

SM2A-03-BLOCK II-(1)
APOLLO OPERATIONS HANDBOOK

SYSTEMS DATA

SECTION 2

SUBSECTION 2.3

STABILIZATION AND CONTROL SYSTEM (SCS)
(SC 106 AND SUBS UNLESS OTHERWISE NOTED)

SCS

2.3.1 INTRODUCTION.

The stabilization and control subsystem (SCS) provides a capability for controlling rotation, translation, SPS thrust vector, and displays necessary for man in the loop control functions.

The SCS is divided into three basic subsystems: attitude reference, attitude control, and thrust vector control. These subsystems contain the elements which provide selectable functions for display, automatic and manual attitude control, and thrust vector control. All control functions are a backup to the primary guidance navigation and control subsystem (PGNCS). The SCS provides two assemblies for interface with the propulsion subsystem; these are common to SCS and PGNCS for all control functions. The main display and controls panel contains the switches used in selecting the desired display and control configurations.

The SCS interfaces with the following spacecraft subsystems:

- Telecommunications Subsystem—Receives all down-link telemetering from SCS.
- Electrical Power Subsystem—Provides primary power for SCS operation.
- Environmental Control Subsystem—Transfers heat from SCS electronics.
- Sequential Events Control Subsystem—Provides abort switching and separation enabling of SCS reaction control drivers and receives manual abort switch closure from the SCS.
- Orbital Rate Drive Electronics for Apollo and LM—Interfaces with the pitch axis of the FDAI ball to give a local vertical referenced display.
- Guidance Navigation and Control Subsystem:

Provides roll, pitch, and yaw total attitude and attitude error inputs for display.

Provides RCS on-off commands to the SCS interface assembly for attitude control.

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Provides TVC servo commands to the SCS interface assembly for automatic thrust vector control

Provides automatic SPS on-off command to SCS interface assembly for ΔV control

Receives switch closure signals from the SCS translation and rotation controls.

- Entry monitor subsystem: the EMS provides SPS enabling/disabling discretes to the SCS thrust on-off logic for the SPS.

- Propulsion subsystem:

The service propulsion subsystem receives thrust vector direction commands and thrust on-off commands from the SCS that can originate in the PGNCS or the SCS.

The reaction control subsystem receives thrust on-off commands from the SCS that can originate in the PGNCS or the SCS.

Detailed descriptions of the SCS hardware, attitude reference subsystem, attitude control subsystem, and thrust vector control subsystem are contained in the following paragraphs.

2.3.2 CONTROLS, SENSORS, AND DISPLAYS.

As an introduction to the stabilization and control system (SCS) a brief description is given of the hardware comprising one complete system. A more detailed discussion follows for the hand controls, displays, and gyro assemblies. The configurations within the SCS resulting from panel 1 switch positions are also presented.

2.3.2.1 SCS Hardware.

The function of the SCS hardware shown in figure 2.3-1 is as follows:

- Electronic Control Assembly (ECA) - Contains the circuit elements required for summing, shaping, and switching of the rate and attitude error signals and manual input signals necessary for stabilization and control of the thrust vector and the spacecraft attitude.
- Reaction Jet and Engine ON-OFF Control (RJ/EC) - Contains the solenoid drivers and logic circuits necessary to control both the RCS automatic solenoid coils and SPS solenoid control valves.
- Electronic Display Assembly (EDA) - Provides the interface between the signal sources to be displayed and the FDAIs and GPI. The EDA also provides signal conditioning for telemetry of display signals.

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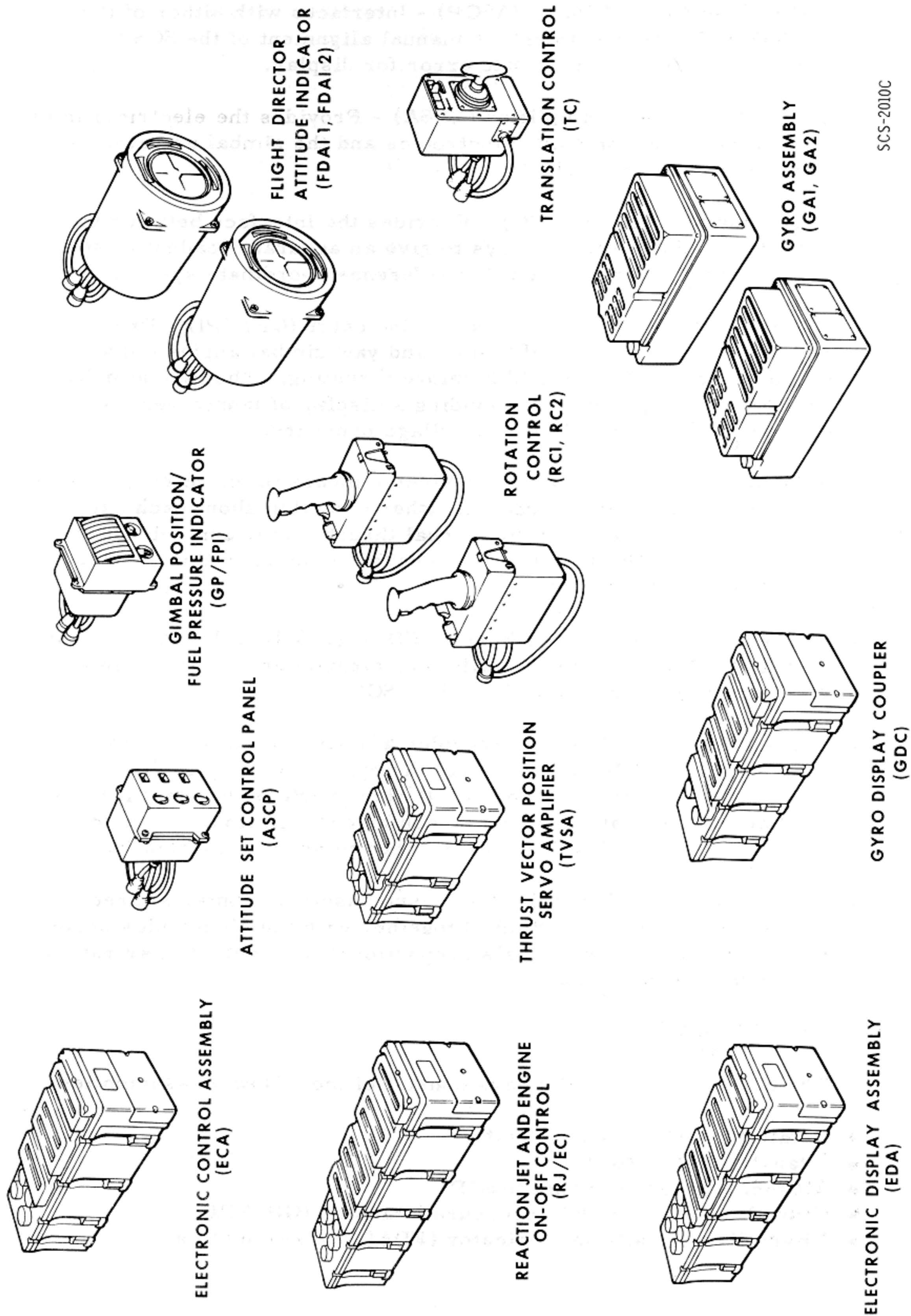


Figure 2.3-1. SCS Flight Hardware



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- Attitude Set Control Panel (ASCP) - Interfaces with either of the total attitude sources to enable manual alignment of the SCS total attitude. Provides an attitude error for display.
- Thrust Vector Servoamplifier (TVSA) - Provides the electrical interface between the command electronics and the gimbal actuator for positioning the SPS engine.
- Gyro Display Coupler (GDC) - Provides the interface between the body rate sensors and displays to give an accurate readout of spacecraft attitude relative to a given reference coordinate system.
- Gimbal Position and Fuel Pressure Indicator (GP/FPI) - Provides a redundant display of the SPS pitch and yaw gimbal angles and a means of manually trimming the SPS before thrusting. The indicator has the alternate capability of providing a display of launch vehicle (S-II and S-IVB) propellant tank ullage pressures.
- Rotation Controls (RC) (2) - Provides a means of exercising manual control of spacecraft rotation in either direction about each axis. Also the RC may be used for manual thrust vector control. It provides the capability to control spacecraft communications with a push-to-talk trigger switch.
- Flight Director Attitude Indicator (FDAI) (2 Only) - Provides to the crew a display of spacecraft attitude, attitude error, and angular rate information from the PGNCs or SCS.
- Translation Control (TC) - Provides a means of exercising manual control over rectilinear motion of the spacecraft in both directions along the three spacecraft axes. It also provides the capability for manual abort initiation during launch by ccw rotation. Transfer of SC control from PGNCs to SCS is accomplished by cw rotation.
- Gyro Assemblies (GA) (2) - Each gyro assembly contains three body-mounted attitude gyros (BMAG) together with the electronics necessary to provide output signals proportional to either angular rate or to angular displacement.

2.3.2.2 Controls and Displays.

The SCS controls and displays consist of the following assemblies:

- Rotation control (RC) - 2 units
- Translation control (TC)
- Attitude set control panel (ASCP)
- Gimbal position and fuel pressure indicator (GP/FPI)
- Flight director attitude indicator (FDAI) - 2 assemblies

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2.3.2.2.1 Rotation Control.

Two identical rotation controls (RCs) are provided. The controls are connected in parallel so that they operate in a redundant fashion without switching. Pitch commands are commanded about a palm-centered axis, yaw commands about the grip longitudinal axis, while roll commands result from a left-right motion (figure 2.3-2). Within the RC there are three command sources per axis:

a. Breakout Switches (\pm BO) - A switch closure occurs whenever the RC is moved 1.5 degrees from its null position. Separate switches are provided in each axis and for each direction of rotation. These six breakout switches are used to provide: command signals to the command module computer (CMC), SCS minimum impulses, acceleration commands, BMAG cage signals, and proportional rate command enabling.

b. Transducers - Transducers produce a-c signals proportional to the rotation control displacement from the null position. These signals are used to command spacecraft rotation rates during SCS proportional rate control and to command SPS engine gimbal position during manual thrust vector control (MTVC). One, two, or all three transducers can be used simultaneously, generating corresponding command signals.

c. Direct Switches - Redundant direct switches will close whenever the control is moved a nominal 11 degrees from its null position (hard-stops limit control movement to \pm 11.5 degrees from null in all axes). Separate switches are provided in each axis and for each direction of rotation. Direct switch closure will produce acceleration commands through the RCS direct solenoids.

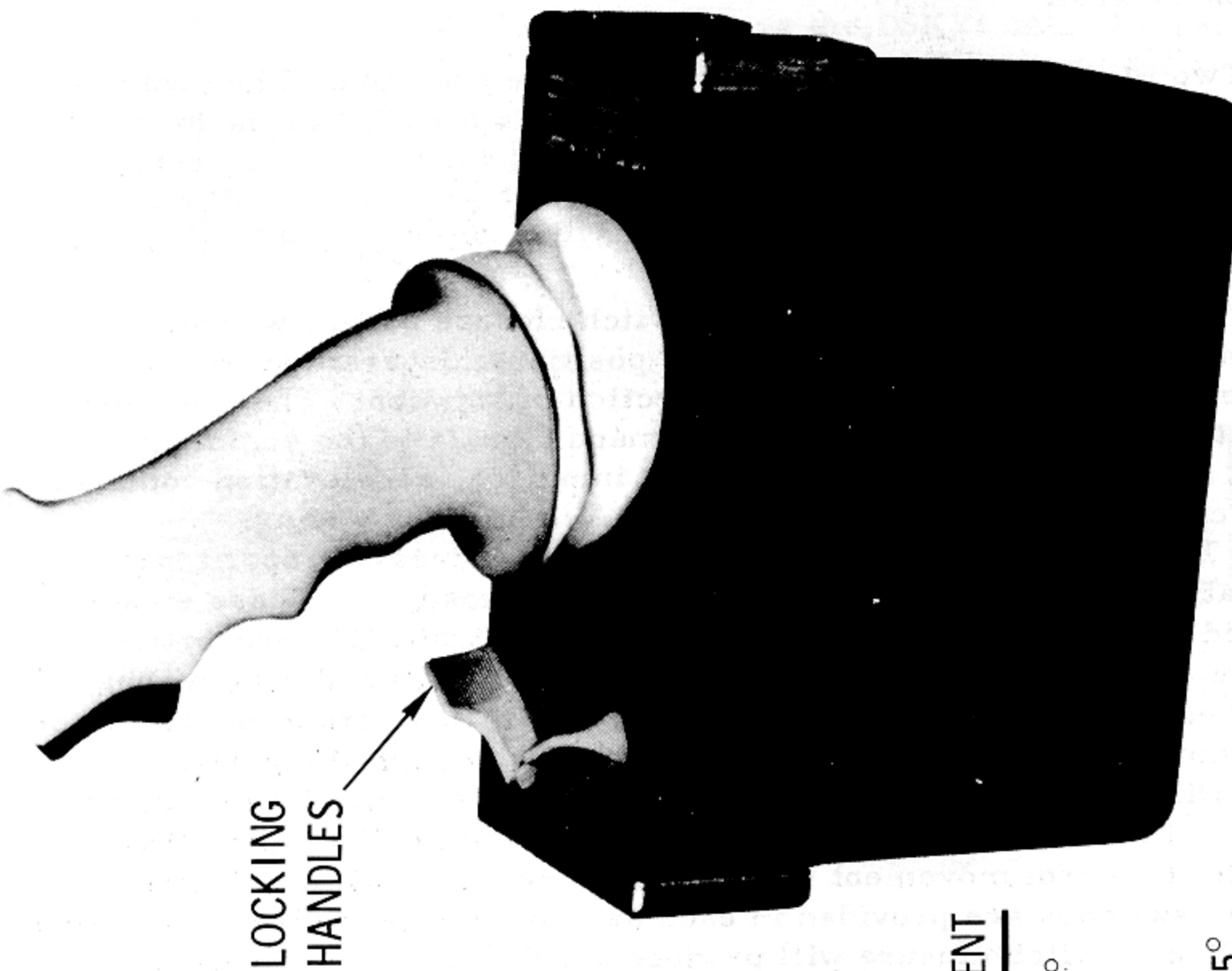
The rotation control is provided with a tapered female dovetail on each end of the housing. This dovetail mates with mounting brackets on the couch armrests. When attached to the armrests, the input axes are approximately parallel with spacecraft body axes. Figure 2.3-12 illustrates control motions about its axis and the resulting commands to the RCS, PGNCS, or SCS. A trigger-type push-to-talk switch is also located in the control grip. Redundant locking devices are provided on each control.

2.3.2.2.2 Translation Control.

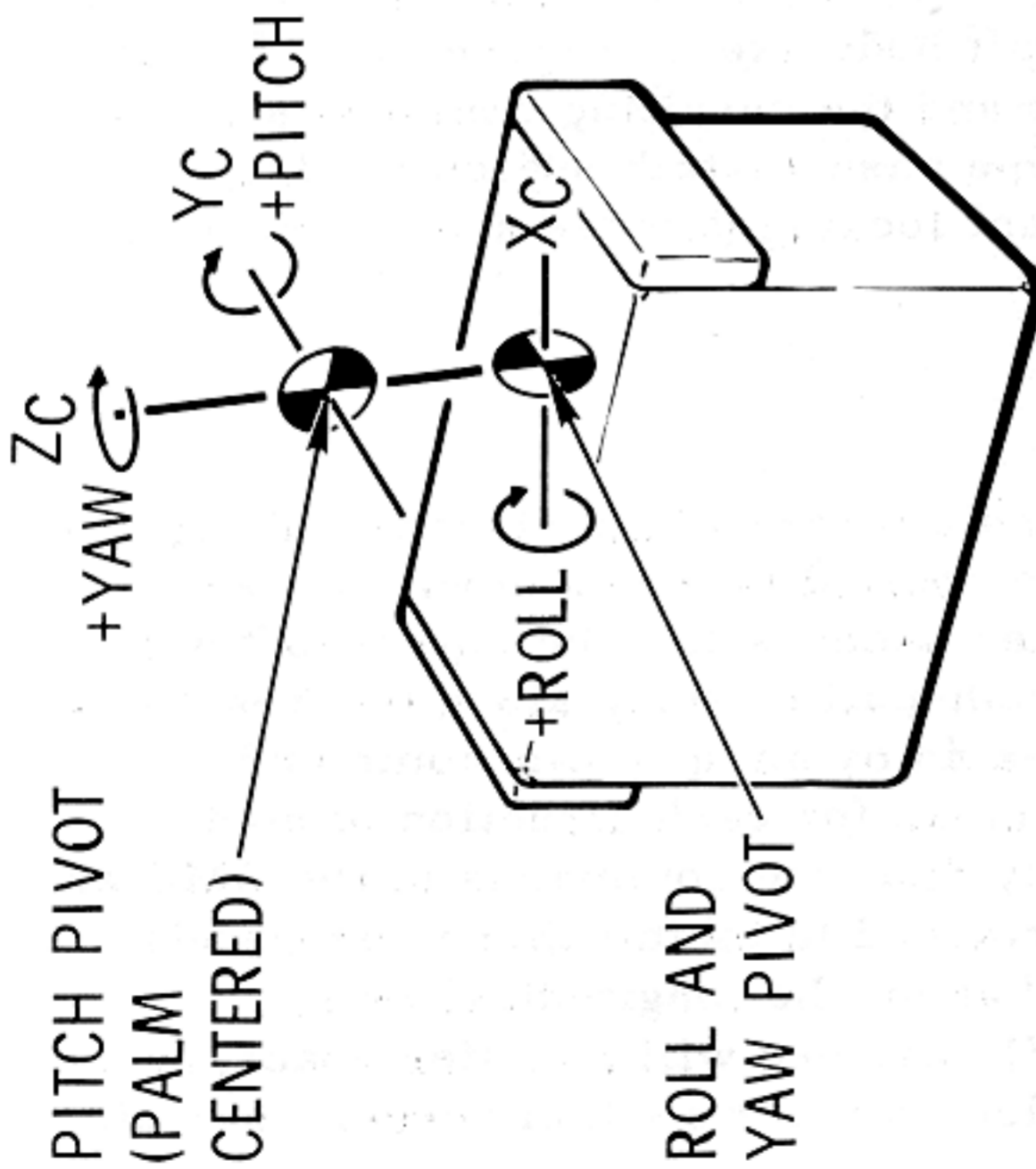
The translation control provides a means of accelerating along one or more of the spacecraft axes. The control is mounted with its axes approximately parallel to those of the spacecraft. The spacecraft will accelerate along the X-axis with a push-pull motion, along the Y-axis by a left-right motion, and along the Z-axis by an up-down command (figure 2.3-3). Redundant switches close for each direction of control displacement. These switches supply discrete commands to the CMC and the RJ/EC. A mechanical lock is provided to inhibit these commands. In addition the T-handle may be rotated about the longitudinal axis:

a. The redundant clockwise (CW) switches will transfer spacecraft control from CMC to SCS. It may also transfer control between certain submodes within the SCS.

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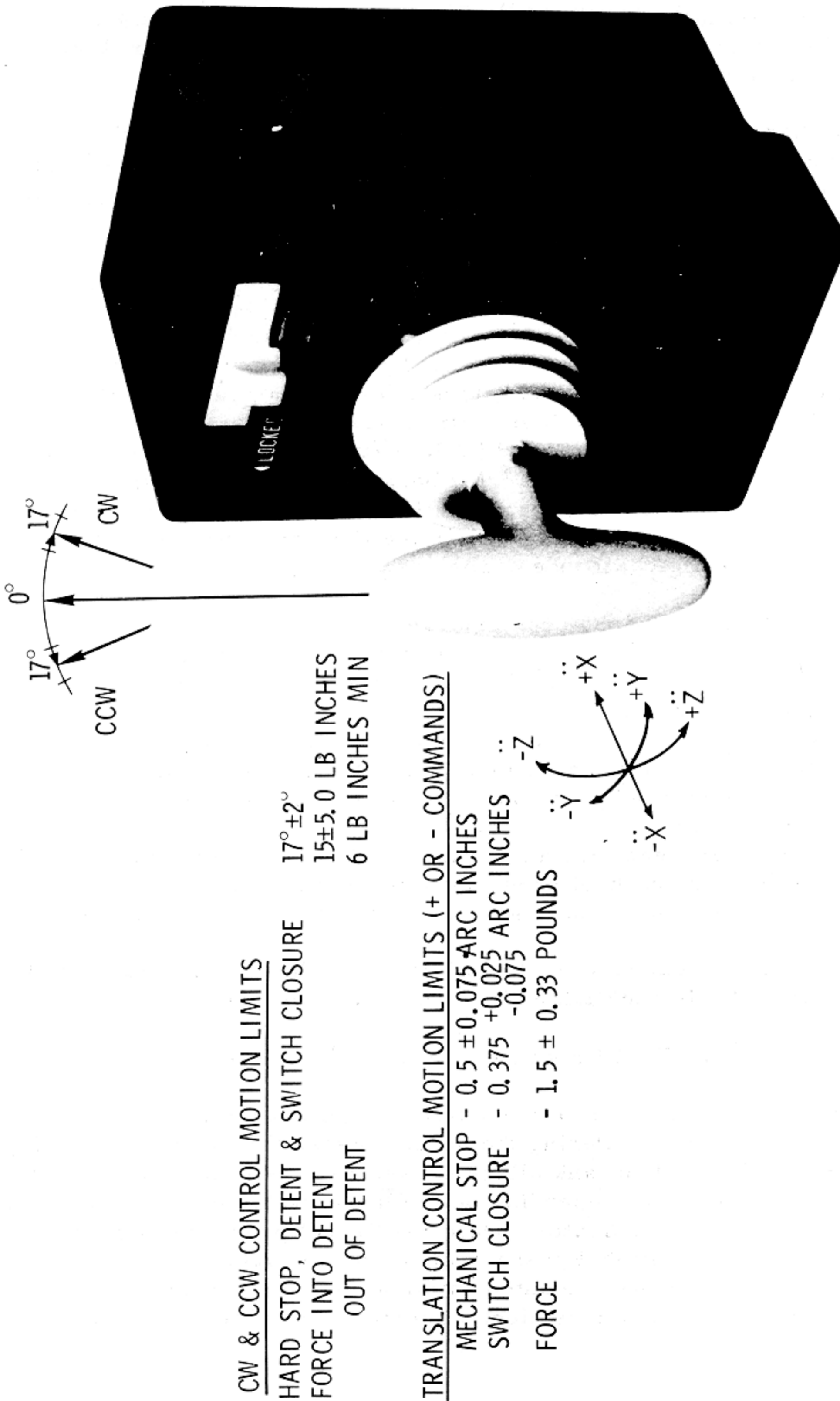
SCS-2002D
 CSM LOGISTICS TRAINING



LOCKING HANDLES
 CONTROLLER LOCK TO ARM BY 50° DISPLACEMENT

ROTATION CONTROL PARAMETERS	DISPLACEMENT
BREAKOUT SWITCH ACTUATION	$1.5 \pm 0.5^\circ$
SOFT STOP	$10 \pm 1^\circ$
DIRECT SWITCH ACTUATION	11°
HARD STOP	$11.5 \pm 0.5^\circ$

Figure 2.3-2. Rotation Control



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Figure 2.3-3. Translation Control

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b. The redundant counterclockwise (CCW) switches provide for a manual abort initiation during the launch phase. A discrete signal from switch closure is fed to the master events sequence controller (MESC) which initiates other abort functions.

