

# FLIGHT CONTROLS

## GENERAL

The flight control system of the Citation X is conventional in that it has a rudder, elevators, ailerons, and roll spoilers. It is unconventional in that they are completely redundantly powered and that the airplane is equipped with dual rudders, both powered from different sources, the lower rudder having an additional independent backup system. The upper rudder is electrically driven. All controls have redundant sources of operating power, as well as having manual backups (except the upper rudder) by conventional cable and follow-up systems. The horizontal stabilizer is of the movable type, powered by dual motors, and is trimmed by an electromechanical linear actuator which is installed at the top of the vertical fin. The actuator operates in response to command inputs from the pilot and copilot trim switches and from the automatic flight control system (AFCS). The secondary trim system (the second motor with its own control module) is controlled by a switch on the pedestal, and receives power from the emergency DC bus. A stabilizer trim indicator is presented on the central engine indicating and crew alerting system (EICAS) display unit (DU); a trim clacker gives audible indication if the autopilot trims for over three seconds.

Lateral and directional trim are achieved by shifting the neutral position of a feel and centering mechanism, by means of a linear electric actuator. The trim switches are located on the aft end of the center pedestal.

All primary control surfaces are powered by the airplane's dual hydraulic systems, power responsibility being approximately divided between the two systems. The two elevators and the lower rudder are each driven by two separate power control units, (PCUs), each one being powered by a different hydraulic system. The lower rudder, which is normally powered by both hydraulic systems, can be powered by an electrically driven self contained, independent hydraulic system which is isolated from the other systems. It is located out of the potential engine rotor burst area. The left and right ailerons are connected together at a central-output-quadrant which is driven by two separate PCUs, each powered by a different hydraulic system. The roll spoilers (1, 2, 9, and 10 in Figure 2-31) are each driven by a single PCU. The roll spoilers operate in conjunction with the ailerons and augment their roll control function.

Speed brake panels (3 and 8 in Figure 2-31) are powered by the A hydraulic system, and panels (4, 5, and 6 in Figure 2-31) are powered by the B hydraulic system.

Pressure loss of one hydraulic system has no major effect on operation of the elevators, ailerons, and rudder. In the event of total hydraulic pressure loss, the surfaces can be controlled manually with some limitations. In case of failure of a hydraulic system controlling a roll spoiler, the roll spoiler will be held in the stowed position. In case of loss of both A and B systems pressure the lower rudder is automatically powered by its own independent electrically driven backup hydraulic system.

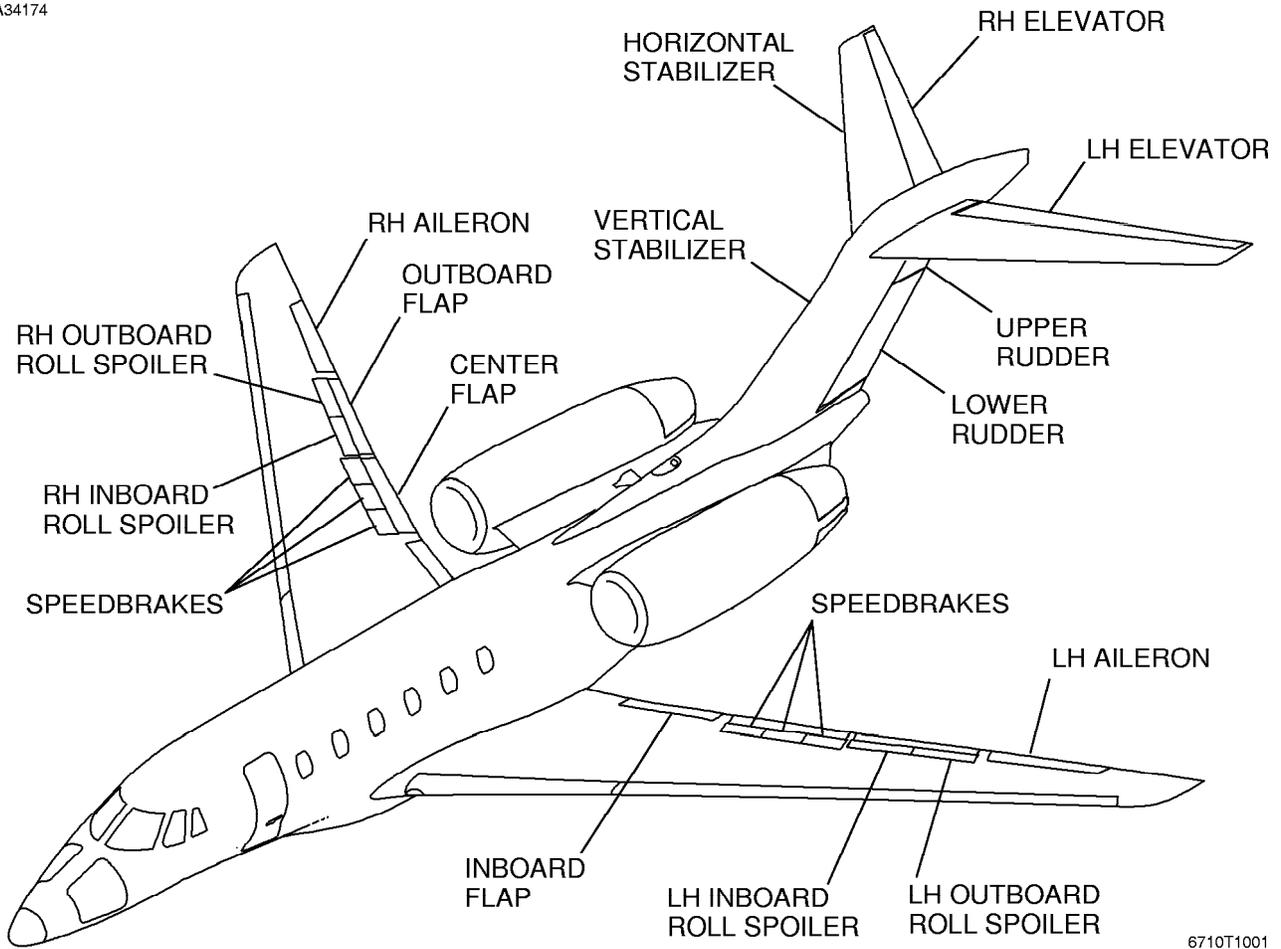
The leading edge slats are extended or retracted using either hydraulic system, since the slat control valve is a single unit with dual valves to accommodate both system A and system B pressure.

The airplane is also equipped with dual yaw stability augmentation systems (YSASs), or yaw dampers, which continuously provide additional yaw control and Dutch roll damping by electrically driving the upper and lower rudders.

A pitch and roll disconnect handle, located at the aft end of the center pedestal, allows disconnection, and thereby isolation, of the pilot's and copilot's pitch and/or roll controls.

# FLIGHT CONTROLS

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Figure 2-21

The flaps are electrically powered and are composed of six segments which move simultaneously. They are driven from a DC power drive unit (PDU) through a flexible shaft drive. Detents for the flap handle are provided at the UP, SLAT, 5-degree and 15-degree positions and a stop detent is installed at the FULL position.

The system power control units (PCUs) have a bungee pushrod installed to connect each PCU input link to an input bellcrank or input quadrant. The pushrod normally works as a solid pushrod to transmit the control signal to each PCU. In the event of a PCU input link jam, which causes the PCU to run away, the pilot can apply a counteracting column force to extend or contract the corresponding pushrod. This allows the other power control unit to force-fight with the runaway PCU. The PCU forces cancel each other out and pilot manual effort is required to counteract aerodynamic forces in one direction, in addition to the normal feel forces.

One of the selectable pages of the engine indicating and crew alerting system (EICAS) is a page which gives a cockpit visible indication of the flight control positions. This indication is primarily designed to enable the crew to see the positions of the flight controls on preflight, since it is not possible for the crew to otherwise visually check the control positions, and to visually indicate failure of the rudder limiter mechanism. Rudder limits and limit displays are discussed under Rudder in this section.

## POWER CONTROL UNIT (PCU) MONITOR SYSTEM

The power control unit (PCU) monitor system is an integrated hydraulic pressure sensing and manual shutoff control system. Sensors, which are pressure transducers, are located in all of the PCUs. They sense hydraulic pressure and react to a difference in the pressure. There are two transducers in each PCU; one in the extend port and one in the retract port. If one PCU in a pair has sustained a runaway, or is force fighting the other PCU, the two adjacent force ratio pressures will be different. If such an abnormal condition exists the monitoring system will cause an amber alerting message (FLT CONTROL FAULT) to appear on the engine indicating and crew alerting system (EICAS) and in the switch/light (flashing mode) of the applicable flight control annunciator. The annunciators are located on the lower center part of the pedestal. The switch/light annunciators are: AILERON/OFF, RUDDER/OFF, LH ELEV/OFF, and RH ELEV/OFF; they serve as annunciators to alert the crew which PCU pair is affected, and as switches to shut off hydraulic power to the affected pair. The top half of the switch will flash. Pressing the illuminated switch will shut off power to the affected control surface, and the control will revert to manual mode. When the control is shut off, OFF will also be illuminated in the switch/light. The EICAS message will not be removed from the display when the PCU is turned off. This is to act as a reminder to the crew, concerning the status of the PCU.

The PCU monitor system is in operation only if both PCUs in each control surface pair are operating at greater than 1000 PSI hydraulic pressure.

## AILERONS AND ROLL SPOILERS

Lateral control is provided by the combined actuation of the ailerons and the roll control spoilers (the two outboard spoiler segments on each wing). Full control wheel rotation in either direction results in 15 degrees travel of the ailerons up or down, and 0 to 40 degrees up travel of the roll control spoilers. Movement of the outboard roll spoiler starts after approximately 3 degrees of aileron displacement, and the inboard roll spoiler starts at approximately 6 degrees of aileron deflection; both are at a maximum at 68 degrees of wheel deflection. The roll spoilers do not operate as speed brakes, in flight or on the ground.

The pilot's and copilot's roll control systems are independent except for the point at which the two systems are interconnected. A manual disconnect handle, which can be used to disconnect both the pitch and roll controls of the pilot's and copilot's respective controls, is located on the aft section of the center pedestal, to be used in the event of a system jam of either the pitch control, aileron, or the roll spoiler system. Both the pilot's and copilot's systems will then operate independently, and the jammed system will be isolated. Pulling the T-handle disconnects both the roll and pitch axes. To re-engage the roll systems in flight, the T-handle must be turned 90 degrees clockwise (to ROLL RECONNECT), which is opposite of the direction for the pitch reconnect.

The pilot's control wheel is connected to the dual aileron power control units (PCU), which are mounted in parallel in the center wing area, by a cable system, bungee pushrods, and a central-input-quadrant. The PCUs drive a central-output-quadrant which mechanically drives the ailerons. The copilot's control wheel is connected to the roll spoiler PCUs through a spoiler mixer assembly. Each spoiler PCU drives an individual roll spoiler panel.

