed), on the side coupled to the autopilot, causes the flight director roll and/or pitch hold mode to be activated.

Control Wheel Master Switch (MSW)

The MSW switch immediately disconnects all autopilot and yaw damper servo drives. The selected flight director modes are not affected. In normal operation, a 28-vdc signal is routed through the normally closed contacts of each MSW and then to the onside IC-600. This input to the IC-600s is the autopilot disengage discrete, and if 28-vdc is removed from this discrete on either IC-600, the autopilot will disengage.

Touch Control Steering (TCS)

TCS allows the pilot to manually fly and retrim the aircraft without disengaging the autopilot. To use the TCS function, the pilot will press the TCS button on the control wheel, maneuver the aircraft to the desired condition, and release the TCS. Operation of the TCS button has no effect on flight director mode of operation. While the TCS button is held depressed (AP engaged), a white "TCS ENG" annunciator appears in the autopilot display area at the top of the PFDs.

Autopilot Engagement/Disengagement

Engagement of the autopilot is achieved via the AP push button on the FGC. Each button has a vertical green bar that illuminates when engaged. Engagement of the autopilot will cause the yaw damper to automatically engage. When engaged, the autopilot will couple to the master flight director, and will follow the guidance commands from the master FD. If no flight director is active, engagement of the autopilot will cause the master FD to default to the basic PIT and ROL modes. When engaged, the appropriate annunciation will be provided on both PFDs, and the green bar on the AP push button will be illuminated. Disengagement of the autopilot via the AP switch will cause the AP annunciation to be removed from both PFDs and the green bar on the AP push button will extinguish. Other actions that will cause autopilot disengagement include:

(1) Control wheel master switch activation.
(2) Pilot initiated trim commands with control wheel trim switch depressed.
(3) Yaw damper disengagement.
(4) AP switch on the FGC disengaged.
(5) Autopilot primary or secondary monitor trip.

For a normal disconnect, AP flashes red for 5 seconds, then is removed. For a monitored disconnect, it flashes red for 5 seconds, then steady, and the aural alert is continuous until the crew cancels it with the MSW.

Yaw Damper Engagement/Disengagement

Selection of the autopilot via the AP push-button will automatically engage the yaw damper. Alternatively the yaw damper may be selected via the YD push button on the FGC. When engaged, the green YD annunciation will be provided on both PFDs, and the green bar on the YD push button will illuminate. Manual disengagement of the yaw damper via the YD button will cause the YD annunciation to flash amber then be removed from both PFDs and the green bar on the YD push button will extinguish. In the case of a monitored disconnect, the YD annunciation will turn amber and flash for five seconds and then remain steady. Other actions that will disengage yaw damper include:

(1) Control wheel master switch activation.
(2) YD Switch on the FGC disengaged.
(3) Yaw damper monitor trip.

Mistrim Annunciation

When the autopilot is engaged and the roll or pitch servo remains energized for a longer than normal period, this condition will be annunciated with a CAS. The autopilot does not have a capability to trim in the roll axis; therefore, if there is a mistrim in the roll axis, this will also display a CAS.

The following CAS illuminations are specific to the autopilot:

CAS	Color	Description
AP AIL MISTRIM	Amber	Autopilot is engaged and autopilot aileron servo is holding excessive torque. Disengage autopilot.
AP ELEV MISTRIM	Amber	Autopilot is engaged and autopilot elevator servo is holding excessive torque. Disengage autopilot.
AP ELEV MISTRIM	White	Autopilot is engaged and autopilot elevator servo is holding torque. Disengage autopilot.

Power Supply Configuration

The power supply for all the AFCS servos is supplied from the L MAIN bus, and is routed through the #2 IC-600 to the servos. The system is designed so that power failure to any major component in the system will result in the system reverting to the safe, disconnect mode. The AFCS SERVOS circuit breaker is located in the FLIGHT group of the pilot's circuit breaker panel.