Neither the CW or CCW functions are inhibited by the locking switch on the front of the controller. The T-handle will remain in the CW or CCW detent position without being held, once it is rotated past approximately plus or minus 12 degrees.

2.3.2.2.3 Attitude Set Control Panel (ASCP).

The ASCP (figure 2.3-4) provides, through thumbwheels, a means of positioning differential resolvers for each of the three axes. The resolvers are mechanically linked with indicators to provide a readout of the dialed angles. The input signals to these attitude set resolvers are from either the IMU or the GDC. The inertial (Euler) attitude error output signals are sine functions of the difference angles between the desired attitude, set by the thumbwheels, and the input attitude from the GDC or IMU. The GDC Euler output can be used to either align the GDC or to provide fly-to indications on the FDAI attitude error needles.

Characteristics of the counters are:

- a. Indicates resolver angles in degrees from electrical zero, and allows continuous rotation from 000 through 359 to 000 without reversing the direction of rotation.
- b. Graduation marks every 0.2 degree on the units digit.
- c. Pitch and roll are marked continuously between 0 and 359.8 degrees. Yaw is marked continuously from 0 to 90 degrees and from 270 to 359.8 degrees.
- d. Readings increase for an upward rotation of the thumbwheels. One revolution of the thumbwheel produces a 20-degree change in the resolver angle and a corresponding 20-degree change in the counter reading.

The counter readouts are floodlighted and the nomenclature (ROLL, PITCH, and YAW) is backlighted by electroluminescent lighting.

2.3.2.2.4 Gimbal Position and Fuel Pressure Indicator (GP/FPI).

The GP/FPI (figure 2.3-5) contains redundant indicators for both the pitch and yaw channels. During the boost phases, the indicators display S-II and S-IVB propellant tank ullage pressures. S-II fuel pressure (or S-IVB oxidizer pressure depending on the launch vehicle configuration) is on the redundant pitch indicators while S-IVB fuel pressure is on the two yaw indicators. The gimbal position indicator consists of two dual servo-metric meter movements, mounted within a common hermetically sealed case. Scale illumination uses electroluminescent lighting panels.

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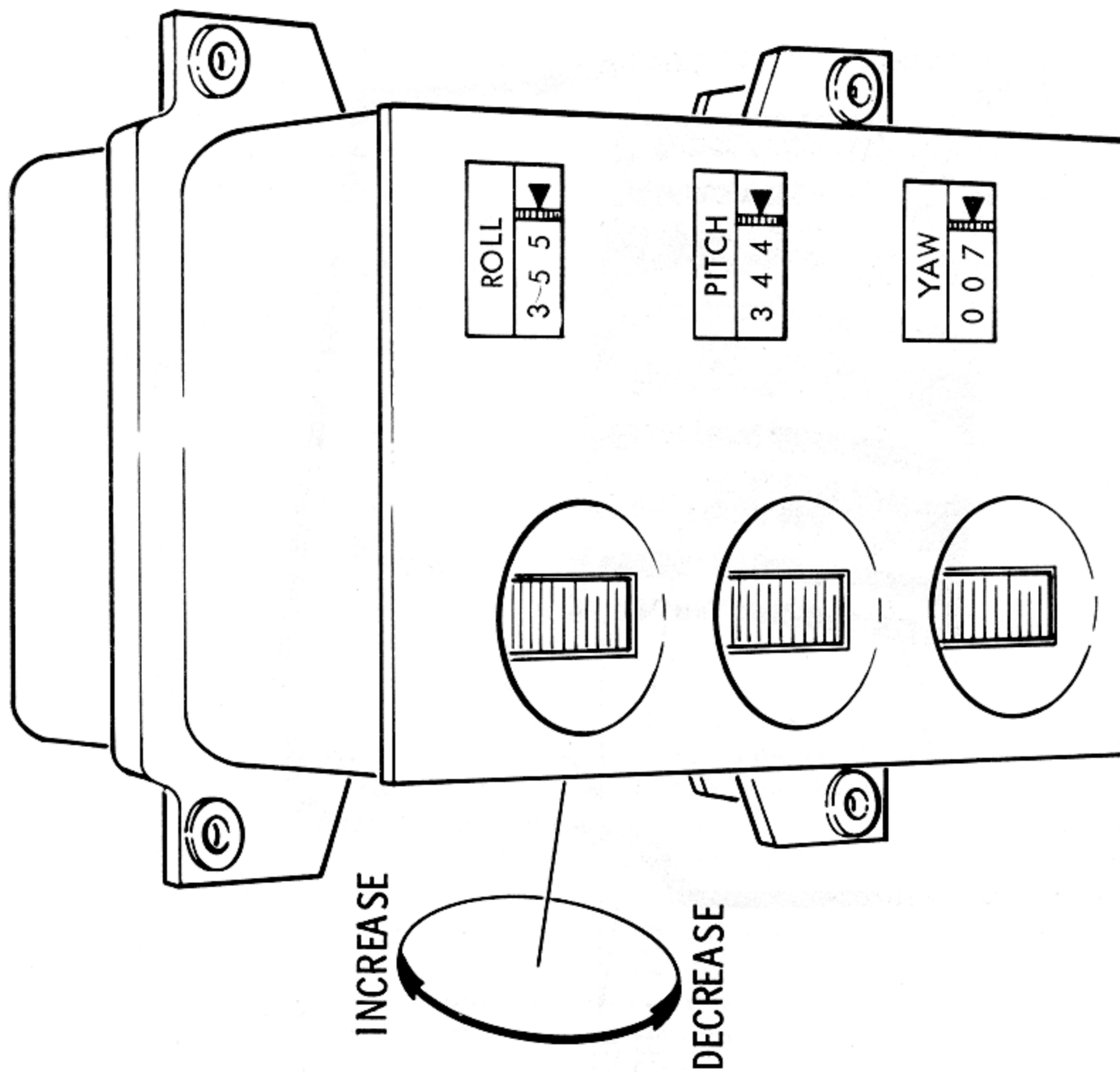


Figure 2.3-4. Attitude Set Control Panel

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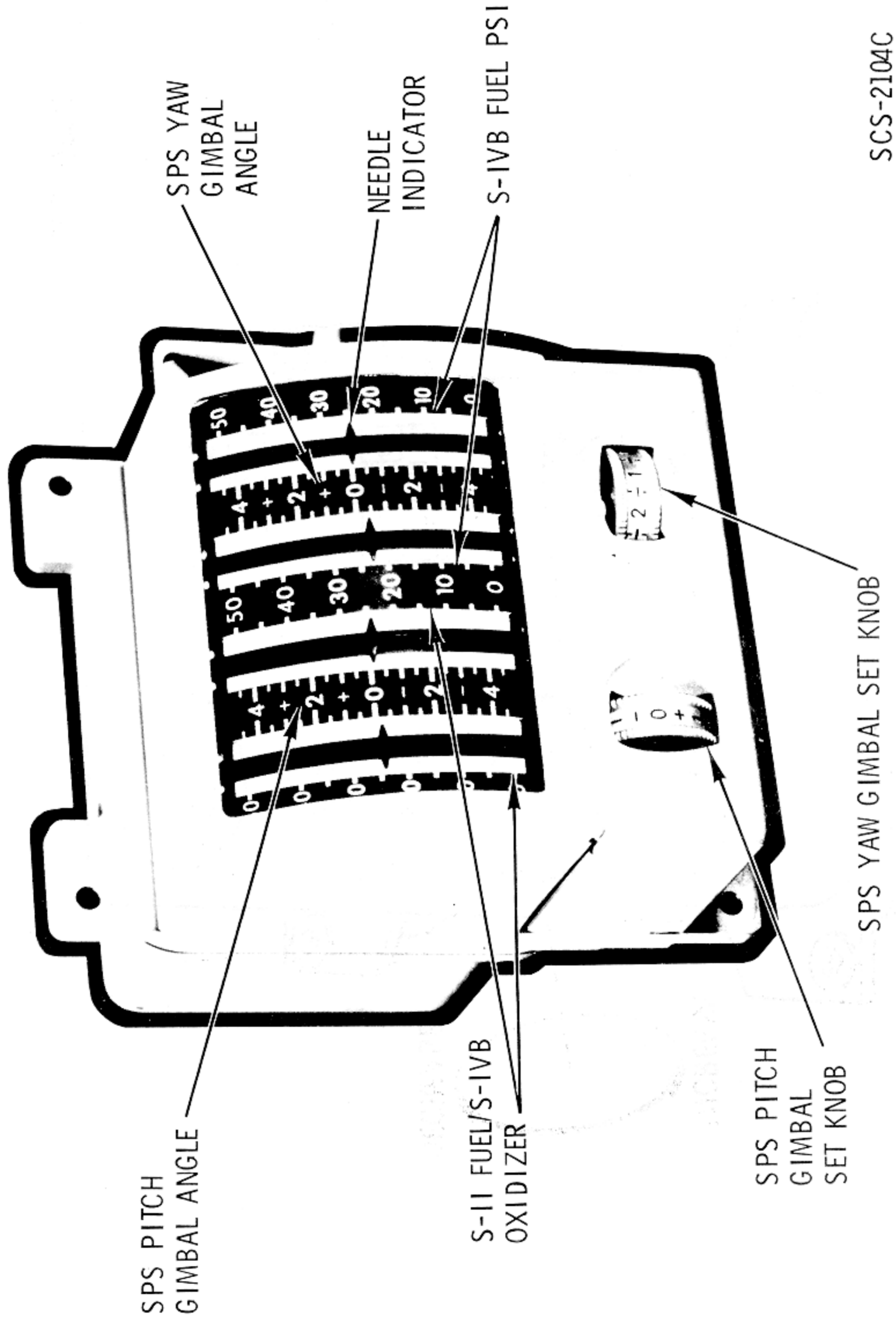


Figure 2.3-5. Gimbal Position Indicator

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For an SCS delta V mode, manual SPS engine gimbal trim capability is provided. Desired gimbal trim angles are set in with the pitch and yaw trim thumbwheels. The indicator displays SPS engine position relative to actuator null and not body axes. The range of the engine pitch and yaw gimbal displays are ± 4.5 degrees. This range is graduated with marks at each 0.5 degree and reference numeral at each 2-degree division. The range of the fuel pressure scale is 0 to 50 psi with graduations at each 5-psi division, and reference numerals at each 10-psi division. A functional description of the GPI display circuitry which shows the redundancy is in paragraph 2.3.5.3.

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2.3.2.2.5 Flight Director Attitude Indicator (FDAI).

The FDAIs provide displays to the crew of angular velocity (rate), attitude error, and total attitude (figure 2.3-6). The body rate (roll, yaw, or pitch) displayed on either or both FDAIs is derived from the BMAGs in either gyro assembly 1 or 2. Positive angular rates are indicated by a downward displacement of the pitch rate needle and by leftward displacement of the yaw and roll rate needles. The angular rate displacements are "fly-to" indications as related to rotation control direction of motion required to reduce the indicated rates to zero. The angular rate scales are marked with graduations at null and \pm full range, and at $\pm 1/5$, $\pm 2/5$, $\pm 3/5$, and $\pm 4/5$ of full range. Full-scale deflection ranges are obtained with the FDAI SCALE switch and are:

- Pitch rate: ± 1 deg per sec, ± 5 deg per sec, ± 10 deg per sec
- Yaw rate: ± 1 deg per sec, ± 5 deg per sec, ± 10 deg per sec
- Roll rate: ± 1 deg per sec, ± 5 deg per sec, ± 50 deg per sec

Servometric meter movements are used for the three rate indicator needles.

The FDAI attitude error needles indicate the difference between the actual and desired spacecraft attitude. The attitude error signal can be derived from several sources: The uncaged BMAGs from GA-1, the CDUs (PGNCS), or the ASCP-GDC/IMU (figure 2.3-10). Positive attitude error is indicated by a downward displacement of the pitch error needle, and by a leftward displacement of the yaw and roll error needles. The attitude error needle displacements are "fly-to" indications as related to rotation control direction of motion, required to reduce the error to zero. The ranges of the error needles are ± 5 degrees or ± 50 degrees for full-scale roll error, and ± 5 degrees or ± 15 degrees for pitch and yaw error. The error scale factors are selected by the FDAI SCALE switch that also establishes the rate scales. The pitch and yaw attitude error scales contain graduation marks at null and \pm full scale, and at $\pm 1/3$ and $\pm 2/3$ of full scale. The roll attitude scale contains marks at null, $\pm 1/2$, and \pm full scale. The attitude error indicators utilize servometric meter movements.

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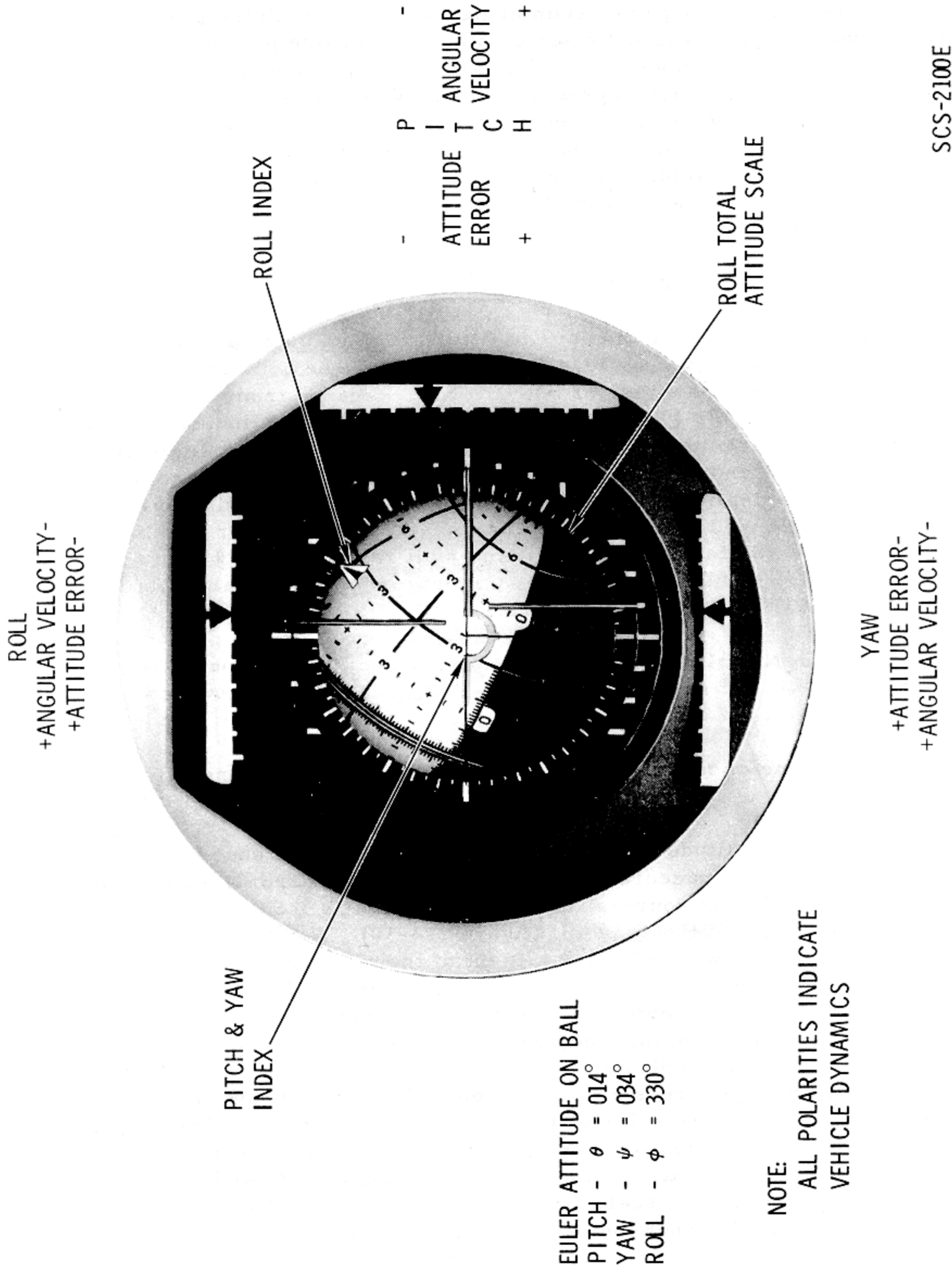


Figure 2.3-6. Flight Director Attitude Indicator

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Spacecraft orientation, with respect to a selected inertial reference frame, is also displayed on the FDAI ball. This display contains three servo control loops that are used to rotate the ball about three independent axes. These axes correspond to inertial pitch, yaw, and roll. The control loops can accept inputs from either the IMU gimbal resolvers or the GDC resolvers. Selecting the source is covered in paragraph 2.3.2.3.

The control loops are proportional servos; therefore, the angles of rotation of the ball must correspond to the resolver angles of the source. The FDAI, illustrated in figure 2.3-6, has the following markings:

- a. Pitch attitude is represented on the ball by great semicircles. The semicircle (as interpolated), displayed under the FDAI inverted wing symbol, is the inertial pitch at the time of readout. The two semicircles that make up a great circle correspond to pitch attitudes of θ and $\theta+180$ degrees.
- b. Yaw attitude is represented by minor circles. The display readout is similar to the pitch readout. Yaw attitude circles are restricted to the intervals - 270 to 360 degrees (0°) and 0 (360°) to 90 degrees.
- c. Roll attitude is the angle between the wing symbol and the pitch attitude circle. The roll attitude is more accurately displayed on a scale attached to the FDAI mounting, under a pointer attached to the roll (ball) axis.
- d. The last digits of the circle markings are omitted. Thus, for example, 3 corresponds to 30, and 33 corresponds to 330.
- e. The ball is symmetrically marked (increment wise) about the 0-degree yaw and 0/180-degree pitch circles. The following comments provide clarification for areas of the ball not shown in figure 2.3-6.
 1. Marks at 1-degree increments are provided along the entire yaw 0-degree circle.
 2. The pitch 180-degree semicircles has the same marking increments as the 0-degree semicircle.
 3. Numerals along the 300- and 60-degree yaw circles are spaced 60-pitch degrees apart. Note that numerals along the 30-degree yaw circle are spaced 30-pitch degrees apart.
- f. The red areas of the ball, indicating gimbal lock, are defined by $270 < \text{yaw} < 285$ degrees and $75 < \text{yaw} < 90$ degrees.

2.3.2.3 Functional Switching Concept.

The Block II SCS utilizes functional switching concepts as opposed to "mode select" switching mechanized in the Block I system.

Functional switching requires manual switching of numerous independent panel switches in order to configure the SCS for various mission functions (e.g., midcourse, ΔV s, entry, etc.). Mode switching would, for example, employ one switch labeled "midcourse" to automatically accomplish all the necessary system gain changes, etc., for that mission phase. Thus mode selection simplifies the crew tasks involved, but limits system flexibility between various mode configurations.

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Function select switching, on the other hand, requires more crew tasks, but offers flexibility to select various gains, display scale factors, etc., as independent system capabilities. Function select switching also allows flexibility to "switch out" part of a failed signal path without affecting the total signal source (e.g., SCS in control of the vehicle with GN displays still presented to the crew).

2.3.2.3.1 Display Switching Interfaces.

The FDAI switches determine the source of display data, the FDAI selected, and the full-scale deflections of the attitude error and rate needles. The source of rate information for display will always be from BMAG 2 unless BMAG 1 is put into a backup rate configuration. Other switches also modify the data displayed and these will be pointed out as they are discussed. Both FDAIs are also assumed to be properly energized from the power switching panel.

2.3.2.3.2 Spacecraft Control Switching Interfaces.

There are two sources of vehicle controls selectable from the SC main display console: SCS or CMC. CMC is the primary method of control and the SCS provides backup control. The vehicle attitude control is obtained from the reaction control engines and the thrust vector control from the service propulsion engine.

2.3.3 ATTITUDE REFERENCE SUBSYSTEM. (Figure 2.3-7)

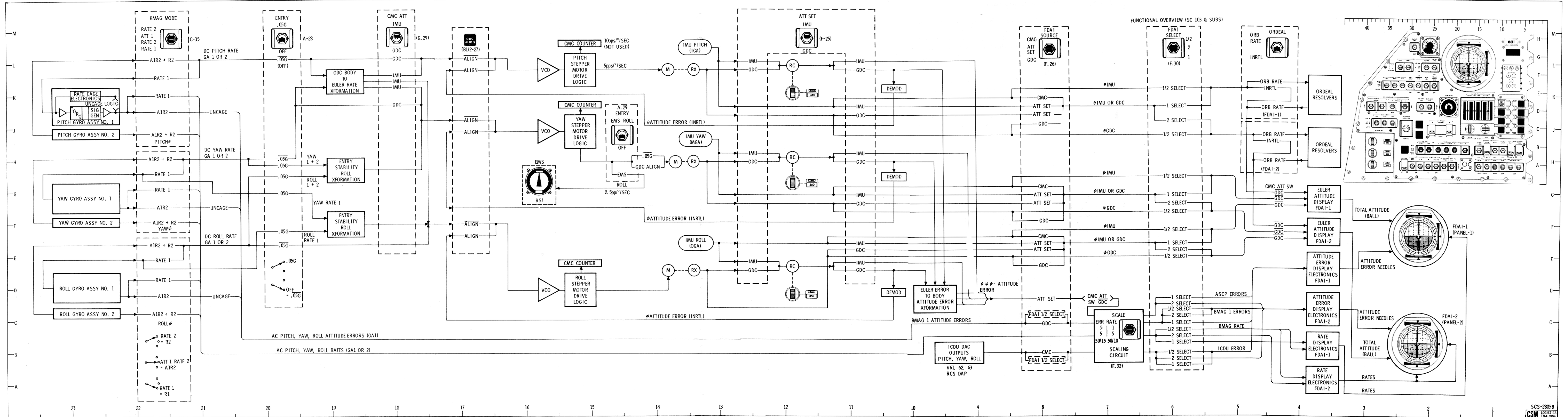
2.3.3.1 Gyro Display Coupler (GDC).

The purpose of the GDC is to provide a backup attitude reference system for accurately displaying the spacecraft position relative to a given set of reference axes. Spacecraft attitude errors can be displayed on an FDAI using the ASCP-GDC difference. This error signal provides a means of aligning the attitude reference system to a fixed reference while monitoring the alignment process on the error needles; or it could be used in conjunction with manual maneuvering of the spacecraft with the error needles representing fly-to-commands.

The GDC can be configured for the following configurations:

- GDC align - Provides a means of aligning the GDC to a given reference.
- Euler - Computes total inertial attitude from body rate signal inputs.
- Non-Euler - Converts analog body rate signals to digital body rate pulses.
- Entry (.05 G) - Provides redundant outputs of attitude changes with respect to the roll stability axis.