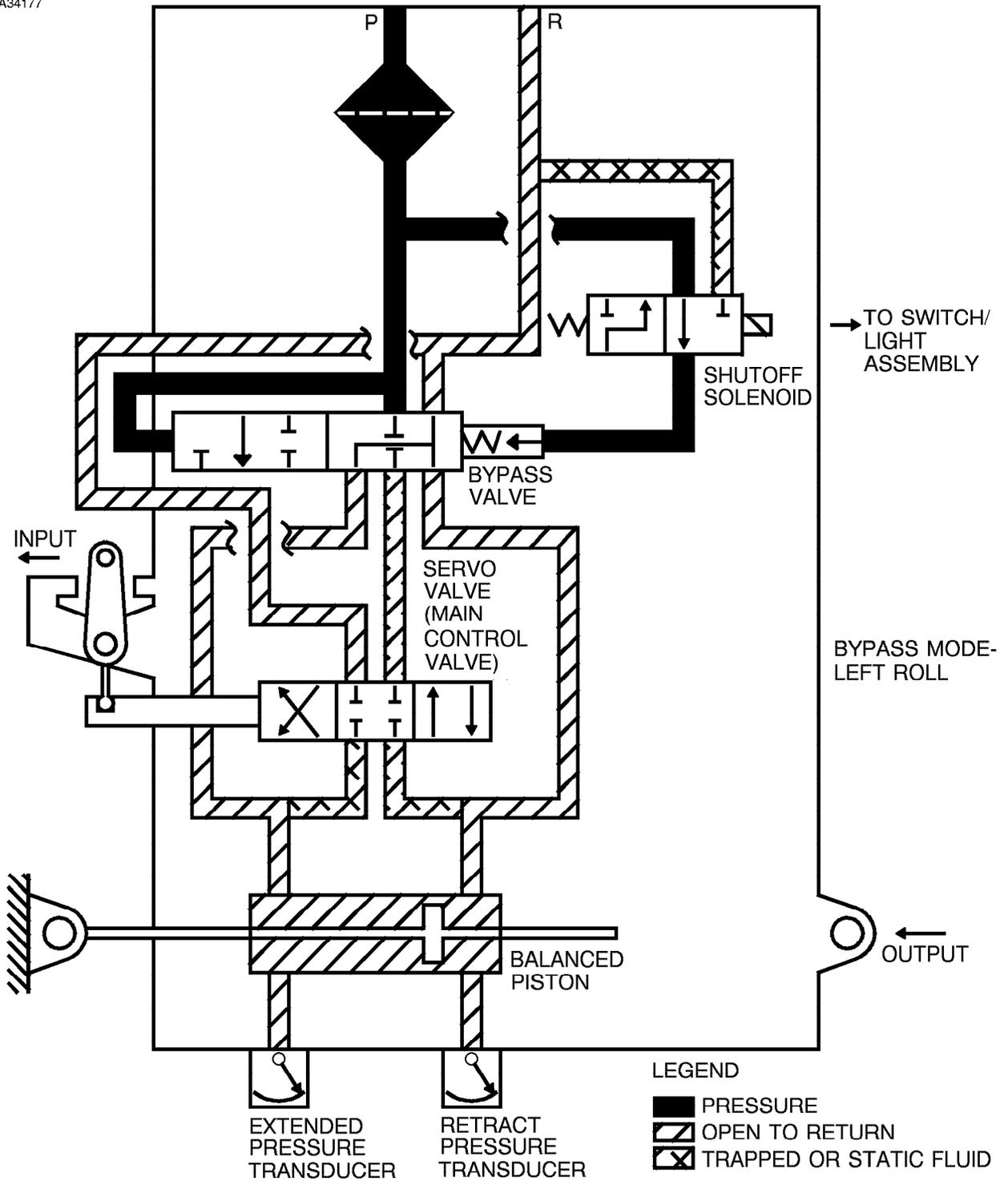
The aileron/spoiler mixer assembly system operates in two ratios: HIGH RATIO when 34 deg of wheel deflection results in maximum aileron deflection. LOW RATIO is when 68 deg of wheel deflection results in maximum aileron deflection. HIGH RATIO is normal when both A and B hydraulic system are pressurized. A cyan AILERON RATIO LOW message is normally annunciated prior to engine start. After normal system pressure is achieved on A and B hydraulic system the aileron /spoiler shifts from LOW RATIO to HIGH RATIO automatically. The AILERON LOW RATIO message then clears. When one or both hydraulic systems are subsequently depressurized, the aileron/spoiler mixer mechanism automatically shifts from HIGH RATIO to LOW RATIO. A cyan AILERON RATIO LOW message will again annunciate on the EICAS display. This ratio shift is needed for reasonable roll forces at the control wheel when continued flight in manual reversion is required.

An artificial feel (bungee) unit is linked to the aileron system to provide artificial feel to the lateral control system.

Lateral trim is electrically operated by an aileron trim switch on the pedestal. Actuation of the trim switch relocates the aileron neutral position, which is reflected in the control wheel position, and shifts the neutral point of the feel and centering unit to move the ailerons and roll spoilers together with the control wheel. An indicator on the pedestal shows the amount of trim displacement in relation to the original neutral position.

# AILERON POWER SYSTEM

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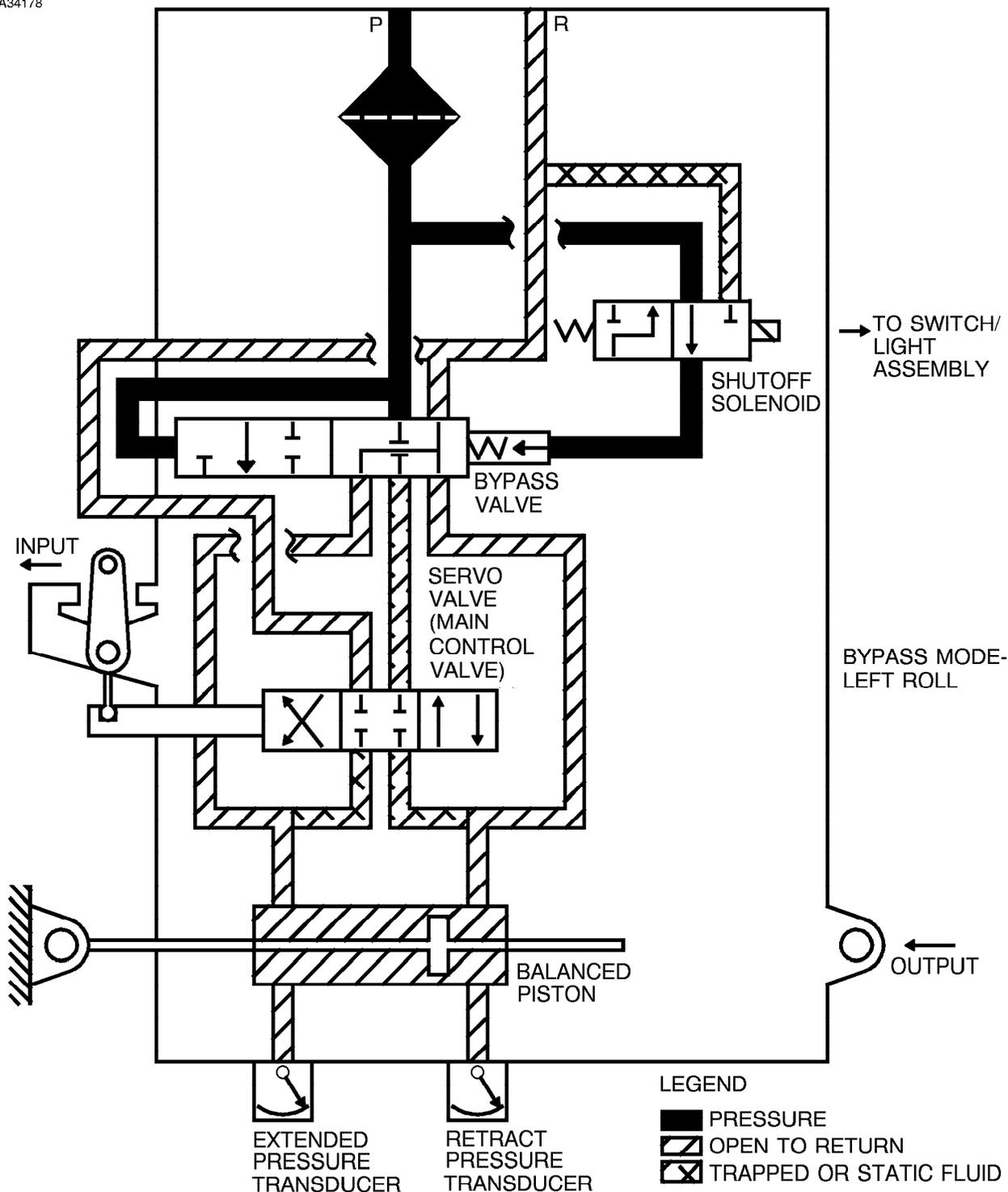


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Figure 2-22 (Sheet 1 of 3)

# AILERON POWER SYSTEM

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Figure 2-22 (Sheet 2)