GO-AROUND (GA) BUTTON

The GA button is located on the outboard side of the left thrust lever. Selection of the GA button disconnects the autopilot (but not the yaw damper) and cancels all other vertical and lateral modes except automatic altitude preselect arm. The flight director provides a wings level command bar display in the lateral axis and a fixed pitch-up vertical command. The pitch command does not guarantee that the go around airspeed will be achieved. If used on takeoff, the pitch attitude will not guarantee achieving the V2.

FLIGHT MANAGEMENT SYSTEM (FMS)

The UNS-1E is a fully integrated Flight Management System (FMS). The FMS provides centralized navigation sensor control, flight planning, lateral and vertical flight guidance, steering enroute, terminal and approach modes of operation. Database management, fuel management, and maintenance functions are also provided by the FMS.

The UNS-1E accepts position information from up to five long-range navigation sensors as well as DME, VOR, or TACAN sensors. The data from these sensors is used to determine the best computed position. The UNS-1E incorporates an internal GPS sensor with Receiver Autonomous Integrity Monitoring (RAIM) as a standard part of the FMS configuration. The FMS interfaces with the IC-600s for a transfer of information to the FMS and lateral and vertical steering commands back to the EFIS and FD/AP.

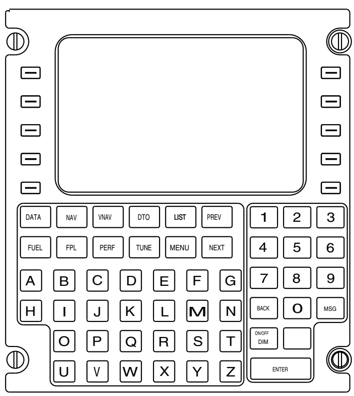
If a single FMS is installed, it can receive ADC, AHRS, EFIS, and AP data from either IC-600 but will use the #1 IC-600 as the primary source. If dual FMSes are installed, FMS 1 will use the #1 IC-600 as the primary source and FMS 2 will use #2 IC-600 as its primary source. With dual FMS installation, the pilots can use either FMS as the navigation source on their PFDs/MFD. In the material that follows, the FMS unit will be referred to as the Control Display Unit (CDU). The CDU contains a color, flat panel display, ten Line Select Keys (LSK) and dedicated function keys.

The FMS is powered from the left essential bus by a 5-amp circuit breaker labeled FMS located on the pilot's circuit breaker panel (AVIONICS [NAVIGATION] group).

CONTROL DISPLAY UNIT (CDU)

The CDU (Figure 5-19) is the primary interface to the pilot. It provides keypad input for selection of NAV modes and entering of waypoints and a display to indicate current operational modes. The CDU contains all the components required for the FMS functions. Other functions that are provided include:

- **Radio tuning and RMU interface** — The CDU allows remote tuning of the VHF COMM, VOR/ILS, ADF and ATC radios.
- **Flight planning** — The CDU can store a fixed number of flight plans into the FMS database.



UNS-1E FMS CDU
Figure 5-19

DATA TRANSFER UNIT (DTU)

The DTU allows updating of the FMS database, storing of pilot data and can also be used to record flight data. Database updates can be obtained on a subscription basis. The data transfer unit is a drive (e.g. 3.5-inch floppy or zip disk) designed for mounting in the aircraft.

CONFIGURATION MODULE

The configuration module is used to store configuration data that is specific to the aircraft in which the FMS is installed.

At start-up, the CDU checks the version number stored in FMS memory against the version number stored in the configuration module. Any discrepancy between version numbers results in an FMS message, "CONFIG UPDATE REQUIRED", on the CDU.

FMS FUNCTIONS

FLP (flight plan) — Before using the FMS for navigation, an active flight plan must be defined within the FMS. The operator may select a previously stored route or create a new one to load as the active flight plan. A route or active flight plan can be created on the FMS as well. Once the departure airport is identified, the UNS-1E will present tailored lists from which the current runway, SID and transition can be selected. Also, both low and high altitude airways can be accessed for route or flight plan creation using the LIST function. Routes or the flight plan may also be constructed waypoint by waypoint, or by combining stored route segments. When en route and nearing the destination, a progression of smart prompts similar to those used on departure may be utilized to input a STAR, the approach and landing runway.