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Figure 2.3-7. SCS Attitude Reference Overview (Sheet 1 of 2)

Figure 2.3-7. SCS Attitude Reference Overview (Sheet 2 of 2)

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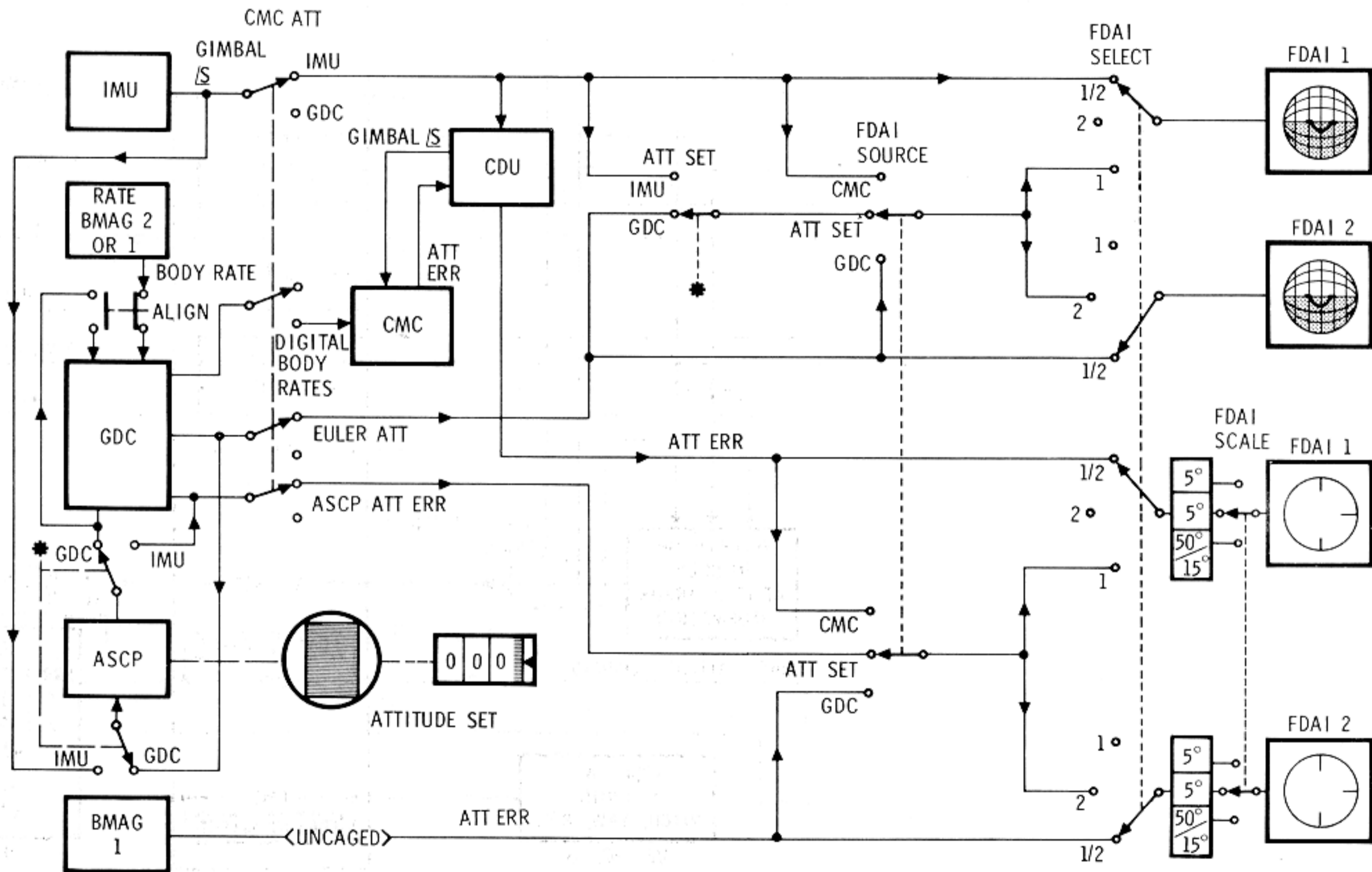
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2.3.3.2 GDC Configurations.

Panel switch positions necessary to obtain each particular GDC function are discussed below.

a. The GDC align mode is used when aligning the GDC Euler angles (shafts) to the desired inertial reference selected by the ASCP thumbwheels (resolvers). This is done by interfacing the GDC resolvers with the ASCP resolvers (per axis) to generate error signals which are proportional to the sine of the difference between the resolver angles. (See figure 2.3-8.) When the GDC ALIGN switch is pressed, these error signals are fed back to the GDC input to drive the GDC/ASCP resolver angular difference to zero. During the align operation all other inputs and functions for the GDC are inhibited. When the EMS ROLL switch is up and the GDC ALIGN switch is pressed, the RSI pointer rotates (open loop) in response to yaw ASCP thumbwheel rotations.



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Figure 2.3-8. FDAI Attitude Select Logic

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b. In the Euler configuration, the GDC accepts pitch, yaw, and roll d-c body rate signals from either gyro assembly and transforms them to Euler angles to be displayed on either FDAI ball. The GDC Euler angles also interface with the attitude set control panel (ASCP) to provide Euler angular errors, which are transformed to body angular errors for display on either FDAI attitude error indicators.

c. With the CMC ATT switch in the GDC position, pitch, yaw, and roll d-c body rate signals from either gyro assembly are converted to digital body rate signals and sent to the G&N command module computer. Power is not only removed from both FDAI ball-drive circuits when this configuration is selected, but ASCP-generated errors are also removed.

d. In the entry mode (≥ 0.05 G), the GDC accepts yaw and roll d-c rate signals from:

1. Either gyro assembly, and computes roll attitude with respect to the stability axis to drive the RSI on the entry monitor system.
2. Gyro assembly 1, and computes roll attitude with respect to the stability axis to drive either FDAI 1 or FDAI 2 in roll only.

2.3.3.3 FDAI Display Sources.

The two FDAIs display total attitude and attitude errors that may originate within the SCS or PGNCS. They also display angular rate from the SCS. The flight crew establishes the FDAI sources by panel switch selection. (See figures 2.3-9 and 2.3-10.)

2.3.3.3.1 Total Attitude and Error Display Sources.

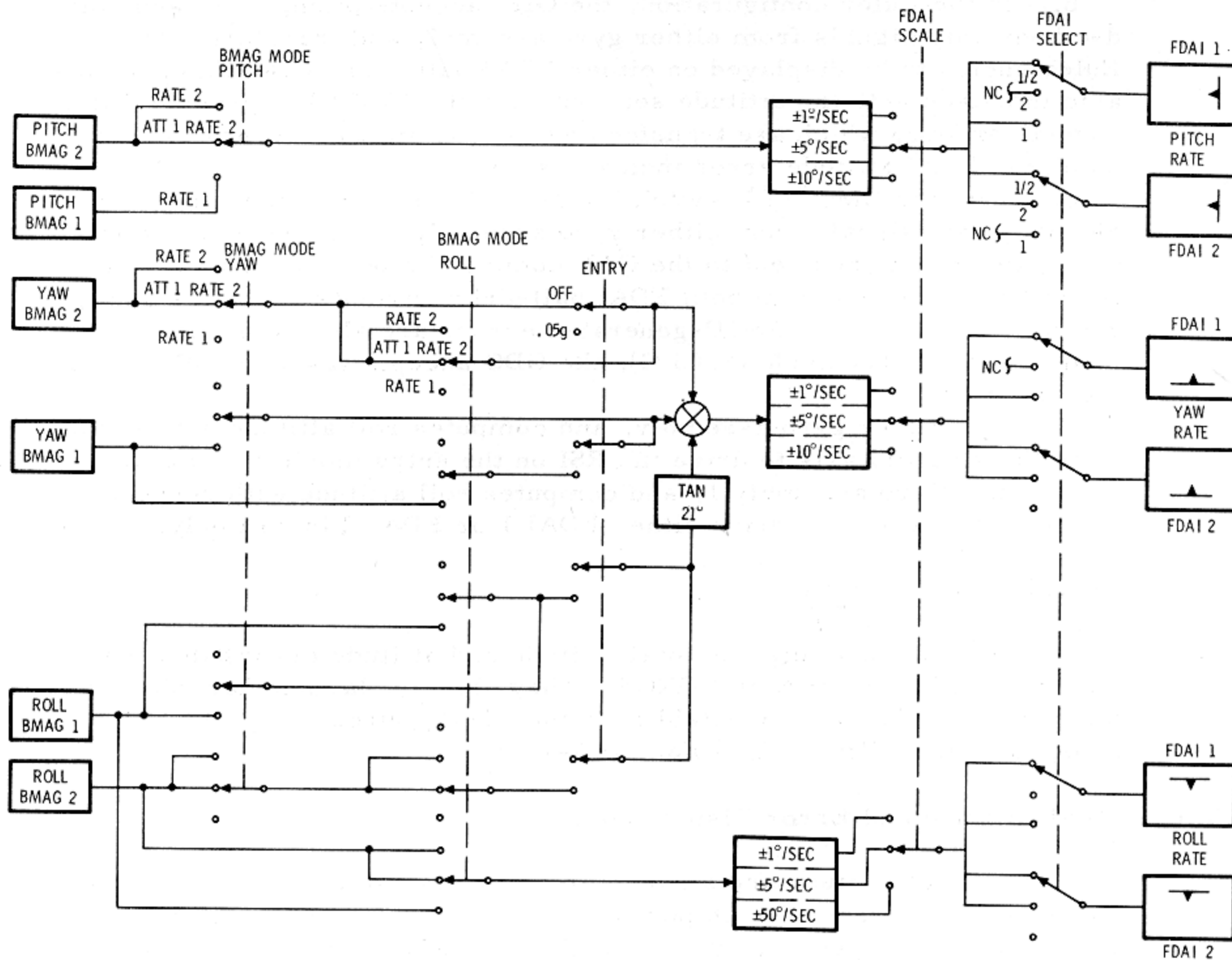
The total attitude and attitude error display selections result from combinations of panel switch positions (figure 2.3-8). When both FDAIs are selected, the platform gimbal angles will always be displayed on FDAI 1 while GDC Euler angles will be displayed on FDAI 2. In order to select the source of attitude display to a particular FDAI, that FDAI and source (G&N or SCS) must be selected (figure 2.3-10). The other FDAI will be inactive. It should be noted that any time total attitude is to be displayed on either FDAI, the CMC ATT switch must be in the IMU position.

The FDAI attitude display may be modified by a NASA-supplied Orbital Rate Display-Earth and Lunar (ORDEAL) unit. The ORDEAL unit is inserted electrically in the pitch channel between the electronic display assembly and FDAI to provide a local vertical display in the pitch axis of either (or both) FDAIs. Controls on the unit permit selection of earth or lunar orbits and orbital altitude adjustment.

The FDAI attitude error display source can be either the SCS or the G&N, with two sources per system. The attitude error sources are as follows:

- a. The BMAG 1 error display is an indication of gimbal precession about its null point, assuming the gyro is uncaged, and may only be

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Figure 2.3-9. FDAI Rate Select Logic

displayed when the SOURCE switch is in the GDC position or when the FDAI SELECT switch is in the 1/2 position.

b. Euler angles from the GDC interface with the ASCP to provide an Euler angle error (GDC-attitude set difference signal) which is then transformed to body angle errors for display on either FDAI. This display source facilitates manual maneuvering of the spacecraft to a new inertial attitude that was dialed in on the attitude set thumbwheels.

c. Inertial gimbal angles from the IMU interface with the ASCP to generate inertial error (IMU-attitude set difference signal) which may be displayed on either FDAI. Thus, if the error needles were nulled using the thumbwheels on the ASCP, the ASCP indicators would then indicate the same inertial reference as the platform.

d. The CMC generates attitude errors that are a function of the program. These will be displayed when the SOURCE switch is in the CMC position, or when the FDAI SELECT switch is in the 1/2 position.

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		ARS CONFIGURATIONS					TOTAL ATTITUDE DISP SOURCES			ERROR DISP SOURCES (4)			RATE DISP SOURCES		
		GDC ALIGN	EULER	NON-EULER	ENTRY 2.05G	IMU	GDC (2)	BMAG #1 (3)	GDC-ATT SET DIFF	IMU-ATT SET DIFF	CDU	BMAG #2	BMAG #1		
B M	PITCH	RATE 2												✓	
		ATT 1/ RATE 2						✓						✓	
		RATE 1													✓
A G M	YAW	RATE 2												✓	
		ATT 1/ RATE 2						✓						✓	
		RATE 1													✓
O D E	ROLL	RATE 2												✓	
		ATT 1/ RATE 2						✓						✓	
		RATE 1													✓
E N T R Y	.05G	.05G			✓		✓ (2)								
		OFF	✓	✓				✓	✓					✓ (4)	✓ (4)
	EMS ROLL	EMS ROLL					✓ (2)								
		OFF													
	FDAI SELECT	(1) 1/2				✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
		2				✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
		1				✓	✓	✓	✓	✓	✓	✓	✓	✓	✓
	FDAI SOURCE	CMC				✓								✓	
		ATT SET					✓				✓	✓			
		GDC						✓							
	ATT SET	IMU					✓						✓		
		GDC	✓					✓				✓			
	CMC ATT	IMU		✓			✓								
		GDC			✓										
	GDC ALIGN	PRESS	✓												
		OFF		✓	✓	✓									
	ATTITUDE SET CONT PANEL	3 THUMB- WHEELS	✓									✓	✓		

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- (1) ATTITUDE AND ATTITUDE ERROR TO FDAI #1 FROM G&N AND TO FDAI #2 FROM THE SCS
- (2) DURING ENTRY, STABILITY ROLL ONLY IS SUPPLIED TO THE FDAI SELECTED AND TO THE ROLL STABILITY INDICATOR ON THE ENTRY MONITOR SYSTEM.
- (3) BMAG UNCAGE LOGIC MUST ALSO BE SATISFIED IN ADDITION TO SWITCHES SHOWN.
- (4) NECESSARY FOR CORRECT YAW DISPLAY DURING NON ENTRY MISSION PHASES.

CHART I

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Figure 2.3-10. ARS Switching

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The rate display sources (figures 2.3-9 and 2.3-10) will always be from either of the two gyro assemblies on a per-axis basis. The normal source for rate display will be the BMAG 2 gyros, and is selected by having the BMAG MODE switches in the ATT 1/RATE 2 or the RATE 2 position. The backup source is selected when the BMAG MODE switch is in the RATE 1 position. This will rate cage the BMAG 1 gyros and switch their outputs to the FDAI rate needles. When the ENTRY -.05 G switch is placed up, the roll rate gyro output is modified by the tangent 21 degrees and summed with the yaw rate. This summation results in a cancellation of the yaw rate sensed due to the CM rolling about the stability axis. Since this is a summation of a-c rate signals and since the gyro assemblies are supplied from separate a-c buses, selecting backup rate (BMAG 1) in yaw will automatically select the backup rate gyro (BMAG 1) in roll and vice versa. This prevents any phase difference from the two buses from affecting the summation of the two rate signals.

2.3.4 ATTITUDE CONTROL SUBSYSTEM (ACS).

2.3.4.1 Introduction.

The SCS hardware used in controlling the spacecraft attitude and translation maneuvers include the gyro assemblies, rotation and translation controls, and two electronic assemblies. The electronic control assembly (ECA) provides commands as a function of both gyro and manual control (RC and TC) inputs to fire the RCS via the reaction jet/engine control assembly (RJEC). Alternate spacecraft attitude control configurations provide several means of both manually and automatically controlling angular rates and displacements about spacecraft axes. Accelerations along spacecraft axes are provided via the TC. The crew uses this control for both docking and delta V maneuvers.

2.3.4.2 Hardware Function (ACS).

While a description of each SCS component was given in paragraph 2.3.2.1, this description considers those functions and interfaces used in the ACS.

2.3.4.2.1 Gyro Assembly - 1 (GA-1).

GA-1 contains three BMAGs that can provide pitch, yaw, and roll attitude error signals. These error signals are used when SCS automatic attitude hold is desired. The signals interface with the electronics control assembly (ECA). The BMAGs can be rate caged independently by control panel switching to provide backup rate information, or held in standby. The GA-1 BMAGs can be uncaged independently (by axis) during SCS attitude hold if the MANUAL ATTITUDE switch is in RATE CMD, the BMAG MODE switch in ATT 1 RATE 2, the ENTRY .05 G switch is OFF and no RC breakout switch is closed (figure 2.3-11).

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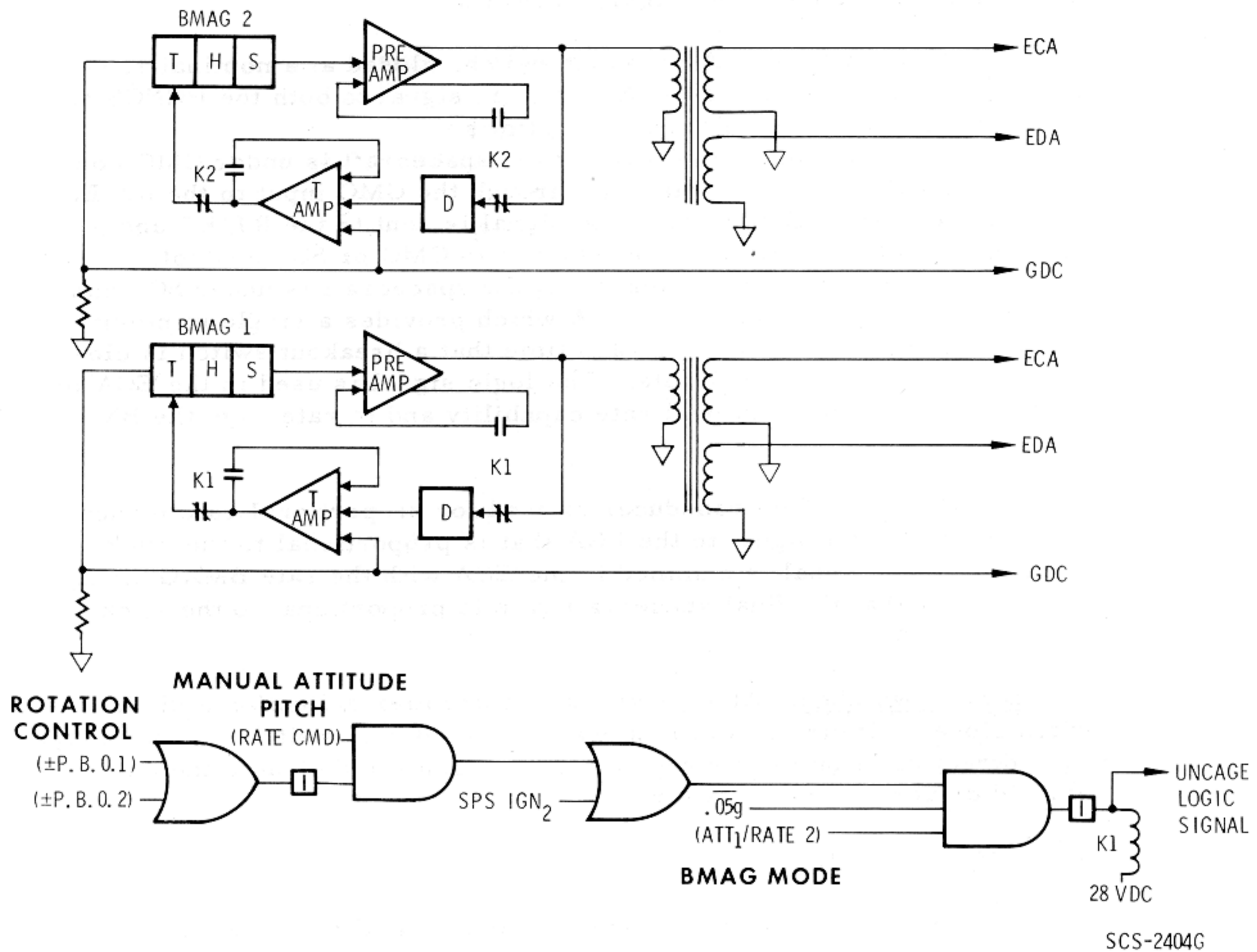


Figure 2.3-11. BMAG Logic and Outputs

2.3.4.2.2 Gyro Assembly - 2 (GA-2).

GA-2 contains three BMAGs that are always rate caged. These BMAGs normally provide pitch, yaw, and roll rate damping for SCS automatic control configuration and proportional rate maneuvering. The rate signals interface with the ECA. When backup rate by axis is selected (RATE 1), the GA-2 signal(s) is not used.

2.3.4.2.3 Rotational Controllers (RC-1 and RC-2).

The RCs provides the capability of controlling the spacecraft attitude simultaneously in three axes. Either controller provides the functions listed below for each axis (pitch, yaw, roll) and for each direction of rotation (plus or minus).

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Within the RC are six breakout switches, three transducers, and twelve direct switches. (See figure 2.3-12.)

Breakout Switches. A breakout switch, closed at a nominal 1.5-degree RC deflection, routes a 28-vdc logic signal to both the PGNCS and the SCS for attitude control inputs as follows:

- a. Rotation Command to CMC. If the spacecraft is under CMC control, the signal commands rotations through the CMC input to the RJ/EC.
- b. Acceleration Command. The signal is sent to the RJ/EC and commands rotational acceleration whether in CMC or SCS control.
- c. Minimum Impulse Command. If the spacecraft is under SCS control, the logic signal goes to the ECA which provides a single minimum impulse command to the RJ/EC each time that a breakout switch is closed.
- d. Proportional Rate Enable. The logic signal is used in the ECA to enable the manual proportional rate capability and to rate cage the BMAGs in GA-1.

Transducer. The transducer is used for proportional rate maneuvers. It provides a signal to the ECA that is proportional to the stick deflection. The signal is summed in the ECA with the rate BMAG signal in such a way that the final spacecraft rate is proportional to the stick (RC) deflection.

Direct Switches. At 11 degrees of controller deflection a direct switch closes. If direct power is enabled, the direct switches route 28 vdc to the direct coils on the appropriate RCS engines and disable the auto coil solenoid drivers in that axis (or axes).

2.3.4.2.4 Translation Controller.

The translation controller provides the capability of manually commanding simultaneous accelerations along the spacecraft X-, Y-, and Z-axes. (See figure 2.3-13.) It is also used to initiate several transfer commands. These functions are described below.

Translation Commands.

- a. CMC Control. If the spacecraft is under CMC control, a translation command results in a logic signal (28 vdc) being sent to the CMC. The CMC would provide a translation command to the RJ/EC.
- b. SCS Control. If the spacecraft is under SCS control, the translation command is sent to the RJ/EC.

Clockwise Switches (CW). A clockwise rotation of the T-handle will disable CMC inputs to the RJ/EC. A logic signal (\overline{CW}) is sent to the CMC when the T-handle is at null.