# AILERON POWER SYSTEM

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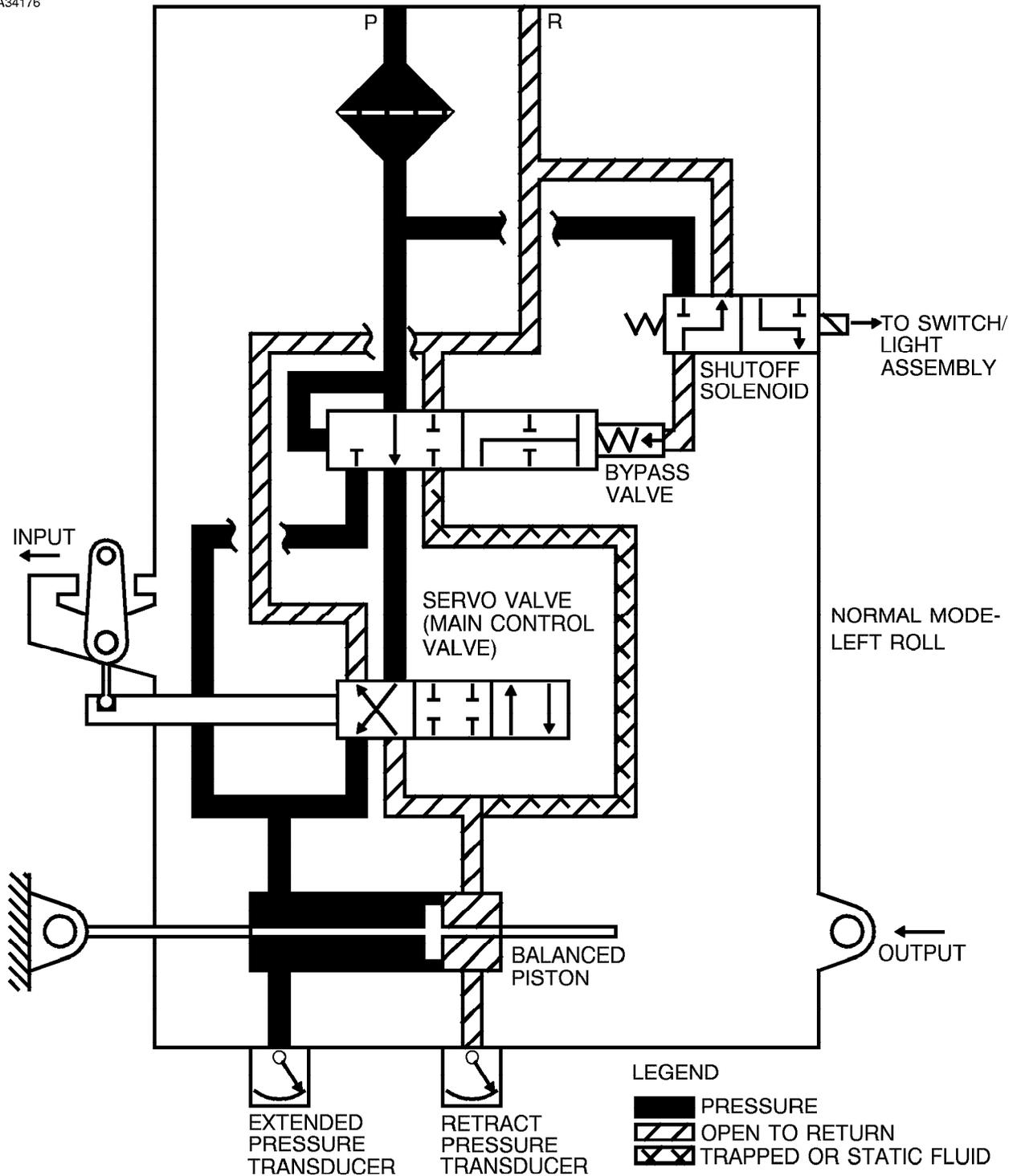
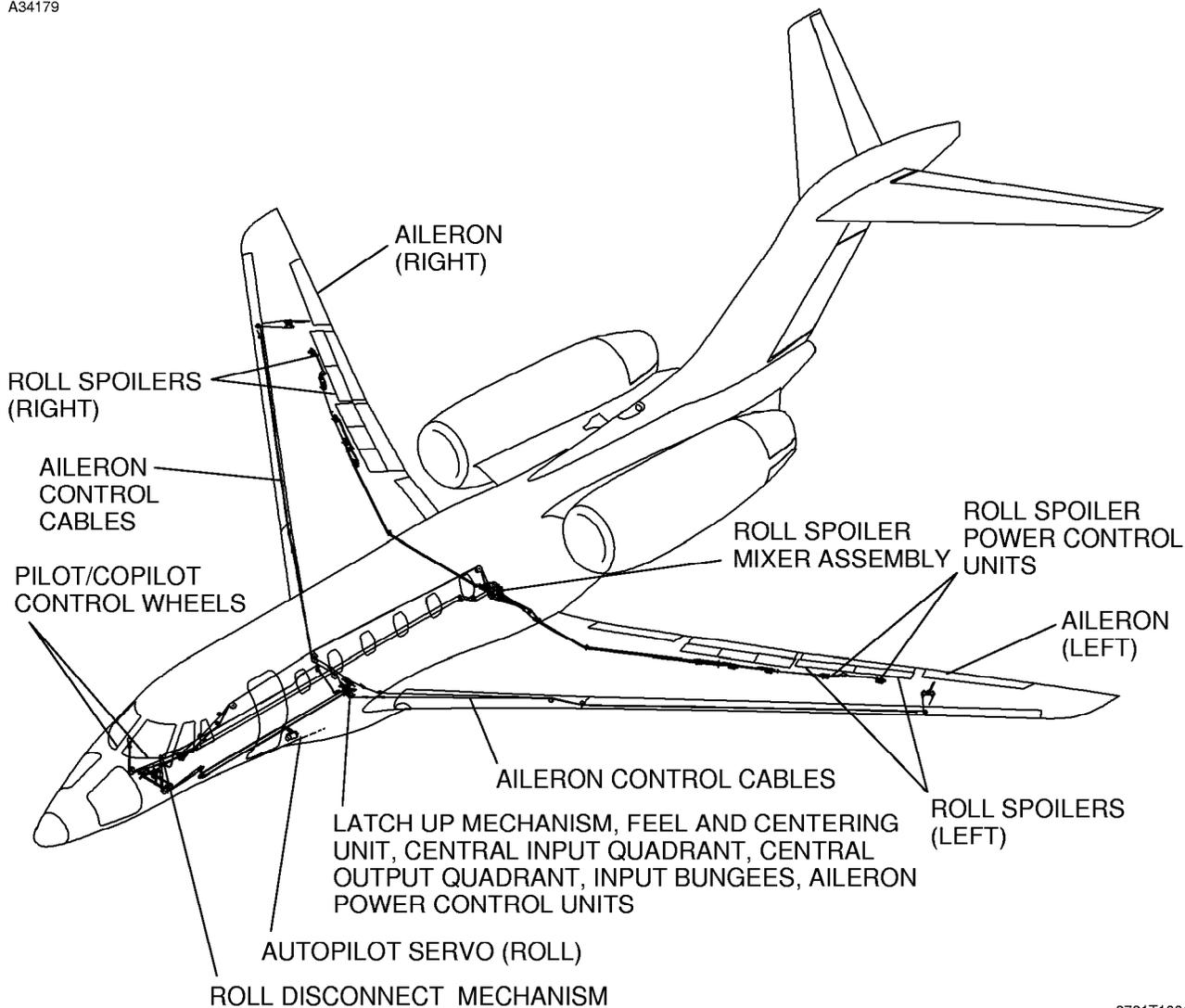


Figure 2-22 (Sheet 3)

# AILERON/SPOILER ROLL CONTROL SYSTEM

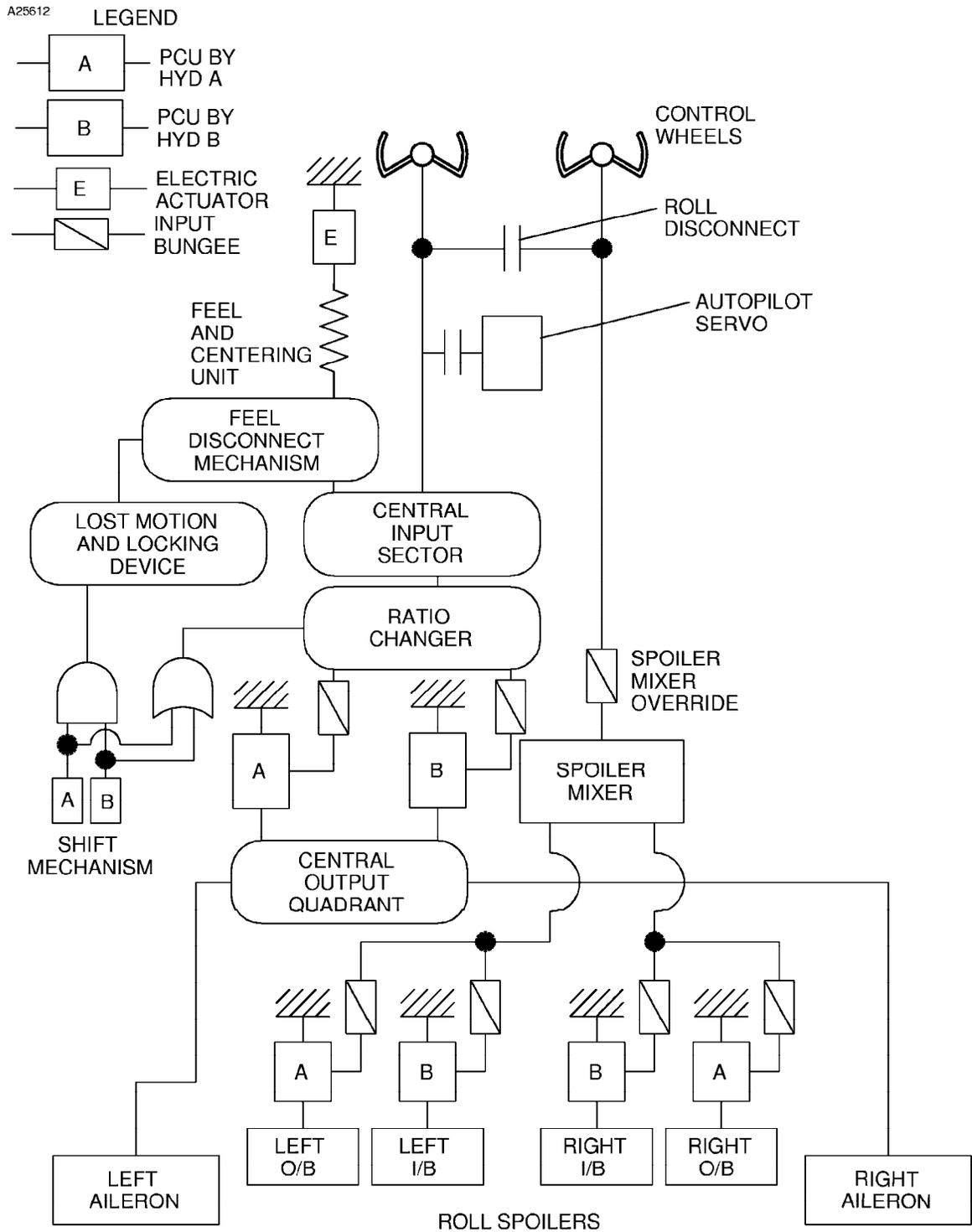
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Figure 2-23

# AILERON/SPOILER ROLL CONTROL SYSTEM DIAGRAM



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Figure 2-24

## ELEVATORS

Pitch control is accomplished through conventional forward and aft movement of the control columns, which operate separate left and right elevators, through two separate power control units, (PCUs) on each elevator. The pilot's control column is linked to the left elevator and the copilot's is linked to the right one. Each of the two PCUs powering a control surface receives hydraulic pressure from a different system. Full elevator travel is through a range of 19 degrees nose up to 15 degrees nose down.

Elevator feel is provided by pitch feel units incorporated in each elevator control system to provide artificial feel for the pilots. The feel units are of electromechanical design, consisting of a mechanical spring cartridge and a linear electric actuator. They work in conjunction with the PCU centering springs, located in each side of the horizontal stabilizer, and generate a feel force pattern in relation to the airplane speed and the elevator position, similar to that which would be present in a manual control system. The electrical inputs to the feel units are generated by the micro air data computers (ADC).

Control column movement is transmitted to the elevator PCUs through control cables, and through bungee pushrods, to the left and right dual PCUs, which drive the left and right elevators in proportion to the amount of input. The bungee pushrods operate under normal conditions as a solid rod to transmit the control inputs to the PCU. In the event of a PCU input link jam the pilot can apply a counteractive force which extends or compresses the bungee pushrod which will allow the operable PCU to assume control of elevator operation.

The pilot and copilot control columns are interconnected through a torque tube which incorporates a pitch/roll disconnect mechanism. In the event of a control system jam, the disconnect mechanism separates the two control systems and allows independent operation of the systems. Pitch disconnect is achieved by pulling a T-handle on the aft right side of the pedestal. The pitch disconnect system can be re-engaged in flight.

Since each pair of elevator PCUs is driven by separate hydraulic systems, loss of one hydraulic system has no major effect on elevator control system operation. In the event of a total hydraulic system failure, the control systems are operable in the manual reversion mode. Both elevators, in this case, are operated by means of cable transmitted control inputs which manually move the actuators. No switch positioning or other act of the pilot is required to cause reversion to the manual mode.

