APOLLO TRAINING

APOLLO
STABILIZATION AND CONTROL SYSTEM

BLOCK II
DETAILED TRAINING PROGRAM

MAY 1969

FOR TRAINING PURPOSES ONLY

CSM LOGISTICS TRAINING

QUESTIONS RELATIVE TO THE CONTENTS OF THIS DOCUMENT
SHOULD BE DIRECTED TO INSTRUCTORS:

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PREFACE

This training manual provides a means of familiarizing the reader with the Block II Stabilization and Control System (SCS). Its primary purpose is not to provide a detailed circuit analysis, but to instead convey a functional description of the system's capabilities. To achieve this objective, functional diagrams are used to depict major signal flow paths.

The text most closely describes the Block II SCS configuration for spacecraft 103 and subsequent. Since this document is not subject to revision control, the subject matter should be considered for training purposes only.

The following documents represent the reference material utilized in the preparation of this familiarization manual. It is recommended that the reader utilize these items as a supplement to the text in order to obtain additional detailed system information.

YG974N	System Technical Development Specification	(TDS 1401)
BG 287	Reaction Jet/Engine On-Off Control TDS	(TDS 1402)
BG 286	Control Electronics Assembly TDS	(TDS 1403)
BG 285	Electronic Display Assembly TDS	(TDS 1004)
BG 289	Gyro Display Coupler Assembly TDS	(TDS 1005)
BG 288	Thrust Vector Position Servo Amplifier Assembly TDS	(TDS 1006)
GG 362	Gyro Assembly TDS	(TDS 1007)
JG 264	Flight Director Attitude Indicator TDS	(TDS 1008)
JG 261	Gimbal Position/Fuel Pressure Indicator TDS	(TDS 1009)

CG 161 Attitude Set Control Panel TDS	(TDS 1010)
CG 160 Translation Control TDS	(TDS 1011)
CG 166 Rotation Control TDS	(TDS 1412)
GG 248C Miniature Integrating Gyro TDS	(TDS 1013)
Servometric Meter Assembly TDS	(TDS 1016)
YG974N1 Interconnecting Wire List	H5986-1
Block II SCS System Schematic	C13373AA01
Block II Phasing Requirements	DR 9249
BG 287 H3 RJ/EC Schematic	C13052-4
BG 285 G6 EDA Schematic	C13064-2
BG 289 G3 GDC Schematic	C13068-2
BG 286 H1 ECA Schematic	C13065-4
BG 284 G4 TVSA Schematic	C13067-1
GG 362 G4 GA Schematic	C13069-1
JG 261 G4 GP/FPI Schematic	C12964-2
JG 264 G8 FDAI Schematic	C12968-3
CG 161 G4 ASCP Schematic	C12959-11G
CG 166 H3 RC Schematic	C13365AA01
CG 160 G6 TC Schematic	C12961-1

INDEX

INDEX

INDEX

INDEX

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS

ix

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS

ABBREVIATIONS USED IN TEXT AND DIAGRAMS

A	Analog (Telemetry Test Point)
A & C	Quads "A & C" of The Service Module Reaction Control System
AC	Alternating Current
AC EXC	Ac Excitation
ACCEL CMD	Acceleration Command
AMP	Amplifier
ARS	Attitude Reference Subsystem
AS	Attitude Set
ASCP	Attitude Set Control Panel
ATT	Attitude
B & D	Quads "B & D" of the Service Module Reaction Control System
BMAG	Body Mounted Attitude Gyro
B.O.	Breakout (Signal created at closure of rotation control switches)
B.O.P.	Breakout Pitch (Signal created at closure of the rotation control pitch breakout)
BUR	Backup Rate
CONV	Converter
CW	Clockwise (Logic signal from rotary motion of translation grip)
$\overline{CW}$	"Not" clockwise (Logic signal from rotary motion of translation grip)
CCW	Counterclockwise (Logic signal from rotary motion of translation grip)
CDU	Coupler Data Unit (Guidance and Navigation Equipment)
CG (c.g.)	Center of Gravity

CKTS	Circuits
CM	Command Module
CMC	Command Module Computer
CMD(S)	Command(s)
CNTL	Control
COS	Cosine
CSM	Command and Service Modules
DB	Deadband
D.C.	Direct Current
DEMOD	Demodulator
DIR SW	Direct Switch (in rotation control)
DISP	Display
D/S	Demodulator-switch
DSKY	Display Keyboard - CM (Command Module Computer)
E	Event (Telemetry Test Point)
ECA	Electronic Control Assembly
EDA	Electronic Display Assembly
EMS	Entry Monitor System
ENG	Engine
EX or EXC	Excitation
FB	Feedback
FDAI	Flight Director Attitude Indicator
FPI	Fuel Pressure Indicator
FREQ DIV	Frequency Divider
FS	Fail Sensed (Logic signal generated from SPS gimbal motor current sensor)

F1	FDAI #1 - Commanders Control Panel Location
F2	FDAI #2 - Navigators Control Panel Location
GA	Gyro Assembly
GDC	Gyro Display Coupler
GMBL	Gimbal
G&N	Guidance and Navigation
GN&C	Guidance, Navigation, and Control
GP/FPI	Gimbal Position and Fuel Pressure Indicator
GPI	Gimbal Position Indicator (portion of GP/FPI)
GSE	Ground Support Equipment
HI	High side of signal flow line
HW	Half Wave (Demodulator)
IC	Integrated Circuit
IGN_2	"Ignition 2" logic signal
IMU	Inertial Measurement Unit
INV	Inverter
IU	Instrument Unit of S-IVB Booster
LEM	Lunar Excursion Module (renamed Lunar Module)
LES	Launch Escape System
LEV DET	Level Detector
LM	Lunar Module
LO	"Low" side of signal flow line
LOI	Lunar Orbit Injection (Retrothrust)
MAX	Maximum
MIN	Minimum

MIN IMP	Minimum Impulse
MIT	Massachusetts Institute of Technology
MN A	Main Bus A (Spacecraft 28vdc)
MN B	Main Bus B (Spacecraft 28vdc)
MOD	Modulator
MS	Milliseconds
MTVC	Manual Thrust Vector Control
MVG	Motor - Velocity Generator
NASA	National Aeronautics and Space Agency
NC	Not Connected (open) or normally closed (contact of relay)
NR	North American - Rockwell
OX	Oxidizer
PGNCS	Primary Guidance, Navigation & Control System
POS	Position
PSI	Pounds per Square Inch
PWR	Power
P, Y, R	Pitch, Yaw, Roll (axis or channel)
RAI	Roll Attitude Indicator
RC	Rotation Control
RC	Control Resolver
RCS	Reaction Control System
RJC	Reaction Jet Control
RJD	Reaction Jet Driver
RJ/EC	Reaction Jet & Engine ON-OFF Control Assembly
RLVDT	Rotary Linear Variable Differential Transducer

RMS	Root Mean Squared
RX	Transmitter Resolver
S/C	Spacecraft
SCS	Stabilization and Control System
SD	Solenoid Driver
SEC	Second
SEL	Select
SEQ	Sequencer
S&ID	Space and Information Division (of NR)
SIG	Signal
SIN	Sine function
SM	Service Module
SMRD	Spin Motor Rotation Detector
SMRCS	Service Module Reaction Control System
SPS	Service Propulsion System
SUM AMP	Summing Amplifier
SW (sw)	Switch
TC	Translation Control
TCA	Temperature Control Amplifier
TEI	Transearth Injection
TIA	Temperature Indicating Amplifier
TM	Telemetry
TVC	Thrust Vector Control
TVSA	Thrust Vector Servo Amplifier (Assembly)
V/F	Voltage to Frequency Converter

VDC	D.C. Voltage
VG	Velocity Generator
W/RD	With Rate Damping (MTVC)
WO/RD	Without Rate Damping (MTVC)

SECTION 1

APOLLO MISSION AND SPACECRAFT

1.1 **INTRODUCTION**

Honeywell has been designated, under contract from North American Rockwell (NR), to design, develop, and manufacture the stabilization and control system (SCS) for the Apollo command and service modules (CM/SM). Honeywell is therefore responsible for the achievement of specified system performances based upon North American supplied mission and interface requirements, along with data defining the spacecraft vehicle dynamics. The basic definition of the stabilization and control system includes a spacecraft "autopilot" subsystem, manual control, and associated cockpit displays.

1.2 **APOLLO MISSION**

A profile of the Apollo mission is shown in Figure 1.1. The primary objective of the Apollo mission is to land a manned vehicle on the lunar surface and safely return the crew to Earth. This ultimate goal will be accomplished using Block II Apollo vehicles. These vehicles will use electronic systems (including the SCS) which have been redesigned and feature advances in the state-of-the-art learned from the earlier Block I vehicles.

The lunar spacecraft in Project Apollo is being developed under direction of NASA's Manned Spacecraft Center at Houston, Texas. The spacecraft has three main elements: a command module (CM), a service module (SM), and a

LUNAR LANDING MISSION PROFILE

MOON AT EARTH LANDING

APOLLO-LEM RENDEZVOUS

LEM ASCENT

TRANSEARTH INJECTION (TEI) ⑦

LUNAR ORBIT ⑥

LEM DESCENT

LUNAR ORBIT INJECTION (RETROTHRUST) (LOR) ⑤

MOON AT EARTH LAUNCH

TRANSEARTH MIDCOURSE ⑧

EARTH PARKING ORBIT ②

EARTH APPROACH

TRANSLUNAR MIDCOURSE ④

ASCENT TO ORBIT ①

ENTRY ⑨

PACIFIC RECOVERY AREA

TRANSLUNAR INJECTION ③
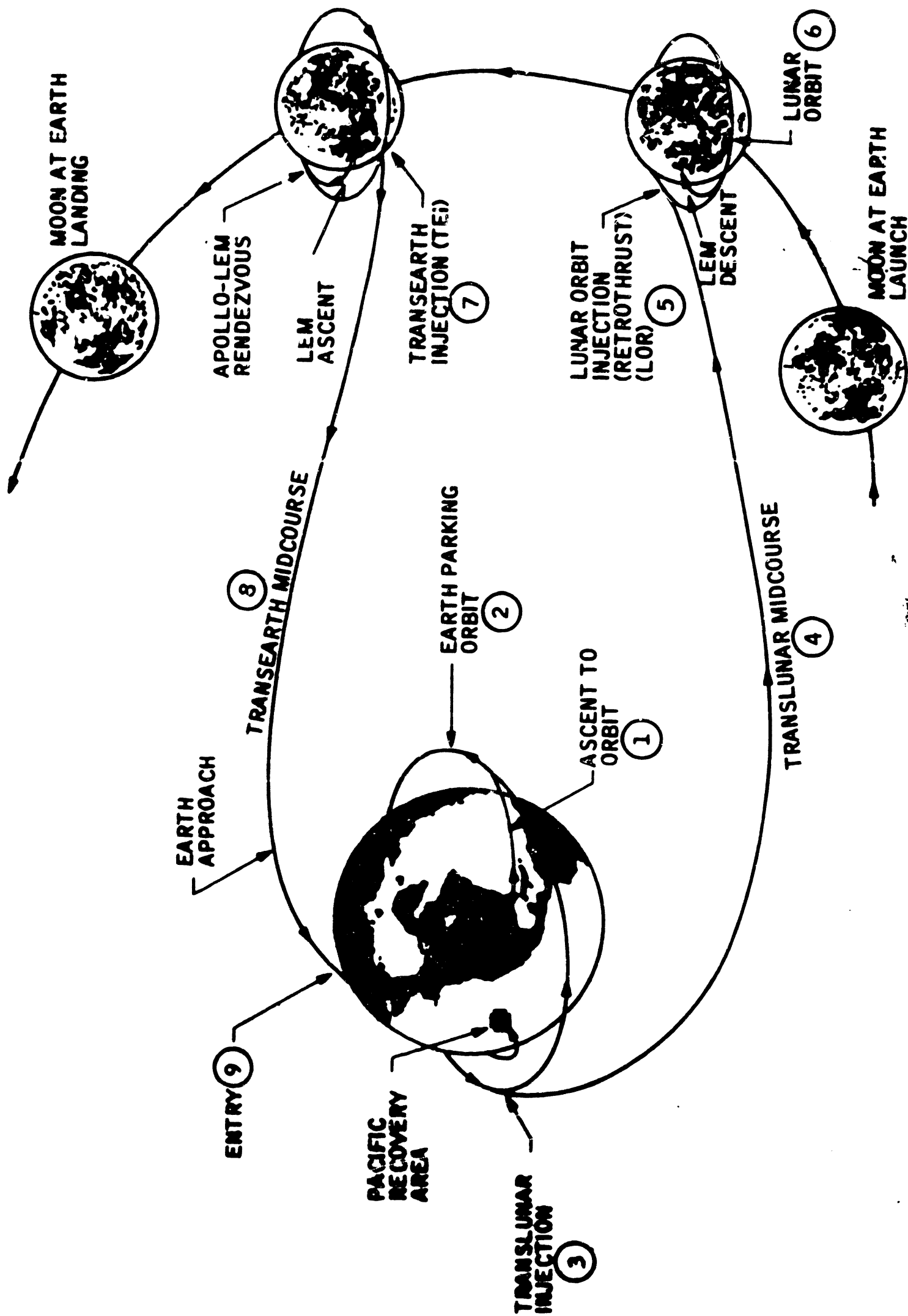
lunar module (LM). The command module carries the three-man crew, plus guidance and control instrumentation. The service module contains instrumentation (to which access by the crew is not necessary during flight) and the primary spacecraft propulsion system. The lunar module is the only part of the spacecraft that lands on the moon. These interface stages of the Apollo vehicle are shown in Figure 1.2.

The 3,000-ton Saturn V will be launched from the Kennedy Space Center, Florida. The first (S-IC), second (S-II) and third (S-IVB) stages are fired in succession to place the third stage and the Apollo spacecraft into a "parking" orbit around the Earth. The first and second stages are jettisoned after cutoff. The launch escape tower is also discarded just after second stage ignition.

After the spacecraft has been checked out in earth orbit, the third stage (S-IVB) is restarted, boosting the vehicle to near escape velocity (about 25,000 miles per hour). The command plus service modules separate after the adapter shroud surrounding the lunar module opens. The command module then docks with the lunar module, and the spent third stage is jettisoned.

Earth's gravity will slow the spacecraft's speed to about 6,500 mph by the end of the first day, and to about 1,500 mph after two days. This portion of the flight is known as the "midcourse" or "translunar coast" phase. As the spacecraft approaches the moon, the large propulsion unit on the service module is fired to slow the Apollo plus LM into a precise circular orbit about 80 miles above the moon's surface.

APOLLO SPACECRAFT Stage Interfaces (Boost Phase)

LES

CM

SM

LEM ADAPTER

LEM

INSTRUMENT UNIT

S-IVB STAGE

1-4
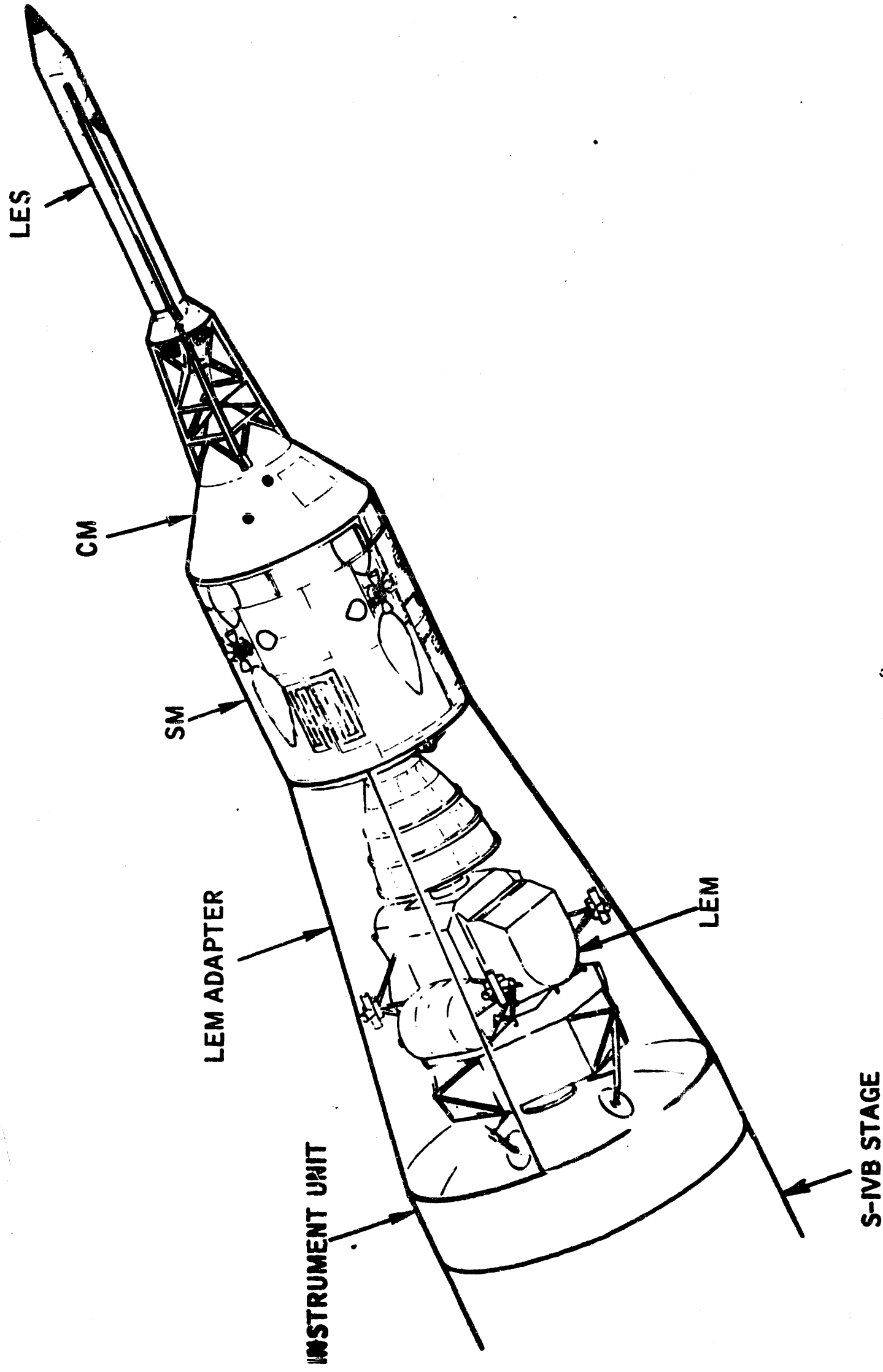
Two astronauts then crawl through the egress hatch of the command module into the lunar module. The LM engine is then fired and the LM begins a descent trajectory to the touchdown site. The Apollo CM/SM remains in the circular lunar orbit.

A large glass cockpit area in the LM allows the two astronauts to have a line of sight view of the touchdown site. With the descent engine firing and landing gear extended, the vehicle automatically descends to within 100 feet of the lunar surface. The vehicle will also be able to hover or move laterally for about 1,000 feet for choosing the best touchdown point.

After the lunar landing, the LM is checked out for the lunar take-off. Only then does exploration of the moon begin. Most of this exploration will be geologic in nature. It will include mapping, photography, observation of surface characteristics, core and surface sampling, and seismic and radiation measurements.

For the return trip to the orbiting Apollo, the LM utilizes the ascent engine. The first LM stage, containing the descent engine, acts as the launch pad and remains on the moon. The orbiting spacecraft (CM/SM) containing the third astronaut, will be above the moon's horizon when the ascent stage of the lunar module is launched. Radar and visual contact are maintained between the two converging vehicles, and final docking will be made under manual control with the LM the active vehicle.

After docking, the two LM astronauts transfer back to the command module.

The lunar module is jettisoned and remains in lunar orbit. Following checkout of the spacecraft, the propulsion system of the service module is restarted, injecting the command and service modules into a trans-earth trajectory. During the trans-earth coast phase the spacecraft velocity is increased by earth's gravitational pull and several mid-course corrections are made if necessary. The service module is jettisoned just before entry into the earth's atmosphere.

The command module is then oriented with heat shield forward for entry. Earth approach must be on a very precise trajectory, to encounter the earth's atmosphere and be "captured" for a safe entry. Too shallow an entry approach would result in an earth miss (skip-out) and too steep an entry would result in excessive G's or heat build up on the command module.

Traveling at approximately 25,000 miles an hour, the module thus ideally enters the atmosphere at a critical entry angle at approximately 400,000 feet. It encounters a heating rate several times higher than those encountered during project Mercury and Gemini entries.

The blunt section (aft heat shield) of the command module ablates as the heat builds up due to the aerodynamic forces. The spacecraft is thus slowed down by this dissipation of energy. Then a sequence of chute deployments begins. A drogue chute first slows the module, and then at 10,000 feet, the main chutes open to bring it to a safe water landing. Earth based radar and optical instruments will track the entry descent, and helicopter recovery teams will cover the crew and command module.

1.3 APOLLO LAUNCH VEHICLES

The launch vehicles used to propel Apollo spacecrafts into space are shown in Figure 1.3.

The Saturn IB is used to launch earth orbiting manned Apollo vehicles. It includes an S-IB and an S-IVB stage.

The Saturn V is able to place more than 120 tons into earth orbit or send more than 45 tons to the moon. The Saturn V, with its Apollo payload is 361 feet tall. Physical and performance characteristics of the stages are as follows.

The S-IC first stage burns over 15 tons of propellants per second during its two and one-half minutes of operation to take the vehicle to a height of about 36 miles and to a speed of about 6,000 miles per hour. The stage is 138 feet long and 33 feet in diameter. It is powered by five F-1 engines with a combined thrust of 7.5 million pounds burning RP-1 kerosene and liquid oxygen. The fueled weight of the stage is 5,028,000 pounds.

The S-II second stage burns over one ton of propellants per second during about six and one-half minutes of operation to take the vehicle to an altitude of about 108 miles and a speed of near orbit velocity (17,400 miles per hour. The stage is 81 feet long and 33 feet in diameter. It is powered by five J-2 engines with a combined thrust of one million pounds burning liquid hydrogen and liquid oxygen. The fueled weight of the stage is 1,064,000 pounds.

APOLLO SPACE VEHICLES

APOLLO SPACECRAFT

LEM ADAPTER

INSTRUMENT UNIT

S-IVB BOOSTER

S-I BOOSTER

224 FT

SATURN IB

LES

CM

SM

APOLLO SPACECRAFT

LEM ADAPTER

INSTRUMENT UNIT (IU)

S-IVB STAGE

S-II BOOSTER

S-IC BOOSTER

61 FT

SATURN V
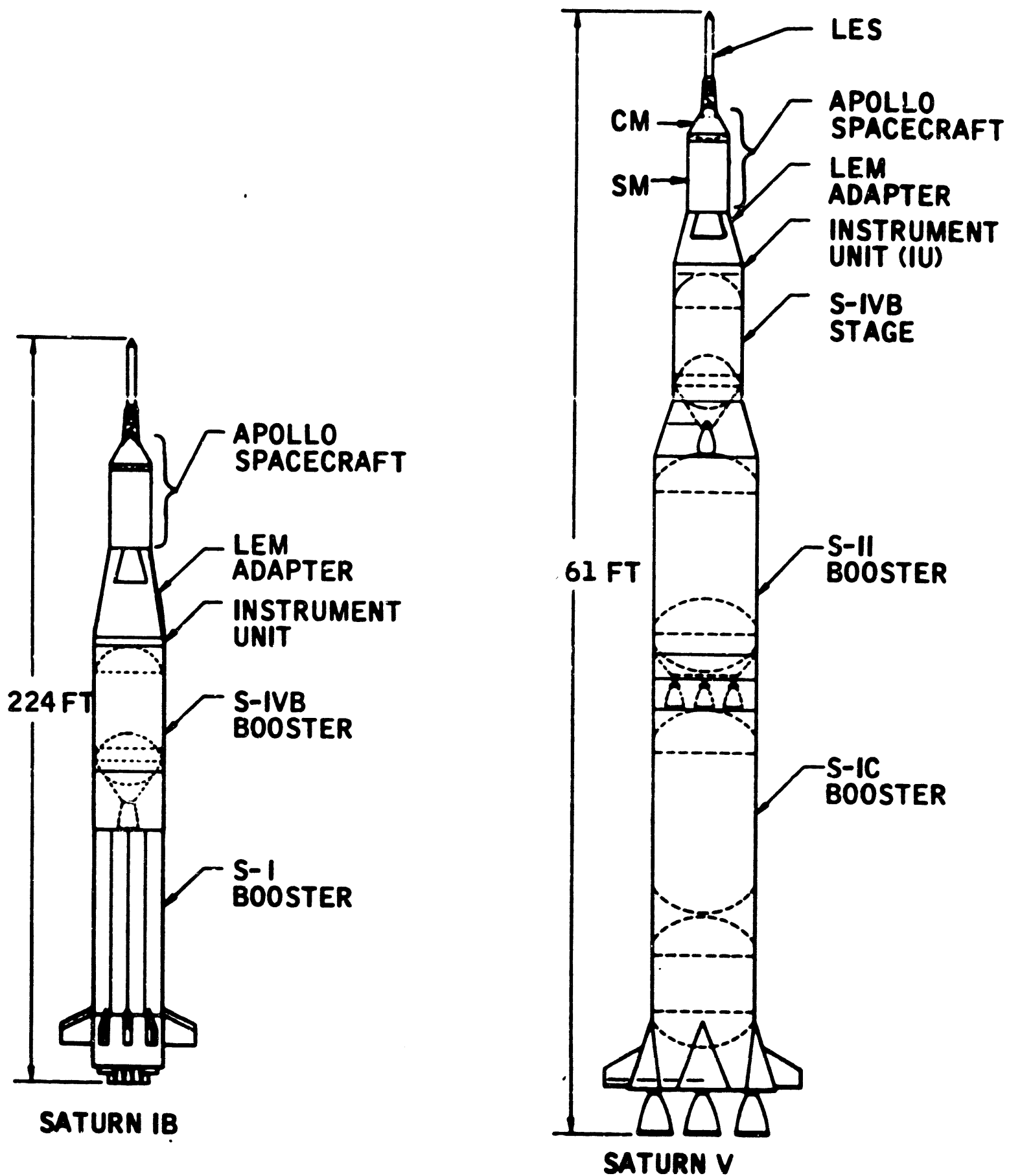
FIGURE 1.3

The S-IVB third stage has two important operations during the Apollo lunar mission. After the second stage drops away, the third stage ignites and burns for about two minutes to place itself and the Apollo spacecraft into the desired earth orbit. At the proper time during this earth parking orbit, the third stage is re-ignited and burns for about six minutes to place the spacecraft on a lunar trajectory at a velocity of 24,900 miles per hour. The stage is 58 feet long and 21.7 feet in diameter. It is powered by a single J-2 engine with a thrust of 200,000 pounds. The fueled weight of the stage is 265,000 pounds.

The Instrument Unit, located atop the third stage, contains the guidance and control equipment for the launch vehicle. It is 3 feet long, 21.7 feet in diameter and weights about 4,100 pounds.

The Apollo Spacecraft consists of the CM/SM and LM. The command module is 12 feet high and 13 feet in diameter and weighs 52,000 pounds fully fueled. The SM contains a 22,000 pound thrust engine. The Lunar Module consists of two stages and has a total weight of 32,000 pounds. The ascent engine develops 3,500 pounds thrust. The descent engine thrust can be varied from 1,050 to 10,500 pounds.

The Launch Escape System is 34 feet long. It contains a 150,000 pound thrust launch escape engine, a 33,000 pound thrust tower jettison engine and a 3,000 pound thrust pitch control engine. Normally the LES is jettisoned about three minutes after lift off. For an abort, the LES would carry the CM up and away from the launch vehicles path.

SPACECRAFT ENGINE AND AXES DEFINITION

Figure 1.4 provides a reference for describing and defining the spececraft reference axes and rotations, gimbal angles and RCS engine functions and nomenclature.

Reference Axes

The reference axes of the spacecraft are orthogonal and are identified as follows: the X-axis is parallel to the nominal launch axis of the spacecraft, and is positive in the direction of initial flight. The Y-axis is normal to the X-axis, and positive to the right of a crewman facing in the + X direction. The Z-axis is normal to both the X and Y axes, and positive in the direction of the crewman's feet.

Rotations

Changes in attitude of the vehicle are defined relative to rotations about the reference axes. Positive rotations are defined as clockwise motions about the reference axes when looking from negative to positive along the axis. Vehicle rotations about the spacecraft X-axis are identified as roll. Roll angular displacements are identified by the symbol "ϕ" and angular velocities by "p" or "$\dot{\phi}$". Pitch angular displacements (θ) are those about the Y-axis. Pitch angular velocities are identified by "q" or "$\dot{\theta}$". Yaw angular displacements and velocities are generated relative to the Z-axis and are identified by "ψ" and "r" or "$\dot{\psi}$" respectively.

Gimbal Angles

The SPS engine is mounted in a double gimbal ring. The engine can be

RCS ENGINE FUNCTIONS

FIGURE 1.4

1-11

deflected about the gimbal hinge points by providing proper command signals to the servo actuators. The gimbal hinge points define lines which are parallel to the spacecraft Y and Z axes. Thus the orientation of the SPS engine is defined by pitch and yaw gimbal angles which are deflections about the respective lines from the nominal engine position. Gimbal angle polarities are defined in the same manner as spacecraft angular displacement polarities as indicated in Figure 1.4.

RCS Engine Nomenclature and Functions

The Service Module contains four clusters (quads) of RCS engines for a total of 16 engines. These engines were originally identified by numbers 1 through 16. As indicated in Figure 1.4, the engines are now identified by quad (A, B, C or D) and number (1, 2, 3 or 4). A translation along any S/C axis or combination of axes can be accomplished by firing the proper combination of RCS engines. Similarily, rotations about any axis or combination of axes can be controlled. The SM RCS Engines table on Figure 1.4 identifies the engines which would be turned ON to accomplish the desired motion. For example: a positive pitch acceleration ($+\ddot{\theta}$) would be provided by firing engines C3 and A3. An upward translational acceleration (along the -Z axis) would be provided by firing the B2 and D1 engines.

The Command Module contains two sets of 6 RCS engines each. The engines for each set are numbered 1 through 6. The first numeral of the engine identifying number designates system 1 or 2. The two systems have independent propellant supplies. The 12 RCS engines provide the CM with rotational

control capability about all three axes. There is no requirement for translation capability with the Command Module.

SECTION 2

INTRODUCTION TO THE SCS

2.1   **GENERAL**

The Stabilization and Control system (SCS) provides a capability for controlling spacecraft rotation, translation and thrust vector control. The SCS also provides the displays necessary to allow the astronauts to monitor automatic operation as well as to exercize optimum control of the spacecraft.

Functionally, the SCS can be divided into three basic subsystems: attitude reference, reaction jet control and thrust vector control. These subsystems contain the elements which provide selectable functions for display, automatic and manual attitude control, manual translation, control, and automatic and manual thrust vector control. All control functions are backup to the primary guidance navigation and control system (PGNCS). The SCS provides two assemblies for interface with the propulsion subsystems; these are common to the SCS and PGNCS for all control functions. The main display and control panels contain the switches used in selecting the desired display and control functions.

The Block II SCS utilizes "functional switching" as opposed to the "mode select" switching mechanized in Block I. Functional switching requires manual switching of numerous independent panel switches in order to configure the SCS for various mission functions (e.g., midcourse, ΔVs, entry, etc.). Mode switching would, for example, employ one switch labeled

"midcourse" to automatically accomplish all the necessary system gain
changes, signal enables and disables, etc. for that mission phase.
Thus mode selection simplifies the crew tasks involved, but limits
system flexibility between various mode configurations. 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., use of the SCS to control the vehicle
while the G&N supplies the display information).

2.2   BASIC SCS FUNCTIONS

The SCS system provides the capability for 3-axis attitude and/or rate
control of the Apollo vehicle.

Switching is provided so that either the G&N system or the SCS has the
ability to control the vehicle and provide attitude information for display.
Flexibility is provided by control panel switching so the pilot may choose
a number of alternate parallel success paths involving the SCS and the
G&N system.

During Delta-V maneuvers, the attitude of the vehicle is controlled by
positioning the gimbals of the SM thrust engine in pitch and yaw and
operating reaction jets in roll. In addition to providing the interface
for G&N control, the system provides the capability for complete SCS
attitude control of the vehicle during thrusting, and on-off control of

the thrust engine. Redundant sets of Body Mounted Attitude Gyros (BMAGs) provide a SCS attitude reference while the velocity increment is measured by the ΔV counter of the NR supplied EMS unit. Additional operational capabilities allow the astronaut to assume manual control of the vehicle during thrusting (MTVC).

During the non-thrusting mission phases, three-axis rotation control is accomplished using reaction jets in all axes. Here again, a complete redundant attitude control capability is provided by SCS sensors and electronics. In addition, a three-axis rotation and translation control capability is provided through the use of the SCS manual controls.

System displays include the presentation of total vehicle attitude from a given reference, attitude error, and maneuvering rate. In addition, other displays show SPS gimbal position, attitude set, and a redundant roll angle display during entry.

2.3   BASIC SCS REQUIREMENTS

In June of 1964, NASA directed the program definition phase of the Block II integrated guidance and control system.

As part of the Block II philosophy, MIT was given responsibility for primary guidance and control of the Apollo vehicle whenever the IMU is operating (except when the Apollo vehicle is attached to the Saturn launch vehicle and during tower aborts.) NAA (with Honeywell as subcontractor) was assigned backup stabilization and control whenever the G&N system is not operating and during tower aborts. Other guidelines for the Block II

system included:

1. No inflight maintenance requirement.

2. Provide flexible manual TVC capability.

3. Provide a capability to hold vehicle attitude in two axes while maneuvering in the third.

4. Incorporate function select switching.

5. Selection of back-up rate will not destroy SCS attitude reference.

The program definition phase covered the period from June 4 to August 18, 1964. This was a period of cooperative effort between NAA, MIT, and NASA to determine the feasibility of the Block II approach, establish requirements in the interface areas and detail the mechanization.

As a backup to the primary GN&C system the SCS provides stabilization and control of the spacecraft for rotation, translation, and SPS thrusting using the command and service module reaction engines and the SPS engine gimbal servo; and it provides the required displays and controls to permit necessary crew interface with the controlling elements. The system is designed for use in backup to, and for interfaces with the primary vehicle guidance and control system.

2.3.1   <u>Automatic Control</u>

The capability for attitude hold in any or all three axes to within the limits of the system deadband, using the SM reaction engines (or the CM engines for the command module - only configuration) is required from S-IVB separation up to 0.05g (entry). Attitude hold capability is provided in axes not being maneuvered in, when in rate command mode. It is not

required in an acceleration command mode.

Body bending filters for SCS TVC are also included in the SCS. These filters are used with the SCS automatic TVC. TVC also can be accomplished by the G&N system using its own filters and supplying guidance commands directly to the SCS servo electronics package.

The system is also capable of being switched from G&N to SCS during SPS thrusting without shutting off the SPS engine. The G&N system provides its own automatic SPS engine gimbal trim. The G&N system outputs to the SCS are analog SPS gimbal position commands and discrete on/off commands to the SPS driver amplifiers for engine ignition control and to the RCS driver amplifiers for attitude, rate and minimum impulse control.

2.3.2   Manual Control

The SCS manual rotation control provides the following types of maneuvering:

a. Rate command

b. Acceleration command (at breakout switch positions)

c. SCS minimum impulse command

d. Direct acceleration control (near the stick hard stops)

Normal manual SMRCS maneuvering rates will probably not exceed 0.85 deg/sec, though the capability for crew selection of higher rate maneuvering does exist.

The capability for performing manual maneuvers thru the G&N by using the SCS manual rotation control is provided; however, SCS proportional rate control inputs to the G&N system are not provided.

Manual thrust vector control (MTVC) is provided only by the SCS and is selectable by axis in pitch and yaw.

Command module maneuvering capability in all three axes is provided up to 0.05g, and roll maneuvering plus roll, pitch and yaw rate damping is provided after 0.05g. Roll-rate capability includes maximum possible rates (50°/sec) for a 180-degree maneuver; that is, accelerate continuously for 90 degrees, then decelerate continuously for 90 degrees.

While in an acceleration command mode the vehicle is in free drift, with no commands present. Also, the G&N system manual minimum impulse controller does not directly interface with the SCS.

2.3.3   <u>Reliability Requirements</u>

The SCS is designed such that no single failure results in the loss of spacecraft or crew. Here it is assumed that, for survival-required functions, redundancy is required either through overlapping G&N - SCS functions or mechanization within the SCS where applicable.

It is assumed that the mission success reliability of the SCS is to be consistent with achievement of the spacecraft electronics system reliability objective of R = 0.984.

The SCS is designed such that elemental failures do not propagate (cause additional failures) within the system.

The pilot can provide attitude control manually in any mission phase if required. This eliminates the requirement for a redundant automatic attitude error source.

Control electronics redundancy is not employed except for the SPS TVC servo, and mission duty cycle is limited to meet the electronics system reliability objective.

The gimbal position display and SPS clutch servo electronics are redundant.

NAA supplied functional switching has been incorporated to ensure that no single element failure in the switching network shall preclude the ability to select an alternate configuration necessary for survival.

2.4   SCS INTERFACES

The SCS interfaces with the following systems:

Telecommunications System - The SCS provides discrete and analog signals to the TM system which indicate system status as well as various performance parameters. This information is fed to the ground stations via down-link telemetry.

Electrical Power System - The Electrical Power System provides the primary power for SCS operation. The SCS devices utilize 28 VDC power from the MN A and MN B busses as well as 115 VAC 400 hz three-phase and single-phase power from the AC 1 and AC 2 busses.

Environmental Control System - The water-glycol solution from the Environmental Control System is fed thru the cold plates on which the SCS devices are mounted. This enables heat to be transferred from the SCS assemblies.

Sequential Events Control System - The SECS provides abort switching and receives manual abort switch closure from the SCS. The SECS also provides enable-disable control for the SCS reaction control drivers.

Entry Monitor System - The SCS provides signals to the EMS which drive
the Roll Attitude Indicator to display spacecraft roll attitude during
the entry phase of the mission. The EMS ΔV counter provides a discrete
SPS engine off command to the SCS to end a thrust maneuver.

Primary Guidance Navigation and Control System - The SCS interfaces with
the Inertial Measuring Unit (IMU), the optical and inertial Coupler Data
Units (CDU), the Command Module Computer (CMC) and the Inverter of the
PGNCS. The IMU provides pitch, yaw and roll total attitude information
to the SCS for display. Attitude error signals are provided to the SCS
via the inertial CDUs for display. The CMC provides RCS on-off commands
and SPS on-off commands to the SCS for control of the reaction jets and
the SPS thrust engine respectively. TVC servo commands are supplied via
the optical CDUs to the SCS for automatic thrust vector control. The CMC
accepts discrete translation and rotation commands from the SCS manual
controls. The SCS also provides spacecraft pitch, yaw and roll rate infor-
mation to the CMC in a backup configuration, which the CMC could utilize to
generate attitude error information. The SCS receives an 800 hz reference
signal from the PGNCS Inverter which is utilized in the mechanization for
driving the FDAI displays.

Reaction Control System - The RCS contains 16 SM reaction jets and 12 CM
reaction jets. Each reaction jet is operated by 2 two-coil solenoid valves,
one valve for fuel control and one valve for oxidizer control. The SCS
provides on-off commands to the two sets of coils for each reaction jet.
One set of coils (automatic) is controlled by an electronic switch (solenoid

driver amplifier). The second set (direct) is controlled by switch closures in the SCS hand controllers. The command signals to the automatic coils can originate in either the PGNCS or the SCS.

Service Propulsion System - The SCS interfaces with the propulsion engine solenoid valves and the gimbal actuators of the SPS. The SCS provides thrust on-off commands to the solenoid valve coils which can originate in the PGNCS or the SCS and EMS. The SCS provides servo commands to the gimbal actuator magnetic clutches to control the position of the SPS engine in pitch and yaw. These commands can originate in the PGNCS or the SCS. The gimbal actuator supplies rate and position feedback signals to the SCS. The gimbal actuator also provides a Fail Sense signal to the SCS if the No. 1 actuator motor draws excessive current.

Launch Vechicle Propellant Pressure Transducers - The SCS receives signals from the S-II fuel pressure transducers and the S-IVB fuel and oxidizer pressure transducers for display purposes.

