TECHNICAL MANUAL

GENERAL SYSTEMS

AUTO FLIGHT SYSTEM

FMS SERIES
F-15C AND F-15D
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NOTE: On a changed page, the portion of the text affected by the latest change is indicated by a vertical line in the outer margin of the page. Changes to illustrations are indicated by vertical lines, miniature pointing hands, shaded areas or a major change symbol.

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INTRODUCTION

PURPOSE AND SCOPE.

This technical manual is one of a series providing organizational maintenance instruction for the Automatic Flight Control System on F-15C and F-15D aircraft.

This manual has description, principles of operation and test equipment requirements for the autopilot system. Illustrations and schematics are incorporated as an aid in using this manual.

APPLICABILITY NOTATIONS.

Information and instructions contained in this manual peculiar to one model are identified as F-15C or F-15D.

Data applicable to particular aircraft within a series are identified by aircraft serial numbers.

ABBREVIATIONS AND SYMBOLS.

For the meaning of each nonstandard definition, symbol and acronym, an independent glossary follows section 11.

IMPROVEMENT REPORTS.

Recommendations for improvements to prescribed requirements and procedures will be submitted by AFTO Form 22, Technical Order System Publication mprovement Report, in accordance with TO 00-5-19. Completed forms shall be forwarded to Warner Robins ALC/LFI, 296 Cochran Street, Robins AFB, GA. 31098-1622.

RECORD OF APPLICABLE TIME COMPLIANCE TECHNICAL ORDERS.

The record of applicable time compliance technical orders is a list of all TCTO's which affect the technical content (text or illustration) of this manual. Only current TCTO's are listed. A TCTO is deleted from the list when any of the below occurs:

a. The equipment configuration to which the TCTO is applicable is no longer covered in the publication.

b. The TCTO is rescinded.

c. The TCTO is superseded or replaced.

Table 1: Record of Applicable Time Compliance Technical Orders

TCTO No.	Title	TCTO Date
	NONE	

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SECTION I

SYSTEM FUNCTIONAL DESCRIPTION

1-1. **AUTO FLIGHT.**

1-2. **DESCRIPTION**. This manual contains maintenance instructions for the Automatic Flight Control Set AN/ASW-38 which is made up of equipment listed in table 1-1. The Automatic Flight Control System (AFCS) provides control augmentation in pitch, roll, and yaw to the aircraft primary flight control system. The AFCS also provides the primary flight control system manual and automatic takeoff trim functions. With other aircraft systems, the AFCS provides automatic attitude and altitude holding. See figure 1-1 for a graphic illustration of system and for component location on the aircraft.

1-3. **Control Augmentation System (CAS).** CAS measures forces put on the control stick and rudder pedals, and compares input commands to aircraft response. If aircraft response exceeds input commands, CAS subtracts from control surface displacement until aircraft response matches force commands. Similarly, CAS adds surface displacement if aircraft response is less than force commands.

1-4. Since CAS commands large control surface movements, each CAS-axis has a self-monitoring dual channel system. If the two channels are not in agreement, the monitor system shuts off CAS for the malfunctioning axis and prevents control surface displacements which are not correct.

1-5. **Attitude Hold.** Automatic attitude holding is arrived at by using inertial navigation set (INS) pitch and roll attitude signals.

Table 1-1. Line Replaceable Units (LRU)

Common Name	S/S/SN	Ref Des	Nomenclature
Accelerometer Assembly	22-10-20	93A-D001	Acceleration Sensor Assembly TR-272/ASW
CAS Control Panel	22-10-21	93Z-H004	Engaging Controller, Automatic Flight Control System C-8982/ASW
Dynamic Pressure Sensor	22-10-22	93A-B003	Dynamic Pressure Sensor TR-276/ASW
Force Sensor	22-10-23	93Z-J005	Stick Force Sensor TR-274/ASW
Force Sensor (Rear Cockpit)	22-10-23	93Z-L002	Stick Force Sensor TR-274/ASW
Pitch Computer	22-10-25	93A-B006	Flight Control Computer CP-1104/ASW
Rate Gyro Assembly	22-10-26	93A-P009	Rate Sensor Assembly TR-273/ASW

Table 1-1. Line Replaceable Units (LRU) (CONT)

Common Name	S/S/SN	Ref Des	Nomenclature
Roll/Yaw Computer	22-10-24	93A-B008	Flight Control Computer CP-1105/ASW

1-6. **Altitude Hold.** Automatic altitude holding is done by using air data computer altitude signals and INS vertical velocity signals.

1-7. **Manual Trim.** Manual trim sets a reference for zero force in any axis. When the trim switch for any axis is actuated, the feel trim actuator is repositioned. Linear variable differential transformers (LVDT) in the actuator, supply dual signals to the roll/yaw computer. The trim signals alter zero force position reference. For example: If the pilot trims into a 2g turn at zero pitch force, longitudinal feel trim (pitch trim) actuator LVDT supply dual signals to the pitch computer (by way of the roll/yaw computer). The 2g becomes the reference for zero stick force instead of the normal 1g. Without the change in reference, pitch CAS authority overrides the trim input and keeps the aircraft flying at 1g.

1-8. **Automatic Takeoff Trim.** When the T/O TRIM switch is pressed, control surfaces and trim actuators for all axes are driven to best takeoff position. When the trim actuators are in the takeoff position, the T/O TRIM light on the CAS control panel comes on. When the switch is released, the light goes out.

1-9. **AFCS Test Set.** The AFCS test set automatically tests the AFCS LRU and related units in the primary flight control system. The AFCS test set diagnostic ability is expanded to isolate trim system malfunctions. With the pressure temperature test set, the AFCS test set detects malfunctions of the dynamic pressure sensor bellows.

1-10. **In-Flight Monitor, AFCS (IFM).** The inflight monitor (IFM) is flyable diagnostic monitoring equipment used to detect, memorize, and indicate in-flight AFCS and manual flight control system failures. Selected failures resulting in CAS disconnects are latched by set/reset indicators on the monitor control unit for postflight diagnosis. Use of the IFM expedites repair of in-flight malfunctions which cannot be duplicated during ground testing procedures In addition, the IFM can be used during ground testing to isolate intermittent malfunctions that the flight line test set fails to detect.

1-11. **COMPONENT DESCRIPTION.** See figure 1-1.

1-12. **CAS Control Panel (22-10-21).** The CAS control panel contains three dual pole lever lock toggle switches, two solenoid toggle switches, one pushbutton switch, and one indicating light. The unit is mounted on the forward section of the left console.

1-13. The three dual pole lever lock switches are the CAS PITCH, ROLL, and YAW. Each switch has three positions: OFF, RESET, and ON. CAS operation in each axis is arrived at when the switch for that axis is set to ON if initial engaging conditions are satisfied. The RESET position resets a failed axis following a malfunction. If a malfunction was momentary, CAS for the failed axis can be reset and engaged again.

1-14. The two solenoid toggle switches (ALT HOLD and ATT HOLD) are spring loaded to OFF and solenoid held to ON. When ALT HOLD and ATT HOLD switches are set to ON, pilot relief functions are established and remain in effect as long as switches remain ON. Switches may disengage manually or automatically. ALT HOLD disengages if the air data computer (ADC) or the INS become unreliable, ATT HOLD disengages if the INS becomes unreliable. Both switches are disengaged simultaneously if the control stick force sensor autopilot disengage switch ADS is pressed.

1-15. The T/O TRIM switch is used to automatically establish takeoff trim. The T/O TRIM light comes on when takeoff trim is

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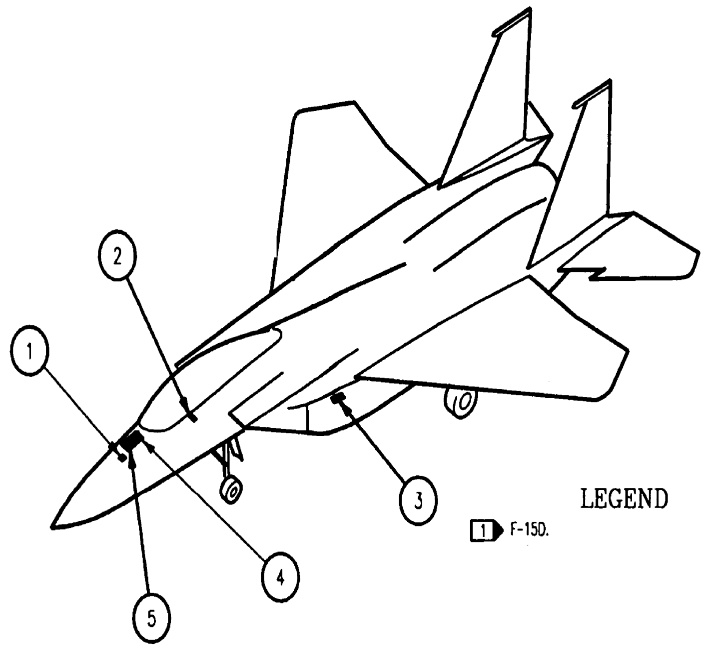

INDEX NO.	COMMON NAME	S/S/SN	REF DES	ACCESS
1	DYNAMIC PRESSURE SENSOR	22-10-22	93A-B003	DOOR 3R
2	ACCELEROMETER ASSEMBLY	22-10-20	93A-D001	DOOR 6R
3	RATE GYRO ASSEMBLY	22-10-26	93A-P009	DOOR 52L
4	PITCH COMPUTER	22-10-25	93A-B006	DOOR 3R
5	ROLL/YAW COMPUTER	22-10-24	93A-B008	DOOR 3R
6	MASTER CAUTION LIGHT	23-24-30		
7	RUDDER PEDAL ADJUST KNOB	27-20-18		
8	FORCE SENSOR	22-10-23	93Z-J005	
9	CAUTION LIGHT DISPLAY PANEL	33-10-28	35Z-J501	
10	CENTER CONSOLE RH CIRCUIT BREAKER PANEL	33-10-46		
11	CENTER CONSOLE LH CIRCUIT BREAKER PANEL	33-10-47		
12	GROUND POWER CONTROL PANEL	39-10-13	52Z-H069	
13	RUD TRIM SWITCH	76-10-23	93S-H018	COCKPIT
		1 ▶ 76-10-24	93S-H018	
14	FLAPS UP/DOWN SWITCH	27-50-13	46S-H006	
15	CAS CONTROL PANEL	22-10-21	93Z-H004	
16	ROLL RATIO SWITCH	39-10-14	19S-H006	
17	FLAP POSITION LIGHT	33-10-56	46DSH023	
18	PITCH RATIO SWITCH	27-40-18	19S-H007	
19	PITCH RATIO INDICATOR	27-40-19	19M-H001	
1 ▶ 20	MASTER CAUTION LIGHT	23-24-31		
1 ▶ 21	RUDDER PEDAL ADJUST KNOB	27-20-18		
1 ▶ 22	REAR FORCE SENSOR	22-10-23	93Z-L002	REAR COCKPIT
1 ▶ 23	RUDDER TRIM SWITCH	27-20-17	93S-K019	
1 ▶ 24	FLAP POSITION LIGHT	33-10-56	46DSK024	

Figure 1-1. Auto Flight System Component Location (Sheet 1 of 4)

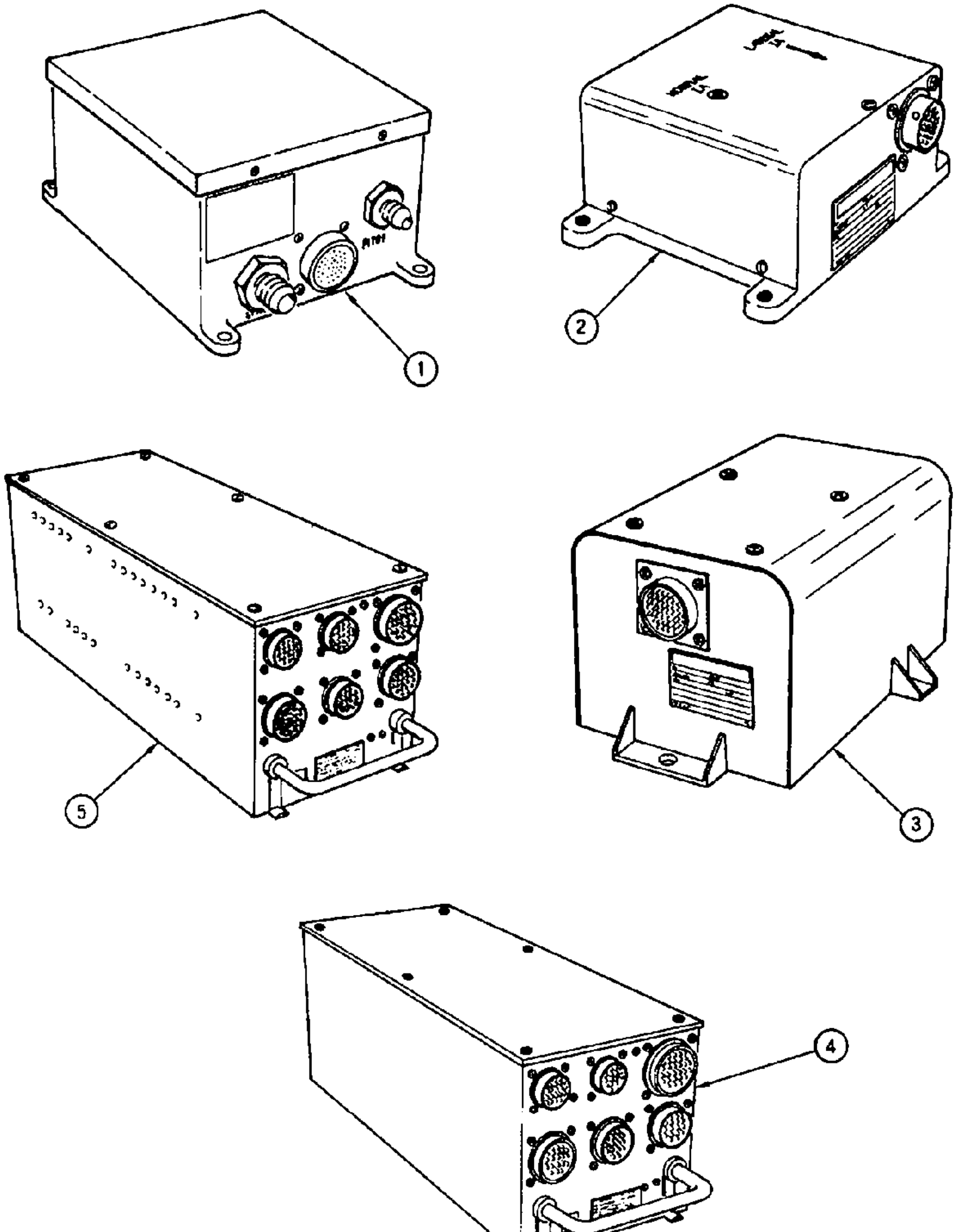

Figure 1-1. Auto Flight System Component Location (Sheet 2)

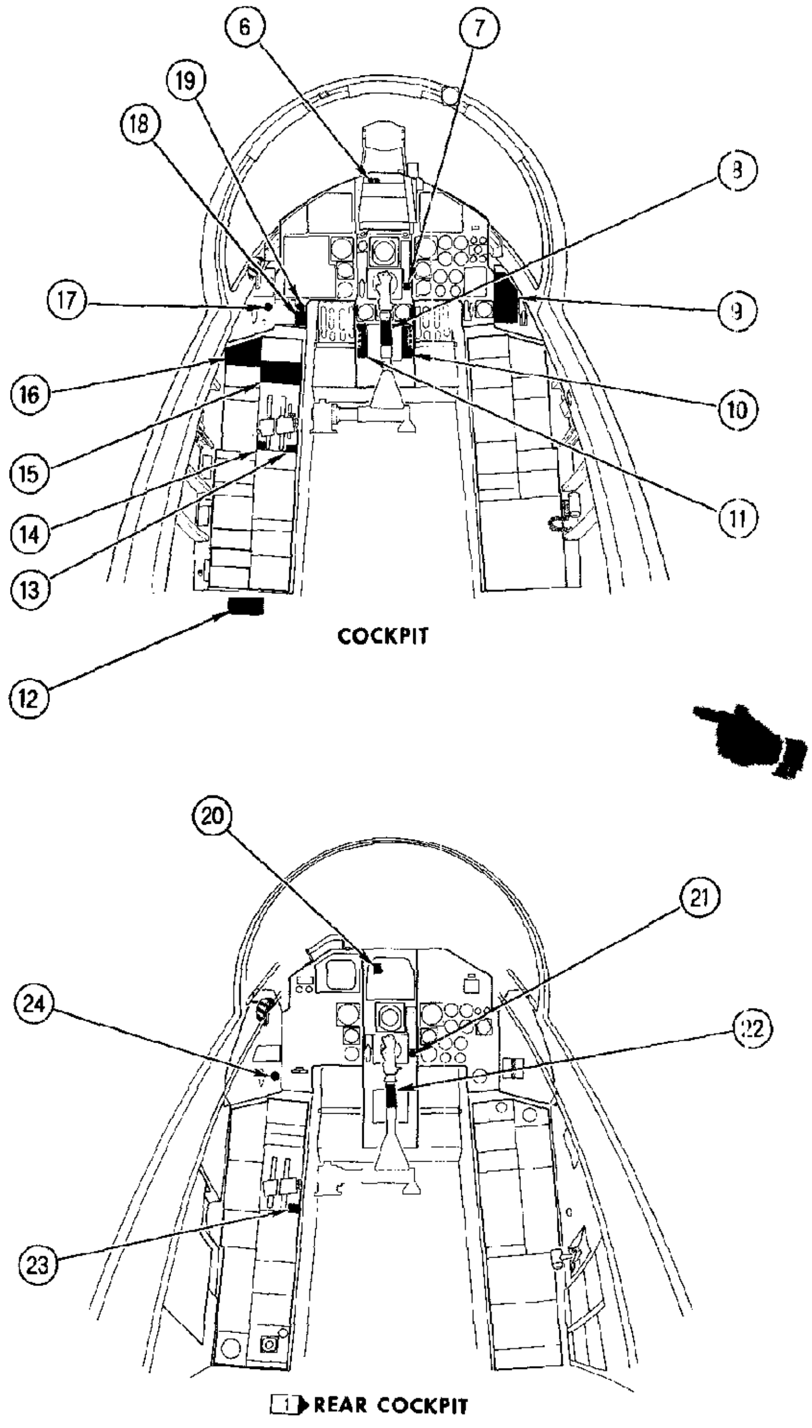

Figure 1-1. Auto Flight System Component Location (Sheet 3)

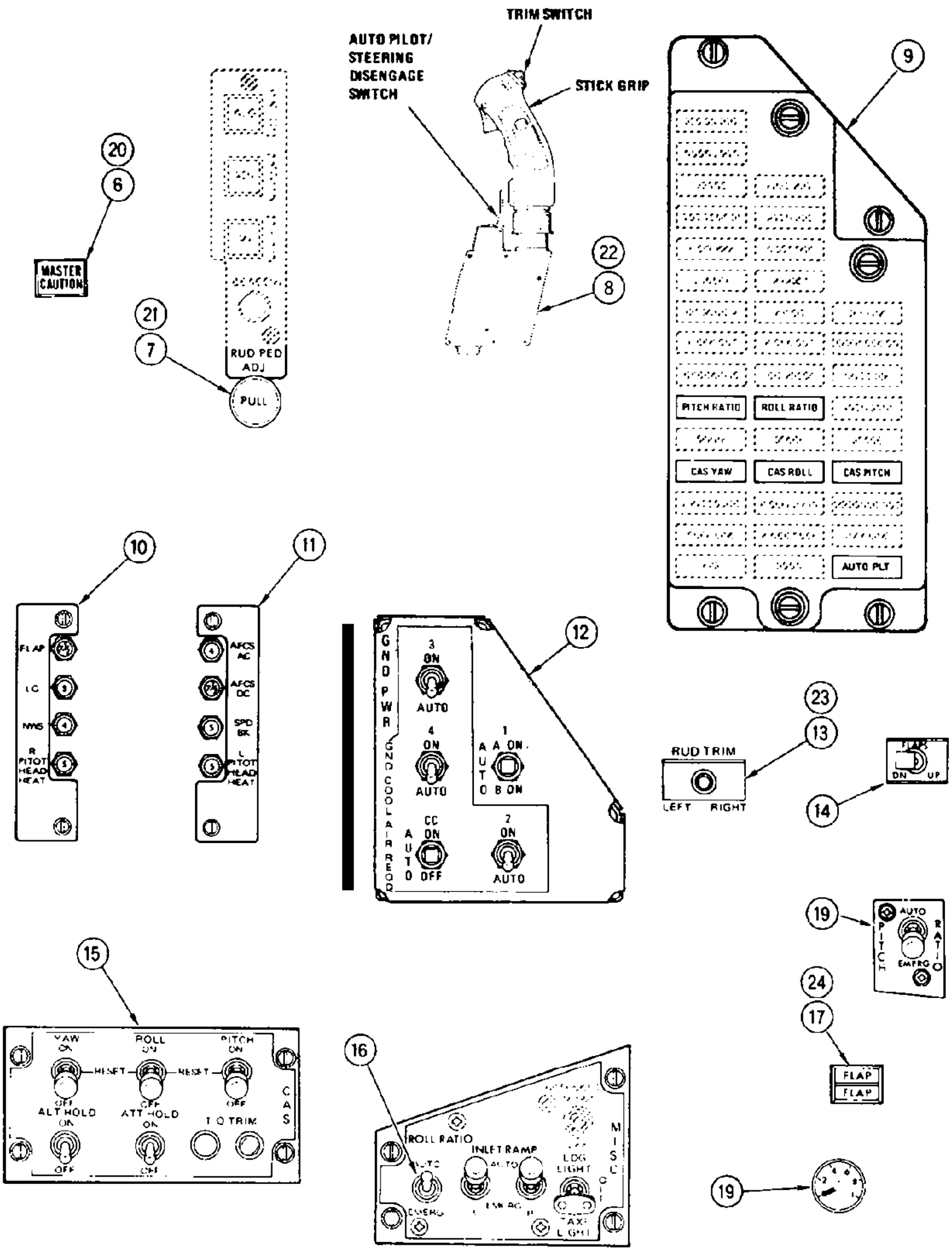

Figure 1-1. Auto Flight System Component Location (Sheet 4)

established. The CAS control panel does not contain a fault indicator.

1-16. **Pitch Computer (22-10-25).** The pitch computer receives and processes inputs from other AFCS components and from other aircraft systems to compute the required pitch outputs to the stabilator actuators in CAS, ALT HOLD and ATT HOLD modes. The computer, in back of door 3R, is similar to the roll/yaw computer in appearance.

1-17. **Roll/Yaw Computer (22-10-24).** The roll/yaw computer receives and processes inputs from other AFCS components and other aircraft systems to compute the required outputs to the stabilator and rudder actuators in CAS, ALT HOLD and ATT HOLD mode. The computer, in back of door 3R, is similar to the pitch computer in appearance.

1-18. **Dynamic Pressure Sensor (22-10-22).** The dynamic pressure sensor contains two differential pressure transducers driven by a bellows. The bellows are connected to pitot pressure and static pressure lines. The transducers supply dual electrical signals proportional to dynamic pressure (qc). The dynamic pressure signals are applied to the roll/yaw computer and are used to reduce total differential stabilator deflection in roll CAS at high speeds. The dynamic pressure sensor is in back of door 3R.

1-19. **Accelerometer Assembly (22-10-20).** The accelerometer assembly contains four accelerometers. Each sensor provides an electrical output signal proportional to aircraft acceleration along the two sensitive axes. Two sensors are oriented to sense normal acceleration and two are oriented to sense lateral acceleration. Normal acceleration outputs are applied to the pitch computer and are used in pitch CAS to monitor aircraft response to pilot commands. Lateral acceleration outputs are applied to the roll/yaw computer and are used to monitor aircraft response to pilot commands in the yaw CAS mode. The acceleration sensor assembly is back of door 6R.

1-20. **Rate Gyro Assembly (22-10-26).** The rate gyro assembly contains six rate gyros in a single housing. Two rate gyros provide pitch rate signals proportional to angular motion about the pitch axis. Two gyros supply roll rate signals and the other two gyros supply yaw rate signals. The pitch rate gyro outputs are applied to the pitch computer to monitor aircraft response to pilot commands in pitch CAS. The roll and yaw rate gyro outputs are applied to the roll/yaw computer and monitor aircraft response in roll and yaw CAS. The rate gyro assembly is in door 52L.

1-21. **Force Sensor (22-10-23).** The force sensor contains strain gage elements to measure forces applied to the control stick. The strain gage elements produce signals proportional to applied forces. The amplified signals are applied to the pitch computer and roll/yaw computer for use in pitch and roll CAS modes. The force sensor also contains the autopilot disengage switch and the nose gear steering switch. The sensor is installed between the control stick grip and the control stick column.

1-22. **RELATED EQUIPMENT.**

1-23. Table 1-2 provides a summary of related systems and functions

Table 1-2. Related Systems

System/Component	Function
1. Pitch and Roll Channel Assembly (PRCA).	a. Receives pitch computer pitch commands.
	b. Reverts to maximum roll rate if yaw increases to 60°/sec or more.
2. Stabilator Actuators.	Receive pitch computer pitch and roll commands.
3. Rudder Actuators.	Receive roll/yaw computer directional commands.

Table 1-2. Related Systems - CONT

System/Component	Function
4. Control Stick Grip.	Provides pitch and roll trim commands to roll/yaw computer.
5. RUD TRIM Switch.	Provides yaw trim signals to roll/yaw computer.
6. Pitch Trim Actuator.	Provides zero reference pitch trim to roll/yaw computer.
7. Roll Trim Actuator.	Provides zero reference roll trim to roll/yaw computer.
8. Yaw Trim Actuator.	Provides zero reference yaw trim and rudder pedal commands to yaw computer.
9. FLAPS Switch.	Alters angle-of-attack limitations on pitch CAS during landing by adding 10° AOA bias.
10. Bypass valve.	Energizes if yaw rate increases to 60°/sec or more.
11. Speed Brake.	Speed brake retracts automatically if true AOA increases to more than 15.5°.
12. Static Pressure Probe.	Provides static pressure to dynamic pressure sensor for qc computation.
13. Pitot Pressure Probe.	Provides pitot pressure to dynamic pressure sensor for qc computation.
14. Angle-of-Attack (AOA) Probe Transducers.	a. Provides left and right AOA signals to pitch computer for use in CAS operation stall inhibit.
	b. Provide left and right AOA signals to roll/yaw computer (by way of pitch computer) to control ARI schedule and roll CAS limiting.
15. Inertial Navigation Set (INS).	a. Provides pitch attitude signals to pitch computer for attitude hold mode.
	b. Provides roll attitude signals to roll/yaw computer for attitude hold mode.
	c. Provides vertical velocity signals to pitch computer for altitude hold mode.
	d. Provides attitude validity signal to pitch computer to enable attitude holding mode.
	e. Provides vertical velocity validity signal to pitch computer to enable altitude holding mode.
16. Air Data Computer (ADC).	a. Provides altitude error signals to pitch computer for altitude hold mode.
	b. Provides altitude error valid signal to pitch computer to enable altitude holding mode.

Table 1-2. Related Systems - (CONT)

System/Component	Function
17. Avionics Status Panel.	Receives AOA failure signals from pitch computer.
18. Caution Lights Logic Unit.	a. Receives CAS on/off signals. b. Receives and processes AUTO PLT caution light signals if ATT HOLD or ALT HOLD switches disengage.
19. Air Inlet Control System.	Provides ARI shutoff signal to roll/yaw computer and attenuates computed roll forces at mach 1.5.
20. Intercom.	Receives warning tone signal from roll/yaw computer if yaw rate goes above 30°/sec.

1-24. **PRINCIPLES OF OPERATION.**

1-25. AFCS electrical signals to the primary flight control systems provide the intelligence for control augmentation and pilot relief modes. To be familiar with AFCS operation, it is required to know primary flight control system operation. TO SR1F-15C-2-27GS-00-1 provides detailed primary flight control system principles of operation.

1-26. **PRIMARY FLIGHT CONTROLS.** The control stick and rudder pedals are the sources of pilot mechanical control. The mechanical inputs are transmitted to flight control surface actuators by way of the control stick boost pitch compensator (CSBPC). The primary flight control system enables the aircraft to continue a mission even after loss of CAS functions. Aircraft maneuvering capability is provided by the CSBPC. The CSBPC is a hydromechanical analog computer made up of two units: a pitch and roll channel assembly (PRCA) and an aileron rudder interconnect (ARD assembly.

1-27. The CSBPC provides the functions below:

a. Hydraulic boost of pilot effort in pitch and roll.

b. Variable mechanical advantage of pitch and roll control systems as a function of airspeed.

c. Automatic series pitch trim as a function of the difference between commanded and actuator aircraft load factor.

d. Decreased lateral control surface deflection and increased directional deflection for lateral stick displacement as a function of increased longitudinal stick deflection.

e. Correct control system mechanical advantages as a function of gear down and flaps down for landing.

1-28. Mechanical control system roll and pitch commands are conditioned in the CSBPC, combined in a lateral/longitudinal mixer linkage, and supplied by way of cables and push rods to aileron and stabilator actuators.

1-29. Spring cartridges and motor screwjacks (at the base of control stick and rudder pedals) provide the pilot with artificial feel force and trim ability.

1-30. **Longitudinal (Pitch) Mechanical Control System.** See figure 1-2. The pitch trim actuator position determines the neutral (zero force) control stick position in pitch.