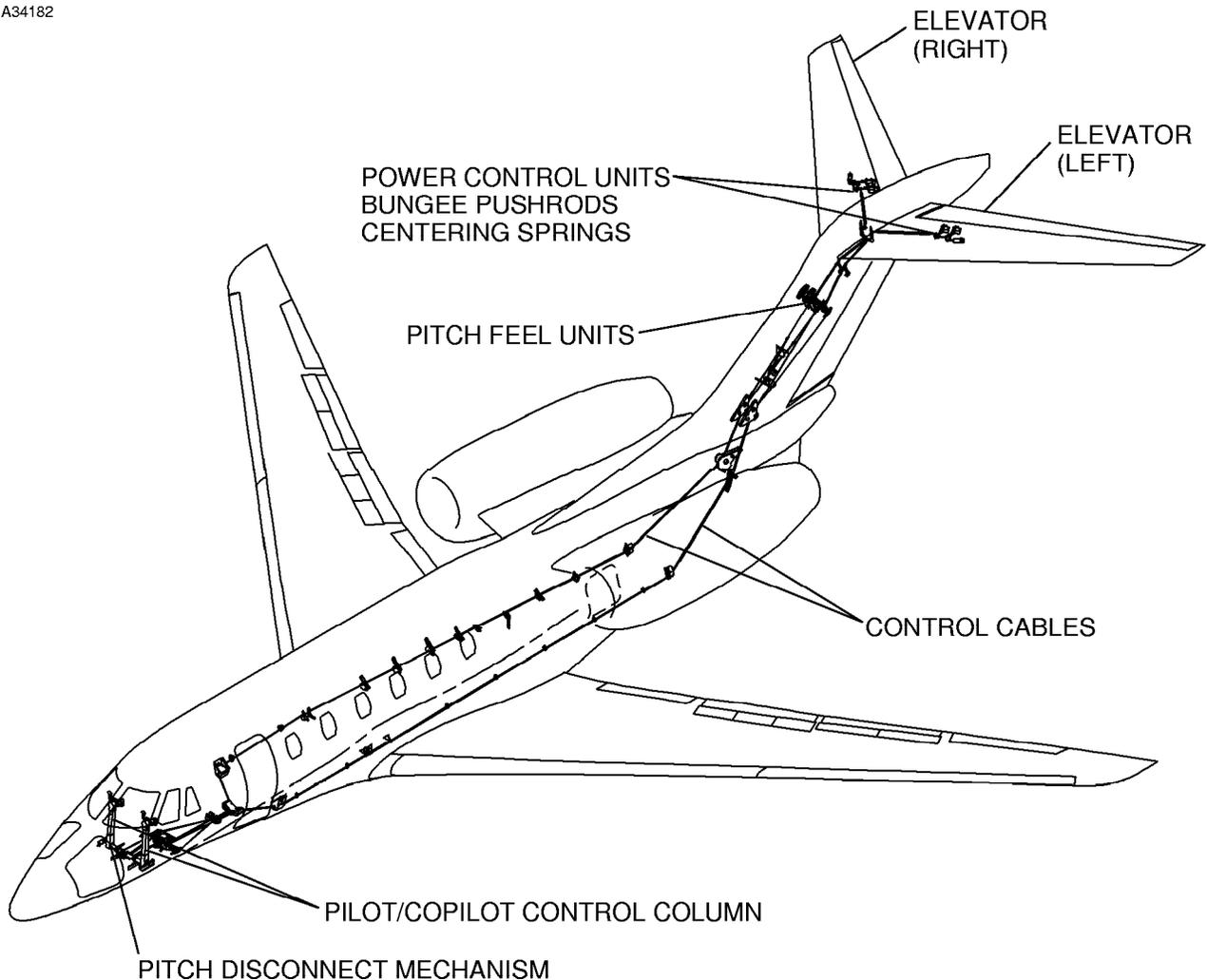
Pitch trim is accomplished by repositioning the entire horizontal stabilizer. Full travel of the stabilizer is from 12 degrees in the airplane nose up direction to 2 degrees in the airplane nose down direction. Primary pitch trim is controlled by switches on the control wheels. The trim switches are in two segments which must be actuated simultaneously. A SECONDARY TRIM ON/OFF switch on the left side of the pedestal, when ON, disengages the primary pitch trim system and arms the secondary system. Pitch trim, with the secondary system, is controlled by a two segment NOSE UP/NOSE DOWN switch on the pedestal. The primary system can thus be disengaged by arming the secondary system, by depressing either the pilot or copilot A/P/TRIM/NOSE WHEEL STEERING DISC switch on the pilot's or copilot's control wheel, or by pulling the PRIMARY STAB TRIM circuit breaker. The secondary system is disengaged by placing the ON/OFF switch to OFF. Pulling the SECONDARY STAB TRIM circuit breaker will also disengage that system.

Pitch trim indication is provided to the pilots on the center display unit (DU) of the engine indicating and crew alerting system (EICAS) display. A calibrated (analog) scale with pointer is always present, displayed in white, on the left side of the display. The STAB trim setting is also presented in digital display. A green area, showing the takeoff trim setting range on the scale, is present on the analog display when the airplane is on the ground, i.e., when the takeoff phase inhibit (TOPI) system is active. In this case T/O will also be annunciated in the display. The digital display will follow the color scheme of the analog pointer, and the pointer will be displayed in the color range of the arc where it is pointing. If stabilizer trim data become invalid or miscompare, the digital display will be dashed out with amber dashes and the analog pointer will be removed from the display.

A clacker sound is provided through the audio to alert the pilots any time the stabilizer trim is actuated by the autopilot for a period longer than three seconds. The avionics power switch must be on for the clacker to sound. Normal trimming by the pilots using the control wheel trim buttons will not actuate the trim clacker.

## ELEVATOR CONTROL SYSTEM

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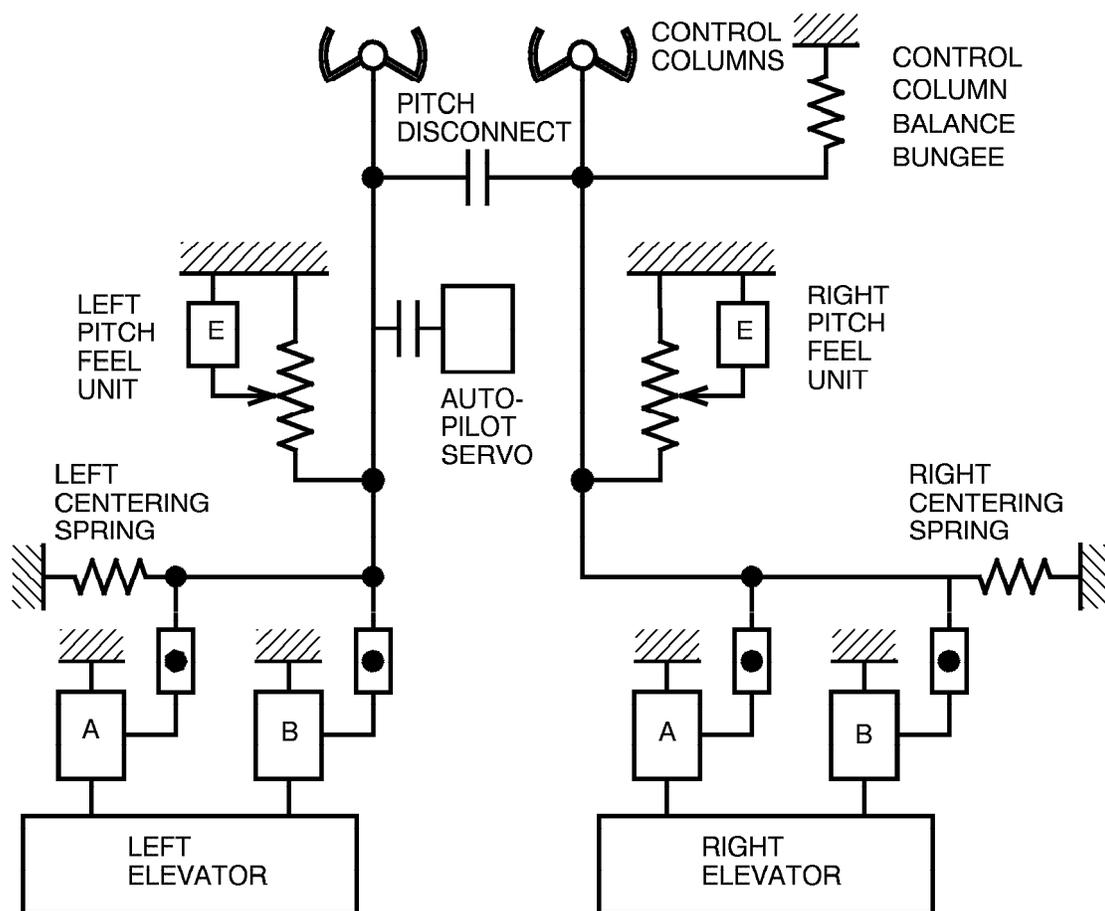
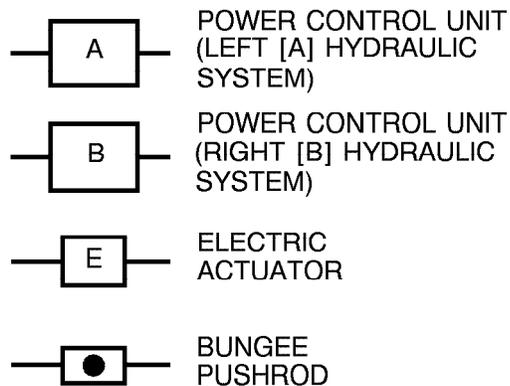
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Figure 2-25

# ELEVATOR CONTROL SYSTEM DIAGRAM

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## LEGEND



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Figure 2-26

## RUDDER SYSTEM

The airplane yaw control is provided by two separately controlled rudders; an upper and a lower. The lower rudder has the larger area and is positioned by two identical power control units (PCUs), each of which is powered by a different hydraulic system. The PCUs are installed in the vertical fin. Both pilots' rudder pedals are connected to the PCUs through a single mechanical control path which includes bungee pushrods.

A yaw feel and centering unit is linked to the lower rudder system to provide artificial feel to the pilots. The feel and centering unit changes the amount of rudder pedal effort required for differing rudder deflections and thereby simulates the feel of a non-powered system. Maximum rudder deflection is 30° for the lower and 18° for the upper, in either direction. Dual yaw damper servos are linked to the lower rudder system (in series) to provide Dutch roll damping and turn coordination.

The smaller upper rudder is driven electrically by two stand alone yaw stability augmentation systems (YSAS). The yaw dampers are referred to as the primary (A) and the secondary (B). Each YSAS has a yaw stability augmentation computer (YSAC) and a servo actuator as component parts. Either the primary or the secondary YSAS continuously provides Dutch roll damping of the airplane as well as tracking of the upper rudder to the rudder pedal commands from the pilots. Selection of the yaw damper is controlled by an amber lighted switch (UPPER YAW DAMP/A/B) on the upper center instrument panel. The selected yaw damper annunciation will be illuminated in the switch. Default selection alternates at each power up in order to even out system usage. The lower rudder also provides yaw damping stability augmentation, however, its yaw damping is provided through PCU displacement of the rudder and is therefore not available in the event of complete hydraulic failure. Normally both the lower and upper rudder systems provide yaw damper function at the same time. If the yaw damper on either rudder completely fails, however, the other system will provide adequate control to maintain the yaw stability of the airplane.