Communications System - The SCS provides a Push-to-Talk trigger switch in each of the Rotation Controls which interfaces with the spacecraft communications system.

2.5   **SCS HARDWARE**

The individual assemblies contained in the Stabilization and Control System are shown in Figure 2.1. The basic functions of each are described here. A more detailed description of the devices is provided in Section 3.

SCS FLIGHT HARDWARE
BLOCK II
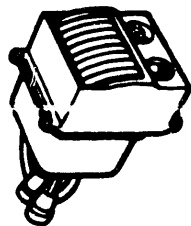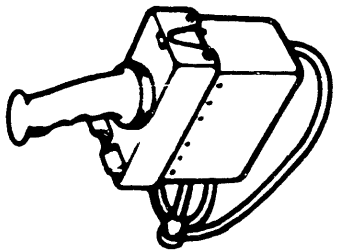
ATTITUDE SET CONTROL PANEL
(ASCP)

GYRO DISPLAY COUPLER
(GDC)

REACTION JET AND ENGINE
ON-OFF CONTROL
(RJ/EC)

GIMBAL POSITION/
FUEL PRESSURE INDICATOR
(GP/FPI)

GYRO ASSEMBLY
(GA1, GA2)

THRUST VECTOR POSITION
SERVO AMPLIFIER
(TVSA)

ROTATION CONTROL
(RC1, RC2)

ELECTRONIC CONTROL ASSEMBLY
(ECA)

TRANSLATION CONTROL
(TC)
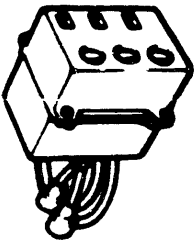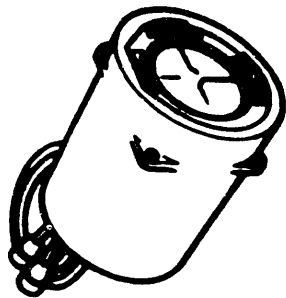
FLIGHT DIRECTOR ATTITUDE INDICATOR
(FDAI 1, FDAI 2)
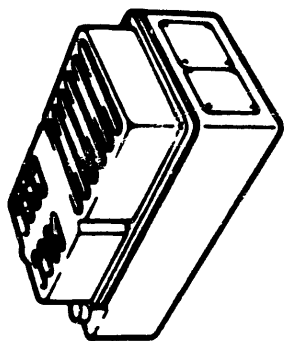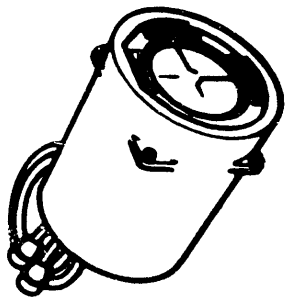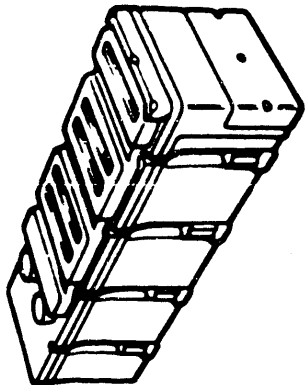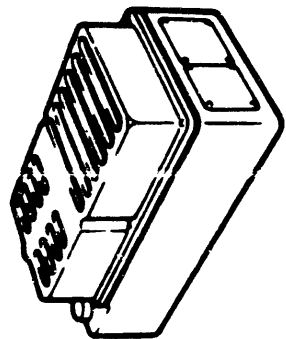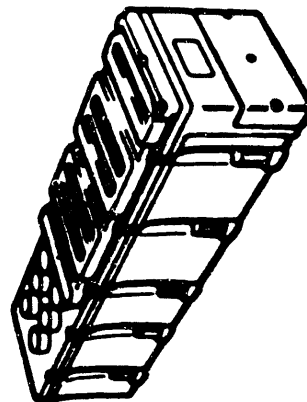
ELECTRONIC DISPLAY ASSEMBLY
(EDA)
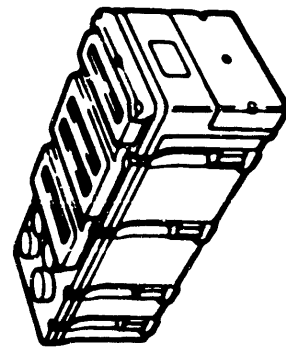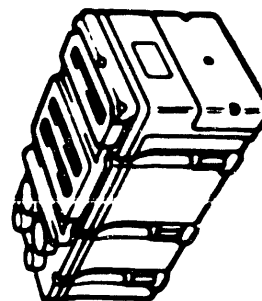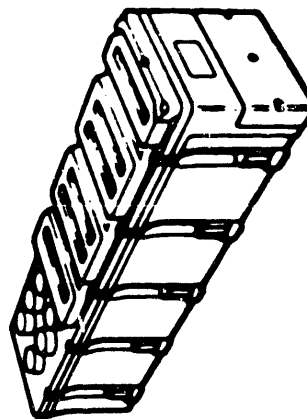
FIGURE 2.1

Reaction Jet and Engine ON-OFF Control (RJ/EC) - The RJ/EC contains the
solenoid drivers and logic circuits necessary to control the reaction jet
automatic solenoid coils and the service propulsion engine solenoid control
valves and relays, as well as related monitoring, isolation and voltage
scaling required for solenoid driver telemetry signal conditioning.

Electronic Control Assembly (ECA) - The ECA contains the circuit elements
required for summing, conditioning and switching the rate and attitude error
sensor signals and the manual input signals necessary to maintain backup
stabilization and control in all axes (pitch, yaw and roll) for thrust vector
and attitude control.

Electronic Display Assembly (EDA) - The EDA contains the signal conditioning
and logic circuits necessary to provide the proper signals to the displays
as well as the monitoring, isolation and signal conditioning for telemetry
of the display signals.

Gyro Display Coupler (GDC) - The GDC provides the interface between the
body rate sensors and the displays to give an accurate readout of spacecraft
attitude relative to a given reference coordinate system.

Thrust Vector Position Servo Amplifier (TVSA) - The TVSA provides the
electrical interface between the command electronics (SCS or G&N) and the
gimbal actuator for positioning the SPS engine, as well as the monitoring,
isolation and voltage scaling required for telemetry outputs.

Gyro Assembly (GA 1 and GA 2) - Each GA contains three sensing elements

(body mounted attitude gyros) and the electronics necessary to provide output signals proportional to angular rate or to angular displacement for each of the three body axes as well as related gyro electronics.

Flight Director Attitude Indicator (FDAI 1 and FDAI 2) - The FDAIs provide the crew with a visual display of spacecraft attitude, attitude error and angular rate information.

Gimbal Position and Fuel Pressure Indicator (GP/FPI) - The GP/FPI provides a redundant display of the service propulsion engine pitch and yaw gimbal angles and a means of introducing manual trim of the engine gimbals. The indicator has the alternate capability of providing a display of S-II or S-IVB fuel and oxidizer pressure when those stages are attached.

Attitude Set Control Panel (ASCP) - The ASCP provides a means of manually establishing an attitude reference coordinate system and a visual readout of the coordinates commanded.

Translation Control (TC) - The TC provides a means of exercizing manual control over rectilinear motion of the spacecraft in both directions along the three body axes. It also provides the capability for manual abort initiation during launch and transfer of spacecraft control from G&N to SCS.

Rotation Control (RC1 and RC2) - The Rotation Controls provide means for exercizing manual control of spacecraft rotation in either direction about its three main axes. The RC may be used for manual thrust vector control in pitch and yaw. It also provides the capability to control spacecraft communications with a press-to-talk trigger switch.

Figure 2.2 provides a view of the CM lower equipment bay and shows the location of the SCS sensors and electronic assemblies. Provision for mounting a Rotation Control at the Navigation station is also indicated. The astronaut couch armrests have mounting brackets for the Translation Control and Rotation Controls. The FDAIs, GP/FPI and ASCP are mounted on the control panel in front of the astronauts.

2.6   SUMMARY OF SCS FUNCTIONAL REQUIREMENTS

General

The SCS hardware performs the following four basic functions:

a) Provides the capability to sense and display an attitude reference.

b) Provides for attitude and translation control of the vehicle during non-thrusting periods using vehicle reaction jets.

c) Provides for attitude control while thrusting by control of the SPS engine gimbals.

d) Provides for ignition control of the SPS engine.

The SCS contains the filtering, shaping, logic and other electronics necessary to mechanize the above capabilities.

Detailed

The SCS in conjunction with the NR furnished switching and the EMS display performs the following functions:

a) Permits control and hold of spacecraft attitude by means of automatic commands from the G&N system through the SCS driver amplifiers, or holds attitude using the SCS attitude reference.

G&C EQUIPMENT LOCATION

Labeled equipment locations include: G&N CONTROL PANEL, ROTATION HAND CNTRL MOUNT, DSKY, G&N SIGNAL COND ASS'Y, EYEPIECE STORAGE, SXT, SCT, PSA, PIPA ELECTRONICS, SCS GYRO ASS'Y #1, SCS GYRO ASS'Y #2, CMC, CDU, RJEC, EDA, GDC, ECA, TVSA
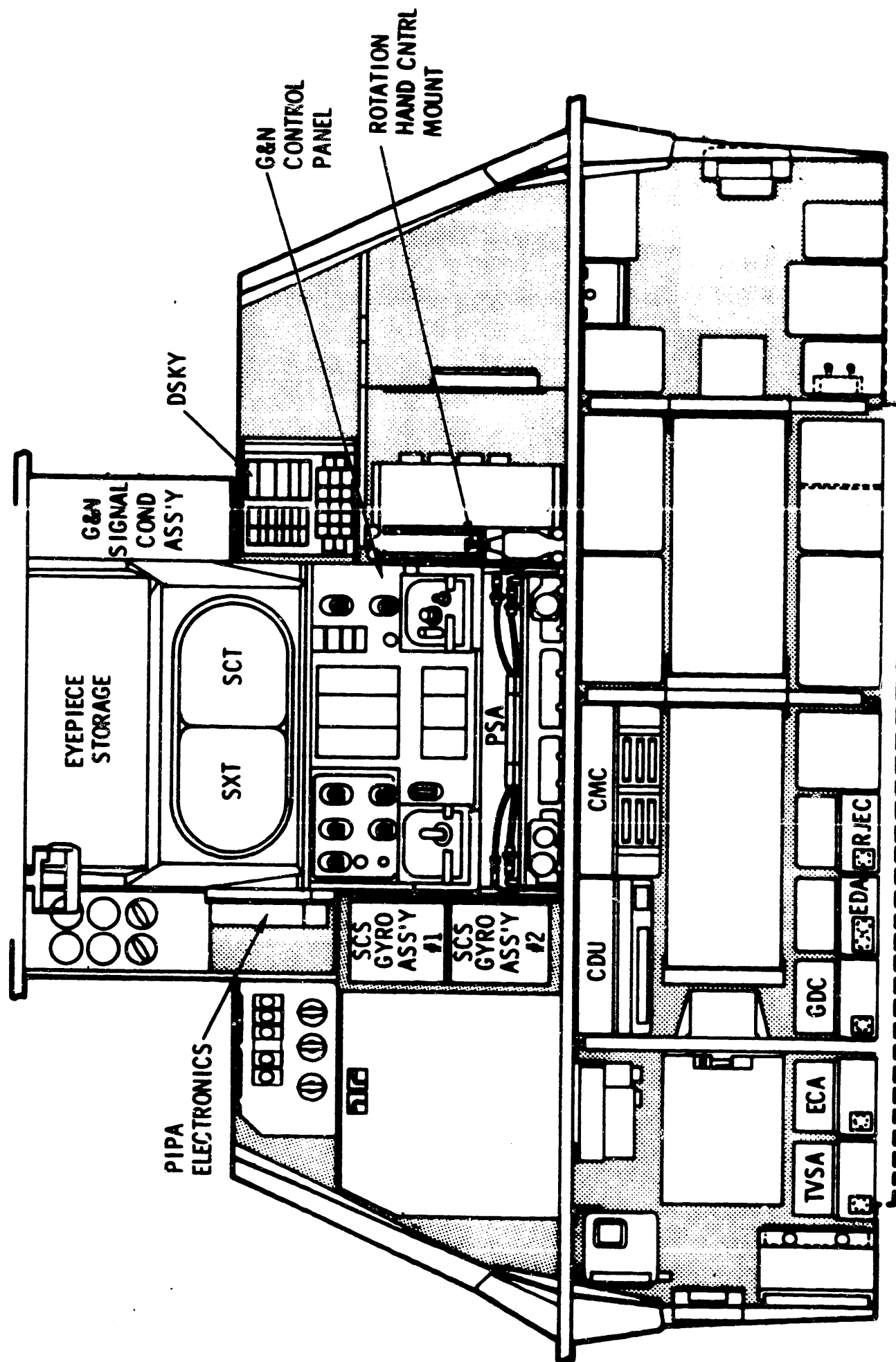
FIGURE 2.2

b) Provides : control of spacecraft attitude with manual rotation controls in the following configurations:

   1) Proportional rate commands through the SCS system.

   2) Acceleration commands through the G&N system or the SCS system.

   3) Minimum impulse commands through the SCS system.

   4) Direct control to the reaction jet solenoids.

c) Displays the spacecraft attitude as follows:

   1) Vehicle attitude with respect to inertial space in three axes utilizing information from the G&N system Inertial Measurement Unit (IMU) or the SCS Gyro Display Coupler (GDC).

   2) Vehicle roll attitude about the spacecraft entry roll axis utilizing information from the GDC.

d) Displays error in spacecraft attitude from one of the following sources:

   1) The G&N system through the G&N CDUs.

   2) The difference between a manual set in attitude and G&N IMU inertial orientation.

   3) The error as computed by the G&N CMC from digital body axis rate signals generated by the SCS GDC.

   4) SCS uncaged attitude gyro outputs.

   5) The difference between the Euler attitude integrated by the GDC and the set in attitude converted to body axis.

e) Displays spacecraft angular rates about the following reference axis:

   1) Each of three spacecraft body axes.

   2) The spacecraft roll and pitch body axes and the entry yaw axis following 0.05g switch actuation.

2-15

f) Provides for manual translation control in six directions through either the G&N system or the SCS system.

g) Provides for rotational rate stabilization of the spacecraft with the SCS system.

h) Provides for manual ullage maneuver.

i) Permits the following four methods of thrust vector control through the SCS servo amplifiers:

    1) Automatic commands from the G&N CDUs.

    2) Automatic control from the SCS.   * (SCS AUTO TVC)

    3) Manual control from the SCS with rate damping. **(MTVC)

    4) Manual control from the SCS without rate damping. *(MTVC)

     * used with and without LEM vehicle attached.
    ** used only without LEM vehicle attached.

j) Provides for manual trimming of the SPS engine gimbals prior to thrusting when the SCS is in control of TVC.

k) Provides for display of engine gimbal angles or the display of S-II and S-IVB booster propellant pressures.

l) Permits automatic G&N ON-OFF control of SPS engine thrust.

m) Provides body mounted gyros (BMAGs) to be used as a backup attitude reference system and to provide rate stabilization.

n) Provides a capability to dial-in information in the following configurations:

    1) Three axis attitude information for use with the display of error in spacecraft attitude.

    2) Three axis attitude information for use in the alignment of the SCS GDC.

3) Redundant entry roll axis attitude information for alignment of the entry roll attitude.

o) Provides the SPS engine with a redundant SCS manual start and automatic turn-off capability.

p) Provides digital pulses proportional to roll axis attitude change during entry to the EMS roll angle display.

SCS HARDWARE DESCRIPTION

3.1   **Controls and Displays**

The SCS provides the following controls and displays which allow the
astronauts to exercise manual control over the spacecraft and also
provide them with visual feedback for the purpose of monitoring various
parameters while the S/C is under automatic or manual control:

> Rotation Control (RC) - 2 each
>
> Translation Control (TC)
>
> Flight Director Attitude Indicator (FDAI) - 2 each
>
> Attitude Set Control Panel (ASCP)
>
> Gimbal Position/Fuel Pressure Indicator (GP/FPI)

The controls and displays are compact, highly reliable, simple to
operate and capable of withstanding the rigorous artificial environment
of outer space.

3.1.1   **Rotation Control - CG166**

The system contains two identical three-axis Rotation Controls. The
controls are connected in parallel to provide redundancy without switching.

Functions

The Rotation Control allows the astronaut to manually insert commands
to control the rotation of the spacecraft about each of its three axes.
The commands can be routed through several alternate control channels
which are selectable by the astronaut through Control Panel switching.

The commands inserted by the astronaut are ultimately used to control S/C rotation by (1) firing the proper RCS engines (pitch, yaw and roll control) or (2) positioning the SPS engine to control the direction of thrust (pitch and yaw control).

The Rotation Control also has a push-to-talk switch (Communication subsystem interface) which is unrelated to vehicle control.

Mounting

Mounting brackets for the Rotation Controls are located on the couch armrests and at the NAV station in the lower equipment bay. The Rotation Control is provided with a tapered female dovetail on each end of the housing for mounting purposes (see Figure 3.1). When attached to the armrests, the command input axes are approximately parallel to the geometrical axes of the spacecraft. The Rotation Control electrical interfaces are made through its 180 inch cable. The long cable allows either control to be mounted at any of the mounting locations. The maximum weight of a Rotation Control is 10 pounds.

Mechanical Features

The Rotation Control contains a spring-restrained handgrip which is free to move in either direction about each of the three orthogonal axes corresponding to the S/C roll, pitch and yaw axes. The handgrip is contoured to allow a firm grip (and thus optimum control capability) with either hand. The handgrip can be displaced from its springloaded null position as indicated in Figure 3.2. To insert a yaw command, the grip is displaced

ROTATION CONTROL

FIGURE 3.1

ROTATION CONTROL MECHANICAL FUNCTIONS

PITCH AXIS

MAX DEFLECTION ALL AXIS

-12° +12°

END VIEW

YAW AXIS

FLEX BOOT

ROLL AXIS

20°

PUSH TO TALK SW

LOCKING DEVICE

- PITCH

YAW

ROLL +

+

-

-

+ PITCH

TOP VIEW
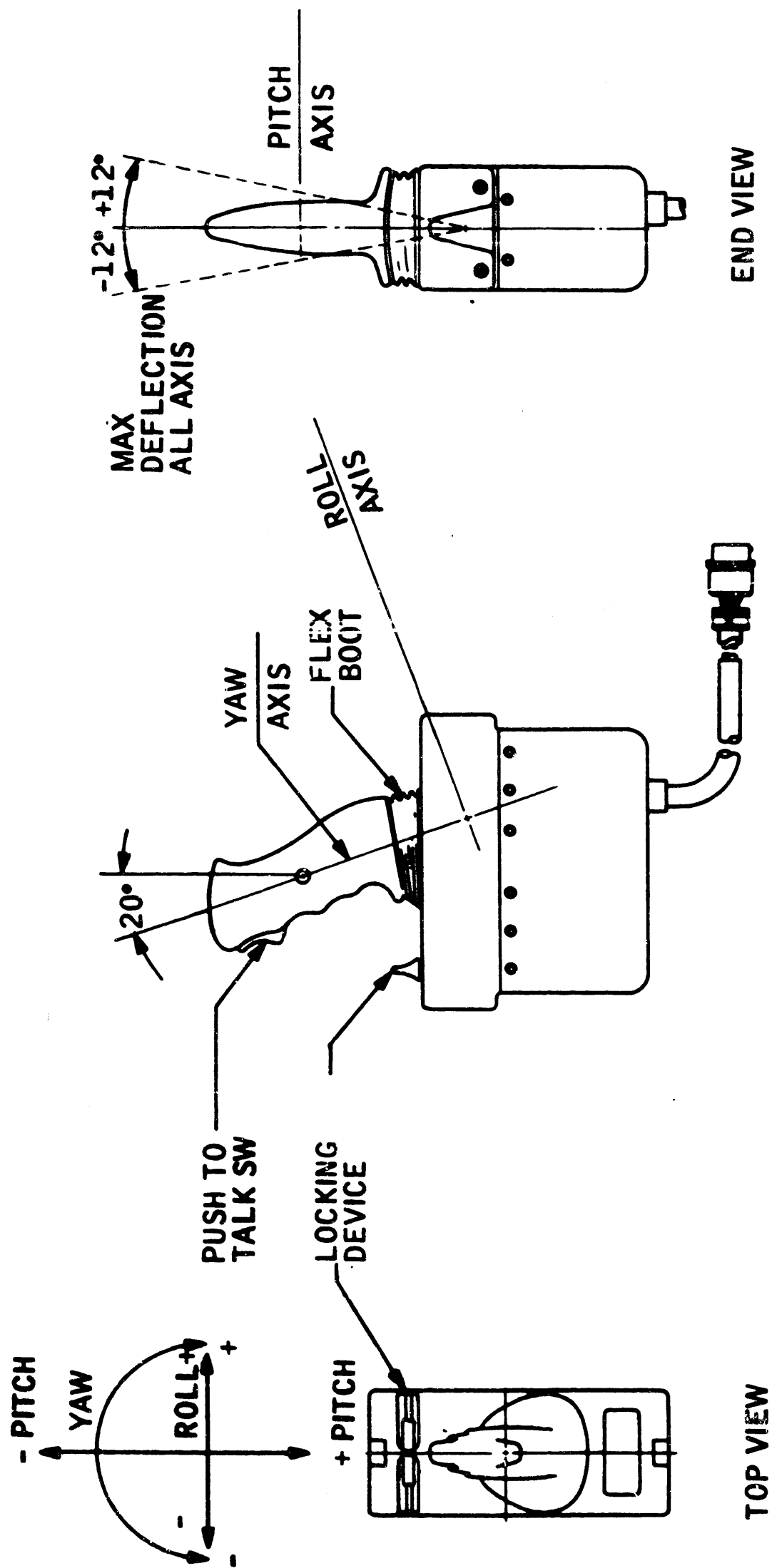
FIGURE 3.2

about an axis which is the centerline of the grip. A pitch command is applied by displacing the grip about an axis through the palm of the hand. To command an attitude change of the S/C about the roll axis the grip is displaced about an axis below the hand perpendicular to the grip centerline (in the plane of the forearm). Commands can be inserted for any combination of axes simultaneously.

The springs which return the grip to null are preloaded, so a finite torque is required to cause an appreciable displacement. As the grip is moved from null, (in either direction about any axis), the restraining torque increases in an approximately linear manner to a soft-stop at 10 degrees. A finite increase in torque is required to move the grip beyond the soft-stop. Displacement of the grip is limited to 12 degrees by a positive mechanical stop called the hard stop. The spring constants in each axis are designed to provide optimum control capability.

Each Rotation Control contains six SPST NO breakout (BO) switches, two in each axis. Rotation of the grip through 1.5 degrees in either direction about any axis will actuate one of the switches. Each Rotation Control also contains six equivalent DPST NO switches. These switches are actuated by displacing the handgrip through 11 degrees (between the softstop and hardstop).

A Rotary Linear Voltage Differential Transformer (RLVDT) is contained in each axis of the RC. The RLVDTs provide analog output signals proportional to the magnitude of the handgrip displacement.

A SPST NO Push-to-Talk switch is located in the handgrip. This trigger switch is actuated with the index finger to engage the S/C intercom.

Provision is made to mechanically lock the handgrip in the neutral position. Two locking handles are linked together and allow the handgrip to be locked or armed from either side of the RC. The locking mechanism does net affect the Push-to-Talk switch. An equivalent DPST NC switch is actuated when the locking handles are placed in the LOCKED position.

Electrical Interfaces

The electrical interfaces of Rotation Control No. 1 (RC1) are indicated in Figure 3.3. RC2 has identical interfaces except as indicated. 28 VDC power is supplied from the Rotation Control Power Switch (separate switch for RC2) to the ARMED contacts of the locking switch. When the handgrip is displaced 1.5 degrees from null about any of the three axes, the proper BO switch is actuated, providing a 28 VDC discrete signal to the devices indicated.

The RLVDTs receive 26 volt 400 hz AC1 (AC2 for RC2) power through the Rotation Control Power switch from a stepdown transformer in the ECA. The RLVDT outputs are proportional to the handgrip displacements in their respective axes. An in-phase signal is provided for positive rotation commands, while negative rotation commands yield a π - phase signal relative to the excitation. The direct switches receive 28 VDC power from the Control Panel 1 Direct Rotation Control Power switch

ROTATION CONTROL INTERFACES
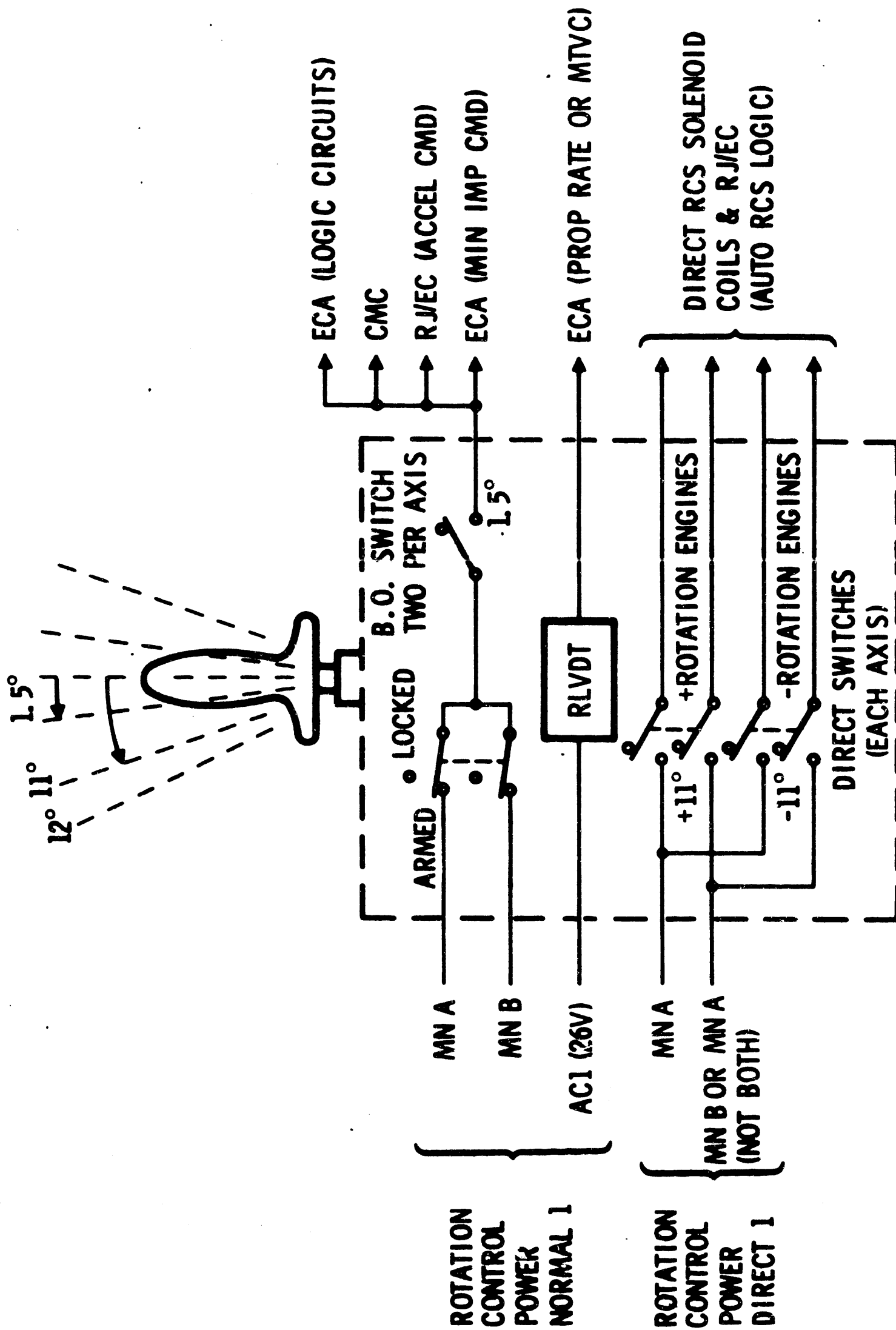
FIGURE 3.3

(separate switch for RC2). When the handgrip is displaced 11 degrees from null about any of the three axes, the proper direct switches are actuated, providing 28 VDC to the direct RCS solenoid coils and a disable command to the RCS logic in the RJ/EC.

3.1.2 Translation Control - CG160

The system contains a single Translation Control with built-in redundancy.

The Translation Control provides the means for inserting manual translation commands into the control system to fire the proper RCS engines. Any combination of commands for translation along the S/C $\pm$ X, $\pm$ Y and $\pm$ Z axes can be inserted simultaneously. The Translation Control also provides the capability to:

      (1) initiate a manual launch abort

      (2) transfer from G&N to SCS control of the S/C

      (3) transfer from SCS Automatic to Manual TVC

      (4) transfer from gimbal servo channel No. 1 to No. 2

Mounting

The Translation Control is provided with a tapered female dovetail. This dovetail mates with mounting brackets on an armrest or at the left side of the NAV station in the lower equipment bay. When attached to the armrest, the T-bar grip is approximately normal to the armrest. The TC is mounted with its axes of operation approximately parallel to the three S/C axes.

Mechanical Features

The TC consists of a spring-restrained T-bar handgrip, which is moveable in four axes, mounted on an environmentally sealed enclosure (see Figure 3.4). The maximum weight of the TC is 5 pounds, 11 ounces. The three principle degrees of freedom are in and out, up and down and left and right. When mounted in the S/C, movement of the T-bar generates commands to provide translation of the S/C in the corresponding direction. Motion of the T-bar handle is restricted to 0.5 inches in any direction from neutral. Switches, actuated at a displacement of 0.35 inches from neutral, provide the means for applying control signals to cause translation of the S/C. A center detent action is provided so that a force of approximately 1.5 pounds is required to obtain appreciable motion of the handle. Spring restraining forces have been engineered to provide good "feel" and smooth control.

The fourth degree of freedom of the control is one of rotation about the X-axis. It may be rotated 17° CW or CCW. The rotary motion features a detent at the center position and one at each extreme of its CW and CCW motion. Separate switches are actuated at ± 14 degrees displacement (a DPST NO switch for CCW rotation and two SPST switches, one NO and one NC, for CW rotations).

A slide-actuated mechanical locking mechanism is located above the T-bar handle to prevent inadvertent actuation. All motions, except the rotary motion of the T-bar handgrip, are inhibited when the mechanism is in the LOCKED position.

TRANSLATION CONTROL
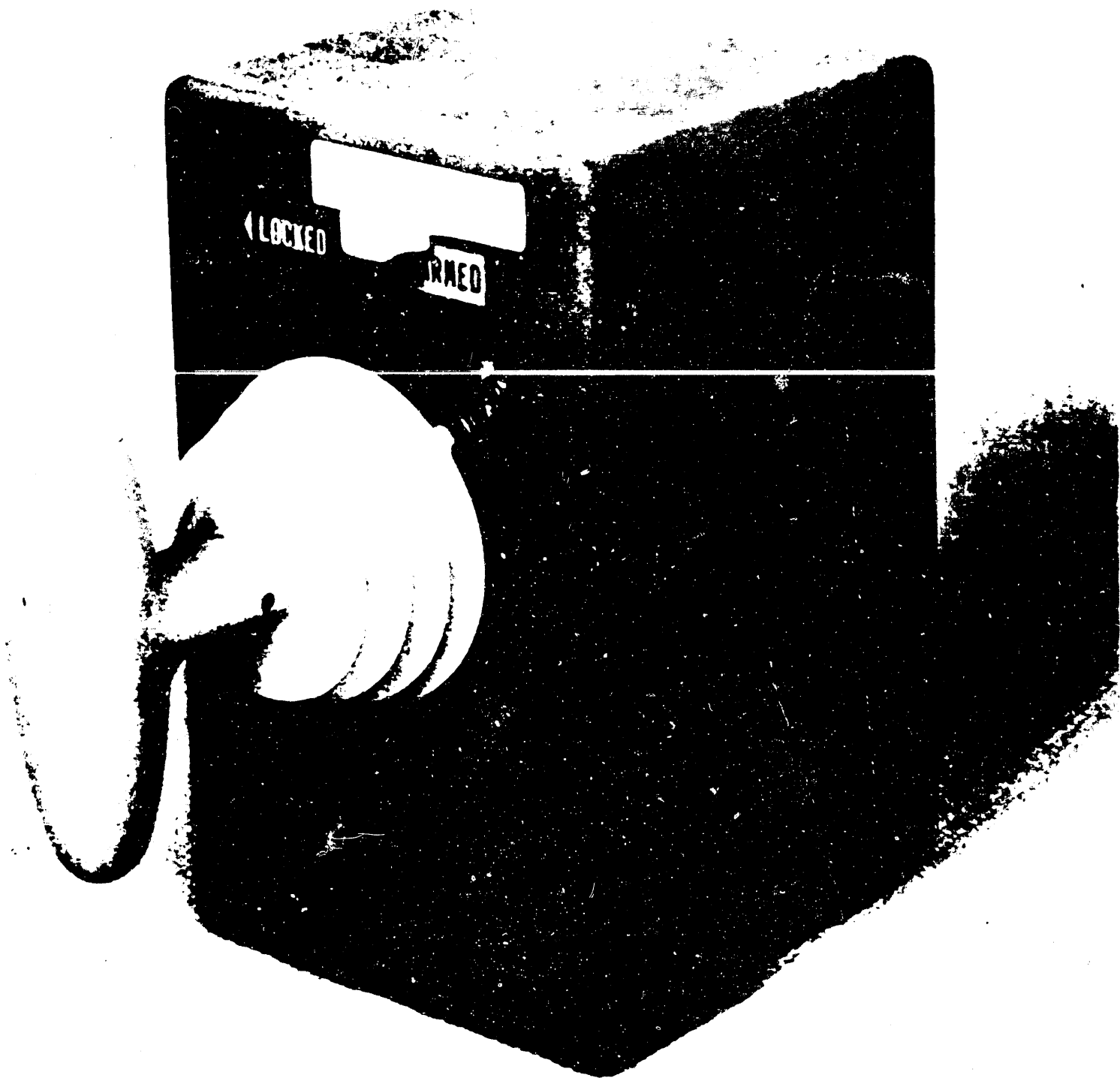
FIGURE 3.4

Electrical Interfaces

All electrical connections are made through two connectors attached to the single 7.5 foot TC cable (see Figure 3.5). MN A and MN B 28 VDC power is supplied to the redundant poles of the six DPST switches (one for each translation direction) through the CM/SM TRANSFER motor-driven switch and through the separate connectors. The redundant outputs are supplied through bus isolation resistors to the CMC and RJ/EC.

Rotating the TC T-bar handle CW closes the NO switch to supply 28 VDC logic power to the ECA and RJ/EC which enables the SCS, and to the TVSA for a transfer command from gimbal servo system No. 1 to No. 2 and the SCS enable command. The CW rotation also opens the NC switch which inhibits G&N control if the SC CONT switch was in the CMC position.

A manual abort is initiated by rotating the TC handle CCW, supplying Entry Battery power through the redundant connectors and NO switches to the Sequential Events Control System.

3.1.3   ## Flight Director Attitude Indicator - JG264

The system contains two identical FDAIs. FDAI 1 is mounted in Control Panel No. 1 and FDAI 2 in Control Panel No. 2.

The FDAIs provide the astronauts with a visual display of the S/C attitude, attitude rate and attitude error in each of three orthogonal axes. Control Panel switching allows the astronaut to display this information on either or both FDAIs from alternate sources. Power and control signals for the

TRANSLATION CONTROL INTERFACES
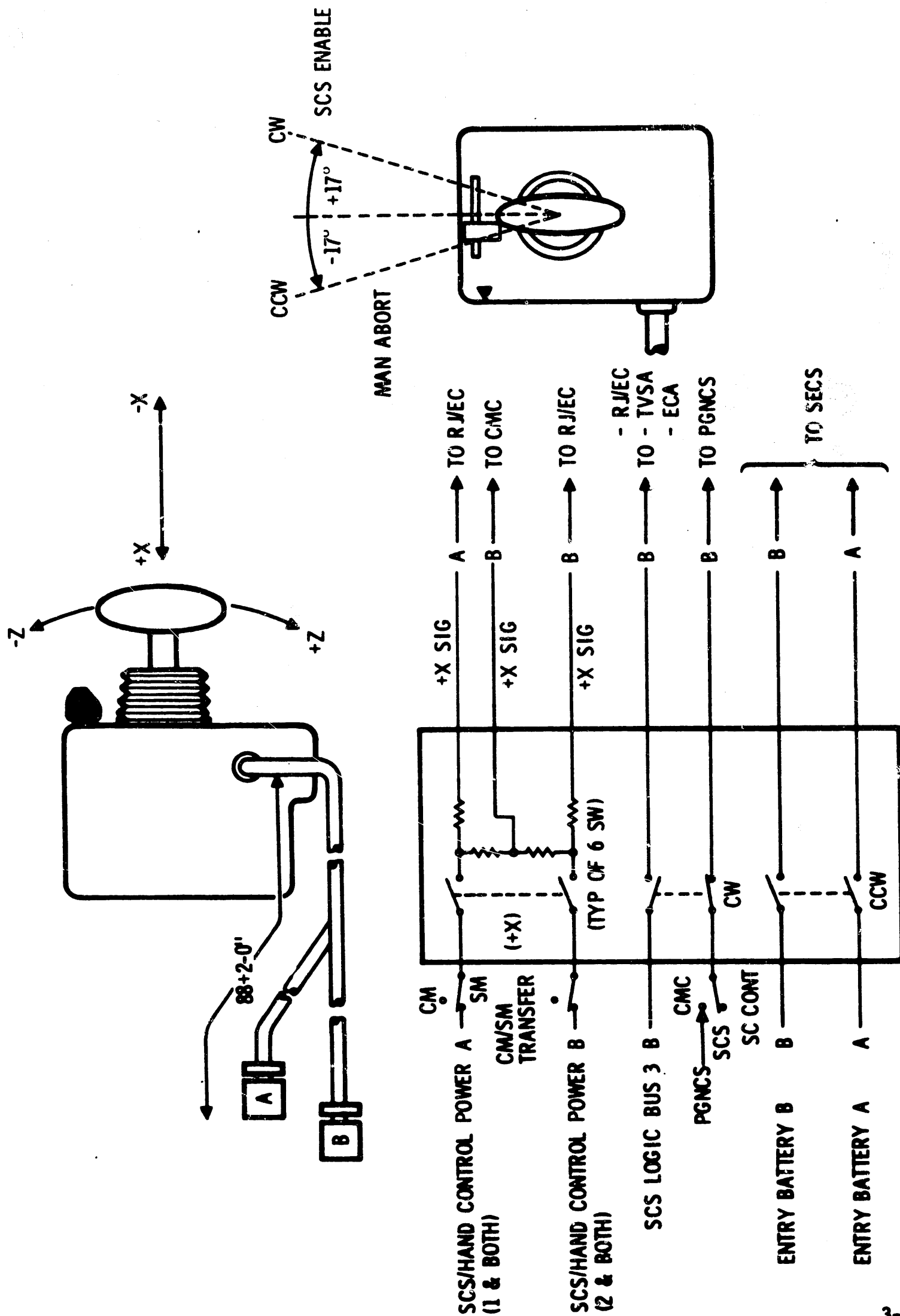
FIGURE 3.5

FDAIs are supplied through the Electronic Display Assembly. Figure 3.6 provides a front view of an FDAI.

Attitude Display

The FDAI presents "inside looking out" attitude information through an easily-read, 4.5-inch diameter, three-axis ball gimbaled inside to outside in the order pitch, yaw and roll. The ball has continuous and independent freedom of motion about each of the three orthogonal axes. Pitch and yaw readouts are made against a fixed cross hair and drone marker centered in front of the ball. Roll attitude is indicated by a pointer (roll index) against a circular scale around the periphery of the ball. Each axis is positively-controlled by a servo drive consisting of a servomotor, resolver and gear train.

The attitude indicator functional mechanization is the same for all three axes. The objective is to position the ball to the same angle as the source resolver for each axis. The source resolver provides AC signals proportional to the sine and cosine of the resolver angle to the FDAI resolver. The FDAI resolver output is proportional to the difference between the source and FDAI resolver angles. This error signal is conditioned in the EDA to drive the FDAI motor-velocity generator which positions the FDAI resolver to null out the error signal.