1-31. The PRCA contains a pitch ratio changer (PRC). The PRC determines control system mechanical advantage (number of degrees of collective stabilator deflection per degree of longitudinal stick deflection) and computes the

1 FORWARD STICK
2 ARI
3 LONGITUDINAL INPUT TO ARI
4 MIXER LINKAGE
5 LATERAL INPUT
6 PRCA
7 LONGITUDINAL MASS BALANCE
8 PITCH TRIM ACTUATOR

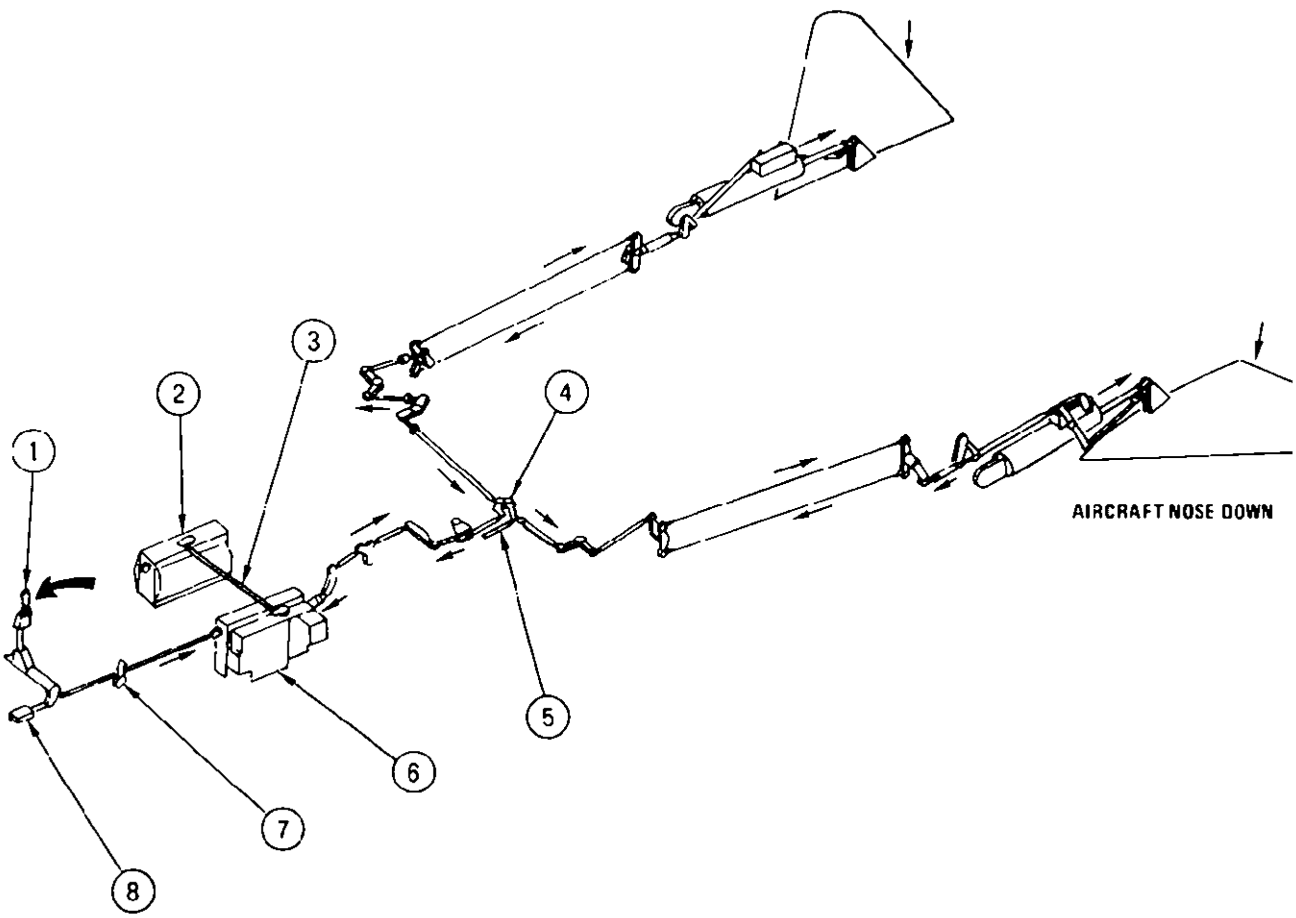

Figure 1-2. Simplified Longitudinal (Pitch) Mechanical Control System Diagram

All data on pages 1-10A and 1-10B deleted.

mechanical advantage. The PRC is a variable linkage positioned by a small hydraulic servoactuator. The actuator control valve is driven by an air data scheduler which uses a bellows to convert pneumatic signals from the aircraft pitot-static system into forces that position the valve as a function of both airspeed and mach number.

1-32. The PRCA also has a pitch trim controller (PTC). The PTC hydraulic actuator supplies series trim to the longitudinal control system. When PTC position is summed with the pilot trim position, the stabilator position changes without resultant stick motion. The PTC automatically compensates for trim changes caused by accelerating from subsonic to supersonic flight, operating flaps or speed brake, or stores separation. The PTC is driven by a control valve which operates as a function of normal acceleration (load factor) and control stick position. The PTC is damped to prevent disturbances from affecting the mechanical control system. PTC travel is limited as the PRC nears minimum ratio. A shutoff valve is added to the PRCA which defeats the PTC nose down trim limiter when (CFT) conformal fuel tanks are installed. When pitch CAS is engaged, the CAS interconnect (CASI) servo controls the PTC.

1-33. The sum of PRC position and PTC position drives a PRCA hydraulic boost actuator to move all the downstream linkage and controls the stabilator actuators. The PRCA boost actuator prevents the pilot from feeling friction and breakout forces in the rest of the linkage and prevents CAS commands inserted in the stabilator actuator from back-driving the control stick. The boost actuator also drives a linkage input to the PRCA roll axis and to the ARI.

1-34. The boosted pitch commands drive the lateral/longitudinal mixer linkage where pitch motions of the control system are summed with roll motions to drive the stabilator actuators. Each stabilator actuator is a dual tandem unit which uses two hydraulic systems simultaneously to drive each stabilator. Each actuator includes the CAS series servo which is used for both pitch and roll CAS functions. Detent springs attached to the input levers drive the stabilators to a position from which CAS can control the aircraft if the primary control system becomes disconnected. The stabilator main ram and CAS ram LVDT signals supply feedback information for pitch and roll CAS operation.

1-35. A longitudinal mass balance is mounted on the linkage between the control stick and the PRCA. The mass balance neutralizes the effect of acceleration on the control stick mass during maneuvers.

1-36. If hydraulic power is lost to the PRCA pitch axis, all PRCA actuators drive to a predetermined failed position and lock. The pilot can then operate the control stick and move the pitch control linkage without boost or pitch trim as in a conventional bellcrank aircraft control system. The same effect is arrived at when the PITCH RATIO switch is set to EMERG. If pitch CAS can be reset and engaged with the PITCH RATIO switch in EMERG, the aircraft can be maneuvered with AFCS signals.

1-37. **Lateral (Roll) Mechanical Control System.** See figure 1-3. The roll trim actuator position determines the control stick neutral (zero force) position in roll. The PRCA roll ratio changer (RRQ multiplies the control stick position by the control system mechanical advantage (number of degrees of aileron and differential stabilator deflection per degree of lateral stick deflection) and computes the mechanical advantage. The RRC is a variable linkage positioned by a small hydraulic servoactuator. The actuator control valve is driven by two inputs.

1-38. The first input is from an air data scheduler which determines the required roll ratio. The ratio is derived from bellows which uses pneumatic inputs from the aircraft pitot-static system to develop a valve control force proportional to airspeed. The computed roll ratio is compared to a second input: A ratio that is a function of pitch command supplied by a linkage to the roll ratio changer from the pitch booster. The ratio changer actuator drives to the smaller of the two ratios thus determined. The arrangement reduces lateral control surface displacements at high speed and during the large pitch maneuvers. Pilot selected stick position, modified by the RRC, drives through a hydraulic boost actuator that moves all the downstream linkage to the stabilator and ailerons actuators.

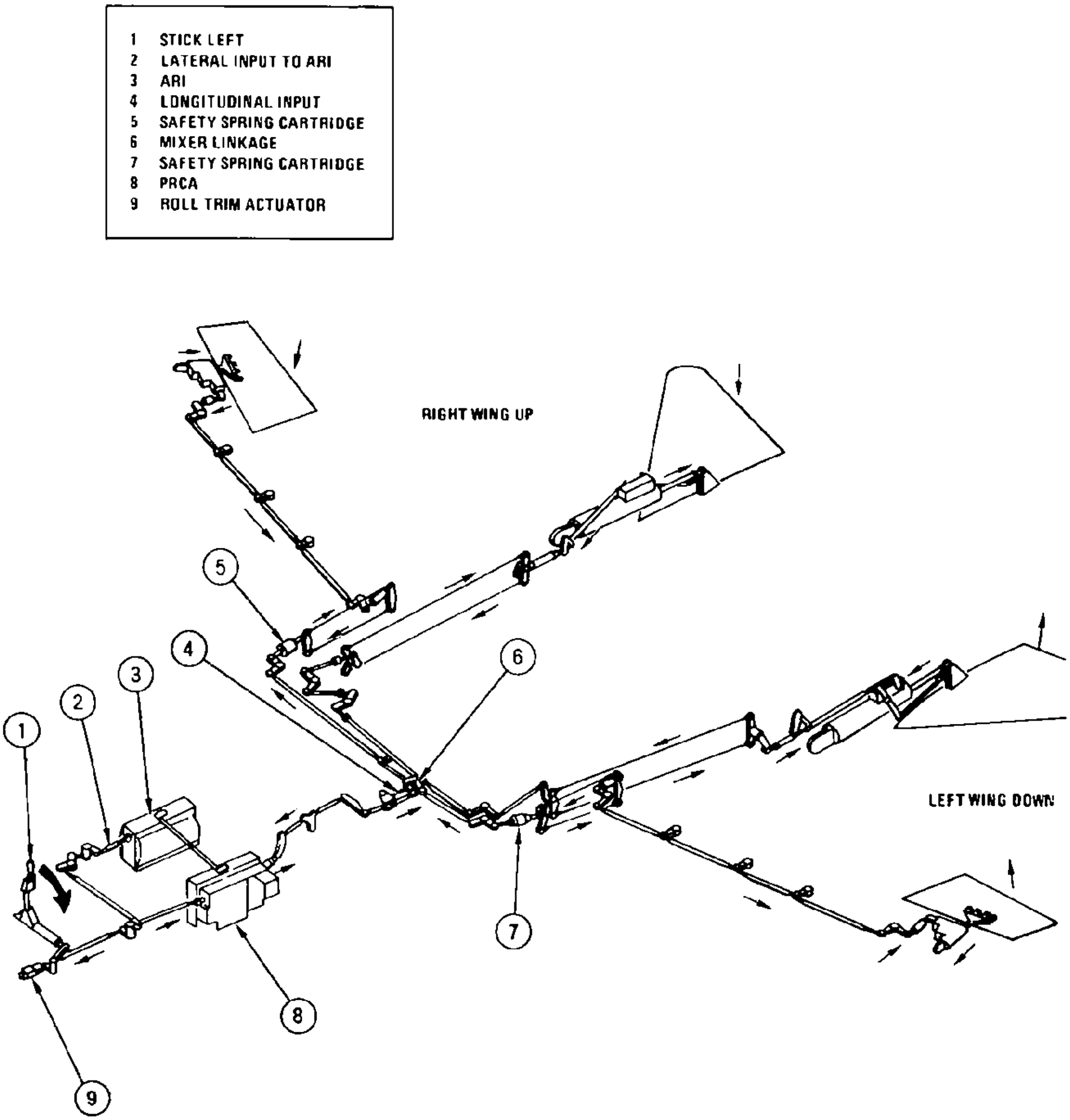

Figure 1-3. Simplified Lateral (Roll) Mechanical Control System Diagram

The roll linkage from the PRCA roll booster is sent to the aileron actuator in each wing through a safety override spring so that loss (jam) of one aileron linkage does not prevent continued use of the remaining roll control system. The linkage also goes to the lateral/longitudinal mixer and operates as described for the pitch system.

1-39. Lateral stick position is also supplied to the ARI. The ARI contains a separate ratio changer and boost actuator. The ARI ratio changer is normally set at zero ratio (no rudder motion in response to lateral stick travel), but the same pitch input which reduces the roll ratio setting in pitch maneuvers also drives the ARI ratio changer. As a result, lateral stick movements result in larger rudder deflections as pitch maneuvering is increased. Pilot selected roll stick position modified by the ratio changer, drives through the boost actuator that moves all the downstream linkage to the stabilator and aileron actuators.

1-40. The rudder schedule is such that aft stick travel increases the amount of right rudder deflection resulting from right stick inputs. Forward stick travel increases the amount of the left rudder deflection resulting from right stick inputs. The ARI is deactivated above mach 1.0 by a hydraulic switching signal supplied by the PRCA pitch axis. A flap down signal operates a solenoid valve which shifts the ARI schedule to the required landing configuration.

1-41. If hydraulic power to the PRCA roll axis is lost or the ROLL RATIO switch is set to EMERG, the ARI is deactivated and prevents any lateral stick movements from affecting the rudders. In addition, the roll ratio changer and ARI ratio changer drive to predetermined failed positions and lock. The two roll boost actuators center and lock. The pilot then controls the ailerons and differential stabilators at a fixed ratio without boost. The PRCA roll axis reverts to the equivalent of a simple bellcrank which provides the pilot with enough handling qualities for a safe return to base. If the pilot switches off the PRCA pitch axis, the roll axis continues to operate and vice versa; however, if either axis is switched off, the mechanical ARI function is deactivated.

1-42. **Directional (Rudder) Mechanical Control System.** See figure 1-4. The yaw trim actuator determines the neutral (zero force) rudder pedal position. The yaw trim actuator also provides the rudder pedal position signals to the AFCS for use in yaw CAS operation.

1-43. The mechanical system from the rudder pedals to the rudder actuators is made up almost completely of push-pull cable. The cable is a long, flexible ball bearing. The control element is a flat steel ribbon. The ribbon runs the length of the cable and rides on steel balls contained in a retainer strip on each side of the ribbon. A flexible case fits over the assembly. Although the ribbon bends in only one direction, it can be twisted so that bends can be made in various directions along its length. Friction and freeplay are higher with this type of control system than push-pull rods and bellcranks. The use of this cable has been confined to the rudder control system where there is more allowable friction and freeplay.

1-44. The rudder cables connect to the ARI box mixer linkage where lateral control inputs are added to the rudder pedal inputs. The roll input booster in the ARI box prevents rudder pedal inputs from feeding back into the lateral control system.

1-45. The rudder actuators have a rotary output and form a part of the rudder hinge. The control valve in the actuators receives enough mechanical input motion to allow one rudder to move through the travel limits even though the other rudder linkage is jammed or disabled.

1-46. **GENERAL AUTOMATIC FLIGHT CONTROL SYSTEM OPERATION.** See figure 1-5. The automatic flight control system aids the primary flight controls. It provides automatic (take off) and manual trim functions. The AFCS also increases control in pitch, roll and yaw, and extends pilot relief functions in attitude and altitude holding modes. The CAS control panel provides switching to engage selected modes.

1-47. When the T/O TRIM switch is pressed, the takeoff trim command is applied to the roll yaw computer. The trim circuits drive the roll trim and yaw trim actuators to neutral. The pitch trim actuator drives the control stick to approximately 1° aft of neutral. The trim actuators move the mechanical flight controls to

1 SAFETY SPRING CARTRIDGE
2 ARI
3 LONGITUDINAL INPUT FROM PRCA
4 LATERAL INPUT FROM STICK
5 LEFT RUDDER PEDAL
6 YAW TRIM ACTUATOR POSITION TRANSDUCER

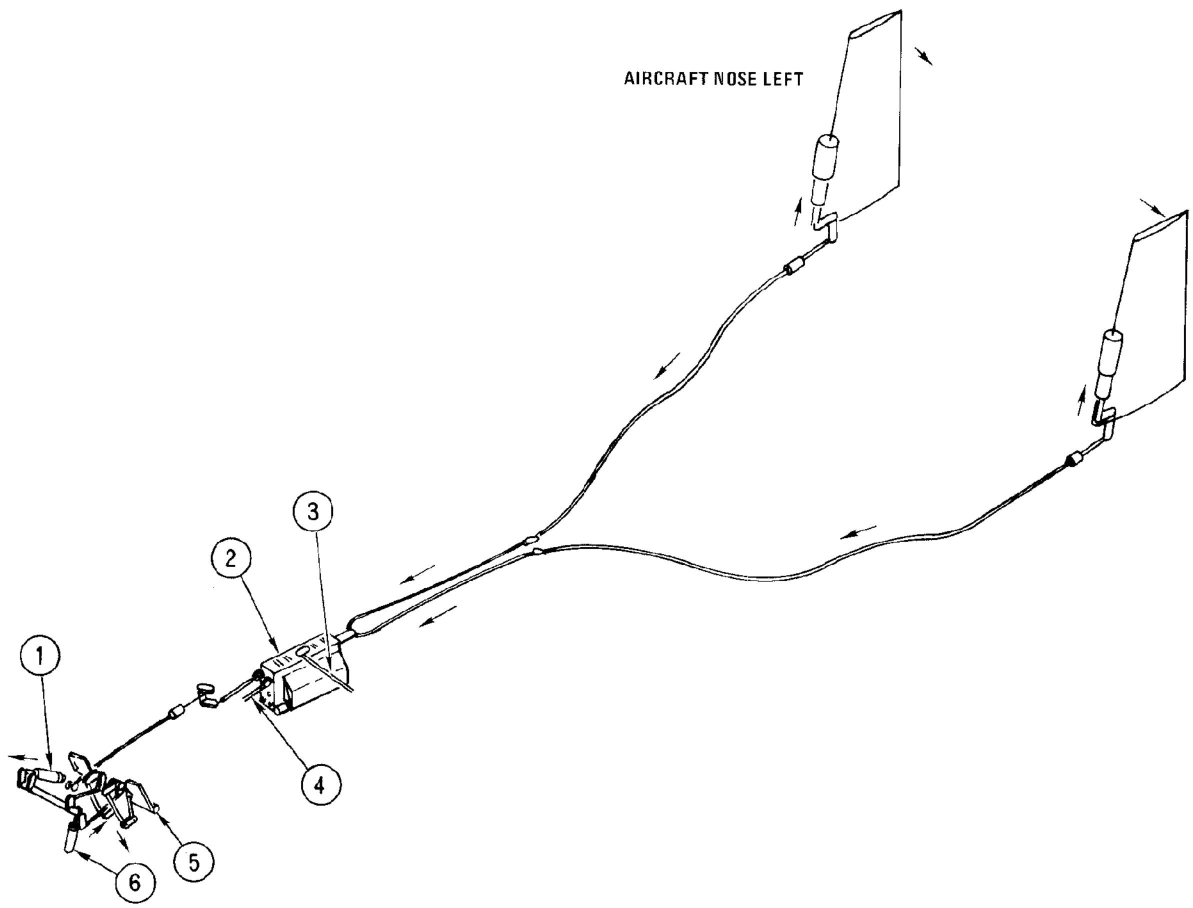

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Figure 1-4. Simplified Directional (Rudder) Mechanical Control System Diagram

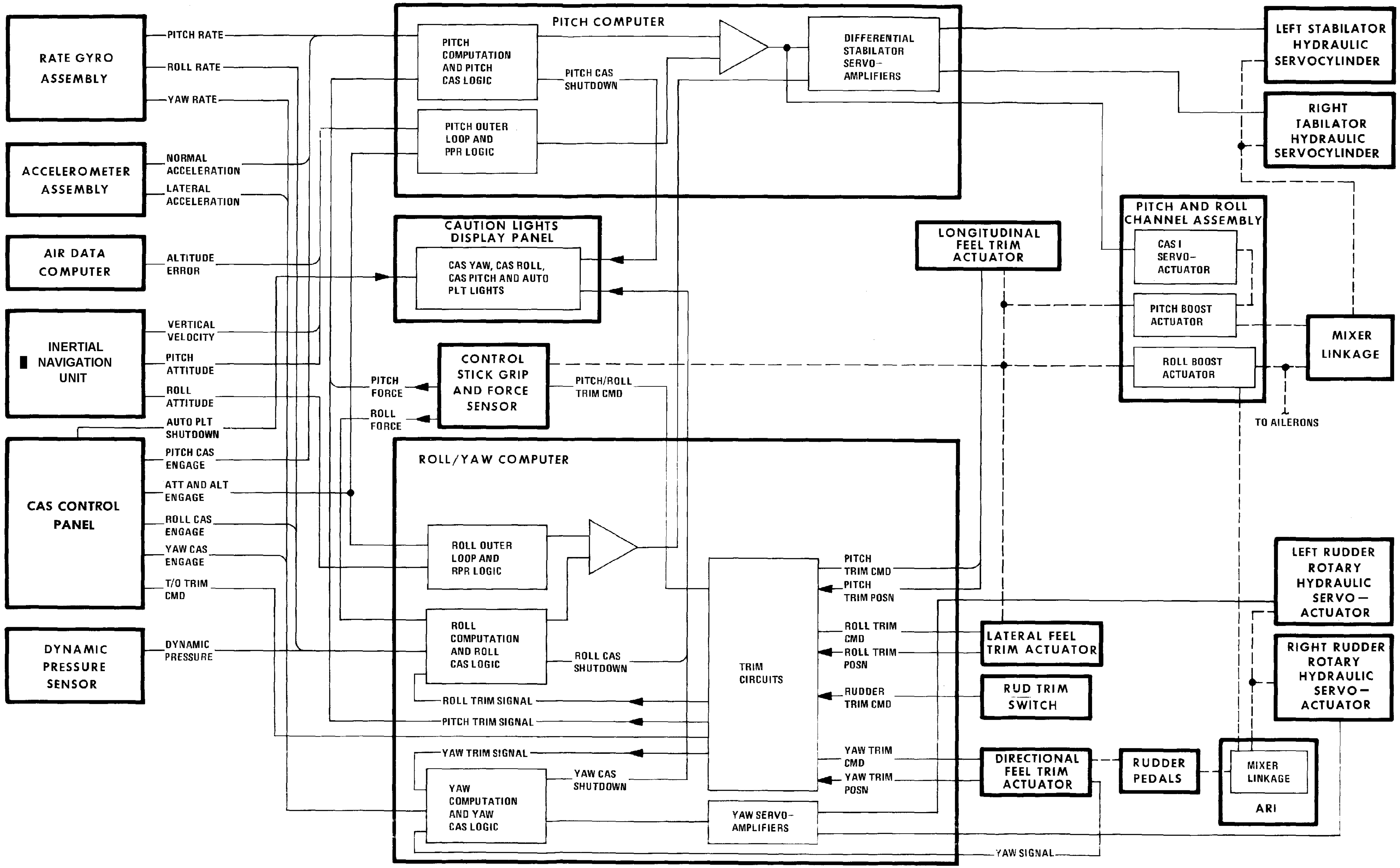

Figure 1-5. AFCS Block Diagram

the takeoff positions and (if hydraulic power is applied) the control surfaces move to neutral in roll and yaw and to approximately 5° nose up in pitch.

1-48. The PITCH, ROLL, and YAW CAS switches provide the inputs to the CAS logic circuits for each axis. The ATT and ALT switches provides the logic to engage the outer loops for pilot relief mode operation.

1-49. The control stick force sensors provide electrical signals proportional to the force applied to the control stick along the longitudinal and lateral axes. The rudder trim actuator position LVDT signals are proportional to rudder pedal position.

1-50. If CAS is engaged, pitch force sensor electrical signals drive the stabilator surfaces together. Roll force signals drive stabilator control surfaces differentially, but ailerons are not affected by roll force signals. Rudder pedal position LVDT signals drive rudder control surfaces.

1-51. When the pilot moves the control stick or rudder pedals, the mechanical control system moves the control surfaces in response to pilot commands. Control stick movement produces pitch and/or roll signals proportional to applied forces. When rudder pedals are moved, LVDT signals proportional to rudder pedal position are produced. As the aircraft responds to mechanical flight control system inputs, rate sensors and acceleration sensors detect and measure aircraft response to the commands.

1-52. The measured aircraft response to pilot commands is in the form of electrical signals. Signals resulting from aircraft response are compared to signals produces by pilot commands. If aircraft response is equal to pilot command, command signals are nulled and control surfaces are not affected. If measured response is smaller than pilot command, CAS adds control surface movement and CAS reduces control surface movement if measured response is larger than pilot command.

1-53. Yaw CAS uses rudder pedal position signals to measure pilot commands. Lateral acceleration sensors and yaw rate sensors measure aircraft response to pilot commands. In roll CAS, control stick roll force sensors measure pilot command and roll rate sensors provide measured response. Dynamic pressure sensors limit roll CAS authority as a function of airspeed.

1-54. Pitch CAS uses control stick force sensors to measure pilot commands. Pitch rate sensors and normal acceleration sensors provide measured aircraft response to pilot commands. Pitch signals are also applied to the PRCA to make sure that the mechanical system follows the CAS commanded maneuver. Since the mechanical system follows CAS commands, pitch disengage transients are minimized. Angle-of-attack sensors (not shown in block diagram) provide the required intelligence for stall warning, roll limiting, and ARI scheduling.

1-55. If ATT is engaged, inertial measurement unit pitch and roll signals are processed in the outer loops and combined with pitch and roll computation signals to command the stabilators to maintain the selected attitude. If ALT is engaged, air data computer altitude signals and inertial navigation unit (INU) velocity signals are used to maintain the required altitude.

1-56. The primary flight controls provide maneuvering abilities if the CAS fails. If the PRCA fails or if the mechanical control system linkages jam or break, AFCS CAS continues to provide maneuvering ability.

1-57. **DETAILED PRINCIPLES OF OPERATION.** Detailed AFCS principles of operation are given by dividing the system into the various functions done:

a. Pitch trim circuit.

b. Roll trim circuit.

c. Rudder trim circuit.

d. Pitch CAS engage logic circuit.

e. Pitch CAS circuit.

f. Simplified PRCA electrohydraulic operation.

g. Simplified stabilator electrohydraulic operation.

h. Differential series servo stabilators.

i. Roll CAS and engage logic circuit.

j. Yaw CAS and engage logic circuit.

k. Simplified rudder electrohydraulic operation.

l. Pilot relief engage logic circuit.

m. Pilot relief circuit.

1-58. Principles of operation are supported by simplified schematics in this section. Most AFCS functions are dual channel but are shown as single channel for clarity. Functions which are not dual channel and dual channel functions which are cross-coupled are so labeled.

1-59. **TRIM.** Manual trim and takeoff trim, position manual controls and control surfaces to required positions. Takeoff trim, positions control surfaces in best takeoff position (neutral roll and yaw and +5° pitch). Manual trim, reduces pilot control forces during steady-state maneuvers and compensates for asymmetrical aircraft loading, center-of-gravity changes, drift, and so forth.

1-60. **Pitch Trim.** See figure 1-6. Pitch trim manually compensates for nose heavy or tail heavy aircraft loading. The trim circuits also have an automatic takeoff trim ability.

1-61. Nose Up trim. To command a nose up trim, the control stick grip trim switch is held aft. When the switch is held aft, a logic low (ground) is applied to logic inverters 1 and 3. The high (5VDC) logic inverter 1 output is applied to OR gate 1. The high OR gate 1 output is immediately applied to transistor switch 1 and OR gate 4. The transistor switch conducts and applies 28VDC to pitch computer nose up relay K1. When K1 energizes, ground is applied to the trim motor. The high OR gate 4 output is applied to AND gate 5. If the trim motor is not at the nose up limit, the high logic inverter 4 output triggers OR gate 5. The combined OR gate 4 and OR gate 5 outputs trigger AND gate 5. The high logic inverter 3 output triggers OR gate 6. The combined AND gate 5 and OR gate 6 outputs trigger AND gate 6. After a small delay, the high AND gate 6 output is applied to transistor switch 2. The delay makes sure K1 is energized before power is applied to the trim motor. Transistor switch 2 applies 28VDC to pitch computer nose up/nose down relay K2. When K2 energizes, 115VAC is applied to the trim motor. The trim motor screwjack moves the control stick aft. As the trim motor screwjack moves, the pitch trim position LVDT A and LVDT B are repositioned. The LVDT signals are applied to the roll/yaw computer and summed with the zero trim bias at summing amplifiers 1A and 1B. The bias is used because the zero pitch trim LVDT output does not relate to the neutral position. The signals are buffered and demodulated. The demodulated signals are applied to the channel A and channel B takeoff trim circuits. The takeoff trim functions are not enabled if the T/O TRIM switch is not pressed. The channel A pitch trim position signal is also applied to the nose up and nose down limit level detectors. When the demodulated LVDT signal relates to the nose up trim limit, the high nose up limit level detector output is applied to logic inverter 4. The low logic inverter 4 output inhibits OR gate 5. The low OR gate 5 output inhibits AND gates 5 and 6. When AND gate 6 is inhibited, transistor switch 2 opens, K2 deenergizes, and the trim motor stops. As long as the trim switch is held aft, K1 is energized; the trim motor does not run because power is removed. If the trim switch is released before the nose up trim limit is arrived at, OR gate 1 and OR gate 6 are inhibited. The low OR gate 6 output inhibits AND gate 6 and transistor switch 2 opens. When transistor switch 2 opens, K2 deenergizes and power is removed from the trim motor. The low OR gate 1 output opens transistor switch 1 after a small delay. The delay makes sure that power is removed from the trim motor before K1 deenergizes.

1-62. Nose Down Trim. To command a nose down trim, the control stick grip trim switch is moved forward. When the switch is moved forward, a logic low is applied to logic inverters 2 and 3. The high logic inverter 2 output is applied to OR gate 3. The high logic inverter 3 output is applied to OR gate 6. The high OR gate 3 output is immediately applied to transistor switch 3 and to OR gate 5. Transistor switch 3 closes and applies 28VDC to nose down relay K3 in the pitch computer. When K3 energizes, ground is applied to the trim motor. The high OR gate 5 output is applied to AND gate 5. If the trim motor is not at the nose down limit, the nose down limit level

LEGEND

1. SOME DUAL CHANNEL TRIM FUNCTIONS ARE COMPLEMENTARY IN NATURE. ONLY THE TAKEOFF TRIM AND TRIM SIGNALS ARE TRUE DUAL CHANNEL FUNCTIONS.
2. COMPONENT NUMBERS ARE FOR REFERENCE ONLY.

(3) SEE SIMPLIFIED RUDDER TRIM SCHEMATIC.

(4) SEE ROLL/YAW DC POWER DISTRIBUTION SCHEMATIC (SD 22-00-16).

(5) SEE SIMPLIFIED PITCH CAS SCHEMATIC.

(6) SEE PITCH AC POWER DISTRIBUTION SCHEMATIC (SD 22-00-01).

(7) SEE LONGITUDINAL CONTROL SYSTEM SIMPLIFIED SYSTEM SCHEMATIC. REFER TO TO SR1F-15C-2-27GS-00-1.