Upon selection of NAV on the MFD (FMS MENU), the IC-600 will display the closest eight navaids (VORs or NDBs) received from the FMS on the MFD MAP as background data. Selection of APT on the MFD, will result in the IC-600 displaying the first four airports received from the FMS on the MFD MAP as background data.

FUEL — Before takeoff, the fuel quantity signal conditioner provides the FMS with fuel quantity on-board. The pilot must accept or change the transmitted fuel quantity on the CDU fuel page. After engine start, the FMS receives real time fuel flow information independent of the aircraft indicating system. APU fuel is not included in fuel flow. Specific range and endurance are provided along with fuel, time and distance predictions for the destination.

PERF (performance) — A performance program in the FMS can compute takeoff speeds, takeoff N_1 , takeoff distance and landing speed. The operator enters pertinent data such as takeoff configuration and environmental conditions mostly through menu selections. V Speeds and balanced field data is calculated and displayed. For landing, Vref is calculated along with approach speeds for different flap settings.

NAV (navigation) — All pertinent en route navigation data is displayed on the first NAV page of the CDU. This page along with the PFD/MFD displays provide complete integrated real time information on flight progress.

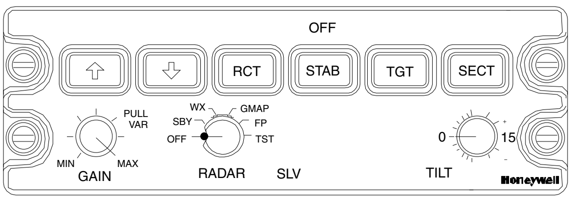
DTO (direct to) — A dedicated function key can be used to navigate from present position directly to any point on or off the present flight plan.

VNAV (vertical navigation) — Vertical navigation pages allow the operator to define waypoints with altitudes or flight levels. Features such as computed top-of-descent, target vertical speed and vertical direct-to are included. The FMS outputs vertical commands to the flight director when selected.

HOLDING PATTERNS — Holding patterns may be programmed at any waypoint on or off the flight plan or stored in the navigation data base as part of a SID, STAR or approach procedure. The holding pattern page provides a graphic depiction of the holding pattern. The pattern is defined with some crew inputs, and when the ACTIVATE line select key is pressed, the aircraft will proceed from its present position directly to the fix and make the appropriate entry (direct, parallel or teardrop), all automatically calculated and flown by the FMS.

WEATHER RADAR

The Primus 650 is the standard radar installed in the Learjet 45. It is an X-band radar system designed for weather detection and analysis. The radar can also be used for ground mapping. The WU-650 is an integrated unit which incorporates the receiver, transmitter and antenna (RTA) into a single unit, located in the nose of the aircraft. The only remaining radar component is the cockpit control panel which is mounted on the center pedestal. The antenna is a 12-inch flat plate that is stabilized with inputs from the #1 AHRS. Weather patterns can be displayed on both PFDs, on the ARC or MAP format, and on the MFD MAP format. The radar generates high-level RF pulses and should be operated with caution while on the ground. When operating on the ground, position the nose of the airplane so that the antenna scan sector is free of large metallic objects such as hangers or other aircraft for a distance of at least 100 feet.



WU-650 WEATHER RADAR CONTROL PANEL
Figure 5-20

The weather radar is controlled from the WC-650 weather radar controller, Figure 5-20. WX is selected for display on the PFDs by selecting the WX button on the display controller, the HSI automatically switches to the ARC mode. WX is selected for display on the MFD MAP by depressing the WX bezel button on the MFD main menu.

A six-position rotary switch is provided on the radar control panel for selecting the different radar modes. They are:

- **OFF** — removes electrical power from the system.
- **SBY (standby)** — in this position the RTA is powered up but does not radiate any RF energy nor does the antenna scan.

- **WX (weather)** — selects the weather radar main operating mode.
- **GMAP (ground map)** — in the ground mapping mode, the system internal parameters are set to enhance returns from ground targets.
- **FP (flight plan)** — selecting this position places the radar in the flight plan mode.
- **TST (test)** — when this mode is selected the weather depiction will be a special colored test pattern to allow verification of system operation.