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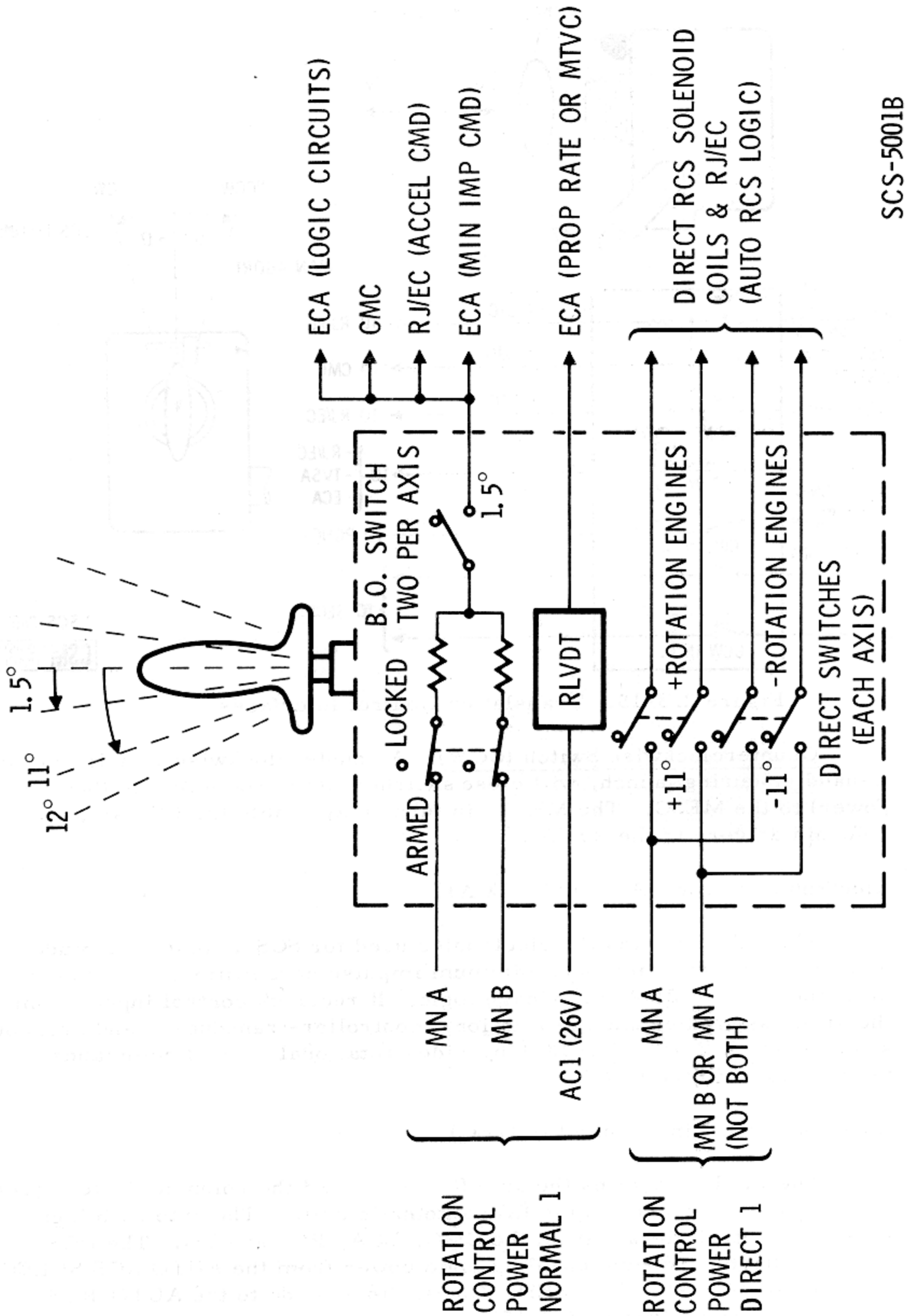


Figure 2.3-12. Rotation Control Interfaces



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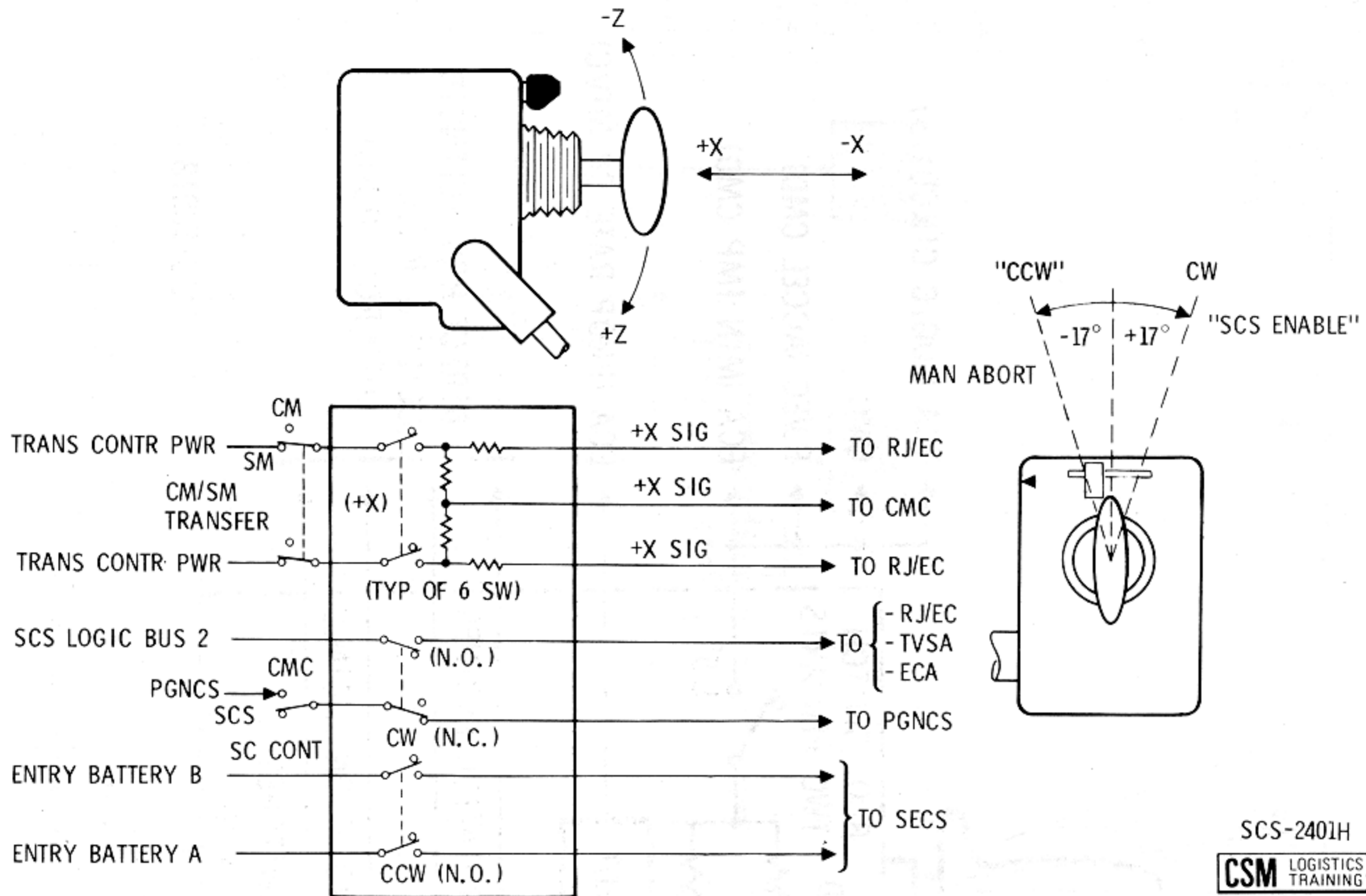


Figure 2.3-13. Translation Control Interfaces

Counterclockwise Switch (CCW). A counterclockwise rotation of the T-handle, during launch, will close switches which route 28 vdc (battery power) to the MESC. The MESC, in turn, may enable the RCS auto coil solenoid drivers in the RJ/EC.

2.3.4.2.5 Electronics Control Assembly (ECA).

The ECA contains the electronics used for SCS automatic attitude hold, proportional rate, and minimum impulse capabilities. It also contains the attitude BMAG(s) uncage logic. It receives control inputs from the gyro assemblies and the rotational controller-transducers and breakout switches (MIN IMP). The ECA provides rotational control commands to the RCS logic in the RJ/EC.

2.3.4.2.6 Reaction Jet Engine Control (RJ/EC).

The RJ/EC contains the auto RCS logic and the solenoid drivers (16) that provide commands to the RCS automatic coils. The auto RCS logic receives control signals from the CMC, ECA, RC, and TC. The RCS solenoid drivers receive enabling logic power from the AUTO RCS SELECT switches on MDC-8. The MESC supplies the 28 vdc to the AUTO RCS SELECT switches (figure 2.3-15).

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2.3.4.3 Reaction Control Subsystem Interface.

2.3.4.3.1 General.

The RCS provides the rotation control torques and translation thrusts for all ACS functions. Prior to CM/SM separation, the SM RCS engines are used for attitude control. The CM RCS is used after separation for control during entry (figures 2.3-14 and 2.3-15). The CM has only 12 RCS engines and does not have translational capability via the TC. After CM/SM separation, the A/C ROLL AUTO RCS SELECT switches have no function, as the 12 CM engines need only 12 AUTO RCS SELECT switches (figure 2.3-15).

An RCS engine is fired by applying excitation to a pair (fuel and oxidizer) of solenoid coils; the pair will be referred to in the singular as a solenoid coil. Each engine has two solenoid coils. One coil is referred to as the automatic coil, the other as the direct coil. Only the automatic coils receive commands from the RJ/EC. The direct commands are routed directly from the RC direct switches (or other switches). The automatic and direct commands are discussed in the following paragraphs.

2.3.4.3.2 Automatic Coil Commands.

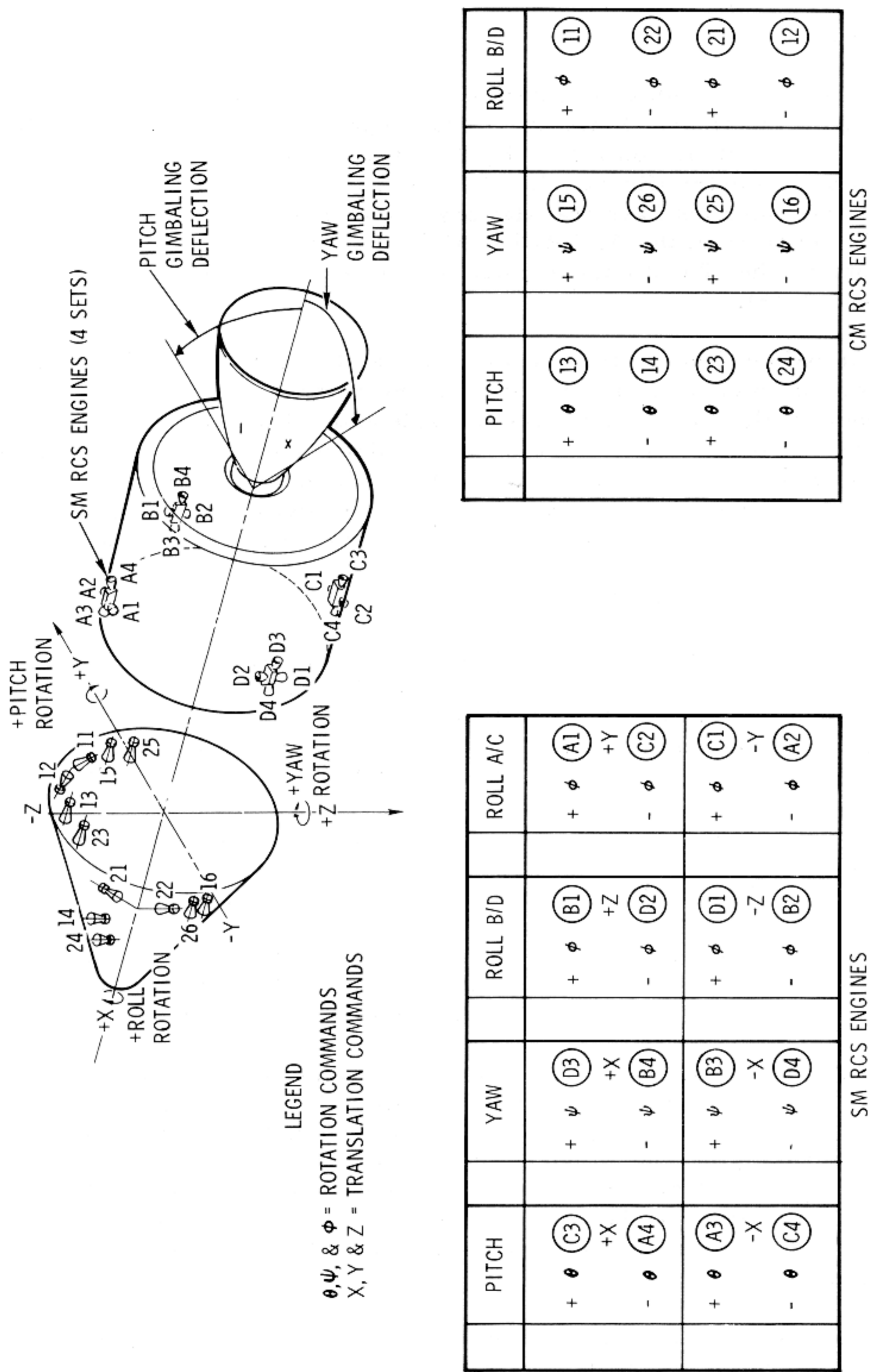
Power. The automatic (auto) coils are supplied 28-vdc power via one set of contacts of the AUTO RCS SELECT switches (figure 2.3-15). The solenoid is operated by switching a ground to the coil through the appropriate solenoid driver in the RJ/EC. The auto coil power is obtained from the STABILIZATION/CONTROL SYSTEM A/C ROLL, B/D ROLL, PITCH and YAW circuit breakers on panel 8. The 28 vdc lines to the auto coils on SM engines (jets) except A₁, A₂, C₁, and C₂ are switched at CM/SM transfer to CM coils. The wires from the A/C ROLL AUTO RCS SELECT switches to SM engines A₁, A₂, C₁, and C₂ are open-ended after transfer. These switches have no function for the CM configuration. Enabling power for the RCS solenoid drivers is supplied to the second set of contacts of the AUTO RCS SELECT switches through the MESC (A and B) from the SCS CONTR/AUTO MNA and MNB circuit breakers (MDC-8).

The CM jets are supplied from two separate propellant systems, 1 and 2. The jets are designated by the propellant system. Each propellant system supplies half the CM jets, distributed such that one jet for each direction (plus and minus) and for each axis (pitch, yaw, and roll) is supplied from the 1 system and the other from the 2 system. When the RCS TRNFR switch is placed from SM to CM, motor switch contacts transfer auto coil power from SM engines to CM engines. Each motor switch contact transfers six engines.

Auto RCS Logic. Commands to the RCS engines are initiated by switching a ground, through the solenoid driver, to the low voltage side of the auto coils. The solenoid drivers receive commands from the auto RCS

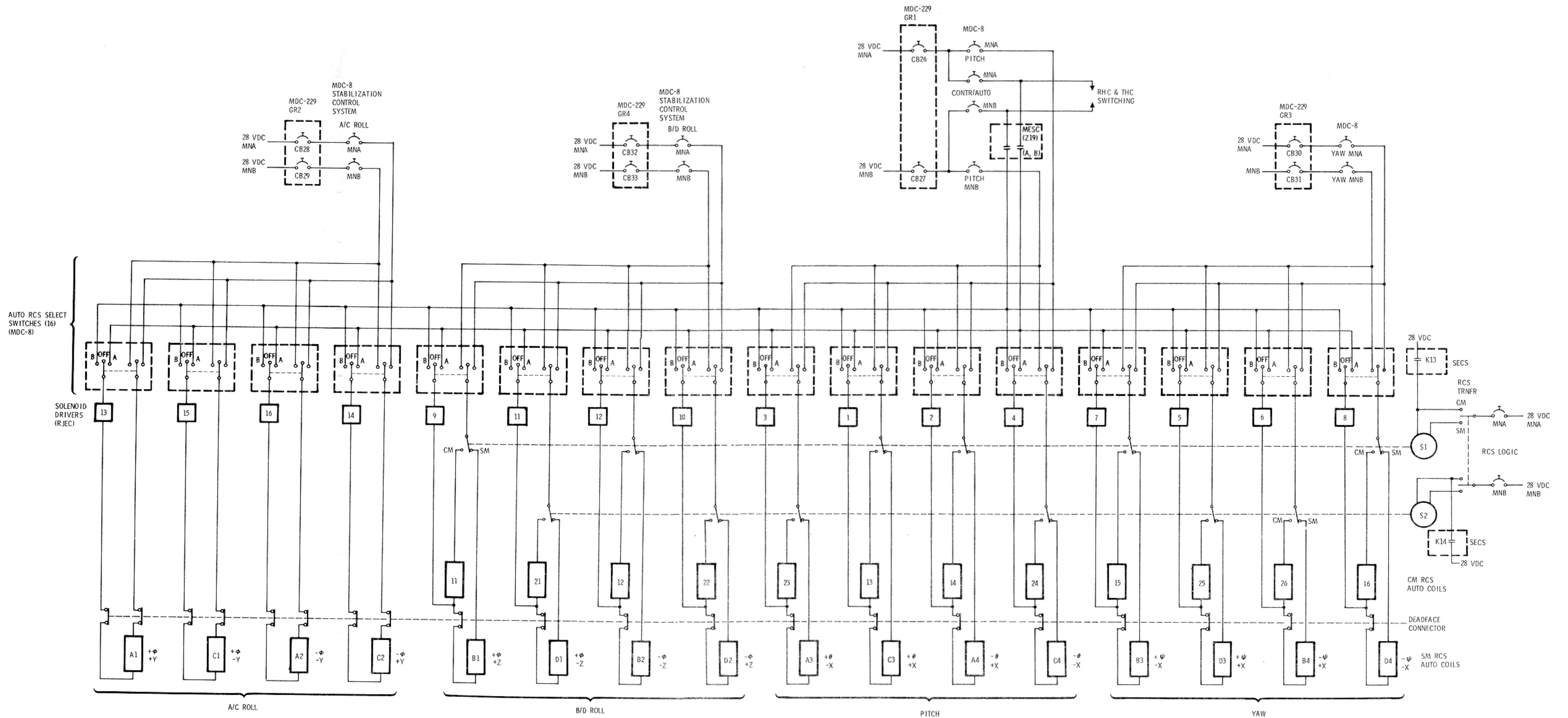
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Figure 2.3-14. SM Jet Functions



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Figure 2.3-15. Auto RCS Enabling Power

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logic circuitry contained in the RJ/EC. The auto RCS logic performs two functions:

- a. Enables the command source selected based on logic signals received from the control panel or manual controls.
- b. Commands those solenoid drivers necessary to perform the desired maneuver.

The logic receives RCS commands from the following sources:

- CMC (provides rotational and translational commands).
- ECA (provides rotational commands for either automatic attitude hold, proportional rate, or minimum impulse control).
- RC-1 and/or RC-2 (breakout switches [BO] provide continuous rotational acceleration).
- TC (provides translational acceleration commands).

The auto RCS logic (figure 2.3-16) is represented by four modules: one module each for pitch and yaw and two for roll (B/D and A/C). The solenoid drivers (four) associated with each module (shown as numbered triangles) correspond to the RCS engine solenoid drivers. The command sources (listed above) are shown as separate inputs to the modules, while enable/disable logic is represented as a single line to each module.

A detailed functional drawing of the pitch auto RCS logic shows how the command priorities are mechanized in the RJ/EC. (See figure 2.3-17.)

The ACS provides the following commands, listed in order of descending priority:

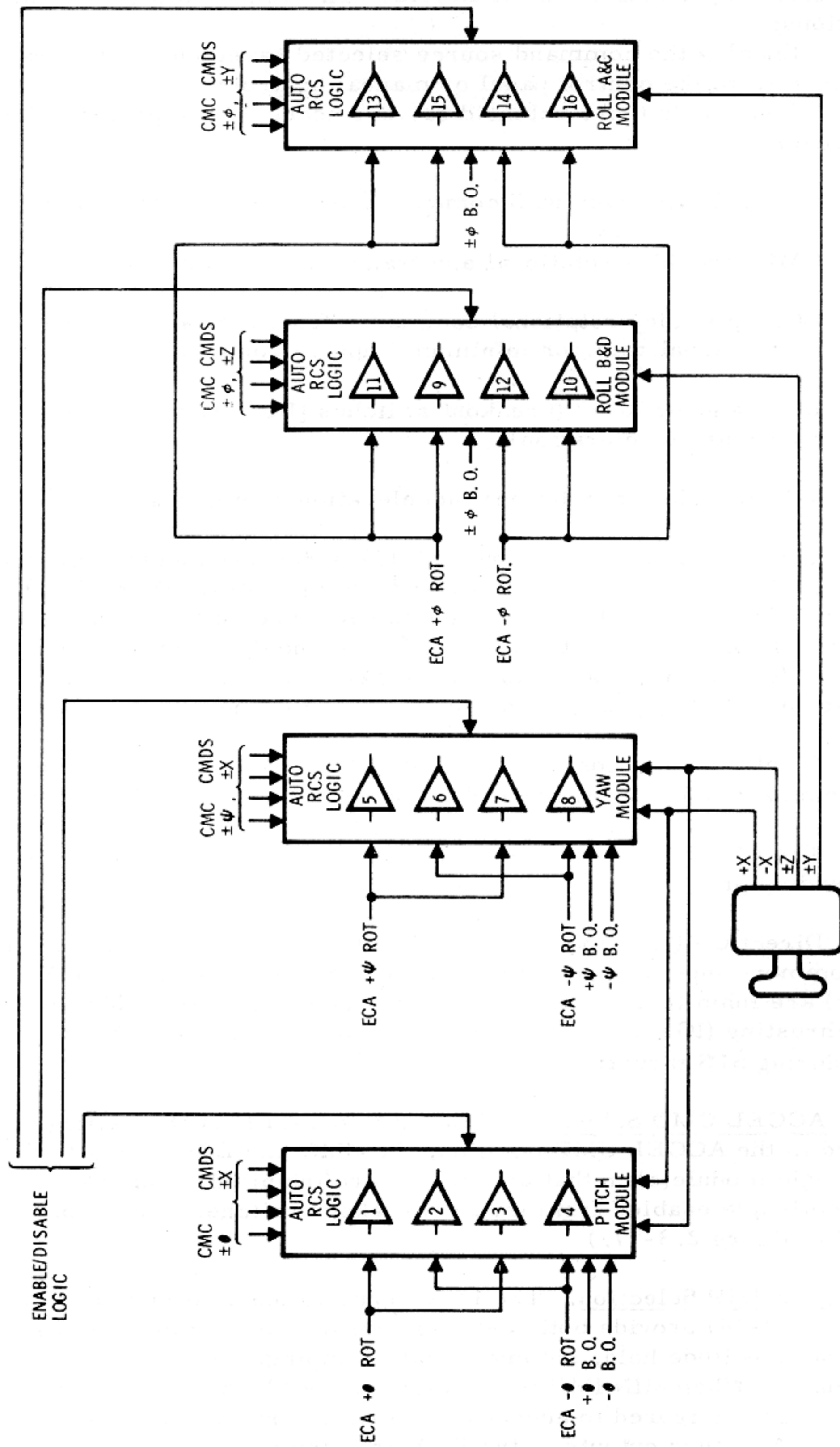
Direct Coil Commands. At the initiation of direct coil commands, all command input channels to the auto RCS logic module(s) in that axis (axes) are inhibited. Pitch and yaw auto commands are inhibited during SPS thrusting (IGN 1). This prevents auto coil commands from firing the RCS during SPS thrusting.