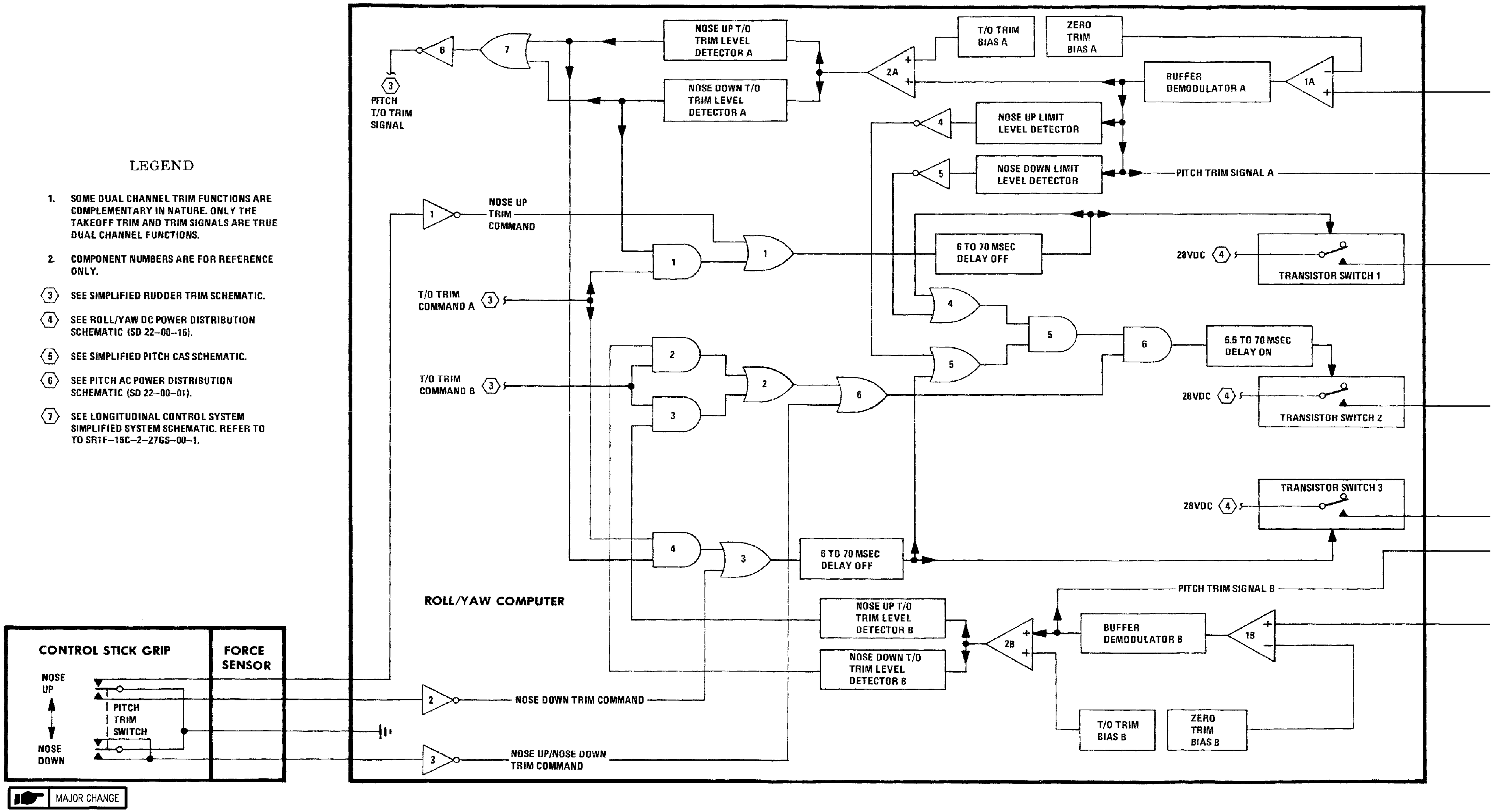

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Figure 1-6. Simplified Pitch Trim Schematic (Sheet 1 of 2)

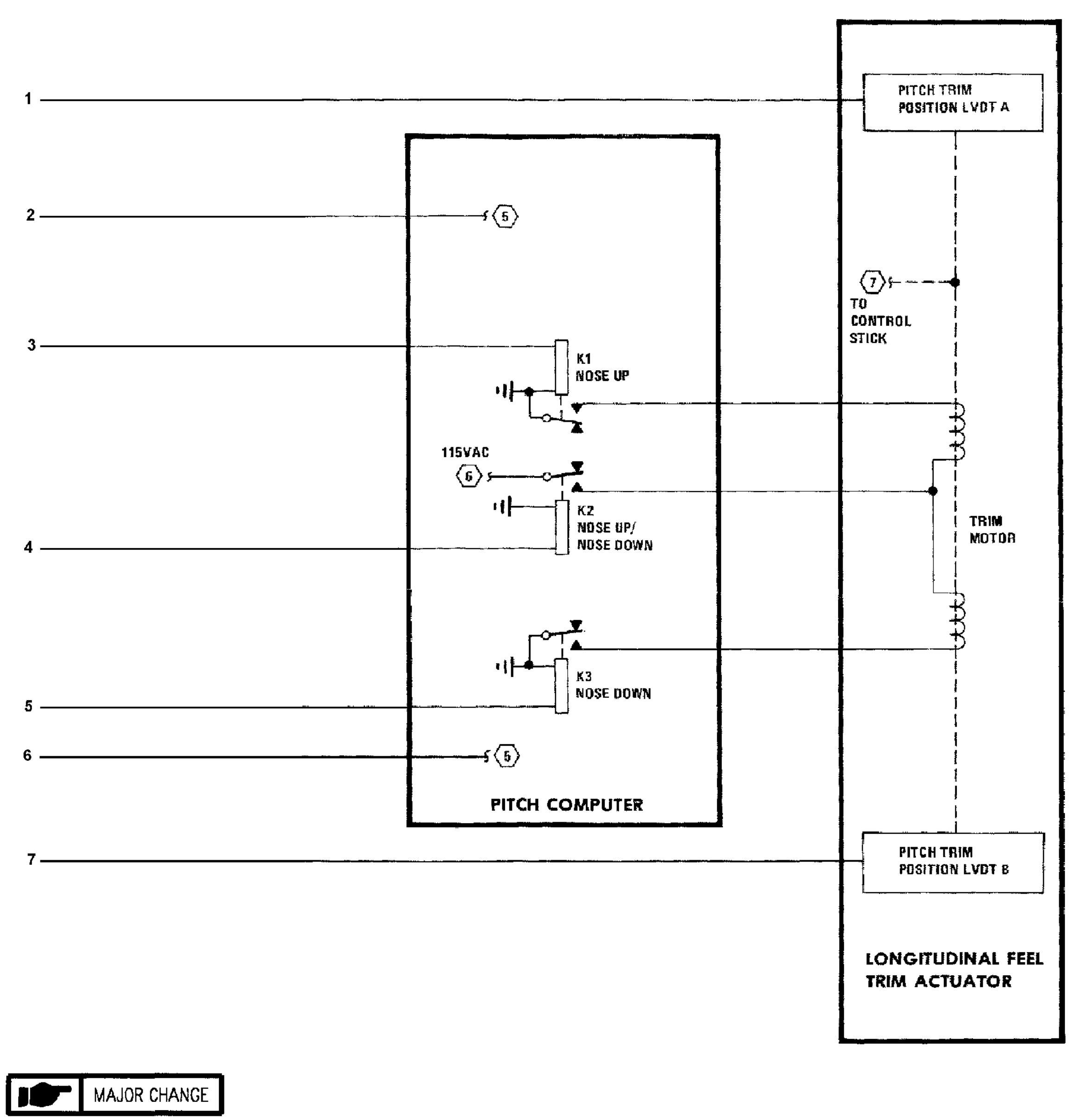

Figure 1-6. Simplified Pitch Trim Schematic (Sheet 2)

detector output is low. The high logic inverter 5 output triggers OR gate 4. The high OR gate 4 and OR gate 5 outputs trigger AND gate 5. The high AND gate 5 output combines with the high OR gate 6 output to trigger AND gate 6. After a small delay, transistor switch 2 closes, and applies 28VDC to the pitch computer nose/up down relay K2. The delay makes sure K3 is energized before power is applied to the trim motor. When K2 energizes, 115VAC is applied to the trim motor. The trim motor screw-jack moves the control stick forward. The pitch trim position LVDT A and LVDT B supply the pitch trim position to the roll/ yaw computer. When the demodulated channel A pitch trim position signal relates to the nose down trim limit, the logic inverter 5 output inhibits OR gate 4. The low OR gate 4 output inhibits AND gates 5 and 6. When AND gate 5 is inhibited, transistor switch 2 opens, K2 deenergizes, and the trim motor stops. As long as the trim switch is held forward, K3 is energized; the trim motor does not run because power is removed. If the trim switch is released before the nose down trim limit is arrived at, the trim motor stops when OR gate 6 is inhibited. K3 deenergizes shortly after OR gate 3 is inhibited.

1-63. Pitch Takeoff Trim. When the T/O TRIM switch is pressed, the high takeoff trim command A is applied to AND gates 1 and 4. T/O trim command B is applied to AND gates 2 and 3. Assuming that pitch trim is full nose down, the nose down takeoff trim level detector outputs (A and B) are high. The channel A level detector output is applied to AND gate 1. The channel B output is applied to AND gate 2. The high AND gate 1 output triggers OR gate 1. OR gate 1 triggers OR gate 4 and drives transistor switch 1. Since pitch trim is full nose down, the nose up limit detector output is low. The high logic inverter 4 output triggers OR gate 5. From this point the logic is the same as nose up trim until the takeoff trim position is arrived at. When the pitch trim position relates to takeoff trim, the takeoff trim nose down level detector outputs change to low. The low level detector outputs inhibit AND gates 1 and 2 and, then, OR gates 1 and 2. When OR gates 1 and 2 are inhibited, the trim motor stops as in manual nose up trim. If pitch trim is full nose up, the nose up takeoff trim level detector outputs are high. AND gates 3 and 4 are triggered if the takeoff trim commands exist. AND gate 4 triggers OR gate 3. The high OR gate 3 output triggers OR gate 5 and drives transistor switch 3. Since pitch trim is full nose up, the nose down limit level detector output is low. The high logic inverter 5 output triggers OR gate 4. From this point, the logic is the same as nose down trim until the takeoff trim position is arrived at. When the pitch trim position relates to the takeoff trim position, the nose up takeoff trim level detector outputs change to low. The low level detectors outputs inhibits AND gates 3 and 4 and, then, OR gates 2 and 3. When OR gates 2 and 3 are inhibited, the trim motor stops as in manual nose down trim. Takeoff trim bias is summed with the demodulated pitch trim signals at summing amplifiers 2A and 2B. The bias is required because the pitch takeoff trim position is approximately 1° of stick movement aft of neutral. When the trim actuator is at takeoff trim position, the nose up and nose down takeoff trim level detector outputs are low. The low channel A level detector outputs inhibit OR gate 7 and the logic inverter 6 output changes to high. The remaining pitch takeoff trim functions are explained with the rudder takeoff trim. Refer to paragraph 1-71.

1-64. **Roll Trim.** See figure 1-7. Roll trim manually compensates for asymmetrical aircraft loading. The trim circuits also have a automatic takeoff trim ability.

1-65. Right Roll Trim. To command a right roll trim, the control stick grip trim switch is held right. When the switch is held right, a logic low (ground) is applied to logic inverters 1 and 3. The high (5VDC) logic inverter 1 output is applied to OR gate 1. The high OR gate 1 output is immediately applied to transistor switch 1 and to OR gate 4. The transistor switch conducts and applies 28VDC to right roll relay K1. When K1 energizes, ground is applied to the trim motor. The high OR gate 4 output is applied to AND gate 5. If the trim motor is not at the right roll limit, the high logic inverter 4 output triggers OR gate 5. The combined OR gate 4 and OR gate 5 outputs trigger AND gate 5. The high logic inverter 3 output triggers OR gate 6. The combined AND gate 5 and OR gate 6 outputs trigger AND gate 6. After a small delay, the high AND gate 6 output is applied to transistor switch 2. The delay makes sure K1 is energized before power is applied to the trim motor. Transistor switch 2 applies 28VDC to the left/right roll relay

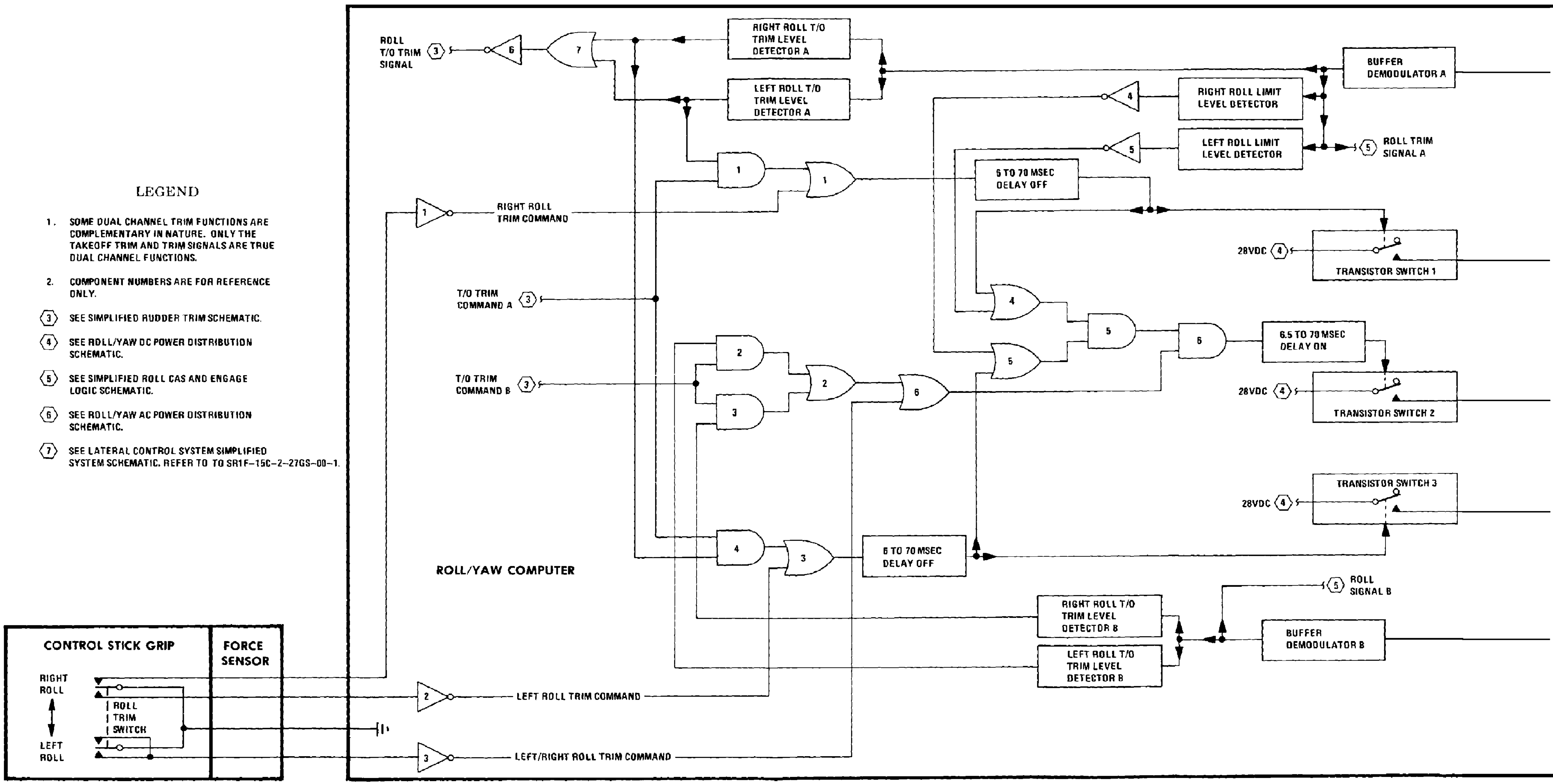

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Figure 1-7. Simplified Roll Trim Schematic (Sheet 1 of 2)

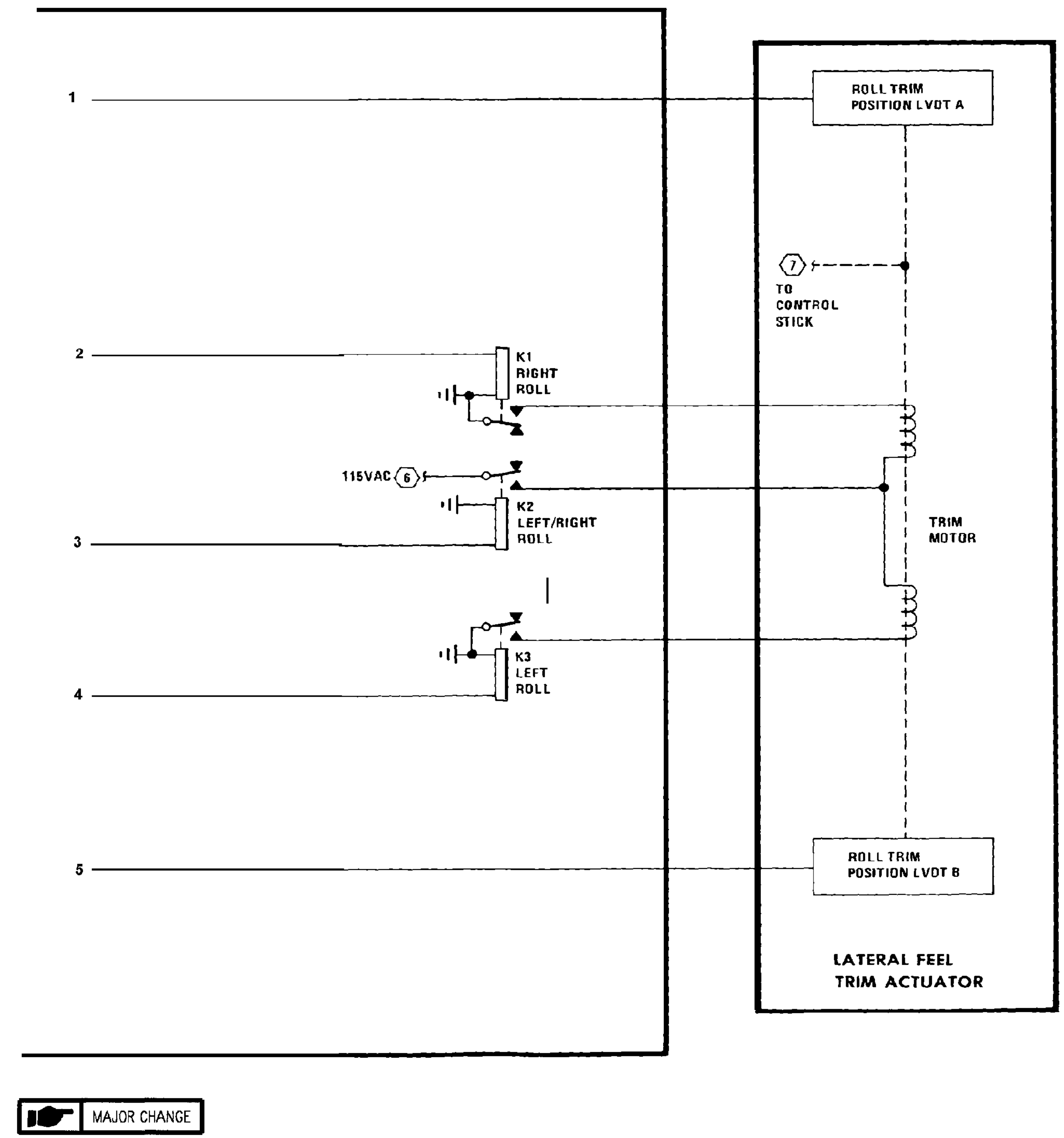

Figure 1-7. Simplified Roll Trim Schematic (Sheet 2)

K2. When K2 energizes, 115VAC is applied to the trim motor. The trim motor screwjack moves the control stick right. As the trim motor screwjack moves, the roll trim position LVDT A and LVDT B are repositioned. The LVDT signals are applied to the roll/yaw computer. The buffered and demodulated signals are applied to the channel A and channel B takeoff trim circuits. The takeoff trim functions are not enabled if the T/O TRIM switch is not pressed. The channel A roll trim position signal is also applied to the right roll and left roll limit level detectors. When the demodulated LVDT signal relates to the right roll limit, the high right roll limit level detector output is applied to logic inverter 4. The low logic inverter 4 output inhibits OR gate 5. The low OR gate 5 output inhibits AND gates 5 and 6. When AND gate 6 is inhibited, transistor switch 2 opens, K2 deenergizes, and the trim motor stops. As long as the trim switch is held right, K1 is energized; the trim motor does not run because power is removed. If the trim switch is released before the right roll trim limit is arrived at, OR gate 1 and OR gate 6 are inhibited. The low OR gate 6 output inhibits AND gate 6 and transistor switch 2 opens. When transistor switch 1 opens, K2 deenergizes and power is removed from the trim motor. The low OR gate 1 output opens transistor switch 1 after a small delay. The delay makes sure power is removed from the trim motor before 1 deenergizes.

1-66. Left Roll Trim. To command a left roll trim, the control stick grip trim switch is moved left. When the switch is moved left, a logic low is applied to logic inverters 2 and 3. The high logic inverter 2 output is applied to OR gate 3. The high logic inverter 3 output is applied to OR gate 6. The high OR gate 3 output is immediately applied to transistor switch 3 and to OR gate 5. Transistor switch 3 closes and applies 28VDC to left roll relay K3. When K3 energizes, ground is applied to the trim motor. The high OR gate 5 output is applied to AND gate 5. If the trim motor is not at the left roll limit, the left roll level detector output is low. The high logic inverter 5 output triggers OR gate 4. The high OR gate 4 and OR gate 5 outputs trigger AND gate 5. The high AND gate 5 output combines with the high OR gate 6 output to trigger AND gate 6. After a small delay, transistor switch 2 closes and applies 28VDC to the left/right roll relay K2. The delay makes sure K3 is energized before power is applied to the trim motor. When K2 energizes, 115VAC is applied to the trim motor. The trim motor screwjack moves the control stick left. The roll trim position LVDT A and LVDT B supply the roll trim position to the roll/yaw computer. When the demodulated channel A roll trim position signal relates to the left roll trim limit, the logic inverter 5 output inhibits OR gate 4. The low OR gate 4 output inhibits AND gates 5 and 6. When AND gate 5 is inhibited, transistor switch 2 opens, K2 deenergizes, and the trim motor stops. As long as the trim switch is held left, K3 is energized; the trim motor does not run because power is removed. If the trim switch is released before the left roll trim limit is arrived at, the trim motor stops when OR gate 6 is inhibited. K3 deenergizes shortly after OR gate 3 is inhibited.

1-67. Roll Takeoff Trim. When T/O TRIM switch is pressed, the high takeoff trim command A is applied to AND gates 1 and 4. The takeoff trim command B is applied to AND gates 2 and 3. Assuming that roll trim is full left, the left roll takeoff trim level detector outputs (A and B) are high. The channel A level detector output is applied to AND gate 1. The channel B output is applied to AND gate 2. The high AND gate 1 output triggers OR gate 1. OR gate 1 triggers OR gate 4 and drives transistor switch 1. Since roll trim is full left, the right roll limiter detector output is low. The high logic inverter 4 output triggers OR gate 5. From this point the logic is the same as right roll trim until the takeoff trim position is arrived at. When the roll trim position relates to takeoff trim, the takeoff trim left roll level detector outputs change to low. The low level detector outputs inhibits AND gates 1 and 2 and, then, OR gates 1 and 2. When OR gates 1 and 2 are inhibited, the trim motor stops as in manual left roll trim. If roll trim is full right, the right roll takeoff trim level detector outputs are high, AND gates 3 and 4 are triggered if the takeoff trim commands are present. AND gate 4 triggers OR gate 3. The high OR gate 3 output triggers OR gate 5 and drives transistor switch 3. Since roll trim is full right, the left roll limit level detector output is low. The high logic inverter 5 output triggers OR gate 4. From this point, the logic is the same as left roll trim until the takeoff trim position is arrived at. When the roll trim position corresponds to the takeoff trim position, the right roll takeoff trim level detector outputs change to low. The low level detector outputs inhibit AND gates 3 and 4 and, then OR gates 2 and 3. When OR gates 2 and 3 are inhibited, the trim motor stops as in manual right roll trim. When the trim actuator is at takeoff trim position, the right roll and left roll takeoff trim level detector outputs are low. The low channel A level detector outputs inhibit OR gate 7 and the logic inverter 6 output changes to high. The remaining roll takeoff trim functions are explained with the rudder takeoff trim. Refer to paragraph 1-71.

1-68. **Rudder Trim.** See figure 1-8. Rudder trim manually compensates for aircraft drift. The trim circuits also have an automatic takeoff trim capability.

1-69. Nose Right Trim. To command a nose right trim, the throttle panel RUD TRIM switch is held right. When the switch is held right, a logic low (ground) is applied to logic inverters 1 and 3. The high (5VDC) logic inverter 1 output is applied to OR gate 1. The high OR gate 1 outputs immediately applied to transistor switch 1 and to OR gate 4. The transistor switch conducts and applies 28VDC to directional feel trim actuator nose right relay K1. When K1 energizes, ground is applied to the trim motor. The high OR gate 4 output is applied to AND gate 5. If the trim motor is not at the nose right limit, the high logic inverter 4 output triggers OR gate 5. The combined OR gate 4 and OR gate 5 outputs trigger AND gate 5. The high logic inverter 3 output triggers OR gate 6. The combined AND gate 5 and OR gate 6 outputs trigger AND gate 6. After a small delay, the high AND gate 6 output is applied to transistor switch 2. The delay makes sure K1 is energized before power is applied to the trim motor. Transistor switch 2 applies 28VDC to the trim actuator nose left/nose right relay K2. When K2 energizes, 115VAC is applied to the trim motor. The trim motor screwjack moves the right rudder pedal forward. As the trim motor screwjack moves, the rudder trim position LVDT A and LVDT B are repositioned. The trim position LVDT signals are applied to the roll/yaw computer. The rudder pedal position LVDT A and LVDT B are also repositioned. The function of the rudder pedal position signal is explained in the yaw CAS theory of operation. The buffered and demodulated signals are applied to the channel A and channel B takeoff trim circuits. The takeoff trim functions are not enabled if the T/O TRIM switch is not pressed.

The channel A rudder trim position signal is also applied to the nose right and nose left limit level detectors. When the demodulated LVDT signal relates to the nose right limit, the high nose right limit level detector output is applied to logic inverter 4. The low logic inverter 4 output inhibits OR gate 5. The low OR gate 5 output inhibits AND gates 5 and 6. When AND gate 6 is inhibited, transistor switch 2 opens, K2 deenergizes, and the trim motor stops. As long as the trim switch is held right, K1 is energized; the trim motor does not run because power is removed. If the trim switch is released before the nose right trim limit is arrived at, OR gate 1 and OR gate 6 are inhibited. The low OR gate 6 output inhibits AND gate 6 and transistor switch 2 opens. When transistor switch 2 opens, K2 deenergizes and power is removed from the trim motor. The low OR gate 1 output opens transistor switch 1 after a small delay. The delay makes sure that power is removed from the trim motor before K1 deenergizes.

1-70. Nose Left Trim. To command a left yaw trim, the RUD TRIM switch is moved left. When the switch is moved left, a logic low is applied to logic inverters 2 and 3. The high logic inverter 2 output is applied to OR gate 3. The high logic inverter 3 output is applied to OR gate 6. The high OR gate 3 output is immediately applied to transistor switch 3 and to OR gate 5. Transistor switch 3 closes and applies 28VDC to nose left relay K3. When K3 energizes, ground is applied to the trim motor. The high OR gate 5 output is applied to AND gate 5. If the trim motor is not at the nose left limit, the nose left limit level detector output is low. The high logic inverter 5 output triggers OR gate 4. The high OR gate 4 and OR gate 5 outputs trigger AND gate 5. The high AND gate 5 output combines with the high OR gate 6 output to trigger AND gate 6. After a small delay, transistor switch 2 closes and applies 28VDC to the nose left/nose right relay K2. The delay makes sure K3 is energized before power is applied to the trim motor. When K2 energizes, 115VAC is applied to the trim motor. The trim motor screwjack moves the left rudder pedal forward. The rudder trim position LVDT A and LVDT B supply the rudder trim position to the roll/yaw computer. When the demodulated channel A rudder trim position signal relates to the nose left trim limit, the logic inverter 5

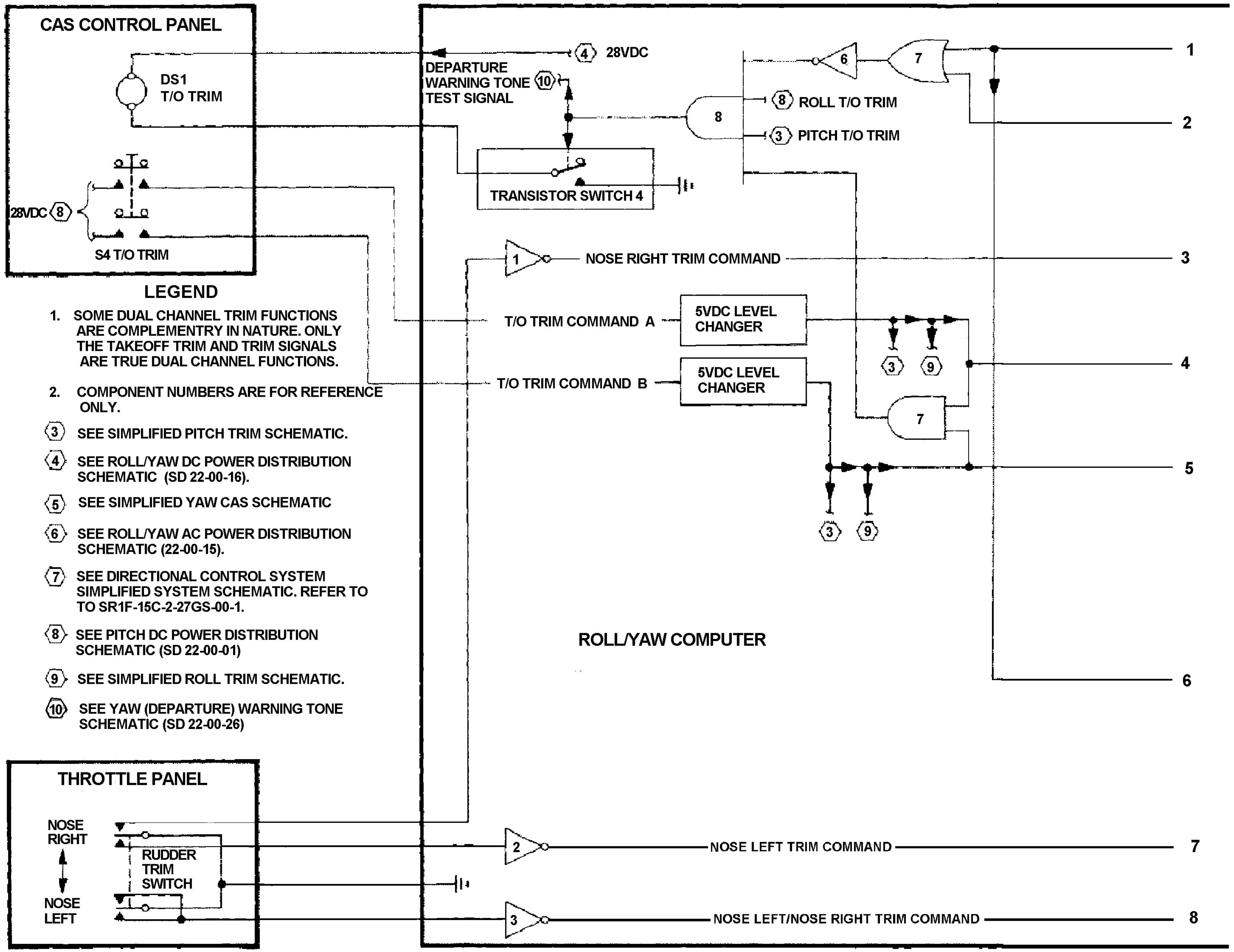

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Figure 1-8. Simplified Rudder Trim Schematic (Sheet 1 of 2)

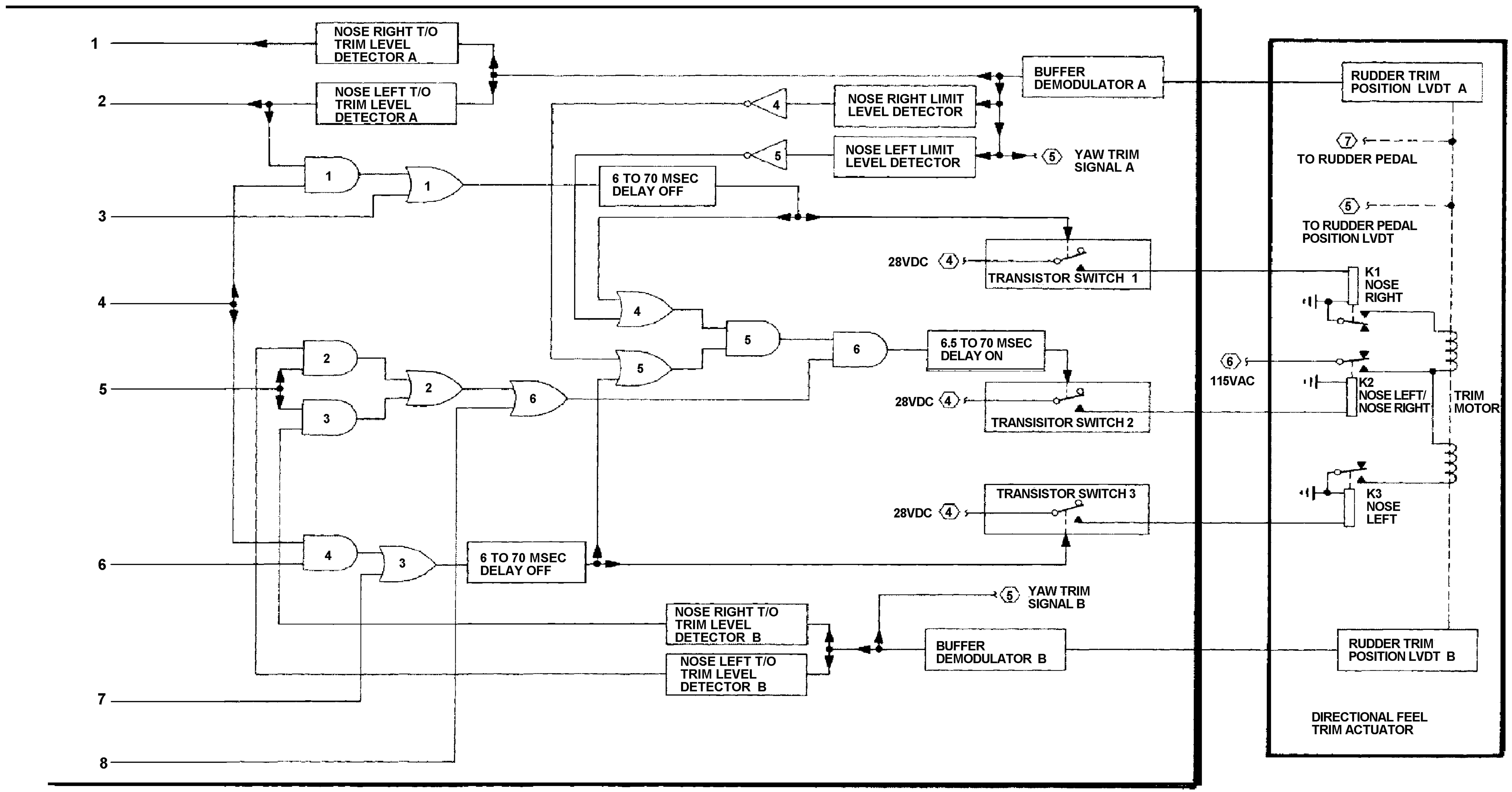

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Figure 1-8. Simplified Rudder Trim Schematic (Sheet 2) output inhibits OR gate 4. The low OR gate 4 output inhibits AND gates 5 and 6. When AND gate 5 is inhibited, transistor switch 2 opens, K2 deenergizes, and the trim motor stops. As long as the RUD TRIM switch is held left K3 is energized; the trim motor does not run because power is removed. If the trim switch is released before the nose left trim limit is arrived at, the trim motor stops when OR gate 6 is inhibited. K3 deenergizes shortly after OR gate 3 is inhibited.