Due to the critical nature of the yaw damper system, several engine instrument and crew alerting system (EICAS) messages are designed to appear in order to alert the crew to any system difficulty. The EICAS messages which are possible in respect to the yaw dampers are: (1) YD FAIL LOWER A (and/or B) (amber) - indicates that either of the yaw dampers has failed in flight, both have failed on the ground, or the selected one has failed during power up. (2) YD FAIL LOWER A (or B) (cyan) - indicates that one yaw damper has failed. (3) YD NOT CENTERED (amber) - indicates upon power up that one or more of the linear actuators is/are not centered. Do not dispatch in this case, until the problem is repaired. (4) YD UPPER FAIL A (or B) (amber) - This message will appear on the ground if the non-selected yaw damper should fail before takeoff. (5) YD OFF LOWER (amber) indicates that the yaw damper is not in operation.

Dual electromechanical rudder travel limiters are installed on the lower rudder to limit the maximum rudder deflection, depending upon the airplane speed. At low airspeed the rudder can deflect to its full limit; at higher airspeeds, the rudder is limited in travel to a variable percentage of its maximum deflection. This protects the airplane structure from being overloaded by excessive rudder input at high speeds. The upper rudder is thereby also protected in its range of movement, which is less than the lower rudder, because the upper rudder receives its inputs electrically from the rudder pedals, which are mechanically limited at higher speeds by the lower rudder limiters. In case of one limiter failure, the remaining one will operate with no loss of efficiency. The rudder travel limiter operates on the principal of a triangular shaped probe, which moves in a slot which travels with the rudder. As the triangular probe moves deeper into the slot, available rudder travel is restricted.

The electromechanical limiters receive their airspeed data from the automatic flight control system (AFCS) and translate it into a mechanical limiter position.

The available deflection versus airspeed information is depicted in Figure 2-28 in this section. Airspeed data for the rudder limiters is obtained from the automatic flight control system (AFCS).

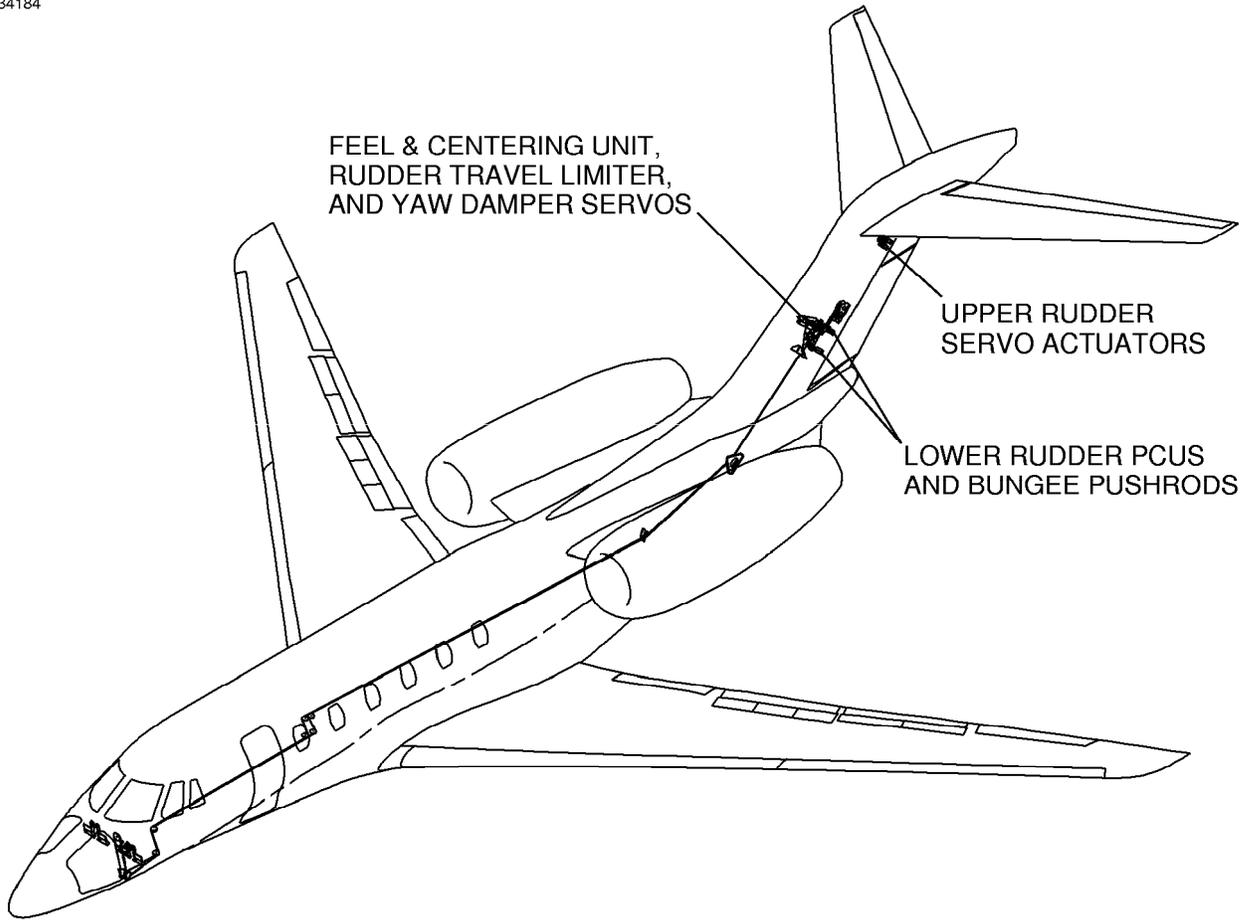
Rudder deflection, and in case of failure of the rudder limiter mechanism, the range of failure is indicated on the control deflection page of the EICAS system. This page is selected by selecting CTL POS on the bezel button of the EICAS display unit (DU). The percentage of available rudder which may be used will be indicated in both digital and analog display. In the case of a single failure, the digital display will be in amber and the analog display will remain green. In case of a dual failure the horizontal bar will become red in the area that represents the portion of the rudder arc which should not be used, and the digital display will also be in red. The pointer will follow the color of the area of the indicator where it is at any time. A red digital indication of 80%, indicates that the rudder limiter has failed at the 80% point of full rudder throw, at which point the rudder will be blocked mechanically. The green area of the analog display will vary with the desired limits of the rudder depending upon the present airspeed. The pilot will then, based on the display, be able to keep application of the rudder out of the restricted (red) range.

An amber RUD LIMIT FAIL A-B message indicates a single limiter failure, either A or B. A red RUDDER LIMIT FAIL message indicates a dual limiter failure. These indications will not be available without a RUDDER LIMIT FAIL message. The control deflection indications will always be available upon selection of CTRL POS on the bezel button. If the data being received by the CAS system is invalid or there is sensor disagreement, the bugs will be changed to amber or will be removed from the display. The digital crew alerting system (CAS) indications will follow the EICAS message color.

Rudder trim is applied through a left-right momentary switch on the lower center pedestal. Trim is displayed on the rudder trim portion of the consolidated aileron/rudder trim indicator, also located on the lower center pedestal. Movement of the lower rudder trim is displayed in units from neutral. An indication of one unit or more of the trim will trigger a no takeoff warning on the EICAS. The rudder trim is achieved by electrically displacing the trim actuator, moving the whole lower rudder control system and shifting the neutral point of the feel and centering unit.