ACCEL CMD Selection. If a MANUAL ATTITUDE switch(es) is placed in the ACCEL CMD position, the CMC and ECA inputs to the auto RCS logic module(s) in that axis (axes) are inhibited. Commands to fire auto coils are enabled from the RC breakout switches. (See bottom "and" gates in figure 2.3-17.)

MIN IMP Selection. The ECA inputs to the auto RCS logic modules (figure 2.3-14) provide both the minimum impulse commands, as well as automatic attitude hold, automatic rate damping, and proportional rate command. When MIN IMP is selected on a MANUAL ATTITUDE switch, the ECA is configured to accept RC breakout commands and supply output pulses. All other outputs of the ECA are inhibited in the ECA.

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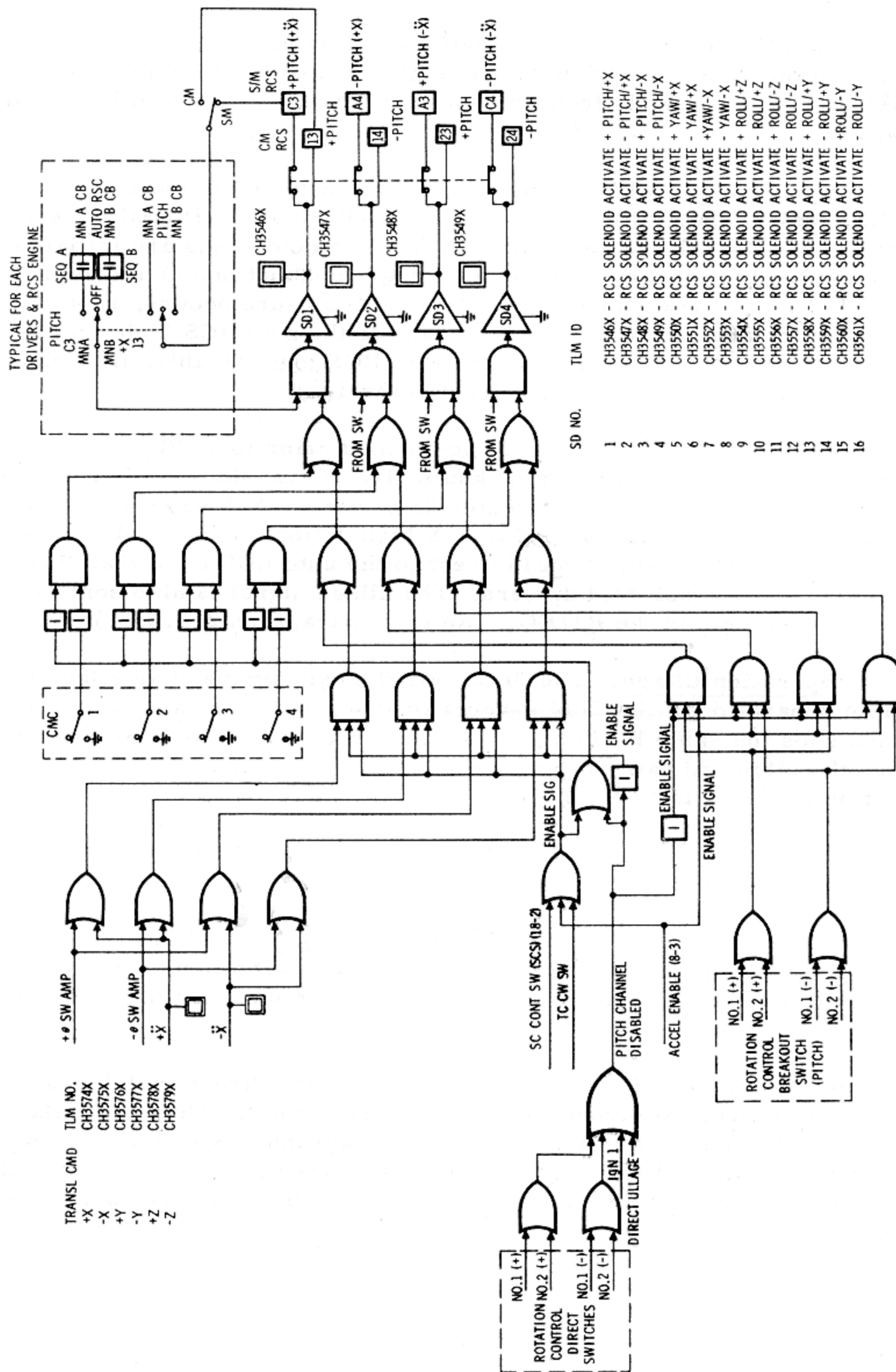


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Figure 2.3-16. Auto RCS Signal Flow

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SCS-5009B

Figure 2.3-17. Auto RCS Logic



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2.3.4.3.3 Direct Coil Commands.

The RCS engines can be operated by applying 28 vdc to the direct coils, as the other side of the direct coils is hard wired to ground. The coils receive commands from the sources described in the following paragraphs (shown in figure 2.3-18).

Direct Rotational Control. The direct switches in the rotation controllers (RCs) are enabled when the ROT CONTR PWR-DIRECT 1 & 2 switches on MDC-1 are up or down. The RCS commands are initiated when the RC is deflected a nominal 11 degrees about one or more of its axes. At this displacement a switch (direct) closure occurs, routing 28 vdc to the appropriate direct coils and to the auto RCS logic (paragraph 2.3.4.3.2). The signal to the auto RCS logic disables the solenoid drivers in the channel(s) under direct control.

Direct Ullage. An ullage is performed prior to an SPS thrust maneuver. Direct ullage is a backup to TC +X translation. Pressing the DIRECT ULLAGE pushbutton routes 28 vdc to the SM direct coils on the pitch and yaw RCS engines used for +X translations. (See table on figure 2.3-14.) A signal (28 vdc) is sent to the auto RCS logic that disables the pitch and yaw solenoid drivers. The ullage signal is also sent to the SPS ignition logic in the RJ/EC. (Refer to paragraph 2.3.5.5.)

Separation Ullage. The SECS (MESC) can command an ullage to enable separation of the CSM spacecraft from the S-IVB adapter. The ullage uses the same RCS engines as the direct ullage command and disables the pitch and yaw solenoid drivers. The enabling logic for this function is shown in figure 2.3-18.

SM/CM Separation. The SM JETTISON CONTROLLER sends commands to SM direct coils for -X translation and +roll rotation.

CM PROPELLANT JETT-DUMP Control. This function is used after the RCS capability is no longer required. Actuation of the CM PROPELLANT DUMP switches will provide commands to the direct coils on all CM engines, except 13 and 23.

At CM-SM separation the lines from the RC direct switches are transferred from SM direct coils to CM direct coils. This is similar to the automatic coil transfer described in paragraph 2.3.4.3.2, except that either of the two transfer motors transfers power to all CM direct coils. The lines for direct or separation ullage (steps b and c), are open ended at CM-SM separation.

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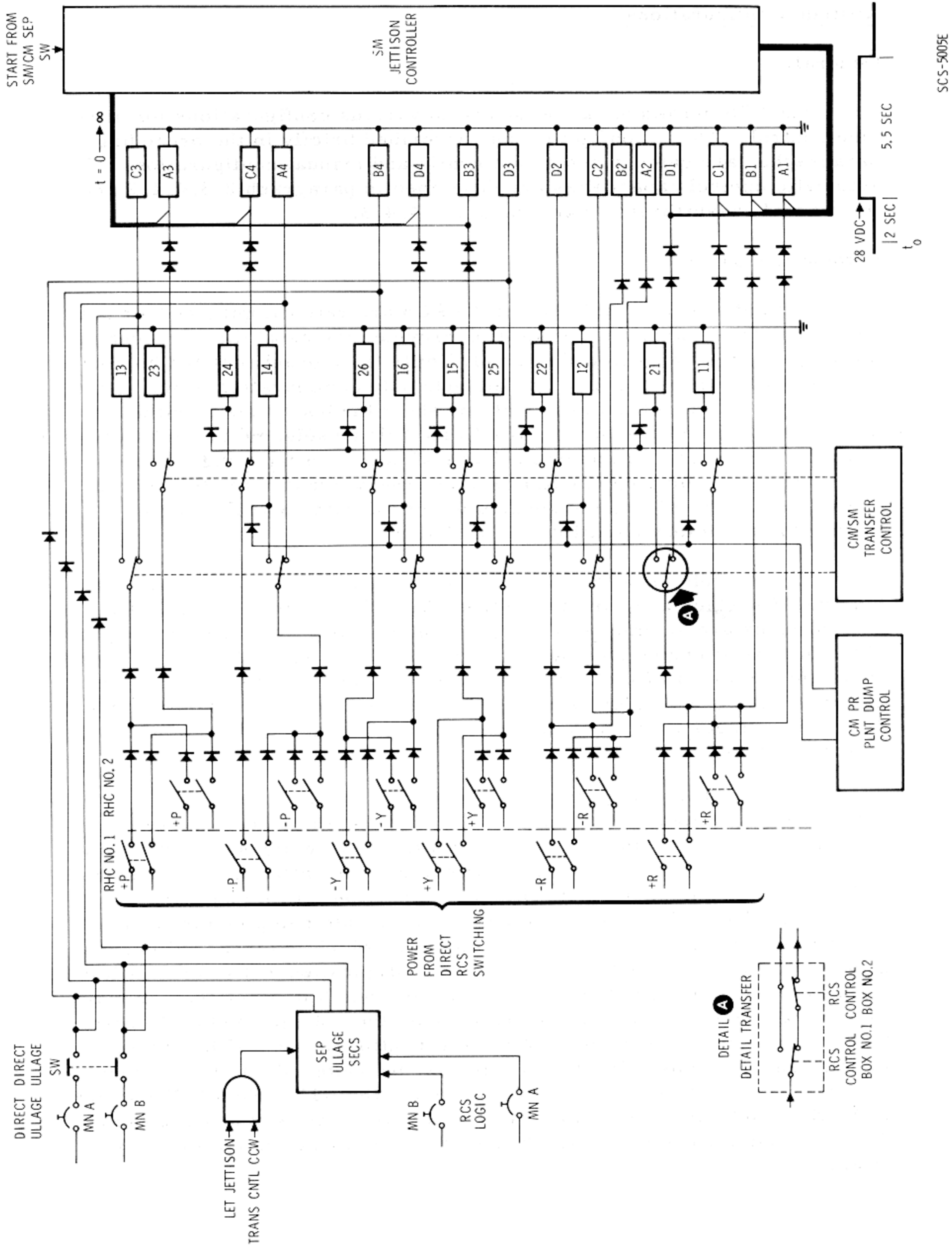


Figure 2.3-18. Direct Control Loop

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2.3.4.4 Attitude Configurations.

2.3.4.4.1 General.

The SCS hardware can be placed in various configurations for attitude control. These configurations, described briefly in the preceding paragraphs, are categorized as automatic and manual configurations. The automatic control capabilities are described in paragraph 2.3.4.4.2 and the manual capabilities in paragraph 2.3.4.4.3.

2.3.4.4.2 Automatic Control.

The automatic capabilities of the ACS are rate damping and attitude hold. The rate damping configuration provides the capability of reducing large spacecraft rates to within small limits (rate deadband) and holding the rate within these limits. The attitude hold configuration provides the capability of keeping angular deviations about the body axes to within certain limits (attitude deadband). If attitude hold is selected in pitch, yaw, and roll, the control can be defined as maintaining a fixed inertial reference. The rate damping function is used together with the attitude hold configuration; therefore, the description of the rate control loop is included in the following attitude hold discussion.

Attitude hold uses the control signals provided by the rate and attitude BMAGs which are summed in the ECA. (See figure 2.3-19.) The control loops are summed at the input to a switching amplifier which provides the on-off engine commands to the auto RCS logic. Each of the three switching amplifiers (pitch, yaw, and roll) has two outputs that provide clockwise and counterclockwise rotation commands. The polarity of the d-c input voltages to the switching amplifiers determines the commanded direction of rotation.

If the switching amplifier input signal is smaller than a specific value, neither output is obtained. This input threshold required to obtain an output is the switching amplifier deadband. Manually-selectable gain authority provides flexibility in the selection of the attitude hold deadband width, the rate damping sensitivity and proportional rate command authority. The RATE switch controls both the rate damping threshold and the proportional rate command authority, which is discussed in paragraphs to follow. Since the attitude hold configuration utilizes the attitude and rate loops, the switching amplifiers will switch on when the summation of attitude error and rate signals equals the voltage deadband. Attitude error signals are scaled (20:1) as a function of the RATE switch. In addition, a deadband limiter circuit may be switched into the attitude error loops. This is accomplished by having the ATT DEADBAND switch in MAX, which, in effect, blocks the first four degrees of attitude error. The rate and attitude error deadbands are summarized in the following table.

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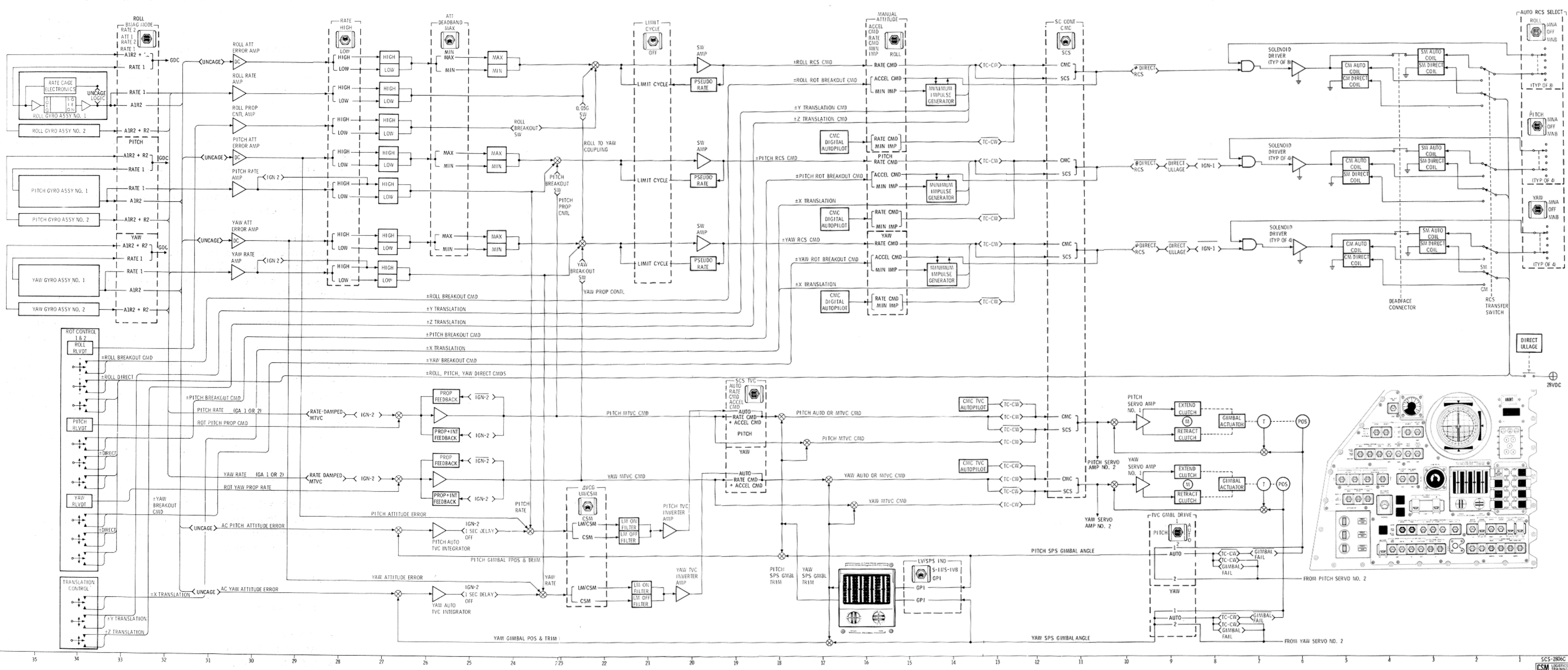


Figure 2.3-19. SCS Attitude and Thrust Vector Control System

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RATE Sw Position	Rate deadband (°/Sec)	ATT DEADBAND Sw Position	
		MIN	MAX
LOW	±0.2	±.2 deg	±4.2 deg
HIGH	±2	±4 deg	±8 deg

During attitude hold it is desirable to maintain minimum rotation rates to conserve propellants. This capability is provided by the pseudo-rate circuit. Pseudo-rate feedback around the switching amplifier is enabled via the LIMIT CYCLE switch. Placing the LIMIT CYCLE up causes the switching amplifier output to pulse off and on when the input level approaches the threshold.

When the pseudo-rate mode is used, the pulse duration from the switching amplifier may be insufficient to insure proper operation of the solenoid valves in the RCS. This applies for operation near the deadband limits. To insure a sufficiently-long pulse to the solenoids, a one-shot circuit is connected downstream from the switching amplifier. The one shot provides a single minimum-impulse command (on-time) for each switching amplifier output pulse. When the switching amplifier pulse width exceeds the one shot on-time, the longer RCS command is initiated. The output pulse width of the one shot is a function of the d-c bus voltage; the pulse width increases as the bus voltage decreases. This is because the solenoid valve pickup time increases as the bus voltage decreases; therefore, a longer RCS "on" command is required. Thus, the one-shot circuit provides compensation for bus voltage variations: the pulse width varies approximately from 13 msec to 17 msec over a bus voltage range of 30 to 25 vdc. The one-shot circuit is also used in manual minimum impulse control. This configuration is described in the next paragraph.

An additional rate control loop is used for the yaw axis only. This loop is enabled during entry, after .05 G, and is used to cancel unwanted yaw rate BMAG signals. The unwanted yaw BMAG signals are those signals resulting from roll maneuvers about the stability X-axis. The 21-degree offset between this axis and the X-axis causes the yaw BMAG to sense a component of the entry roll rate.

2.3.4.4.3 Manual Control.

Following are the manual attitude control capabilities.

- DIRECT
- ACCELERATION CMD
- MINIMUM IMPULSE
- PROPORTIONAL RATE

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The commands listed are initiated by manual inputs to either rotation controller. With the exception of direct, the RC commands rotations through the RCS auto coils.

The manual rotation control capabilities are discussed further in the following paragraphs.

Proportional Rate. Proportional rate provides the capability to command spacecraft rates that are proportional to the RC deflection. The RC transducer output is summed (by axis) through the breakout switch logic path (figure 2.3-19) with the rate signal from the BMAG. Initially, the RC output (commanded rate) will be larger than the BMAG output (actual rate) so that the summed signals will be greater than the switching amplifier threshold. The RCS engines will fire until the summation of the rate and commanded rates are within the switching amplifier deadband. When the RCS engines stop firing, the spacecraft will continue to rotate at a constant rate until a new rate is commanded.

Since the MANUAL ATTITUDE switch must be in RATE CMD for proportional rate, the spacecraft will be under automatic control when the RC is released.

The rate commanded by a constant stick deflection is a function of the ratio of the control loop gains. The ratio has two possible values which are selected by the RATE switch. The nominal rate commanded at maximum stick deflection (soft stop), for both rate switch positions, are shown in the following list.

RATE Switch Position	Maximum Prop. Rate CMD (By Axis)	
	Pitch & Yaw	Roll
LOW	0.7 deg/sec	0.7 deg/sec
HIGH	7 deg/sec	20 deg/sec

The switching chart shows the LIMIT CYCLE switch in the OFF position. Performing a proportional rate maneuver with pseudo-rate enabled (switch-on), required more RCS fuel than the same maneuver without pseudo-rate feedback.

Minimum Impulse. Minimum impulse provides the capability of making small changes in the spacecraft rate. When minimum impulse is enabled, the switching amplifier output is inhibited. Thus, the spacecraft (attitude) is in free drift in the axis where minimum impulse is enabled, if direct control is not being used.

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	MANUAL						AUTOMATIC	
	DIRECT	ACCELERATION CMD	TRANSLATION	MINIMUM IMPULSE	PROPORTIONAL RATE (4)	RATE DAMPING	ATTITUDE HOLD	
MANUAL ATTITUDE	ACCEL CMD	✓	✓			✓		
	RATE CMD		✓			✓		
	MIN IMP		✓				✓	
PITCH	ACCEL CMD	✓	✓			✓		
	RATE CMD		✓			✓		
	MIN IMP		✓				✓	
YAW	ACCEL CMD	✓	✓			✓		
	RATE CMD		✓			✓		
	MIN IMP		✓				✓	
BMAG MODE	RATE 2					✓		
	ATT 1/RATE 2					✓		
	RATE 1					✓		
	RATE 2					✓		
	ATT 1/RATE 2					✓		
	RATE 1					✓		
SC CONT	CMC		✓			✓		
	SCS		✓			✓		
	CW		✓			✓		
TRANS CONTROL	NEUTRAL		✓			✓		
	UP/DOWN							
RHC DIRECT PWR		✓	(3)	(3)	(3)	(3)	(3)	
ROT CONTROL	B.O. SW							
	DIRECT SW	CLOSE			CLOSE(2)	OPEN	OPEN	
LIMIT (1) CYCLE						✓		
ENTRY	.05G							
	OFF						✓	

- (1) NOT REQUIRED TO ENABLE A PARTICULAR FUNCTION.
- INDICATES DESIRED POSITION FOR RCS PROPELLANT CONSERVATION.
- (2) IF B.O. SW IS OPEN THE S/C WILL BE IN FREE DRIFT.
- (3) IF "ON", DIRECT SW IN ROTATION CONTROL MUST BE "OPEN".
- (4) MAXIMUM RATE ATTAINABLE IS FUNCTION OF RATE-HIGH/LOW SWITCH

GENERAL COMMENTS:

- A. THE CAPABILITIES, IN GENERAL, ARE LISTED IN ORDER OF THEIR PRIORITY.
- B. WHEN MORE THAN ONE SWITCH POSITION IS CHECKED (✓) THE CAPABILITY WILL BE ENABLED IN EITHER POSITION.

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Figure 2.3-20. ACS Control Capabilities

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Minimum impulse control is commanded by the RC breakout switch. This switch provides a 28-vdc logic signal to the one-shot circuit in the ECA. The one shot (paragraph 2.3.4.4.2) provides a command to the auto RCS logic for a nominal 15 ms. Additional minimum impulse commands are obtained each time a breakout switch is closed (by repeated opening and closing of the breakout switch).

Acceleration Command. When acceleration command is enabled and a breakout switch is closed, continuous commands are sent to the appropriate RCS auto coils. The SC CONT switch has no function in enabling the acceleration command capability, which is second in priority only to direct coil operations. (Refer to paragraph 2.3.4.3.2.)