The VG provides a feedback signal to ensure stability of the servo loop. A pictorial representation of the FDAI/IMU gimbal relationship is given in Figure 3.7. The torque motors drive the IMU gimbals to keep the

FLIGHT DIRECTOR ATTITUDE INDICATOR

ROLL
+ANGULAR VELOCITY-
+ATTITUDE ERROR-

ROLL INDEX

PITCH & YAW
INDEX

EULER ATTITUDE ON BALL
PITCH - θ = 014°
YAW - ψ = 034°
ROLL - ϕ = 330°

NOTE:
ALL POLARITIES INDICATE
VEHICLE DYNAMICS

```
   -
ATTITUDE  P
ERROR     I   ANGULAR
          T   VELOCITY
   +      C
          H
              +
```

ROLL TOTAL
ATTITUDE SCALE

YAW
+ATTITUDE ERROR-
+ANGULAR VELOCITY-
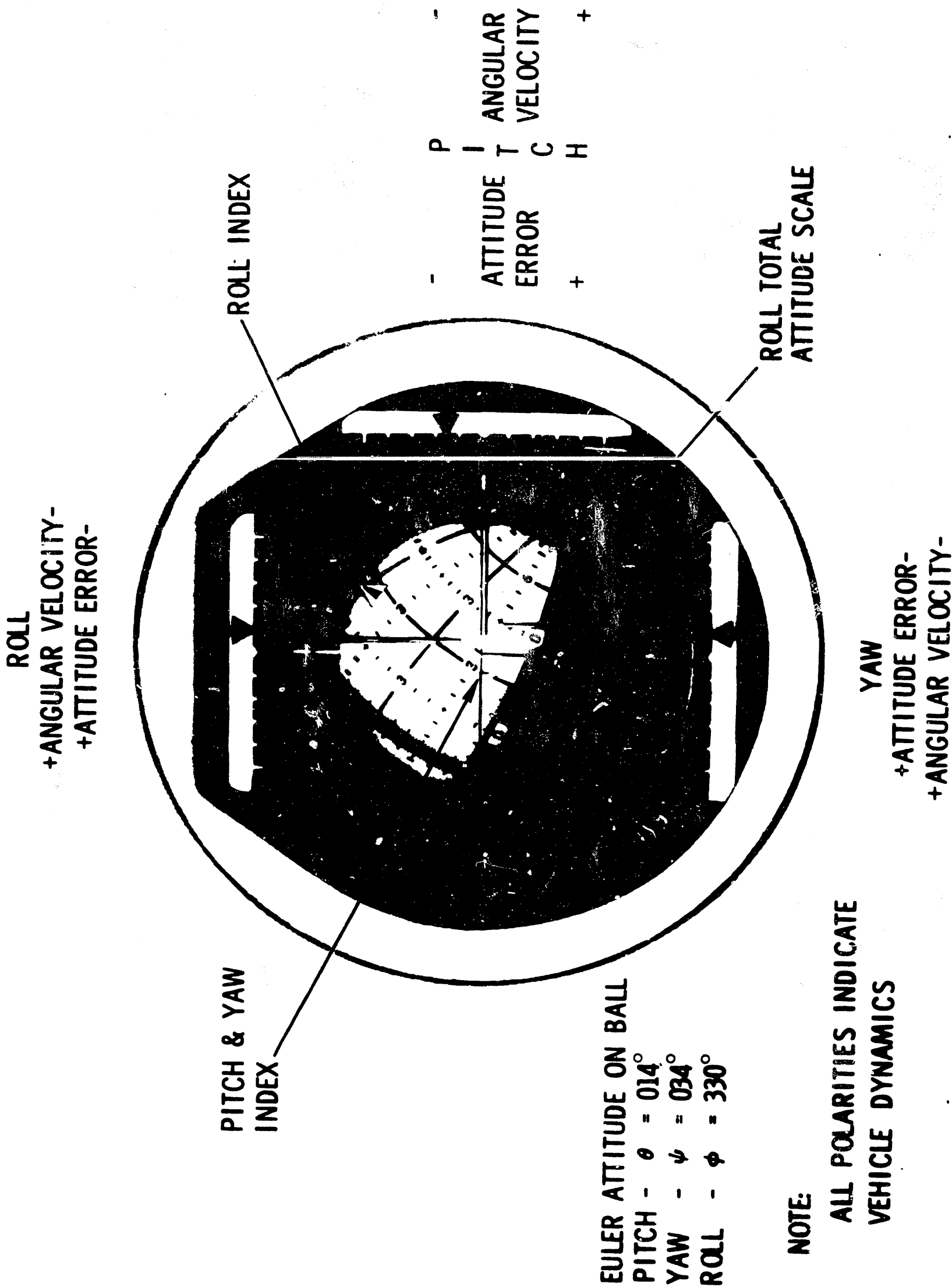
FIGURE 3.6

FDAI/IMU GIMBAL RELATIONSHIP

- FIGURE 3.7

3-15
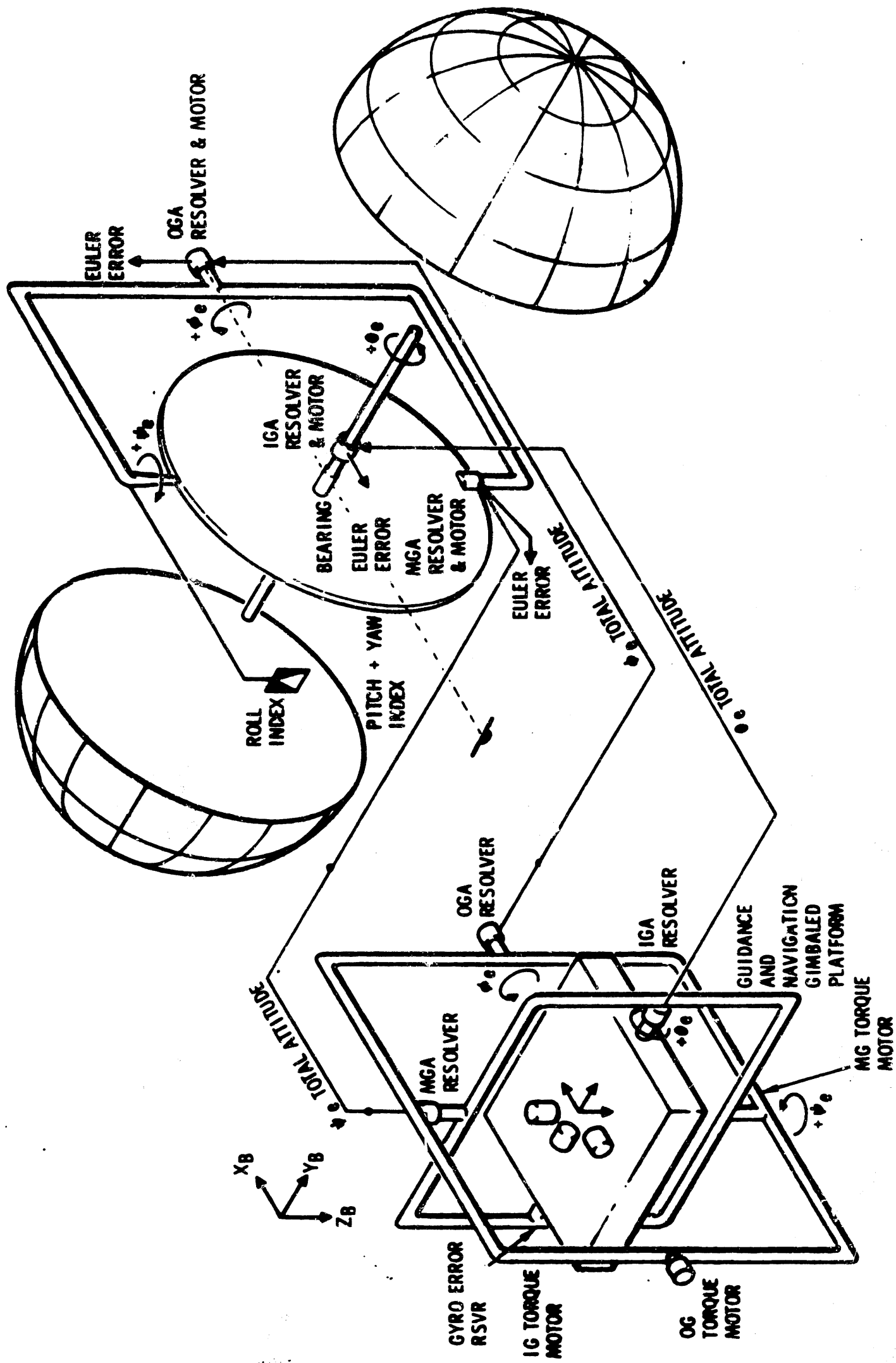
platform inertially fixed in space as the S/C attitude changes. Hence the IMU gimbal angles describe the S/C attitude relative to the platform inertial reference. The IMU gimbal resolver outputs to the respective FDAI resolvers are shown here by single lines. The euler error signals (difference between IMU and FDAI resolver angles) are fed from the FDAI resolvers to the respective servo amplifiers in the EDA to drive the proper FDAI motors until the euler error signals are nulled. The servo loop is designed to drive the ball at a max slew speed of 60 degrees per second.

A portion of the attitude indicator markings described below are visible in Figure 3.6. The pitch lines are great semi-circles which converge at 90° yaw and 270° yaw. The yaw lines are small circles which form parallel planes on either side of the 0° yaw plane. The lower half of the ball is black and the upper half gray (except the gimbal lock areas). The black-gray intersection (horizon line) represents a plane passing through 0° and 180° pitch and 90° and 270° yaw. The red circular gimbal lock areas cover ± 15 of spherical surface from their centers located at 90° and 270° yaw. Major pitch and yaw lines are provided and marked every 30°. (The last zero on each number is omitted). As the ball is rotated about the pitch axis all numbers appear right side up to a stationary observer. Major tic graduations appear every 5° between the pitch and yaw coordinate lines. Minor tic graduations appear every 1° on each hemisphere along the 0° yaw plane and on top of the horizon plane (0° and 180° pitch) for $0^\circ \pm 30$ yaw. Scale marks are provided every 5° on the roll bezel ring. Roll attitude

is read at the position of the roll index (white triangle with black triangle inside). 0° roll is at the top (12 o'clock position). Positive roll attitude changes cause the roll index to move counter clockwise. Hence the 3 o'clock position looking at the face of the FDAI represents 270° and the 9 o'clock position 90° roll attitude.

The body axis symbol, against which pitch and yaw attitude are read off the ball, consists of a drone with cross lines. The intersecting lines inside the drone are yellow. Extensions of these lines through the wings and tail of the white drone are black and provide mounting support for the drone to the roll bezel ring.

Motion of the ball within the FDAI case, as observed by the astronaut looking into the FDAI face, will always be in a direction opposite the motion of the spacecraft. If the S/C pitches nose up the ball is driven down. Rotation of the S/C nose to the right results in the ball being driven to the left. Likewise, a CW roll of the S/C causes the ball to rotate CCW.

Attitude Error Display

The attitude error indicators consist of three bow-shaped pointers positioned by servometric mechanisms located in the rear of the case. These error pointers curve in front of the ball and are read with respect to the fixed attitude scale marks on the periphery of the ball. The indicator scales (roll at the top, pitch to the right and yaw at the bottom) represent zero at the center marks of each scale. The pitch and yaw graduation

marks are located at null, $\pm$ 1/3 full scale, $\pm$ 2/3 full scale and $\pm$ full scale. Full scale values are selectable at 5° or 15° attitude error by a control panel switch. Roll graduation marks are located at null, $\pm$ 1/2 full scale, and $\pm$ full scale. Full scale values of 5° or 50° roll attitude error are selectable. The scale marks are white on a black background. The pointers are yellow. The attitude error indicators are sometimes described as having "fly to" needles or pointers. (The attitude error polarities for each scale are indicated in Figure 3.6). This description is appropriate because the attitude error can be reduced by inserting a command through the Rotation Control to move the S/C toward the error pointer, i.e. if the pitch error pointer is above the null mark, (negative attitude error) moving the RC handgrip such that the knuckles move up will cause the S/C to pitch up and thus cause the attitude error pointer to move towards null. An analysis of the yaw and roll axes yields similar results.

Attitude Rate Display

The attitude rate indicators consist of three black triangular-shaped pointers (one each for roll, pitch and yaw rate) positioned by servometric mechanisms. Readouts are made against individual rate scales. The scales have graduation marks at null, $\pm$ full scale and at 1/5 full scale increments. Full scale values for pitch and yaw rate of 1, 5, or 10 °/sec are selectable, while values of 1, 5, or 50 °/sec are provided for roll rates. The black pointers are silhouetted against the white electroluminescent lamp background. The graduation marks are white on a black background.

The angular rate indicators also provide the astronaut with "fly-to" indications. "Fly-to" commands made with the Rotation Control reduce the rates toward null in the same manner as described for the attitude error indicator. If the objective is to reduce the attitude error in a particular axis to zero, it is necessary to null the rate indicator in that axis at the time the attitude error indicator is at null.

Case Structure

The case is machined from forged aluminum. The cylindrical case provides the required mechanical strength with the lowest possible weight (9 pounds maximum). The glass window, which is coated with a reflection reducing coating, is retained in the case with a semi-rigid epoxy. The aluminum rear cover contains a molded-in gasket, a fill hole and four locked-in type studs for attaching the S/C coldplate to the device.

A round header with attached lead wires is soldered to the case. The wires are separated into two bundles, each covered with silicone tubing. The header is potted for humidity protection and strain relief. The connectors have rubber seals with undersized holes (for humidity protection) through which the individual lead wires are fed. There are four mounting lugs on the outside front of the case for mounting the device to the S/C Control Panel.

Some basic construction features of the FDAI are included in the following paragraphs which require reference to the assembly drawing JG264G.

Ball and Yoke Assembly

The ball structure consists of a round (diameter of the ball) narrow frame on which are mounted the pitch and yaw resolvers, motors and gear trains. The pitch resolver and motor are mounted at right angles to the frame and the two hemispheres are coupled to the double-ended resolver shaft, one directly and the other through a pin and coupling. The yaw resolver and motor are mounted parallel to the frame and the resolver is directly coupled to the yaw axis pivot point.

The ball frame (with the hemispheres) is attached (gimbaled) in the two piece yoke casting. The yaw axis slip ring assembly, containing a ball bearing, is at one of the pivot points. The outer race of the bearing is attached to the yoke and the inner race to the frame. At the other pivot (ball bearing) a gear is rigidly attached to the yoke. This gear is meshed with the yaw gear train.

The roll axis slip ring assembly (also containing a ball bearing) is attached to the rear of the yoke. The inner race of the bearing is attached to the yoke, the outer race to the rear support. The yoke is supported and pivoted in front by a 5 inch diameter ball bearing assembly, which is mounted in the bearing support. Four 3-3/4 inch standoffs attach the front bearing support to the rear support. Both front and rear supports are attached to the case. The roll axis resolver, motor and gear train are mounted on a gear plate which is attached to the rear support. The resolver shaft is directly coupled to the roll axis slip ring assembly ball bearing inner race.

3-20

Servometric Meters

Servometric meters are used (rather than galvanometric) for the attitude error and rate indicators for several reasons. They are more accurate, more reliable, lighter, more resistant to vibration and use less space. The dynamic characteristics (response time, overshoot, etc.) of the servometric indicators are accurately controlled by the design of their servo loops.

A servometric indicator consists of a torquing coil, pointer, and a feedback potentiometer. The scaled and conditioned DC attitude error or rate signal is fed to the servo amplifier (located in the EDA). The amplifier supplies a DC current to the torquer coil, driving the shaft to which are attached the pointer and the feedback pot wiper. Negative feedback is supplied to the servo amplifier by the feedback pot wiper. The shaft is thus held in the position necessary to keep the sum of the input and feedback currents to the servo amplifier at null. When not powered, the indicator will not show a null, but will remain at the position commanded at the time power is removed.

The six attitude error and rate servometric meters are attached to the rear frame of the FDAI. Two meters are attached to each of three sides of the rear support. The two meters on each side are mounted in tandem with the front meter mounting plane stepped down and the rear meter slightly overlapping it. The six pointers visible at the face of the FDAI are attached to the meter armatures by two-section tubular members which are kept rigid by tension wires within the tubes.

Electroluminescent Lighting

White electroluminescent lighting (EL) provides illumination of all display elements in the FDAI. EL illumination is uniform, glare-free, requires low power, and enhances reliability due to its freedom from catastrophic failure.

The EL lighting transformer is mounted on the rear support of the FDAI. This transformer provides power to all the FDAI lamps. The maximum lighting power consumed by the lamps is 4.5 volts.

A conically shaped 4.5 inch diameter EL lamp is attached to a ring frame which is mounted on the front bearing support. The center of the lamp is about 1/2 inch back from the front of the ball. This lamp illuminates a portion of the ball. (The roll attitude scale and the drone are also attached to the ring frame).

A larger diameter, similarly shaped EL lamp is mounted on another ring frame attached to the first. The center of this lamp is about 1/8 inch ahead of the ball. This lamp lights the roll attitude scale, the error scales, and the roll index. (The attitude error scales are mounted on standoffs to the front of the roll attitude bezel ring).

Three white and black painted translucent attitude rate scales are attached to the above mentioned EL lamp frame. Three EL lamps behind the rate scales illuminate the scales.

Another frame containing a cylindrical EL lamp around the periphery of
a 4.5 inch diameter, .5 inch thick two-piece acrylic lighting wedge is
attached to the above mentioned frame. The lighting wedge reflects the
light from the EL lamp surrounding it to illuminate the front of the ball,
the drone and the error pointers. A mask which outlines the indicator
scales is attached to the case between the lighting wedge and the glass
window to prevent extraneous glare from reaching the viewer.

3.1.4   Attitude Set Control Panel - CG161

The ASCP shown in Figure 3.8 is mounted in the lower left corner of control
panel 1 with four bolts. The display is designed to provide maximum
readability from the nominal viewing position which is at a 22 inch
distance and an angle of 40° to 46° from the normal to the panel in a
vertical direction upward. Figure 3.9 depicts the astronaut's view of
the ASCP face.

Function

The ASCP provides the means for manually inserting desired attitude
information into the SCS system in the form of three angles. The ASCP
receives signals which represent the actual attitude of the spacecraft
relative to an arbitrary inertial (fixed) reference frame. Output
signals are provided which represent the attitude error or the difference
between the actual and desired total attitude of the spacecraft. These
output signals can be used (1) to drive the attitude error needles on
either FDAI, providing the astronaut with a visual indication of the
spacecraft attitude error, and (2) to align the GDC to a fixed reference
frame.

3-23

ATTITUDE SET CONTROL PANEL
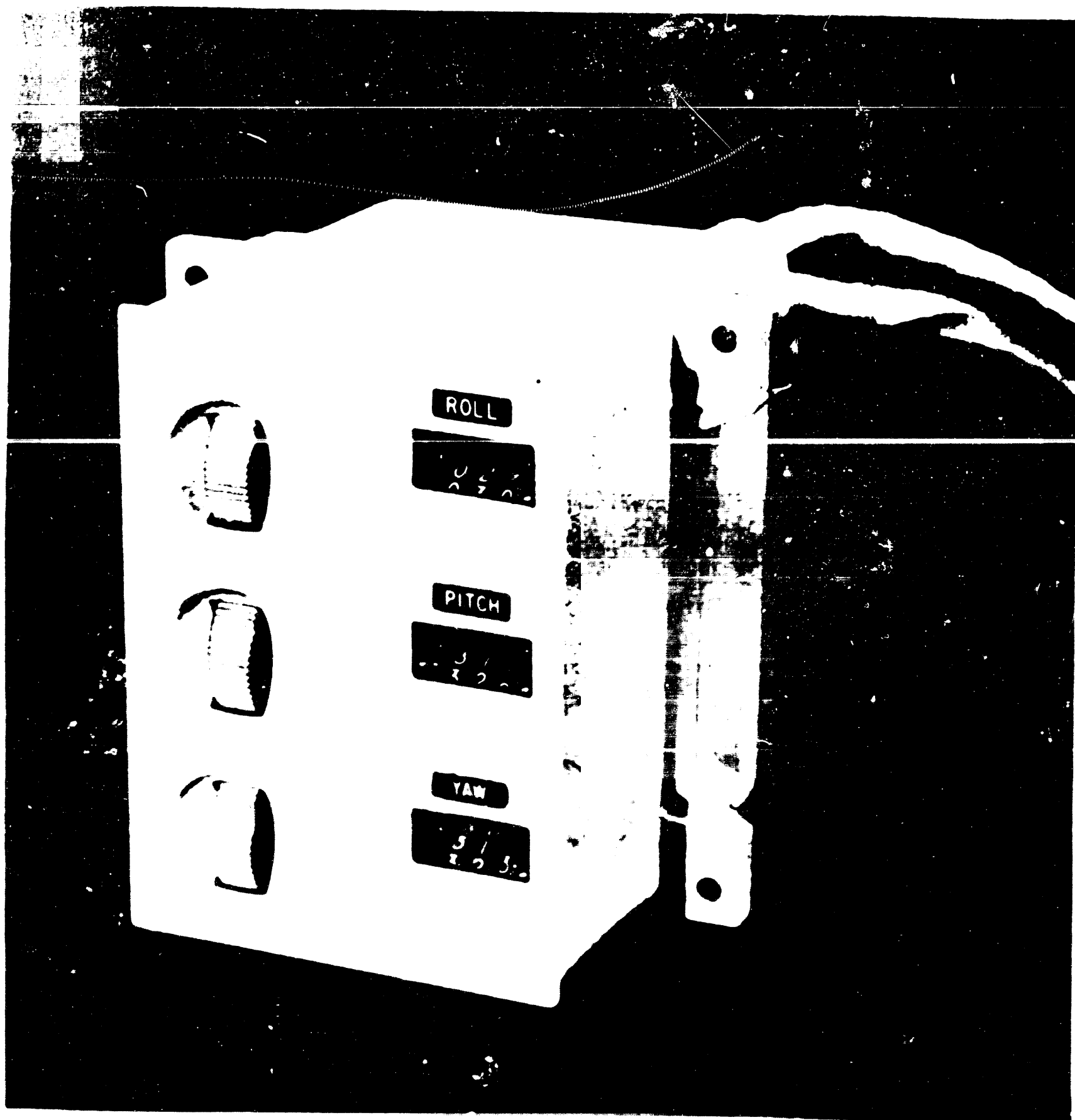
ASCP FACE FORMAT (As Viewed From 45 Degrees)
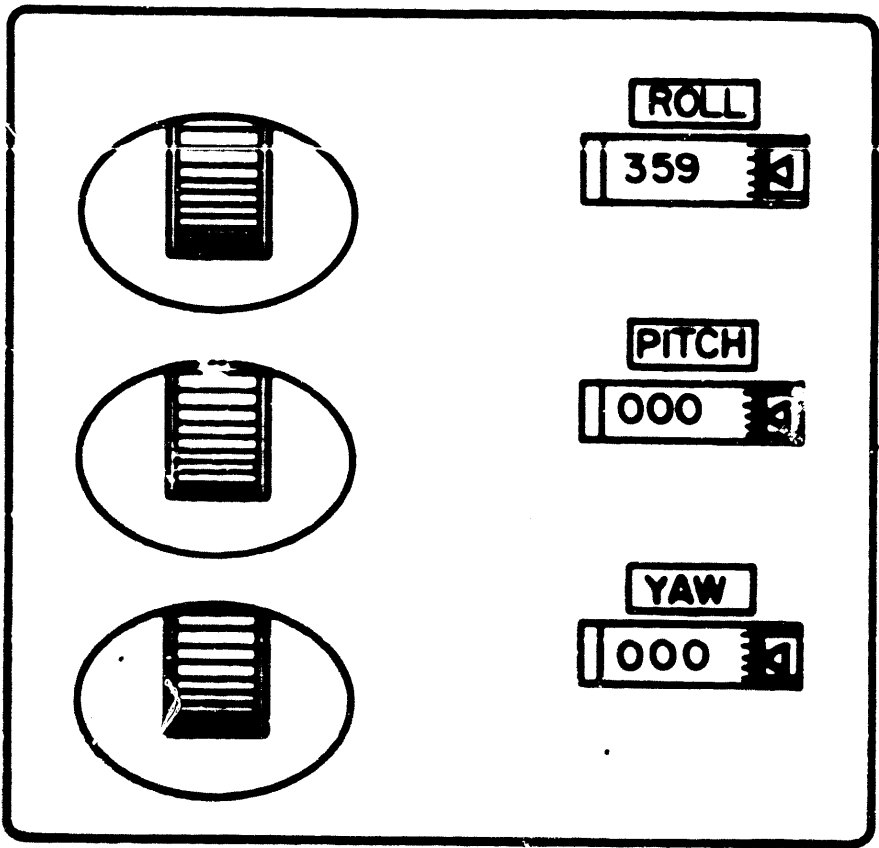
: 3.2

The ASCP provides the means for manually positioning a resolver for each of three axes with a thumbwheel. These resolvers are mechanically linked to indicators which provide visual feedback to the astronaut of the command attitude angles which he manually dials in. The input signals to the attitude set resolvers are four wire resolver outputs from the IMU or GDC. The input signals are sine and cosine functions of the IMU or GDC attitude in each axis. The attitude error output signals are sine functions of the differences between command attitude (positions of the attitude set resolver rotors) and the actual attitude (positions of the IMU or GDC transmitter resolver rotors).

Mechanical Features

The ASCP case is machined from forged aluminum. Electrical access for system operation is made through two 19-pin connectors and 24 inch cables. The maximum weight of the ASCP including connectors and cables, is 3 lbs. 6 oz. The basic elements of the ASCP are the case, 2 connectors, 2 cables, 3 thumbwheels, 3 counters, 3 resolvers, 3 gear assemblies, and 9 electroluminescent lamps.

Custom anti-backlash gears and precision resolvers are used to provide a position accuracy of 0.1 degrees. Dual friction brakes are used to provide constant thumbwheel torque and greater stability under vibration. The force required to rotate the thumbwheels is 10 ± 5 ounces while a maximum of 36 ounces is required initiate rotation.

The counters indicate resolver angles in degrees from electrical zero and allow continuous rotation from 000 thru 359 to 000 without reversing direction of rotation. The counters display three digits, with graduations every 0.2 degrees at the unit digits. The pitch and roll indicators are marked continuously between 0 and 359.8 degrees. The yaw indicator is marked continuously from 0 to 90 degrees and from 270 to 359.8 degrees. Only the unit digits and 0.2 degree graduations appear between 90 degrees and 270 degrees except that the 180 degree number is marked.

The counter readings increase with upward rotation of the thumbwheels. The thumbwheel gearing is such that one revolution of the thumbwheel will produce a 20 degree change in the resolver angle and a corresponding 20 degree change in the counter reading. The thumbwheels are recessed into the front panel to prevent accidental movement.

Each counter readout is floodlighted by two EL lamps and the nomenclature (ROLL, PITCH & YAW) is backlighted by separate EL lamps.

Electrical Interfaces and Interconnections

115V 400 hz variable lighting power is supplied to the nine electroluminescent lamps which backlight the PITCH, YAW & ROLL nomenclature and floodlight the counter readouts. The maximum lighting power consumed by the ASCP lamps is 1.75 watts.

The input signals to the ASCP synchro resolvers are four-wire transmitter resolver outputs of either 800 hz, 26 VAC maximum amplitude or 400 hz, 10 VAC maximum aplitude. The output signals are related to the attitude

error by the equation

$$e \text{ output} = E \sin B,$$

where E is a maximum value of 49.6 VAC for an input of 26V, 800 hz or 19.1 VAC for an input of 10V, 400 hz and B is the attitude error.

The ASCP resolver inputs originate in the IMU or GDC, but are enabled and disabled by relays in the EDA. The attitude error outputs are supplied to both the EDA and the GDC.

3.1.5   <u>Gimbal Position/Fuel Pressure Indicator - JG261</u>

The GP/FPI shown in Figure 3.10 is mounted in the low center position of control panel 1 with four bolts. The nomenclature for the scales and the thumbwheels is located on the control panel. The GP/FPI vertical scales are designed for maximum readability and minimum parallax error. The nominal viewing position is at a 22 inch distance from the GP/FPI face. The range of optimum viewing points are between angles of 20° and 45° from the normal to the panel in the vertical direction upward.

<u>Functions</u>

The GP/FPI performs three functions:

- displays the angular position of the SPS pitch and yaw gimbals
- provides a means of manually positioning the pitch and yaw gimbals prior to thrusting
- displays the fuel pressure of the SII booster and the fuel and oxidizer pressure of the SIVB booster

FUEL PRESSURE/GIMBAL POSITION INDICATOR

SPS YAW GIMBAL ANGLE

NEEDLE INDICATOR

S-IVB FUEL PSI

SPS PITCH GIMBAL ANGLE

S-II FUEL/S-IVB OXIDIZER

SPS PITCH GIMBAL SET KNOB

SPS YAW GIMBAL SET KNOB

FIGURE 3.10

3-29
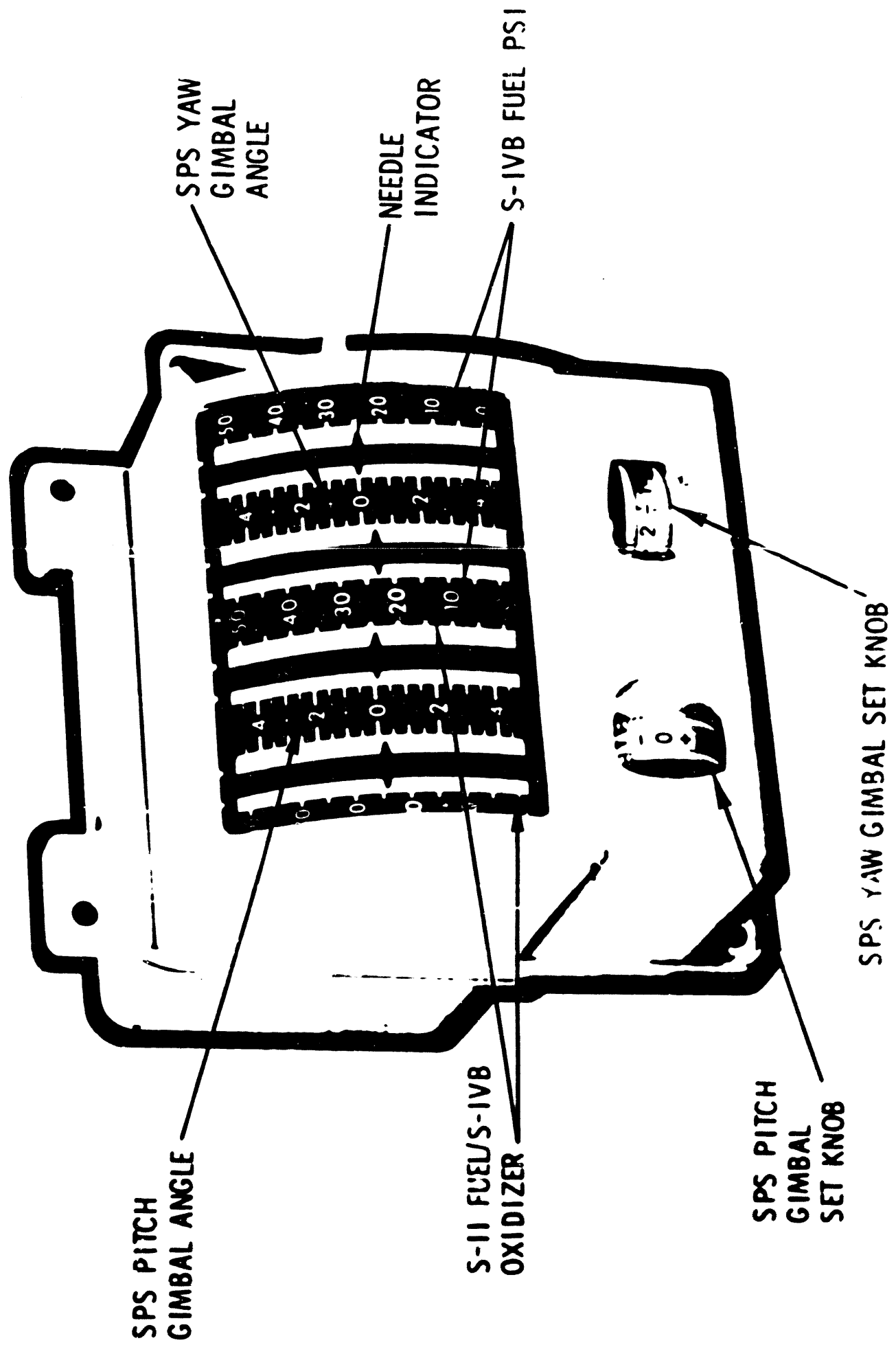
Pitch and yaw gimbal position is displayed by redundant servometric meter movements. Two movements in each axis respond to gimbal position signals from separate control loops in the EDA. The gimbal position indicator displays gimbal position angles between +4.5 and -4.5 degrees in both the pitch and yaw axes.

Two thumbwheels, one for pitch and one for yaw, are provided to manually trim the gimbal position. Each thumbwheel is mechanically coupled to two electrically isolated potentiometers which convert thumbwheel angular position into electrical signals for gimbal positioning purposes.

The servometric meters used to display gimbal position for pitch and yaw are also used to display fuel pressure of the SII booster and fuel and oxidizer pressure for the SIVB booster respectively. The fuel pressure displays indicate pressures in the range of 0 to 50 psi.

Mechanical Features

Electrical access for system operation is made through two 19-pin connectors and 15.5 inch cables. The maximum weight of the GP/FPI including connectors and cables is 2 lbs. 14 oz. The basic elements of the GP/FPI are the case, 2 gimbal trim assemblies, 4 servometric meter assemblies, 5 electroluminescent lamp-scale assemblies, 2 connectors and 2 cables.

The force required to rotate the trim controls is 10 ± 5 ounces at the thumbwheel surface while a maximum of 32 ounces is required to initiate rotation.

The transparent surfaces of the indicator face are coated with anti-glare and reflection-reducing materials.

Electrical Interfaces and Interconnections

115V 400 hz variable lighting power is supplied to the five electrolumin-escent lamps thru the control panel 8 integral lighting rheostat. These lamps consume a maximum power of 0.75 watts. The other power consumption of the GP/FPI is included in that specified for the EDA.

The GP/FPI receives four separate DC signals to drive the torquer coils and +15 VDC reference power for the servometric meter feedback potentio-meters from the EDA. The potentiometer wiper voltage is fed back to the EDA.

The GP/FPI receives +15 VDC and -15 VDC reference voltages for the four gimbal trim potentiometers (two in each axis) from the TVSA. The pot-entiometer wiper voltages, which are proportional to the thumbwheel settings, are fed back to the TVSA for gimbal positioning purposes.

3.2 ELECTRONIC ASSEMBLIES

The SCS system electronics is contained in the following assemblies:

      Gyro Assembly (2 Units)

      Gyro Display Coupler

      Electronic Display Assembly

      Electronic Control Assembly

      Reaction Jet & Engine ON/OFF Control Assembly

      Thrust Vector Position Servo Amplifier Assembly

In general, each SCS assembly is completely independent of the other except for signal flow interfaces. In many cases, component designs used in Block I have been adapted and improved, where possible, for use in Block II. New component designs have been made where system requirements have changed or where significant improvements in weight, power consumption, or reliability were possible.

A reduction in weight and volume was attained by use of hard-wire (welded matrix layers) interconnected modules inside the units (made possible by the deletion of in-flight maintenance), use of integrated circuits (micromin), and electronic switches (when possible) instead of relays.

The over-all system contains approximately 10% fewer electronics parts than the Block I system which contained approximately 9500 electronic and electromechanical parts.

Temperature Control ·

Each of the SCS Electronic Assemblies and the FDAI is cooled by conduction to a cold plate. The bases of the assemblies are machined to precise flatness requirements to ensure maximum heat transfer between the device and the cold plate. The SCS devices are designed to meet all operating requirements with cold plate surface temperature between $55^{o}F$ and $127^{o}F$. The coldplate temperature is controlled by the water-glycol solution pumped through it from the Enviromental Control System. The devices will also meet all operating requirements during entry when coolant flow is

shut off for 2.5 minutes and the coldplate temperature rises to 160°F.
The SCS devices are designed so each module or assembly within the device
provides a means of conducting heat from the submodule or subassembly
directly to the device mounting base.

Maintainability

There will be no in-flight replacement or maintenance of the SCS hardware,
nor any field repair, adjustment or calibration of the devices. The
devices are designed for ease of repair at Honeywell's facility. The
installed SCS is repairable by replacement of the faulty device. Sufficient
signals are brought out to the test connectors to positively isolate a
faulty component. The devices are tested in the bench-maintenance areas
and returned to Honeywell for repair if failure is verified. The only
access allowed to a device, other than the interface connectors, is the test
connector located under a removable cover. Removal and replacement of SCS
devices can be accomplished with standard hand tools. Any SCS device
can be replaced without electrical alignment, calibration or testing of
the SCS.

Failsafety

The SCS is designed to be failsafe. Failsafe is defined to mean that a
failure within the SCS will not:

a) propagate and cause additional functional and/or performance
   failures and/or performance failures within the affected device,

b) induce a failure in any other SCS device,

c) affect the function and performance of a redundant path within the SCS, or

d) induce a failure in, or affect the function and performance of any redundant path of interfacing spacecraft systems.

No fuses are used in the SCS. The need for protection against failure specifically includes; for example, that a failure in one RCS driver amplifier and/or associated switching logic will not cause a failure in any other driver amplifier.

Test Provisions

The SCS provides test points, circuitry and signal conditioning required for operation with Ground Support Equipment, the Telemetry System and Honeywell Test Sets. Ground Support equipment consists of Honeywell Bench Maintenance Equipment, Honeywell Component Test Sets, Honeywell System Test Set and the NAR/NASA Automatic Checkout Equipment. The monitoring of SCS performance after the hatch is closed, during the launch count down, will be by the Telemetry System only. There will be no SCS GSE umbilical monitoring.

The SCS circuitry provides for parallel type testing by the GSE for subsystem spacecraft checkout. This is accomplished by stimulating the SCS and measuring the responses through the GSE checkout connectors, without breaking any interface connections. Test circuitry is provided to:

a) Torque the gyros individually from the GSE in the rate and attitude configurations;

b) Measure the values of interface signals (inter-component and inter-system) as required for correct system operation in the spacecraft;

c) Stimulate each telemetry signal conditioner (without moving the S/C) through the GSE checkout connector;

d) Verify servo control transfer redundancy of the thrust vector control system.

3.2.1 Packaging (See Figure 3.11)

Chassis - The components of each assembly are contained in a welded aluminum alloy enclosure, designed to provide vacuum sealing and electrical continuity. The base (coldplate mounting surface) is held to a close flatness tolerance and is protected by a rigid handling plate when the device is not in use. The chassis is bolted to the coldplate or handling plate with 6 to 10 screws. The cover is fitted with a rubber gasket and clamped with hex head screws to the milled flange on the chassis to provide an hermetic seal. Rubber gaskets are used at the connector and valve sealing surfaces for the same purpose. The seal-off valve is installed to provide a means of evacuating and backfilling the device with an inert atmosphere. A front cover plate (attached with two screws) can be removed for access to the test connector and seal-off valve without breaking the hermetic seal.