# RUDDER SYSTEM

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Figure 2-27

# RUDDER CONTROL SYSTEM SCHEMATIC

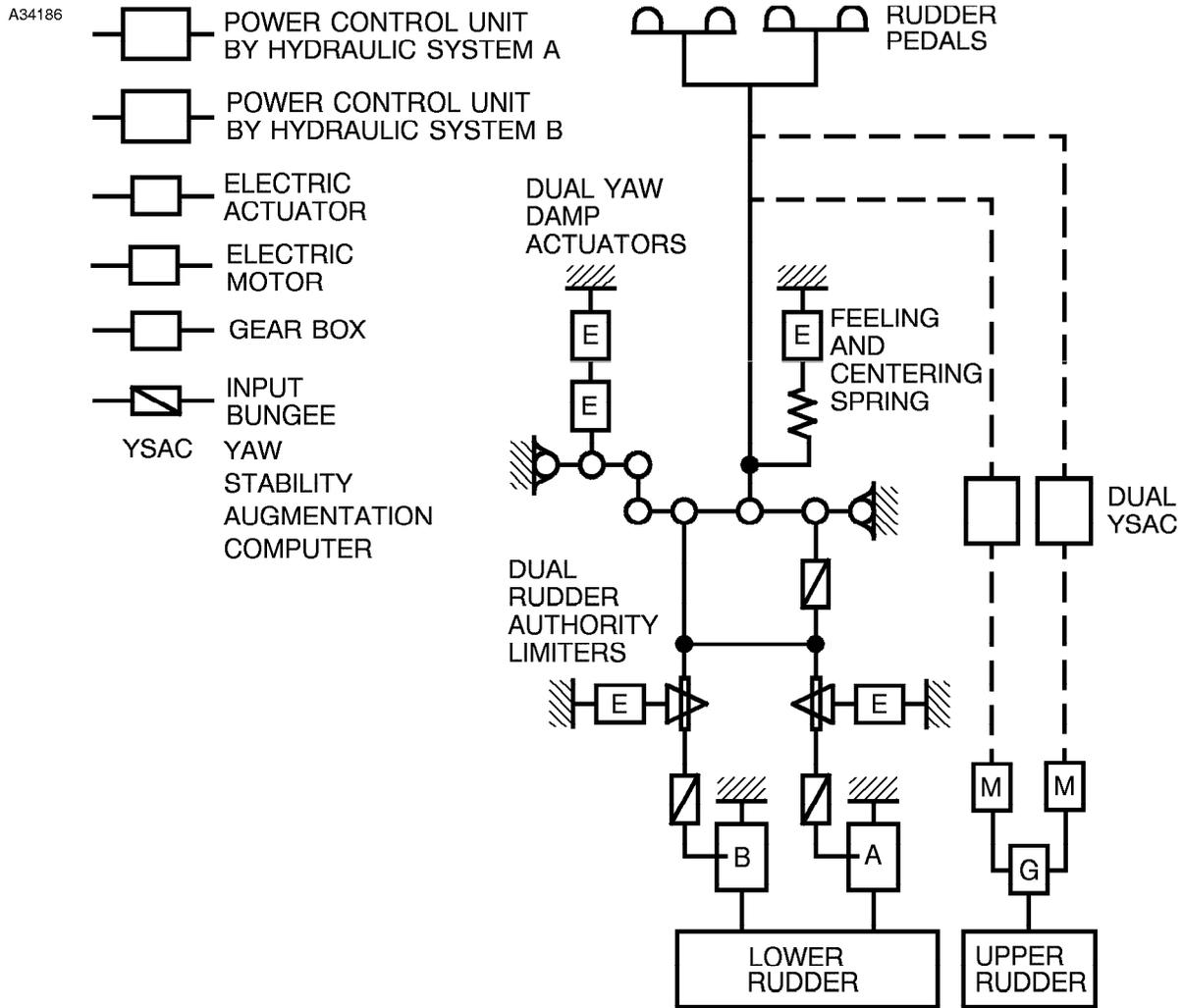
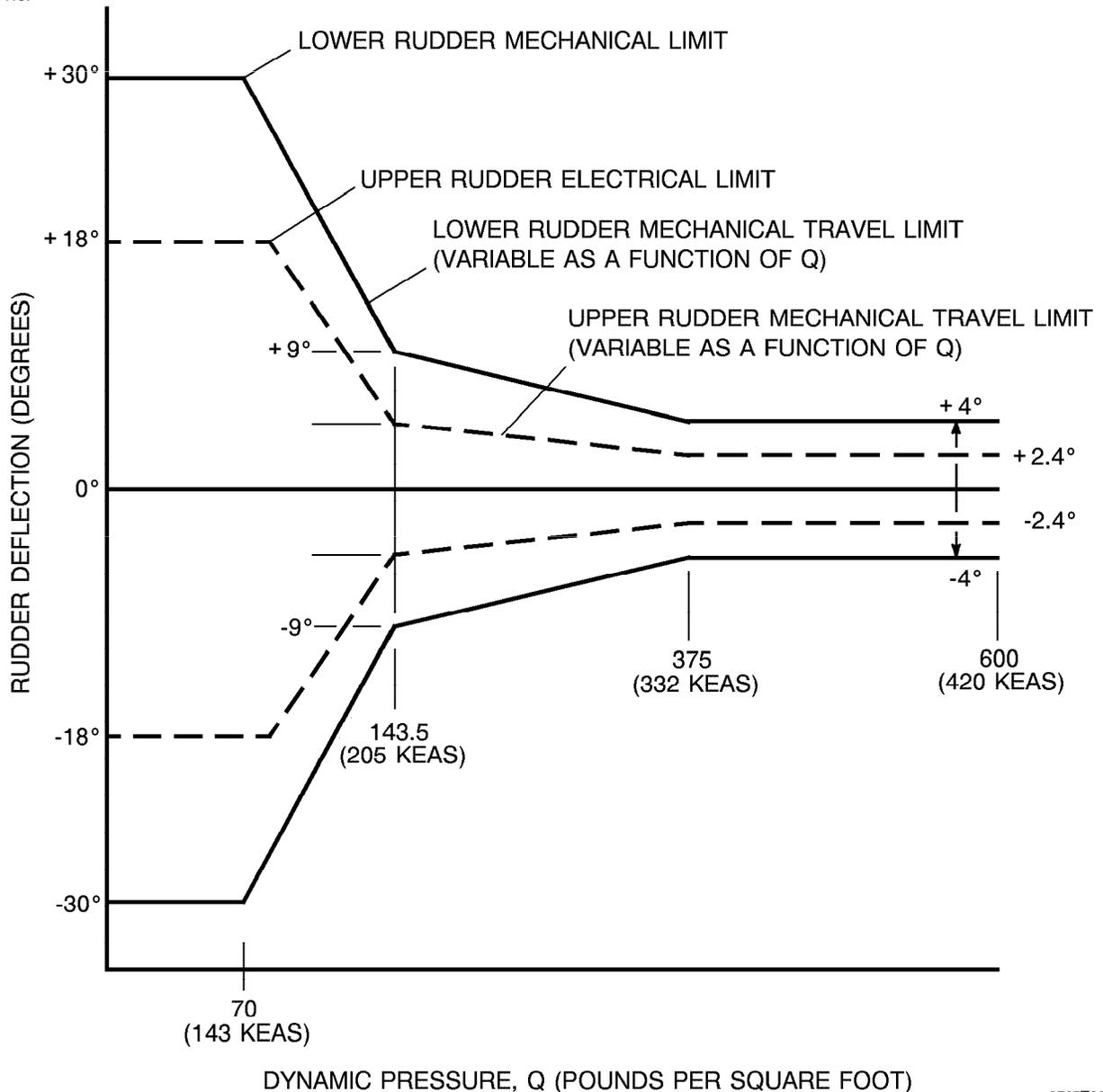


Figure 2-28

# RUDDER TRAVEL LIMITING

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Figure 2-29

## NOSE WHEEL STEERING

The nose wheel steering is powered by the A hydraulic system. Steering is accomplished through a hydraulic rack and pinion type power steering unit mounted on top of the nose landing gear assembly. The unit is controlled by a steering handwheel mounted on the pilot's side console and/or through the pilot's or copilot's rudder pedal inputs. The handwheel provides steering of up to 80 degrees in either direction and the rudder pedals provide a maximum of 12 degrees of steering. The inputs are cumulative through a cable and pulley mixer assembly, so by using the handwheel and the rudder pedals together, a maximum of 92 degrees deflection of the nose wheel is possible in either direction.

The power steering unit also provides nose gear centering upon retraction, since the nose wheels must be centered in order to enter the nose wheel well. With the gear fully extended the centering function is accomplished by allowing retract pressure on both sides of each rack, which forces them to the centered position. A centering logic valve, which is located within each steering rack, regulates the differential pressure by venting one side or the other to return until the racks are centered.

A nose wheel accumulator is installed in the system in order to absorb shocks in the system and to provide a separate trapped source of hydraulic power in case of complete system failure. The accumulator is of the sliding piston type and is serviced with 1500 PSIG of nitrogen.

#### NOTE

The nose gear torque links must be disconnected during towing operations or the system may be damaged. The torque link disconnect pin is removed by removing a safety pin from the shaft by pushing a release button and pulling out the pin. The torque links are spring loaded to extend horizontally from the nose gear strut when the pin is removed.

The nose wheel steering accumulator bleed switch is powered through the FWD/AFT COMP LTS circuit breaker which is on the hot battery bus, so power is available to it at all times.

The nose wheel steering hydraulic pressure is controlled by the normally open nose wheel steering shutoff valve. When the nose landing gear squat switch signals that the airplane is airborne, the nosewheel steering shutoff valve is powered closed. The nose wheel steering shutoff valve can also be powered closed by momentarily pressing the A/P/TRIM/NWS DISC switch on either control wheel. The switch must be held depressed for as long as the nose wheel steering is not desired. Steering will resume when the switch is released. With the nose wheel steering hydraulic power cut off, the nose wheel will caster and the shimmy damping will be effective.

Two redundant systems keep hydraulic power from the nosewheel steering system when it is not desired. These are the hydraulic blocking (sequence) valve blocks pressure when the gear is not fully extended, and the nosewheel steering (solenoid) shutoff valve, which is powered closed when the nose gear squat switch is signaling that the airplane is airborne.

The hydraulic blocking valve is located in the circuit between the accumulator and the nose wheel steering unit. It is powered open when either main landing gear squat switch signals a landing. The function of the blocking valve is to assure a fully charged nose wheel steering accumulator upon touchdown.

# NOSE WHEEL STEERING FUNCTIONAL DIAGRAM

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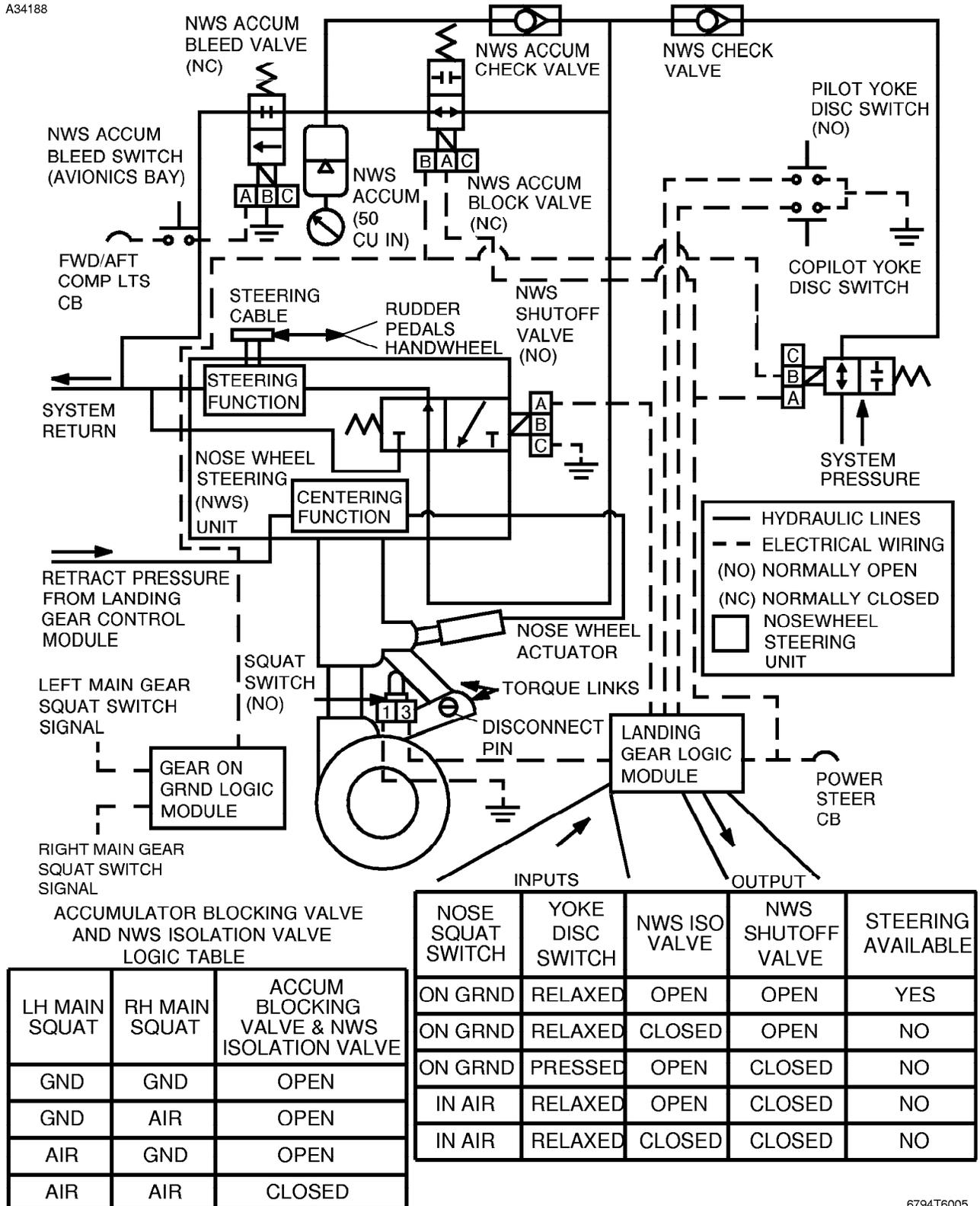