1-71. **Rudder Takeoff Trim.** When the T/O TRIM switch is pressed, the high takeoff trim command A is applied to AND gates 1 and 4. The takeoff trim command B is applied to AND gates 2 and 3. Assuming that rudder trim is full left, the nose left takeoff trim level detector outputs (A and B) are high. The channel A level detector output is applied to AND gate 1. The channel B output is applied to AND gate 2. The high AND gate 1 output triggers OR gate 1. OR gate 1 triggers OR gate 4 and drives transistor switch 1. Since rudder trim is full left, the nose right limit level detector output is low. The high logic inverter 4 output triggers OR gate 5. From this point, the logic is the same as nose right trim until the takeoff trim position is arrived at. When the yaw trim position relates to takeoff trim, the takeoff trim nose left level detector outputs change to low. The low level detector outputs inhibit AND gates 1 and 2 and then OR gates 1 and 2. When OR gates 1 and 2 are inhibited, the trim motor stops as in manual nose left trim. If the rudder trim is full right, the nose right takeoff trim level detector outputs are high. AND gates 3 and 4 are triggered if the takeoff trim commands exist. AND gate 4 triggers OR gate 3. The high OR output triggers OR gate 5 and drives transistor switch 3. Since rudder trim is full right, the nose left limit level detector output is low. The high logic inverter 5 output triggers OR gate 4. From this point, the logic is the same as nose left trim until the takeoff trim position is arrived at. When the yaw trim position relates to the takeoff trim position, the nose right takeoff trim level detector outputs change to low. The low level detector outputs inhibit AND gates 3 and 4 and, consequently, OR gates 2 and 3. When OR gates 2 and 3 are inhibited, the trim motor stops as in manual nose right trim. When the trim actuator is at takeoff trim position, the nose right and left yaw takeoff level detector outputs are low. The low channel A level detector outputs inhibit OR gate 7 and the logic inverter 6 output changes to high. The high logic inverter 6 output combines with the pitch and roll takeoff trim signals at AND gate 8. The takeoff trim commands A and B trigger AND gate 7. The high AND gate 7 output triggers AND gate 8 when all trim actuators are at takeoff trim. AND gate 8 activates transistor switch 4. Switch 4 provides a ground for the T/O TRIM light. When the T/O TRIM switch is released, AND gate 7 is inhibited and the T/O TRIM light goes out.

1-72. **PITCH CAS OPERATION.** When forward or aft force is applied to the control stick, the mechanical flight control system deflects the stabilators. Simultaneously, the control stick pitch force sensors produces signals proportional to the applied force. The normal accelerometers and the pitch rate gyros measure aircraft response to the command pitch input. The measured response is compared to the command electrical force inputs from the pitch force sensors.

1-73. If no difference exists between commanded pitch and aircraft response, the pitch force signals are nulled. If aircraft response is more or less than commanded, the difference is applied (through a ±10° limiter) to the stabilator servoactuators. The servoactuators add or subtract as much as 10° of collective stabilator displacement to the position established by the hydromechanical control system. Most control augmentation components are dual channel. If the dual channels differ by an excessive amount, pitch CAS disengages automatically and the servoactuators drive to the neutral position.

1-74. Limited pitch commands are applied to the PRCA CAS interconnect (CASI) servo. The CASI servo drives the PRCA pitch trim compensator and forces the mechanical system to track the pitch CAS. The tracking function minimizes differences between CAS and mechanical commands so that if pitch CAS fails, the mechanical system can take control at the point of failure.

1-75. **PITCH CAS ENGAGE LOGIC**. See figure 1-9. The pitch CAS logic circuit indicates how the pitch reset/engage switches are used to energize the PRCA and stabilator shutoff valve and to turn off the caution panel CAS PITCH light. When the PITCH CAS switch is set from OFF to RESET and then to ON, 28VDC is applied to switches 1 and 2. At the same time, the 15 to 100 msec one-shot multivibrator provides a pulseto OR gate 2 and applies a high to AND gate 7. When OR gate 2 goes high, one high input is provided to AND gate 2. If a pitch CAS fail signal does not exist at logic inverter 11, AND gate 7 is triggered and provides a high input to OR gates 4 and 5 for the length of the reset pulse. OR gate 4 provides a high input to AND gates 2, 4 and 8.

1-76. OR gate 2 and OR gate 4 trigger AND gate 2. The high AND gate 2 output is applied to OR gate 2 and to transistor switch 1. The high AND gate 2 output at OR gate 2 keeps the OR gate triggered and the reset pulse is no longer required to trigger OR gate 2. Transistor switch 1 applies 28VDC to the PRCA shutoff valves and excitation voltage to the CASI fail switches. If the CASI servo differential hydraulic pressure is normal, all fail switches close and trigger AND gates 9 and 10. The high AND gate outputs trigger AND gate 1. The high AND gate 1 output is applied to logic inverter 6. The low logic inverter 6 output inhibits AND gate 5. Refer to paragraphs 1-85 and 1-86 for an explanation of the AND gate 5 function.

1-77. To activate switch 2, AND gate 8 must be high. The requirements below must be satisfied to trigger AND gate 8.

a. The OR gate 4 output must be high.

b. The AND gate 3 output must be high.

c. The logic inverter 7 output must be high.

d. The logic inverter 8 output must be high.

1-78. The OR gate 4 output is high during the reset cycle. The AND gate 3 output is high as long as the differential stabilators have not failed and yaw rate is less than 41.5°/sec. The logic inverter 7 output is high as long as AND gate 6 is low. The logic inverter output 8 is high as long as AND gate 5 is low. When AND gate 8 is triggered, switch 2 closes. Switch 2 applies 28VDC to the stabilator shutoff valves and (through a shutdown time delay and 5VDC level changer) provides a second high input to OR gate 4. The one-shot multivibrator output is no longer required to trigger OR gate 4.

1-79. When the one-shot reset signal triggers OR gate 5, the high OR gate 5 output is applied to logic inverters 9 and 10 and to AND gates 4, 5 and 6. Logic inverter 9 (channel B) supplies pitch CAS on/off data to the signal data recording set. Channel A keeps the CAS PITCH light from coming on as long as OR gate 5 has a high output. Logic inverter 10 supplies a pitch CAS shutdown signal to the pitch CAS circuit and to the pilot relief logic circuit if the OR gate 5 output goes low.

1-80. The high OR gate 4 output is applied to one side of AND gate 4. The other side of AND gate 4 receives an input from OR gate 5 and locks OR gate 5 to high as long as OR gate 4 is high.

1-81. **Pitch CAS Shutdown.** AND gate 6 provides a method of disengaging pitch CAS on failure and the ability of operating roll CAS without engaging pitch. The AND gate 8 output is also required for roll CAS operation. If pitch CAS fails, OR gate 3 is triggered; if pitch CAS is engaged, there is a high output from OR gate 5 at AND gate 6. When AND gate 6 is triggered, logic inverter 7 goes low, AND gate 8 goes low, switch 2 opens and the stabilator shutoff valves close. The pitch CAS fails signal is also applied to logic inverter 11. When the logic inverter 11 output is low, AND gate 7 is inhibited and pitch CAS cannot be reset.

1-82. After a time delay, the high input to OR gate 4 from the 5VDC level changer goes low driving the OR gate 4 output to low with the results below:

a. AND gate 8 is inhibited by a second low input from OR gate 4.

b. AND gate 4 goes low, inhibits OR gate 5 and logic inverter 9 goes high (CAS PITCH light comes on).

c. AND gate 6 goes low and logic inverter 7 goes high (but AND gate 8 remains low because of a low from OR gate 4).

d. Logic inverter 10 goes high (pitch CAS shutdown).

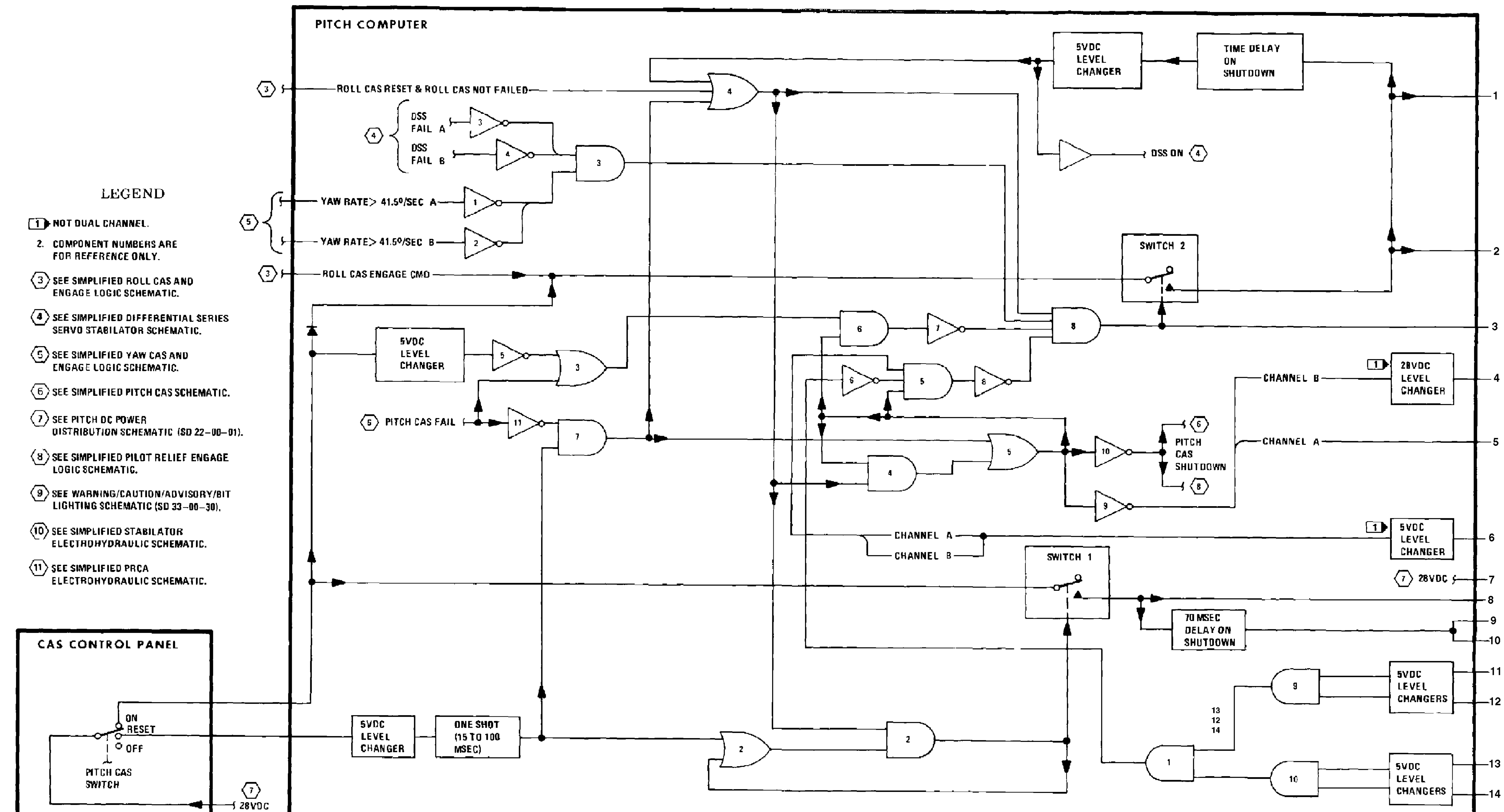

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Figure 1-9. Simplified Pitch CAS Engage Logic Schematic (Sheet 1 of 2)

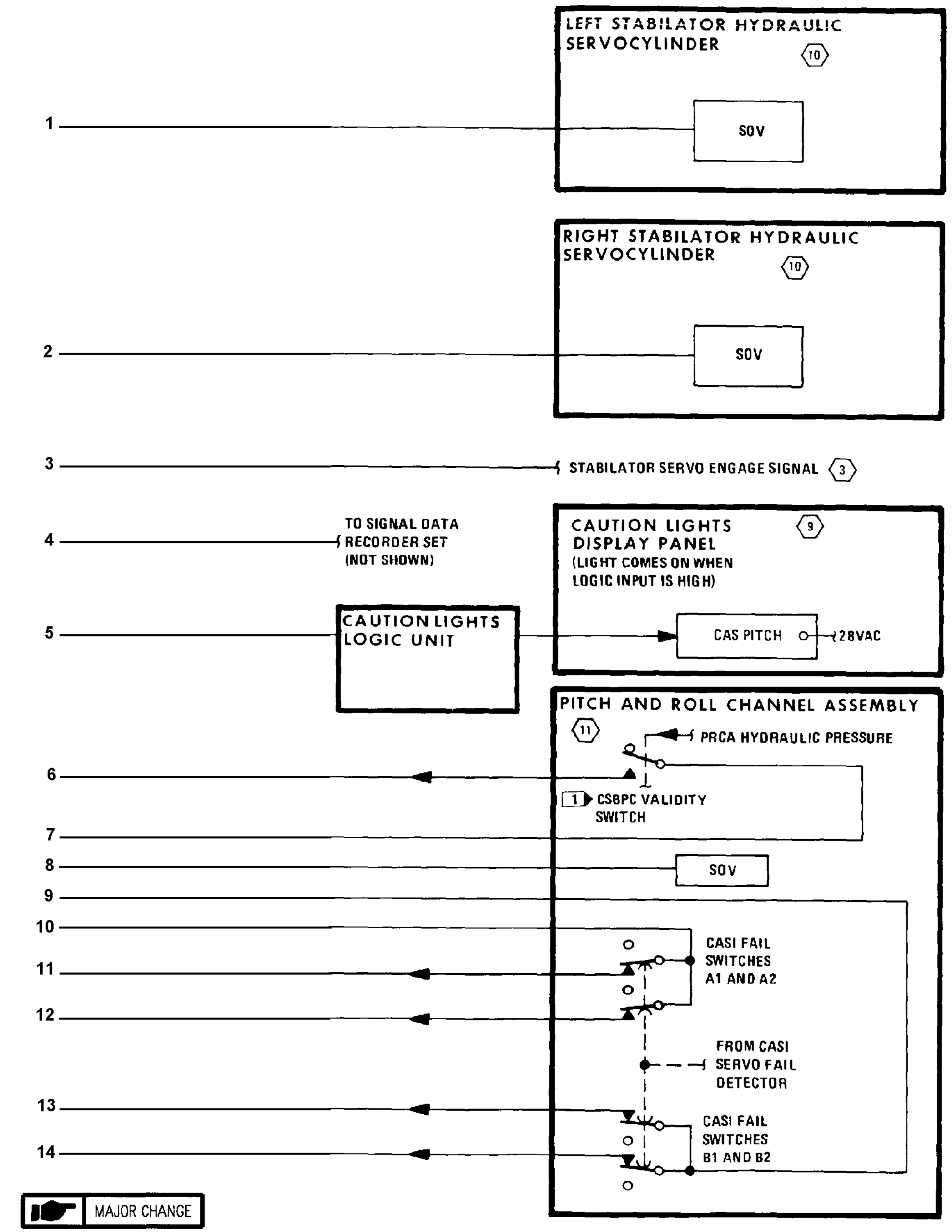


Figure 1-9. Simplified Pitch CAS Engage Logic Schematic (Sheet 2)

e. AND gate 2 goes low and inhibits OR gate 2.

f. Switch 1 opens and disables the PRCA CAS interconnect (CASP AND gate 1 goes low and inhibits AND gate 2 more).

1-83. If the pitch CAS fails momentarily (nuisance failure), pitch CAS can be reset and engaged again. If the failure is constant, AND gate 7 is inhibited and OR gate 4 is not enabled during the reset cycle, which inhibits AND gate 8.

1-84. **Roll CAS Logic After Pitch CAS Failure.** If roll CAS is engaged at the time pitch CAS fails or is disengaged, roll CAS also disengages when AND gate 8 goes low. If the pitch CAS failure was not because of a differential series stabilator (DSS) failure and other parameters are within established limits, roll CAS can be reset and reengaged. Engagement is arrived at by triggering OR gate 4 with a signal supplied when roll CAS is reset (as long as roll CAS has not failed). OR gate 4 triggers AND gate 8, switch 2 closes, and locks in OR gate 4. Since the PRCA 's not required for roll CAS operation, switch remains open.

1-85. **Pitch CAS Logic After CASI Failure.** AND gate 5 allows pitch CAS operation if a CASI failure occurs. During normal operation, two of the three inputs to AND gate 5 are high and one is low so that AND gate 5 has low output to logic inverter 8. The two high inputs to AND gate 5 are from OR gate 5 and from the CSBPC validity switch. If one of the CASI fail switches opens, the inverted AND gate 1 output triggers AND gate 5 and the low logic inverter 8 output inhibits AND gate 8.

1-86. If PITCH RATIO is set to EMERG, the CSBPC validity switch opens, AND gate 5 is inhibited and pitch CAS can be reset and engaged again without the CASI function.

1-87. **PITCH CAS CIRCUIT.** See figure 1-10. The pitch CAS circuit shows how control stick pitch force sensor signals are used to drive stabilator control surfaces and the PRCA electrohydraulic valves.

1-88. Demodulated pitch force signals are applied to switch 1 and to the 10.64 lb deadband breakout. Channel A signals are also applied to a 1.0 lb breakout and a 3.5 lb breakout for use in developing pitch control stick steering (CSS) logic. The channel A and B force signals are applied through switch 1 to the 1.0 lb breakout (if ALT HOLD is not engaged) or to the 3.5 lb breakout (if ALT HOLD is engaged) and combined with the 10.64 lb deadband output. The composite pitch force signals (which duplicate electronically the mechanical action of the dual gradient feel trim actuator on the force applied to the control stick) are applied to summing amplifier 7. The amplifier 7 output is filtered by a structural filter to reduce response at the aircraft longitudinal bending frequencies. Computed pitch is then amplified and summed with integrated pitch at the pitch CAS limiter. Inverted limited pitch drives the CASI electrohydraulic valve (EHV). Limited pitch and inverted limited pitch are applied to the differential series stabilator servoamplifier which, in turn, drive the stabilators. When the stabilators move, the aircraft pitches up or down. Pitch rate gyros and normal accelerometers sense aircraft movement.

1-89. **Acceleration and Pitch Rate.** If the landing gear handle is up, 7.5VDC positive limiter bias is applied to the acceleration fader, cancelled pitch rate fader, and inverting amplifiers 4 and 5. The inverting amplifiers supply negative fader bias. Buffered normal acceleration signals are applied to the fader. Channel A of the buffered normal acceleration signal is also applied to the pilot relief circuits for use in the outer loop disengage logic. Acceleration is a part of the feedback into the pitch CAS loop.

1-90. During non-terminal flight phases, the required feedback parameters are normal acceleration and cancelled pitch rate. During terminal flight phases (landing), the required feedback parameters is uncancelled pitch rate. Normal acceleration feedback is removed during landing to prevent the large normal accelerometer outputs (which would go with any hard landing) from entering the pitch CAS loop. Pitch rate feedback cancellation is removed simultaneously for stable control. Modular relay panel relays deenergize when the cockpit landing gear handle is moved down and start the terminal flight phase. When the landing gear handle is moved down, 28VDC from the modular relay panel is removed from a 5VDC level changer and logic inverter 3 provides a high output to switch 2 and

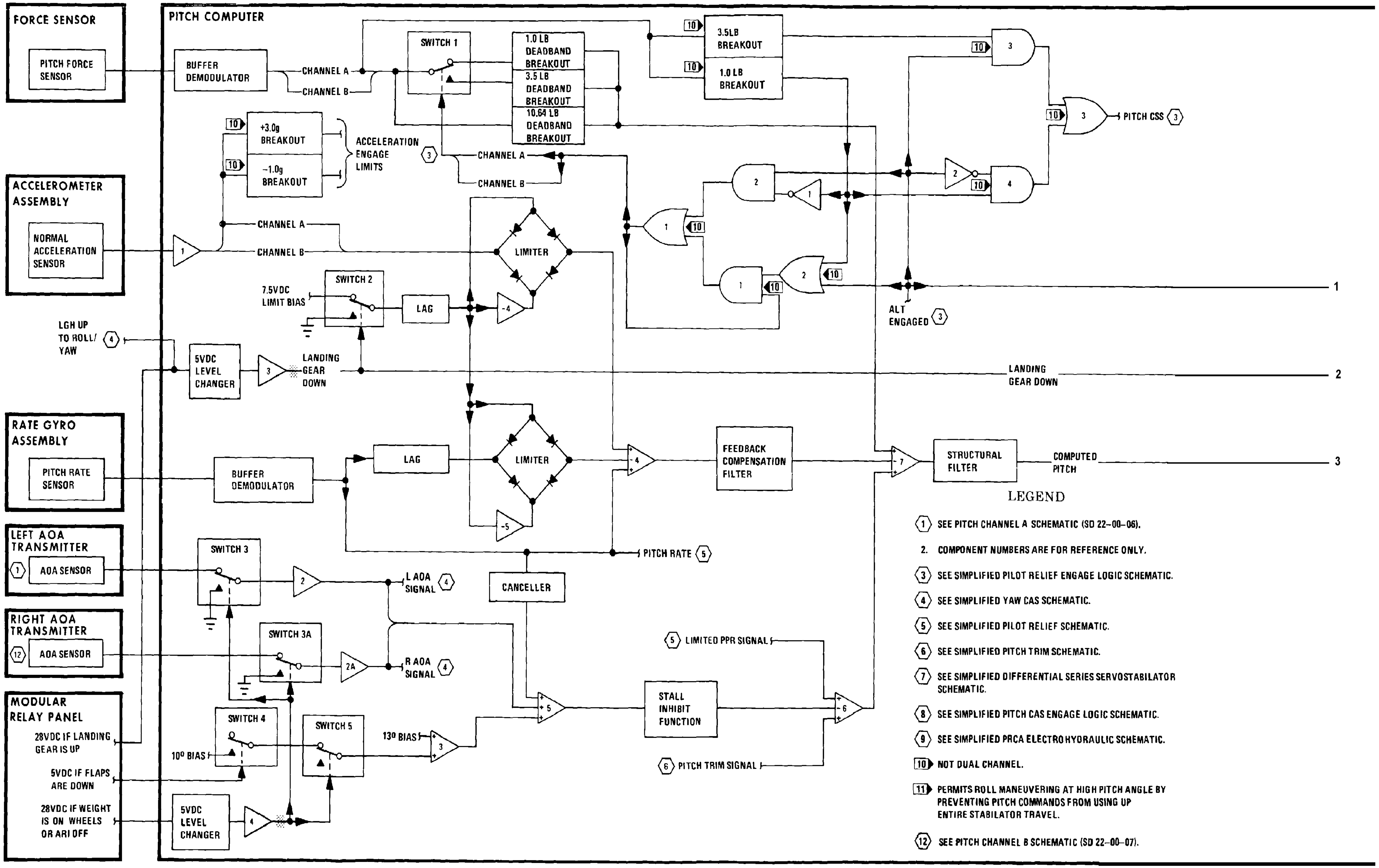
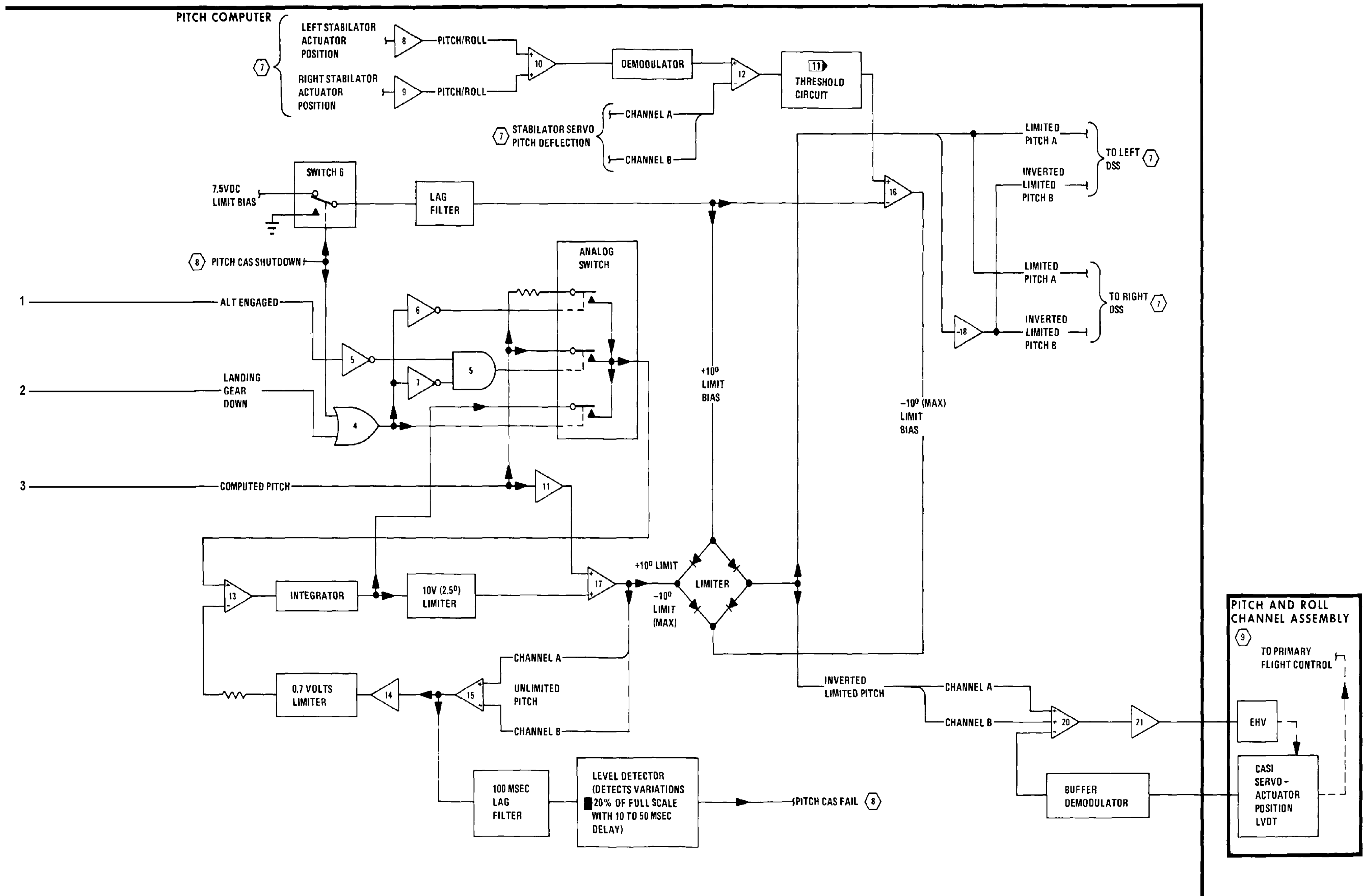


Figure 1-10. Simplified Pitch CAS Schematic (Sheet 1 of 2)



1-10. Simplified Pitch CAS Schematic (Sheet 2)

OR gate 4. When switch 2 is activated, limiter bias is removed slowly.

1-91. To prevent switching transients, a fader circuit slowly removes and applies acceleration signals. The fader circuit is made up of a variable limiter controlled by a lag network through which limiter bias is applied or removed. Limiter authority is exponentially controlled with a 1 second time constant so as to smoothly remove (or apply) acceleration signals to the pitch CAS loop.

1-92. Pitch rate cancelling is arrived at by subtracting lagged pitch rate from unfiltered pitch rate. The lagged pitch rate component is controlled by a fader circuit in the same way as normal acceleration signals. The fader circuit provides a smooth transition from cancelled to uncancelled pitch rate when the landing gear handle is operated. Pitch rate is also applied to the pilot relief circuit and to the stall warning circuit. Feedback parameters are filtered by a compensation network which provides specified augmented short period damping and natural frequency requirements through the flight envelope.

1-93 **Stall Inhibit Function**. The angle-of-attack probes supply AOA signals to buffer amplifier 2 through normally closed (NC) switch 3. The AOA signal is then transmitted to the roll/yaw computer for ARI scheduling and CAS authority limiting. The AOA signal is added to cancelled pitch rate and 13° AOA bias at summing amplifier 5. The summing amplifier 5 output is applied to a stall inhibit (stall warning) circuit. As the aircraft nears a stall attitude, a threshold circuit feeds a negative pitch command into the CAS so that a larger force is required by the pilot to command an increase in nose up attitude. The cancelled pitch rate input allows the circuit to prepare for a stall condition.