Modules - The electronic circuitry in the devices is packaged in modules. The module size is determined by the module frame which is the basic structural member of each module. The frame is cast aluminum, rectangular in shape, with a flange on three sides providing a mounting surface and a

TYPICAL BLOCK II ELECTRONIC ASSY
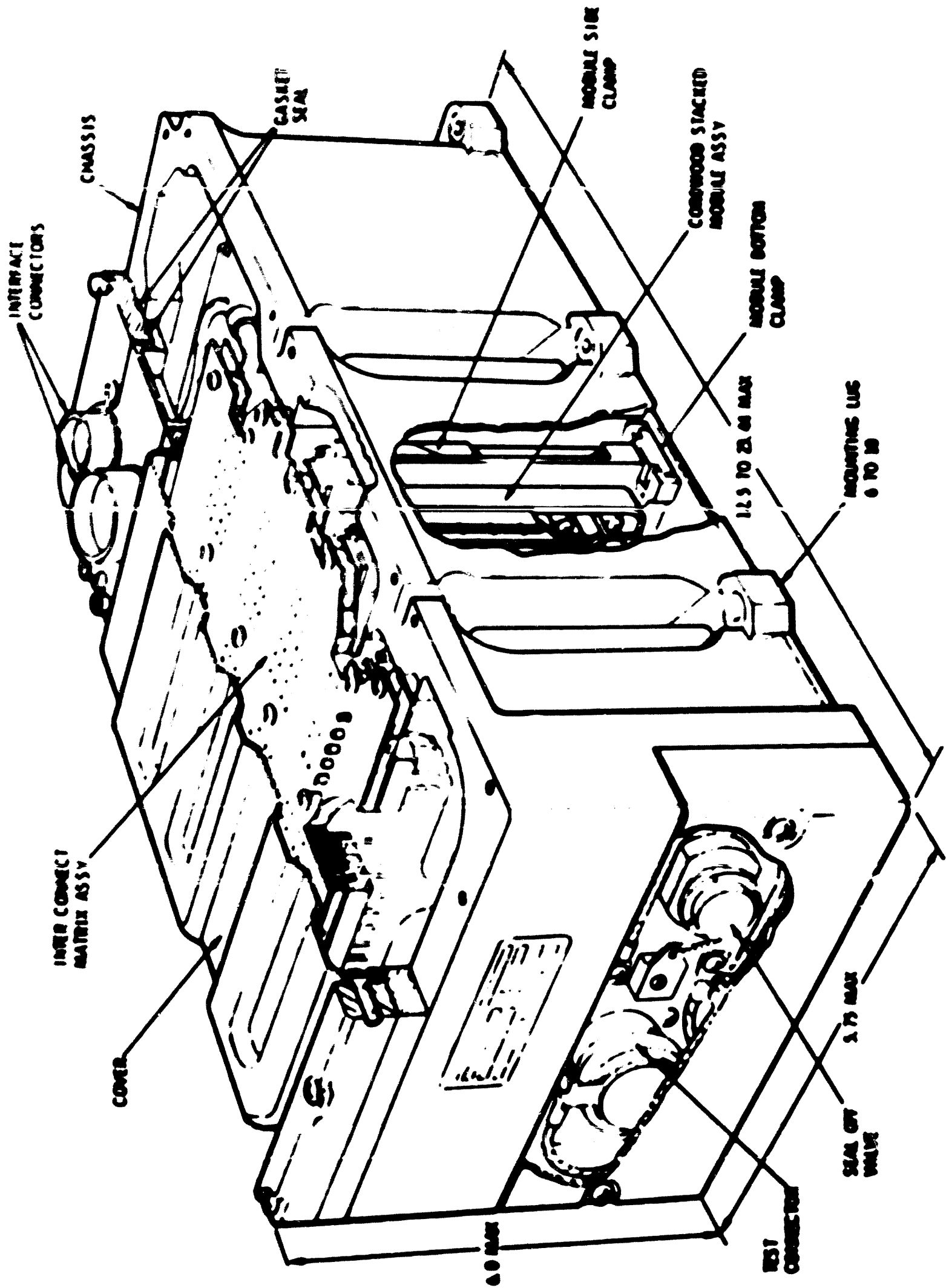
FIGURE 3.11

heat transfer path to the chassis. The module flanges mate with channels provided in the chassis walls. The modules are mounted in the chassis by two screws through the flange at the base of the module and with removable wedges which clamp the module sides on the channels provided in the chassis.

The components are installed in the module per the pattern printed on the .005 inch thick glass epoxy jig wafer. Two jig wafers are used, one on each side of the module frame. The components are mounted directly on the frame or suspended in cordwood fashion between the jig wafers. Holes are drilled in the jig wafers to permit the component leads to extend past the jig wafer face for connecting to other components. The electrical connection between components is made by welding nickel bus wire to the component leads which is routed per the pattern printed on the jig wafer face. Component leads which interface with other modules or with external signals do so through a four row multipin header assembly. The pins of the header are then welded to a matrix. Each module is conformally coated with an epoxy material to provide vibration resistance.

Matrix - (Interconnect Assembly) Two methods are used for internal device wiring. Point to point wiring is used between the device connectors and the matrix assemblies. Module interconnections are provided by the one or two matrix assemblies. A matrix assembly consists of a matrix and top and bottom boards. The matrix is a multilayer (up to 9 layers) arrangement of nickel busses which are routed on both sides of a mylar wafer per the printed pattern. The busses are routed on both sides to allow connections across the module as well as from module to module without crossing the uninsulated

nickel bus material. A hole is punched in the mylar to permit welding a bus on one side to a bus on the other side of the mylar. Multilayers are required to provide the necessary interconnections as well as the separation of various signals to prevent crosstalk. The busses are cemented to the mylar to prevent shorting of adjacent busses. Insulator wafers are used between the layers to prevent layer to layer shorts. The matrix is bolted to the module frames and the chassis for rigid support. Where necessary, the matrix top board has a copper clad on it to provide a heat transfer path and a mounting base for the device calibration resistors.

3.2.2 **Gyro Assembly - GG362**

The SCS contains two identical Gyro Assemblies. Their designations, GA1 and GA2, are determined by their mounting locations in the Command Module lower equipment bay. The Gyro Assembly senses rotational motion about the spacecraft axes and provides scaled signal outputs which provide spacecraft attitude and angular rate information for display and control of the spacecraft attitude and angular rate. When mounted in the spacecraft, GA2 always provides rate information. GA1 can be configured, through Control Panel switching, to provide attitude or backup rate information.

The Gyro Assembly consists of three miniature integrating GG248 gyros (with their input axes aligned in an orthogonal triad) and associated electronics in an environmentally sealed case. The theory of operation of the miniature intergrating gyro is provided in Appendix I. The construction of the electronics in the sensor assemblies is similar to that used for all Block II devices, and is interconnected by welded buses (matrix interconnect layer).

The decision to combine gyros and associated electronics in the same package was reached when the need for in-flight gyro replacement was deleted, after consideration of advantages offered by reduced interconnecting spacecraft cabling, and easier gyro checkout.

The gyro assembly includes externally adjustable drift trim resistors and a test connector separated from the interface connectors.

To meet operational accuracy requirements, the gyro axes must be maintained parallel to the spacecraft axes. To make this possible, the mounting and reference surfaces in the gyro assembly are precisely machined, and the gyros are aligned by spinning-in (to a minimum electrical null), providing the most precise input axis alignment possible. The installation of the gyro assembly must permit final adjustment by means which are compatible with use of the cold plate. The temperature of each gyro is individually controlled to the exact temperature at which calibration was made, eliminating an additional source of error.

The temperature of the bath at which a gyro is calibrated is nominally the operating temperature of the gyro ($\pm 1^\circ$F). The resistance of the temperature sensor element in the gyro corresponding to this actual temperature is recorded; and when the gyro is installed in the gyro assembly, its temperature control amplifier is trimmed to control at the recorded resistance within $\pm 0.1^\circ$F.

The gyro mounting block is designed such that under any combination of environmental extremes of coldplate temperature, ambient temperature, gyro

warm-up time, and with minimum power consumption, the gyro operating temperature will be controlled within the required limits. Thermally-sized epoxy fiberglass inserts for insulating gyros from the mounting block, and polyurethane hoods for the exposed end of each gyro are means used to accomplish this end.

Thermal capability and high density electronic packaging are achieved by using a heat sink structure for supporting heat-generating components and transmitting the heat to the coldplate.

Reliability is enhanced by using advanced welding techniques (such as per-cussive welding) thus eliminating solder connections, normally a significant source of reduced reliability because of component damage and variable quality of joints.

To meet the environmental requirements the gyro assembly is sealed and has a leak rate determined by analysis of the environments including shelf life and mission.

To minimize errors, a sensor installation would have the input axis of the sensor exactly coincide with the corresponding axis of the spacecraft. Practically, it is impossible to locate each gyro exactly at the c.g. of the spacecraft and coincident with each axis. Therefore the sensor is displaced and its sensitive axis maintained parallel with its correspond-ing spacecraft axis.

Mounting and alignment surfaces are provided on the base casting for

positioning in the spacecraft. Mounting surfaces for the gyros are precisely machined, relative to the mounting and alignment surfaces, so that the over-all error between the gyro sensitive axis and the assembly mounting surface is less than 10 arc minutes.

The only remaining controllable cause of error is the installation in the spacecraft. To minimize this error, adjustment must be provided in the spacecraft in all three axes. Placement or location of the alignment surfaces on the gyro assembly must coincide with the stops provided in the spacecraft.

Mechanical Description

The maximum weight of a Gyro Assembly is 22 pounds, 7 ounces.

External - The device housing is a welded aluminum enclosure approximately 6" high, 7" wide, and 14" long. It is hermetically sealed to 1×10^{-6} cc/sec indicated leak rate with 10% helium fill gas. The base of the package has two dowel pin holes in line with the longitudinal axis. The dowel pin holes and flat base define an orthogonal triad to which three internal single degree of freedom gyros are aligned. The device is cooled by conduction through the base by contact to a coldplate surface.

The Gyro Assembly uses three electrical connectors for:

Operational power inputs (J1-19 pin)

Operational signal outputs (J2-37 pin)

Ground check-out (J3-55 pin)

Connectors J1 and J2 are located on the Connector Housing mounted vertically

side by side at one end of the device. Connector J3 is on the opposite
end of the device mounted horizontally and recessed so that a protective
plate over the connector is flush with the end of the device. A device
"fill valve" is mounted on the connector housing beside the J3 connector.
This valve facilitates the purging and filling of the device with a
"dry" fill gas after hermetic sealing.

A "trim window", recessed and protectively covered, is located on the same
end of the device as the J3 connector. This "trim window" allows for
external recalibration of G-insensitive torques. The Gyro Assembly is a
"limited Life" device i.e., it must be recalibrated at 6 month intervals.

Internal - The three gyros are orthogonally mounted such that the roll
and yaw gyro output axes are perpendicular to the base and the pitch gyro
output axis horizontal and along the width of the base. The roll and yaw
gyros are mounted side by side on a beam that goes across the width of
the device. The pitch gyro is mounted in a yoke attached to the device
base on the J3 side of the roll/yaw beam.

A "stack" of five structurally similar modules is bolted to the device base
on the connector housing side of the roll/yaw beam. Each module consists
of a machined aluminum frame casting with mutually perpendicular cross
members. Electronic components within the modules are either mounted
directly to the frame (for maximum thermal dissipation) or are suspended,
cordwood fashion, between two fiberglass jig wafers. Each module is
conformally coated to provide structural integrity. Electrical connections

between the modules are made in a welded matrix. The electrical connections to the welded matrix (and all other electrical interconnections) are hard wired and soldered to terminals.

Electronic components are mounted inside the connector housing which is attached to the chassis by screws. There is a separate rubber faced gasket between the housing and chassis for hermetic sealing. The same type of gasket is used between the cover and chassis. Additional electronic components are mounted around and above the pitch gyro which is on the J3 connector end of the roll/yaw beam.

Spin Motor Excitation

Three phase 115 volt, 400 hz power is supplied to the three GG248 spin motors through three single phase transformers whose primaries are connected in a four-wire wye, neutral grounded, (see Figure 3.12). The secondaries are connected in a three-wire wye with phase B grounded. Each secondary voltage is 15V rms resulting in a line-to-line voltage of 26V rms. Three impedance matching capacitors are connected line-to-line. The transformer secondaries feed the three gyro spin motors which are connected in parallel.

Power Supply

The Gyro Assembly circuits use +20V DC, -15V DC and $\pm$32V DC power. The GA-power supply provides these voltages. The power supply consists of one 3-phase full wave rectifier which provides $\pm$40V DC. The unfiltered voltages are fed to regulators which provide the $\pm$32V DC power. These voltages drive two more regulators which provide +20V DC and -15V DC power for the electronic amplifiers.

SPIN MOTOR SUPPLY

FIGURE 3.12

3-44
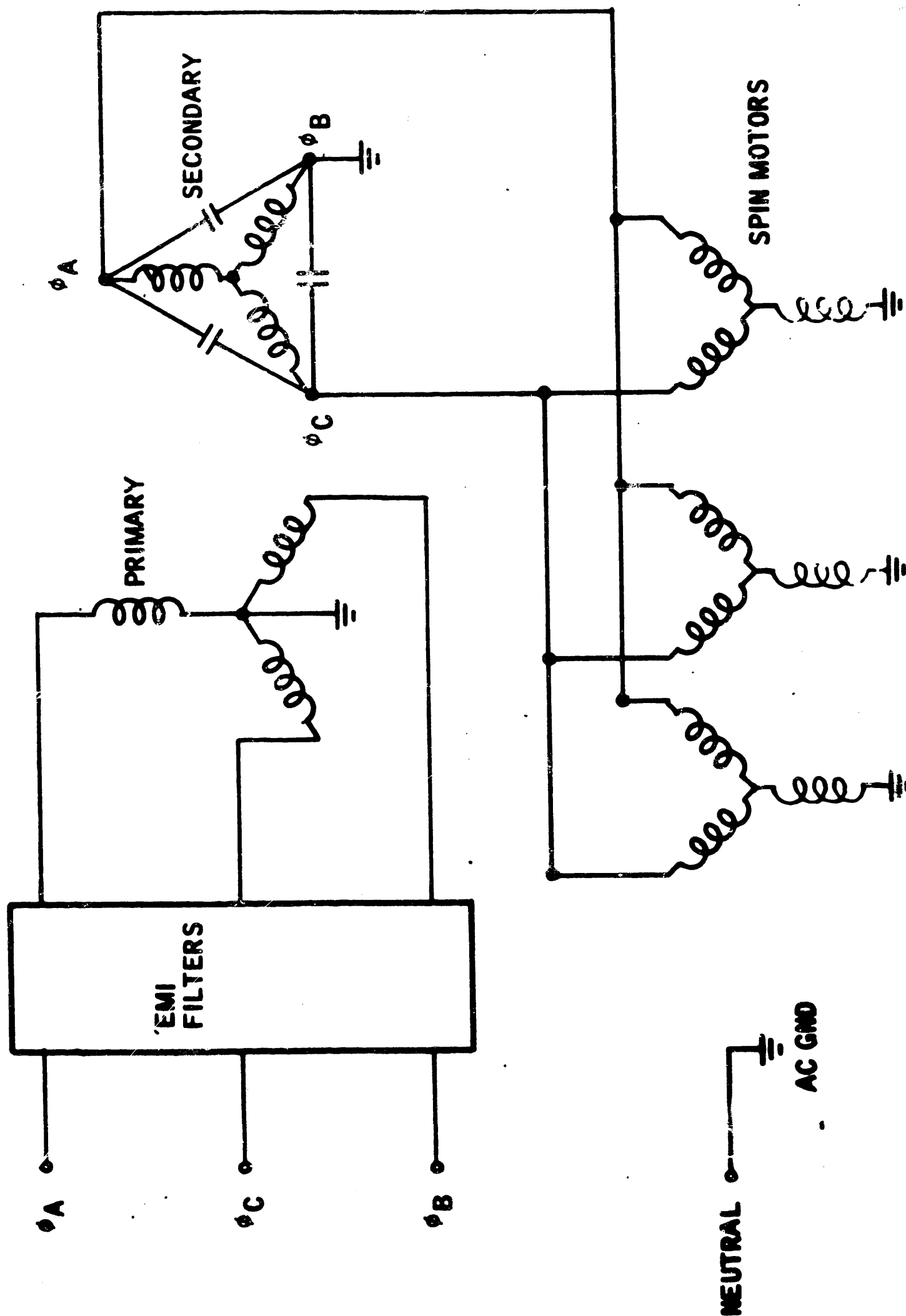
Signal Generator Excitation & Compensation

Excitation is provided to the three signal generators from a transformer supplied with 115V $\emptyset$A 400Hz power. (ref. Figure 3.13) The phase of the excitation voltage is adjustable at the device level through selection of the phase trim resistor. The three gyro signal generator primary windings are connected in series to allow for simultaneous phase shifting of the three gyro output voltages. The phase of the individual gyro outputs is adjusted by means of the phase trim capacitors at the harmonic filter outputs.

Compensation for fixed gyro torques (g insensitive) is made by shunting a portion of the signal generator primary current through the secondary compensation winding. The magnitude of the compensation current in each signal generator is controlled by the value of the drift trim select resistors.

Compensation for the internal elastic restraint of the gyro flex leads is made by selection of proper capacitors shunting the signal generator secondaries. The capacitors allow circulating currents 90° out of phase with the respective S.G. secondary voltages, i.e., gyro gimbal positions. These currents effectively develop torques compensating for the flex lead elastic restraints.

The signal generator quadrature voltage is minimized for each gyro by the quadrature trim circuits. The compensating voltage is developed at the junction of two select resistors supplied by the 20V $\emptyset$A secondary. These $\emptyset$A voltages are fed to the S.G. secondaries through the phase shifting resistor-capacitor networks.

SIGNAL GENERATOR EXCITATION & COMPENSATION CIRCUITRY
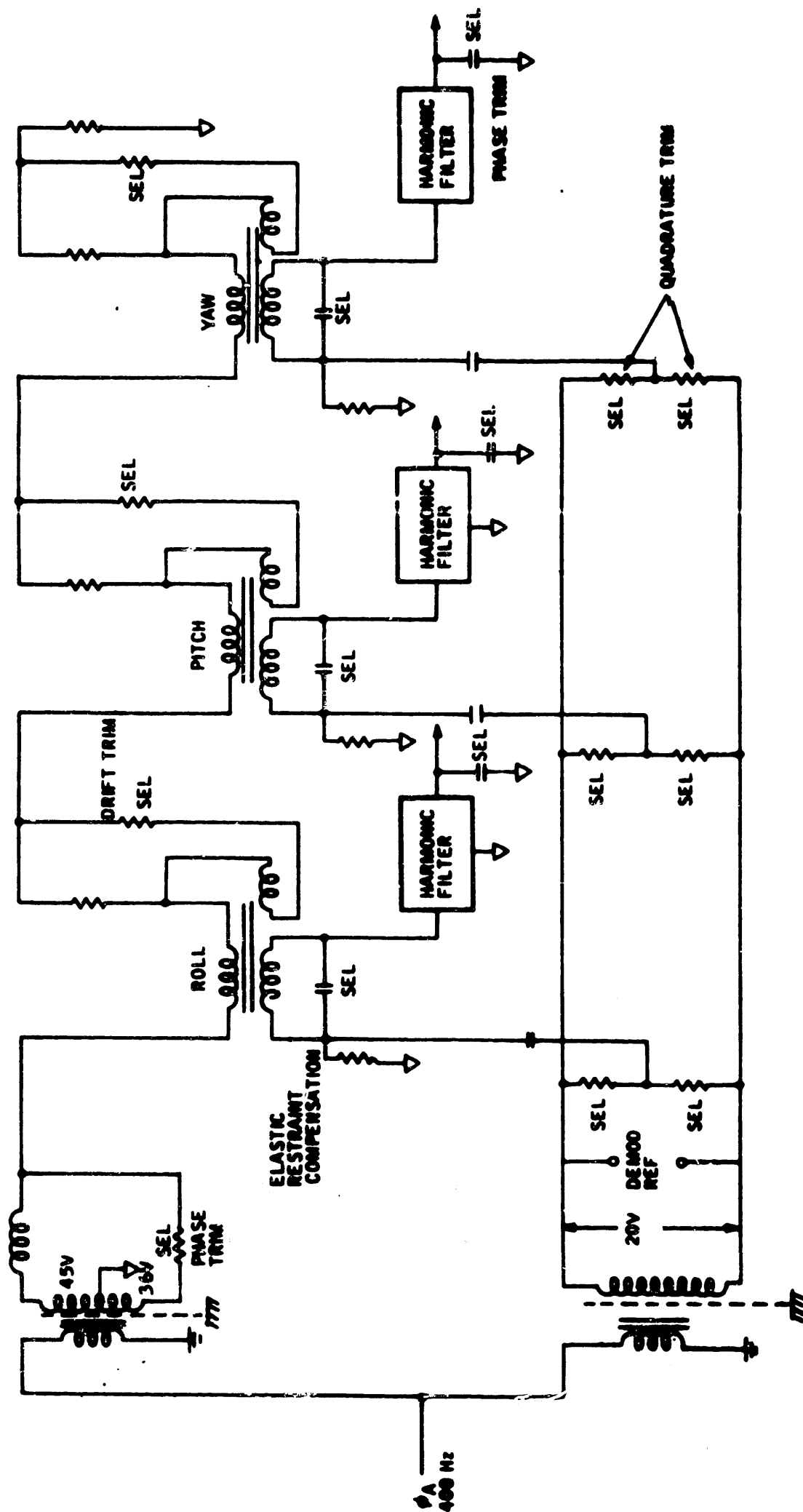
FIGURE 3.13

Loop Electronics

The loop electronics is identical for all three gyros, except for phasing
on the DC rate output. Figure 3.14 shows a block diagram of the loop
electronics for a single axis. The gyro signal generator output signal
is fed through the harmonic filter (bandpass filter with a center frequency
of 400 hz) to the pre amp. The information signal which modulates the
400 hz carrier signal is then amplified by the pre amp with a gain of
18 V/V. The attitude scale factor is controlled by adjustment of this
pre amp gain. The pre amp output feeds a second amplifier whose gain is
27 V/V (Rate Mode) with relay K1 de-energized or 1 V/V (Attitude Mode)
with relay K1 energized. The output of this amplifier feeds the output
transformer. The two secondaries couple isolated outputs to the display
and control electronics in the EDA and ECA respectively. In the rate
mode the amplifier output is fed through relay K1 contacts to a full wave
demodulator. The demod reference voltage is obtained from the 20V
secondary shown in Figure 3.13. The DC rate output polarity is controlled
by the phasing of the demod reference. The demod output is filtered by
an R-C network and amplified by the torquer amplifier which supplies current
through the relay contacts to the gyro torquer coil and current sampling
resistor. The DC rate output signal is developed across the selected
current sampling resistance (nominally 10 ohms) which determines the output
scale factor. This output signal is also fed back to the torquer amplifier
to control its transfer function. A low impedance feedback path is provided
around the torquer amplifier in the attitude mode to keep its output at
null.

GYRO LOOP ELECTRONICS
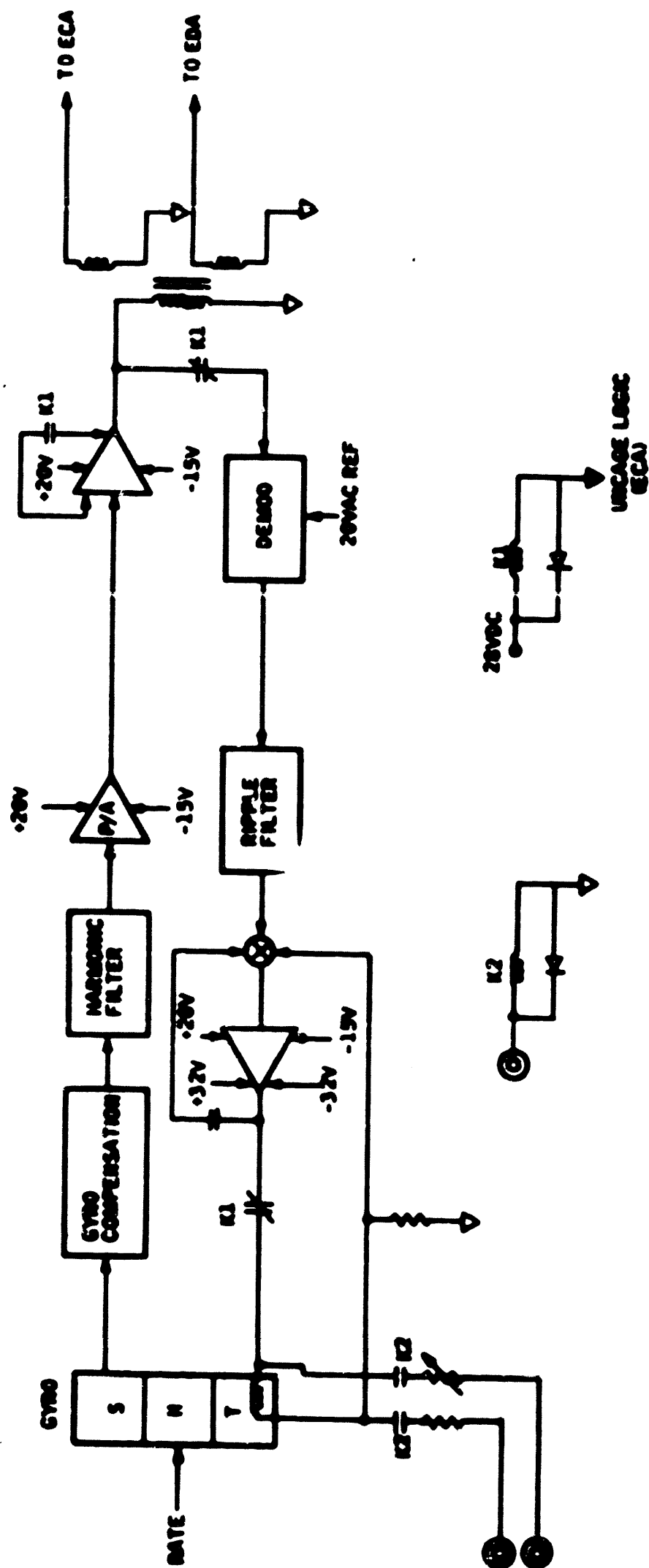
FIGURE 3.14

Relay K2 provides the capability of electrically torquing the gyro with a test signal. Application of 28V DC to the K2 coil will close the normally open K2 contacts. Application of a DC test signal to the appropriate test points will provide a current through the torquer coil when relay K2 is energized.

Spin Motor Rotation Detector (SMRD)

The function of the SMRD circuitry is to provide a telemetry signal which indicates the operational status of the gyro spin motors. Input signals for the SMRD circuitry are obtained from pick-up coils within the gyros. (see Figure 3.15) Three identical circuits (one for each gyro) provide outputs to a gating transistor. The gating transistor is turned on when all rectifier outputs reach the switching level, providing a 5V DC telemetry output. If one or more of the rectifier outputs is below the switching level, the gating transistor remains off and a 0 VDC telemetry signal is provided.

The gyro pick-up coil provides a 0 to 800hz signal (proportional to the spin motor speed) to the amplifier which is tuned for maximum gain at 800hz. The amplifier output voltage is rectified and scaled such that the spin motor must be operating at a minimum of one-half synchronous speed to provide an ON command to the gating circuit. All three rectifiers must provide ON commands for the gating transistor circuit to supply the 5V DC TM output.

SCS GYRO ASSEMBLY NO.1

SMRD TM

EQUIVALENT OF GATING TRANSISTOR CIRCUIT

CH3623X

CH3624X

CH3623X - GYRO 1 COMB SPIN MOTORS RUN DET

CH3624X - GYRO 2 COMB SPIN MOTORS RUN DET

GYRO ASSY NO. 2 (IDENTICAL CKT)

RECT

RECT

RECT

800 CPS NOTCH FILTER

800 CPS NOTCH FILTER

800 CPS NOTCH FILTER

AC AMP

AC AMP

AC AMP

+28 VDC

+28 VDC

+28 VDC

ROLL SMRD COIL

PITCH SMRD COIL

YAW SMRD COIL

FIGURE 3.15

3-50
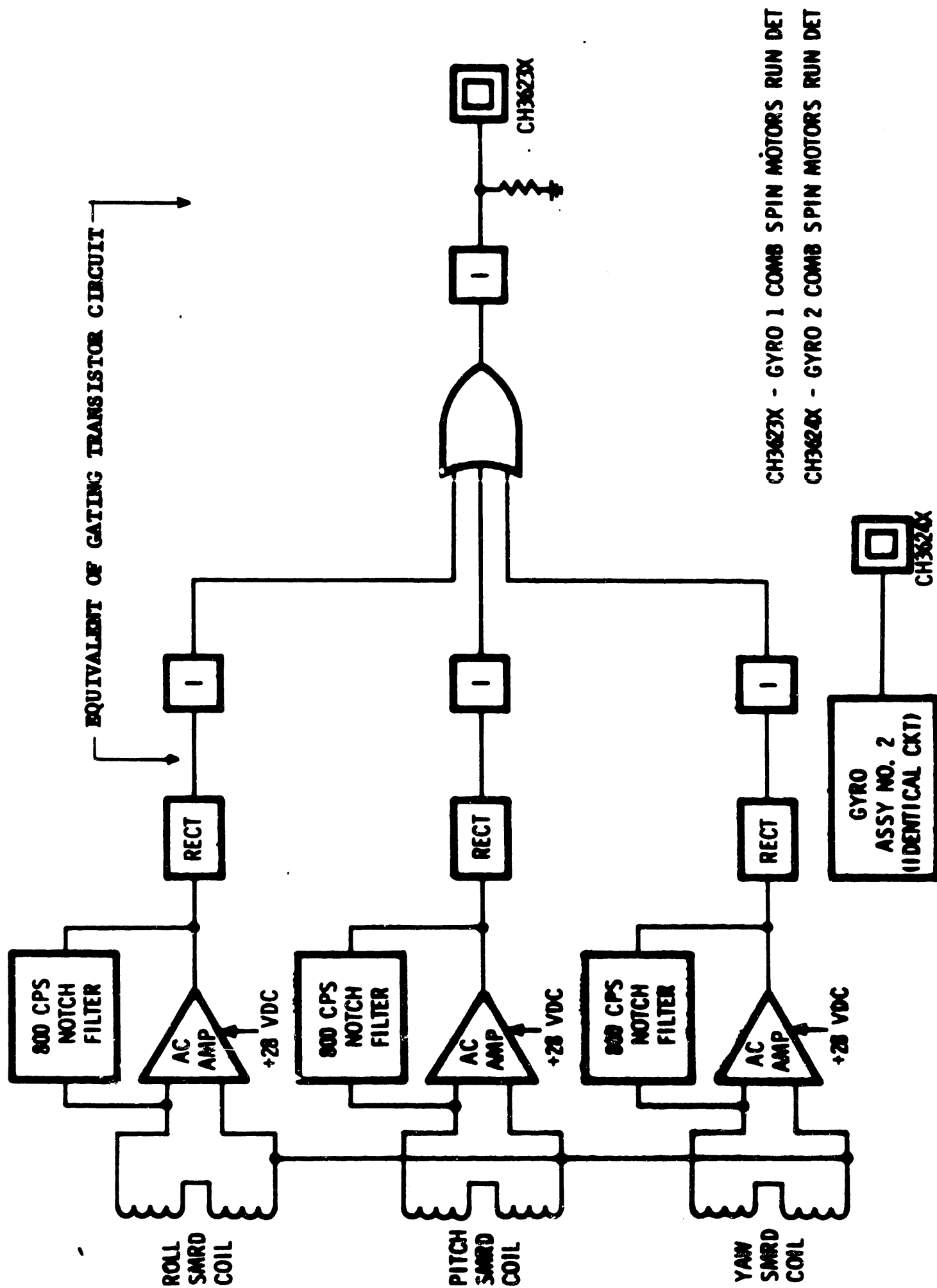
TCA/TIA Circuitry

The MIG GG248 is designed for operation at 170°F. The temperature is critical in that it determines the viscosity and density of the fluid and thus affects the gyro parameters. The gyro is maintained at the proper operating temperature by controlling the power applied to the gyro heater. This function is provided by the temperature control amplifier circuit. It is imperative that the gyro temperature status be know by the crew. The temperature indicating amplifier circuitry provides the necessary signals to control the status lamps (outband lamps) on Control Panel 2 as well as the discrete TM signal which indicates the gyro temperature status.

The TCA/TIA mechanization is shown in Figure 3.16. The TCA circuit consists of the amplifier and the gyro temperature sensor (780 ohms at 170°F) which is one leg of a 780 Ω bridge circuit. When the bridge is unbalanced (cold gyro) the error signal from the bridge is amplified and turns on an output transistor in series with the gyro heater. When the bridge approaches balance (gyro gets warm) the output transistor turns off sufficiently to stabilize the gyro temperature to within a 1°F band.

The TIA bridge circuits are in parallel with the TCA bridge circuits. The TIA outputs are ORed into the lamp and TM driver which provides a ground output for the outband condition and +28V DC (+5V DC at the voltage divider output to TM) for the inband condition. The inband condition is satisfied when all three gyro temperatures are between 168°F and 172°F. The outband condition is satisfied when one or more of the gyro temperatures is outside the 168 - 172°F range.

TCA/TIA CIRCUITRY

ROLL TCA/TIA

+28VDC

780Ω   780Ω

REF   REF

780Ω   ROLL GYRO TEMP SENSOR

+28VDC

ROLL TCA

ROLL TCA/TIA BRIDGE

+28VDC

ROLL TIA

ROLL GYRO HEATERS

+28VDC

CONTROL PANEL

+28VDC

OUTBAND IND

+28VDC

DRIVER

T/M

PITCH TCA/TIA

SAME AS ROLL

PITCH GYRO HEATERS

+28VDC

YAW TCA/TIA

SAME AS ROLL

YAW GYRO HEATERS
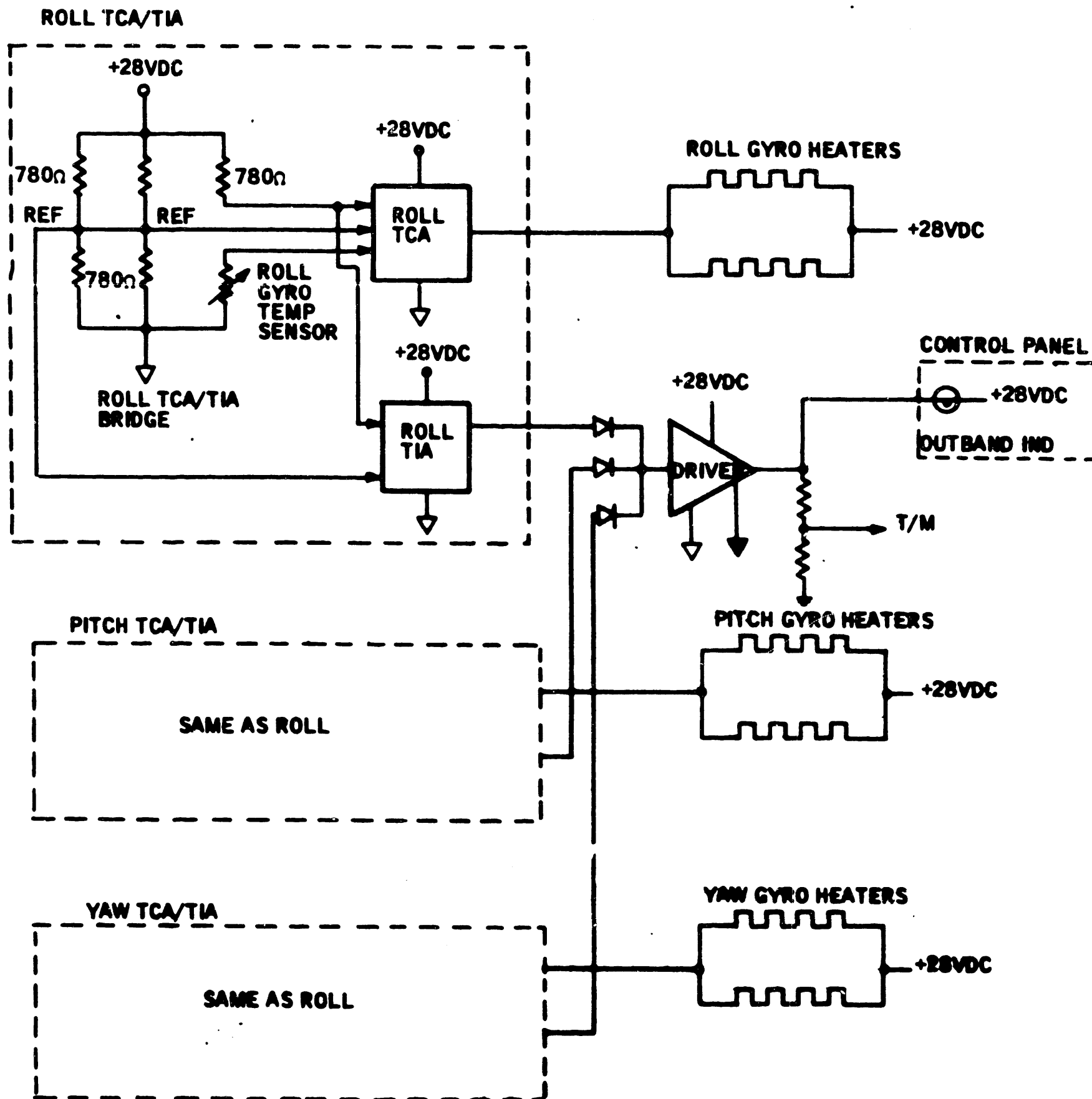
+28VDC

FIGURE 3.16

Gyro Assembly Performance Data

A GA consumes a maximum of 60 watts steady state plus 40 watts transient 28V DC power. Maximum 115V AC power consumption is 34.6 watts steady state plus 86.1 watts transient.

Listed below are a number of GG362 parameters, characteristics and performance requirements:

Spin Motor

Synchronous speed	24,000 rpm
Starting power	5 watts
Running power	3 watts
Run-up time	25 sec, max.
Run-down time	45 sec, min.

Temperature Sensor

Resistance	$780 \pm .2 \Omega$ @ $170 \pm 1 °F$
Current	10 ma
Sensitivity	$1.5 \Omega/°F$ @ $170°F$

Heaters

Voltage	28V DC
Power	28 watts max (2 heaters)

Mechanical Performance Specifications

Angular Momentum	1×10^5 gm-cm^2/sec
Time Constant	400 microseconds max.
Gimbal Inertia	200 gm-cm^2

Damping Constant	460,000 dyne-cm/rad/sec
Gimbal Freedom	$\pm 4.4°$
Input Angle Freedom	$15°$min, $25°$max

Input-Output Performance Specifications

Input Rate	$50°$/sec max (30 sec ON, 60 sec OFF) $25°$/sec continuous
AC rate output	$.125v/°$/sec $\pm 1.75\%$ (400hz)
AC rate output null	5.0 mv max.
DC rate output	$.1 \ v/°$/sec $\pm 0.3\%$
DC rate output null	.05 mv max.
AC attitude output	$.3V/°$IA $\pm 3.5\%$ for $0°$ to $8°$ IA $\pm 5.5\%$ for $8°$ to $15°$ IA
AC attitude output null	15.0 mv max.
g – insensitive drift	$1.73°$/hr.
g – sensitive drift	$4°$/hr/g
g^2 – sensitive drift	$0.42°$/5 minute period during entry
Loop gain	$65°$/sec per degree, min

3.2.3   <u>Gyro Display Coupler - BG289</u>

The GDC is the computing component of the SCS Attitude Reference Display Subsystem. It is essentially a computer which is mechanized to perform transformations between two reference frames, i.e. the fixed (inertial) reference and the movable (spacecraft) reference axes. As its name implies, the GDC provides the interface between the SCS attitude sensors (gyros) and the displays (FDAIs). The functional mechanization of the GDC is described in section 4.9.

Mechanical Description

The maximum weight of the GDC is 25 pounds, 4 ounces. The electronics for the signal conversion is contained in ten interconnected modules within the environmentally sealed chassis. Electrical access for system operation is provided through two 55-pin connectors. A 55-pin connector is provided for interface with the ground test equipment. The ten modules containing all the GDC electronics are:

A1	Modulator
A2	Pitch Voltage to Frequency Converter (V/F)
A3	Roll V/F
A4	Yaw V/F
A5	Regulator and Oscillator
A6	Yaw and EMS Digital Output
A7	Roll and Pitch Digital Output
A8	Power Supply
A9	Align and Logic
A10	Resolver and Amplifier Assembly

Electrical Interfaces and Interconnections

The Gyro Display Coupler Assembly provides the necessary interface between the body rate sensors and the FDAIs to give an accurate indication of the spacecraft attitude relative to a given reference coordinate system.

The GDC interfaces with the CMC of the G&N system, the spacecraft control panel, the Entry Monitor System, and GSE. The GDC also has interconnections with the EDA, the Gyro Assemblies and the ASCP of the SCS.

The GDC accepts DC body rate signals from the Gyro Assemblies, 400hz
Euler Attitude Error signals from the ASCP, and 28vdc logic signals,
28vdc power and 115vac 400hz power from the S/C Control Panels. The
GDC conditions the DC body rate signals and provides digital pulses to the
G&N computer and to the Entry Monitor System. The GDC Euler attitude
error signals from the ASCP are transformed to body referenced error
signals in the GDC and fed to the EDA to drive the attitude error needles.
The GDC Euler attitude signals are also routed to the EDA to drive the
FDAI ball.