Figure 2-30

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## SPEED BRAKES

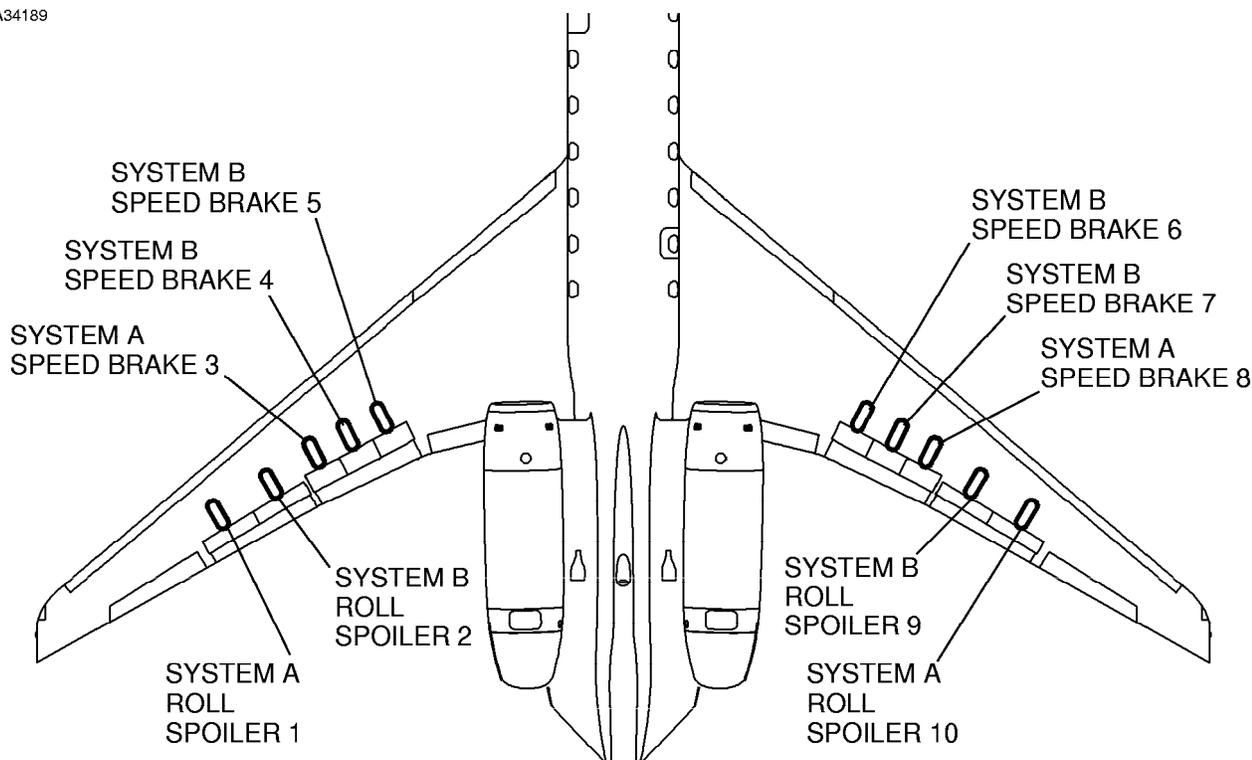
The speed brakes are comprised of spoiler panels numbered 3 through 8; three of them on the trailing edge of each wing. The panels are mechanically controlled and hydraulically actuated, each panel having its own actuator. They are connected to the control lever, located on the throttle quadrant, by cable, pushrods, and bellcranks to the hydraulic actuators which power them. Speed brake actuation can be modulated from 0 to 40 degrees of deployment. The hydraulic actuators for the speed brakes are essentially the same as the spoiler actuators which operate with the ailerons, except that the speed brakes are controlled only by the speed brake lever and the spoiler panels respond only to control wheel input.

In case of complete hydraulic system failure, or the failure of an actuator, the speed brake panels can retract but will not extend, so loss of range due to floating of speed brakes and spoilers does not present a problem.

The center speed brake panel of the left wing and the outboard panel of the right wing are monitored for position. The speed brake monitoring system puts out signals to the engine indicating and crew alerting system (EICAS) and to the angle-of-attack system. If the monitored speed brake panels are extended over 5 percent, white filled bars will be displayed on the synoptic wing in the EICAS display. If speed brake data is invalid or out of range or if there is a five percent split in the extension of the two monitored panels, the synoptic wing indication of all six panels will change to amber. When the speed brakes are stowed no symbology is present.

## SPEEDBRAKES/SPOILERS

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Figure 2-31

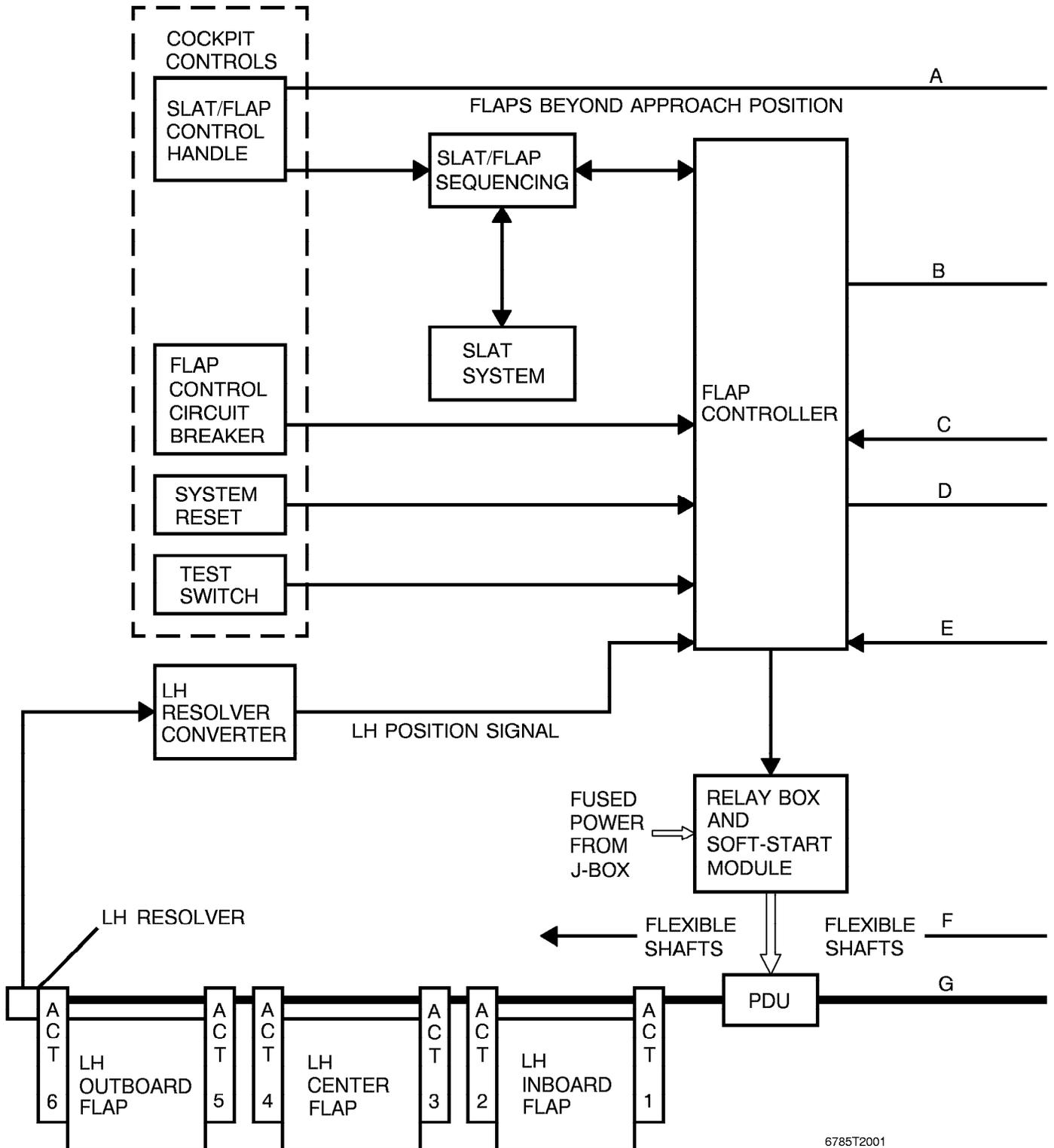
## FLAPS

The trailing edge flaps are mechanically controlled and driven by a direct current (DC) electric power drive unit through ballscrew actuators and flex-drive shafts. The flaps are in three segments on each wing. They use the Fowler flap motion in that the flaps extend as they are lowered, thereby effectively increasing the wing area when the flaps are extended. Extension time for both full up to full down, and full down to full up, is approximately 27 seconds. The major control and monitoring function of the flaps is accomplished by the flap controller, which can sense several types of malfunctions and shut down the flap system, if required. The controller runs a self test during each preflight to verify the proper functioning of all fault detection circuitry. Each fault that the controller senses has a built in test equipment (BITE) code which it will set, and which can be read after flight, through a window in the controller case.

A FLAP RESET button is located immediately forward of the flap control handle; if a condition which has immobilized the flaps has cleared, the flaps will regain operation upon the pressing of the FLAP RESET button. If the condition has not cleared the flaps will remain inoperative. If the controller senses a fault and disables the flaps, an amber FLAPS FAIL message will appear in the engine instrument and crew alerting system (EICAS) display. Flap positions of 0, 5, 15, and 35 degrees are selectable in detents.

# FLAP SYSTEM BLOCK DIAGRAM

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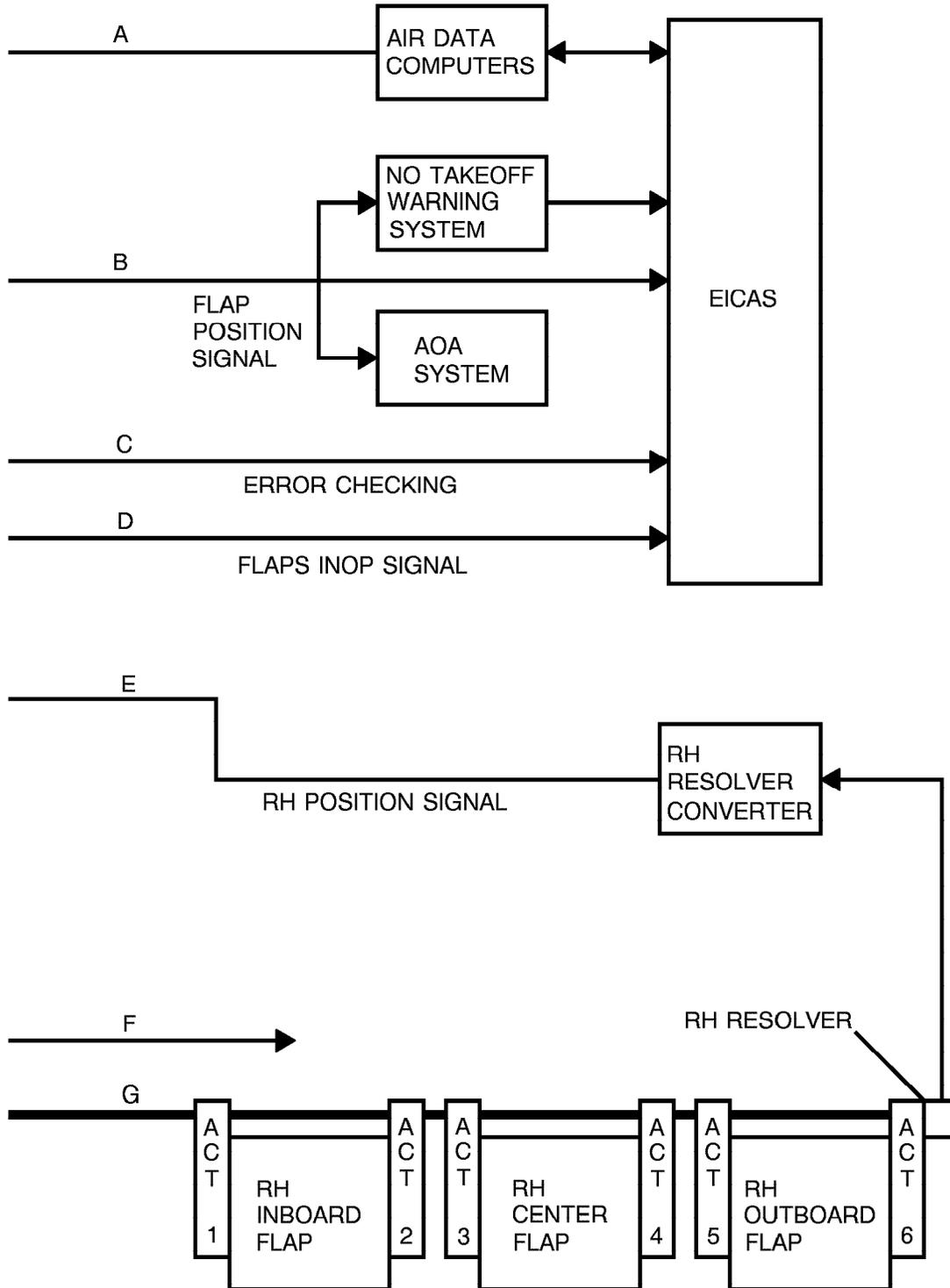