Power is provided to the RTA and cockpit controller from the right avionics main bus with a 7.5-amp circuit breaker located in the INSTRUMENT/INDICATIONS group of the copilot's circuit breaker panel.

AVIONICS COOLING

INSTRUMENT PANEL COOLING

The instrument panel cooling system is located forward of the throttle quadrant and is provided to draw flight deck ambient air through the instrument panel and thus prevent overheating of avionics displays and instruments. The system consists of an avionics cooling fan, an on/off thermostat switch and an overtemperature thermostat circuit. The avionics cooling fan is activated when the temperature sensing switch reaches 90° F (32° C). The fan automatically turns off when the temperature has been reduced below 70° F (21° C). If the temperature reaches an extreme of 135° F (57° C) the overtemperature circuit is energized and an annunciation is posted on the CAS. The CAS remains displayed until the temperature has been reduced to 125° F (51.7° C).

The following CAS illumination is specific to the avionics cooling system:

CAS	Color	Description
INSTR PNL TEMP	White	The temperature at the instrument panel is higher than normal.

The instrument panel cooling system is powered from the L MAIN bus and protected by the INSTR FAN 3-amp circuit breaker located on the pilot's circuit breaker panel (ENVIRONMENTAL group).

MISCELLANEOUS

COCKPIT VOICE RECORDER (CVR)

A solid state Cockpit Voice Recorder (CVR) is installed in the Learjet 45. The standard CVR is a three-channel unit providing 30 minutes of recording. An optional unit is available which provides 120 minutes of recording. Two of the channels are used to record pilot and copilot audio. The third channel is used for the area microphone. Located in the tailcone, the CVR is painted international orange with reflective tape added to aid in recovery following a mishap. It also has an underwater locator beacon installed on one end of the unit. The recording is converted to a digital format and stored in crash protected memory. The area microphone is located in the upper center area of the instrument panel. An erase button and headphone jack are located on the CVR panel just beneath the copilot audio control panel. The CVR performs a self-test at power-up and has a continuous self monitor. If a fault is detected at any time, an annunciation is posted on CAS.

The following CAS illumination is specific to the CVR:

CAS	Color	Description
CVR FAIL	White	The cockpit voice recorder has failed.

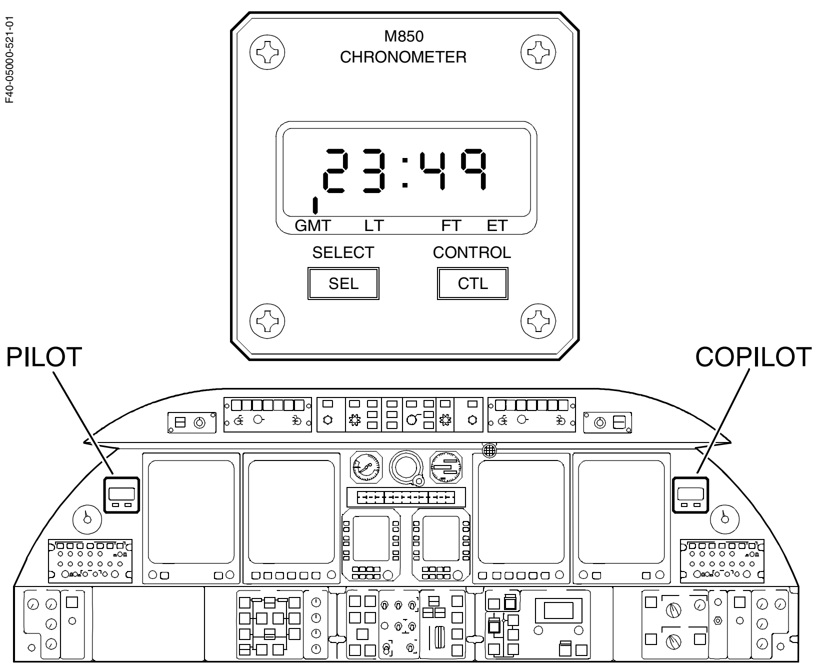
The erase function is initiated by pressing the erase button on the CVR panel. An interlocking device only allows this function to work when the airplane is on the ground and parking brake is set. When erase function is complete, a three-second tone is output to the headphone jack.