The GDC consumes a maximum of 49.9 watts steady state plus 10 watts
transient 28vdc power. Maximum 115vac power consumption is 60.2 watts
steady state plus 33.5 watts transient.

Modulator Module

The modulator converts the D.C. rate signals received from the GA into
2.2K hz sine wave outputs for use in the Euler Mode. The output is
proportional to the input in magnitude, and reverses phase for input
polarity reversal. The design features an internal gain which is
inversely proportional to signal input to provide a signal-to-noise ratio
that is much higher than a constant AC gain modulator. Hence, the output
waveform is relatively free of noise. The module contains two identical
modulator circuits. One receives inputs from the pitch channel of either
GA1 or GA2. The other is fed by the Yaw channel of either GA. Each mod-
ulator output feeds into a separate roll resolver buffer amplifier.

Voltage to Frequency Converter Modules

The Voltage to Frequency Converter (V/F) converts analog voltage signals into proportional digital output pulses. The GDC contains three V/F modules. Each V/F provides two outputs, one for each input polarity. The signals converted by the V/Fs are as follows:

1) DC body rates from either GA in the Single Axis Mode.

2) 2.2Khz AC euler rate from the resolvers in the Euler Mode.

3) Demodulated 400hz attitude set signals from the align demods in the Align Mode.

4) The sum of yaw DC body rate and roll DC body rate from a GA (to provide roll stability attitude information) in the .05g (Entry) mode. (roll and yaw V/Fs only)

The V/F consists of an integrator, a 2.2Khz demodulator, and complementary positive and negative reset circuits. See Figure 3.17.

The integrator is a chopper stabilized amplifier (CSA) with capacitive feedback. The paths followed by the high and low frequency inputs are indicated on the figure. The low, or signal frequencies are less than 40hz. The nominal open loop gain of the CSA (which determines the basic integrator accuracy) is 10^7 with a 50K ohm input resistor.

The demodulator is a half-wave field effect transistor demodulator. The FET zero offset characteristic along with pinch-off delay spike cancellation provide a low threshold device.

The complementary resets (only one is shown in the figure) are constant

V/F BLOCK DIAGRAM

FIGURE 3.17

3-58
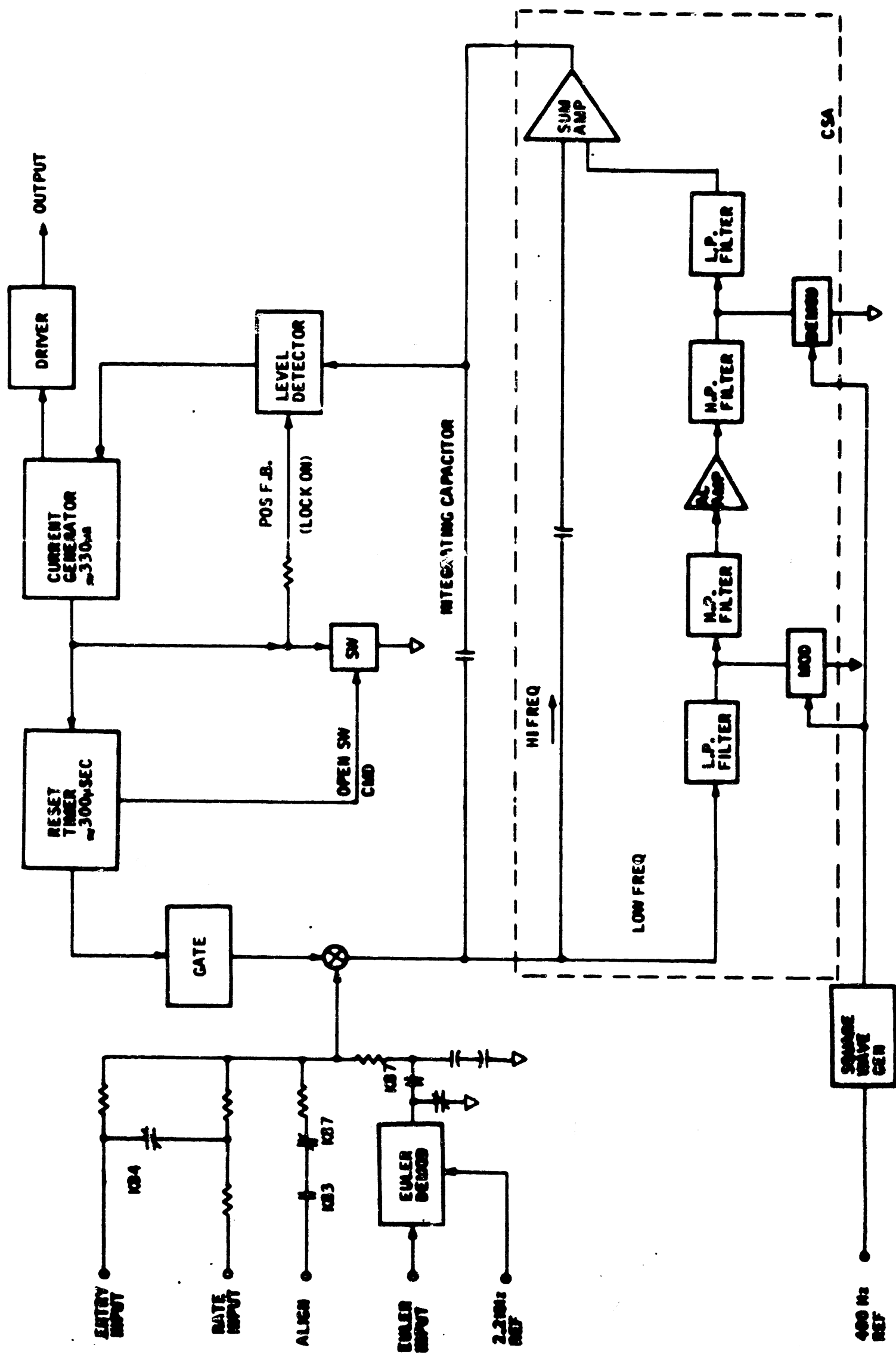
charge resets. Constant charge is realized by a precise current flow for a precise time.

The scaling of the V/F is shown by the equation:

$$\text{frequency output} = \frac{\text{input current}}{(\text{reset current})\ (\text{reset time})}$$

Input current is the net D.C. current to the CSA sum point from the demod or the D.C. input network. Thus the normal input is being integrated even during the reset time. This is necessary to obtain the high degree of precision required from the Voltage to Frequency Converter.

The D.C. rate input in the Single Axis Mode is applied to the CSA whose output is the integral of the input. When the output reaches 1.5 volts, a reset circuit trips resulting in a pulse output which resets the integrator toward zero. The rate of integration is dependent on the amplitude of the input. Applying the opposite polarity input drives the integrator in the reverse direction and in turn operates the complementary reset circuit.

In the Euler Mode, the input signal is received from the 2.2Khz resolver and fed to the demodulator. The demodulated signal is then applied to the CSA. In the Align Mode, the demodulated 400hz align signal is applied directly to the CSA. In the Entry Mode, the yaw and roll D.C. rate signals are scaled and summed prior to application to the CSA.

The V/F outputs are fed to the Digital Output modules. The outputs

(2 for each V/F) are taken from the current generator circuits in the complementary reset circuits and fed through a driver, or buffer stage, which is also a part of the reset circuit.

Regulator and Oscillator Module

The Regulator and Oscillator Module provides regulated voltages for power and reference as needed in the other GDC modules. The voltages generated are +20vdc, -20vdc and 5vrms 2.2Khz. Two identical D.C. regulators are utilized to provide the required redundancy for the Entry Mode. One regulator powers the pitch and yaw axes modules while the other powers the roll axis modules. The oscillator provides the 2.2Khz reference voltage for the modulators and the V/F modules.

Digital Output Modules

The function of the Digital Output Modules is to convert the digital pulses provided by the V/F modules into the required pulses for:

1) The body rate signals to the CMC (roll, pitch and yaw axes)

2) Controlling the direction and rate of the stepper motors in the Resolver Assembly (roll, pitch and yaw axes)

3) Controlling the direction and rate of the stepper motor in the Entry Monitor System (GDC yaw channel only)

There are two Digital Output modules, one for roll and pitch and the other for yaw and EMS. Each module contains two up-down binary counter circuits and two stepper motor drive circuits. The up-down counters provide division of the V/F pulses by two for the pulses fed to the CMC; by four

3-60

to pulse the roll, pitch and yaw stepper motor drive circuits; and by eight to pulse the EMS stepper motor drive circuit. The stepper motor drive circuit for the EMS differs from the other three only in the size of the voltage dropping resistor needed to provide the proper voltage amplitude at the stepper motor coils.

The up-down counter contains two flip-flop circuits and four and-gate logic stages to provide staircase operation for the bi-directional output pulses. An inhibit function is available for preventing the division by four and eight, thereby inhibiting pulses into the stepper motor drive circuits but retaining the outputs to the CMC. Each CMC pulse is equivalent to 0.1° of axis rotation. Each pulse fed to the roll, pitch and yaw stepper motor drive circuits is equivalent to 0.2° of axis rotation, while each pulse to the EMS stepper motor drive circuit represents 0.4° of the GDC yaw axis rotation.

The stepper motor drive circuit has two double ended flip-flop circuits to provide for alternate reversal for both directions. The flip-flops control the excitation phase to the stepper motor coils. Each input pulse reverses the excitation phase, causing the motor to step one increment. The steering gates control the direction of rotation of the stepper motor rotor.

Power Supply Module

The Power Supply Module contains two 400hz transformers, EMI filters and two rectifier circuits. Passive electromagnetic interference (EMI) filters

are incorporated on all incoming lines. The step-down transformers provide 25 vac for resolver excitation, 10 vac for demodulator reference and the necessary inputs to the rectifier circuits. Each rectifier supplies +30vdc and -30vdc to the voltage regulators in the Regulator and Oscillator module. Redundant elements are provided to meet the redundancy requirement during entry.

Align and Logic Module

The Align and Logic module contains three 400hz chopper-demodulator circuits and the logic elements necessary to control the relays which configure the GDC circuits for one of four modes.

Each demodulator receives a 400hz signal from an Attitude Set Control Panel resolver. When the Align Mode is enabled the demod supplies a DC output to the corresponding V/F circuit. The V/F processes this signal and feeds it down stream to align the GDC resolver shafts to the same angles as the respective ASCP resolvers.

The logic circuitry receives three switching signals from Control Panel switches. The circuitry performs the necessary summing of the signals to control four sets of relays in GDC modules. The states of these relays control the mode configuration of the GDC.

Resolver and Amplifier Module

The Resolver and Amplifier module contains the components necessary for performing the body to euler and euler to body transformations. The module contains three stepper motors, three gear trains, three resolver assemblies and six amplifier circuits.

The three stepper motors, one for each axis, are identical. The stepper motor receives excitation pulses from its respective digital output circuit and steps its rotor through a 45° increment for each pulse received. The rotor shaft is mechanically coupled to the gear train input. The gear train provides a 225:1 reduction. The resolver shaft is mechanically coupled to the gear train output. Hence, the resolver shaft is driven in 0.2° increments to correspond with the commanded GDC axis rotation. The stepper motors can drive the resolver shafts at rates up to 60°/ sec.

The module contains three resolver assemblies, one for each axis. The roll and yaw resolver assemblies are identical, having three separate pancake type resolvers in tandem on a single shaft. The pitch assembly is a single pancake type 400hz resolver transmitter on a shaft. The roll and yaw resolver assemblies each contain an identical 400hz resolver transmitter and two computational resolvers. One of the computational resolvers (body to euler rate) is operated at 2200hz while the other (euler to body attitude) is operated at 400hz. The body error outputs for display are fed directly from the computation resolvers to the EDA. However, the euler attitude information for display is obtained from the resolver transmitters of the respective axis.

GDC Assembly Calibration

After the ten modules are assembled in the device, the GDC is calibrated to achieve the state-of-the-art accuracy imposed by the system requirements. There are a total of 57 resistors selected during the device calibration.

A7   Attitude Error

A8   Attitude Reference Relay No. 1

A9   Attitude Error Relay

A10 Signal Conditioner

A11 GP/FPI No. 1

A12 GP/FPI No. 2

A13 Power Supply

A14 Reference Transformer

A15 Power Transformer

Electrical Interfaces and Interconnections

The EDA provides the necessary interface between the information generating units of the SCS and other S/C systems and the SCS displays, namely the FDAIs and the GP/FPI. (NOTE: ordeal in the only exception - its input is between the EDA and the pitch axis of the FDAI ball drive).

The EDA interfaces with the IMU and CDU of the G&N system, the SII fuel tank pressure system and SIVB fuel and oxidizer tank pressure systems, the S/C Control Panel, the Telemetry system and GSE. The EDA also has interconnections with the ECA, GDC, GA2, FDAI 1, FDAI 2, GP/FPI, ASCP and TVSA devices of the SCS.

The EDA consumes a maximum of 50.2 watts steady state plus 10 watts transient 28vdc power. Maximum 115vac power consumption is 34.4 watts steady state plus 20.2 watts transient. This allocation includes all AC power sent to the diplays, except for lighting.

Functional Description

The EDA contains the electronics required to drive the system displays and provides the electrical interface between the system and the spacecraft telemetry equipment. The basic functions performed by the EDA are:

1) Accept and condition all display input signals.

2) Provide all the displays signal and power switching functions.

3) Position the servometric meters in the FDAIs and the GP/FPI.

4) Operate the FDAI attitude servo loops.

5) Condition signals for telemetry.

6) Generate all required reference power not supplied by the S/C power source, the G&N system or the Gyro Assemblies.

The following paragraphs elaborate on these functions and relate them to the EDA modules and the SCS subsystems to which they are applicable. Redundant circuitry, with the exception of logic, is provided for each display function to prevent a single failure from disabling both displays.

Attitude Rate Display

The attitude rate display function requires electronics for six independent servo loops (separate circuits for each axis and for each FDAI). The mechanization is described in section 4.10.3. The components used in this mechanization are contained in modules A3 and A5. 400hz AC rate signals are received from GA1 and GA2. Twenty-eight solid state switches inhibit input signals and provide the gain changes for each mode. Eight AC amplifiers amplify the rate input signals. Six demods demodulate the amplified AC signal. Three relays (six sets of contacts) provide the proper 400hz

reference to the demods. Six dc amplifiers drive the FDAI servometric rate indicators. Two relays (3 sets of contacts) feed the proper DC rate signal to the signal conditioners.

Attitude Error Display

The attitude error display function also requires electronics for six independent servo loops. The mechanization is described in section 4.10.2. The components in this mechanization are contained in modules A7 and A9. Attitude error signals are received from the following sources:

1) 800hz signals from the CDU in the G&N system.

2) 400hz signals from GA1.

3) 400hz or 800hz signals from the ASCP.

   Error information from the ASCP is either:

   a) the difference between the IMU attitude signals and the attitude dialed into the ASCP, or

   b) the difference between the GDC information and the attitude dialed into the ASCP.

   The two sources are obtained by exciting the ASCP resolvers from either the G&N or the GDC. This switching function is accomplished by 6 relays (6 sets of contacts) in module A8.

Thirty-four solid state switches enable or inhibit input signals and provide the gain changes for each mode. Eight AC amplifiers amplify the attitude error input signals. Six demods demodulate the amplified AC signal. Three relays (six sets of contacts) provide 400hz or 800hz reference power to the demods. Six DC amplifiers drive the FDAI servometric attitude error

indicators. Two relays (3 sets of contacts) route the proper DC attitude error signal to the signal conditioners. (A single set of contacts of one of the relays in module A5 is also used).

Attitude Display

The attitude display function requires electronics for six servo loops to drive the two FDAI balls. The mechanization is described in section 4.10.1. The components in this mechanization are contained in modules A4, A6 and A8. 800hz attitude signals are received from the IMU gimbal resolvers and 400hz attitude signals from the GDC resolver transmitters. Six relays (12 sets of contacts) inhibit either of the inputs to either of the FDAI channels. Six relays (6 sets of contacts) enable the signal flow paths to either of both FDAIs. Two relays (4 sets of contacts) control application of the 400hz reference power to the FDAI motor - velocity generator excitation windings. Six AC amplifiers demodulate the amplified error signals. Two relays (4 sets of contacts) provide 400hz or 800hz reference power to these demods. Six 400hz demods demodulate the feedback signal from the FDAI velocity generators. Six DC amplifiers amplify the DC error signals. Six 400hz modulators modulate the amplified DC error signal. The modulator outputs are fed to the motor control windings.

Gimbal Position/Fuel Pressure Display

The EDA provides four channels of electronics to drive four GP/FPI indicators. The indicators display gimbal position or booster tank pressure. Separate scale markings are provided for each function. The

mechanization of the four servo loops is described in section 6.5.5. The components used in the mechanization are contained in modules All and A12. DC pitch and yaw gimbal position signals are fed to this module from the four position transducers in the Gimbal Actuator Assemblies of the SPS system through the TVSA. SII and SIVB tank pressure signals are received from four transducers, two in each of the respective tank pressure systems. Four relays (4 sets of contacts) inhibit gimbal position or tank pressure input signals. Four DC amplifiers amplify the difference signals (sum of selected signals and the indicator feedback pot wiper voltages) and drive the servometric meters. Four relays (4 sets of contacts) alter the gain of the DC amplifiers as a function of the mode selected. Four relays (4 sets of contacts) provide the zero reference voltage required for the mode selected.

Signal Conditioner

The EDA contains the electronics for 13 analog signal conditioner channels. The signal conditioning channels provide the ground isolation, biasing and filtering required to condition the input signal for the spacecraft telemetry equipment. The operational amplifiers used in the mechanization are contained in module A10. The amplifiers condition and provide to the telemetry system the following signals: roll, pitch and yaw displayed rate; roll, pitch and yaw attitude error; pitch and yaw SPS gimbal position; roll, pitch and yaw commands by the Rotation Control; and Pitch and Yaw TVC total error.

Two additional signal conditioner channels (for pitch and yaw differential clutch current to the SPS servo actuators) are located in the ECA.

Logic

Input signals to the EDA must be conditioned to provide the proper display output for different modes of operation. Different mode configurations are established by the EDA logic which controls relays and electronic switches which in turn control amplifier gains and allow input and output lines to be enabled or inhibited. The EDA logic states are determined by logic inputs from the spacecraft control panel switches. The components in the mechanization are contained in modules A1 and A2 and consist of resistors, diodes and switching transistors.

Power Supplies and Transformers

The power supplies and transformers required for the display and telemetry functions provided by the EDA are contained in modules A13, A14 and A15. Separate power supplies are energized by AC bus 1 and AC bus 2 ØA to provide regulated $\pm$ 15V DC power to all No. 1 and No. 2 display channels respectively. This excitation voltage is provided to the display control amplifiers in the EDA and the feedback potentiometers of the servometric meters in the FDAIs and the GP/FPI. The power transformers for these two power supplies also provide the demod reference voltages and the FDAI motor - velocity generator reference to the respective attitude display channels. The third power supply is fed by AC bus 1 and provides regulated $\pm$ 15V DC, -10V DC and -2.5V DC to the signal conditioner amplifiers.

Five reference transformers in module A14 provide the remaining reference voltages required to drive the displays. One transformer receives 28V 800hz power from the IMU and provides 0° and 180° phased reference power to the attitude error and attitude display circuits. Two transformers receive 115V 400hz power from GA1 and GA2 respectively, and supply 0° and 180° phased reference power to the rate and attitude error display circuits. Two transformers receive 115V 400hz ØB and ØC power (one from AC bus 1 and the other from AC bus 2) and supply a single phase (90° out of phase with ØA) reference to the modulators in the FDAI ball drive circuits.

3.2.5 Electronic Control Assembly - BG286

The primary purpose of the ECA is to provide the switching and control electronics required for backup control of the Apollo Spacecraft. It also provides the monitoring, isolation and voltage scaling required for signal conditioning.

Mechanical Description

The maximum weight of the ECA is 16 pounds, 11 ounces. The electronic circuitry is packaged in ten interconnected modules within the environmentally sealed chassis. Electrical access for system operation is provided through two 41-pin connectors and two 55-pin connectors mounted at the rear top surface of the package. A 55-pin test connector located at the front of the package provides connections for the Ground Support Equipment. The modules containing the ECA electronics are:

      A1   Pitch Thrust Vector Control

      A2   Pitch Reaction Jet Control

A3   Rotational Control

A4   Yaw Reaction Jet Control

A5   Yaw Thrust Vector Control

A6   Roll Reaction Jet Control

A7   Power Supply

A8   Gyro Uncage Logic

A9   Signal Conditioning

A10 Manual TVC Logic

Electrical Interfaces and Interconnections

The ECA interfaces with the S/C Control Panel and GSE. The ECA has

interconnections with the GA1, GA2, RC1, RC2, RJ/EC and TVSA devices

of the SCS.

The ECA consumes a maximum of 14.4 watts steady state plus 7.0 watts

transient 28vdc power. Maximum 115VAC power consumption is 13.7 watts

steady state plus 9.4 watts transient.

Functional Description

The ECA accepts signals from the GA1, GA2, RC1, RC2 and TVSA devices.

These signals are conditioned, shaped, limited, switched and summed as

required to provide pitch and yaw SPS gimbal servo commands as well as

pitch, yaw and roll reaction jet driver commands. The ECA generates the

DC supply voltages and reference voltages required by its circuits and

provides excitation for the Rotation Control transducers utilizing the .

115V 400hz spacecraft power. The ECA provides command signals to the RJ/EC

and TVSA devices.

The following paragraphs provide brief descriptions of the circuitry involved in performing the various functions. A more detailed description of the functional mechanization and operation of the circuits is provided in the subsystem descriptions of sections 5 and 6.

Thrust Vector Control Modules

The ECA contains two identical TVC mechanizations, one for control of the pitch SPS engine gimbal and the other for yaw. The electronics are contained in identical modules A1 and A5. Input signals are received from the TVSA (gimbal trim and gimbal position), and from the Reaction Jet Control modules (AC attitude, DC attitude and DC rate). The circuits provide gimbal command outputs to the TVSA. Each module contains one demodulator, four integrated circuit (IC) operational amplifiers with signal conditioning and shaping components, one relay and three solid state switches. The functional operation of these circuits is described in section 6.5.2.

Reaction Jet Control Modules

The ECA contains three identical RJC mechanizations (except for a difference in a few component values for roll) which make available command signals for controlling the spacecraft reaction jets. The electronics is contained in modules A2, A4 and A6. Input attitude and rate signals are received from GA1, GA2 and the Rotational Control module. AC attitude, DC attitude and DC rate signals are fed to the respective TVC modules. Each module receives logic command signals from the S/C control panel, the RJ/EC logic (A2 and A4 only) and module A8. Each module supplies a + and a - rotation

command signal to its respective jet logic channel in the RJ/EC. Each module contains 8 solid state switches, 2 demod-switches, 2 relays, 2 demodulators, 6 IC operational amplifiers with signal conditioning and shaping components, and a pseudo rate - minimum impulse circuit. The functional operation of these circuits is described in section 5.9.

Rotational Control Module

The ECA contains rotational control circuitry in module A3. 400hz rate command signals are received from the three transducers in each Rotation Control and 400hz rate signals from the three gyros in each Gyro Assembly. Logic signals are received from modules A10 and A8 and from the S/C control panel. Pitch, yaw, and roll proportional rate command signals are fed to modules A2, A4 and A6. Pitch and yaw MTVC commands are routed to the respective channels in the TVSA. The module contains 10 demodulators, 5 IC operational amplifiers with signal conditioning and shaping components, 10 solid state switches and 4 relays.

Power Supply

The ECA power supply module A7 contains two identical power supplies, each consisting of a transformer bridge rectifier and voltage regulator. Power supply No. 1 is powered by AC bus 1, while AC bus 2 drives power supply No. 2. The module provides 26vac transducer excitation voltage for the Rotation Controls (bus 1 for RC1 and bus 2 for RC2.) Reference voltages of the proper phase and from the AC bus corresponding to the input signal source are provided to all the ECA demodulators.

Power Supply No. 1 provides regulated $\pm$15V DC power to all the Auto TVC circuits and to the RJC circuits, except the Rotational Control amplifier circuits. Power Supply No. 2 provides regulated $\pm$14.5V DC power to all the Manual TVC circuits, including the Rotational Control amplifiers which provide the proportional rate command signals to the RJC mechanization.

Gyro Uncage Logic

Module A8 contains three identical Gyro Uncage Logic mechanizations, one for each of the three gyros in GA1. Each mechanization provides a ground return for the GA1 uncage relay in its respective axis when the uncage logic equation is satisfied. The input logic signals consist of the IGN 2 signal from the RJ/EC, the .05 G ENTRY, BMAG MODE (PITCH, YAW and ROLL) and MAN ATT (PITCH, YAW and ROLL) Control Panel switches, and the Rotation Control breakout switches in each axis. Besides the uncage logic signal to GA1, the mechanization also supplies logic signals to each of the Reaction Jet Control modules, A2, A3 and A4.

Manual TVC Logic

The logic for controlling the Manual TVC circuits in the ECA is contained in module A10. The module contains two identical mechanizations, one each for the pitch and yaw channels. The logic receives inputs from the SC CONT, BMAG MODE and SCS TVC control panel switches, the CW switch in the Translation Control, and the IGN 2 signal from the RJ/EC. The logic provides control signals to 8 electronic switches and 4 relays in module A3.

Signal Conditioning

Module A9 contains two signal conditioning circuits which have no functional relationship to the other ECA circuitry. The module receives two input signals, pitch and yaw differential clutch current, from the TVSA. The conditioned signals are fed to the Telemetry system. The signal conditioning amplifiers receive their operating voltages from the RJ/EC device.

3.2.6   ## Reaction Jet & Engine ON/OFF Control Assembly - BG287

The primary purpose of the RJ/EC is to provide driver circuits for the SM and CM reaction jet solenoids and the SPS engine solenoids. The RJ/EC also contains the logic circuitry associated with each driver amplifier.

Mechanical Description

The maximum weight of the RJ/EC is 20 pounds, 8 ounces. The electronic circuitry is packaged in seven interconnected modules within the environmentally sealed chassis. Electrical access for system operation is provided through two 21-pin, two 32-pin and two 61-pin connectors mounted at the rear top surface of the package. A 55-pin test connector located at the front of the package provides connections for the Ground Support Equipment. The RJ/EC has space for ten modules. However all the electronics is contained in seven modules.

        A1   Yaw Reaction Jet Driver

        A2   Pitch Reaction Jet Driver

        A3   (growth space)

        A4   Roll B&D Reaction Jet Driver

A5   Roll A&C Reaction Jet Driver

A6   (growth space)

A7   SPS Engine Logic and Driver

A8   Driver Power Supply

A9   Signal Conditioning Power Supply

A10 (growth space)

Electrical Interfaces and Interconnections

The RJ/EC provides the necessary interface between the command sources and the SM and CM reaction jet auto solenoids and the SPS engine solenoids. The command sources for reaction jet control are the CMC, ECA, RC breakout switches and TC switches. The command sources for controlling the SPS engine solenoids are the CMC, EMS and Control Panel 1 switches.

The RJ/EC consumes a maximum of 2.1 watts steady state plus 1.0 watt transient 28vdc power. Maximum 115vac power consumption is 4.0 watts steady state plus 1.0 watt transient.

Functional Description

For reaction jet control, the RJ/EC accepts command signals from the CMC, the switching amplifier - minimum impulse control circuits of the ECA, the TC switches and the RC breakout switches. Logic signals from Control Panel 1, TC CW switch and SPS Ignition signals generated within the RJ/EC control the state of the reaction jet logic which enables and disables the command inputs to the proper jet drivers.

For SPS engine solenoid control, the RJ/EC accepts command signals from the CMC, the THRUST ON pushbutton on Control Panel 1 and the automatic off signal from the EMS ΔV counter. The SPS Thrust Control logic receives logic signals from Control Panel 1 switches for enable and disable of the command signals to the SPS engine solenoids. The Thrust Control Logic also provides an enable-disable logic signal to the reaction jet logic for the pitch and yaw RCS engines.

The following paragraphs provide brief descriptions of the module contents. The functional mechanization of the drivers and logic is described in sections 5 and 6.

Reaction Jet Drivers

The RJ/EC contains four identical Reaction Jet Driver modules, A2 for pitch, A1 for yaw, A5 for roll A&C and A4 for roll B&D reaction jet solenoids. Each module contains four identical pre-amp and driver amp stages and the enable-disable logic circuitry for controlling inputs to the pre-amps. Each module receives command inputs from the CMC, the TC switches (through diode OR gates in module A8), the ECA + and - rotation command outputs, and both Rotation Control + and - breakout switches. Logic control inputs are received from the SC CONT and MAN ATT Control Panel 1 switches, TC CW switch and module A7. Pre-amp enable power is received through MESC contacts and driver-amp bias is provided by module A8. Each module has four outputs, one from each driver-amp, which supplies a ground return for the respective reaction jet solenoid to turn on the reaction jet.

SPS Engine Logic and Drivers

Module A7 contains the SPS thrust ON/OFF logic and the SPS solenoid drivers. The module receives enable voltage from the control panel 1 ΔV THRUST switches, logic control signals from the control panel 1 S/C CONT and DIRECT ULLAGE switches and TC CW and +X switches, ar command signals from the CMC, EMS and control panel 1 SPS THRUST wit. The module provides control signals for the SPS solenoid valves and logic signals to modules A1 and A2 (pitch and yaw disable) and the ECA (Engine IGN 2). The module contains 2 pre-amps, 2 driver-amps, 4 delay circuits, the diode logic mechanization and 7 TM event voltage dividers.

Driver Power Supply

Module A8 contains two -4vdc power supplies which provide bias voltage to the RJC and SPS driver-amplifiers. (The bias voltage is not essential for proper operation of the drivers. It serves only to prevent turn on of the drivers due to large 28V DC transients). The module receives 115vac power from the SIG COND/DRIVER BIAS POWER switches on control panel 7 and supplies -4vdc bias power to modules A1, A2, A4, A5 and A7. The module contains two identical power supplies consisting of a transformer, rectifier and filter for each. Each power supply provides bias to eight RJC and one SPS driver-amplifiers.

Signal Conditioner Power Supply

Module A9 contains a Signal Conditioner Power Supply identical to the one in the EDA modules A13 and A15. The power supply receives 115vac

power from S12 on control panel 7. It supplies four DC voltages to the signal conditioning amplifiers in the ECA module A9.

3.2.7 Thrust Vector Servo Amplifier Assembly - BG288

The TVSA provides the necessary electrical interface between the command electronics and the actuator for positioning the SPS engine. The functional mechanization of the TVSA is described in Section 6.

Mechanical Description

The maximum weight of the TVSA is 12 pounds, 8 ounces. The electronics is contained in five interconnected modules within the environmentally sealed chassis. Electrical access for system operation is provided through two 61-pin and two 19-pin connectors. A 55-pin connector is provided for interface with the ground test equipment. The five modules containing the TVSA electronics are:

        A1   Demodulator

        A2   Yaw Servo Amplifier

        A3   Pitch Servo Amplifier

        A4   Logic

        A5   Power Supply

Electrical Interfaces and Interconnections

The TVSA provides the interface between the SCS and G&N command electronics and the SPS Servo Actuator. The TVSA accepts signals from the CMC, ECA and GP/FPI. These signals are summed, shaped, limited and switched as required to provide pitch and yaw gimbal servo commands to the SPS Servo Actuator.

The TVSA consumes a maximum of 13.4 watts steady state plus 60.0 watts transient 28vdc power. Maximum 115vac power consumption is 21.9 watts steady state plus 19.4 watts transient.

Demodulator Module

Module A1 contains all the demodulators utilized in the TVSA mechanization. It also contains IC DC amplifiers for the SPS Servo Actuator position feedback signals and transistor buffers for the demod reference signals. The module receives the demod reference voltages from module A5 and pitch and yaw rate and position signals from the SPS Servo Actuator transducers. It provides DC rate and position signals to modules A2 and A3. The module contains 16 buffer transistors, 8 demodulators and 4 amplifiers.

Servo Amplifier Modules

The TVSA contains two identical servo amplifier modules, A2 and A3, for the yaw and pitch channels respectively. Each module contains identical redundant control channels. Each channel receives DC input signals from the CMC, ECA, GP/FPI trimpots and module A1. It receives logic signals for controlling its solid state switches from the SC CONT and SCS TVC control panel switches, the Translation Control CW switch and the RJ/EC. Each channel provides drive current to its respective SPS Servo Actuator extend and retract clutches. A module contains 8 solid state switches, 4 IC DC amplifiers, two clutch drive circuits and the logic mechanization which drives the solid state switches.

Logic Module

The TVSA logic, except that which controls the command source selected, is contained in module A4. This module also contains the filters for the 15vdc power used for GP/FPI trim pot excitation. The module receives 28vdc power inputs from S8 and S9 on Control Panel 7. It receives logic inputs from the control panel 1 TVC Gimbal Drive switches, Translation Control CW switch, the EDA and TVC Failure Signals from the SPS Servo Actuator. Signal inputs are received from module A1 and the GP/FPI trim pots. The logic controls gimbal trim signals to the ECA and gimbal position signals to the ECA and EDA. The module contains 4 TT-type RC filters, 8 relays and the associated summing control components.

Power Supply

Module A5 contains two redundant power supplies. This module receives 115vac power from S8 and S9 on control panel 7. The module provides ± 15vdc regulated power to modules A1, A2 and A3. ± 15vdc non-regulated, non-filtered power is supplied to module A4. Bus 1 and Bus 2, 20vac power is supplied to A1 for demod reference. Bus 1 and Bus 2, 13vac power is supplied to the SPS servo actuator position and rate transducers. The module contains 2 EMI filters, 4 transformers, 2 bridge rectifiers and 4 voltage regulators.

SECTION 4

ATTITUDE REFERENCE SUBSYSTEM (ARS)

4.1   INTRODUCTION

The purpose of the SCS Attitude Reference Subsystem is to provide the
astronauts with a visual display of the spacecraft attitude in space.
An indication of the spacecraft attitude is required for monitoring
automatic operation and for performing manual maneuvers.

As man ventures farther into space, leaving the earth behind, he requires
a new reference for his attitude orientation. Until now, man has used
the earth as his reference. Even in orbit around it (during Mercury
and Gemini flights) local vertical was the usual attitude reference for
the astronaut and the displays in the capsule. It was not always
essential that an attitude display be provided since that information
was available in the form of the earth's horizon. Some entries were
performed during Mercury flights using only that reference.

However, during flights farther into space, the earth no longer remains
a near body that can be used as a reference. Furthermore, the apparent
earth size and position will be continually changing as the vehicle moves
out into space. Thus a fixed reference frame for attitude orientation
must be chosen. A fixed reference frame available to the astronaut is
in the form of an inertial sphere (e.g., the celestial sphere).

Vehicle Reference Frames

If the spacecraft is to be maneuvered to an attitude relative to the

celestial sphere, then the spacecraft itself must have a set of coordinates to be used in the alignment. As a general reference frame, a rectangular coordinate system has been assigned to the spacecraft and is generally referred to as the spacecraft "body" coordinates. The +X (roll) axis is chosen along the longitudinal axis of the spacecraft in the direction of lift-off. The +Y (pitch) axis is at 90° to the X axis, laterally through the vehicle in the direction of the astronaut's right shoulder. The +Z (yaw) axis is at 90° to the other two axes and is vertically down with respect to the astronauts. Since the CM center of mass is offset from the capsule longitudinal center line, aerodynamic forces on the capsule (when it enters the earth's atmosphere) will orient it to a stable attitude relative to the flight path. These entry coordinate axes are known as the "STABILITY" or "WIND" axes. These axes are displaced from the Body axes, in the X-Y plane by an angle α.

There are two stable attitudes for flight in the atmosphere, heat shield forward or apex forward. Prior to encountering the earth's atmosphere, the capsule must be oriented with the heat shield forward to provide a safe entry. With the heat shield forward, the stable attitude of the capsule relative to the flight path is as shown in Figure 4.1. The pitch stability axis is identical to the pitch body axis (+Y axis) which is directed out of the page. The roll and yaw stability axes are defined as the roll and yaw body axes rotated +21° about the pitch (Y) axis.

EARTH ENTRY ORIENTATION
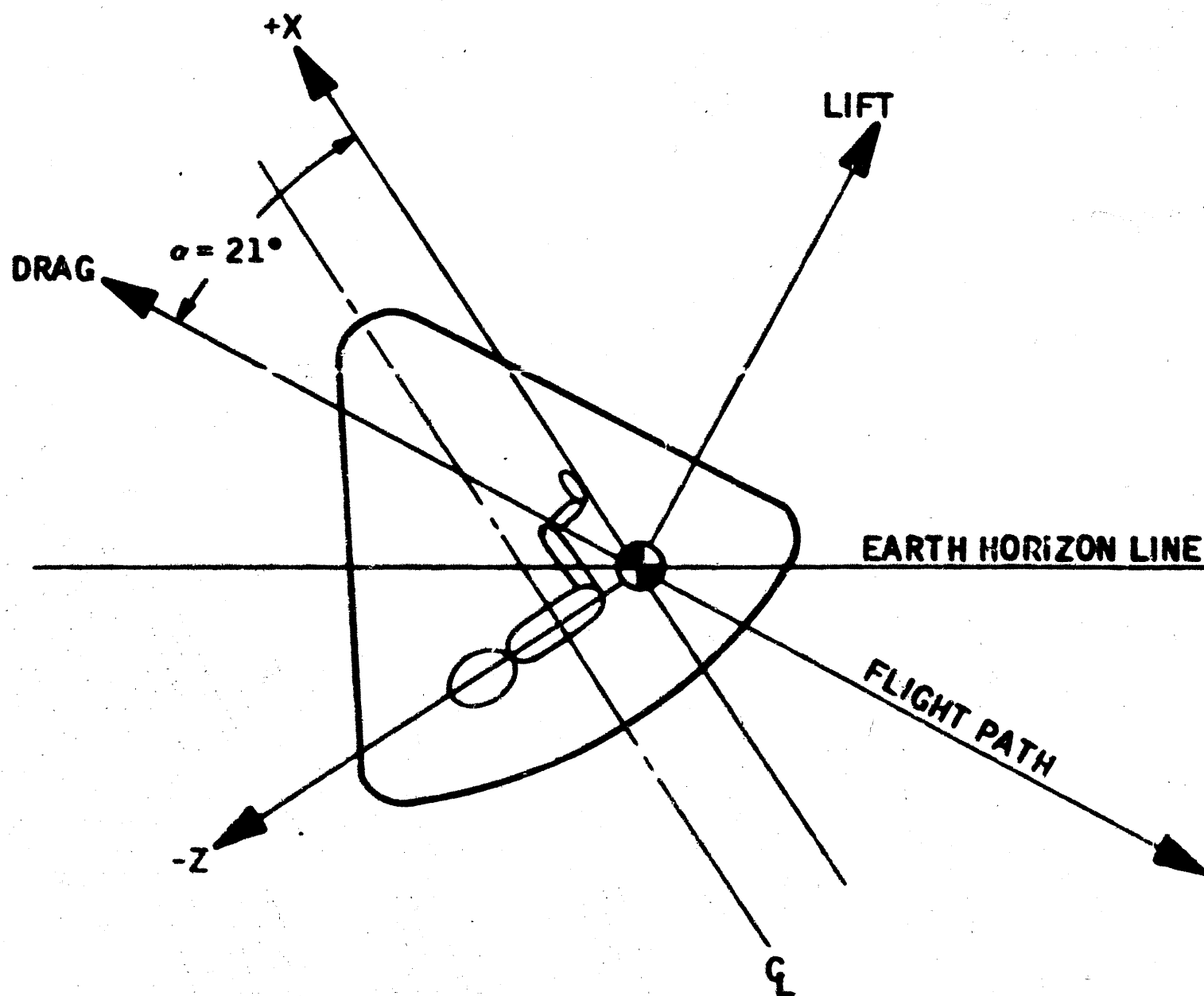
FIGURE 4.1

Inertial Reference Frame

Spacecraft attitude in space has meaning only if it is referred to
a fixed or inertial reference frame. The inertial reference frame
is defined by an orthogonal triad of axes which is fixed with respect
to the celestial sphere. The three axes are referred to as the
inertial pitch, yaw, and roll axes. The relationship between the
three axes can be defined by the equation $\bar{i} \times \bar{j} = \bar{k}$ where $\bar{i}$, $\bar{j}$, and $\bar{k}$ are
unit vectors in the direction of the inertial roll, pitch, and yaw axes
respectively.