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Figure 2-32 (Sheet 1 of 2)

# FLAP SYSTEM BLOCK DIAGRAM

A34196



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Figure 2-32 (Sheet 2)

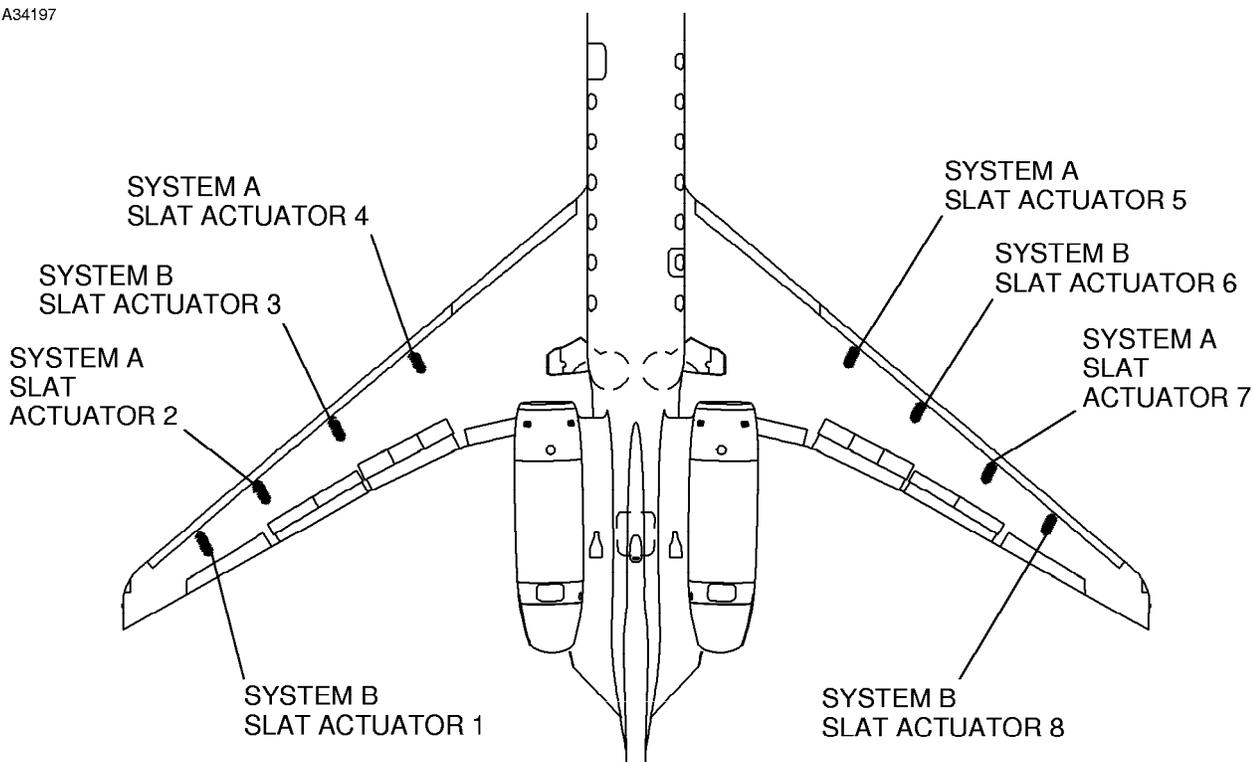
## SLATS

Leading edge slats are provided on each wing for added low speed lift, and controllability enhancement. Deployment occurs when the flaps are lowered and the flap handle passes the SLAT detent on the flap handle. The slats can extended independly of the flaps by selecting the SLAT ONLY position on the flap quadrant. The slats are retracted when UP is selected on the flap handle and the flaps are retracted to less than 1.5°. The slats are controlled by the slat control valve, which contains four solenoid operated pilot valves and an interlocked spool. Hydraulic pressure is routed from hydraulic systems A and B into the appropriate stow and deploy lines. Four hydraulic actuators are used on each wing, for a total of eight. Every other actuator on each wing is operated by the opposite hydraulic system. Hydraulic system A provides hydraulic pressure for slat actuators number 2, 4, 5, and 7. Hydraulic system B provides the power for slat actuators number 1, 3, 6, and 8.

Flap/slat sequencer logic modules are incorporated to process electrical signals pertinent to the flap/slat systems. The slats react to one extend signal which is not related to handle position. Angle-of- attack (AOA) data from the AOA sensor is provided to the flap/slat sequencer modules. If a very high angle-of-attack condition is sensed, the slats will be extended as an automatic corrective action to the situation. If this automatic extension function should fail, a red AUTO SLATS FAIL message will appear in the crew alerting system (CAS) display and a double chime will sound. In this case there will be no analog display.

## SLATS

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Figure 2-33

Slat position sensing is provided by four proximity switches located in the leading edge of the wing. Position signals are sent through the flap/slat sequencer to the data acquisition units (DAU) and are then displayed on the lower right corner of the engine indicating and crew alerting (EICAS) system display, on the synoptic wing. Slat positions are represented by two bars ahead of the displayed synoptic wing. They are shaded white for normal extension, are absent for retraction, and are amber for an asymmetrical condition. The amber display only indicates that there is an asymmetrical condition; it does not indicate the extent of it, or any detail. An amber digital crew alerting system (CAS) display SLATS ASYMMETRY will also appear at the same time.

If the slats fail to extend or fail to retract when commanded, an amber CAS message SLATS FAIL will be annunciated.

## STALL WARNING - STICK SHAKER

The stall warning system receives its information from the angle-of-attack (AOA) system. The angle-of-attack system is completely redundant in that it is comprised of two separate systems, a left and a right. Slotted probes, which act as position transducers, are mounted on each side of the forward fuselage. The probe/transducers transmit AOA data to the angle-of-attack computers, mounted in the pilot's and copilot's side consoles, which use the probe/transducer information and airplane configuration (flap, slats, and speedbrakes) to compute a "normalized" angle-of-attack. The resulting information from the left system is displayed on the pilot's optional AOA indicator on the left switch panel, and on the left side of the pilot's electronic attitude director indicator (EADI) display on the electronic flight instrument system (EFIS), and on the optional indexer. Information from the right system is presented on the left side of the copilot's EADI display. The EFIS displays are presented in a low speed awareness format. The stick shakers, which are mounted about nine inches down from the control wheel on the forward side of each pilot's control column, are energized by inputs from the angle-of-attack computers when they sense an impending stall. An electric motor with rotating weights induces a vibration feel to the control column, alerting the pilots to the approaching stall.

The aircraft is monitored for excessive angles-of-attack. At certain high altitudes, above 35,000 feet, these high angles-of-attack could disturb airflow into the engines enough to cause one or both to flame out. To prevent this from occurring, the minimum speed warning system was incorporated.

If the critical angle-of-attack is reached, with aircraft altitude above 35,000 feet MSL, EICAS will alert the crew with a red CAS message, MINIMUM SPEED. The pilot must push forward on the control column, to reduce AOA, and increase airspeed immediately to prevent further airspeed degradation. Once the AOA is decreased sufficiently, the MINIMUM SPEED message will extinguish.

The Minimum Speed protection is inhibited at flight altitudes less than 35,000 feet and/or any time the slat/flap handle has been placed into any detent.

If the engine indicating and crew alerting system (EICAS) senses a fault in the AOA system it will present an amber STALL WARN L-R message to warn the crew that the stall warning system (either left or right, as indicated) is not reliable. A left system warning means that the left stick shaker will not operate, the AOA indicator will show off, the optional indexer will be inoperative, and the pilot's low speed awareness display will be inoperative. A right system warning means that the copilot's fast/slow indicator and the stick shaker will be inoperative.

A message of AOA PROBE FAIL L-R on the EICAS indicates that there is a fault in the respective probe transmitter. That particular angle-of-attack system should not be used; the indications driven by the applicable (left or right) system, as listed above, should be disregarded.

An EICAS message AOA HEAT FAIL L-R indicates failure of the applicable AOA probe heater. In icing conditions, information from that system should be disregarded or used with caution.

When the airplane is powered up, the optional analog (pointer) type angle-of-attack indicator will go through a self test including the following:

1. The OFF flag pulls out of view.
2. The pointer is within the display range of the indicator

The rotary TEST switch on the center pedestal provides a means of checking the stick shakers and other features of the angle-of-attack system on preflight. To test the system select AOA on the test switch and observe the following:

1. The three lights in the AOA indexer shall illuminate one at a time during the test sequence.
2. The AOA indicator shall slew to the zero end of the scale.
3. The AOA indicator OFF flag shall be displayed as well as the following CAS messages:
  - a. STALL WARN L-R
  - b. AOA PROBE FAIL L
4. After a few seconds, the AOA OFF flag shall be removed and the indicator will move up scale.
5. As the pointer approaches .60, the INDEXER lights (if installed), shall cycle from yellow to green to red.
6. At approximately .80 normalized angle-of-attack (AOAN), the stick shaker should actuate. Vibration from left stick shaker should be felt through both control columns.
7. System should return to normal operation after a few seconds.

Digital crew alerting system (CAS) messages of an amber AOA PROBE FAIL L-R, and a red AUTO SLATS FAIL will appear in the engine indicating and crew alerting system (EICAS). If hydraulic pressure is available, the slats will deploy during this test. The AUTOSLATS FAIL message should extinguish when the rotary test switch is rotated out of the AOA position.