Direct. Direct control is similar to acceleration command except that the direct RCS coils are used. Also, instead of a breakout switch providing the firing command, the RC direct switch is used to provide 28 vdc straight to the direct coils (figure 2.3-19). Power to the RC direct switches is controlled by the two ROT CONTR PWR DIRECT switches on MDC-1, one switch controlling the 28 vdc for each RC. (See figure 2.3-26.) During direct control in an axis, all auto coil commands in that axis are inhibited in the auto RCS logic (figure 2.3-17).

2.3.4.5 Translation Control.

When power is supplied to the translation control (TC), a manual translational command fires auto coils to give acceleration(s) along an axis (or axes). The TRANS CONTR PWR switch on MDC-1 supplies 28 vdc to the TC translational switches (figure 2.3-26).

TC inputs are routed as logic inputs to the auto RCS logic when the spacecraft is under SCS control. However, during CMC control, TC commands arrive at the auto RCS logic via the CMC. (See figure 2.3-17.) Since the TC uses only SM RCS engines, after CM/SM separation the TC has no translation function.

Other translational control is possible from inputs other than the TC. These are direct ullage, CSM/LV separation ullage, and CM/SM minus —X translation (SM JETT CONT). These translation commands utilize direct coils. (See figure 2.3-18.)

Certain panel switch combinations are necessary for each ACS capability that has been discussed. For a summary, see figure 2.3-20.

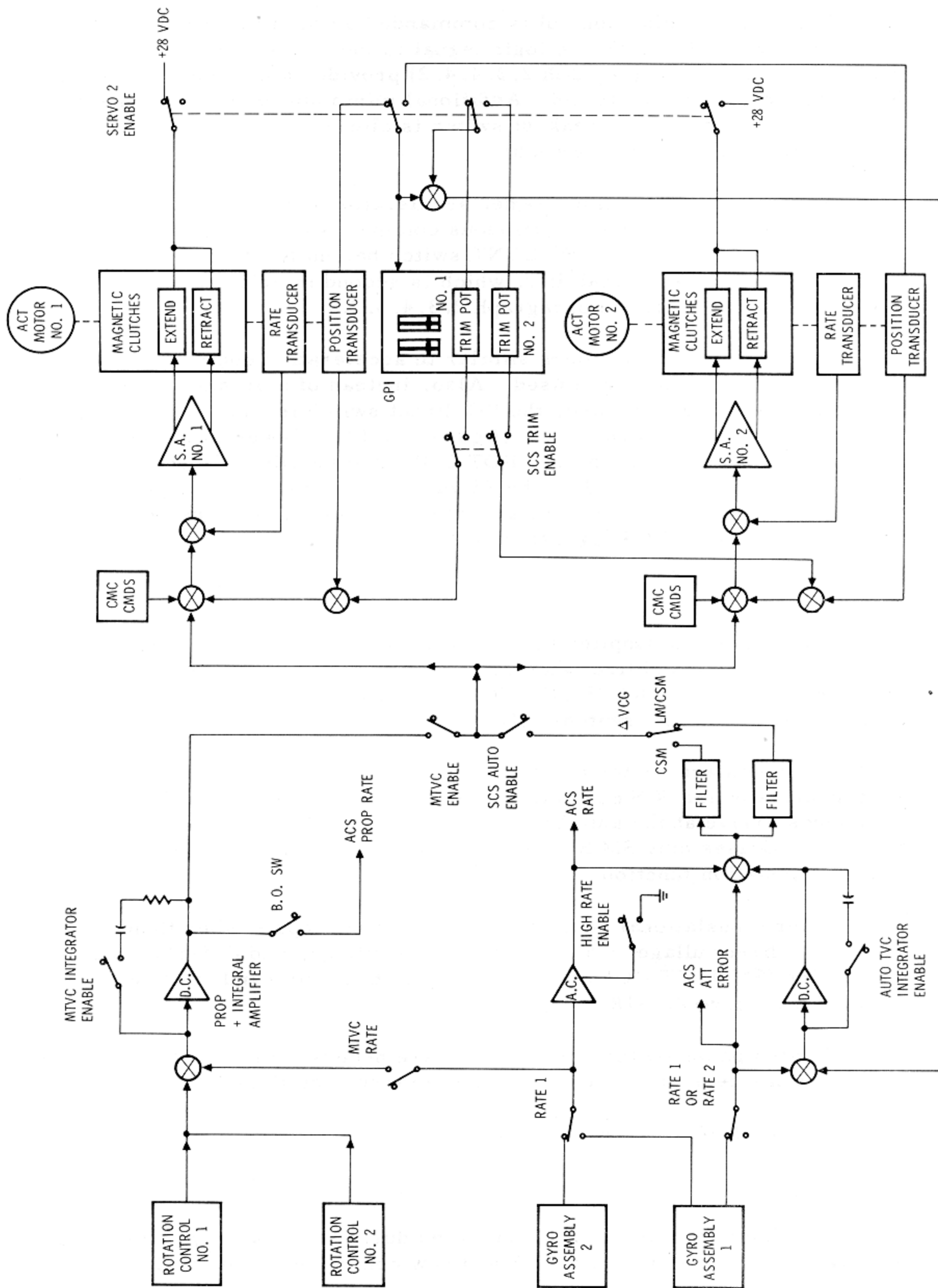
2.3.5 THRUST VECTOR CONTROL (TVC).

2.3.5.1 Introduction.

The spacecraft attitude is controlled during a delta V by positioning the engine gimbals (TVC) for pitch and yaw control while maintaining roll

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Figure 2.3-21. TVC Signal Flow

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attitude with the attitude control subsystem. The SCS electronics can be configured to accept attitude sensor inputs for automatic control (SCS auto TVC) or rotational controller (RC) inputs for manual thrust vector control (MTVC). Manual TVC can be selected to utilize vehicle rate feedback signals summed with the manual inputs; this comprises the MTVC/RATE CMD configuration. Selecting MTVC without rate feedback describes the MTVC/ACCEL CMD configuration. A different configuration can be selected for each axis; for example, one axis can be controlled manually while the other is controlled automatically.

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The following paragraphs present the characteristics of the SCS/TVC configurations. A switching table, specifying the panel switching and logic signals required for enabling each configuration, is included. The operation of the engine ignition/thrust on-off logic is also described.

2.3.5.2 TVC Panel Configurations.

On the simplified TVC signal flow diagram shown in figure 2.3-21 functional enabling switches are used for reference. The TVC switching table (figure 2.3-22) relates the functional switching and panel switching to the TVC configuration desired. Both figures are applicable to either the pitch or yaw TVC channel.

In general, it is possible to enable a functional switch through several (alternate) panel configurations. The alternate configurations usually require the CW logic signal which is obtained from a clockwise rotation of the translation controller (TC) T-handle. This provides a convenient means of transferring from one TVC configuration to another during the thrusting maneuver. The CW signal will also enable transfer from servo No. 1 to servo No. 2 (figure 2.3-22) under certain conditions. Thus, it is possible to transfer to a completely redundant configuration by using the TC clockwise switch.

The gimbal servo control loop consists of a servoamp that drives two magnetic clutch coils; one coil extends the actuator; the other retracts the actuator. Gimbal rate and position transducers provide feedback for closed loop control. Two servo control channels are provided in each axis, pitch and yaw. The active channel is selected through functional switch servo 2 enable (figure 2.3-21). Primary control utilizes servo No. 1. Servo No. 2, in an axis, can be engaged either by selecting 2 position on the TVC GMBL DR switch or by automatic transfer. Automatic transfer will occur, if the TVC GMBL DR switch is in the AUTO position and either the FS (fail sense) or CW logic signal is present. The CW logic will enable transfer to servo No. 2 in both axes, whereas, the FS logic will enable transfer only in the axis where it is present. The fail sense signal is generated in the motor excitation circuitry of servoactuator No. 1, occurring when an overcurrent is sensed. The transfer logic described is included in the switching table (figure 2.3-22).

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	CMC DAP (1)	SCS AUTO		MTVC		SCS GMBL TRIM	SERVO 1	SERVO 2 (3)
		SCS	AUTO	RATE CMD	ACCEL CMD			
SC CONT	✓	✓		✓	✓	✓		
BMAG MODE (PITCH & YAW)	CMC							
	SCS		✓	✓	✓	✓		
	RATE 2							
ATT 1/ RATE 2		✓						
	RATE 1							
SCS TVC (PITCH & YAW)	AUTO	✓	✓	✓				
	RATE CMD			✓				
	ACCELCMD				✓			
TVC GMBL DRIVE (PITCH & YAW)	1						✓	
	AUTO						✓	
	2							✓
XLATION CONTROL	NEUTRAL	✓	✓	✓			✓	
	CW				✓	✓		
IGN2 (2)		✓	✓	✓	✓			
FAIL SENSE SIG (PITCH OR YAW)								✓

- (1) CMC MODE SWITCH MUST BE IN AUTO
- (2) IGN2 SIGNAL APPEARS AT SPS IGNITION AND IS DELAYED OFF 1 SEC AT THRUST TERMINATION.
- (3) SERVO LOOP SWITCHING BY AXIS WITH FAIL SENSE LOGIC SWITCHING

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Figure 2.3-22. TVC Switching

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2.3.5.3 GPI Signal Flow.

The gimbal position display (figure 2.3-5) is used as a monitor of SPS pitch and yaw gimbal deflections from actuator null during CMC and SCS control of a ΔV . Prior to an SCS ΔV , the SPS engine must be positioned with the trim thumbwheels on the GPI. In this case, the GPI will display the trim gimbal angles that are set with the thumbwheels.

Since there is only one display panel of gimbal position, there are redundant indicators, servometric meter drivers, and power supplies associated with both the pitch and yaw position displays. (See figure 2.3-23.) When servo channel No. 1 is controlling the SPS actuator, the position input to both GPI indicators (pitch and yaw) is supplied from the No. 1 position transducer. If actuator control is transferred to the No. 2 servo, then the No. 2 position transducer drives both indicators in that axis. If the FDAI/GPI POWER switch is in BOTH position then all four indicators are powered. With the switch in position 1, the first and third indicators are enabled. The second and fourth indicators are energized with the switch in position 2.

2.3.5.4 SCS Auto TVC.

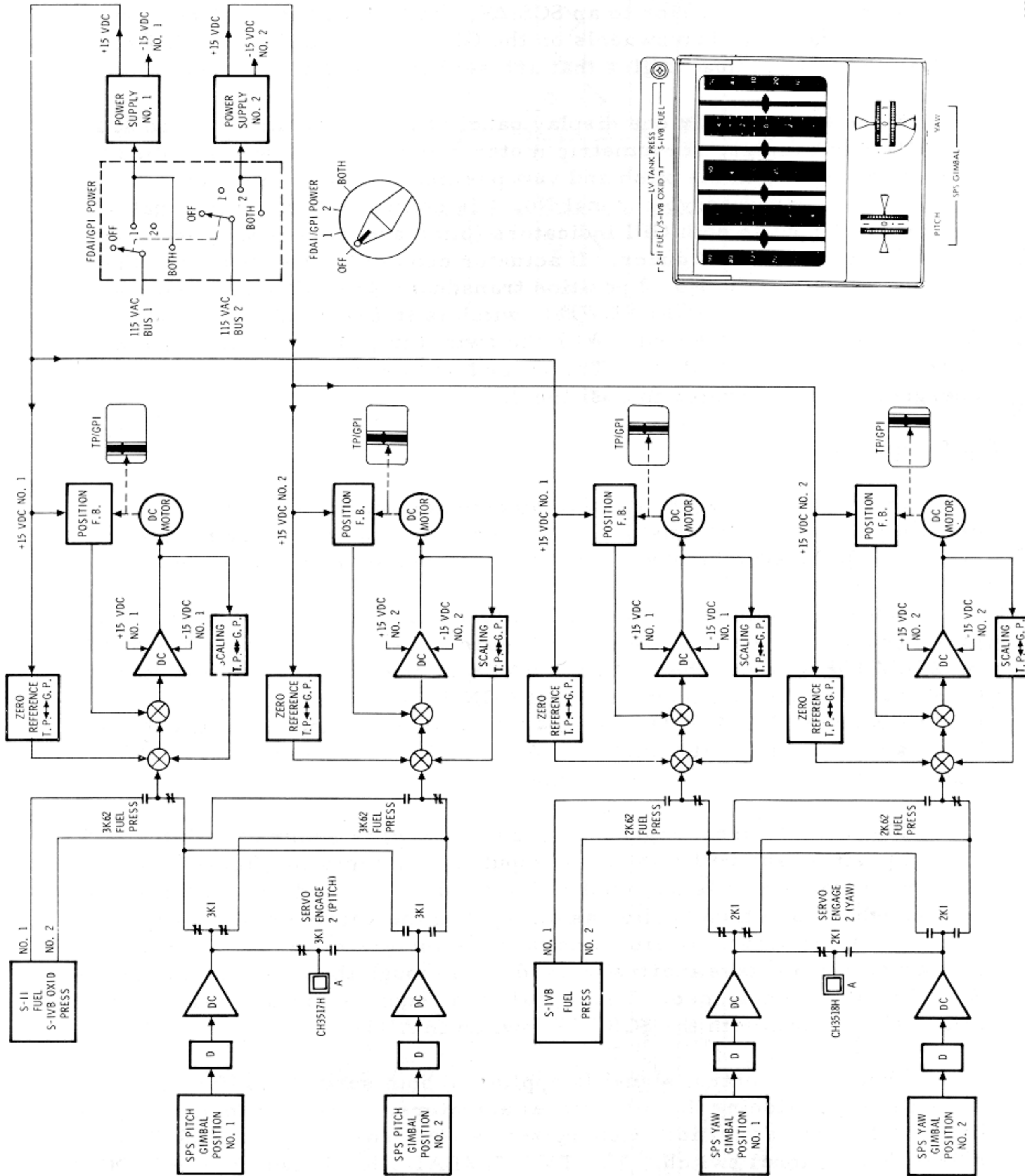
In order to configure the SCS electronics for an SCS auto TVC, certain panel switches must be positioned. In addition, other manual or automatic logic switching will affect the control signals and servo loops.

Since SCS auto TVC requires attitude error signals from GA-1, the gyro uncage logic must be satisfied (figure 2.3-11). This requires that the BMAG MODE switches be in ATT 1 RATE 2, the ENTRY-.05 G switch be OFF, and that the SPS ignition signal (IGN 2) be present. For attitude hold (paragraph 2.3.4.2.1), the IGN 2 logic was not needed as GA-1 can be uncaged by placing the MANUAL ATTITUDE switches to RATE CMD while having no breakout switch input.

The attitude error signal (in pitch and yaw) is summed with the SPS gimbal position and GPI trim at the input to an integrator (figure 2.3-22). The integrator output is summed with attitude error and rate, filtered for body-bending, and then applied as an input to the servo amplifiers (primary and secondary). During a delta V the integrator output insures that the thrust vector stays inertially fixed even though the cg shifts as the propellants are consumed. The signal path requires that the delta V is under SCS control with the SCS TVC switch in AUTO.

Though the control signal is applied to both servo amplifiers, only one will be positioning the SPS gimbal actuators. Selection logic controlling which servo amplifier is energized is represented by the SERVO 2 ENABLE functional switch. The TVC GIMBAL DRIVE switches on MDC-1 have AUTO positions which provide an automatic transfer from servo 1 to servo 2 if either a TC-CW switch is closed or an over-current logic

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Figure 2.3-23. GPI Signal Flow

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signal is sent from the SPS. Positioning the TVC GIMBAL DRIVE switches to 1 or 2 selects the desired servo loop, but overrides the TC—CW or over-current transfer.

Pre-thrust gimbal trim is accomplished by manually turning the trim wheels on the gimbal position indicator (GPI) to obtain the desired indicator readout. The trim wheel in each axis is mechanically connected to two potentiometers. As shown in figure 2.3-21, one potentiometer is associated with servo No. 1 and the second with servo No. 2. It is desirable to pretrim before an SCS delta V, to minimize the transient duration and the accompanying quadrature accelerations. It is also desirable to set the trim wheels properly before a CMC delta V if the SCS AUTO configuration is to serve as a backup. This will enable the SCS to relocate the desired thrust direction if a transfer is required after engine ignition.

2.3.5.5 Manual Thrust Vector Control.

Manual control of the thrust vector utilizes crew commands via the RC to position the gimbaled SPS. There are two types of MTVC: MTVC with rate damping (rate command) and MTVC without rate damping (acceleration command). Either mode of MTVC is selectable by panel switching. In addition, TC-CW logic provides either an automatic transfer from a PGNCS-controlled delta V or from an SCS auto delta V. (See figure 2.3-22.)

In order to provide ease of manual control, a proportional plus integral amplifier is incorporated in the MTVC signal flow path. The operation of this circuit can be described by considering the response to a step input; the output will initially assume a value determined by the proportional gain and the input amplitude. It will then increase, from this value, as a straight-line function of time. The slope of the line is a function of the input amplitude and the integrator constant. When the input is removed, the output will then drop by the initial value. With no additional inputs the output will theoretically remain constant (in practice, it will slowly decay). The circuit (integrator) provides the following capabilities:

- a. Maintain a gimbal deflection after returning the RC to rest.
- b. Make corrections with the RC about its rest position, rather than holding a large displacement.
- c. With no manual inputs, SC rate is damped out in the RATE CMD configuration.

The selection between the RATE CMD and ACCEL CMD configurations is made by enabling rate signals in the RATE CMD mode with the IGN 2 logic signal present (thrust on). This enables rate BMAG signals to be summed with RC inputs. The position of the BMAG MODE switch determines which rate source (BMAG 1 or 2) is summed, through its associated functional switch. Placing the SCS TVC switch in the ACCEL CMD position disables the rate command mode.

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The RATE CMD configuration is analogous to the proportional rate capability described in the ACS (paragraph 2.3.4) except there is no dead-band. With no manual input, the thrust vector is under rate BMAG control. If there is an initial gimbal cg misalignment, an angular acceleration will develop. The rate BMAG, through the proportional gain, will drive the gimbal in the direction necessary to cancel this acceleration. With no integrator, a steady-state rate would be required to hold the necessary gimbal deflection (through cg). However, due to the integrator, the rate is driven to zero. When an RC input (manual) is present, a steady-state vehicle rate will be established so that the integrator input goes to zero when the output value is sufficient to place the thrust vector through the cg. When the manual input is removed the rate is driven to zero.

When rate feedback is inhibited by selecting ACCEL CMD, the RC input must be properly trimmed to position the thrust vector through the cg. However, positioning the thrust vector through the cg only drives the rotational acceleration to zero. Additional adjustments (RC trimming) are necessary to cancel residual rates and obtain the desired attitude and positioning vector.

2.3.5.6 Engine Ignition, Thrust On-Off Logic.

This section describes the configurations available for ignition on-off control. Panel switch positions and/or logic signals necessary for a particular configuration are considered. The functions of output (logic) signals are given.

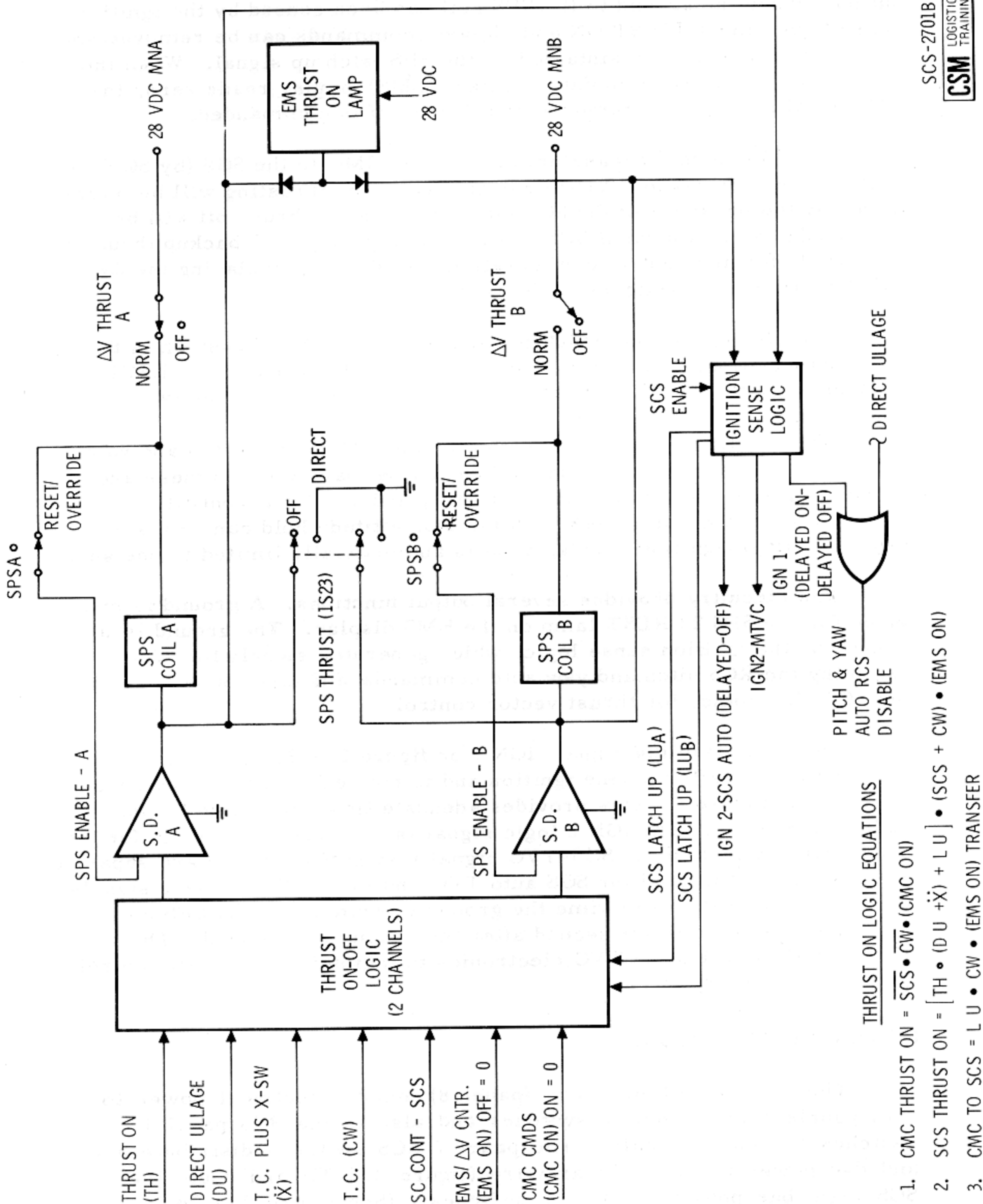
Redundant d-c power is supplied to redundant SPS coils and solenoid drivers (as shown in figure 2.3-24) via the ΔV THRUST (A and B) switches.

With the switch positions shown in figure 2.3-24, engine ignition is commanded by placing a ground on the low side of SPS coil No. 1. Thrust-off is commanded when the ground is removed. The ground switching can be accomplished in two basic ways. One method is to position the SPS THRUST switch from the NORMAL to the DIRECT ON position for engine turn-on, and later placing the ΔV THRUST A and B from NORMAL to OFF to terminate thrust. The second method is to switch the ground through the solenoid driver as commanded by the thrust on-off logic.