Euler Angles

The total attitude of the vehicle is defined by the orientation of
the S/C body axes relative to an arbitrarily chosen fixed reference
frame. The mathematical transformation between a fixed and a movable
reference frame is called an Euler transformation (so named in honor
of the mathematician who derived it).

The total attitude of the vehicle is described by three Euler angles:
roll (ϕ), pitch (θ), and yaw (ψ). All Euler angles are zero when the
body roll (X), pitch (Y) and yaw (Z) axes are parallel to the respective
fixed reference roll ($\bar{i}$), pitch ($\bar{j}$) and yaw ($\bar{k}$) axes. The Euler angles
are those three ordered vehicle rotations (in the sequence pitch-yaw-roll)
about the body axes, from the all-angles-zero attitude, required to
arrive at the given attitude, provided that $-90° < \psi < +90°$, and $0° < \theta$,
$\phi < 360°$. The Euler angles are identical to the gimbal angles obtained
from a three-gimbal platform (with the gimbal sequence pitch-yaw-roll

4-4

from inside to out) having its reference element aligned to the chosen reference frame.

4.2   SCS ARS COMPONENTS

The SCS devices used in the Attitude Reference Subsystem include:

        Flight Director Attitude Indicators (FDAI 1 and FDAI 2)

        Gyro Assemblies (GA 1 and GA 2)

        Gyro Display Coupler (GDC)

        Attitude Set Control Panel (ASCP)

        Electronics Display Assembly (EDA)

The display portion of the ARS consists of the two FDAIs. S/C total attitude is indicated by the position of the ball. The difference between the desired and actual S/C attitude is indicated by the FDAI attitude error needles. The angular rate of change of S/C attitude is described by the FDAI rate needles.

The rate sensing elements of the ARS are contained in the Gyro Assemblies. The Gyro Assemblies also provide electronic signals which indicate changes in spacecraft attitude about each of the three body axes. The GA outputs consists of 400 hz rate and attitude error signals and DC rate signals.

The GDC contains the electronics and electro-mechanical components necessary to compute S/C inertial and roll stability attitude information from the DC body rate signals provided by the Gyro Assemblies.

The ASCP contains three resolvers (pitch, yaw, and roll) which are

positioned at the angles dialed in on the thumbwheels. The resolvers provide attitude error signals which are sine functions of the resolver angles (desired attitude) minus the input inertial angles (actual attitude).

The EDA contains signal conditioning and servo electronics necessary to drive the FDAI indicators and logic networks to allow selection of the desired source of information and the display(s) to be driven. The EDA logic states are determined by control panel switch positions.

4.3 SCS ARS INTERFACES

The SCS Attitude Reference Subsystem has interfaces with the G&N System, the Entry Monitor System (EMS), and Control Panel Switches.

The G&N System provides the primary attitude and attitude error information which will be displayed on a FDAI. Spacecraft total attitude information is provided by the Inertial Measuring Unit (IMU). Desired attitude information is generated in the Command Module Computer (CMC) and the attitude error (difference between desired and actual attitude) is provided by the G&N system through its Coupler Data Unit (CDU).

The Roll Attitude Indicator (RAI) on the EMS display shows the position of the S/C Lift Vector during the earth entry phase of the mission. This indicator is driven by signals generated in the SCS Gyro Display Coupler after .05g.

Control panel switches allow the S/C Commander to select alternate configurations for displaying the attitude information. The switches furnish logic signals to the GDC and EDA to control the signal flow paths.

4.4 G&C ATTITUDE REFERENCE

A block diagram showing the hardware involved and the basic signal flow paths for the Attitude Reference Subsystem is shown in Figure 4.2. A brief description of the basic signal flow paths follows. The various signal flow paths are enabled and disabled as a function of the position of control panel switches which are discussed in section 4.5.

4.4.1 Total Attitude

Total attitude of the S/C is displayed on either or both FDAI's. Pitch and yaw Euler angles are read at the point on the ball under the drone. The roll Euler angle is read at the point indicated by the roll index on the circular scale. The two sources of attitude information are the IMU and the GDC. During entry roll stability attitude information can be displayed on the RAI and on the roll scale of one of the FDAI's.

Primary Source

The Inertial Measuring Unit of the G&N system is the primary source of total attitude information. The information consists of two 800 hz signals from each of the three gimbal resolvers. The amplitudes of the signals are proportional to the sine and cosine of the respective gimbal angles. These six signals are fed through the select logic, signal

G & C ATTITUDE REFERENCE
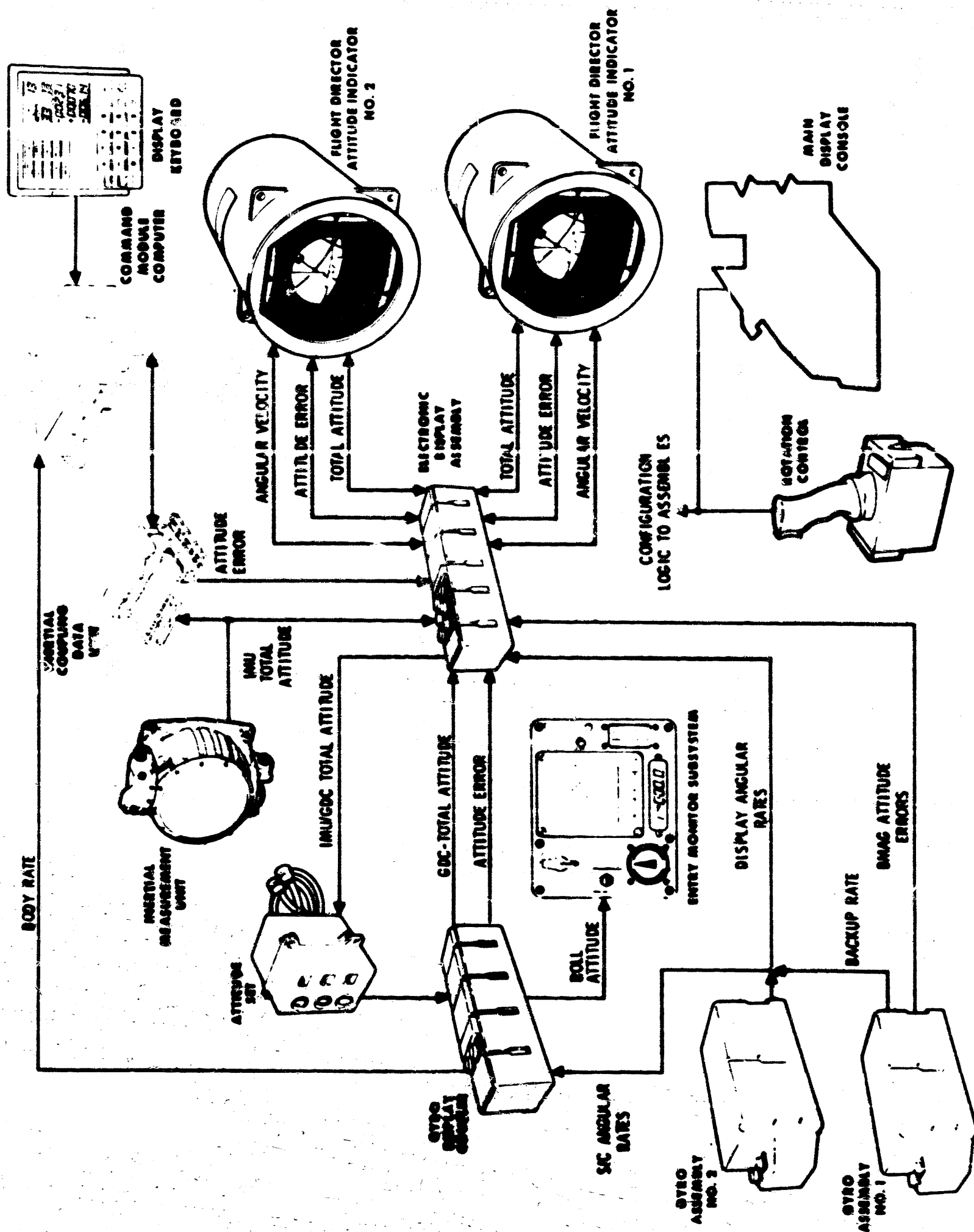
FIGURE 4.2

conditioning and servo electronics in the EDA to control the position of the three gimballed ball. Through this mechanization, the FDAI ball is driven to represent the gimbal angles of the IMU. Either FDAI, but not both, can be driven from the IMU source.

Backup Source

The SCS provides backup total attitude information which is computed in the GDC using inputs from one of both Gyro Assemblies. One function of the GA - GDC combination is to generate Euler angle signals (equivalent to the gimbal angle signals provided by the IMU resolvers). The GA generates accurate DC signals proportional to vehicle rates about each of the S/C axes. The GDC performs a transformation which converts the body rates to Euler rate signals which position shafts to equivalent gimbal angles. Resolvers attached to these shafts provide total attitude information to the EDA for positioning a FDAI ball.

Entry Configuration

During entry the GDC configuration is changed to allow generation of redundant roll stability attitude signals. The Euler attitude mechanization is disabled during this time. In the entry configuration the GDC computes the vehicle attitude relative to the S/C roll stability axis (lift vector position). This information is used to drive a FDAI ball in roll only (pitch and yaw channels are disabled) and the Roll Attitude Indicator of the Entry Monitor System.

4.4.2   <u>Attitude Error</u>

Four sources are available for driving the attitude error needles on either FDAI. The roll, pitch, and yaw attitude errors are determined by the position of the needles relative to the scale marks. Full scale deflection values for the indicators (both rate and attitude error) are selected with a single cont   panel switch.

<u>CDU Source</u>

The Inertial Coupler Data Unit of the G&N system is the primary source of attitude error information. The IMU gimbal resolver signals are converted to digital signals in the CDU and transferred to the CMC for comparison to the desired attitude. The attitude error is determined and transformed to S/C referenced attitude errors which are fed back to the CDU. The CDU performs a digital to analog conversion and supplies the 800 hz attitude error signals to the EDA logic circuits. The three signals are fed through the logic, scaling, signal conditioning and servo electronics in the EDA to the pitch, yaw, and roll servometric attitude error indicators in one of the FDAIs.

<u>GA 1 Source</u>

The uncaged BMAGs in GA 1 provide a second source of attitude error information. They supply 400 hz pitch, yaw and roll attitude error signals to the EDA for processing to drive the indicators in one of the FDAIs. This source is available <u>only</u> when the GA 1 BMAGs are uncaged.

GDC - ASCP Source

Another source of attitude error information is the GDC - ASCP. The three GDC transmitter resolvers feed the Euler angle signals to the respective ASCP resolvers. The Euler angle error signals at the ASCP resolver outputs are fed back to the GDC for transformation to body referenced attitude error signals. These 400 hz signals are sent to the EDA for processing to drive the indicators in one of the FDAIs.

IMU - ASCP Source

Three axis attitude error information referenced to the inertial reference frame (IMU stable platform) is available for driving the attitude error indicators in either FDAI. The three IMU gimbal resolver signals are routed through the EDA to the ASCP resolvers. The ASCP resolver outputs are sine functions of the difference angles between the corresponding gimbal and ASCP resolvers. These 800 hz error signals are routed back through the GDC and the EDA electronics to drive the FDAI attitude error indicators.

4.4.3   ## Angular Rate

GA 2 is the normal source of angular rate information. Backup rate information is available (by-axis) from GA 1. The 400 hz signals from the BMAGs in either Gyro Assembly are fed through the EDA logic, scaling, signal conditioning and servo electronics to drive the servo-metric rate indicators in either or both FDAIs.

During entry the yaw rate indication must be proportional to rates

about the yaw stability axis. This is necessary to prevent yaw body rates from being indicated while the S/C is being maneuvered to change the lift vector position. Entry yaw rate signals are generated by properly scaling and summing the roll BMAG output with the yaw BMAG output to cancel the rate sensed by the yaw gyro. The 400 hz yaw and roll (modified by $\tan \alpha$) BMAG outputs are summed to provide the stability yaw rate signal. Since the GA 1 and GA 2 BMAG 400 hz power is obtained from non-synchronized busses, it is imperative that yaw and roll rate signals are obtained from the same Gyro Assembly. The select logic in the EDA is mechanized to accomplish this (see Figure 4.5), by using GA 1 outputs for both signals if backup rate is selected in either the yaw or roll axis.

4.5   ARS RELATED CONTROL PANEL SWITCHES

The following is a brief discussion of the basic function of the control panel switches as they affect the ARS. The information displayed on the FDAIs is a function of the positions of S2, S3, S4, S5, S6, S20, S21, S22, S37, S50, and S51 (see Figure 4.3).

CMC ATT

The CMC ATT switch (S2) determines the source of attitude information supplied to the G&N computer. The normal source is the IMU (up position). The GDC (down position) can be used to supply backup attitude information to the CMC in case of an IMU failure. Neither FDAI ball will be driven with the switch in the GDC position.

CONTROL PANEL NO.1

FIGURE 4.3

4-13

FDAI SCALE

The FDAI SCALE switch (S3) determines the full scale range for the attitude error and rate needles. Selection of the up position results in full scale deflections of 5° (attitude error) and 1°/sec (rate). The center position yields full scale deflection for 5° and 5°/sec. In the down position, full scale deflection is obtained for 50° and 50°/sec. on the roll needles and 15° and 10°/sec. on the pitch and yaw needles.

FDAI SELECT

The FDAI SELECT switch (S4) determines which FDAI is active to display information from the selected sources.

The 1/2 position enables both FDAIs and selects all sources of display. FDAI 1 will display G&N source information: total attitude to the ball from the IMU and attitude error to the error indicators from the CDU. FDAI 2 will display SCS source information: total attitude to the ball from the GDC and attitude error to the error indicators from GA 1. The rate indicators of both FDAIs are normally driven from GA 2. If GA 1 is selected and used as a source of rate information (backup rate may be selected by axis) the error indicators on FDAI 2 will be at null. In this position the FDAI SOURCE switch and the ATT SET switch will not be functional for selecting display sources.

The 2 or 1 position of this switch will enable the respective FDAI. The sources of total attitude and attitude error to be displayed is now determined by the FDAI SOURCE and ATT SET switches.

FDAI SOURCE

The FDAI SOURCE switch (S5) determines the source of total attitude and attitude error to be displayed on the selected FDAI. This switch is not functional when the FDAI SELECT switch is in the 1/2 position.

In the CMC position G&N source information is displayed: total attitude from IMU and attitude error from the CDU.

In the GDC position SCS source information is displayed: total attitude from the GDC and attitude error from GA 1 (if the gyros are not in a rate caged configuration).

In the ATT SET position the ATT SET switch determines the source of total attitude and attitude error information. The same source selected for total attitude display is interfaced with the Attitude Set Control Panel. The output error signal from the ASCP is proportional to the difference between the selected source and the settings of the ASCP. This generated error signal is displayed on the error indicators.

ATT SET

The ATT SET switch (S6) performs two different functions. First as indicated above, it determines the source of total attitude and attitude error information for displays and secondly it enables alignment of the SCS attitude reference subsystem. The first function is performed only when the FDAI SELECT switch is not in the 1/2 position and the FDAI SOURCE switch is in the ATT SET position. The second function is performed any time this switch, ATT SET, is in the GDC position.

In the IMU position the IMU drives the ball and also interfaces with the ASCP to generate an error signal for display on the error indicators.

In the GDC position the GDC serves as the ball drive and interfaces with the ASCP. In addition to displaying this error it may also be used to align the attitude reference system to a fixed inertial reference by pressing the GDC ALIGN push button. The new inertial reference aligned to is that dialed into the ASCP.

GDC ALIGN

Depressing the GDC ALIGN push button (S37) with the ATT SET switch in the GDC position, aligns the GDC to the attitude dialed into the ASCP. It should be remembered that these two switch positions are the only panel 1 switching necessary for this alignment.

ENTRY EMS ROLL

Placing the ENTRY switch (S50) in the EMS ROLL position allows the GDC yaw channel to provide command signals to the Roll Attitude Indicator stepper motor in the EMS display. This signal flow path is disabled when the switch is in the OFF position.

ENTRY .05 G

Placing the ENTRY switch (S51) in the .05 G position configures the GDC to compute stability roll attitude in its yaw and roll channels. Proper selection of the FDAI SELECT, FDAI SOURCE and the ENTRY EMS ROLL switches allows the stability roll attitude (lift vector position) to be displayed on either FDAI and the RAI of the EMS.

BMAG MODE - ROLL, PITCH and YAW

The BMAG MODE switches, ROLL (S20), PITCH (S21), and YAW (S22) provide
the capability of selecting BMAG configurations by-axis. The ATT 1
RATE 2 position provides rate signals from BMAG 2 and angular error
signals from BMAG 1 (if uncaged). In the RATE 2 position rate signals
are provided by BMAG 2 and in the RATE 1 position, BMAG 1 supplies rate
signals.

4.6   ATTITUDE REFERENCE MODES AND SELECT LOGIC

There are six basic configurations or modes of operation for the Attitude
Reference System. These modes are functions of the EDA select logic states
which are determined by control panel switch positions. The various modes
are defined by the source of attitude and error information. Alternate
success paths for providing a visual display of spacecraft attitude, error
and rate information are available within each mode.

Figures 4.4 and 4.5 are representations of the select logic in the form
of single line block diagrams which functionally depict the signal flow
paths from source to display as functions of control panel switching.

4.6.1   Primary Mode

With all system switching in the primary positions (CMC ATT - IMU, FDAI
SELECT - 1, FDAI SOURCE - CMC, ENTRY (both switches) - OFF, ATT SET - GDC
and BMAG MODE (ROLL, PITCH, AND YAW) - RATE 2) the primary mode of the
Attitude Reference Subsystem is obtained without further switching being
performed. The FDAI SCALE switch does not affect the mode of operation.

FDAI ATTITUDE SELECT LOGIC
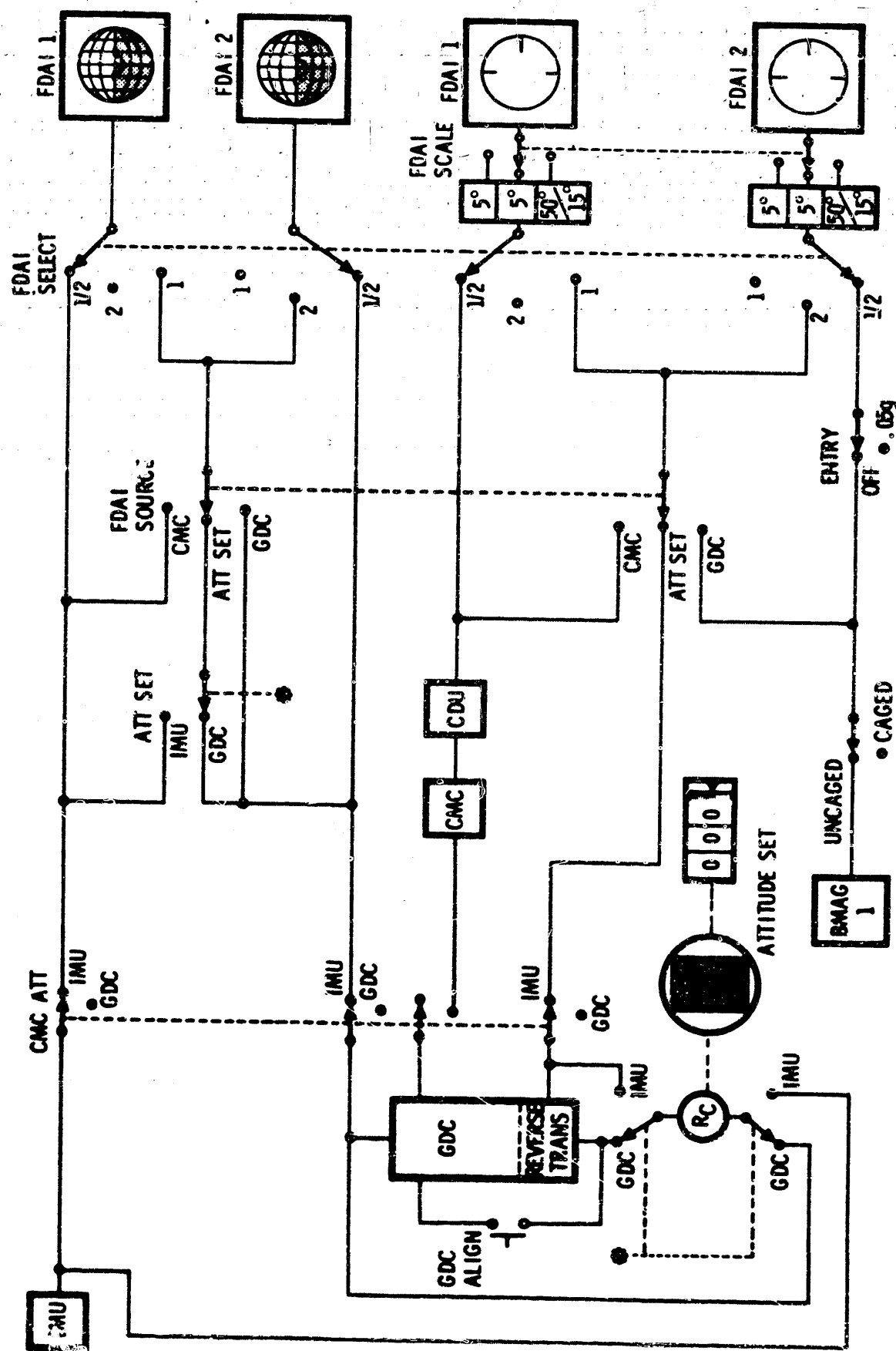
FIGURE 4.4

FDAI RATE SELECT LOGIC

FIGURE 4.5

4-19
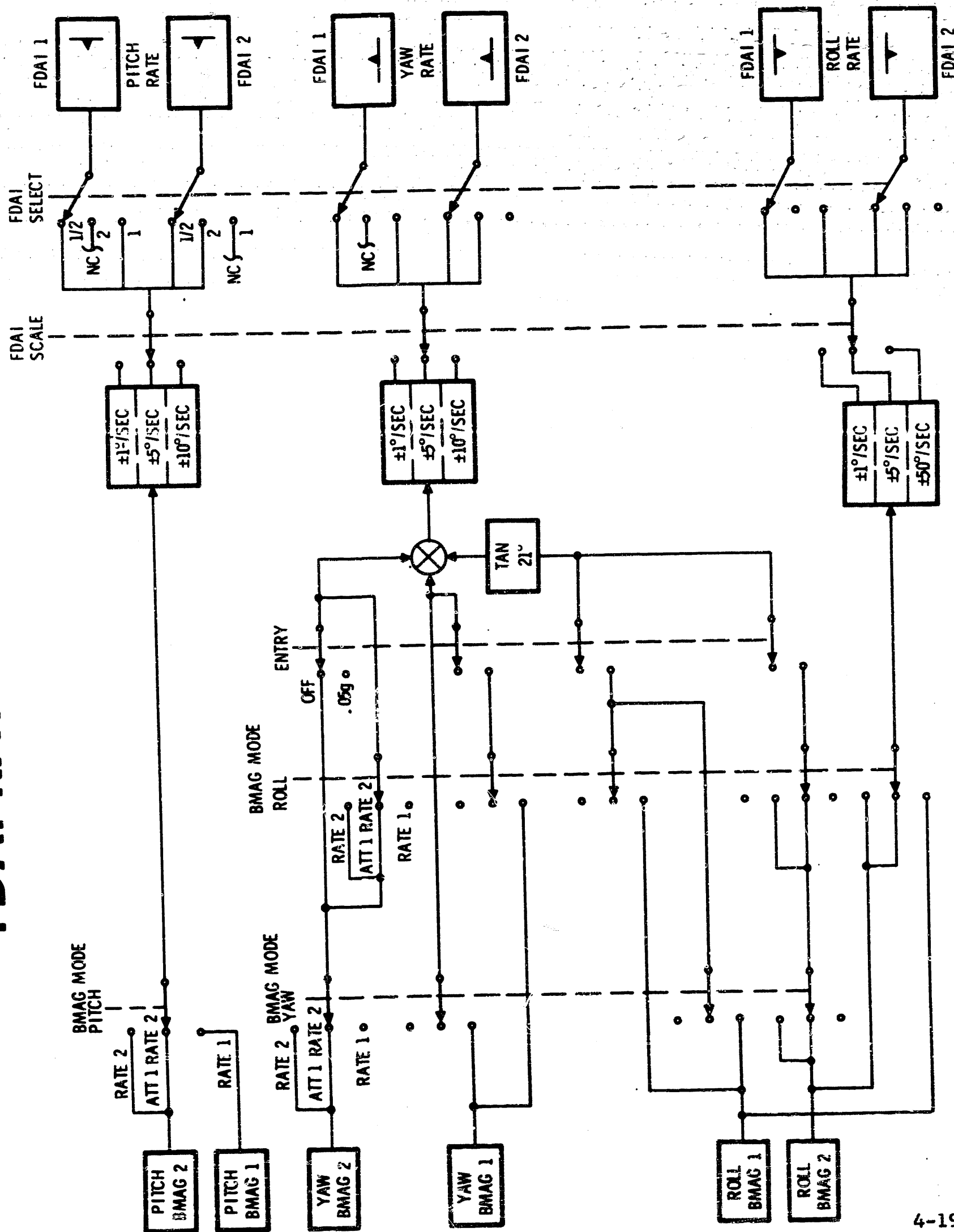
In this mode, spacecraft attitude, error, and rate are displayed on FDAI 1.

Rate information is derived from the GA 2 caged BMAGs. Attitude and error information is derived by the G&N (attitude from the IMU and error from the CMC-CDU) and is displayed directly without further processing (other than scaling) by the SCS.

The attitude information displayed on the FDAI ball and roll index is in inertial or fixed axis coordinates, while the attitude error and rate indications are in body axis coordinates.

FDAI 2

The information described above can be displayed on FDAI 2 if the FDAI SELECT switch is placed in the "2" position. In this case, FDAI 1 is not used.

Both FDAIs

When the FDAI SELECT switch is placed in the 1/2 position, both FDAI 1 and FDAI 2 are turned on. In addition to the display of G&N information on FDAI 1, a simultaneous display of body rate, Euler attitude, and body error signals derived within the SCS is shown on FDAI 2. The manner in which the attitude and error signals are obtained within the SCS is identical to that to be described for the IMU Backup Mode (Configuration A).

Entry

At the time the earth entry phase of the mission begins, the astronaut

will place the two entry switches (S50, S51) to the EMS ROLL and .05 G positions. All BMAGs are placed in a rate configuration. The body rate information from the GA 2 pitch and roll BMAGs is displayed on the rate indicators of FDAI 1. Modified roll rate information from the GA 2 roll BMAG is added to the yaw rate information from the GA 2 yaw BMAG to cause an apparent shift in the yaw sensing axis. The resulting signal provides an indication of rate about the yaw stability axis on the yaw indicator of FDAI 1.

The rate signals from the GA 2 roll and yaw BMAGs are also summed and applied to the yaw GDC channel to provide roll stability attitude inform- ation to the Entry Monitor System (EMS). The rate signals from the GA 1 roll and yaw BMAGs are summed and supplied to the roll GDC channel to generate redundant roll stability attitude information for display on FDAI 2 if both FDAIs are selected.

Backup Rate

Backup rate can be achieved by-axis by placing the BMAG MODE switches in the Rate 1 position. This action causes the GA 1 BMAGs to be caged and thus supply the required rate signals. The GA 2 BMAGs are always caged but the output rate signals are not used under these conditions. Placing the BMAG MODE switches in the ATT 1 RATE 2 position does not affect the information displayed in the primary mode.

CMC Backup Mode

With all system switching in the primary positions, the CMC Backup Mode

is obtained when the following switching is performed: FDAI SOURCE to ATT SET and ATT SET to IMU.

In this mode, the system description is identical to that of the Primary Mode except for attitude error information.

The G&N attitude information (from the IMU) is subtracted from the attitude dialed into the Attitude Set Control Panel. The different signals are then used to position the attitude error needles on the FDAI. The errors displayed are inertial referenced errors.

This mode is primarily intended for use if the G&N CMC is not operable, but the mode could be used at any time.

The sub-mode switching described for the Primary Mode (except "Both FDAIs") is also applicable to the CMC Backup Mode. The CMC Backup Mode is not available when the FDAI SELECT SWITCH is placed in the 1/2 position.

4.6.3   G&N Backup Mode

With all system switching in the primary positions, the G&N Backup Mode is obtained by placing the FDAI SOURCE switch in the ATT SET position.

In this mode, the GDC provides the source for the display of total attitude and attitude error on FDAI 1. The body rates for display are always derived from the SCS gyros.

Gyro rate signals are converted to Euler angles in the GDC and displayed on the FDAI ball. The attitude error for display is derived in two steps.

First the differences between the GDC and the Attitude Set Control
Panel resolver angles are taken. Then the difference is converted
to body axis errors in the GDC and displayed on the attitude error
needles.

The sub-mode switching described for the Primary Mode (with the ex-
ception of "Both FDAIs") is applicable to the G&N Backup Mode.

4.6.4   **IMU Backup Mode** (Configuration A)

With all system switching in the primary positions, the IMU Backup
Mode (Configuration A) will be obtained when the FDAI SOURCE switch
is placed in the GDC position and the BMAG MODE (ROLL, PITCH, YAW)
switches are set to the ATT 1 RATE 2 position.

In this mode, the information for the display of total attitude on
the FDAI 1 ball is derived within the GDC.

The rate information for display is obtained from BMAGs in the rate
mode. The attitude error display is driven by signals derived from
BMAGs in the attitude mode.

The sub-mode switching described for the Primary Mode is applicable
to the IMU Backup Mode (Configuration A) with the following exceptions:

      1)   Selection of Backup Rate results in a loss of body
            attitude error information.

      2)   There is no attitude error information available for
            the entry mode.

4.6.5   IMU Backup Mode (Configuration B)

With all system switching in the primary positions, the IMU Backup

Mode (Configuration B) is obtained by placing the CMC ATT switch in

the GDC position.

In this mode the GDC provides attitude information to the CMC and

allows the G&N system to drive the attitude error indicator of the

FDAI thru the CDU (if a CMC Program is available to do it). Body

rate information is still displayed on the FDAI rate needles. In

this configuration total attitude information is not available from

any source for display on either FDAI ball.

4.6.6   Alignment Mode

With all system switching in the primary positions, the Align Mode

will be obtained when the GDC ALIGN button is depressed.

In this mode the GDC is aligned to the attitude dialed into the ASCP.

All axes are aligned simultaneously. This mode takes precedence over

all other modes using the GDC and is generally used to align the GDC

prior to using one of these modes.

Even though it plays no role in the alignment of the GDC, nulls will

be observed on the FDAI attitude error indicators at alignment completion

when the FDAI SOURCE switch is in the ATT SET position and FDAI 1 or

FDAI 2 is selected with the FDAI SELECT switch. The alignment may also

be observed on the FDAI ball if the FDAI SOURCE switch is in either the

ATT SET or the GDC position.

If the ATT SET switch is placed in the IMU position, the IMU provides
an input to the ASCP and a closed loop for the GDC alignment is not
present. However, in this configuration the attitude set thumbwheels
can be positioned to null out the attitude error and provide an
accurate readout of the S/C attitude with respect to the G&N stable
platform on the ASCP. The ATT SET switch can then be placed in the
GDC position and the GDC ALIGN button depressed to accurately align
the GDC to the stable platform reference.

4.7   ARS FUNCTIONAL OPERATION

Figure 4.6 shows by way of a block diagram the functional signal flow
paths through the GDC and the interfacing hardware for these paths.
The blocks in the figure which are not part of the GDC are the BMAGs
(located in GA 1 and GA 2), the Roll Attitude Indicator (located in
the EMS), the IMU (part of the G&N), the control resolvers (located
in the ASCP), the EDA and the FDAI ball.

The primary purpose of the GA-GDC combination is to provide a backup
attitude reference system for accurately displaying the attitude of
the S/C relative to a fixed reference frame. The GDC is mechanized
to operate in any one of the following modes:

a)   Euler - computes total attitude from body rate signal inputs
     to drive the FDAI ball and transforms inertially referenced
     errors to drive the FDAI attitude error needles.   .

b)   Single Axis - computes digital body rate signals from DC body
     rate signal inputs for use by the G&N system.

ATTITUDE REFERENCE SUBSYSTEM

FUNCTIONAL OPERATION

FIGURE 4.6

c) .05g (entry) - computes roll stability attitude from body rate signal inputs to drive the RSI of the Entry Monitor System and one of the FDAI's.

d) GDC Align - provides a means of aligning the GDC reference to a given reference through the Attitude Set Control Panel.

4.7.1   Euler Mode

In the Euler mode, the GDC takes pitch, yaw, and roll DC body rate signals from either GA 1 or GA 2. These body rate signals are converted to Euler rate signals by the Body-to-Euler transformation mechanization. This transformation generates pitch and yaw Euler Rate signals (and the required term, which when summed with roll body rate, will provide the roll Euler rate signal) using body rate inputs from the pitch and yaw gyros. The Euler rate signals are converted to pulses by the Voltage to Frequency converters. These pulses drive stepper motors which drive the transmitter resolver (RX) rotors through gear trains such that the rotor position for each resolver represents that particular Euler angle.

The transmitter resolver outputs furnish the drive signals to the servo electronics in the EDA which drives the FDAI ball in three axes to the same angular positions as the transmitter resolvers.

Transmitter resolver outputs are also supplied to the control resolvers (RC) in the ASCP. The control resolver outputs (functions of the difference between RX and RC rotor positions) are fed back to the GDC. These resolver outputs are inertially referenced attitude error signals.

It is desirable to provide an indication of the vehicle maneuvers necessary to null the attitude errors. Hence the Euler attitude errors are converted to body referenced attitude errors by the Euler-to-Body transformation mechanization. The Euler-to-Body mechanization receives signals from the pitch and yaw control resolvers to generate pitch and yaw body attitude errors and the term which, when added to the roll control resolver output, represents roll body attitude error. These body attitude errors are supplied to the EDA servo electronics which drives the FDAI servometric meter movements.

4.7.2   Single Axis (Non-Euler) Mode

In the Single Axis Mode the GDC takes pitch, yaw and roll DC body rate signals from either GA 1 or GA 2. These signals are fed directly (by axis) to the Voltage to Frequency converters. The V to F converter output pulses (digital body rate signals) are sent to the G&N Command Module Computer. The GDC thus provides the CMC with a source of attitude information if the IMU fails. The lines from the GDC V to F Converter to the CMC are always enabled in the GDC. The enable-disable logic for these lines is contained in the G&N computer.

4.7.3   .05 G (Entry) Mode

In the Entry Mode, the GDC takes yaw and roll DC body rate signals from:

1) Either Gyro Assembly and computes the roll attitude about the roll stability axis to interface with the RAI on the Entry Monitor System (EMS).

2) Gyro Assembly 1 and computes roll attitude about the roll stability

axis to drive either FDAI 1 or FDAI 2 in roll only.

The roll attitude is computed in the GDC by mechanizing the following equation: $\phi_s = (p \cos \alpha - r \sin \alpha) d t$ where ϕ_s is the roll angle about the stability axis, p and r are the roll and yaw DC body rates respectively, and α is the angle between the CM body and stability axes.

4.7.4   Align Mode

In the Align Mode the GDC takes AC attitude error signals from the Attitude Set Control Panel to align the GDC to the commanded angles dialed into the ASCP whenever the GDC ALIGN switch is depressed. The control resolver outputs are fed (by axis) to the V to F converters to drive the transmitter resolver rotors (through the stepper motors and gear trains) to the same angles as the respective control resolver rotors.

The error signals are disconnected internally in the GDC whenever the Attitude Set switch is placed in the IMU position. During the Align operation, all other inputs and modes of operation are inhibited.

When the RMS ROLL switch is ON and the GDC Align switch is depressed (with the ATT SET switch in the GDC Position), the RAI pointer rotates in response to yaw ASCP thumbwheel rotations.

4.8   ATTITUDE REFERENCE REDUNDANT FEATURES

Backup Rate Redundancy

In the Euler and Single Axis modes, the GDC normally operates on

rate signals from GA 2. However, if a gyro fails in this assembly, it can be switched out internally in the GDC and the corresponding gyro in GA 1 switched in.

During the .05 G mode, the GDC roll axis normally operates on roll and yaw rate signals from GA 1, where as the GDC yaw axis normally operates on roll and yaw rate signals from GA 2. If a gyro in GA 2 fails, it is possible to switch out that gyro and switch in the corresponding gyro from GA 1 without disrupting GA 1 normal operation with the GDC roll axis. However, if a gyro fails in GA 1, it is not possible to switch out that gyro nor switch in the corresponding gyro from GA 2.

Redundant Roll Stability Attitude

During the .05 G mode (or during alignment immediately before the .05 G mode) the roll axis of the GDC operates independently of the yaw axis such that no single failure in the GDC causes loss of both channels.

Two power supplies are used (each one receiving AC power from independent sources). One supplies power to the GDC roll axis and the other to the yaw axis. 28V DC power is supplied to the two channels from alternate busses also.

Alternate Reference for Redundant FDAIs

Since the GDC is required to interface with either FDAI 1 or FDAI 2 and since FDAI 1 operates from 115vac bus 1 and FDAI 2 operates from 115vac bus 2, the output resolver reference is switched as a function of the

FDAI selected. Logic signals (which indicate the FDAI selected) from the Electronics Display Assembly are used to switch the reference to the GDC output resolvers.

4.9   GYRO DISPLAY COUPLER MECHANIZATION

Figure 4.7 is a block diagram showing the functional mechanization of the GDC signal flow paths from the Gyro Assembly interface to the θ, ψ, and φ shafts and the CMC and RAI interfaces. Figure 4.8 is a block diagram showing the signal flow paths from the transmitter resolvers on the GDC shafts and the IMU resolvers to the FDAI balls and (through the ASCP control resolvers) to the FDAI attitude error needles. The following five sections describe the mechanization depicted on these two figures. All relays are shown in the de-energized state. Relays are energized by satisfying the specified logic.

4.9.1   Euler Mode Mechanization

In this section the signal flow paths involved in driving an FDAI ball to display inertial attitude of the S/C and in driving the FDAI attitude error needles to display the difference between actual S/C attitude and the attitude dialed into the ASCP are considered.

4.9.1.1   Basic Functions

The GDC is required to perform two basic functions in the Euler Mode:

1) Compute, store, and continuously update the total attitude of the S/C utilizing DC body rate signals from the Gyro Assemblies.