1-94. If flaps are down, switch 4 and N.C. switch 5 apply another 10° bias to amplifier 3. The increased bias allows an increased AOA signal to be applied to the stall inhibit circuitry before the stall warning takes effect. When weight is on wheels, the AOA signal and the 10° bias are removed and the stall inhibit function is not in operation. Summing amplifier 6 combines stall inhibit signals with pitch pilot relief and pitch trim signals and supplies the third input to summing amplifier 7. Left and right AOA signals are monitored by the pitch computer and if the difference between roll/yaw AOA A and B exceeds 1.20 volts ASP 15 will set on the avionics status panel.

1-95. **Pitch Control Stick Steering.** Before ALT is engaged, a single pound of longitudinal force can enter the pitch CAS loop. If ALT engaged, a 3.5 lb force is required to override pilot relief inputs. When ALT is first engaged, switch 1 applies force to the 1.0 lb level detector. If ALT is engaged and pitch force is less than 1.0 lb, AND gate 2 and OR gate 1 are triggered. The OR gate 1 output controls switch 1, removes the pitch force input from the 1.0 lb deadband breakout, and applies it to the 3.5 breakout. Only pitch force signals in excess of 3.5 lb can now be applied to summing amplifier 7. The OR gate 1 output is also applied to AND gate 1. Since the other input to AND gate 1 exists if ALT is engaged or pitch force is greater than 1.0 1b, both conditions must be removed before switch 1 reverts to the 1.0 lb deadband breakout.

1-96. Even though stick force may exceed 1.0 lb before ALT hold switch is engaged when OR gate 3 is triggered by AND gate 4, MT and ALT hold switch can be engaged. OR gate 3 is triggered by AND gate 3 after ALT hold switch is engaged when stick force exceeds 3.5 lb. When ATT and ALT hold switches are engaged, only the hold circuits are interrupted when OR gate 3 is triggered, but ATT and ALT hold switches remain engaged.

1-97. **Pitch Limiting.** Pitch limiter bias is provided through switch 6 unless a pitch CAS shutdown signal exists. Positive bias is applied directly, but negative bias is varied as a function of the stabilator pitch position mechanical component.

1-98. Each stabilator main ram provides position signals to the pitch computer. The left and right main ram positions from buffer amplifiers 8 and 9 are added in summing amplifier 10 to remove the roll component from the main ram position. The main ram pitch position signals are combined with the CAS ram pitch position signals at summing amplifier 12 to produce a signal proportional to mechanical pitch position. The amplifier 12 output is applied to the threshold circuit. The threshold circuit provides no output if the mechanical pitch position is less than 16° trailing edge up. From 16° to 25°, the threshold

All data on pages 1-30A and 1-30B deleted.

circuit output increases so that, at 25% negative bias at summing amplifier 16 is reduced to the equivalent of 1°.

1-99. If a pitch CAS shutdown signal exists, switch 6 removes the lag filter bias and applies a ground. The lag filter reduces the pitch limiter to zero smoothly with a time constant of approximately 1 second.

1-100. **Integrator Operation.** Computed pitch from the structural filter is divided into two paths; a proportional path and an integral path. The proportional path is through amplifier 11 to summing amplifier 17. The integral path depends on the analog switch condition.

1-101. If pitch CAS is not shut down and the landing gear handle is up, OR gate 4 is inhibited and the logic inverter 6 output is high. A high output from logic inverter 6 closes the analog switch upper contacts to provide reduction of normal gain. Computed pitch is applied to summing amplifier 13 and combined with the difference between unlimited pitch channels A and B.

1-102. If channels A and B are not the same, the limited amplifier 14 outputs add to or subtracts from computed pitch at. amplifier 13 and tends to equalize the two channels. Integrated pitch is added to computed pitch as summing amplifier 17. If the difference between the two channels can be reduced enough, the pitch CAS fail level detector output remains low. If the difference is very large (or even a smaller difference sustained over a longer period), a pitch CAS fail signal is produced and pitch CAS shut down. If pitch CAS is shut down, OR gate 4 is triggered, the analog switch center and upper contacts open and the bottom contacts close. The integrator circuit then becomes a closed loop with no output to summing amplifier 17.

1-103. If ALT is not engaged, logic inverter 5 provides a high input to AND gate 5. If OR gate 4 is inhibited, logic inverter 7 provides a second high input to AND gate 5. AND gate 5 is triggered, the analog switch center contacts close and provide integrated signal strength of normal value. Limited pitch is applied to summing amplifier 20 and combined with the feedback signal from the CASI servoactuator position LVDT. The summed output is amplified by current amplifier 21 and drives the PRCA EHV. Limited pitch (channel A) and inverted limited pitch (channel B) from inverting amplifier 18 drive the stabilizer hydraulic servocylinder EHV. The stabilizers are mechanized to command a nose down (trailing edges down) if limited pitch commands are positive.

1-104. **Simplified PRCA Electrohydraulic Operation.** See figure 1-11. When hydraulic pressure is applied to aircraft, the pressure switch pressurizes and drives the CSBPC validity switch to the closed position. The 28VDC CSBPC validity signal is applied to the pitch computer to satisfy pitch CAS engage logic circuitry. Pressure is also available at shutoffvalves A and B. When pitch CAS is engaged, the shutoff valves open and apply pressure to electrohydraulic valves A and B, to the CASI fail detector, and to the CASI servolock. Pressure applied to the CASI servolock frees the servovalve to receive commands from electrohydraulic valves A and B. CASI servo commands reposition the EHV to apply pressure to the servovalve and to the CASI servo fail detector. If the sum of EHV A and EHV B pressures equals applied pressure, the CASI fail switches close and 28VDC is applied to the pitch computer to indicate correct CASI servo operation. The CASI servovalve movement is transmitted mechanically to the primary flight controls. CASI servo position LVDT signals supply feedback to the CASI servoamplifier to null the servo commands.

1-105. The sum of control pressures A and B at the CASI servo fail detector could be more than applied pressure (if one EHV allows total hydraulic pressure to pass) or less than total pressure (if one EHV shuts off all pressure). If pressures are out of balance, the CASI fail switches open and command a pitch CAS shutdown. When pitch CAS shuts down, SOV A and SOV B close and the CAS lock extends to center the CASI servovalve. If hydraulic pressure to the PRCA fails, the CSBPC validity switch opens when the pressure switch depressurizes. When the CSBPC validity signal is removed, pitch CAS shuts down.

1-106. DIFFERENTIAL SERIES STABILATOR SERVOCYLINDER.

1-107. **Simplified Stabilator Electrohydraulic Operation**. See figure 1-12. When hydraulic pressure is applied to aircraft, the pressure is immediately available to the servovalve and to the differential pressure sensor (DPS). The DPS valve shuts off control pressure to the servovalve CAS ram and (if no pressure is available at the CAS lock) the CAS ram is in the locked position. If the control stick is at neutral in

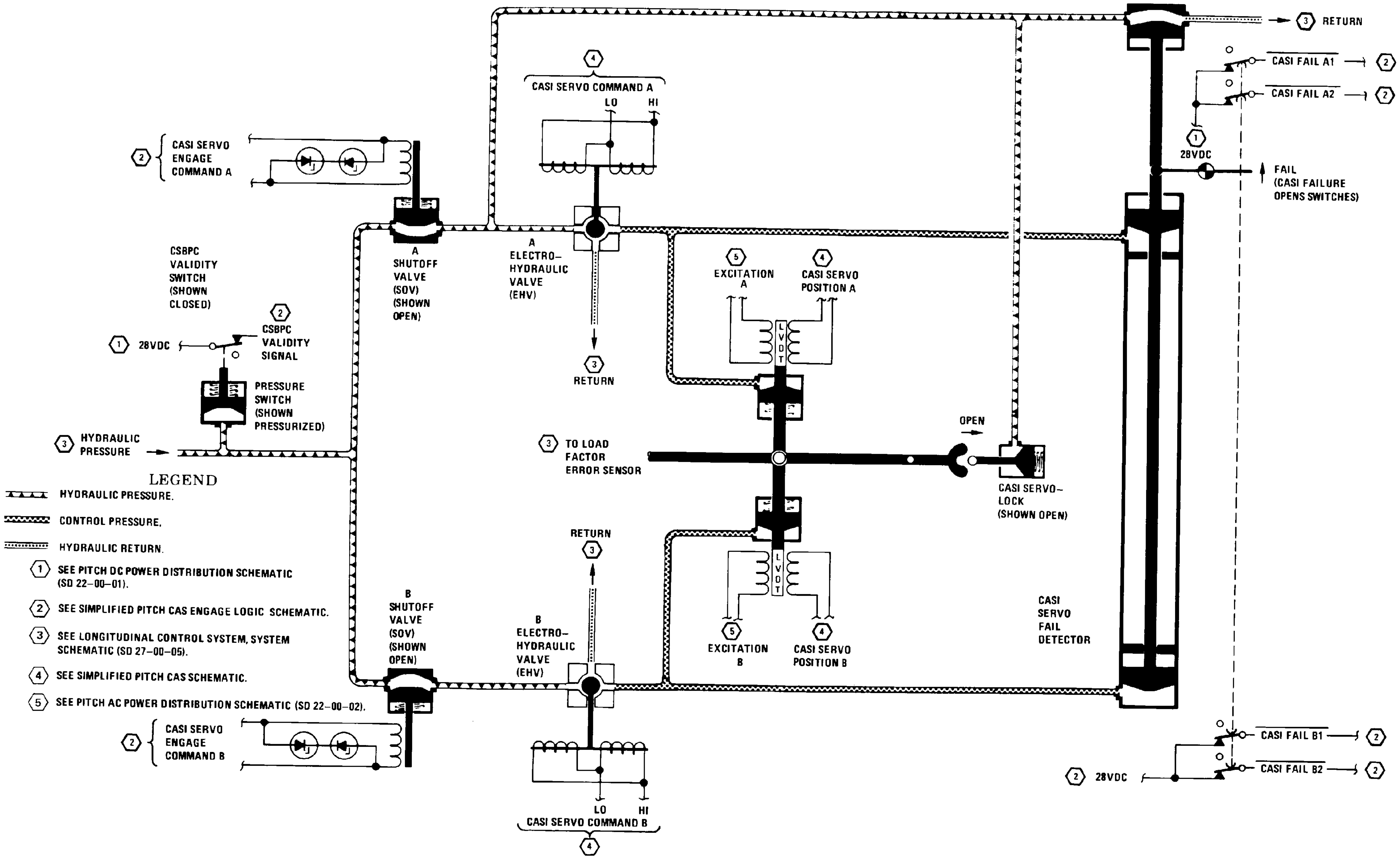


Figure 1-11. Simplified PRCA Electrohydraulic Schematic

pitch, the servovalve remains at neutral, hydraulic pressure is returned through the servovalve return ports, and the main ram also remains at neutral. If the control stick is moved aft, the servovalve plunger moves (up) and ports control pressure C2 and C4 to the main ram. The main ram retracts and drives the stabilator trailing edges up to command and aircraft nose up attitude. If the stick is moved forward, the servovalve plunger moves (down) and ports control pressure Cl and C3 to the main ram. The main ram extends and moves the stabilator trailing edges down to command a nose down attitude.

1-108. When pitch/or roll CAS is engaged, stabilator servo engage commands A and B open shutoff valve (SOV) A and SOV B. When both valves open, hydraulic pressure is available to EHV A and EHV B and to the CAS lock. Assuming that stabilator servo commands A and B are at neutral, equal control pressures from EHV A and EHV B are applied to the DPS. If the sum of pressure A and B equals total hydraulic pressure, the DPS applies the EHV control pressure to the CAS ram. If the sum of control pressures A and B is not equal to total hydraulic pressure, a DPS fail signal is produced and control pressures are blocked from the CAS ram. Hydraulic pressure applied to the CAS lock unlocks the CAS ram, but since the servo command is neutral, the CAS ram does not move.

1-109. Since CAS can supply control surface commands even if the mechanical control system malfunctions, assume that the manual controls are disconnected. If the pilot applies a forward force to the control stick, Nose down commands exist at EHV A and EHV B. Valves A and B provide more pressure to the trailing edge down ports and reduced pressure to the trailing edge up ports. Since the EHV B trailing edge up port is capped, increase EHV A trailing edge down and decrease EHV B trailing edge up pressure is applied to the CAS ram through the DPS. If total pressure at the DPS remains equal to total hydraulic pressure, a DPS fail signal is not produced. The increased EHV A trailing edge down pressure drives the CAS ram and the servovalve dynamic sleeve down and causes the main ram to extend as if a mechanical forward stick input had moved the plunger, but the CAS command is limited to a maximum of 10°. If an aft force is applied to the control stick, EHV B supplies the trailing edge up command and the main ram retracts.

1-110. The CAS ram LVDT and the main ram LVDT provide feedback information to the CAS circuit to cancel servo commands when the stabilator is in the required position. CAS during normal operation can add to or subtract from mechanical inputs by moving the servovalve dynamic sleeve together with or in opposition to mechanical inputs. Left and right stabilators are identical and interchangeable, but the components of the servo command are 180° out of phase.

1-111. **Left Differential Series Stabilator Servocylinder (DSS).** See figure 1-13. Limited pitch and limited roll (channel A) is applied to buffer amplifiers 14 and 13 respectively. Buffer amplifier 13 has a gain of 0.5 and buffer amplifier 14 a gain of 1.0. While a maximum pitch input can command ±l0° of stabilator movement, a maximum roll input can command ±5°. Amplifier 13 amplifier 14 outputs are added at summing amplifier 17, combined with the integrated DPS signal at summing amplifier 24 and applied to summing amplifier 27. The amplifier 27 output drives stabilator EHV A. Inverted limited pitch and roll signals are applied to buffer amplifiers 6 and 5, summed at amplifier 11, combined with integrated DPS signals at amplifier 21 and applied to summing amplifier 25. The amplifier 25 output drives stabilator EHV B. Normally, the A and B signals are near each other (if they were not, pitch CAS would shut down) but they are independently derived and may not be exactly alike. The channel A and channel B servo commnands are inverted, but the channel A and channel B servo loops are reversed so that resultant motion is in the same direction. The channel A and channel B series servos are actually force summed to drive a common ram.

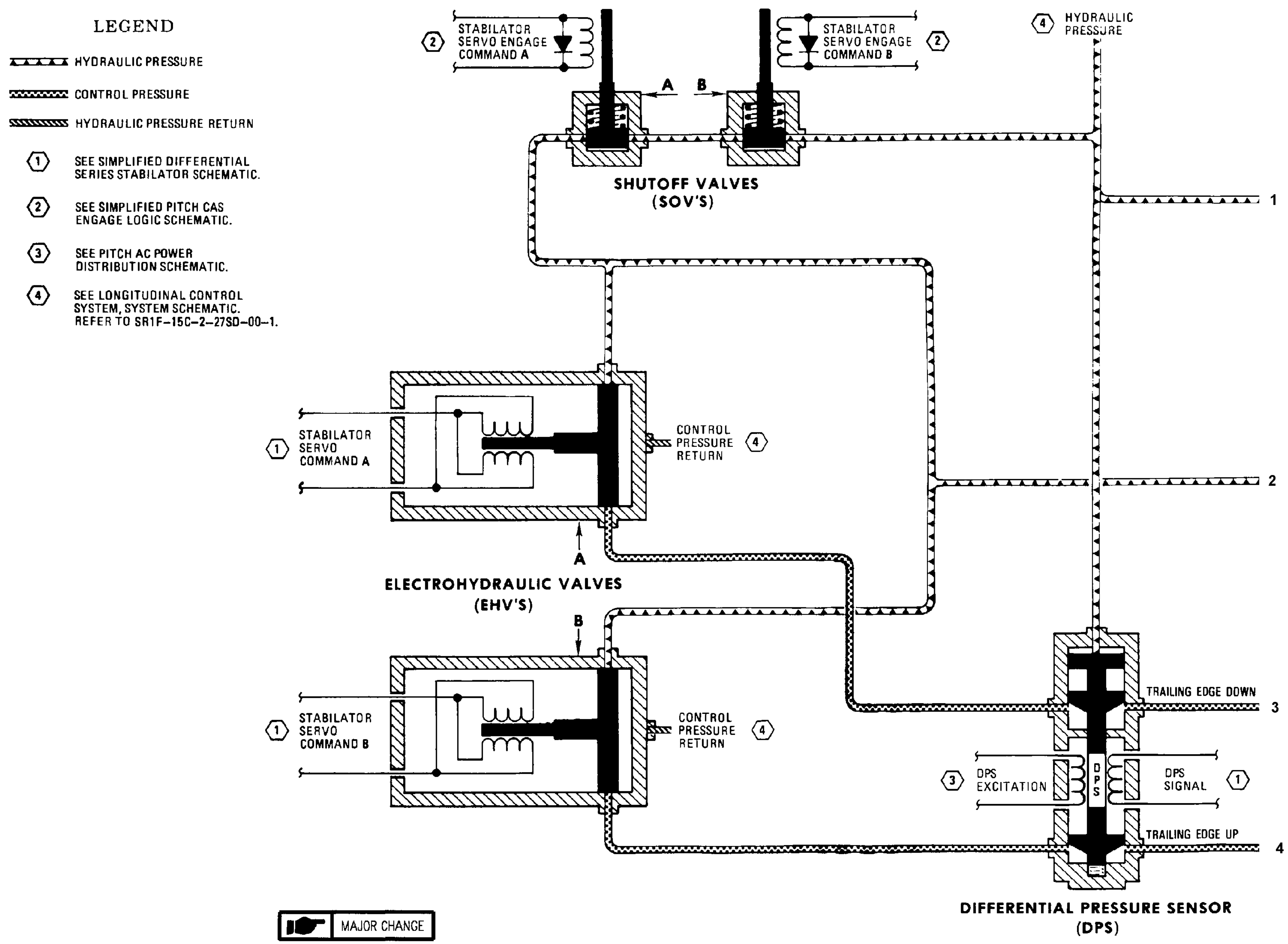


Figure 1-12. Simplified Stabilator Electrohydraulic Schematic (Sheet 1 of 2)

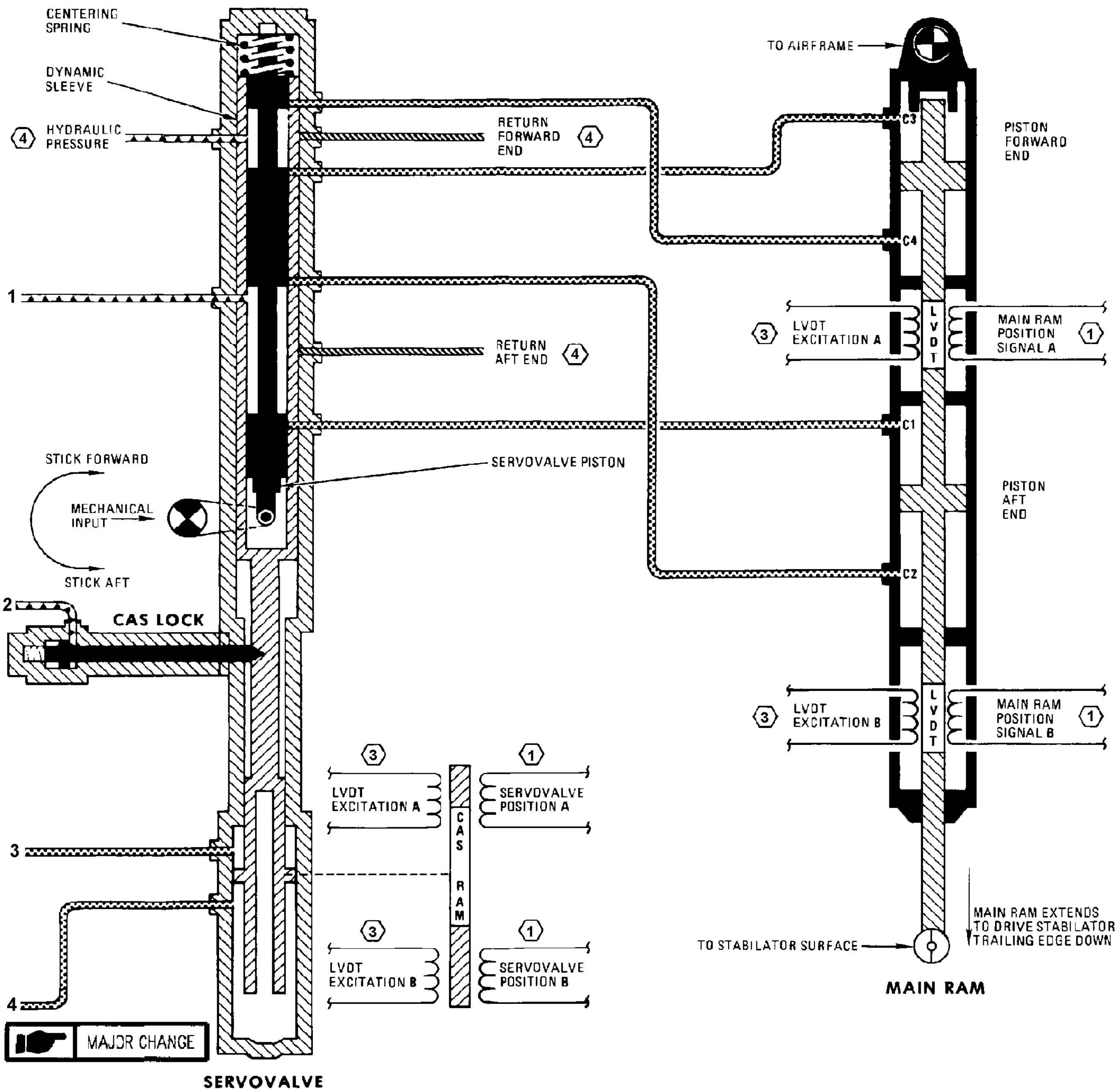


Figure 1-12. Simplified Stabilator Electrohydraulic Schematic (Sheet 2)

Excessive difference between servo commands causes the ram to stall and provides failure transient control. A positive pitch and/or roll signal from channel A combines with the inverted channel B signals to drive the stabilator traling edge down. Conversely, a negative channel A input drives the trailing edge up. When the servovalve drives the main ram, the main ram position LVDT and the CAS ram position LVDT change position. The main ram position LVDT outputs are applied to the signal data recorder set and to the pitch CAS circuit where, combined with CAS ram pitch position signals, they control pitch CAS limiter operation. Refer to paragraph 1-96. CAS ram position LVDT outputs are also used to null pitch and/or roll commands at summing amplifiers 25 and 27.

1-112. Differential Pressure Sensor. Since EHV A and EHV B are driven by similar but separate signals, there may be some difference between commands. The DPS monitors total hydraulic pressure and compares the total to the sum of pressures from EHV A and EHV B. It is preferred that the sum of hydraulic pressures from EHV A and B equal total hydraulic pressure. If no difference exists between total pressure and the sum of A and B, the DPS output is small. The small output prevents a total DPS failure from appearing as a normal condition. The DPS output is demodulated and applied to summing amplifiers 1 and 3. The 1.46VDC bias cancels the small DPS signal. The center inputs at amplifiers 1 and 3 are different if DPS pressures are out of balance. Before pitch and/or roll CAS engagement, switches 1 and 3 are closed and the amplifier outputs are cancelled out. With pitch and/or roll CAS engaged, the DSS-ON signal grounds the bottom inputs into amplifiers 1 and 3 and activates the DPS integrator and fail detection network. Demodulated DPS summing amplifier 3 outputs are applied to the DPS integrator. The integrator output is added to the pitch/roll signal at summing amplifier 24 and to the inverted pitch/roll signal at summing amplifier 21. The DPS integrator tries to equalize the pressures from EHV A and B. If the integrator equalizes pressure before the integrator output arrives at 7.2 volts, pitch and/or roll CAS operate normally. If the integrator output goes above 7.2 volts, the 7.2VDC level detector output goes high, the equalizer fail signal triggers OR gate 1 and the DSS fail B signal is applied to the pitch CAS engage logic circuit.

1-113. Demodulated DPS summing amplifier 1 outputs can also produce a DSS fail signal. The amplifier output is applied at a 3.6 sec lag network which is applied to a 6.0VDC level detector. If amplifier output exceeds 6.0VDC by a very large amount (or a small amount for an extended period of time), the 6.0VDC level detector output goes high. The DPS fail signal triggers OR gate 2 and produces the DSS fail A signal which is applied to the CAS engage logic circuit.

1-114. **Right Differential Series Stabilator Servocylinder.** See figure 1-13. The right differential stabilator operates in the same way as the left, but the roll signal inputs are reversed to achieve differential stabilator movement. Pitch A is combined with inverted roll A to drive EHV A. A positive roll signal (inverted) drives the stabilator trailing edge down. Roll B and inverted pitch B drive EHV B so that positive roll signals result in a right roll command (left stabilator trailing edge down and right stabilator trailing edge up).

1-115. Differential Pressure Sensor. The right stabilator DPS operates in the same way as the left DPS but the integrator 7.2VDC level detector output triggers OR gate 2 if the equalizer fails and produces the DSS fail A signal. The 6.0 VDC level detector output produces the DSS fail B signal. The cross coupling between A and B channels improves the dual channel aspect in this critical area.

1- 116. **ROLL CAS OPERATION.** When a left or right force is applied to the control stick, the primary flight control system deflects ailerons and stabilators differentially. Simultaneously, the roll force sensors produce roll force signals proportional to applied force. The roll rate gyros measure aircraft response to the command roll input. The measured response is compared to the commanded electrical force input from the roll force sensors.

1-117. If no difference exists between commanded roll and aircraft response, the roll force signal is nulled. If aircraft response is more or less than commanded, the difference is applied

LEGEND

1. SEE SIMPLIFIED ROLL CAS AND ENGAGE LOGIC SCHEMATIC.
2. SEE SIMPLIFIED PITCH CAS SCHEMATIC.
3. SEE SIMPLIFIED PITCH CAS ENGAGE LOGIC SCHEMATIC.
4. SEE SIMPLIFIED STABILATOR ELECTRO-HYDRAULIC SCHEMATIC.
5. DUAL CHANNEL COMPONENTS SHOWN INDIVIDUALLY.
6. COMPONENT NUMBERS ARE FOR REFERENCE ONLY.

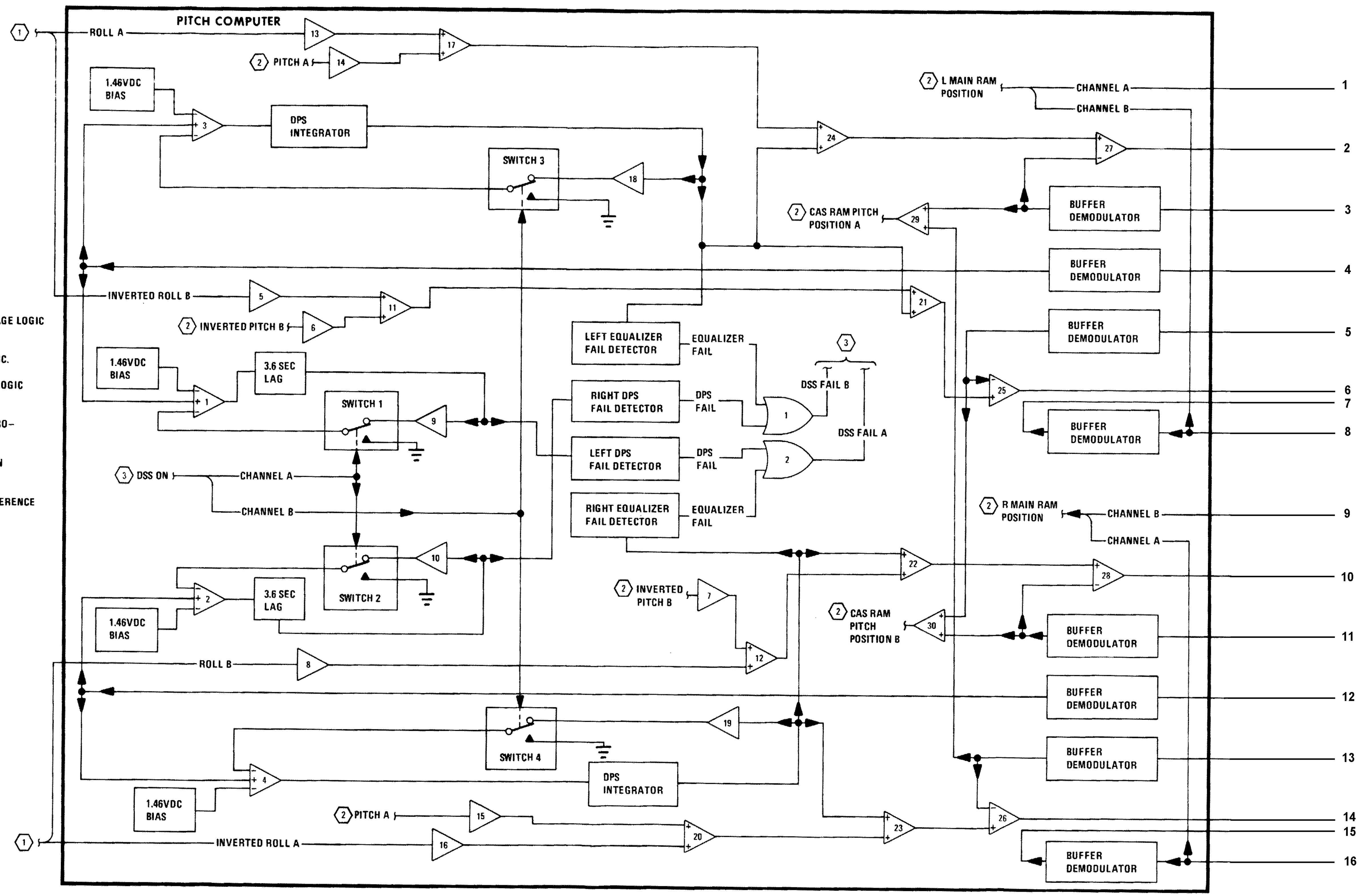


Figure 1-13. Simplified Differential Series Stabilator Servocylinder Schematic (Sheet 1 of 2)

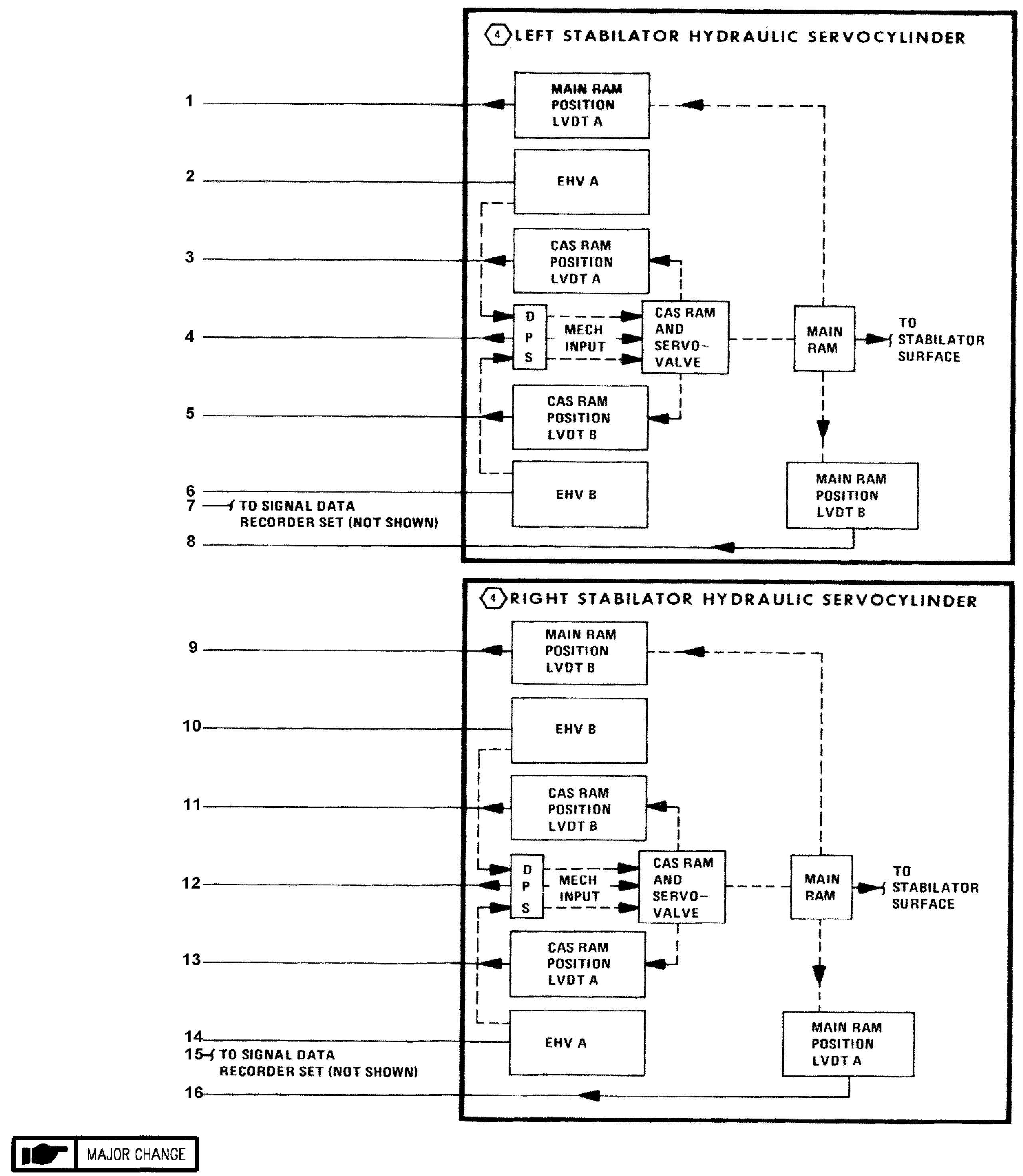


Figure 1-13. Simplified Differential Series Stabilator Servocylinder Schematic (Sheet 2)

(through a variable limiter) to the pitch computer DSS Roll CAS can add or subtract as much as 5° of differential stabilator displacement to the position established by the primary flight control system. Roll CAS commands do not affect the ailerons.