Voice recorder system power is 28-vdc supplied through a 3-amp CVR circuit breaker located on the pilot's circuit breaker panel (INSTRUMENTS/INDICATIONS group).

CLOCKS

Each instrument panel is equipped with a multi-function chronometer to display GMT, local time (LT), flight time (FT), and elapsed time (ET). The SEL button selects what is to be displayed and the CTL button controls what is being displayed (Figure 5-21). Pressing SEL sequentially selects GMT, LT, FT or ET for display. FT starts counting when the main gear weight-on-wheels switches transition to the air mode and stops counting when they transition back to ground mode. The CTL button resets FT back to zero when held down for three seconds. ET is started and reset when the CTL button is pushed momentarily. Depressing the SEL and CTL buttons simultaneously enters the set mode and GMT or LT can be set. The CTL button is then pressed to increment the flashing digit to the desired value. Pressing the SEL button enters that value and toggles to the next digit to be set.

Power for the chronometers is 28-vdc supplied through a 1-amp L and R CLOCK circuit breaker located on pilot's and copilot's circuit breaker panels respectively (INSTRUMENT/INDICATIONS group).



MULTI-FUNCTION CHRONOMETER AND INSTRUMENT LOCATION
Figure 5-21

HOURMETER-AIRCRAFT (OPTIONAL)

An optional hourmeter may be installed to measure aircraft accumulated time. The typical location for the hourmeter is on the RH side of the pedestal, just aft of the SELCAL decoder. It is wired to the right hand main gear weight-on-wheels switch, through a switch in the lower entry door frame. It will measure accumulated time as soon as the aircraft lifts off. The hourmeter receives 28-vdc from a 1-amp HOURMETER circuit breaker located in the INSTRUMENTS/INDICATIONS group of the copilot's circuit breaker panel.

FLIGHT DATA RECORDER (FDR) (OPTIONAL)

The flight data recorder, which may be installed, is a 25-hour Solid-State Flight Data Recorder (SSFDR) with Underwater Locator Beacon (ULB) and remote mounted tri-axial accelerometer.

The following CAS illumination is specific to the flight data recorder:

CAS	Color	Description
FDR FAIL	White	The flight data recorder has failed.

EMERGENCY LOCATOR TRANSMITTER (OPTIONAL)

Dorne & Margolin ELT14

The Dorne & Margolin ELT14 system simultaneously transmits distress signals on the frequencies of 121.5 and 243.0 MHz. The system will automatically activate under emergency conditions or may be manually activated with a cockpit-mounted switch. The system consists of a transmitter, antenna, remote control and monitor unit, and associated airplane wiring.

TRANSMITTER AND ANTENNA

The transmitter and antenna are installed in the airplane tail section. Power for the transmitter is provided by an internal battery pack. The transmitter incorporates a three-position switch (ARM/OFF/ON). Access to the transmitter is through an access cover placarded "ELT LOCATED HERE." The antenna is externally mounted and connects to the transmitter with antenna cable.

Transmitter Switch (ARM/OFF/ON)

Because of its location, this switch is not generally used by the crew. In the OFF position, the transmitter will not transmit distress signals. This position is normally used only while servicing the airplane. In the ON position, distress signals will be transmitted continuously. In the ARM position, the transmitter will automatically activate if the airplane stops abruptly. The switch should be in the ARM position for flight.

REMOTE CONTROL AND MONITOR UNIT

The remote control and monitor unit is installed in the cockpit. Power for this unit is provided by an internal coin cell. A three-position ON/ARM/RESET switch provides the remote control for the ELT transmitter. The ON and ARM positions function the same as described for the transmitter switch. Once activated, the transmitter may be returned to an armed status using the RESET function. The ELT can be reset but not switched off from this control unit.