2) Compute and continuously update the S/C attitude error in each

GYRO DISPLAY COUPLER
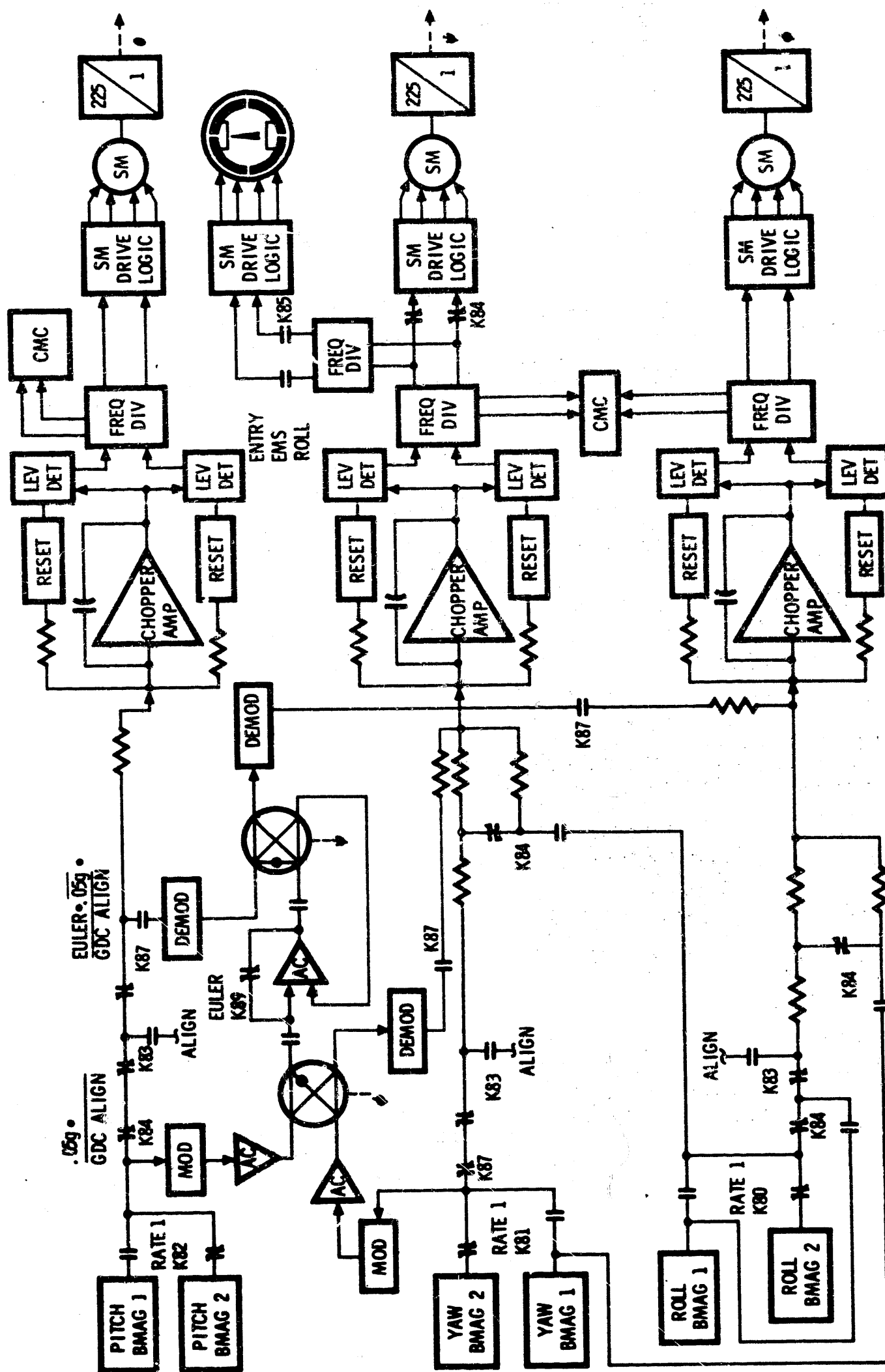
FIGURE 4.7

FDAI SIGNAL FLOW

FIGURE 4.8

4-33
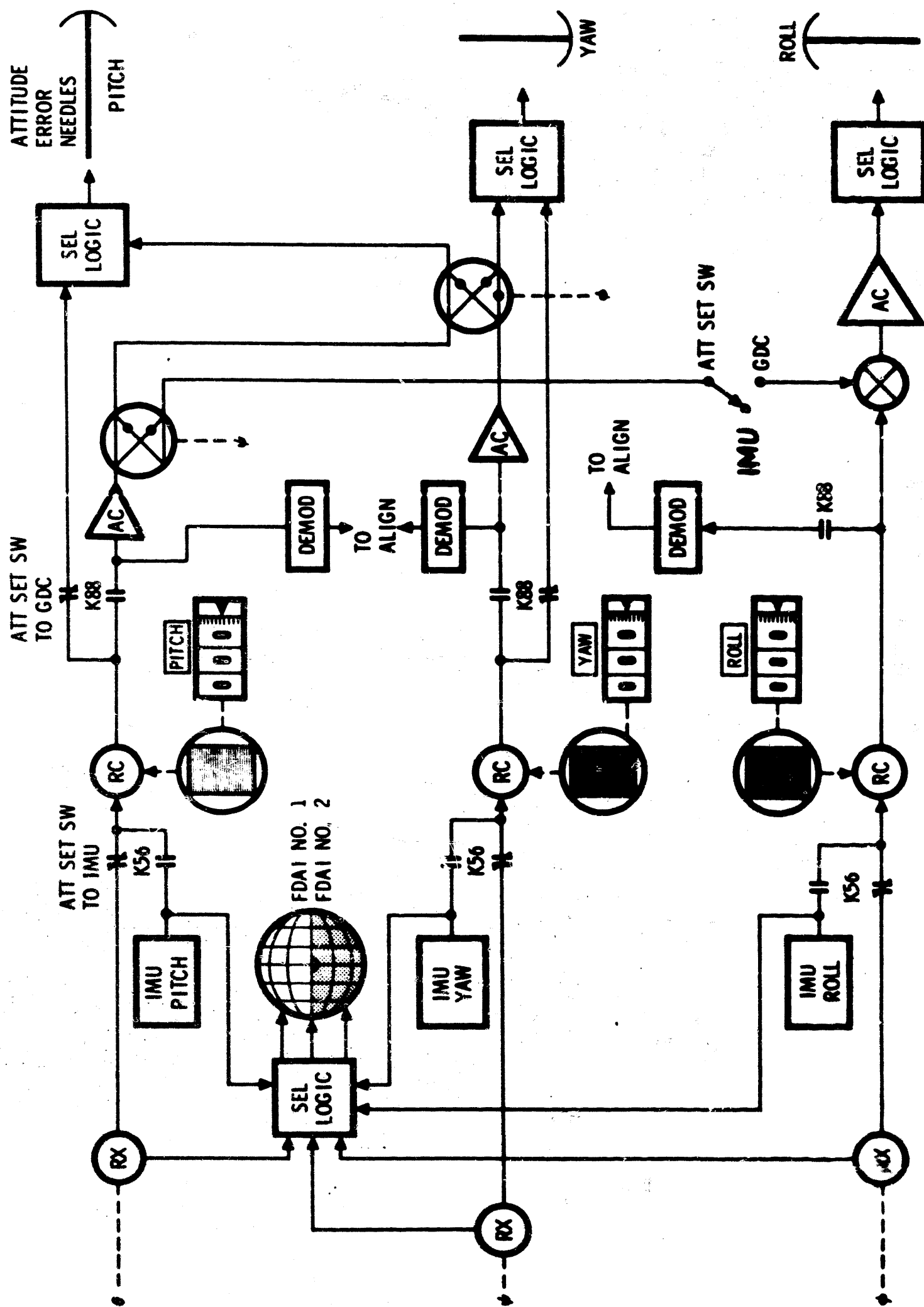
of the three S/C body axes utilizing the inertial referenced
(Euler) attitude error signals from the ASCP.

The Euler angles are computed in the GDC by mechanizing the following
equations:

$$\theta = \int \dot{\theta} \, dt = \int \frac{1}{\cos \psi} (q \cos \phi - r \sin \phi) \, dt$$

$$\psi = \int \dot{\psi} \, dt = \int (r \cos \phi + q \sin \phi) \, dt$$

$$\phi = \int \dot{\phi} \, dt = \int (p - \dot{\psi} \sin \theta) \, dt$$

for $- 100°/sec < \dot{\theta}, \dot{\psi}, \dot{\phi} < + 100°/sec$

and $- 60° < \psi < + 60°$

where q, r, and p are pitch, yaw and roll body rates
respectively; and θ, ψ, and ϕ are pitch, yaw and roll
Euler angles respectively.

The body attitude error angles are computed in the GDC by mechanizing
the following equations:

$$\Delta \theta_b = \Delta \psi \sin \phi + \Delta \theta \cos \psi \cos \phi$$

$$\Delta \psi_b = \Delta \psi \cos \phi - \Delta \theta \cos \psi \sin \phi$$

$$\Delta \phi_b = \Delta \phi + \Delta \theta \sin \psi$$

for $-60° < \psi < + 60°$

where $\Delta \theta_b$, $\Delta \psi_b$ and $\Delta \phi_b$ are the pitch, yaw and roll
body angle errors respectively;

and $\Delta \theta$, $\Delta \psi$ and $\Delta \phi$ are the pitch, yaw and roll Euler
angle errors respectively.

4.9.1.2   DC to AC Conversion

The pitch, yaw and roll gyros in GA 2 (or GA 1 if backup rate is
selected with the appropriate BMAG MODE switches) supply DC signals
proportional to the vehicle angular rates about the respective axes.
For positive S/C angular rates, the pitch and yaw gyros supply -0.1
vdc/°/sec and the roll gyro output is +0.1 vdc/°/ sec. Since the
Euler transformation mechanization utilizes resolvers, the pitch
and yaw DC signals must be converted to AC. (A study of the Euler
equation shows the roll body rate (p) does not have to be modified.)
The DC signals are cycle-chopped by modulators to provide 2200 hz
signals with amplitude equal to that of the DC signal. The choice of
2200 hz is based on the accuracy requirements of the transformation
and the necessity of eliminating 400 hz and 800 hz harmonic interference.
Each of the two modulated signals feed high gain buffer amplifiers.

4.9.1.3   Forward Euler Transformation

The forward transformation (body rate to Euler rate is usually referred
to as the forward transformation while the inertial attitude error to
body attitude error is identified as the reverse transformation)
generates three signals which are functions of the pitch (q) and yaw (r)
body rates and sine or cosine functions of the yaw (ψ) and roll (ϕ) Euler
angles. The resolvers shown in Figure 4.7 are mechanically attached to
the ϕ and ψ GDC shafts as specified. (See Appendix II for explanation
of the resolver representation.)

The Euler Mode is functional when relays K 87 and K 89 are energized.

(The "Euler" logic term is generated when the CMC ATT switch is in the IMU position). Notice the logic on relays K 87: Euler . $\overline{.05g}$. $\overline{GDC\ ALIGN}$. This indicates that the .05g (entry) and GDC align modes take preference over the Euler mode, i.e., the Euler mode is inhibited during the entry phase of the mission and during alignment of the GDC. Relay K 89 serves to prevent the input buffer amplifier for the ψ resolver from saturating in a non-Euler mode when the ψ resolver is positioned near 90° or 270°. Thus when the Euler Mode is mechanized, all relays are in the state shown except relays K 87 and K 89.

Now consider the signals which are conditioned by the forward Euler transformation mechanization consisting of the ψ and ϕ resolvers and the three buffer amplifiers. The 2200 hz modulated pitch rate (-q) and yaw rate (-r) signals are fed to the buffer amplifiers. Input compensation windings within the ϕ resolver provide a second input to the buffer amplifiers to minimize distortion of the signals. The amplifiers provide the necessary gain and inversion of the signals. Thus the signals at the ϕ resolver inputs are +q and +r. The lower output from the ϕ resolver is q sin ϕ + r cos ϕ. This signal represents yaw Euler rate ($\dot{\psi}$). The signal is then demodulated and the DC signal is sent to the yaw V to F converter.

The upper output of the ϕ resolver (q cos ϕ - r sin ϕ) is fed to the third buffer amplifier. Notice that this amplifier has a feedback path through the cosine winding of the ψ resolver. Hence the gain of the buffer amplifier is $-(\cos \psi)^{-1}$ or -secant ψ. An additional

4-36

inversion of this signal is obtained through the compensation winding of the ψ resolver. The compensation winding output is thus secant ψ (q cos ϕ - r sin ϕ). This pitch Euler rate signal ($\dot{\theta}$) is demodulated and sent to the pitch V to F converter.

The ψ resolver input, - secant ψ (q cos ϕ - r sin ϕ), is coupled through the sine winding to provide - tan ψ (q cos ϕ - r sin ϕ) at the output. This signal is demodulated and sent to the roll V to F converter where it is summed with the roll rate signal (+p). Hence the input to the roll V to F converter input is p - tan ψ (q cos ϕ - r sin ϕ) which is roll Euler rate ($\dot{\phi}$).

Thus the three Euler rate signals are fed to their respective V to F converters. The signal flow paths for the three signals from these points to drive their respective ⏦DC shafts are identical.

4.9.1.4 Voltage to Frequency Converter

The V to F converter consists of a chopper stabilized DC amplifier and complementary positive and negative level detectors and reset circuits. Its function is to convert analog DC signals into proportional digital output pulses. Two outputs are generated, one for each input polarity.

The design incorporates accurate integration of the input by means of the chopper stabilized amplifier with capacitive feedback. When the integrator output reaches the level for which the level detectors are calibrated, the appropriate level detector provides a digital

pulse to the output and a command to the reset circuit. The reset circuit generates an accurate current pulse to the integrator input which resets the integrator. The transfer function of the V to F converter is 200 pulses per second output for each volt input. Since the input scale factor is 0.1 vdc per degree per second, the V to F converter output scale factor is 20 pps per $^\circ$/sec. In other words each pulse out of the V to F converter represents a change of 0.05 degree at the input.

4.9.1.5   <u>Frequency Divider</u>

The frequency dividers consist by binary counters. The counters divide the V to F converter digital pulse train by 2 (for outputs to the CMC) and by 4 in the pitch, yaw and roll channels, and also by 8 in the yaw channel (for driving the RAI in the EMS). However, in the Euler mode we are concerned only with the division by 4 in each channel. The frequency divider output scale factor is thus 5 pps per $^\circ$/sec, i.e. each pulse represents 0.2 degree.

4.9.1.6   <u>Stepper Motor Drive Logic</u>

The stepper motor drive circuit controls the excitation phase of the two coils on the stepper motor. Each pulse causes the motor to step one increment. The phase of the pulses determines the direction the motor steps. Thus, the direction and speed of the motor is controlled by this drive circuit.

4.9.1.7 Stepper Motor - Gear Train

The position of the resolver shaft is controlled by the stepper motor through a 225:1 gear train reduction. The stepper motor drives the gear train input shaft in 45 degree increments. Hence, the resolver shaft (being connected to the gear train output) is driven in increments of 0.2 degrees. Recall that the pulses which drive the stepper motor represent 0.2 degrees attitude change at the V to F converter input. Thus the shaft position for each channel represents the Euler angle in the corresponding axis.

4.9.1.8 Resolver Assemblies

The GDC contains three resolver assemblies, one on each of the shafts which are connected to the gear train outputs. The pitch resolver assembly has a single 400 hz transmitter resolver (RX) on its shaft. The yaw and roll resolver assemblies are identical, each having one 400 hz transmitter resolver and two computational resolvers in tandem on their shafts. One of the computational resolvers (Euler rate) is operated at 2200 hz while the other (body attitude error) is operated at 400 hz. The body attitude error output to the displays is obtained directly from the computation resolvers. The Euler attitude information for display is obtained from the transmitter resolver of the respective axis.

4.9.1.9 FDAI Ball Drive Circuitry

The function of the FDAI ball drive circuitry is to position the ball in the 3 axes to the same angles as the 3 GDC resolver shafts. As

indicated in Figure 4.8, the FDAI ball can be driven from the IMU resolvers or the GDC RX resolvers. The select logic in the EDA determines which control path is enabled. The select logic is controlled by the control panel switches. In the Euler mode we are concerned with driving the ball to the attitude represented by the GDC resolver shaft positions.

Figure 4.9 is a pictorial drawing showing the GDC shafts, the FDAI gimbal resolvers and motors and a simplified version of the servo drive electronics located in the EDA. The 400 hz total attitude signals are fed from the GDC RX resolvers to the respective FDAI gimbal resolvers. If an FDAI gimbal resolver is positioned at an angle different from the respective GDC resolver, an Euler error signal is fed from the FDAI resolver to the respective EDA servo amplifier. The servo amplifier output drives the respective FDAI gimbal motor until the Euler error signal is nulled (at which time the FDAI gimbal angle is equal to the respective GDC shaft angle). Dynamic stability of the servo loop is ensured by providing a negative feedback signal to the servo amplifier proportional to the velocity of the gimbal motor from the velocity generator (tachometer) which is part of the motor assembly.

4.9.1.10   <u>FDAI Attitude Error Indicator Drive Circuitry</u>

In the Euler Mode the attitude error needles can be driven by the SCS from either the uncaged BMAGs or the ASCP-GDC mechanization, depending upon the state of the select logic. With the BMAG source, GA 1 provides

FDAI/GDC SHAFT RELATIONSHIP

ROLL INDEX

PITCH + YAW INDEX

OGA RESOLVER & MOTOR

EULER ERROR

IGA RESOLVER & MOTOR

BEARING

EULER ERROR

MGA RESOLVER & MOTOR

EULER ERROR

TOTAL ATTITUDE

TOTAL ATTITUDE

+400 HZ EXC.

EULER ERROR

X_B

Y_B

Z_B

"MGA" RESOLVER

TOTAL ATTITUDE

"OGA" RESOLVER

"IGA" RESOLVER

ATTITUDE REFERENCE SYSTEM STRAP DOWN PLATFORM

+XB

+YB

+ZB

FIGURE 4.9

4-41

400 hz signals proportional to the difference between the actual attitude of the S/C and the attitude of the S/C at the time the BMAGs were uncaged. These body referenced attitude error signals are fed through the EDA logic, scaling, signal conditioning and servo drive electronics to deflect the appropriate FDAI attitude error needles.

When utilizing the attitude set mechanization, the 400 hz total attitude signals are fed from the GDC transmitter resolvers (RX) to the respective ASCP control resolvers (RC) through the normally closed K 56 relay contacts (see Figure 4.8). The control resolver outputs are proportional to the difference between the Euler attitude (RX angles) and the desired attitude (RC angles) dialed in the ASCP via the thumbwheels. The resultant attitude error signals out of the control resolvers are referenced to the inertial (fixed) reference frame. It is desirable to display the attitude errors relative to the S/C (body) axes because the changes in attitude are commanded and performed relative to these axes via the Rotation Control, electronics and RCS engines. Thus the Euler attitude error signals at the control resolver outputs are fed through the normally open K 88 relay contacts to the reverse (Euler to body) transformation.

The reverse transformation equations (as previously stated) are:

$$\Delta\theta_b = \Delta\psi \, \sin\phi + \Delta\theta \, \cos\psi \, \cos\phi$$

$$\Delta\psi_b = \Delta\psi \, \cos\phi - \Delta\theta \, \cos\psi \, \sin\phi$$

$$\Delta\phi_b = \Delta\phi + \Delta\theta \, \sin\psi$$

where $\triangle\theta_b$, $\triangle\psi_b$ and $\triangle\phi_b$ are the attitude errors referenced to the pitch, yaw and roll body axes; $\triangle\theta$, $\triangle\psi$ and $\triangle\phi$ are the attitude errors referenced to inertial reference frame; and θ, ψ and ϕ are the Euler angles.

$\triangle\theta$ from the pitch RC is fed thru a buffer amplifier whose output is $-\triangle\theta$. This signal is coupled through the ψ resolver to yield $-\triangle\theta \cos\psi$ and $+\triangle\theta \sin\psi$. The latter signal is summed with $\triangle\phi$ from the roll RC in the AC amplifier to yield the $\triangle\phi_b$ signal to the roll attitude error electronics in the EDA. The former signal is fed through a buffer amplifier to provide $+\triangle\theta \cos\psi$ at the top input to the ϕ resolver. The yaw RC output is fed through a buffer amplifier to provide $-\triangle\psi$ at the lower input to the ϕ resolver. These two signals are coupled through the ϕ resolver to provide $\triangle\theta_b$ and $\triangle\psi_b$ signals to pitch and yaw attitude error electronics respectively in the EDA. The EDA electronics contains the logic, scaling, signal conditioning and servo electronics for properly driving the FDAI attitude error needles.

4.9.2   <u>Single Axis (Backup IMU) Mode Mechanization</u>

In this section the signal flow paths involved in providing digital body rate pulses to the Command Module Computer of the G&N system are considered. The Single Axis mode is enabled by placing the CMC ATT switch in the GDC position.

Analog Body Rate Signals

The pitch, yaw and roll gyros in GA 2 (or GA 1 if backup rate is selected with the appropriate BMAG MODE switches) supply DC signals proportional to the S/C angular rates about the respective axes. For positive S/C angular rates, the pitch and yaw gyros supply -0.1 vdc/^{o}/sec and the roll gyro output is +0.1 vdc/^{o}/sec. In the Single Axis mode none of the relays are energized. Hence, it is apparent that the Single Axis mode is enabled only if the other three modes are disabled. When in the Single Axis mode, the gyro DC rate outputs are fed directly (through the K 83, K 84, and K 87 normally closed contacts) to the respective V to F converters.

Generation of Digital Body Rate Signals

The V to F converters act on the inputs to produce digital body rate signals (pulse trains) as described in section 4.9.1.4. These signals are fed to the frequency dividers (see section 4.9.1.5). The frequency divider output scale factor at the CMC interface is 10 pps/^{o}/sec. Thus each pulse represents a change in attitude about the respective axis of 0.1 degree. It should be noted that the phasing of the roll gyro output differs from that of the pitch and yaw gyros. The correction for this difference is performed at the interface between the V to F converters and the CMC. It should also be pointed out that it is doubtful whether the CMC will be programmed to utilize these body rate pulses. However, the SCS is still required to meet all the performance specifications for this mode.

4.9.3	.05g (Entry) Mode Mechanization

In this section the signal flow paths involved in driving an FDAI ball and the Roll Attitude Indicator of the EMS to display the relative position of the CM lift vector about the roll stability axis are considered.

Basic Functions

The GDC is required to perform three basic functions in the Entry Mode:

1) Compute, store and continuously update the S/C attitude relative to the roll stability axis for display on an FDAI utilizing DC body rate signals from GA 1.

2) Compute and continuously update the S/C attitude relative to the roll stability axis for display on the RSI utilizing DC body rate signals from GA 1 or GA 2.

3) Ensure that no single point failure results in loss of the ability to display Roll Stability attitude.

Roll stability attitude is derived from yaw and roll body rates (r and p respectively) by mechanizing the equation:

$$\phi_s = \int \dot{\phi}_s \, dt = \int (p \cos \alpha - r \sin \alpha) \, dt$$

Generating Roll Stability Rate

As indicated by the equation above, the GDC is mechanized to derive roll stability attitude by integrating roll stability rate. Roll stability rate is generated by summing modified yaw and roll body rates. The relay control logic shown in Figure 4.7 indicates that the .05g (entry) mode take precedence over all modes except the GDC Align Mode.

Selection of the Entry mode energizes relays K 84. This removes inputs to the pitch channel V to F converter and opens inputs to the yaw channel Stepper Motor Drive Logic. Thus the pitch and yaw GDC shafts cannot be driven in the entry mode.

Energizing K 84 provides summing of GA 2 yaw and roll outputs in the yaw channel V to F converter; and likewise summing of GA 1 outputs in the roll channel. The summing resistors are chosen for each channel such that the roll gyro outputs are modified by the constant $\cos\alpha$, and the yaw gyro outputs by $\sin\alpha$. Recall that for positive S/C rates the respective gyro outputs are +p and -r. Thus the inputs to the yaw and roll channel V to F converters represent roll stability rate.

<u>Redundant Features</u>

Selecting backup rate in either yaw or roll allows the respective GA 1 gyros to supply signals to both the yaw and roll channels. There is no redundant rate source for the roll channel. The GDC power supplies, logic, and signal flow mechanization are designed such that any single point failure in the sensors, GDC or displays will not prevent the ability to display valid roll stability attitude.

<u>Display Drive Circuitry</u>

The mechanization for driving a FDAI in roll from the GDC roll channel is identical to that described in Sections 4.9.1.4 to 4.9.1.9. In the yaw channel the V to F converter output pulse train is divided by 8

(Reference Section 4.9.1.5) yielding pulses which represent 0.4 degree each. If the ENTRY EMS ROLL switch is ON, relay K 85 is energized, allowing the pulses into the stepper motor drive logic (Reference Section 4.9.1.6). The commands from this circuitry are fed to the stepper motor - gear train-indicator assembly (RAI) in the Entry Monitor System.

4.9.4   ALIGN MODE

In this section the signal flow paths involved in driving the GDC resolver shafts and the RAI to the angles commanded (via the ASCP) are considered.

Aligning the GDC

To properly align the GDC, the attitude of the S/C relative to the chosen reference frame must be known. As described in section 4.1.3, S/C attitude is defined by pitch, yaw and roll Euler angles. These angles can be determined by utilizing the G&N optics or interrogating the G&N computer via the Display Keyboard. The angles are then dialed into the ASCP via the thumbwheels. The Align mode is then enabled by energizing relays K 88 (Figure 4.8) and K 83 (Figure 4.7) which provides a closed loop for each axis. Note that the Align mode takes precedence over all the other modes. If RX and RC are not positioned to the same angles, a 400 hz error signal present at the RC output is fed through the NO K 88 contacts to the demodulator. The demodulator output is fed through the NO K 83 contacts to the V to F converter. From this point the GDC is mechanized as described in Sections 4.9.1.4 to 4.9.1.8 to drive the proper RX to the RC angle. When the two resolver angles are equal no error signal is present and the stepper motor is not driven.

The GDC can be accurately aligned to the IMU inertial reference frame as follows: Set the control panel switches to display IMU-ASCP attitude errors. Relays K 56 will thus be energized and K 88 de-energized. Next adjust the ASCP thumbwheels to obtain nulls on the FDAI attitude error needles. The ASCP resolvers are now set at the IMU referenced S/C attitude angles. Then move the ATT SET switch from IMU to GDC (which de-energizes K 56 and energizes K 88 relays) and depress the GDC ALIGN button which energizes relays K 83. The GDC shafts will then be positioned to the ASCP RC angles, aligning the GDC to the IMU reference frame.

Aligning the RAI

Prior to encountering the earth's atmosphere for the entry phase of the mission, the CM will be oriented with lift vector up or down depending upon the calculated entry angle. The RAI must be aligned to agree with the S/C entry attitude. This is accomplished by energizing K 88 and de-energizing K 56 relays (ATT SET switch to GDC), energizing K 85 relay (ENTRY switch to EMS ROLL), energizing K 83 relay (depress GDC ALIGN button) and rotating the yaw ASCP thumbwheel until the RAI needle is in the desired position.

4.10   SOURCE TO DISPLAY MECHANIZATION

This section describes the functional mechanization of the basic signal flow paths for driving the FDAI ball, attitude error needles and rate needles.

4.10.1   Attitude Indicator Mechanization

Figure 4.10 presents a block diagram description of the attitude indicator mechanization for a single axis. The mechanization is the same for the pitch, yaw, and roll axes. The proper FDAI is enabled by energizing relays K2, K3 and K4. K2 and K3 provide the attitude signals to the FDAI resolver and the reference voltage to the FDAI motor - VG windings. K4 enables the motor drive buffer amplifier. The K1 relays are shown in the state necessary for the FDAI to display IMU gimbal position. The IMU gimbal resolver outputs are 800 hz signals equal to A sin θ and A cos θ where A is the value of the RMS excitation voltage and θ is the gimbal shaft angle. These signals are fed to the FDAI resolver whose output is equal to B sin (θ - δ), where δ is the FDAI resolver angle. The resolver output is amplified and fed to the demodulator with the 800 hz reference voltage. The demod output is amplified, modulated at 400 hz and applied to the motor control winding thru the buffer amplifier. If θ is equal to δ this control voltage is zero and the motor is not driven. This is the condition attained when the spacecraft attitude is not changing. Now if the S/C attitude changes, the gimbal position will be changed to keep the IMU platform fixed in inertial space. The change in gimbal position causes the amplitudes of the two gimbal resolver outputs to change, resulting in an error signal out of the FDAI resolver, which generates a control voltage on the motor winding. The motor drives the VG, resolver and ball until the resolver output is nulled. In this way the FDAI resolver and ball are maintained at the same angle as the IMU gimbal. The VG signal provides negative feedback to the DC amplifier, controlling the

ATTITUDE INDICATOR MECHANIZATION, ONE AXIS
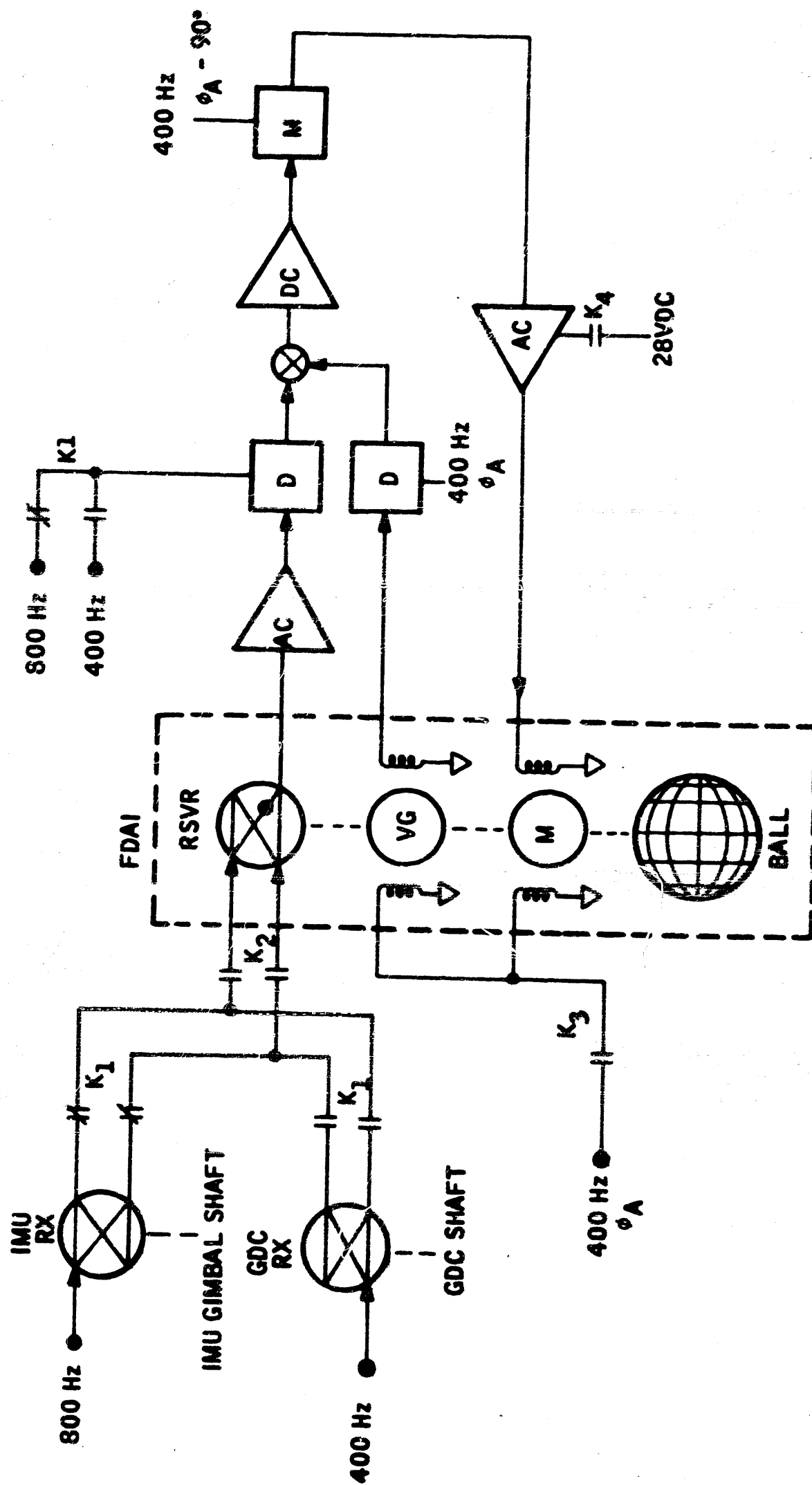
FIGURE 4.10

maximum speed of the motor and maintaining a stable loop. The FDAI ball has a maximum slew speed of 60 degrees per second in each axis.

Energizing the K1 relays replaces the IMU gimbal resolver signals with the GDC resolver outputs. The error signal demod reference is also changed to 400 hz to be compatible with the source. Except for these two changes, the mechanization is the same as described above.

4.10.2   Attitude Error Indicator Mechanization

Figure 4.11 presents a block diagram description of the attitude error indicator mechanization for the pitch and yaw axes. The four sources of attitude error signals are shown on the left. Separate, identical circuits are mechanized for each FDAI. Signals from each source are fed through separate electronic switches to an AC Amplifier. Only one of the four switches is enabled to allow an error signal to the amplifier. (see Figure 4.4 for a description of the select logic.) The FDAI scale change is effected by SW 5. Enabling SW 5 increases the amplifier gain by a factor of three, changing the attitude error indicator full scale deflection value from 15° to 50°. The state of relay K1 is controlled to provide the proper demod reference voltage for the source selected. The demod output is a DC voltage proportional to the magnitude of the error signal generated by the source. The DC amplifier generates a current proportional to its input voltage which drives the torquer coil of the servometric mechanism. The indicator needle and feedback pot wiper are mechanically linked to the torquer coil. The DC amplifier provides an output until the feedback voltage

PITCH (OR YAW) ATTITUDE ERROR INDICATOR MECHANIZATION, ONE CHANNEL

FIGURE 4.11

4-52
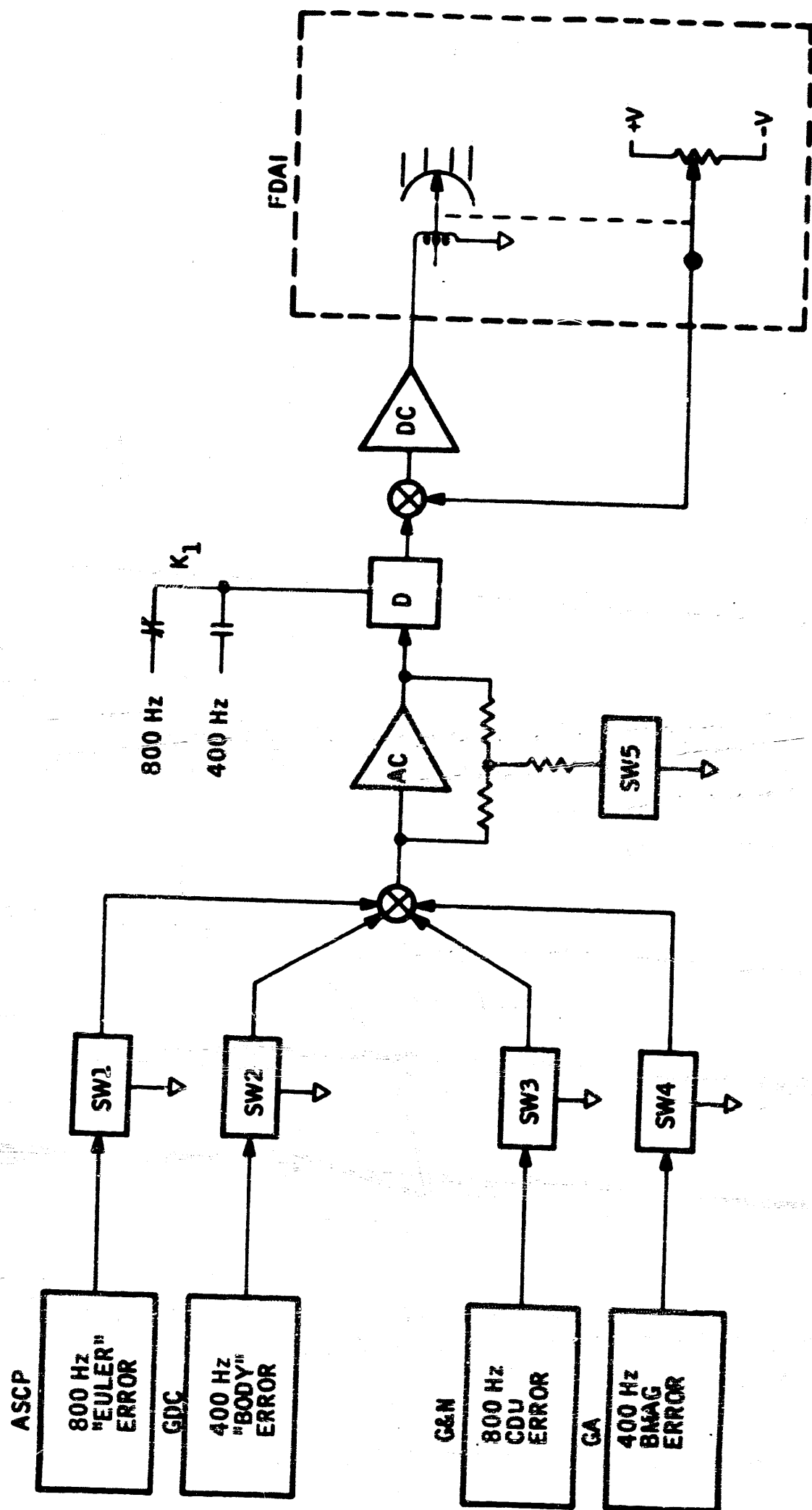
from the potentiometer is equal in magnitude and of opposite sign to the demod output. At this time, the position of the indicator needle on the scale describes the magnitude and polarity of the attitude error as determined by the source.

Figure 4.12 shows the mechanization of a roll attitude error channel. Only the two differences between it and the pitch and yaw mechanizations will be discussed, namely the ASCP and GDC source switching and the scale change switching.

The source switching mechanization is different in the roll channel due to the summing requirements for the roll body attitude error signal (see Section 4.9.1.10). The Euler error source (IMU-ASCP) is enabled by providing continuity through SW 3 with SW 1 and SW 2 disabled. The GDC "body" error source is enabled by enabling SW 1, which allows summing of the two signals which provide the roll body error signal at the AC amplifier input, and SW 2 which changes the amplifier gain to proper value. SW 3 is also enabled to provide continuity to the scale controlling amplifier.

The roll channel must provide three different gain values as functions of the source and scale selected. For low (5°) scale, maximum gain is required for all sources. (The scale factors for signals from the individual sources are adjusted by SW 1 through SW 5). Maximum gain is provided by the AC amplifier with SW 6 and SW 7 enabled, providing continuity to ground. For high (50°) scale, the channel gain is

ROLL ATTITUDE ERROR INDICATOR MECHANIZATION, ONE CHANNEL

FIGURE 4.12

4-54
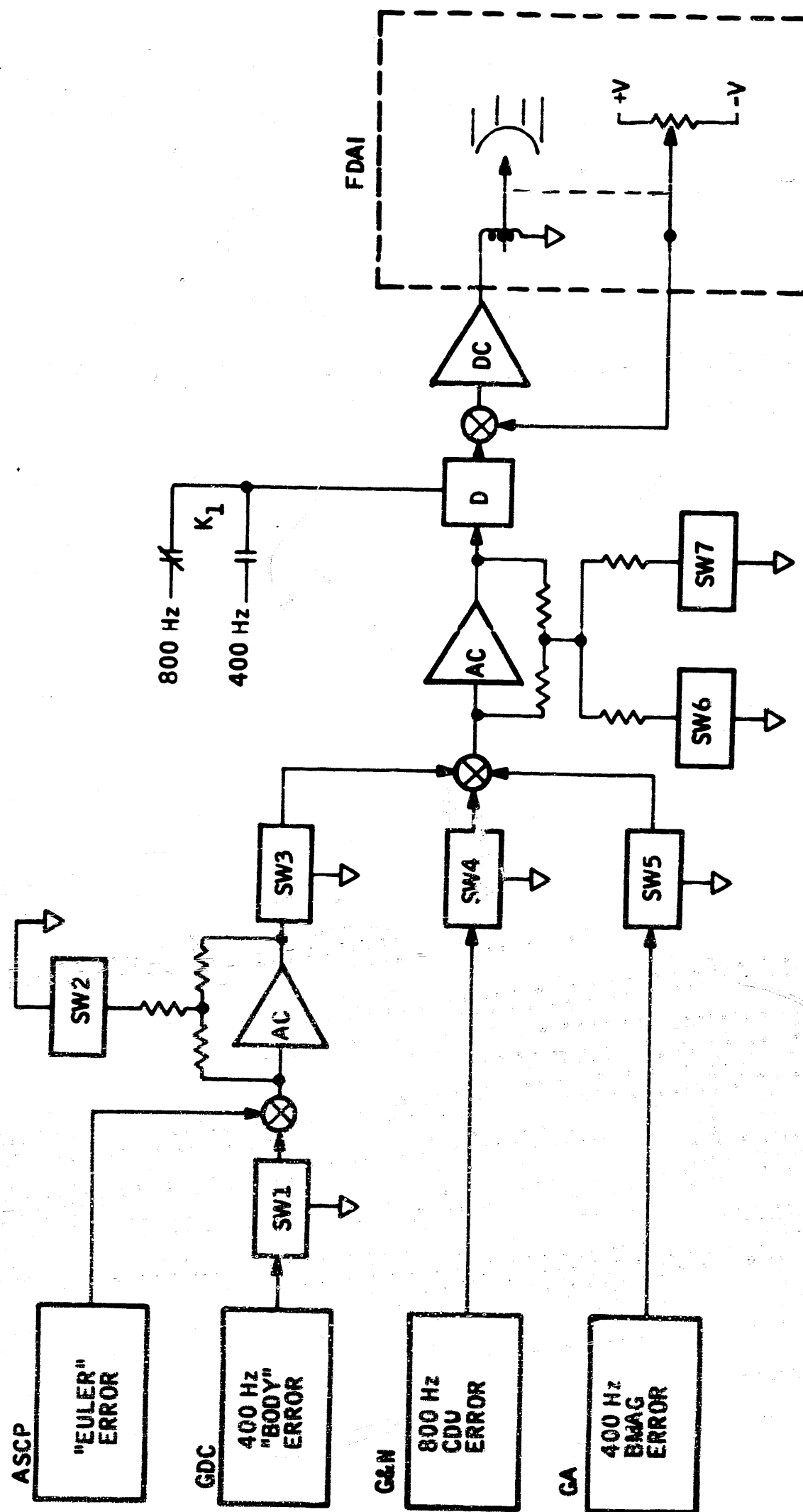
decreased by a factor of 10 for all sources except the G&N CDU by

disabling SW 6 and SW 7. The 50° scale is normally selected only during

the entry phase of the mission. The CDU scale factor is normally 0.3

volts per degree. Since the CDU output is limited to 5 volts (about

17 degrees) the CDU roll scale factor is decreased to 0.075 volts

per degree at the entry interface. Thus, when the FDAI is displaying

attitude errors from the CDU source with the high scale selected,

the roll channel gain is decreased (from the low scale value) by a

factor of 2.5 instead of 10. This is accomplished by disabling SW 6

and leaving SW 7 enabled.