1-118. In the primary control system, roll ratio is varied as a function of airspeed. In the AFCS, airspeed (in theform of dynamic pressure) and/or AOA signals effects the limits of roll CAS authority by varying the roll CAS limiter bias. If airspeed is less than 544 knots and AOA is between +7° and -1° the roll CAS authority is ±5°. the roll CAS authority limits are obtained by subtracting the AOA schedule from the airspeed schedule. A airspeed increases above 544 knots, the airspeed schedule bias on the roll CAS limiter is decreased so that, at 800 knots the roll CAS authority is limited to 1.1°. As AOA increases between 7° and 20.3°the AOA schedule bias on the roll CAS limiter is decreased so that at 20.3° the roll CAS authority is limited to 0°. If AOA is less than -1° the roll CAS authority is also limited to 0°.

1-119**. ROLL CAS ENGAGE LOGIC.** See figure 1-14. Roll CAS maybe engaged independently of pitch CAS. Roll CAS is engaged when the AND gate 2 output goes high. The five inputs required to trigger AND gate 2 are as listed below:

a. If a roll CAS fail signal does not exist, logic inverters 1 and 2 supply high inputs to AND gates 1and 2.

b. If yaw CAS is engaged, a second high is supplied to AND gate 2.

c. When the roll CAS switch is set from OFF to ON, the reset contacts close momentarily and trigger the one-shot multivibrator with the results below:

1. The multivibrator triggers AND gate 1 and OR gate 2.
2. AND gate 1 supplies the reset signal to the pitch CAS engage logic circuit.
3. OR gate 2 supplies the third input to AND gate 2.
4. The conditions remain in effect for 15 to100 msec.

d. When the roll CAS switch is moved to ON:

1. The fourth input to AND gate 2 is supplied through a 5VDC level changer.
2. A roll CAS engage command is supplied to the pitch CAS engage logic circuits and combined with the roll reset signal to engage the stabilators.
3. When the stabilators are engaged, the pitch computer supplies the fifth input to AND gate 2.
4. If the reset signal is still present, AND gate 2 is triggered and the AND gate 2 output locks OR gate 2 to high.

e. When OR gate 2 is high, logic inverter 3 goes low and the caution panel CAS ROLL light goes off.

1-120**. Roll CAS Disengagements.** Stabilators control the aircraft in pitch and roll. If pitch CAS malfunctions, the stabilator shutoff valves close when pitch CAS shuts down. When the valves close, roll CAS commands cannot drive the stabilators. If pitch CAS malfunctions, roll CAS also shuts down (the stabilator servo engaged signal is removed from AND gate2). If the pitch CAS failure does not affect roll CAS operation, roll CAS may be reset and reengaged. Roll CAS also disengages if yaw CAS is disengaged or fails. If the difference between the dual channel roll signals is large enough to produce a roll CAS fail

1-121. **ROLL CAS CIRCUIT.** See figure 1-14. Stick force sensor roll force signals are applied to a 3.0 lb, deadband breakout amplifier. Channel A signals are also supplied to a 1.0 lb level detector. If roll force exceeds 1.0 lb, the level detector output interrupts roll attitudes circuits. The roll force breakout output is applied to a transistor switch. If a Mach < 1.5 signal exists, the switch is closed and roll force signals are applied to summing amplifier 4 with virtually no attenuation. If Mach is greater than 1.5, the switch opens and roll force signals applied to amplifier 4 are attenuated by approximately 50 percent. Roll force signals are combined with roll trim and roll attitude signals at amplifier 4. The amplifier 4 output is applied to summing amplifier signal, roll CAS disengages (paragraph 1-11).

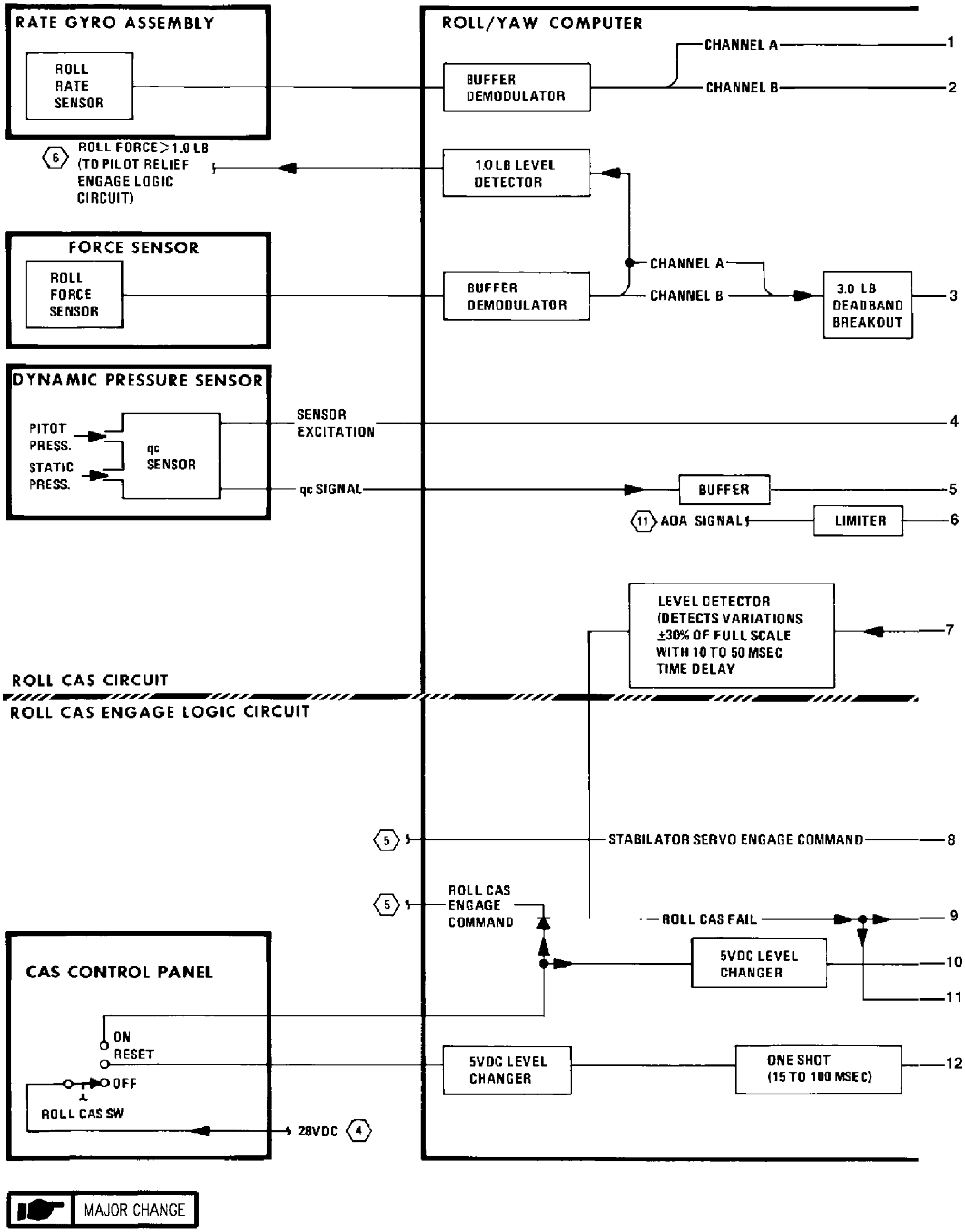
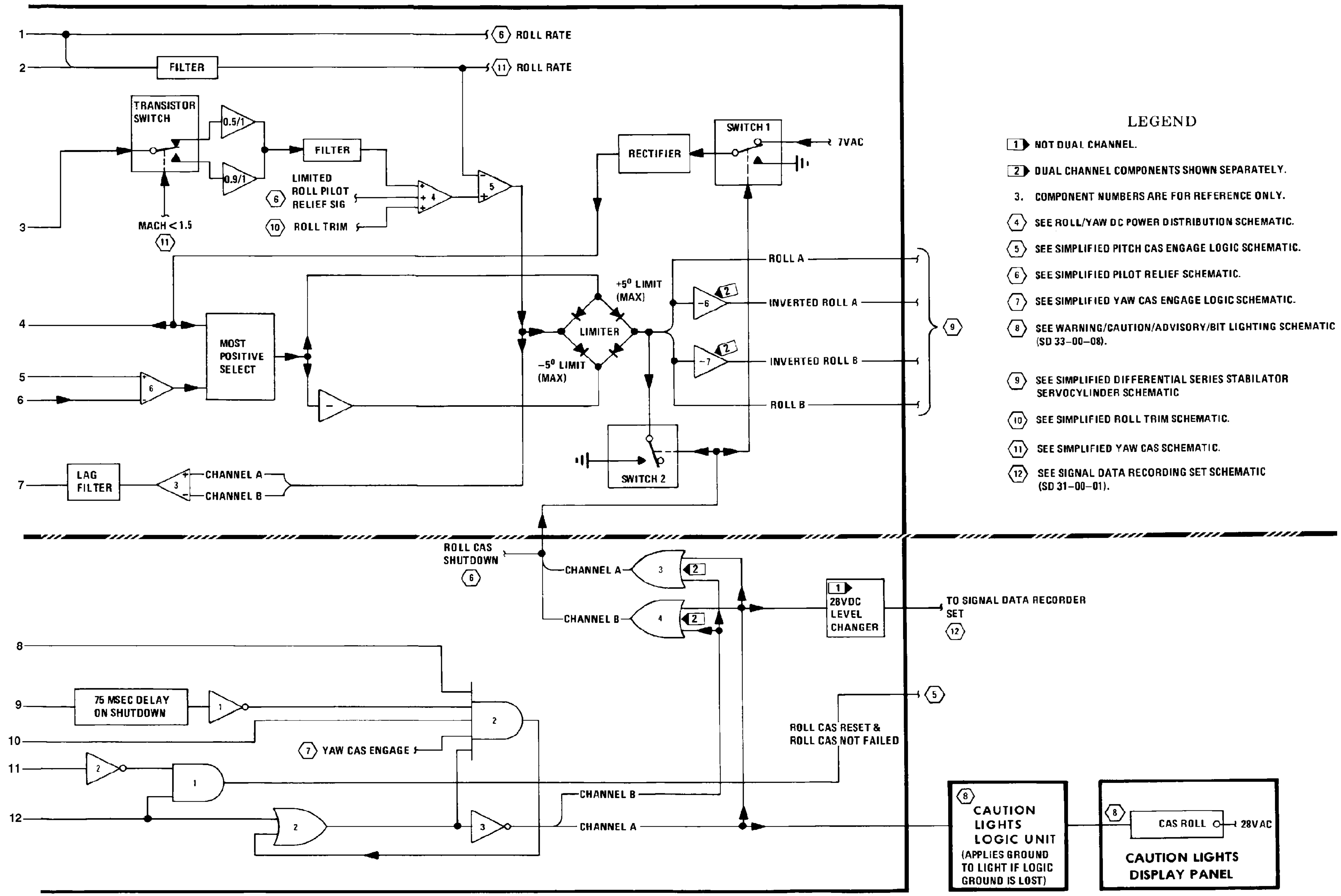


Figure 1-14. Simplified Roll CAS Engage Logic Schematic (Sheet 1 of 2)



MAJOR CHANGE

Figure 1-14. Simplified Roll CAS Engage Logic Schematic (Sheet 2)

5. The amplifier 5 output is applied to the roll CAS limiter. Limiter roll and inverted limited roll drive the stabilator surfaces differentially adding to or subtracting from the roll attitude commanded by the primary flight control system. As the aircraft changes attitude, the roll rate sensor supplies the feedback signal. Demodulated roll rate is applied to summing amplifier 5. When the roll rate signal equals roll force, the roll force signal is nulled. Roll rate is also applied to the yaw CAS ARI circuits for performing coordinated turns.

1-122. Variable Limiting. Stabilators are more effective at higher speeds. At an airspeed of 800 knots, the roll CAS limits are narrowed to 1.1°. When AOA is +23° or -1.0° roll CAS limits are narrowed to 0°. If roll CAS is operating correctly, rectified 7VAC (through switch 1) is applied to the dynamic pressure sensor for excitation voltage. Qc and AOA signals are conditioned and applied to summing amplifier 6. Rectified 7VAC and summing amplifier 6 output is processed through the most positive select circuitry and applied to the limiter, establishing the positive and negative limits

1-123. **Roll CAS Failure.** Summing amplifier 5 channel A and channel B outputs are compared at summing amplifier 3. The difference between channels is applied through a time delay (lag network) to a level detector. If the difference is greater than +30 percent of full scale, the level detector output is applied to a second time delay circuit. To prevent nuisance disengagements while providing positive failure detection, the monitor characteristics are such that large errors are detected with a minimum of delay while small errors must be maintained over a longer period of time to produce a failure signal. When a roll CAS fail signal is produced, logic inverter 1 goes low and inhibits AND gate 2.

1-124. **Roll CAS Shutdown**. When AND gate 2 goes low, OR gate 2 goes low with the results below:

a. AND gate 2 is inhibited by a second low input and roll CAS operation cannot be reestablished without a reset signal.

b. Logic inverter 3 goes high and the caution panel CAS ROLL light comes on. c. OR gates 3 and 4 are triggered and switches 1 and 2 go to ground.

1-125. When switches 1 and 2 go to ground, dynamic pressure sensor excitation is removed and the limiter output is grounded. The CAS shuts down and prevents roll signals from being applied to the differential series stabilator circuits.

1-126. **YAW CAS OPERATION.** When the left or right rudder pedal is pushed, the mechanical controls move the rudders. Simultaneously, yaw trim actuator rudder pedal position LVDT signals produce a CAS yaw command. As the aircraft moves, yaw rate sensors and lateral acceleration sensors generate feedback signals. If the sum of lateral acceleration and yaw rate signals is equal to the sum of rudder pedal position and ARI command signals, signals to the rudder servoamplifiers are nulled. If the sums of the signals are different, yaw CAS adds to or subtracts from the rudder positions commanded by the mechanical system until the electrical signals are nulled.

1-127. The mechanical system can command ±15° of rudder deflection. If yaw CAS is engaged, maximum rudder deflection is ±30°.

1-128. **YAW CAS ENGAGE LOGIC.** See figure 1-15. Yaw CAS is engaged when the outputs of AND gates 4A and 4B are high. The channel A logic is primarily related to the left rudder actuator. The channel B logic is primarily concerned with the right rudder actuator. Since A logic is explained and the B logic differences are described. When the engaging controller YAW CAS switch is OFF, the outputs of logic inverters1A, 4A and 5A are high, AND gates 1A and 2A are triggered and the low logic inverter 2A and 3A outputs inhibit AND gate 3A. As the YAW CAS switch is moved from OFF to ON, a momentary reset signal is produced. The reset signal triggers the one shot multivibrator and the multivibrator output changes the logic inverter 1A output to low. AND gates 1A and 2A are inhibited for the length of the reset pulse and the high logic inverter 2A and 3A outputs trigger AND gate 3A. If a rudder servo fail signal does not exist and yaw rate is less than 41.5°/sec, AND gate 4A is triggered when the YAW CAS switch is set to ON. The high AND gate 4A output closes transistor switch 1A and, through logic inverter 5, inhibits AND gate 1A. The transistor switch 1A output energizes the left rudder actuator solenoid valve and, through logic inverter 4A, inhibits AND gate 2A. When the one shot reset pulse decays, AND gates 1A and 2A remains inhibited by logic inverts 4A and 5A. The delay circuit in the shutdown logic make sure of positive engagement and reduces the possibility of nuisance disengagements. The channel B logic differs from the channel A logic in two areas: the right rudder solenoid valve is energized when ground is applied and ground is supplied when transistor switch 2 closes. The solenoid valves are in series and both must be energized to operate yaw CAS. As yaw CAS commands rudder movements, the main control valve LVDT in each actuator supplies position signals to the roll/yaw computer. The signals are compared in summing amplifiers 1A and 1B. The level detectors are activated by signal differences and produce the rudder servo fail signals. The level detector monitor characteristics are such that large errors are detected with a minimum of delay while small errors must be maintained over a longer period to produce a rudder servo fail signal. The monitor characteristics reduce the chance of nuisance disengagements while providing positive failure detection ability. The channel A and channel B yaw CAS on signals are applied to AND gate 5 and to the roll CAS channel. The high AND gate 5 output is applied to logic inverts 7 and 8. The low logic inverter 8 output opens the caution light logic unit transistor switch and the CAS YAW light goes out.

1-129. **YAW CAS CIRCUIT.** See figure 1-16. The dual channels of the yaw CAS circuits are parallel within the AFCS components. The A channel drives the left rudder servoactuator while the B channel drives the right. Since the two channels operate in the same way, only the A channel is explained. The channel A yaw CAS command signal starts in the directional feel trim actuator rudder pedal position LVDT A. The demodulated rudder pedal position signal is applied to the summing amplifier 4A and combined with the electrical ARI signal from the pulse width multiplier. The amplifier 4A output is applied to the structural filter. The structural filter is tuned to the aircraft lateral bending mode and filters out oscillations. The structural filter output is applied to summing amplifier 7A. The amplifier 7A output is applied to servoamplifier 9A and to branch balancing amplifiers 6A and 6B. Servoamplifier 9A drives the left rudder electrohydraulic valve. If yaw CAS is engaged, the left rudder actuator solenoid valve applies hydraulic pressure to the EHV through the CAS mode switching valve. The EHV applies control pressure through the switching valve to the main control valve (MCV). When the MCV moves, the rudder actuator and the MCV dynamic sleeve also move. The MCV position LVDT provides feedback and fault detection signals. Feedback is applied to servoamplifier 9A. Fault detection is shown in figure 1-15. As the aircraft maneuvers in flight, yaw rate sensors and lateral acceleration sensors detect aircraft response to the rudder pedal command. Demodulated and cancelled yaw rate is summed with lateral acceleration at amplifier 2A. The sum of lateral acceleration and yaw rate is summed with the branch balancing integrator output at amplifier 5A. The amplifier 5A output cancels the rudder pedal position and ARI signal at amplifier 7A when aircraft response equals desired command. Branch balancing is effective through the action when the channel A and channel B outputs of amplifiers 7A and 7B are compared and integrated through the analog switch. Branch balancing is inhibited when the landing gear handle is moved to the down position.

1-130. **ARI Function.** The pulse width multiplier provides the outputs required for coordinated turns and is the CAS equivalent of the primary flight control system ARI. The pulse width multiplier is a variable gain amplifier. If the aircraft flies straight and level in pitch, the aircraft can be rolled without inducing any rudder movement. But, as the angle-of-attack increases, roll rate signals are amplified so that at 31.6° AOA, the pulse width multiplier provides a gain of 9.4 to the roll rate inputs. The amplified roll rate signals drive the rudder servoactuators for coordinated turns. As aircraft airspeed increases above Mach 1.5, the left and right air inlet controller Mach > 1.5 switches close. When either switch closes, the Mach < 1.5 signal is lost. The transistor switch closes and applies the Mach limits bias to the summing amplifier. The bias opposes the AOA signal and reduces the pulse width multiplier rate of gain.

1-131. **AC Interlock and Yaw Rate Limits.** Buffered yaw rate signals are applied to the ±41.5°/sec level detector. If yaw rate exceeds ±41.5°/sec, the high level detector output inhibits logic inverter 1A and shuts down yaw CAS. Since yaw CAS must be engaged to engage roll CAS, roll CAS also shuts down. The excessive yaw rate discrete signal is also delayed and applied to the pitch CAS engage logic so pitch CAS also shuts down after a small delay. The 26VAC applied to the rate sensor is compared with the computer 26VAC. The 26VAC interlock output of the yaw rate gyro is routed through the force sensor on F-15C/D and rear force sensor on F-15D before being compared to the computer 26VAC. If voltage to the rate sensor is interrupted, the amplifier 1A output triggers the level detector. The level detector output prevents yaw CAS and pitch CAS engagement.

1-132. **SIMPLIFIED RUDDER ELECTROHYDRAULIC OPERATION.** See figure 1-17. Before yaw CAS is engaged, the solenoid valve is deenergized, the return port is closed, and hydraulic pressure is applied to the top of the CAS mode switch valve. The CAS mode

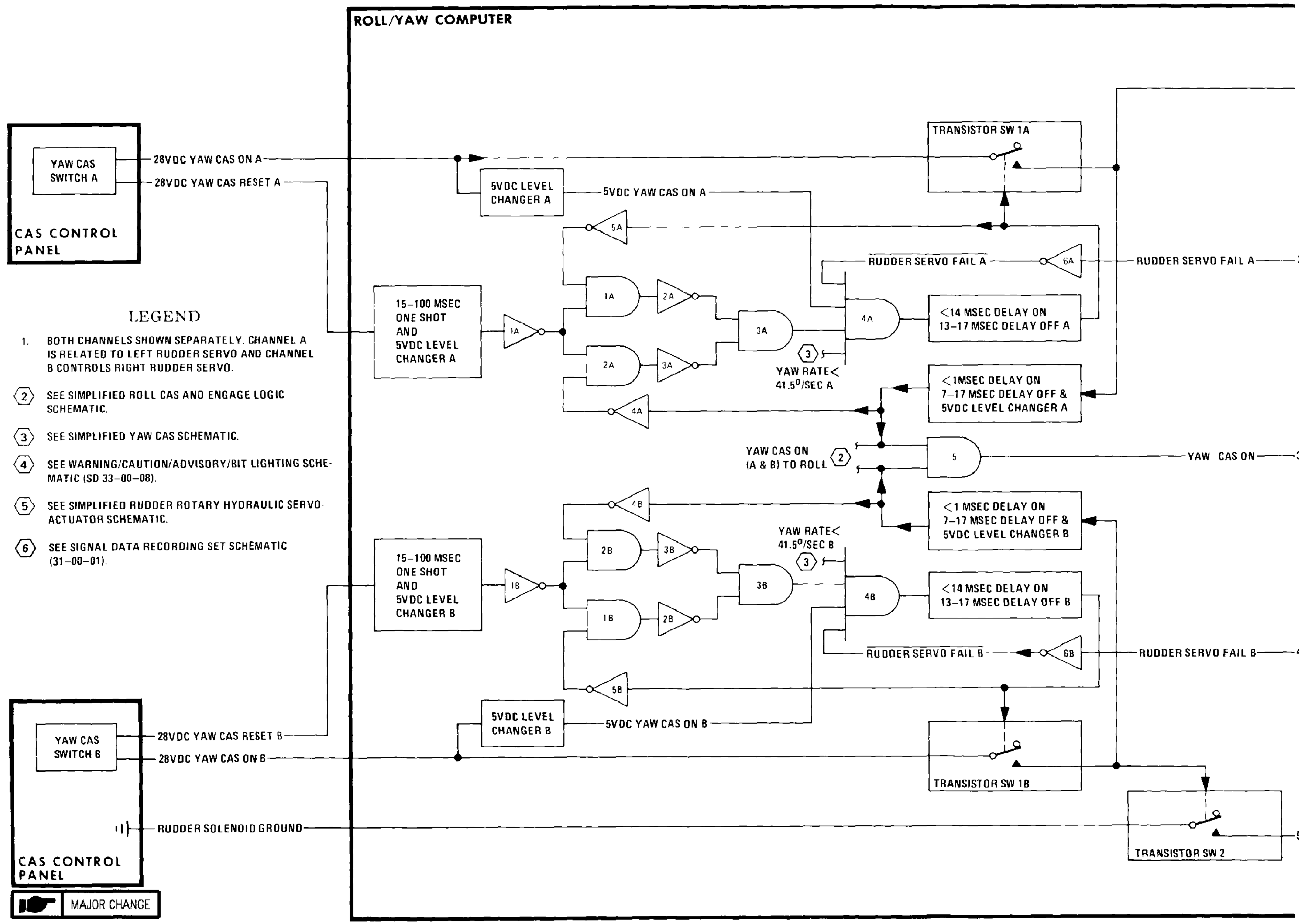
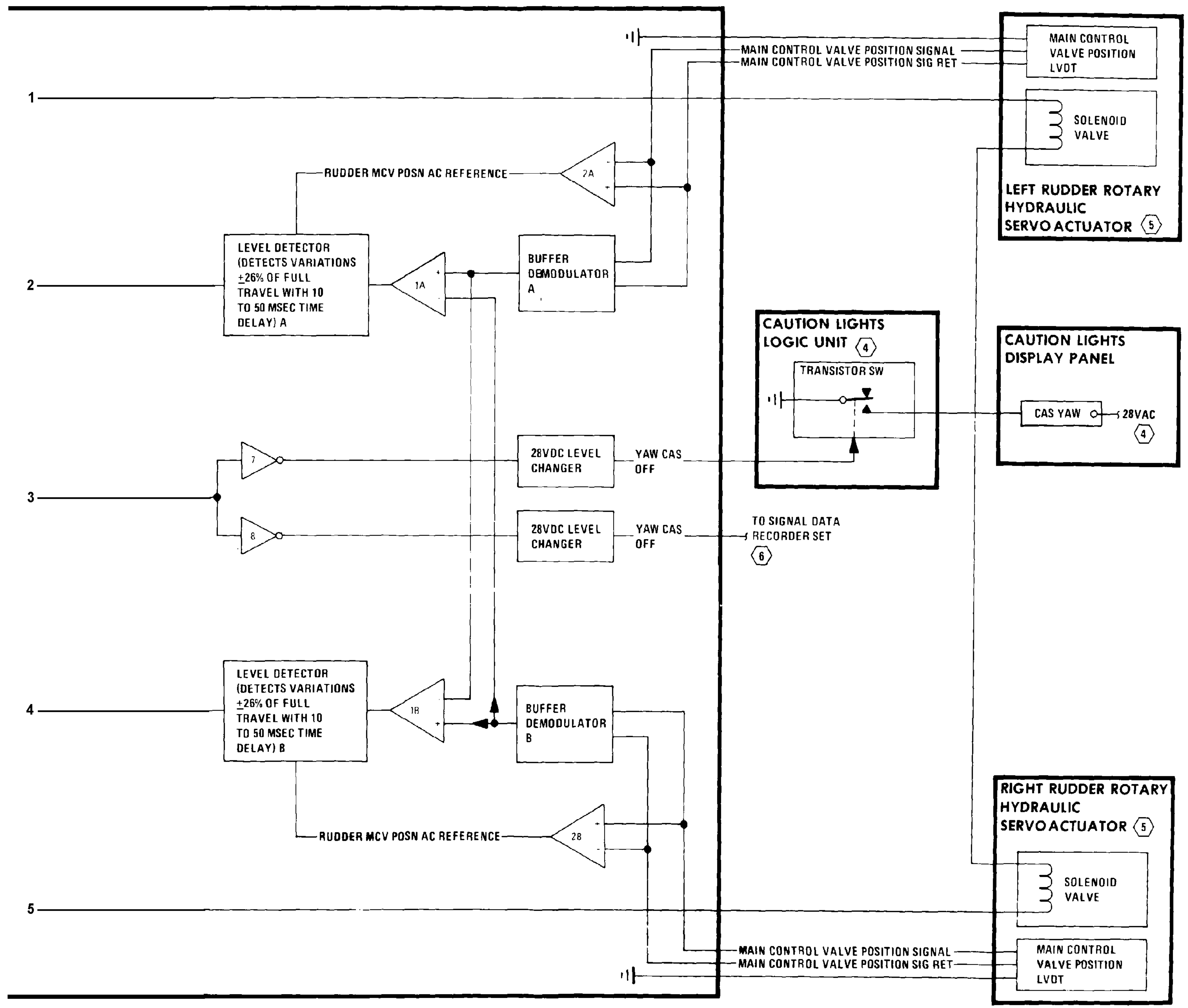


Figure 1-15. Simplified Yaw CAS Engage Logic Schematic (Sheet 1 of 2)



MAJOR CHANGE

Figure 1-15. Simplified Yaw CAS Engage Logic Schematic (Sheet 2)