Engine ignition will be commanded by the thrust on-off logic when any one of the thrust-on logic equations shown in figure 2.3-24 is satisfied. The CMC commands thrust-on (equation 1) by supplying a logic 0 to the thrust on-off logic when the SC CONT switch is in the CMC position and the translation controller (TC) is not clockwise (\overline{CW}). When the CMC changes the logic signal from a 0 to a 1, thrust-off is commanded.

For the SCS control configuration the SC CONT sw must in the SCS position or the TC handle clockwise (CW). A thrust-on enabling signal is obtained from the EMS/ ΔV display. Thrust-on is then commanded by

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Figure 2.3-24. Engine Ignition-Thrust On-Off Logic

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commanding a +X-axis acceleration and pressing the THRUST ON push-button. When the ground to the SPS coil has been sensed by the ignition sense logic, the THRUST ON and +X-axis commands can be removed and engine ignition will be maintained by the SPS latch up signal. When the ΔV counter on the entry monitor system (EMS) display reads zero, the EMS enabling signal is removed and thrust-off is commanded.

If TVC control is transferred from the CMC to the SCS (by SC CONT switch to SCS or TC to CW) after engine ignition, thrusting will be maintained by the presence of the SCS latch up signal. Thrust-off will be commanded as in a normal SCS control configuration. A backup thrust-off command, for any control configuration, is obtained by placing the ΔV THRUST (A and B) switches to the OFF position.

The +X logic signal which is necessary to enable thrust-on in the SCS configuration, can be obtained from either the DIRECT ULLAGE pushbutton or the TC +X contacts. The difference between the two commands are:

- a. Direct ullage uses the direct coils and inhibits the pitch and yaw solenoid drivers; thus, attitude hold cannot be maintained in these axes. Ullage-ignition overlap time is completely under manual control.
- b. When commanding A+X with the TC, attitude hold can be maintained. Ullage-ignition overlap time is automatically limited to one second.

The circuitry provides several output functions. A ground is provided for the SPS THRUST lamp on the EMS display. The ground is also sensed by the ignition sense logic, which generates signals for both disabling the RCS pitch and yaw auto commands and also for configuring the SCS electronics for thrust vector control.

The RCS disabling signal, IGN 1 on figure 2.3-24, is not present until one second after engine ignition and is not removed until one second after engine turn-off. This provides adequate time for engine thrust buildup and decay. The IGN 2 logic signal is required in the logic for the functional switches in the SCS-TVC signal flow paths. There are separate IGN 2 signals generated for SCS auto TVC and for MTVC. These signals are generated at the same time the ground is switched to the SPS coil, but are not removed until one second after the ground is removed. The delayed OFF enables the TVC electronics to maintain spacecraft control during thrust decay.

2.3.6 POWER DISTRIBUTION.

The SCS circuit breakers (panel 8) supply electrical power to both panels 1 and 7 power switches and also to the SCS panel 1 switches for logic signals. The panel 7 SCS switches distribute a-c and d-c power to the SCS hardware (figure 2.3-26) and route the SCS logic bus power to panel 1 switches. (See figure 2.3-26.) The power switching for the two rotation hand controllers and the translation hand controller is on panel 1. (See figure 2.3-26.)

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Figure 2.3-25. Deleted

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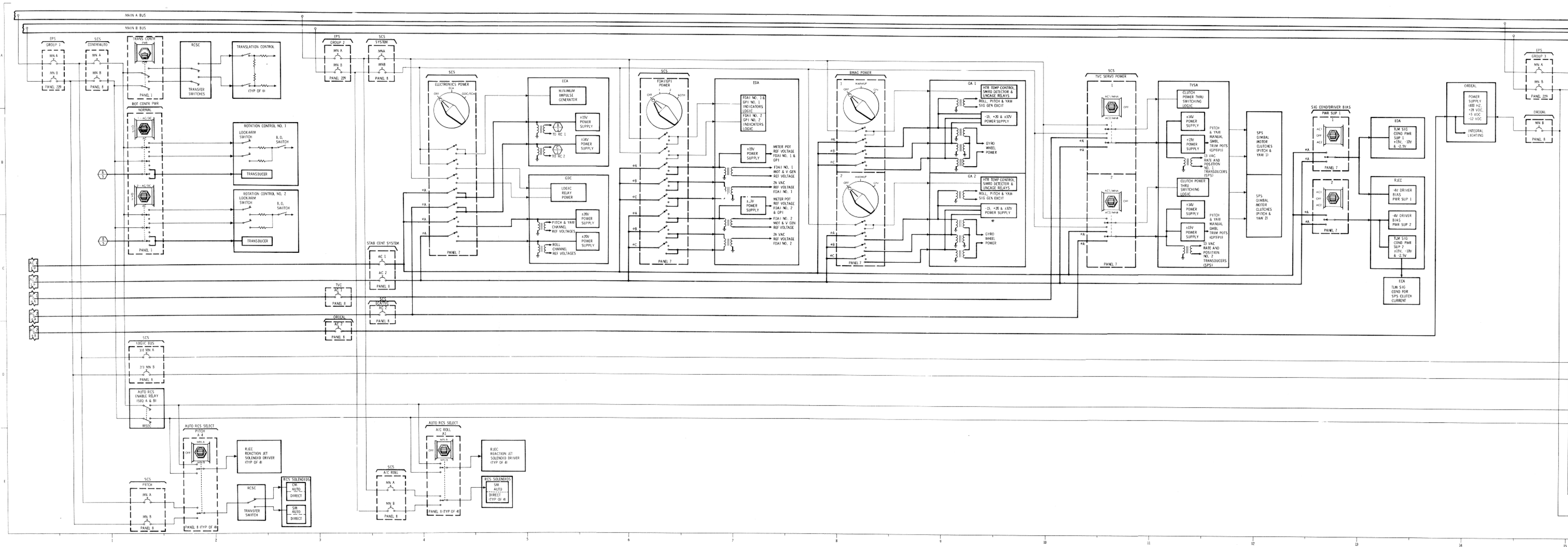


Figure 2.3-26. SCS D-C Power Distribution (Sheet 1 of 2)

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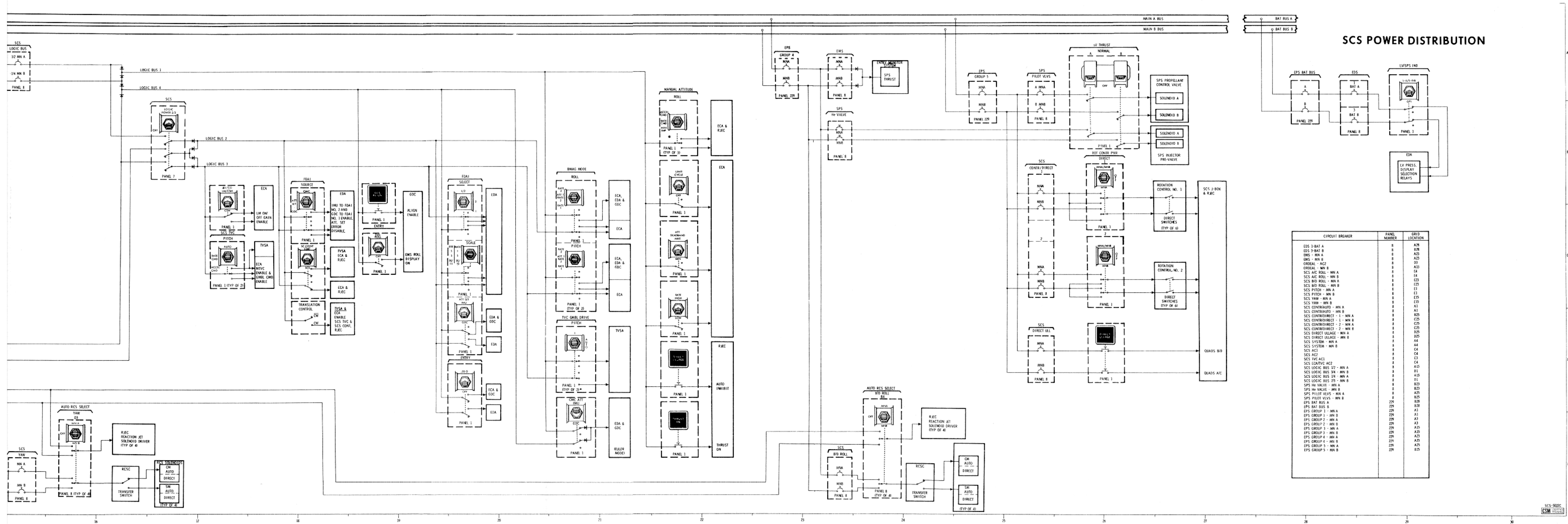


Figure 2.3-26. SCS D-C Power Distribution (Sheet 2 of 2)

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Figure 2.3-27. Deleted

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The SCS performance data is included in the CSM Spacecraft Operational Data Book (SNA-8-D-27). For the SCS operational limitations and restrictions refer to AOH, Volume 2, including the Malfunction Procedures.

2.3.7 ENTRY MONITOR SYSTEM.

The entry monitor system (EMS) provides a visual monitor of automatic primary guidance navigation and control system (PGNCS) entries and delta velocity maneuvers. The EMS also provides sufficient display data to permit manual entries in the event of PGNCS malfunctions together with a command sent to the SCS for SPS engine cutoff. The delta velocity display can also be used as the cue to initiate manual thrust-off commands if the automatic-off commands malfunction. During rendezvous the EMS provides a display of VHF ranging information.

Self-test provisions are provided by a function switch for the three operational modes (entry, delta V, and VHF ranging) to provide maximum system confidence prior to actual use.

The EMS performance data is included in the CSM Spacecraft Operational Data Book (SNA-8-D-27). For the EMS operational limitations and restrictions refer to AOH, Volume 2, including the Malfunction Procedures.

2.3.7.1 Entry Functions.

The EMS provides five displays and/or indications that are used to monitor an automatic entry or to aid in performing a manual entry.

2.3.7.1.1 Threshold Indicator (.05 G).

The threshold indicator, labeled .05 G, illuminates when the atmospheric deceleration is sensed. The altitude at which this indicator is illuminated is a function of the entry angle (velocity vector with respect to local horizontal), the magnitude of the velocity vector, geographic location and heading, and atmospheric conditions. Bias comparator circuits and timers (figure 2.3-28) are used to initiate this indicator. The signal used to illuminate the indicator is also used internal to the EMS to start the corridor evaluation timer, scroll velocity drive, and range-to-go circuits.

2.3.7.1.2 Roll Stability Indicator.

The roll stability indicator (RSI) provides an indication of lift vector position throughout entry. With the ATT SET switch in the GDC position, the RSI will be aligned prior to 0.05G by rotating the yaw thumbwheel on the attitude set control panel with the EMS ROLL switch in the entry position while pressing the GDC ALIGN button. During entry, stability axis roll attitude will be supplied to the RSI by the gyro display coupler. There are no degree markings on the display, but the equivalent readout will be zero when the RSI points toward the top of the control panel. During the entry RSI rotates in the opposite direction to the spacecraft roll.

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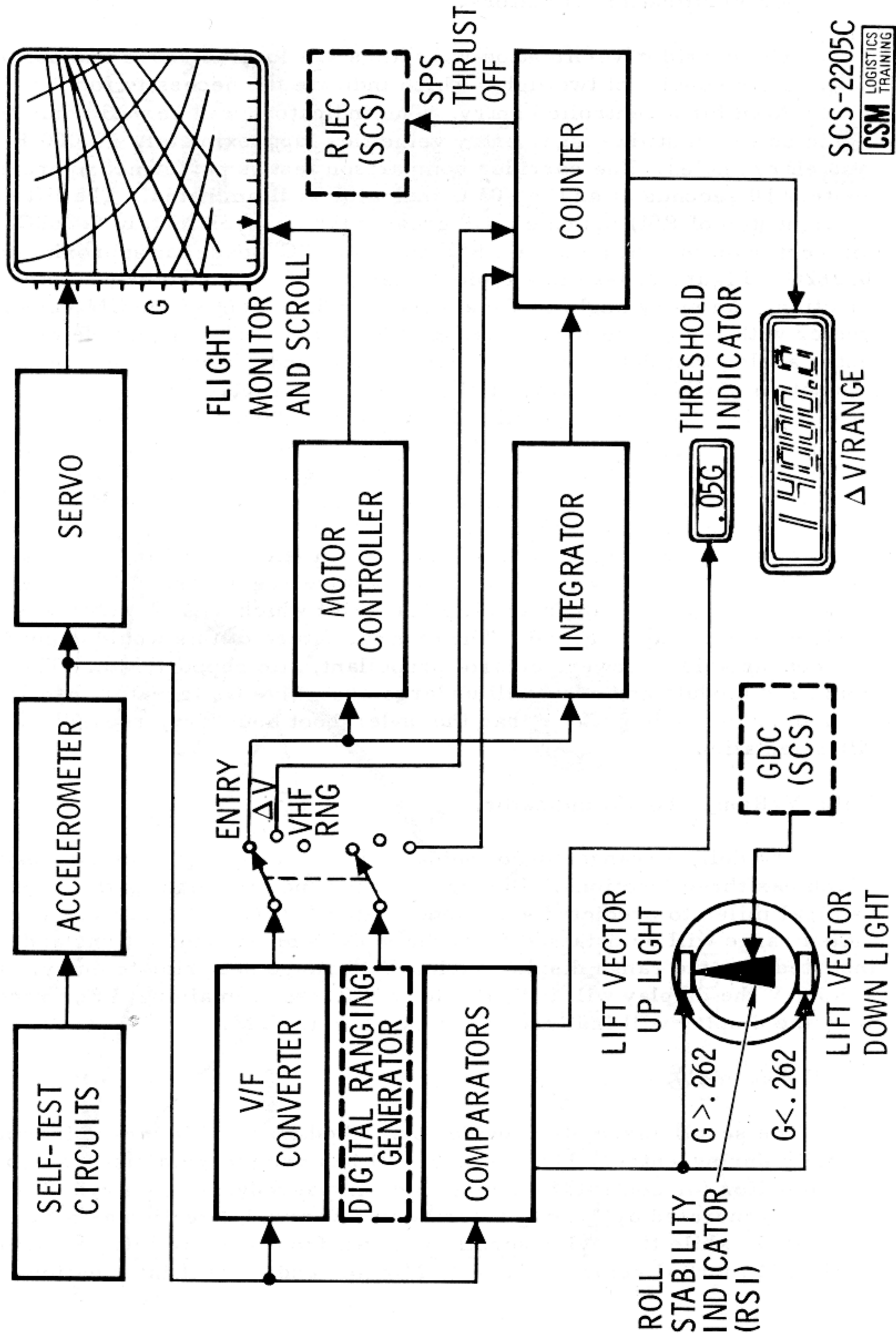


Figure 2.3-28. EMS Block Diagram

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2.3.7.1.3 Corridor Verification Indicators.

The corridor verification indicators are located above and below the RSI. They consist of two lights which indicate the necessity for lift vector up or down for a controlled entry. The indicators will be valid only for vehicles which utilize lunar entry velocities (approximately 35,000 FPS) and entry angles. The corridor comparison test is performed approximately 10 seconds after the .05 G indicator is illuminated. The lift vector up light (top of RSI) indicates "G" greater than approximately 0.262G. The lift vector down light (bottom of RSI) indicates "G" less than approximately 0.262G. Figure 2.3-29 is a typical example of the corridor evaluation function. An entry angle is the angular displacement of the CM velocity vector with respect to local horizontal at 0.05G. The magnitude of the entry angles that determines the capture and undershoot boundaries will be a function of CM lift-to-drag (L/D) ratio. The angles shown are for a L/D of 0.3 to 0.4. The EMS positive lift overshoot boundary is that entry angle that produces approximately 0.262G at approximately 10 seconds after the .05 G indicator is illuminated. An entry angle greater than the EMS positive lift overshoot boundary will cause the upper corridor verification light to be illuminated. Conversely, an entry angle less than the positive overshoot boundary will light the lower corridor light. Entry angles less than the capture boundary will result in noncapture regardless of lift orientation. Noncapture would result in an elliptical orbit which will re-enter when perigee is again approached. The critical nature of this would depend on CM consumables: power, control propellant, life support, etc. The command module and crew will undergo excessive Gs (greater than 10G) with an entry angle greater than the undershoot boundary, regardless of lift orientation.

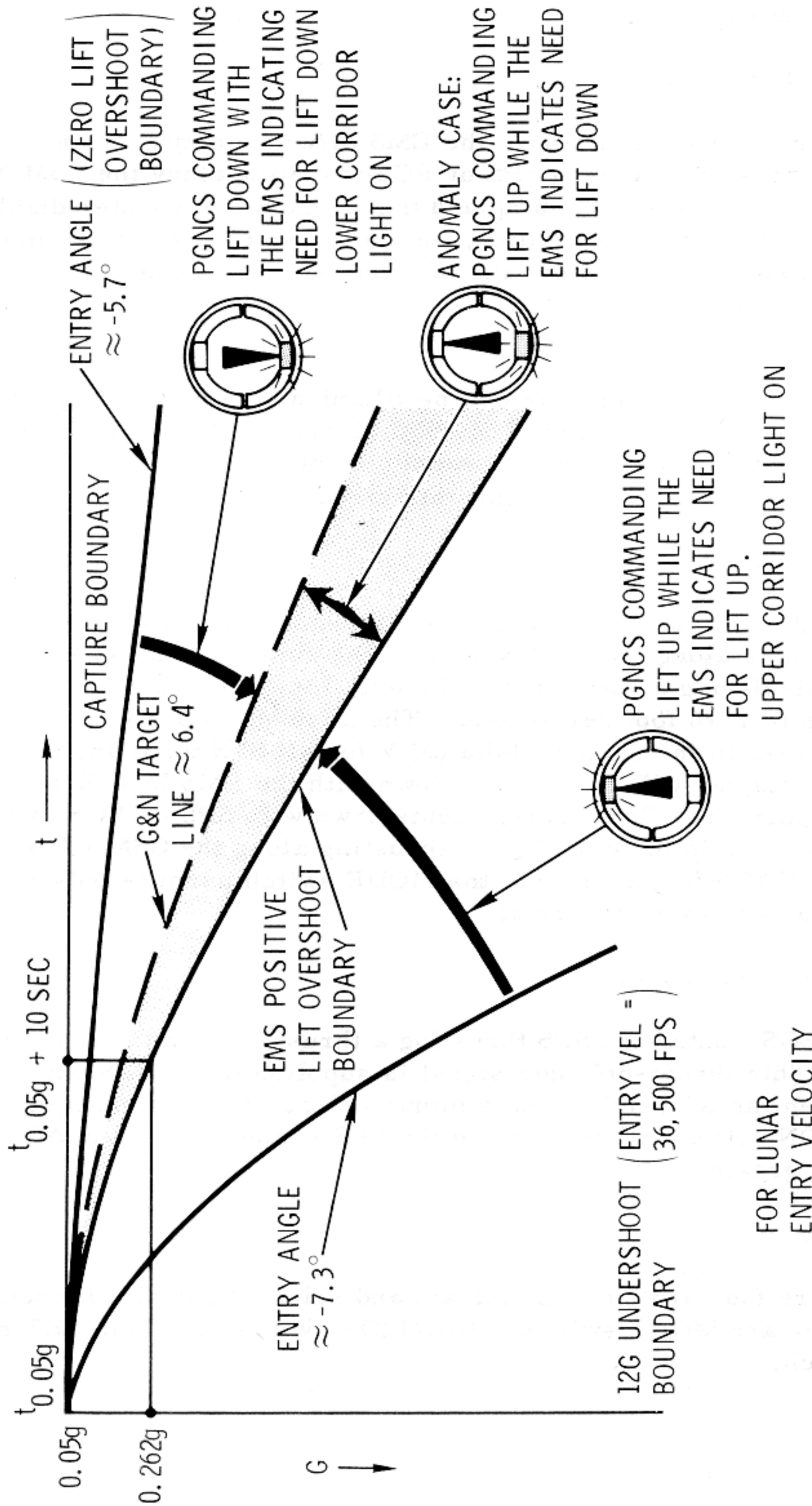
2.3.7.1.4 Delta V/Range-To-Go Indicator.

The delta V/range-to-go indicator is an electronic numeric readout which has three functions. During entry the inertial flight path distance in nautical miles to predicted splashdown after 0.05G is displayed. The predicted range will be obtained from the PGNCS or ground stations and inserted into the range display during EMS range set prior to entry. For a delta V the display will indicate the ΔV (ft/sec) remaining. For rendezvous the display will indicate the distance to the LM.

2.3.7.1.5 Scroll Assembly.

The scroll assembly provides a scribed trace of G versus inertial velocity during entry. The mylar scroll has printed guidelines which provide monitor (or control) information during aerodynamic entry. The entry trace is generated by driving a scribe in a vertical direction as a function of G level, while the mylar scroll is driven from right to left proportional to the CM inertial velocity change. Monitor and control information for

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Figure 2.3-29. EMS Corridor Evaluation

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safe entry and range potential can be observed by comparing the slope of the entry trace to the slope of the nearest guidelines (G onset, G offset and range potential).

2.3.7.2 Delta Velocity Functions.

In addition to entry functions, the EMS provides outputs related to delta velocity maneuvers during SPS or RCS thrusting along the CSM X-axis. Both the "SPS THRUST" lamp and the ΔV numeric counter display information during a ΔV . In addition, an automatic thrust-off command signal is supplied to the SCS when the ΔV counter reaches zero.

2.3.7.2.1 SPS Thrust-On Indicator.

The SPS thrust-on indicator will be illuminated any time a ground is present on the low side of either of the SPS bipropellant solenoid control valves if either of the EMS circuit breakers on panel 8 are set. None of the EMS or MDC switches will inhibit this circuit.

2.3.7.2.2 Delta Velocity Indicator.

The electro-luminescent (EL) numeric readout displays the delta velocity remaining along the CSM X-axis. The numeric display has the capability of displaying a maximum of 14,000.0 fps down to a -1000.0 fps. The readout is to 1/10 foot per second. The ΔV /EMS SET rocker switch will be used to set in the desired delta (Δ) V for all SPS thrusting maneuvers. The ΔV display will count up or down with the EMS MODE switch in the NORMAL position. The display counts down with SPS or RCS thrusting along the CSM +X-axis or up with RCS thrusting along the CSM -X-axis. The BACKUP/VHF RNG position of the MODE switch permits only a decreasing readout during thrusting.

2.3.7.2.3 SPS Thrust-Off Command.

During SCS-controlled SPS thrusting a thrust-off command is supplied by the EMS. This thrust-off logic signal is supplied to the SPS engine on-off circuit when the ΔV display reads minus values of ΔV . Consequently, the THRUST ON button will not turn on the SPS engine unless the ΔV display reads zero or greater.

2.3.7.3 EMS Switches.

There are four switches to activate and select the desired function in the EMS. They are MODE switch, FUNCTION switch, ΔV /EMS SET switch, and GTA switch.

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2.3.7.3.1 MODE Switch.