A red LED, mounted in the end of the switch handle, provides the crew with the ELT status. The LED indicates ELT status as follows:

LED is:	ELT Status:
On continuously	The ELT is transmitting.
Flashing slowly (80 times per minute)	The ELT transmitter is switched OFF or the transmitter battery needs replacement.
Flashing quickly (5 times per second)	The remote control/monitor unit coin cell needs replacement.
Extinguished	The ELT is armed.

OPERATION

To arm the transmitter for automatic activation the ON/ARM/RESET Switch is placed in the ARM position. If the red LED flashes slowly, check that the transmitter switch is in the ARM position. If the transmitter switch is in the ARM position and the LED continues to flash, the transmitter battery needs replacement. To manually activate the transmitter, place the ON/ARM/RESET Switch to the ON position and check that the red LED is on continuously. To reset the transmitter, momentarily place the ON/ARM/RESET Switch to RESET and check that the red LED extinguishes.

ARTEX ELT 110-406

The ARTEX ELT 110-406 transmits on 121.5, 243.0 and 406.025 MHz. The ELT may be manually activated with a cockpit mounted switch or will automatically activate during an impact. Once activated, the ELT transmits the standard swept tone on 121.5 and 243.0 MHz. During that time the 406 MHz transmitter turns on and an encoded digital message is sent to the satellite. The information contained in the message includes:

- Serial number of the transmitter.
- Country code.
- Manufacturer.
- Position coordinates (optional).

The information sent to the satellite is programmed at the factory and contains a unique number that can be used to identify the beacon. The ELT 110-406 system consists of a transmitter, antenna, cockpit switch and indicator light, buzzer (aural monitor), and associated airplane wiring.

TRANSMITTER AND ANTENNA

The transmitter and antenna are installed in the airplane tail section. Power for the transmitter is provided by an internal battery pack which consists of 4 D size lithium manganese dioxide cells connected in series. The ELT unit incorporates an ON-OFF switch. Because of its location, this switch is not generally used by the crew. Access to the transmitter is through an access cover placarded "ELT LOCATED HERE." The antenna is externally mounted and is connected to the transmitter using antenna cable.

Transmitter Switch (ON-OFF)

In the ON position, the transmitter will transmit distress signals continuously. In the OFF position, the transmitter is armed to activate either automatically (impact) or manually (remote control from the cockpit switch). If removed from its mounting rack, the transmitter will be deactivated.

COCKPIT SWITCH AND INDICATOR LIGHT

The ELT 110-406 remote control (cockpit panel switch) provides manual On, Armed, and Reset modes. A 28-vdc indicator light, powered through the ELT WARN circuit breaker on the options circuit breaker panel in the tailcone, flashes continuously if the ELT has been activated and is transmitting.

BUZZER

The buzzer (aural monitor) provides a distinct signal (loud, siren-type sound) enabling a search and rescue team to locate an aircraft with a transmitting ELT in a confined area with a large number of aircraft (such as an airport). The buzzer is installed in the tail section and is powered by the ELT battery pack. The buzzer does not operate continuously, but sounds at predetermined intervals, and runs for shorter periods toward the end of battery life.

OPERATION

Under normal operation the cockpit switch is in the ARM position. The switch on the ELT unit will be positioned OFF. With these switch settings, the ELT will automatically activate on impact. To manually activate the ELT, set the cockpit switch to the ON position. When the ELT is activated, the presence of the emergency swept tone, a flashing cockpit indicator light, and the buzzer in the tail indicates a normally functioning unit. If the ELT is activated, it can be reset. This is done by moving the cockpit switch to ON and then immediately back to ARM.

The 406 MHz transmitter will operate for 24 hours and shut down automatically. The 121.5 and 243.0 MHz transmitter will continue to operate until the unit has exhausted the battery power. The 406 MHz transmitter transmits a digital message that allows search and rescue authorities to retrieve information from a database. Information contained in the database that may be useful include:

- Type of aircraft.
- Address of owner.
- Telephone number of owner.
- Aircraft registration number.
- Alternate emergency contact.