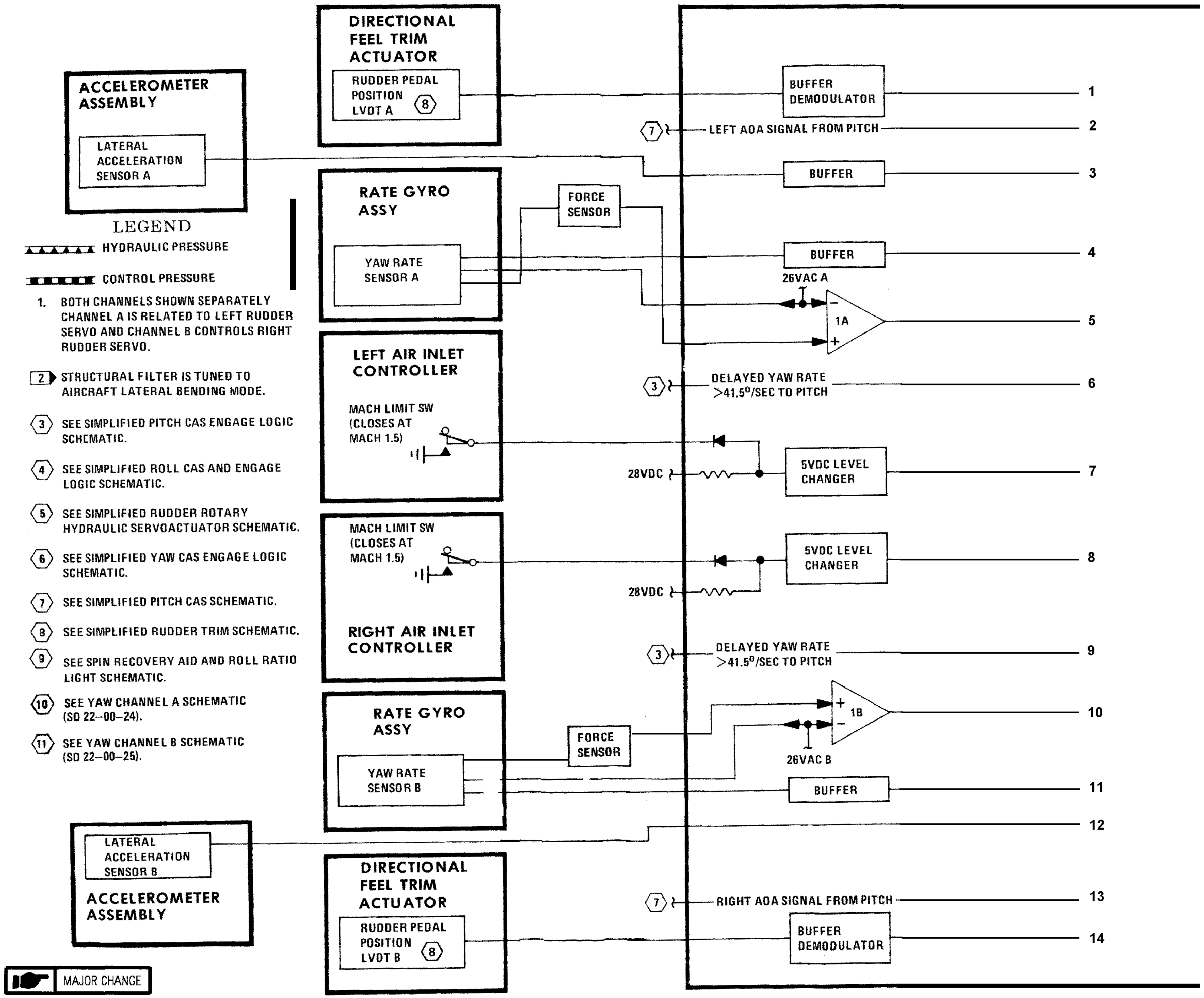


Figure 1-16. Simplified Yaw CAS Schematic (Sheet 1 of 3)

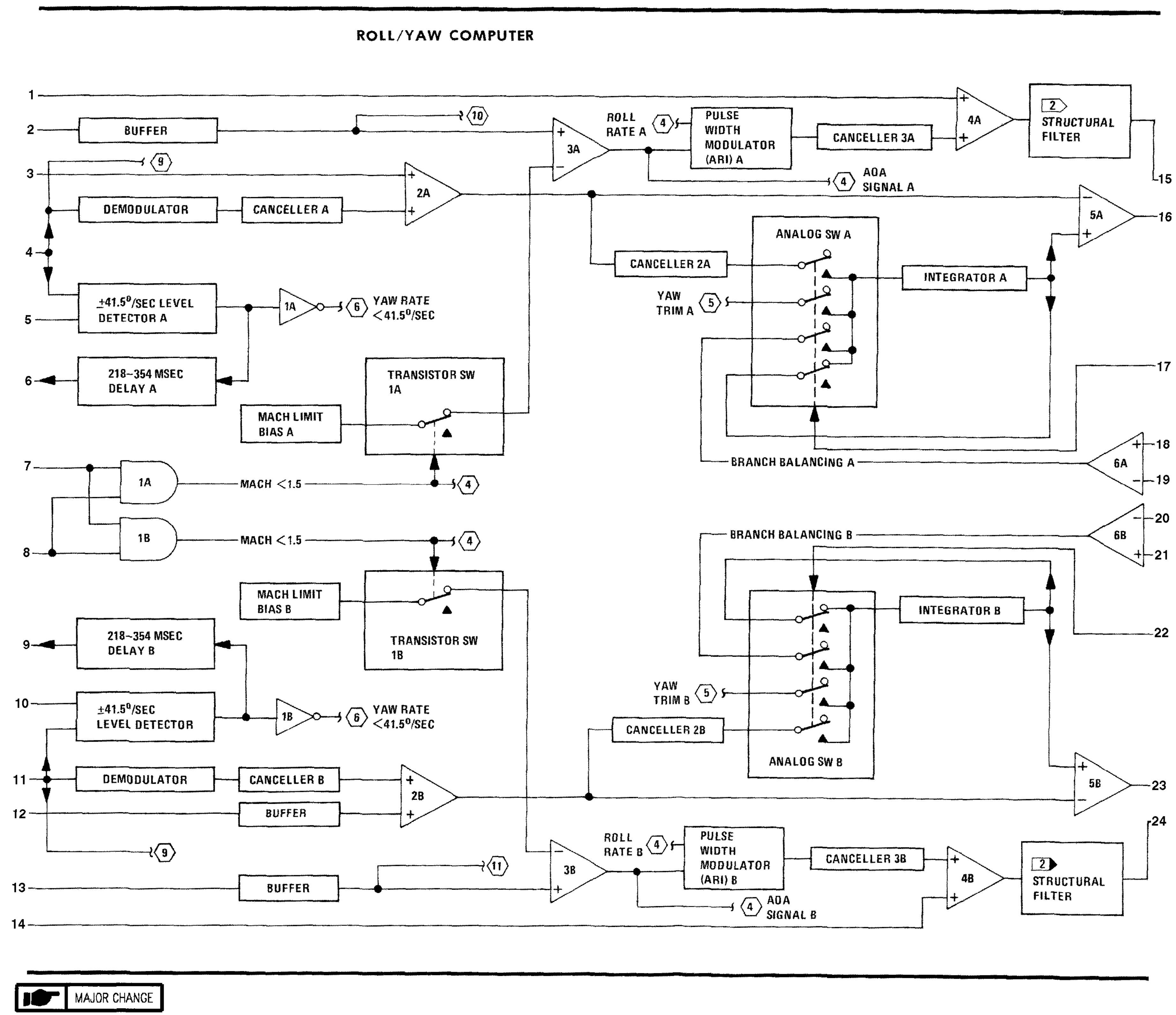
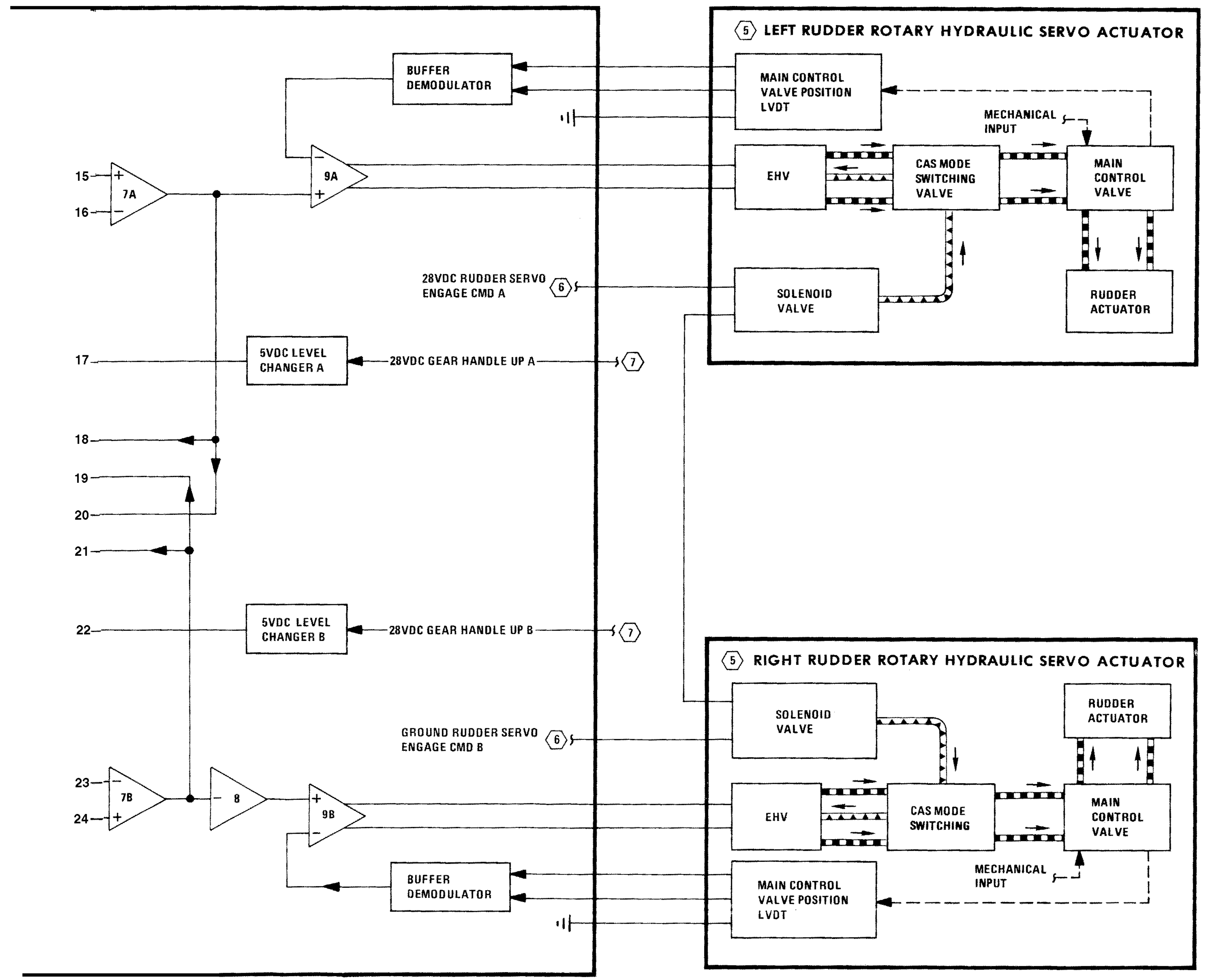


Figure 1-16. Simplified Yaw CAS Schematic (Sheet 2)



MAJOR CHANGE

Figure 1-16. Simplified Yaw CAS Schematic (Sheet 3) switch valve plunger is held down, removing hydraulic pressure from the EHV and isolating the servovalve from EHV control pressures. The solenoid valve also applies locking pressure to the MCV dynamic sleeve. As the rudder pedals are moved, the MCV piston moves and ports regulated hydraulic pressure to the rotary valve. As the valve rotates in the command direction (moving the rudder), the helical coupling translates the rotation into linear extention and retraction of the mechanical follow-up linkage. The follow-up linkage moves the MCV piston in a direction opposite to the direction commanded by the rudder pedal. When the rudder has moved to the commanded position, the MCV piston is centered and rotation stops.

1-133. When yaw CAS is engaged, the solenoid valve energizes, the return port opens, and the pressure port closes. As the pressure at the top of the CAS mode switch valve is removed, trapped hydraulic pressure is ported to return. Since pressure is available at the bottom port, the plunger moves up and pressure is applied to the EHV. Since locking pressure is also removed from the MCV dynamic sleeve, the sleeve is free to receive EHV rudder left (RL) and rudder right (RR) commands. When a right servo command is received, the EHV plunger moves down and applies RR control pressure (by way of the open CAS mode switching valve) to the MCV. The MCV sleeve moves up and ports RR control pressure to the hydraulic rotary shaft cavities. The sleeve movement moves the MCV position LVDT. The rudder MCV position signal nulls the servo command when the dynamic sleeve has moved the required distance. As the rotary shaft rotates to the right, the mechanical follow-up linkage extends and drives the MCV plunger up to block the MCV RR control pressure port. The mechanical follow-up is active even if mechanical controls are jammed or disconnected. RL control pressure rotates the rotary shaft to the left and retracts the mechanical follow-up linkage.

1-134. Just as primary directional controls move the rudders independently of CAS commands, CAS commands deflect the rudders without mechanical inputs. The systems work together and only limited rudder deflection is available independently. The limitation is restricted by the relation between MCV plunger and dynamic sleeve. If CAS is off, the sleeve is stationary and the amount of mechanical follow-up motion to block the MCV RL or RR control ports is relatively small. If CAS is engaged, the dynamic sleeve also moves to command a rudder deflection. The follow-up linkage must move a greater distance to block the ports; so, a greater rudder movement is arrived at when yaw CAS is engaged.

1-135. When the MCV is pressurized, the damping pressure reservoir is also pressurized. The damping pressure smooths out rudder movements during normal operation and prevents rudder oscillations in flight if pressure is lost to the rudder actuator. Rudder actuators move together in normal flight, but independently of one another. Even if one rudder actuator fails, the other actuator can continue to function.

1-136. **PILOT RELIEF MODES.** AFCS CAS operation fulfills the primary AFCS function and

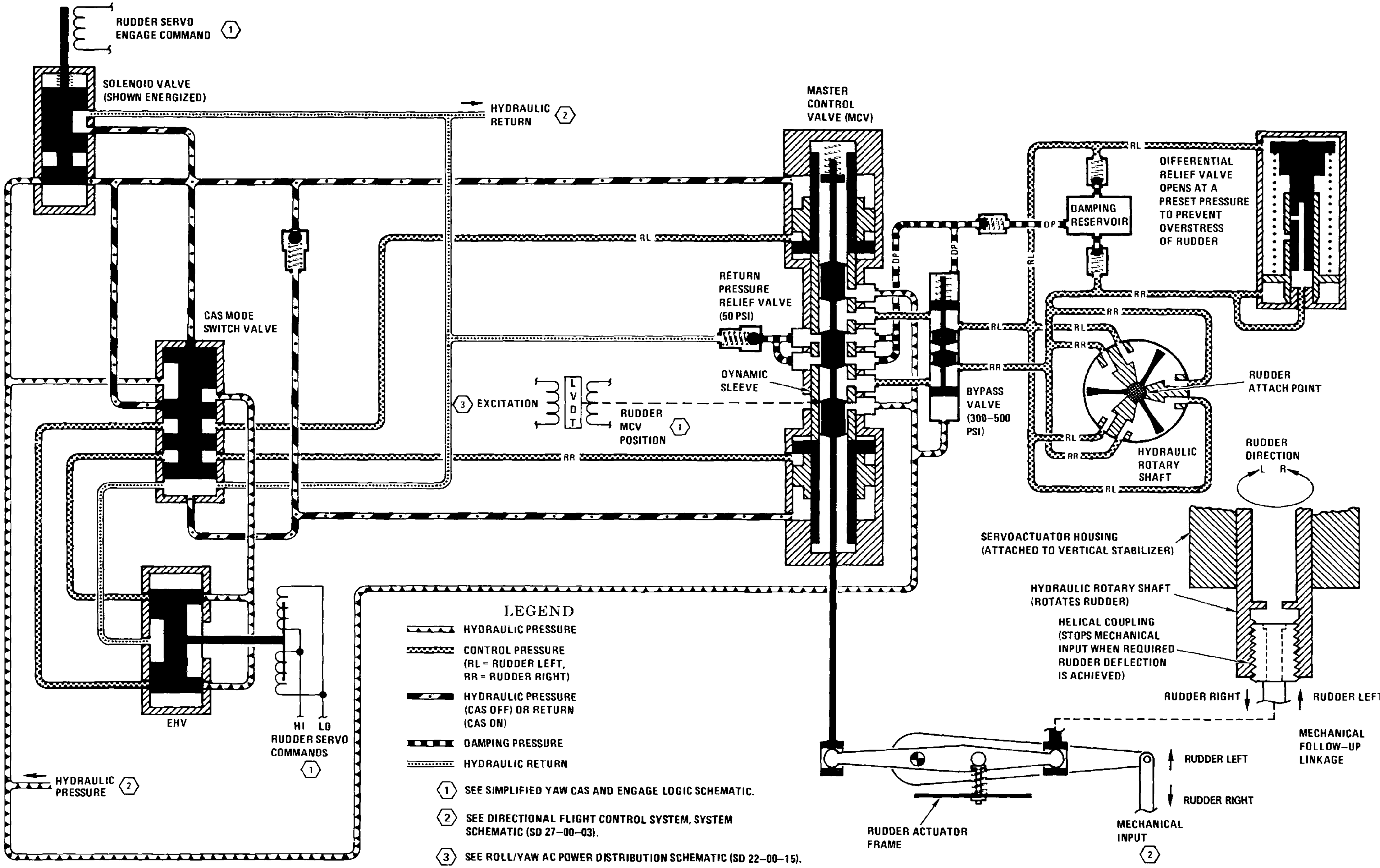


Figure 1-17. Simplified Rudder Rotary Hydraulic Servoactuator Schematic

pilot relief modes provide a secondary improvement of flying characteristics. Since CAS operates as a closed loop and pilot relief pitch and roll signals are inserted into the CAS loop, the pilot relief loops are called pitch outer loop and roll outer loop. There is no pilot relief function in yaw.

1-137. Two pilot relief modes are available: attitude hold (ATT HOLD) and altitude hold (ALT HOLD). To engage ATT HOLD, all CAS axes must be operating correctly. To engage ALT HOLD, ATT HOLD must be engaged.

1-138. **Attitude Holding.** To hold a fixed position in pitch and roll, the engaging controller ATT HOLD switch is set to ON. If pitch outer loop (POL) integrator commands are within 1.0 ±0.26g and roll outer loop (ROL) commands are within ±3.5°/sec, INS attitude signals become the AFCS reference attitude. If there is a change in aircraft attitude, the AFCS outer loops inject signals into the applicable CAS loops to drive the aircraft back to the reference attitude. If a new reference attitude is required, a force is applied to the control stick. If the force is enough to establish control stick steering, the attitude holding function is interrupted, but the ATT HOLD switch remains ON. When force is removed from the control stick, the attitude holding function is restored with the newly acquired reference attitude.

1-139. **Attitude Hold Interrupt.** The ATT HOLD function is not in operation if pitch attitude is greater than ±45° roll attitude is greater than ± 62.5°, or W-pitch control stick steering (CSS) is in effect, but the ATT HOLD switch remains ON. When the condition preventing ATT HOLD operation is removed, the ATT HOLD operation is started or resumed if interrupted. The reference attitude is the attitude at the time the inhibiting signal was removed.

1-140. The ATT HOLD switch physically disengages (moves to OFF) under any of the conditions below:

a. Normal acceleration is greater than +4.0g or less than 0.0g for more than 500 msec.

b. INS attitude signals are not valid.

c. Any CAS axis is disengaged (manually or automatically).

d. ROL signal exceeds ±17.4°/sec for more than 1 second.

e. Stick force sensor Autopilot Disengage Switch (ADS) is pressed.

f. ATT HOLD switch is manually positioned to OFF.

1-141. **Altitude Holding.** To hold a fixed altitude, the ATT HOLD switch is engaged. When ATT HOLD is engaged, the CAS control panel ALT HOLD switch is set to ON. If air data computer (ADC) altitude error and INS vertical velocity signals are valid and vertical velocity is within ±2000 feet per minute, the ALT HOLD switch is magnetically held to the ON position. The ALT HOLD function is not in operation if pitch CSS is effective or roll attitude exceeds ±62.5°, but the ALT HOLD switch remains ON. When the condition inhibiting ALT HOLD operation is removed, the ALT HOLD operation is started, or resumed if interrupted. The reference altitude is the altitude at the time the last inhibiting signal is removed.

1-142. Once a reference altitude has been established, ADC altitude error signals and INS vertical velocity signals are processed in the POL and fed into the pitch CAS loop. The POL signals command the required control surface movements to arrive at and hold the reference altitude. If vertical velocity is within ±2000 feet per minute, the aircraft is within altitude hold limits, which remain the same after altitude hold engage.

1-143. **Altitude Hold Interrupt.** The ALT HOLD function is not in operation if pitch CSS is in effect or if roll attitude is greater than ±62.5°, but the ALT HOLD switch remains ON. When the condition preventing ALT HOLD operation is removed, the reference altitude is established and ALT HOLD operation starts.

1-144. The ALT HOLD switch physically disengages (moves to OFF) under any of the conditions below:

a. When ATT HOLD disengages for any reason. Refer to paragraph 1-140.

b. Altitude error validity or vertical velocity validity fail.

c. ALT HOLD switch is physically moved to OFF.

d. Vertical velocity exceeds +2000 feet per minute.

e. ALT HOLD mode operation during the transonic region, from 0.9 to 1.1 mach, may or may not result in pitch oscillations and/or altitude hold mode disengagement.

1-145. **PILOT RELIEF ENGAGE LOGIC CIRCUIT.** See figure 1-18.

1-146. **Attitude Hold Engage.** When the ATT HOLD switch is set to ON, 28VDC (through the ADS interlock) is applied to the ATT HOLD switch bottom contact, to the ALT HOLD switch top contact, to a 5VDC level changer and to the pilot relief circuit (for roll attitude engagement).

1-147. The 28VDC applied to the ATT HOLD switch bottom contacts is applied to switch 2. If OR gate 4 is low, the logic inverter 6 output is high and switch 2 closes; the 28VDC is applied to the switch holding coil and the ATT HOLD switch is held to ON position. OR gate 4 is low as long as all of the inputs are low. If normal acceleration is not greater than +4.0g or less than 0.0g, the top input is low; if pitch CAS is not shut down, the second input is low; if the INS is valid, logic inverter 2 provides the next low; if the roll attitude interlock is low, the fourth low input is provided; and before engagement only, if POL integrator commands are within 1.0 ±0.26g, the fifth low input is provided. After engagement, the same voltage which is applied to the holding coil activates OR gate 1 and keeps the logic inverter 3 output low. If any input to OR gate 4 goes high, the gate is triggered, logic inverter 6 goes low, switch 2 opens and the ATT HOLD switch deenergizes.

1-148. When the ATT HOLD coil is energized, top inputs to AND gates 1 and 5 are high. If pitch attitude is less than ±45°, roll attitude less than ±62.5°, and pitch CSS is not in effect, OR gate 2 output is low and the bottom AND gate 1 input is high. AND gate 1 is triggered and provides the second high input to AND gate 5 and one of the high inputs to OR gates 5 and 7.

1-149. AND gate 5 triggers OR gate 6. The high OR gate 6 output provides the required switching voltages for the PPR limiter bias and the POL integrator. The OR gate 5 output activates the POL switches that supply pitch pilot relief (PPR) signals to the pitch CAS loop. If ALT HOLD is not engaged, OR gate 5 also triggers AND gate 6 through OR gate 7 to disable the pitch synchronizer.

1-150. **Altitude Hold Engage.** If the ATT HOLD switch is ON, 28VDC is available at the top contacts of the ALT HOLD switch. If ALT HOLD is set to ON, the 28VDC is supplied to a 5VDC level changer and (through the switch bottom contacts) to switch 1. If both altitude error and vertical velocity error are valid and vertical velocity is within ±2000 feet per minute, 28VDC from switch 1 is applied to the holding coil and to another level changer. When the ALT HOLD switch coil holds the switch to the ON position, highs exist at the following points below: At the top input of AND gate 3, at the bottom input of AND gate 4, at logic inverter 7 and at the bottom input of AND gate 2. If roll attitude is less than ±62.5° and pitch CSS is not in effect, the OR gate 3 output is low and the logic inverter 5 output is high. AND gate 2 is triggered and provides high input to AND gates 3 and 4, to the pitch CAS analog switch, to the CSS circuit and to the altitude hold switches. The second high inputs to AND gates 3 and 4 provide alternate inputs to OR gates 5 and 6. Even if pitch attitude goes past the ±45° level and the AND gate 1 output goes low, OR gates 5 and 6 outputs remain high (as long as AND gates 3 and 4 are triggered). The OR gate 6 output controls the POL fade/engage switch and the OR gate 5 output controls the POL switches as in ATT HOLD. Since ALT HOLD is engaged, the logic inverter 7 low output inhibits AND gate 6. When the AND gate 6 output is low, the synchronizer is enabled and pitch attitude error signals are nulled.

1-151. **PILOT RELIEF CIRCUIT.** See figure 1-19. The pilot relief circuit indicates how INS pitch attitude signals are used to hold pitch attitude during ATT HOLD operation and how ADC altitude error signals and INS vertical

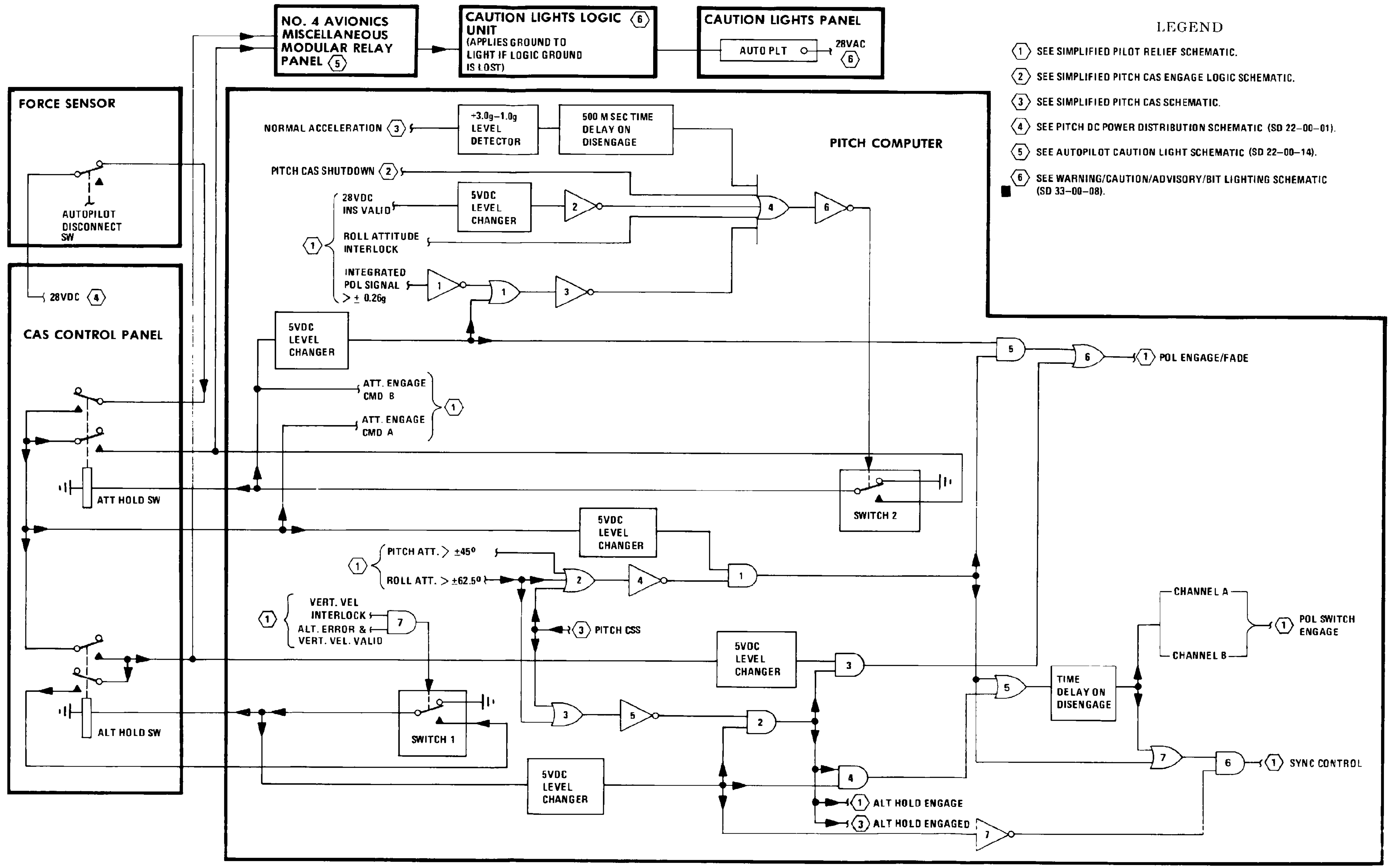


Figure 1-18. Simplified Pilot Relief Engage Logic Schematic

velocity signals are used to hold altitude during ALT HOLD operation. The circuit also shows that INS roll signals are used to hold roll attitude in ATT HOLD and ALT HOLD operation. Extensions of the logic required for both modes are also shown.

1-152. **Solid State Sync Drives.** The pitch computer solid state sync drive and the roll/yaw computer solid state sync drive are required to detect attitude errors. Since both sync drives operate in the same way, only the pitch computer solid state sync drive is explained.

1-153. Demodulated pitch attitude signals are applied to summing amplifier 8. The amplifier output is applied to zero level detector. The zero level detector output is applied as an add/subtract signal to a pulse width modulator which is driven by a 250 kHz clock. The pulse width modulator output is applied (through normally closed (NC) switch 5 contacts, a filter and an integrator) to summing amplifier 8. If pitch attitude is at 0°, the 250 kHz clock drives the pulse width modulator (without interference from add/subtract signals) so that the inverting input to amplifier 8 is a series of equally spaced negative and positive pulses. The amplifier 8 output is a symmetrical sawtooth with an average value of zero. If pitch attitude goes positive, the zero level detector supplies a subtract signal changing the pulse width modulator output ratio so that the positive going pulses are longer than the negative going pulses. The summing amplifier output still maintains an average value of zero but the sawtooth is asymmetrical to subtract from the positive attitude signal. Conversely, if the attitude signal is negative, the zero level detector add signal causes the summing amplifier output to add to the negative attitude signal.

1-154. When ATT HOLD is engaged, switch 5 goes to ground and the pulse width modulator output is removed from amplifier 8. Since the amplifier output was held at an average value of zero before engagement, any deviation from pitch attitude at the time of engagement is supplied as a pitch attitude error to the POL circuit. If ALT HOLD is engaged, control voltage is removed from switch 5 and the synchronizer is again enabled providing a zero average altitude error output.

1-155. **Pitch Computer Logic.** See figure 1-19.

1-156. Attitude Hold. The inertial navigation unit (INU) INS validity switch provides a 28VDC signal to the pilot relief circuit if attitude signals are valid. Buffered and demodulated pitch signals are applied to ±45° level detectors. If pitch attitude exceeds ±45°, OR gate 2 is triggered and attitude holding operation is inhibited. If ATT HOLD is engaged and ALT HOLD is disengaged, switch 5 goes to ground and pitch attitude error is applied to summing amplifier 12.

1-157. The POL engage/fade signal from the pilot relief engage logic circuit exists if condition required for ATT HOLD operation are satisfied. The POL engage/fade signal is applied directly to switches 1, 4 and 6. Switch 1 applies maximum limiter bias to the fader. Switches 4 and 6 integrate the unlimited pitch pilot relief (PPR) signals.

1-158. Altitude Hold. The ADC supplies a 15VDC signal to the INU if altitude signals are valid. If vertical velocity is valid, the INU supplies an altitude error and vertical velocity valid signal to the pilot relief engage logic. If ALT HOLD is engaged and operating, the ALT HOLD engaged signal is applied to a 12VDC level changer and to switches 2 and 3. Switches 2 and 3 control the altitude error signal input to summing amplifier 12. The 12VDC level changer disables the ADC altitude error synchronizer and ADC deviations from reference altitude are applied to buffer amplifier 1. INU vertical velocity is applied to buffer amplifier 2.