The MODE switch has three positions: NORMAL, STBY, and BACKUP/VHF RNG. The STBY position applies power to the EMS circuits; it inhibits system operation but does not inhibit set functions. The NORMAL position permits the self-tests to function. It also is the normal position for operations when the FUNCTION switch is in the ENTRY and ΔV positions. The BACKUP/VHF RNG position is used as a backup in the entry and delta V operations and is the proper position during VHF ranging. The BACKUP/VHF RNG position will be used as a backup to initiate the scroll velocity drive and the range display countdown in the event of failure of the .05 G circuits. The BACKUP/VHF RNG position energizes the .05 G light, but does not activate the corridor verification circuits for a display.

2.3.7.3.2 FUNCTION Switch.

The FUNCTION switch is a 12-position switch which is used to select the desired function in the EMS. Three positions are used for delta V operations. Eight positions are used for entry, entry set and self-test. The remaining position is OFF. One position is used for VHF ranging.

<u>Switch Position</u>	<u>Function</u>
OFF	Deactivates the EMS except the SPS THRUST ON light and the roll stability indicator.
EMS test 1	Tests lower trip point of 0.05 G - threshold comparator and enables slewing of the scroll.
EMS test 2	Tests the high trip point of the .05 G - threshold comparator.
EMS test 3	Tests lower trip point of the corridor verification comparator and enables slewing of the ΔV /RANGE display for EMS test 4 operations.
EMS test 4	Tests the range-to-go integrator circuits, G servo circuits, G-V plotter and range-to-go circuits.
EMS test 5	Tests high trip point of corridor verification comparator and enables slewing of scroll.
RNG SET	Establishes circuitry for slewing the ΔV /RANGE display.
Vo SET	Establishes circuitry for slewing the scroll to the predicted inertial velocity at 0.05G.

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<u>Switch Position</u>	<u>Function</u>
ENTRY	Operational position for monitoring the CM earth atmosphere entry mode.
ΔV TEST	Operational position for self-test of delta V circuits.
ΔV SET/VHF RNG	Establishes circuitry for slewing the ΔV /RANGE display. Enables VHF ranging display.
ΔV	Operational position for accelerometer to drive the ΔV /RANGE display for X-axis accelerations.

2.3.7.3.3 ΔV /EMS SET Switch.

The ΔV /EMS SET switch, a five-position rocker switch, is used to drive either the ΔV /RANGE display or the EMS scroll. With the FUNCTION switch in the ΔV SET/VHF RNG, RNG SET, and EMS TEST 3, depressing the ΔV /EMS SET switch from null to a soft stop (either INCR or DECR) will change the display readout at 0.25 unit per second. Depressing the ΔV /EMS SET switch through a soft stop to a hard stop results in a change of 127.5 units per second. With the FUNCTION switch in the Vo SET, EMS TEST 1, and TEST 5 position, depressing the ΔV /EMS SET switch results in driving the EMS scroll. Depressing the ΔV /EMS SET switch to the soft stop drives the scroll at approximately 0.0164 inch per second (30 fps per second). Depressing through to the hard stop drives the scroll at approximately 0.263 inch per second (480 fps per second). The scroll mechanism puts a constraint on the reverse slewing of the scroll (ΔV /EMS SET switch INCR). The scroll may be slewed only one inch to the right after scroll slewings to the left of at least three inches.

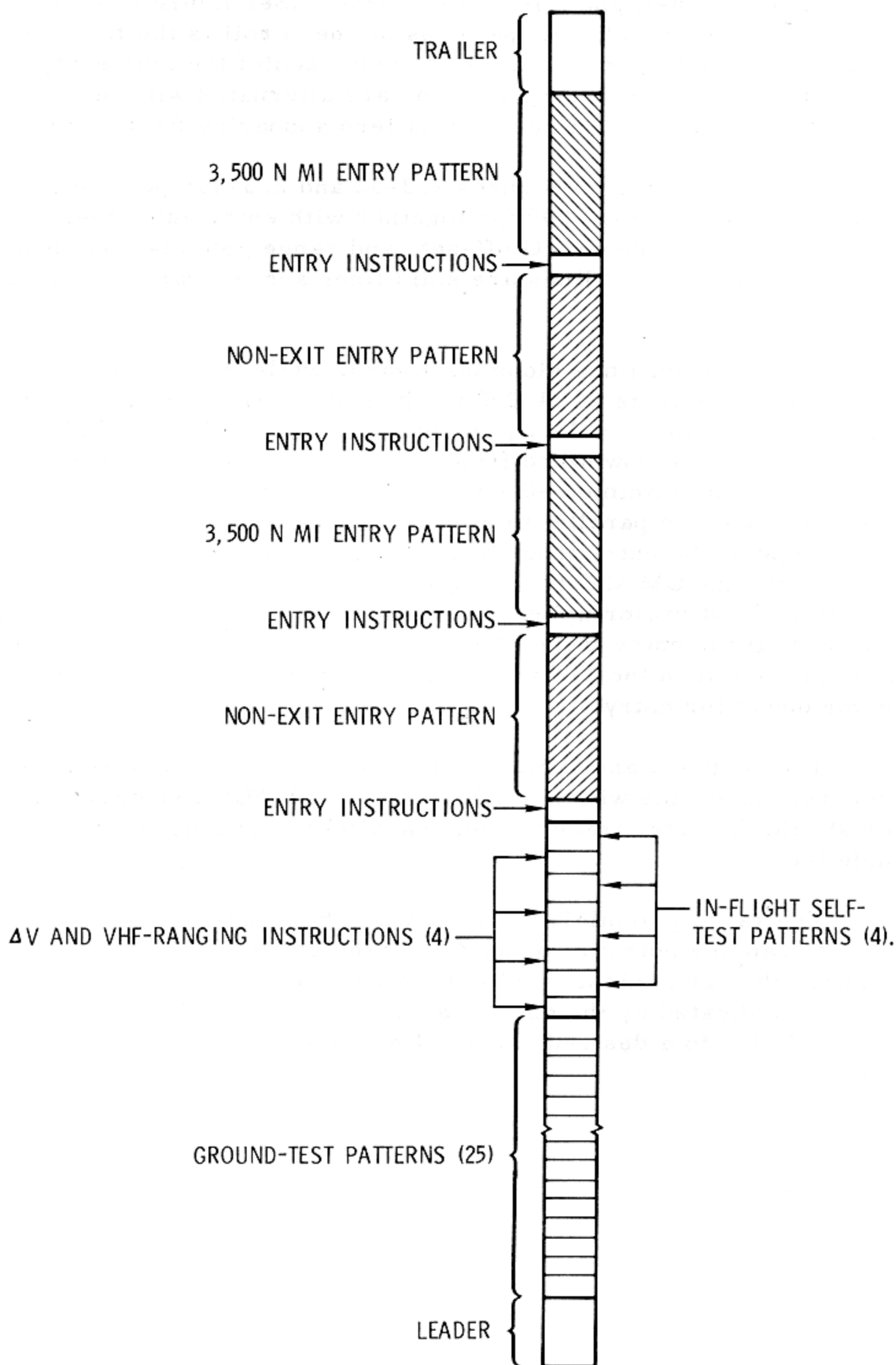
2.3.7.3.4 GTA Switch.

The GTA switch provides a ground test capability. With the coverplate removed, the GTA switch will be placed up to simulate 0G in the vertical stack configuration of the SC. An adjustment pot is available to calibrate 0G when the GTA switch is on and the EMS is operating. For the coverplate to be closed, the GTA switch must be off which removes the simulated 0G function for ground test.

2.3.7.4 Entry Scroll.

The EMS mylar scroll, contained in the EMS scroll assembly, contains four entry patterns together with entry in-flight test patterns and the instructions for entry, delta V and VHF ranging. (See figure 2.3-30.)

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Figure 2.3-30. EMS Scroll Format

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There are four sets of ΔV and VHF ranging instructions that are alternated with four entry in-flight self-test patterns. (See figure 2.3-31.) Following the fourth in-flight self-test patterns on the scroll is the first set of entry instructions. Entry instructions precede each of the four entry patterns. Lunar-return non-exit entry patterns are alternated with lunar-return 3500 NM exit patterns, a non-exit pattern appearing first on the scroll.

Each entry pattern (figures 2.3-32 and 2.3-33) has velocity increments from 37,000 to 4,000 fps together with entry guidelines. These lines are called G on-set, G off-set, and range potential guidelines. The G on-set and G off-set lines are solid lines and the range potential lines are broken.

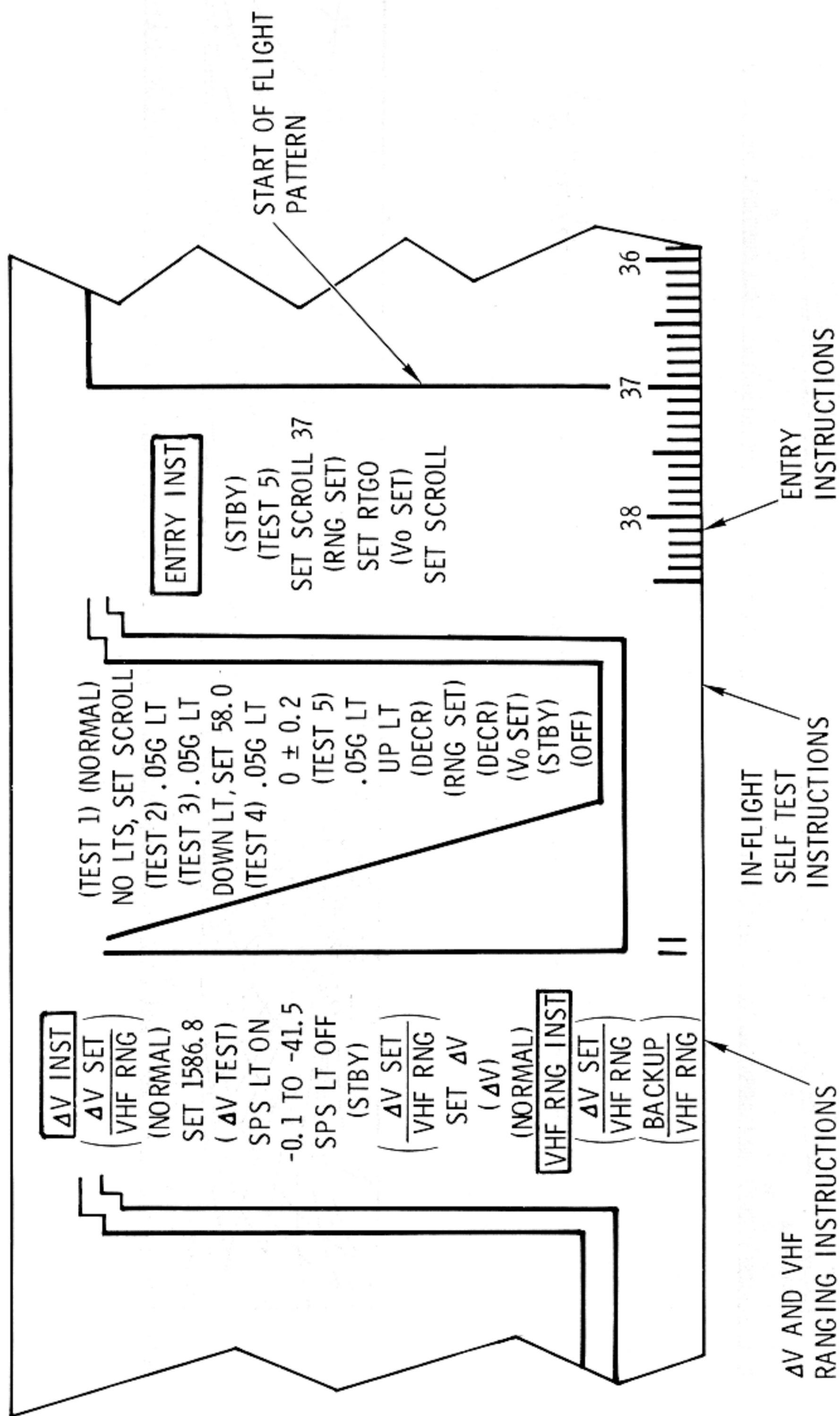
The G on-set lines slope downward, while the G off-set lines ray upward and terminate at 24,000 fps just to the right of the vertical line at 25,500 fps (minimum velocity for earth orbit). Below 24,000 fps the G on-set lines slope downward from the full-lift profile line which represents the steady-state minimum-G entry profile. During entry the scribe trace should not become parallel to either the nearest G on-set or G off-set line. If the slope of the entry trace becomes more negative than the nearest G on-set line, the CM should be oriented such that a positive lift vector orientation (lift vector up) exists in order to prevent excessive G buildup. However, if the entry trace slope becomes more positive than the nearest G off-set line then the CM should be oriented to produce negative lift (lift vector down) for entry.

The G on-set and G off-set lines are designed to allow a 2-second crew response time with a single system RCS/SCS 180-degree roll maneuver should the entry trace become parallel to the tangent of the nearest guideline.

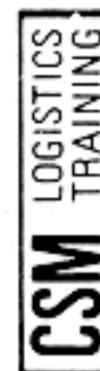
The range potential lines, shown in hundreds of nautical miles, indicate the ranging potential of the CM at the present G level. The crew will compare the range displayed by the range-to-go counter with the range potential indicated by the entry trace. The slope and position of the entry trace relative to a desired ranging line indicates the need for lift vector up or down.

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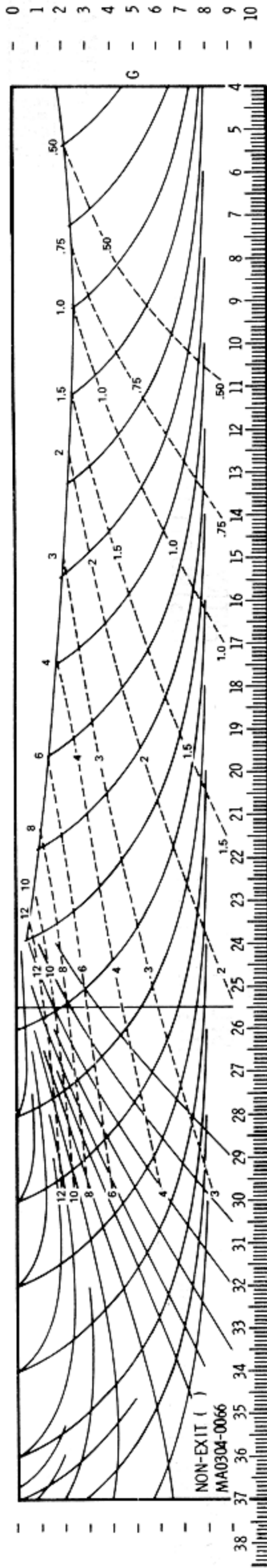


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Figure 2.3-61. EMS In-Flight Instructions for ΔV , VHF Ranging, Self-Test and Entry

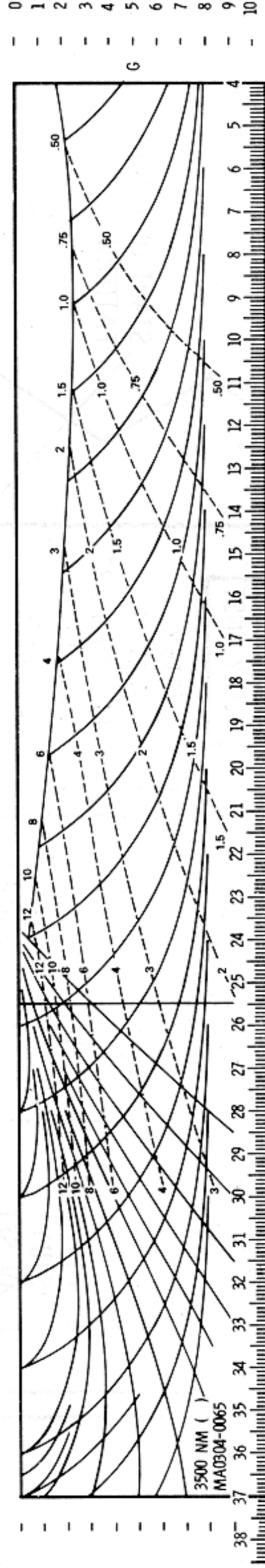
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Figure 2.3-32. EMS Lunar Non-Exit Range Limit Pattern (7/22/68)



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 LOGISTICS TRAINING
 CSM

Figure 2.3-33. EMS Lunar 3500 NM Range Limit Pattern (7/22/68)

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2.3.7.5 EMS Functional Data Flow.

The following functional discussion of the EMS relates system mechanization to the EMS operation. (See figure 2.3-34.)

2.3.7.5.1 Accelerometer.

The accelerometer, which is aligned to within ± 2 degrees of the SC X-axis, is the only sensor in the EMS. It has three outputs: low level G to threshold and corridor circuits, high level G to the flight monitor G axis during entry, and an output to the A/D converter which is used to drive the $\Delta V/RANGE$ display and mylar scroll. The difference in the low and high level G outputs is scale factor.

2.3.7.5.2 Threshold and Corridor Verification Circuits.

The threshold and corridor verification circuits use the accelerometer low level G output. The .05G comparator will trigger and illuminate the threshold light (.05 G) if a G level of $0.05G \pm 0.005G$ is present for 1 ± 0.5 seconds. If the G level drops to $0.02G \pm 0.005G$, the light will be extinguished. The corridor evaluation will occur 10.053 ± 0.025 seconds after the .05 G threshold lamp is illuminated. The lift vector up light will illuminate if the G force is greater than approximately $0.262 \pm 0.009G$. The lift vector down light will be illuminated if the G force is less than approximately $0.262 \pm 0.009G$. There will be only one corridor verification light turned ON for corridor evaluation. The corridor lights will be turned off when the flight monitor G axis drive passes the 2G level.

2.3.7.5.3 Scroll Assembly G Axis Drive Circuits.

The scroll assembly G axis drive circuits receive the accelerometer high G level output signal and position the G axis scribe in vertically. The scribe drive is a normal closed-loop servo circuit with velocity and position feedback. The loop is biased from zero by the magnitude of the accelerometer input.

2.3.7.5.4 Scroll Assembly Velocity Axis Drive Circuits.

The scroll assembly velocity axis drive circuits use the accelerometer A/D converter output to drive the scroll from right to left. The A/D converter output is about one pulse for each 0.1 fps of velocity change. The motor control circuits and stepper motor cause the scroll to move from right to left and the present inertial velocity is read on the scroll. Before entry scroll is initialized to the inertial velocity by setting the FUNCTION switch to the Vo SET position and using the $\Delta V/EMS$ SET switch to slew the scroll to the predicted inertial velocity value at 0.05G.

STABILIZATION AND CONTROL SYSTEM

SYSTEMS DATA

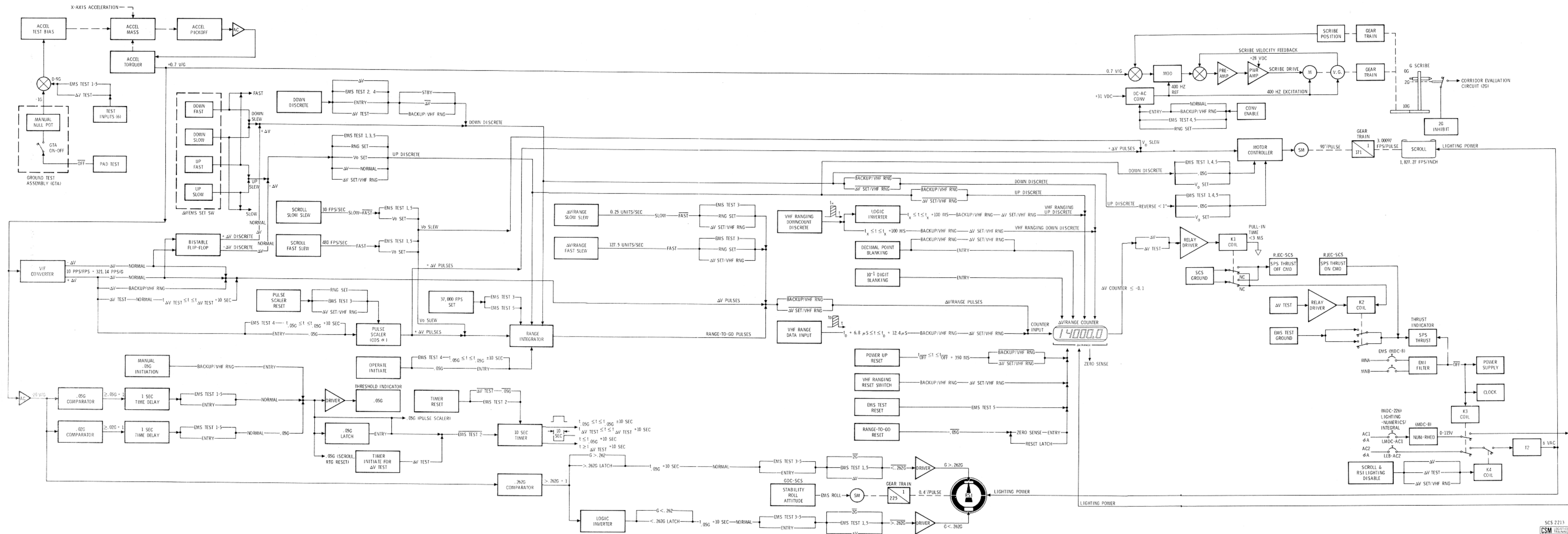


Figure 2.3-34. EMS Functional Block Diagram

STABILIZATION AND CONTROL SYSTEM

SYSTEMS DATA

2.3.7.5.5 ΔV /RANGE Display Circuits.

The ΔV /RANGE electronics directly controls the numeric display value except during VHF ranging operations. The display will be initialized by a combination of the FUNCTION switch and ΔV /EMS SET switch except during VHF ranging operations. During ΔV operations, the accelerometer A/D converter output pulses are used to increment or decrement display value. When the display decreases to a value of -0.1 fps, a signal is supplied to the SCS for an automatic SCS control SPS OFF command. For entry, the display will read range to go, being decremented by the range integrator. The output of the range integrator will decrease as a function of the inertial velocity stored in it at any time. The range integrator is decremented to that it contains the CM present inertial range-to-go if properly initialized. The divider network sends pulses to the flight monitor velocity axis drive in order to drive the scroll from right to left after 0.05G is sensed. If the 0.05G function should fail, placing the MODE switch to the BACKUP/VHF RNG position will initiate the divider network operation to drive the range-to-go display and the flight monitor scroll from right to left as a function of G level.

2.3.7.5.6 Roll Stability Indicator Drive.

The RSI drive function, controlled by the yaw axis of the GDC in the SCS, requires the correct positive of the two ENTRY switches (.05G and EMR ROLL) for its correct operation during entry. This function is described as a normal GDC function in paragraph 2.3.3.2.

2.3.7.5.7 Thrust-Off Function.

The thrust-off function will provide a logic function for a SCS thrust-off command any time the ΔV /RANGE counter goes to -0.1 fps. During a delta V mode operation, a relay energizes and provides a ground to the SCS. This function operates in conjunction with the ΔV and ΔV TEST positions of the FUNCTION switch.