1-159. Altitude error (change in altitude) and vertical velocity are summed at amplifier 5 to arrive at an altitude error in feet.

1-160. As the AFCS drives the aircraft towards reference altitude, the altitude error is reduced. Summing amplifier 9 combines altitude error with vertical velocity and cancelled pitch attitude. When the ALT HOLD engaged signal is not present, logic switch 2 is closed and switch 3 is open. This permits the output of amplifier 9 to appear at the input of the fader only. When the ALT HOLD engaged signal is present, logic switch 2 opens and 3 closes. This grounds the fader input and removes the ground on switch 2 allowing the output of amplifier 9 to be applied to summing amplifier 12. The fader output is such that it initially opposes the output of amplifier 9 providing an easy-on circuit preventing large errors from being immediately switched into amplifier 12.

1-161. Pitch Pilot Relief Signal (PPR). Summing amplifier 12 is the source of PPR signal. If ALT HOLD is engaged, the input is altitude error plus vertical velocity and cancelled pitch attitude; if only ATT HOLD is engaged, the signal is pitch attitude error. The signal is applied to summing

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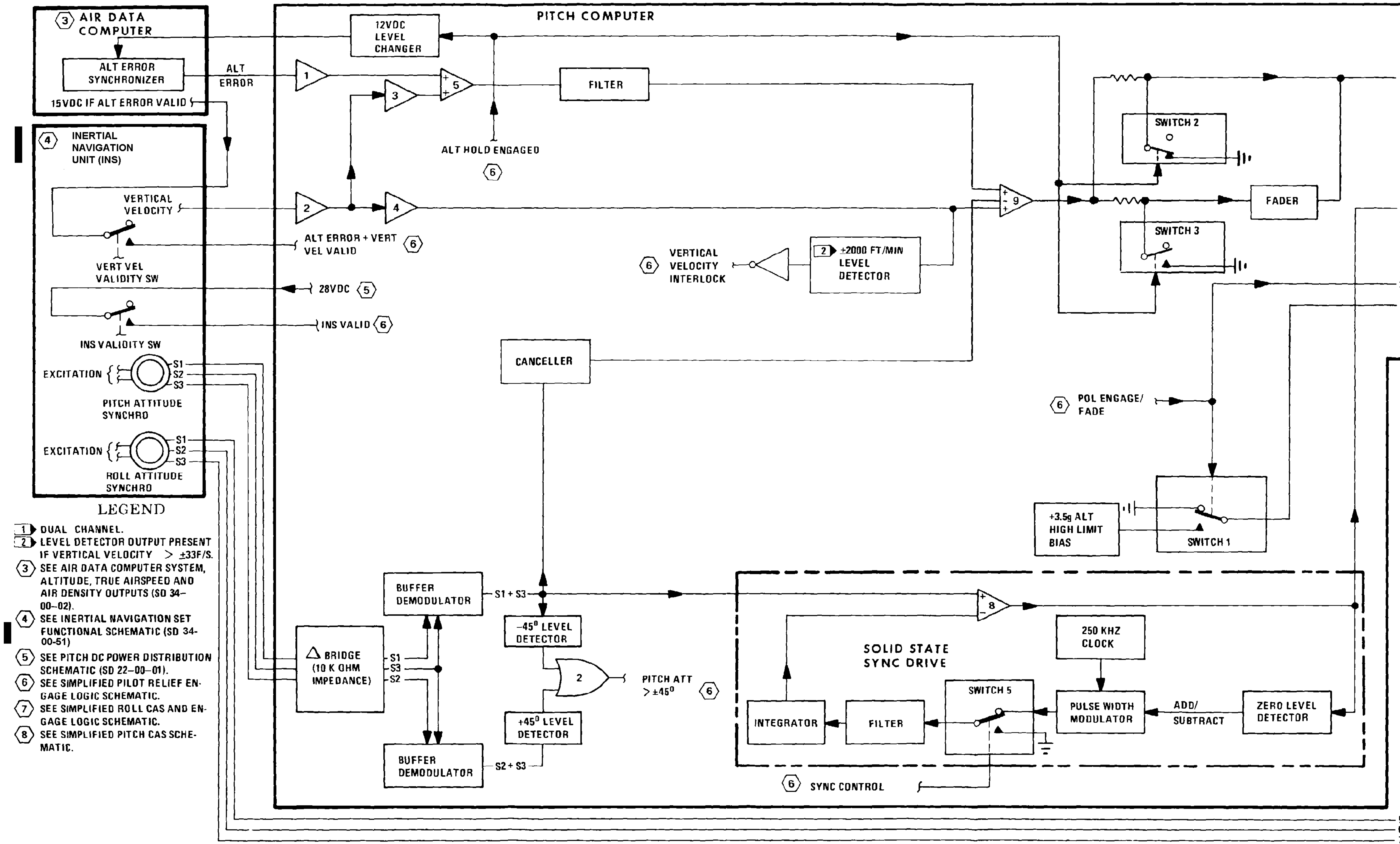


Figure 1-19. Simplified Pilot Relief Schematic (Sheet 1 of 2)

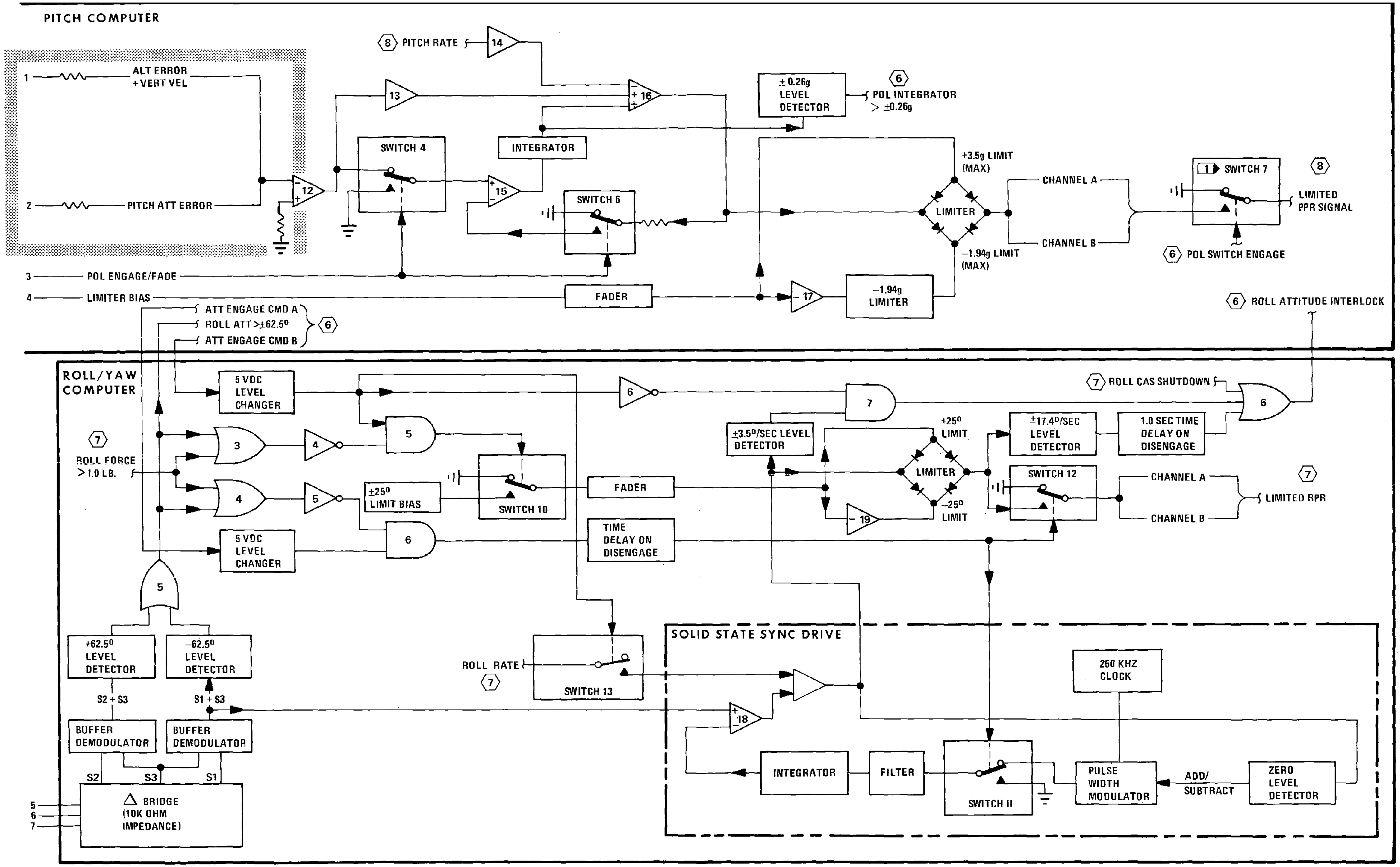


Figure 1-19. Simplified Pilot Relief Schematic (Sheet 2)

amplifier 16 and combined with pitch rate and integrated PPR signals. Amplifier 16 output is applied to the PPR limiter. Positive limit bias is applied directly from the fader. Negative limit bias is developed by inverting amplifier 17 through the -1.94g limiter. Integrated PPR signals are also applied to the 0.26g detector. The limiter output is divided into two channels and inserted into the pitch CAS loop through switch 1 (POL switch).

1-162. **Roll Yaw Computer Logic.** See figure 1-19. INU roll attitude is buffered, demodulated, and applied to the roll solid state synchronizer and to the +62.5° level detectors. The roll solid state synchronizer works the same as the pitch synchronizer but is not enabled when ALT HOLD is engaged. For all practical purposes, the roll axis is in ATT HOLD even when ALT HOLD is engaged.

1-163. Attitude Hold. Pitch computer attitude engage commands A and B are applied to 5VDC level changers. Command A is applied to AND gate 6, command B is applied to AND gate 5 and to logic inverter 6. Assuming that OR gates 3 and 4 are inhibited, highs from logic inverters 4 and 5 trigger the AND gates. AND gate 5 activates switch 10. Switch 10 applies bias to the roll pilot relief limiter and opens the limits to +25°. AND gate 6 activates switches 11 and 12. Switch 11 disables the solid state synchronizer to establish the reference roll attitude. Switch 12 applies the limited Roll Pilot Relief (RPR) signal to the roll CAS circuit.

1-164. Pre-Engage Logic. The roll solid state synchronizer tries to maintain the roll attitude output to an average zero level before ATT HOLD is engaged. If the aircraft is rolling, the synchronizer may not be able to maintain the zero level output. Logic inverter 6 provides a high input to AND gate 7. If roll rate is greater than ±3.5°/sec, the high ±3.5°/sec level detector output triggers AND gate 7. AND gate 7 triggers OR gate 6 and the roll attitude interlock signal prevents ATT HOLD engagement. If roll attitude is greater than ±62.5°, the high output from either level detector triggers OR gate 5. When the OR gate 5 output is high, the logic input to the pitch computer inhibits pitch attitude operation. OR gate 5 also triggers OR gates 3 and 4 and then inhibits AND gates 5 and 6. If roll force is greater than 1.0 lb (left or right), OR gates 3 and 4 are also triggered with the same results, but only the roll channel is inhibited because the roll force logic (roll CSS) is not transmitted to the pitch computer.

1-165. Post-Engagement Logic. Roll CSS may be started after engagement. If lateral stick force exceeds ±1.0 lb, roll attitude holding is interrupted and resumes (with a new reference roll attitude) when the force is removed. If the limited RPR signal exceeds ±17.4°/sec, the high level detector output triggers OR gate 6 and produces the roll attitude interlock signal and disengages the ATT HOLD switch. OR gate 6 can also be triggered if roll CAS shuts down (manually or automatically). Since yaw CAS must be engaged and operating correctly before roll CAS can be engaged, a yaw CAS shutdown also triggers OR gate 6 and produces a roll attitude interlock signal.

1-166. **AUTOPILOT CAUTION LIGHT.** The AUTO PLT caution light comes on if the ATT HOLD and/or the ALT HOLD switches disengage. The caution light is turned off by pressing the MASTER CAUTION light switch (master caution reset).

1-167. **Initial Condition.** When power is initially applied, relays K1 through K7 in the No. 4 avionics modular relay panel are deenergized. Ground for the caution lights logic unit is supplied through energized contacts of K1, K5, or K6 for AUTO PLT light to be off. When ground to the caution lights logic unit is removed, a ground is supplied to the caution light display panel which lights the AUTO PLT light. Since all relays are deenergized, the AUTO PLT caution light comes on. If the MASTER CAUTION light-switch is pressed one time, 28VDC is applied to the coils of relays K6 and K7. When K6 energizes, ground is supplied to the caution lights logic unit and the AUTO PLT light goes off. Energized contacts of K6 also apply 28VDC to the coils of K6 and K7 to hold the relays energized after the MASTER CAUTION light switch is released. The holding voltage for relays K6 and K7 is interlocked through deenergized contacts of K2 and K3. The contacts of K7 are used to provide a warning if the ALT HOLD switch disengages.

1-168. **Attitude Hold Disengage Warning.** If the ATT HOLD switch is set to ON, relays K1 and K2 energize. When K2 energizes, K6 and K7 deenergizes. Ground for the caution lights logic unit is supplied by the energized contacts of K1. The deenergized contacts of K4, K5, and K6 form an interlock for the ground signal. If the ATTHOLD switch disengages (moves to OFF), K1 deenergizes. When K1 deenergizes, the AUTOPLT light comes on. The MASTER CAUTIO and AUTO PLT lights are reset by momentarily pressing the MASTER CAUTION light switch.

1-169. **Altitude Hold Disengage Warning.** If the ATT HOLD and ALT HOLD switches are both set to ON, the relay logic is changed. The AUTO PLT light comes on, even if only the ALT HOLD switch disengages. When ATT HOLD is engaged,K1 and K2 energize. When ALT HOLD is engaged, K3 energizes. When K3 energizes, K4 and K5 also energize. The energizing voltage for K4 comes from 6CBC010, through energized contacts of K1, energized contacts of K3, and 93CRL028. When K4 energizes, 28VDC is applied through energized contacts of K4 and deenergized contacts of K7, back to the coil of K4. Relay K4 is now latched in the energized position even if K3 deenergizes. The energizing voltage for K5 is the same as the energizing voltage for K4, but does not go through 93CRL028. Energized contacts of K5 supply the ground signal to the caution lights logic unit, through deenergized contacts of K6, and the AUTO PLT light goes off.

1-170. If ALT HOLD deenergizes, K3 deenergizes. When K3 deenergizes, K5 deenergizes, but K4 remains energized. Energized contacts of K4 break the ground path to the caution lights logic unit and the AUTO PLT light comes on. If the MASTER CAUTION light switch is pressed, K6 and K7 energize. When K7 energizes, K4 deenergizes. When K6 energizes, ground is applied to the caution lights logic unit through energized contacts of K6. IF ATT HOLD is still engaged, K1 and K2 are energized. Now, when the MASTER CAUTION light switch is released, K6 and K7 deenergizes. The circuit now functions as described in paragraph 1-168 when the ATT HOLD switch is set to ON.

1-171. **AUTOMATIC SPEED BRAKE RETRACT.** If the aircraft nears a stall condition with the speed brake extended, the speed brake automatically retracts. The circuitry that activates the automatic retraction of the speed brake energizes when the true angle-of-attack is greater than 15.5° measured at approximately 25.5 units on the ANGLE OF ATTACK indicator. A transistor switch in the roll/yaw computer applies ground to speed brake auto retract relay 48K-L007. When the relay energizes, 28VDC is applied to solenoid A of the speed brake selector valve. At the same time, 28VDC is removed from solenoid B. The action of 48K-L007 solenoids of the selector valve is the same as setting the speed brake control switch to RETRACT.

1-172. **SPIN RECOVERY AID.** When the landing gear handle is up, 28VDC is applied to the coil of relay 41K-L095. A discrete ground for relay coil 41K-L095 is supplied through transistor switches in the roll/yaw computer. Yaw rate signals from the yaw rate gyro are supplied to the roll/yaw computer. If aircraft yaw rate exceeds ±60°sec, level detectors trigger the transistor switches which open the ground and deenergizes relay 41K-L095. With relay 41K-L095 deenergized, the roll gear down solenoid in the PRCA is energized, giving full roll approval.

1-172A**. SPIN RECOVERY AID DISPLAY (SRAD)** When the central computer (CC) detects a spin condition, the CC blanks the multi-purpose color display (MPCD) vertical situation display (VSD) and VSD repeater displays in both cockpits and displays a spin recovery aid display (SRAD). A spin condition is recognized when the yaw rate exceeds 57°/second or the yaw rate exceeds 41.5°/second and continues above 30°/second for more than 5 seconds with an indicated airspeed less than 175kts. The SRAD format displays the alphanumeric SPIN RECOVERY, AOA, calibrated airspeed and baro corrected altitude. In addition, the altitude flashes in below 10,000 feet AGL. See Figure 1-20. An arrow indicates stick position (left/right) and bars indicate recommended throttle position on the display. Along with the throttle position indicators, the alphanumeric THROTTLES will be displayed. During an inverted spin, the alphanumeric CONTROL NEUTRAL will be displayed with no stick arrow. If the INS data goes invalid during a spin, the spin arrow and throttle position indicators will be removed from the display. When the SRAD is active and the INS goes invalid and weight on wheels is set, the SRAD will be removed.

1-172B. If AOA is invalid then the AOA value and symbol will be removed from the display If indicated airspeed or baro corrected altitude is invalid then OFF will be displayed in place of the number. The baro corrected altitude is referenced to the elevation of the nearest waypoint (nearest destination altitude). If the baro corrected altitude is less than 10,000ft above the nearest destination altitude then the display altitude will flash at a 2.5Hz rate.

1-172C. If overload warning system (OWS) is requested while SRAD is displayed on the VSD, SRAD format will be removed and OWS Recall format will be displayed. With OWS Recall format displayed not the VSD and with the aircraft in a spin, if OWS is no longer requested, the Radar format will be displayed on the VSD. If a new spin is detected, SRAD format will be displayed regardless of active VSD format (OWS Recall or Radar).

1-172D. After recovery from the spin (yaw rate <20°/second), SPIN RECOVERY will be removed and RECOVER will be displayed for 4 seconds. The throttle bars will slew to the same height (within 2 seconds) and the stick position arrows will be removed. After 4 seconds, the displays return to their previously selected formats.

1-172E. The SRAD format can be deselected using PB11 on the MPCD display. Once the SRAD is deselected, it will not be reactivated until the aircraft recovers from the spin and a spin condition is again detected. Initiated BIT from the BCP, PACS manual selection or select jettison will deactivate the SRAD. The CC commands VTRS to record the SRAD and continue to record for 30 seconds after RECOVER is displayed.

1-173. **YAW (DEPARTURE) WARNING TONE.** If the aircraft yaw rate arrives at 30°/sec, a warning tone cycling at 2 Hz for 30°/sec, and increasing linearly up to 10 Hz for 60°/sec is audible in the cockpit of the F-15C/D.

1-174. **In-flight Condition.** At a yaw rate of 30°/sec a signal from the yaw rate gyro is applied to a 30°/sec level detector and a 60°/sec tone control scheduler. The output of the 30°/sec level detector closes transistor switch 1, making 28VDC available to transistor switch number 2. The signal from the 60°/sec tone control scheduler schedules signals to the control oscillator between 30°/sec and 60°/sec. The control oscillator then sends a signal at 2 Hz; for 30°/sec and increases linearly up to 10 Hz for 60°/sec to transistorswitch 2. Transistor switch 2 closes and opens on command from the control oscillator supplying an interrupted 28VDC of 2 Hz to 10 Hz to the integrated communication control panel (ICCP), then to the pilots headset, warning of the yaw rate condition.

1-175. **Ground Test Condition.** Ground testing is made possible by pressing the take off trim switch which supplies 5VDC to a logic inverter. When the logic, inverter is low, transistor switch 3 closes putting a 7VAC signal to a summing amplifier, simulating a yaw rate greater than 30°/sec. The simulated signal is processed the same as the inflight condition producing an audible tone.

1-176. **ANGLE-OF-ATTACK WARNING TONE.** The AOA warning tone is a programmable 900 Hz tone with two 10 Hz. rate pulses. The AOA warning tone is activated when AOA is valid, the AOA cockpit units (CPU's) exceed the AOA tone limit, and the landing gear is up or unknown. On power up, with weight on wheels, the AOA tone limit defaults to OFF or 30 CPU's based on aircraft configuration. The ability to change the default value exists by using the NAV CONTROL INDICATOR (NCI) panel. With CCC selected on the DATA SELECT switch and M2 selected on the DEST DATA switch, enter the AOA limit using the keyboard. The number on the right Display Readout Devices (DRD) on the NCI panel, is used as the AOA tone limit if the number is a whole number between 20 and 50 (inclusive). All other values are illegal and will display all 9's on the right DRD. Numbers between 46 and 50 (inclusive) are accepted but are used to turn off the system. If AOA is greater than 18 CPU's, it will replace the UD)AOA XX). The MACH or TOD display will not return until the AOA has gone below 17 CPU's. If the programmed AOA is exceeded, AOA units will flash on the HUD.

1-177. The aircraft configuration is checked by the PACS every 3.5 seconds after communication is established with the PACS. AOA tone limit defaults to 30 CPU's when any of the below occur:

1. CFTs are reported

2 Air to Ground stores are reported

3. Miscellaneous stores are reported.

4. PACS is not online within 5 seconds after CC power up.

5. OWS 10 Hz validity flag is invalid.

AOA tone limit defaults to off with all other aircraft configurations.

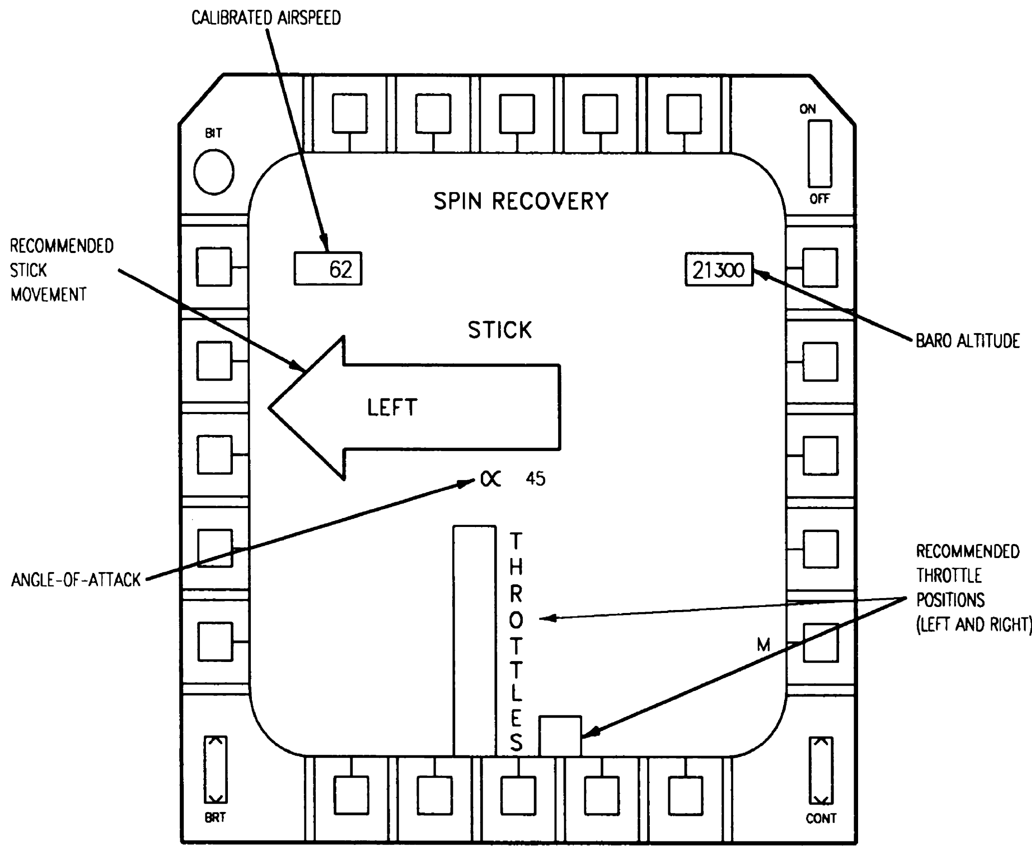
1-178. When a spin is detected by the AFCS, yaw rate warning tone is given priority over OWS, Altitude and AOA tone for use of the 900 Hz tone. The tone priority is listed below:

1. Yaw Rate

2. OWS

3. Altitude

4. AOA (Can be heard with OWS voice warning).



MAJOR CHANGE

Figure 1-20. Spin Recovery Aid Display (SRAD)

SECTION II

SUPPORT EQUIPMENT LIST

2-1. TEST EQUIPMENT.

2-2. To do maintenance on the system or components, the test equipment listed in table 2-1 should be used. Alternate equipment with equal or greater range and accuracy than theequipment listed may be used. The listed test equipment is required for doing organizational maintenance on the AFCS.

Table 2-1. Test Equipment List

Equipment Number	Nomenclature	Use and Application
AN/PSM-37( )	Multimeter	System Analysis
AN/USM-341	Digital Multimeter	System Analysis
DPP-10 0-10 lb.	Gage, dial	Trim Switch Force Checkout
H-133C/AIC	Assembly, Microphone-Headset	Operational Checkout
TTU-205C/E (18910010000)	Test Set, Pressure-Temperature	Dynamic Pressure Sensor Checkout
68D110007-1003	Indicator, Stick Rigging	Roll Trim Measurement Checkout
68D150026-1001	Strap, Ground	Grounding TTU-205C/E Test Set
68D270028-1001	Assembly, Cable Adapter	System Analysis
68D270027	Breakout Box, AFCS	System Analysis
68D270029-1001	Cable Assembly, CAS Shutdown Inhibit	System Analysis
292E800G3 and 29E800G4	Test Set, Automatic Flight Control System AN/ASM-497	Operational Checkout
936E225G( )	In-Flight Monitor, AFCS	Operational Checkout

2-3. SPECIAL TOOLS.

2-4. The special tools listed in table 2-2 are used to do maintenance of Auto Flight system.

Table 2-2. Special Tools List

Tool Number	Nomenclature	Use and Application
MDE321450-1	Adapter Assembly, Torque Wrench, Stick Force Transducer Connector	Stick Force Sensor Installation
68D050026-1001	Kit, Rigging Pin	Yaw Trim Measurement. Checkout
68D210008-1003	Adapters Kit, Pitot-Static	Dynamic Pressure Sensor Checkout
500T72	Torque Tester, Probe	Test AOA Probes

2-5. **AFCS BREAKOUT BOX.**

2-6. AFCS breakout box provides the capability to make voltage and resistance measurements on connectors J4 and J6 of the pitch computer and with cable adapter assembly, PN 68D270028-1001, J4 and J6 of the roll/yaw computer. This capability augments the existing troubleshooting. procedures to allow more positive fault isolation of the stabilator and rudder actuators.

2-7. **IN-FLIGHT MONITOR (IFM).**

2-8. **DESCRIPTION**. The IFM is flyable diagnostic monitoring equipment for a portion of the F-15 flight control system. Monitoring is provided for the AFCS and elements of the mechanical flight control system that interface with it. These, elements are the control surface actuators, feel trim actuators, CASI, and switches that affect flight control performance. Monitoring circuitry is contained in the unit which is an electronic LRU designed and qualified for flight in the F-15 aircraft. The front face of the monitor contains 21 fault indicators. These indicators latch and store in-flight failures which cause CAS disconnects. Postflight diagnosis of fault indicators allows repair of intermittent malfunctions that cannot be duplicated on the ground. In addition, the IFM can be used during ground testing to isolate intermittent malfunctions that the flight line test set fails to detect.

2-9. **Installation and Self Test (22-10-12).** Following installation, during postflight and prior to each IFM flight, a self test should be done to verify IFM integrity. The self test is done by operating switches on the IFM front panel. Switches are to be held momentarily in the specified position and released. A not failed condition is displayed by the indicator showing all black (reset). A failed condition is displayed by the indicator when the center is all white (set).

2-10. **Postflight Procedures (22-10-13).** Postflight procedures are used to record indicators that failed in flight and to certify IFM integrity by doing a self test. A partial IFM disconnect is done to aid AFCS test set use. Following malfunction diagnosis and repair, a partial reconnect and self test is done when additional IFM flights are desired.

2-11. **Removal (22-10-14).** When all IFM flights are completed, the IFM is removed and aircraft forms are documented.

2-12. **Analysis Matrix (22-10-15).** The IFM analysis matrix is a diagnostic table for cross-referencing IFM and AFCS test set data. Probable causes listed in the problem source column are in descending order from most to least probable. CAS axes disengagement from the aircraft forms must be correlated with the matrix. When AFCS test set indications agree with those listed for specific IFM faults, the malfunction is considered solved. When AFCS test set and IFM data disagree and further troubleshooting fails to isolate the IFM fault, probable problem source corrective action should be taken in the order listed. Consideration should be given to making an IFM flight after each problem source replacement.

GLOSSARY

Nonstandard abbreviations and symbols are described below. All abbreviations and symbols used in the maintenance manual set are described in TO SR1F-15C-2-00GV-00-1.

ABBREVIATIONS

Act. - Actuator
ADC - Air Data Computer
ADS - Autopilot/Steering Disengage Switch
AFCS - Automatic Flight Control System
ALT - Altitude
AOA - Angle-of-Attack
ARI - Aileron Rudder Interconnect
ASP - Avionics Status Panel
ATT - Attitude

BIT - Built-in Test

CAS - Control Augmentation System
CASI - CAS Interconnect
CSBPC - Control Stick Boost and Pitch Compensator (consists of PRCA and ARI)
CSS - Control Stick Steering

DPS - Differential Pressure Sensor
DSS - Differential Series Stabilator

EHV - Electrohydraulic Valve

IFM - In-Flight Monitor
INS - Inertial Navigation Set
INU - Inertial Navigation Unit

L - Left
LGH - Landing Gear Handle
LRU - Line Replaceable Unit
LVDT - Linear Variable Differential Transformer

MCV - Master Control Valve
MN - Manual

ND - Nose down
NU - Nose up
NC - Normally closed

POL - Pitch Outer Loop
POSN - Position
PPR - Pitch Pilot Relief
PRC - Pitch Ratio Changer
PRCA - Pitch/Roll Channel Assembly
PTC - Pitch Trim Compensator

qc - Dynamic Pressure

R - Right
RL - Rudder Left
ROL - Roll Outer Loop
RPR - Roll Pilot Relief
RR - Rudder Right

SOV - Shutoff ValveHUD - Head Up Display
SRAD - Spin Recovery Aid Display

TE - Trailing Edge
T/O - Takeoff
TOT - Takeoff Trim

VS1 - Vertical Speed Indicator

WOW - Weight-On-Wheels

DEFINITIONS

LEVEL CHANGER - A device to change 28VDC to 5VDC logic or vice versa.

LEVEL DETECTOR - A variable gain device which provides an output only if indicated conditions are satisfied.

SYMBOLS

$\overline{\text{ALT ENGAGE}}$ In diagrams, line over words indicates logic is a log 1 (high) when signal does not exist. (Alt not engaged).