4.10.3   Rate Indicator Mechanization

Figure 4.13 depicts the mechanization for driving the rate indicators

on an FDAI. The mechanization is identical for FDAI 1 and FDAI 2.

A description of the logic for the switches will not be emphasized

here since it is given in Figure 4.5. The mechanization allows the

rate indicators of either or both FDAIs to be driven from the caged

BMAGs of either GA by axis.

The source for each axis is selected by electronic switches SW 1

through SW 6. During entry the yaw rate indication must be

proportional to yaw "stability" rate (see Appendix 3). This

requires summing a modified roll body rate signal into the yaw axis.

SW 7 and SW 8 provide this capability. During entry SW 8 will be

enabled only if both SW 4 and SW 5 are enabled; and SW 7 will be

enabled only if SW 2 is enabled. SW 2 will be enabled if either SW 4

or SW 5 is disabled during entry.

RATE INDICATOR MECHANIZATION, ONE CHANNEL

FIGURE 4.13

4-56
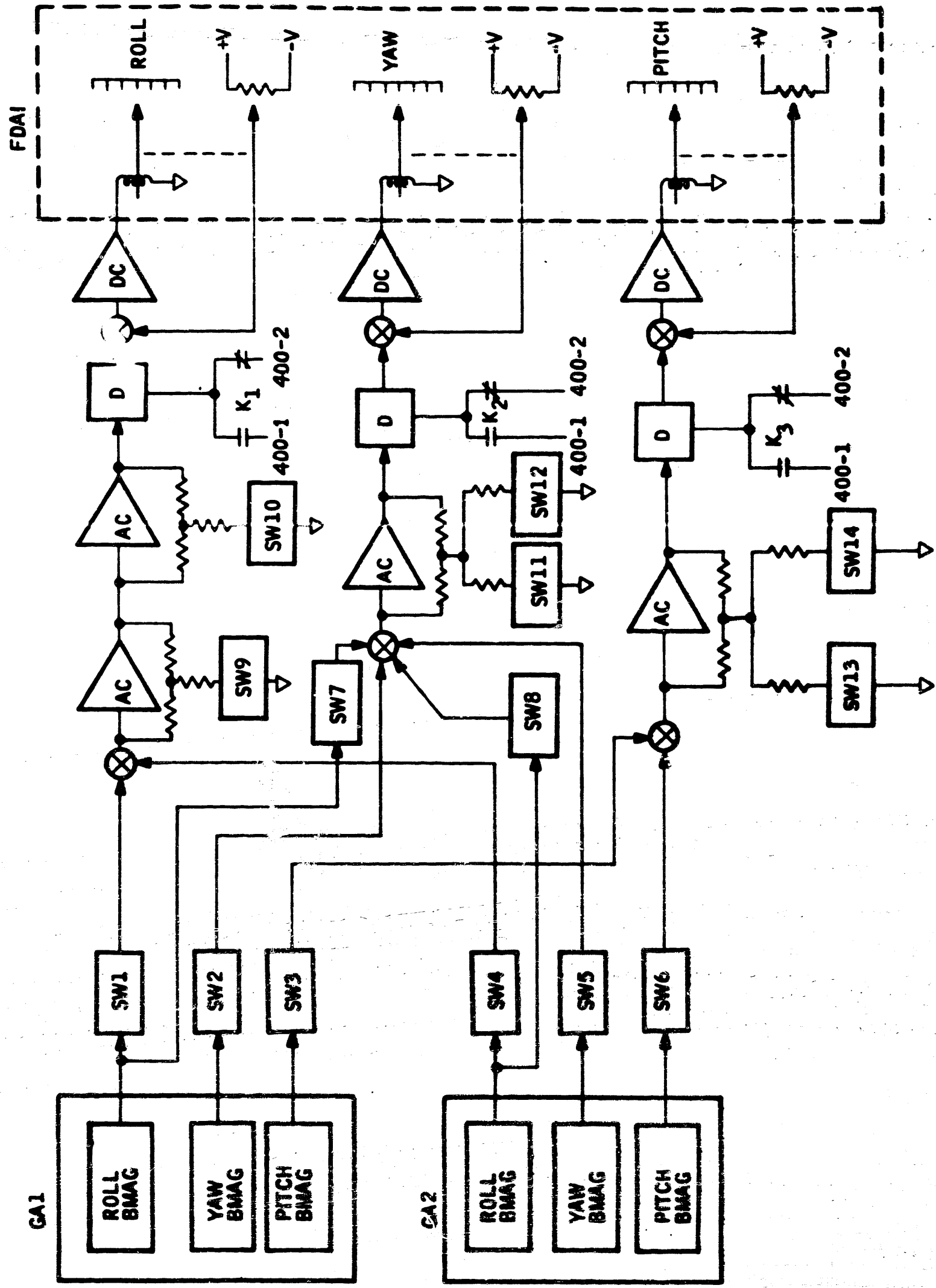
The single FDA SCALE switch on control panel 1 determines the gain of each axis for both FDAIs for the rate and attitude error indicators simultaneously. The rate scale factors are controlled by SW 9 through SW 14. The pitch and yaw scaling amplifiers are identical. The AC amplifiers provide maximum gain for the 1°/sec full scale deflection value with SW 11 through SW 14 closed. The amplifier gains are decreased by a factor of five to provide 5°/sec full scale deflection values by opening SW 12 and SW 14. The gains are decreased by an additional factor of two by opening SW 11 and SW 13 to provide a full scale deflection value of 10°/sec.

Since the roll scales are changed by a factor of 50, two AC amplifiers must be utilized to provide the required linearity and accuracy. Each amplifier provides maximum gain for a full scale deflection value of 1°/sec with SW 9 and SW 10 closed. Opening SW 10 decreases the gain of the second amplifier by a factor of five, changing the full scale value to 5°/sec. Full scale values are increased to 50°/sec by opening SW 9 to decrease the first amplifier gain by a factor of ten.

K1, K2, and K3 are controlled to provide the bus 1 reference voltage when a GA 1 source is being used and the bus 2 reference when the channel is supplied by a GA 2 source. The operation of the DC amplifier and the servometric meters is the same as described for the attitude error mechanization.

4.10.4 **ARS Switching Summary**

Chart 4-1 summarizes the effect Control Panel 1 switches have on the Attitude Reference System configurations and on the total attitude, attitude error, and attitude rate display sources.

4.11 **ATTITUDE REFERENCE SUBSYSTEM PERFORMANCE REQUIREMENTS**

This section gives a description of the accuracy requirements on the total attitude, attitude error and rate information which is displayed on either FDAI.

4.11.1 **Total Attitude**

First we will consider the requirements on the total attitude display. Since the accuracy of the total attitude presented by the FDAI is a function of several factors, we will consider first the general requirements on the FDAI ball and servo loop and then the accuracy requirements for various types of maneuvers.

4.11.1.1 **FDAI Ball and Servo Loop**

The total attitude indicated on the FDAI ball will not differ from the attitude derived by the source (IMU or GDC) by more than $1.5^\circ + 0.25^\circ$ per $^\circ$/sec change of the source attitude. The damping ratio of the FDAI is 0.7. Any change in the source attitude of 0.4° will produce a perceptible change in the attitude displayed. The attitude can be read to within 2° when the ball is positioned near zero degrees yaw. The maximum slew speed of the ball is at least 50°/sec in each axis.

ARS SWITCHING

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

(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 4.1

4.11.1.2 Low Rate Maneuver

If the body axis rates are limited to 1°/sec in each axis for a period of time not greater than five minutes, and the total change in attitude is not greater than 200° roll, 60° pitch and 60° yaw, and the yaw euler angle at no time exceeds $+60^\circ$, then the maximum error at the completion of the maneuver will be 2.5° roll, 1.5° pitch, and 1.5° yaw.

4.11.1.3 Large Angle Maneuver

If the body axis rates are limited to 5°/sec, and the resulting euler rates are less than 60°/sec in each axis for a period of time not greater than five minutes, and the total change in attitude is not greater than 180° roll, 180° pitch, and 60° yaw, and the maneuver is initiated with all euler angles zero, then the maximum error at the completion of the maneuver will be 5.0° in each axis.

4.11.1.4 Single Axis Maneuver

With the same conditions imposed as for the large angle maneuver, except as indicated, the performance requirements for maneuvers about a single body axis are as stated below.

If the total change in attitude is not greater than 185° roll, 5° pitch and 5° yaw, then the maximum error at completion of the maneuver will be $0.9^\circ + 0.00723 \, \Delta\phi$ roll, 0.9° pitch and 0.9° yaw.

If the total change in attitude is not greater than 5° roll, 185° pitch and 5° yaw, then the maximum error at completion of the maneuver will be 0.9° roll, $0.9^\circ + 0.00723 \, \Delta\theta$ pitch and 0.9° yaw.

If the total change in attitude is not greater than 5° roll, 5° pitch and 65° yaw, then the maximum error at completion of the maneuver will be 0.9° roll, 0.9° pitch and $0.9^\circ + 0.00723 \Delta \psi$ yaw.

4.11.1.5   **High Rate Maneuver**

If the body axis rates are limited to 15°/sec in each axis for a period of time not exceeding five minutes, and the euler angles never exceed 20° in any axis, and the total attitude change does not exceed 150° in any axis, and if the initial and final euler angles are zero, then the maximum error at the completion of the maneuver will be 1.05° in each axis.

Any one of the euler angles may exceed 20° if the $(\cos \psi(t) \times \cos \phi(t))$ is greater than $\cos 60^\circ$, and the body axis rates are limited to 15°/sec roll, 50°/sec pitch and 50°/sec yaw resulting in euler rates less than 60°/sec in each axis, with the maximum error at the completion of the maneuver still limited to 1.05° in each axis.

4.11.1.6   **Pre-Entry Maneuver**

The maximum errors in the display of total attitude for a period of 25 minutes during pre-entry will be 5° in each axis. The constraints imposed on the maneuver are as follows: the maneuver period is not more than five minutes of the 25 minutes; the maneuver is initiated with a yaw euler angle of 0 degrees and a roll euler angle of 0 degrees or 180 degrees; the maximum body axis rate is 50°/sec in each axis; the maximum excursions about the body axes from the initial attitude

are not more than 270° roll, 180° pitch and 60° yaw; the total change in attitude about the body axes is not more than 360° roll, 240° pitch and 120° yaw; the maneuver is completed by axis in the order pitch-roll-yaw, with one rotation completed before the second is begun.

4.11.1.7 Launch Attitude Indication

The maximum errors in the display of total attitude for a period of 20 minutes after a GDC alignment during the Apollo launch phase of the mission will be 10° in each axis if the following constraints are imposed during the maneuver: maximum body axis rates are 20°/sec roll, 5°/sec pitch and 5°/sec yaw; the maximum deviations about the body axes from the attitude at GDC alignment are $-60° \leq$ roll $\leq +60°$, $-30° \leq$ pitch $\leq +120°$, and $-30° \leq$ yaw $\leq +30°$; the total change in attitude about the body axes is not more than 90° roll, 180° pitch and 60° yaw.

4.11.1.8 Entry Roll Attitude Displays

The performance requirements for the display of lift vector position (stability roll attitude) on the FDAI and the Roll Stability Indicator of the Entry Monitor System are tabulated below for five cases:

CASE	t_{max} (minutes)	F_{max} (g-sec)	Q_{max} (degrees)	R_{max} (degrees)	Accuracy FDAI (deg)	Tolerance RSI
I	2	350	360	360	4	2.8
II	5	850	360	2000	9	6.5
III	20	1400	360	2000	10	7.5
IV	20	1400	1000	2000	12	10
V	20	1400	1000	3000	14	12

"t" is the elapsed time after the entry interface; "F" is the total force integral, i.e. $F = \int_0^t$ (force in "g's) along the positive entry roll axis) dt; "Q" is the total cumulative maneuver angle (algebraic), i.e., $Q = \int_0^t$ (stability roll rate) dt; and "R" is the total cumulative maneuver angle (absolute), i.e. $R = \int_0^t$ (absolute value of the stability roll rate) dt.

4.11.2 Attitude Error

The accuracy of the attitude error indication on the FDAI is 1.75% of the full scale value. The attitude error indicator is capable of being interpreted to within 5% of the full scale deflection value when observed from the nominal viewing position. The damping factor of the servometric display is a minimum of 10. The response time of the servometric display is a maximum of 0.75 seconds.

4.11.3 Attitude Rate

The accuracy of the attitude rate indication on the FDAI is 1.5% of the full scale value. The viewing requirements, damping factor, and response time of the servometric rate indicators are the same as those of the attitude error indicators.

SECTION 5

REACTION JET CONTROL SUBSYSTEM

5.1   <u>INTRODUCTION</u>

This section describes the methods and mechanizations available for controlling the spacecraft via the Reaction Control System. The function of the Reaction Jet Control subsystem is to provide translation and rotation control of the spacecraft. This control is accomplished by providing ON/OFF command signals to the proper RCS thrusters. An ON command to the automatic or direct coils energizes separate solenoid valves in the fuel and oxidizer lines. Opening the valves allows the propellants to enter the thrust chamber. The use of hypergolic propellants eliminates the need for an ignition system, since ignition occurs upon contact of the fuel and oxidizer.

CM and SM Control Systems

The Apollo S/C can be controlled by firing various combinations of the 16 thrusters on the Service Module or the 12 thrusters on the Command Module. The same control circuits are utilized whether the spacecraft is controlled by the CM or SM thrusters. In a normal lunar mission, the SM RCS is used until the SM is jettisoned prior to earth entry. The CM RCS will normally not be used prior to separation of the CM from the SM. The basic difference between the two systems is that the SM RCS offers translation control (linear acceleration along any S/C axis, or any combination of the spacecraft axes). Either the CM or SM RCS (but only one at a time) can be controlled to provide various

types of attitude or rotational control about each S/C axis and any combination of S/C axes.

Attitude and Translation Control Capabilities

The mechanizations for providing attitude (rotational) control of the spacecraft are numerous and fairly involved, while the translation control provisions are relatively few and simple. Translation of the spacecraft through the use of the RCS is accomplished primarily by providing manual inputs through the Translation Control. Control of the spacecraft rotation is possible by various manual and automatic means.

Manual and Automatic Control Capabilities

The Reaction Jet Control subsystem provides both manual and automatic control mechanizations. The automatic control mechanization is slightly more complex, but much of the circuitry is the same for manual and automatic control. However, the automatic control mechanizations (except for CM-SM separation and a few abort sequences) provide control signals to the RCS auto coils of the solenoid valves. Manual control of the spacecraft is possible by providing signals to either the auto coils or the direct coils.

Auto Coil vs Direct Coil RCS Control

Both the CM and SM RCS have auto and direct coils. Spacecraft translation and rotation can be controlled by either the direct or auto coils. The normal method of controlling the spacecraft (except during CM-SM separation

prior to entry and a few abort sequences) is through the auto coils.
The direct coils are usually utilized in an emergency situation.

Primary and Backup Modes

Commands for controlling the spacecraft via the RCS originate in
either the G&N or the SCS system. Selection of alternate mechanizations
for various types of control is provided by control panel switches.
The switches provide discrete signals to the SCS logic which provides
enable-disable control for the various signal flow paths.

5.2   SCS RJC COMPONENTS

The SCS devices utilized in the Reaction Jet Control subsystem include:

Reaction Jet & Engine ON/OFF Control Assembly (RJ/EC)

Electronic Control Assembly (ECA)

Gyro Assemblies (GA 1 and GA 2)

Rotation Controls (RC 1 and RC 2)

Translation Control (TC)

The RJ/EC provides the command signals to the auto coils of 16 SM or
12 CM RCS solenoid valves. None of the reaction jets can be turned on
via the auto coils except through the ground supplied by the RJ/EC.
The device receives logic and command inputs from Control Panel 1 and
the G&N System as well as from the ECA, RC 1, RC 2 and TC. Control
Panel 8 provides 28 VDC power to the RJ/EC.

The ECA provides command signals to the RJ/EC for control of the space-
craft rotation about each of its axes. The ECA receives logic and command

5-3

signals from Control Panel 1 and GA 1, GA 2, RC 1 and RC 2. Control Panel 7 provides AC and DC power to the ECA.

The Gyro Assemblies, GA 1 and GA 2, provide AC attitude error and/or rate signals to the ECA. These signals are proportional to rotations about the three spacecraft axes. The Gyro Assemblies receive AC and DC power from Control Panel 7.

The Rotation Controls provide logic and/or command signals to the G&N CMC, the RCS direct coils, the ECA and the RJ/EC as functions of the RC handle position, which is controlled by the astronaut. The Rotation Controls are provided with DC power from Control Panel 1 and AC power from the ECA.

The Translation Control provides logic and command signals to the G&N CMC, the SECS and the RJ/EC as functions of the handle position, which is controlled by the astronaut. The TC receives 28 VDC power from Control Panel 1.

5.3   SCS RJC SUBSYSTEM INTERFACES

Numerous alternate control paths are available for controlling the attitude and linear acceleration of the Apollo spacecraft. Each of these paths involves an interface with the SCS.

Control Panel 1 switches provide the capability to select the desired control mode. Power is supplied to the RJC subsystem via switches on Control Panels 1, 7 and 8 and circuit breakers on Control Panel 8.

The G&N system provides commands to the reaction jet auto coils via 16
lines from the CMC to the RJ/EC. Translation and rotation of the space-
craft can be controlled manually thru the interfaces of the TC, RC 1 and
RC 2 with the CMC.

The SCS RJC subsystem interfaces with both the auto coils and the direct
coils of the RCS solenoid valves. As mentioned previously, the RCS
engines can be fired via the auto coils only thru the RJ/EC. The direct
coils are used to fire the RCS engines with commands from the Rotation
Controls, Direct Ullage button, SECS or the SM Jettison Controller.
The RJ/EC prevents activation of the auto coils for the RCS thrusters
which are being controlled by the direct coils.

The SCS RJC subsystem also interfaces with the SECS. An abort can be
commanded manually via the TC interface with the SECS. The SECS also
provides the enable-disable command for the RJC subsystem via the RCS
latching relay.

The SCS RJC subsystem also interfaces with the SCS Thrust Vector Control
and Engine Ignition subsystems. These interfaces are described in detail
in following paragraphs.

5.4   G&C REACTION JET CONTROL

Figure 5.1 is a block diagram drawing of the G&C Reaction Jet Control
mechanization. This drawing shows the SCS and G&N devices involved in
controlling the state of the RCS auto and direct coils.

G&C ATTITUDE CONTROL
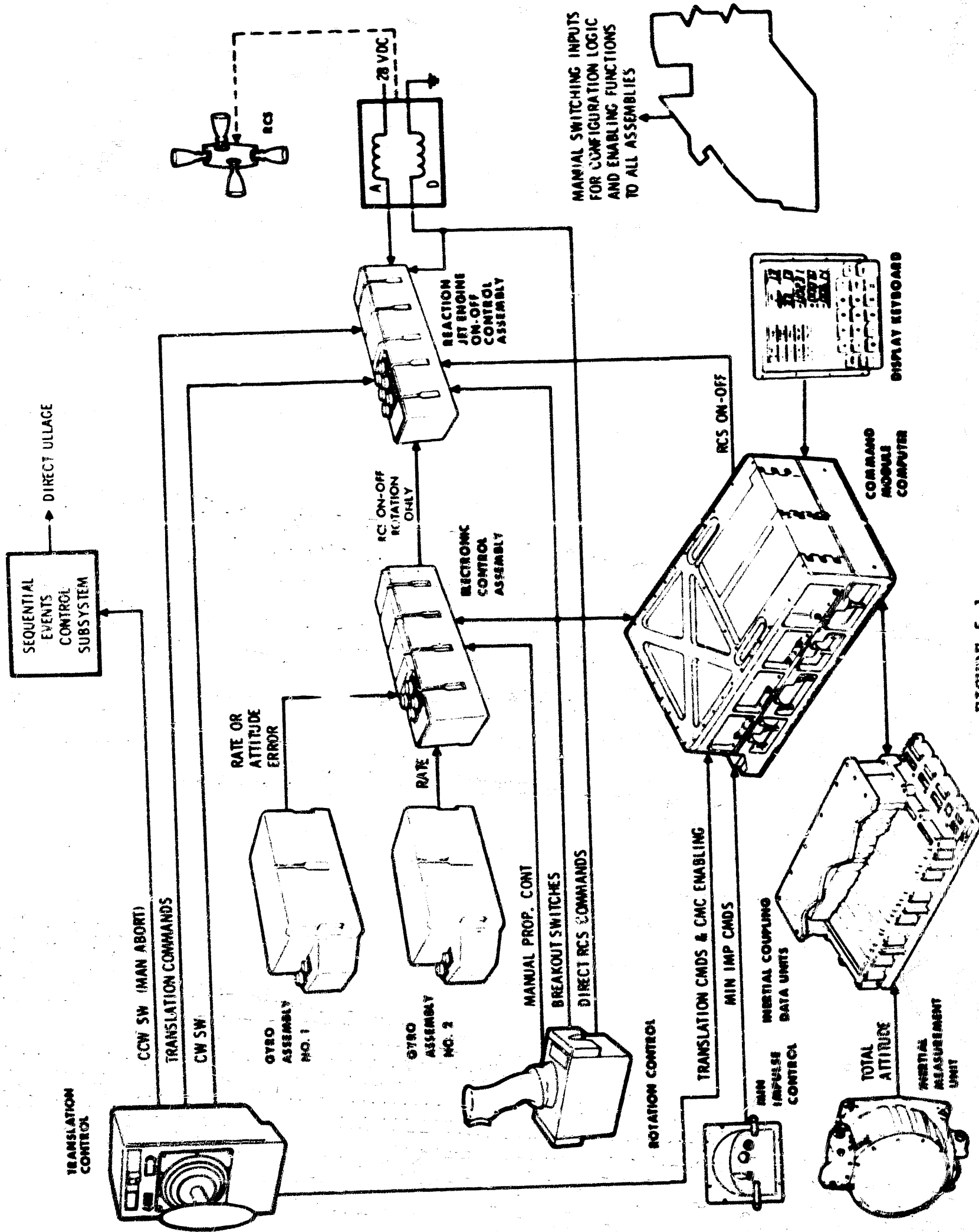
FIGURE 5.1

G&N Control

The G&N system provides RJC commands via discrete signals on the 16 lines from the CMC to the RJ/EC. These commands are used only when the G&N control mode is enabled via Control Panel 1 switches. The type of control mode is selected by the astronaut via one of the two Display Keyboards which provide commands to the CMC programmer. The CMC provides the "desired" attitude information to the CDUs. The CDUs generate error signals which are proportional to the difference between the desired and actual attitude and feed them to the CMC. The CMC provides commands to fire the proper reaction jets to maintain zero error signals via the 16 lines to the RJ/EC.

Manual translation commands can be supplied to the CMC via the TC. Manual rotation commands are fed to the CMC from RC 1, RC 2 and the G&N Minimum Impulse Control in the LEB Nav Station. G&N Control can be disabled by rotating the TC handle CW.

SCS Control

The SCS provides various alternate methods of reaction jet control. Attitude Hold is provided by mechanizing the GA 1 gyros to sense changes in attitude while GA 2 generates signals proportional to rotational rates about the S/C axes. These rate and attitude error signals are conditioned and summed in the ECA to provide command signals to the RJ/EC which controls firing of the reaction jets to maintain the spacecraft attitude within the desired deadband.

Rate Damping is provided by the same mechanization as described above, except that only rate signals are fed to the ECA from GA 1 or GA 2. This mechanization prevents the spacecraft rotational rates from exceeding selected values.

Proportional Rate Control is provided by the same mechanization as Rate Damping except that rate command signals from the Rotation Control(s) are summed in the ECA rate channels. This mechanization provides spacecraft rotational rates proportional to RC stick displacement.

Acceleration Command capability is provided by use of discrete command signals from the RC breakout switches to the RJ/EC solenoid drivers. This mechanization places the spacecraft in rotational free drift in the selected axis when the breakout switches are open and provides rotational acceleration (by keeping the proper jets turned on) when the breakout switches are closed.

Direct RCS Control via the connection between the Rotation Control Direct Switches and the RCS Direct Coils. This mechanization yields the same S/C control characteristics as Acceleration Command.

The five methods of spacecraft rotational control mentioned above are selectable by axis. That is, the pitch, yaw and roll rotations of the spacecraft are controlled independently by the mechanization selected via control panel switches.

Translational control of the spacecraft along any combination of the S/C

axes in either direction is possible simultaneously with rotational control, except when Direct RCS Control is being utilized. The TC provides the discrete command signals from its switches as a function of the TC handle position.

5.5   RJC RELATED CONTROL PANEL SWITCHES

The following is a brief discussion of the basic functions of the control panel switches as they affect the Reaction Jet Control subsystem. Block II Apollo utilizes the functional switching concept as opposed to mode switching. It should be apparent from the preceding paragraphs that a large number of mode switches would be required to provide all the redundant mechanizations available. Basically, when the mode switching concept is employed the amount of redundancy is necessarily decreased to prevent the mechanization from getting exceedingly complex. Functional switching allows a relatively large increase in redundant mechanizations with a minimal increase in the number of switches and mechanization complexity.

The control panel switches are identified by numbers such as 1S59. This number identifies switch number 59 on Control Panel 1. In general the switch numbers increase with changes in position from left to right and top to bottom on the control panel. Figures 5.2, 5.3, and 5.4 depict Control Panels 1, 7 and 8.

SC CONT (1S18)

The SC CONT switch places the SCS or CMC in control of the spacecraft.

The CMC position (up) places the S/C under CMC control if the Translation

CONTROL PANEL NO.1

FIGURE 3.2

CONTROL PANEL NO. 7

FIGURE 5.3

CONTROL PANEL NO. 8
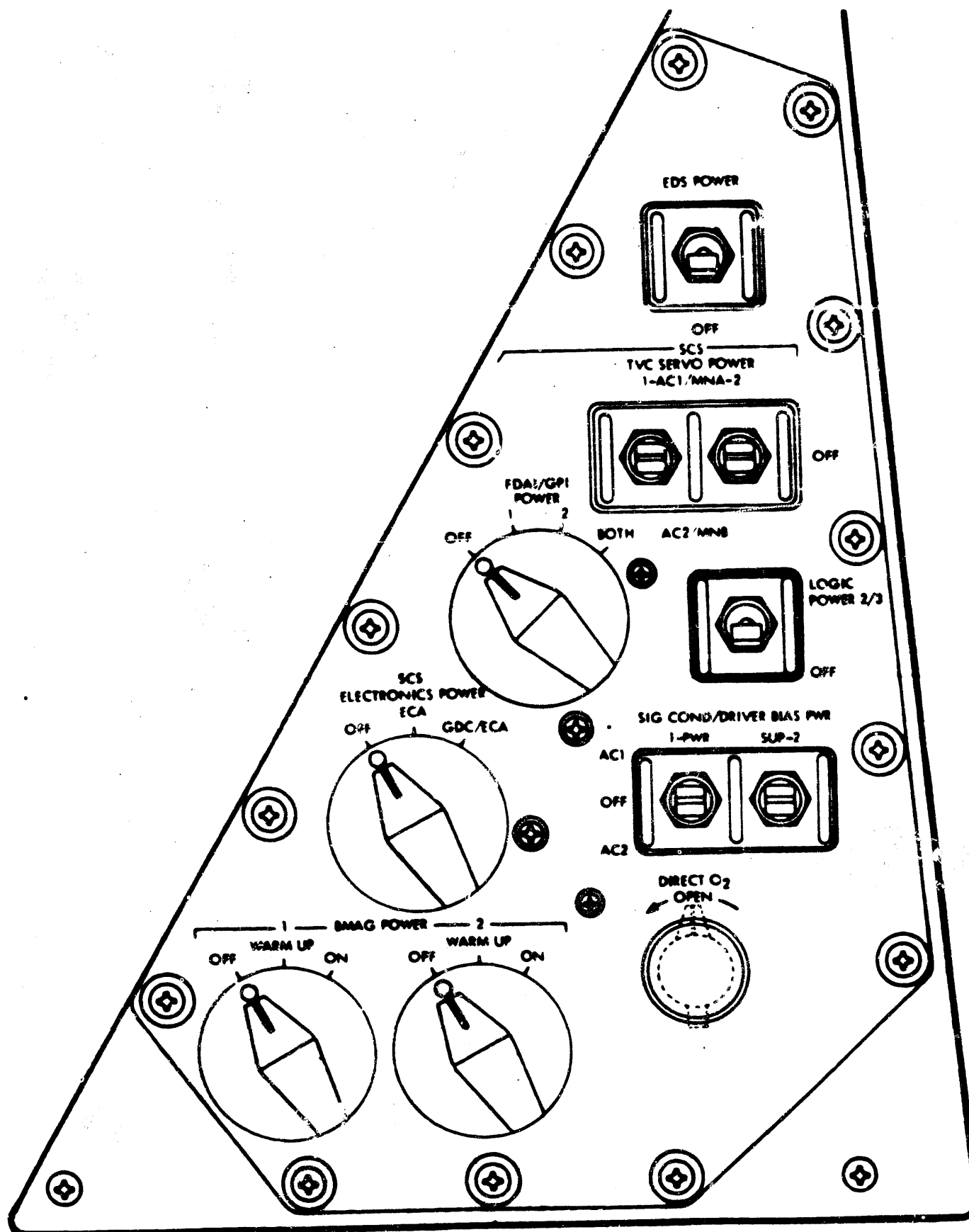
FIGURE 5.4

Control is not also turned CW. The type of computer control will be determined by the CMC Mode switch (1S19) and inputs to the computer for program selection. Also, a MANUAL ATTITUDE switch (1S7, 1S8, or 1S9) must not be in the ACCEL CMD position or that axis will not be under computer control regardless of the position of 1S18. Instead, it places that axis in the Acceleration Command Mode with outputs from the RC 1 and RC 2 breakout switches being utilized.

The SCS position (down) places the S/C under SCS control. Again, if a MANUAL ATTITUDE switch (1S7, 1S8, or 1S9) is in the ACCEL CMD position, that axis is placed in the Acceleration Command Mode and neither the SCS nor the CMC can control S/C rotation about the axis.

CMC MODE (1S19)

The CMC MODE switch selects the mode of operation for the CMC. This switch will engage S/C control via the CMC only if the SC CONT switch is in the CMC position, the CW switch in the Translation Control is not engaged, and the MANUAL ATTITUDE switches are not in the ACCEL CMD position. This switch has no inputs to the SCS hardware but is described to summarize basic CMC control capability.

The AUTO position (up) permits the CMC to control spacecraft attitude as a function of the computer program. The Display Keyboard inputs can be used to modify the programs, select the maneuver rates for attitude control, set up a Thrust Vector Control program, or any other capabilities programmed within the computer. The Display Keyboard inputs are the only normal inputs for spacecraft control.

The HOLD position (center) configures the CMC for an attitude hold configuration. It performs no automatic function except to maintain the attitude error at or below the programmed deadband with a rate deadband for drift rate control. This position does permit the breakout switch signals from the SCS Rotation Controls to generate maneuver commands in the computer which result in a manually commanded attitude change. The MANUAL ATTITUDE switches again cannot be in the ACCEL CMC position for this function to be operative.

The FREE position (down) prevents the computer from generating commands to maintain the spacecraft at a specific attitude or control an angular drift rate. The G&N Minimum Impulse Control in the lower equipment bay (with the MANUAL ATTITUDE switches not in the ACCEL CMD position) can generate commands to the computer that will result in a minimum impulse firing of the selected RCS thrusters. Closing the RC breakout switches results in continuous RCS thruster firing.

MANUAL ATTITUDE (1S7, 1S8, 1S9)

The MANUAL ATTITUDE switches provide enable inputs by axis during SCS control of the S/C to perform various control functions.

The ACCEL CMD (up) position enables the breakout switches of the Rotation Controls to the solenoid drivers and creates a free drift condition with no commanded input. It allows rotational acceleration by energizing the auto coils of the proper RCS thrusters.

The RATE CMD (center) position allows proportional rate control. The

magnitude of the commanded rate is proportional to the displacement of the Rotation Control. However, commands are not enabled unless the breakout switches are closed. This mechanization prevents inadvertent commands from being recognized by the control electronics.

The MIN IMP (down) position places the SCS in a free drift mode (no attitude hold or rate damping) with no commanded inputs. Minimum impulse commands are generated by closing the breakout switches in the Rotation Control. A nominal 15ms minimum impulse signal is fed to the RCS auto coils.

LIMIT CYCLE (1S10)

The LIMIT CYCLE switch in the up position enables SCS automatic pseudo rate feedback. Enable of the feedback electronics results in pulse modulation of the electrical signal used to turn the RCS engines on and off. This serves a useful function when the S/C is under SCS control for limit cycle action by generating minimum impulses which minimizes rates, resulting in RCS propellant conservation.

ATT DEADBAND (1S11)

The ATT DEADBAND switch controls the attitude deadband excursion to $\pm4.2^{\circ}$ in the MAX position or $\pm0.2^{\circ}$ in the MIN position when the SC CONT switch (1S16) is in the SCS position and the RATE switch (1S12) is in the LOW position. With the Rate switch in the high position the attitude deadband is $\pm8^{\circ}$ or $\pm4^{\circ}$ with 1S11 in the MAX and MIN positions, respectively.

RATE (1S12)

The RATE switch controls the maximum proportional rate command available from the Rotation Control with the S/C under SCS control. These proportional rates will be available in a particular axis only if the respective MANUAL ATTITUDE switch (1S7, 1S8, or 1S9) is in the RATE CMD position. The maximum rates obtainable in the HIGH position are 20°/sec in roll and 7°/sec. in pitch and yaw. The maximum rates in the LOW position are 0.7°/sec. in roll, pitch, and yaw. The HIGH and LOW positions also set the autopilot rate deadbands to $\pm 2^{\circ}$/sec and $\pm 0.2^{\circ}$/sec respectively.

BMAG MODE (1S20, 1S21, 1S22)

The BMAG MODE switches permit alternate selection of sensor outputs in GA 1 and GA 2 by axis.

- The RATE 2 (up) position generates a "rate damping only" mode, using rate from the BMAG in GA 2. The BMAG in GA 2 also supplies rate for display and to the GDC to compute the inertial attitude for display. The BMAG in GA 1 is caged in this switch position and therefore no automatic attitude hold capability is available from the SCS.

- The ATT 1/RATE 2 (center) position allows rate signals from the BMAG in GA 2 and attitude error signals from the BMAG in GA 1 to be used for control and display purposes.

- The RATE 1 (down) position cages the BMAG in GA 1 and inhibits outputs from the BMAG in GA 2. The BMAG in GA 1 supplies the rate signals for rate control, rate display, and to the GDC to calculate total attitude. With the BMAG in GA 1 caged, no automatic attitude hold capability is available in that axis.

DIRECT ULLAGE (1S24)

Linear acceleration along the +X S/C axis is obtained by depressing the direct ullage button. This function is made available as a backup method of forcing the SPS propellants to the rear of the tanks prior to SPS engine ignition. Depressing the button causes the rear facing SM reaction jets to fire by energizing their direct coils. The auto coils for the pitch and yaw reaction jets are disabled while the DIRECT ULLAGE button is depressed.

ENTRY (1S51)

The ENTRY switch is normally in the OFF position. The astronaut places the switch in the .05G position during reentry of the CM into the earth's atmosphere when the .05G sense lamp on the EMS display is illuminated. Placing the switch in the .05G position automatically affects the RJC mechanization as follows:

- The GA 1 BMAG's are caged

- The SCS Attitude Hold mode is disabled

- The Roll rate signal is coupled into the yaw channel

TRANS CONT PWR (1S66)

The TRANS CONT PWR switch provides an enable-disable capability for manual translation control of the spacecraft via the TC. Placing the TRANS CONT PWR switch in the up position provides 28VDC to the TC translation switches through the CM/SM transfer mechanization. The RCS TRNFR switch must be in the SM position to enable the TC translation switches.

ROT CONT PWR - NORMAL (1S64, 1S65)

The NORMAL ROT CONT PWR switches 1 and 2 control AC and DC power to the transducers and LOCK/ARM switches of RC 1 and RC 2 respectively. In the OFF (center) position, no power is available to the breakout switches or the transducers. In the AC/DC (up) position, both the breakout switches (through the RC LOCK/ARM switch) and the transducers are supplied power. In the AC (down) position, only the transducers are activated. The latter position allows the SPS engine gimbals to be controlled by the RC while maintaining attitude hold via the RJC subsystem. This capability is described in section 6.

ROT CONT PWR - DIRECT (1S13, 1S67)

The DIRECT ROT CONT PWR switches 1 and 2 control 28 VDC power to the Direct switches of RC 1 and RC 2 respectively. In the OFF (center) position, no power is available to the Direct switches. In the MNA/MNB (up) position the Direct switches are supplied MN A+B power. In the down position RC 1 Direct switches receive MN A power while RC 2 Direct switches receive MN B power.

BMAG POWER (7S6, 7S7)

Power to the Gyro Assemblies is controlled by the two BMAG POWER switches. Switch 1 controls MNA and Bus 1 power to GA 1, while switch 2 controls MNB and Bus 2 to GA 2. In the OFF position, the GA receives no power. In the WARM UP position DC power is supplied to the GA temperature control and indicating circuits. The BMAG POWER switches should be left in the WARM UP positions until their respective BMAG TEMP lamps (Control Panel 2 Caution and Warning Lamps) are extinguished. Operational power (115V 400 hz) and 28VDC are supplied to the GA when the BMAG POWER switch is in the ON position.

SCS ELECTRONICS POWER (7S5)

Power to the ECA is controlled by the SCS ELECTRONICS POWER switch. In the OFF position, the ECA receives no power. The ECA receives MNB, bus 1 and bus 2 power in the ECA and GDC/ECA positions.

SIG COND/DRIVER BIAS PWR (7S11, 7S12)

The RJ/EC contains two 4VDC power supplies which provide bias power to the

solenoid drivers. The function of the bias voltage is to prevent the drivers from inadvertently energizing the solenoid auto coils when high level noise signals are present on the input lines. However, the system requirements are met without the driver bias voltage. SIG COND/DRIVER BIAS PWR switch 1 provides power to one -4VDC power supply, switch 2 to the other. Each power supply provides driver bias power to eight RJC solenoid drivers. Bus 1 power drives the power supply in the AC 1 position, bus 2 in the AC 2 position.

AUTO RCS SELECT (8S17 thru 8S32)

All 16 SM or 12 CM RCS solenoid valve auto coils and the 16 RJ/EC drivers receive 28 VDC enabling power through the 16 AUTO RCS SELECT switches. These switches provide the capability of inhibiting individual reaction jet firing via the auto coils.

5.6   AUTO RCS LOGIC

The logic and the solenoid drivers which control the state (energized or deenergized) of the CM and SM RCS solenoid auto coils are contained in four identical modules of the RJ/EC. Figure 5.5 provides a diagram which shows the Auto RCS logic signal flow paths. This drawing describes the inputs to the modules.

Logic Inputs

As indicated in Figure 5.5, each module receives similar logic inputs. First of all, each module receives an enable/disable logic input which affects all four solenoid drivers in the module. Each module receives
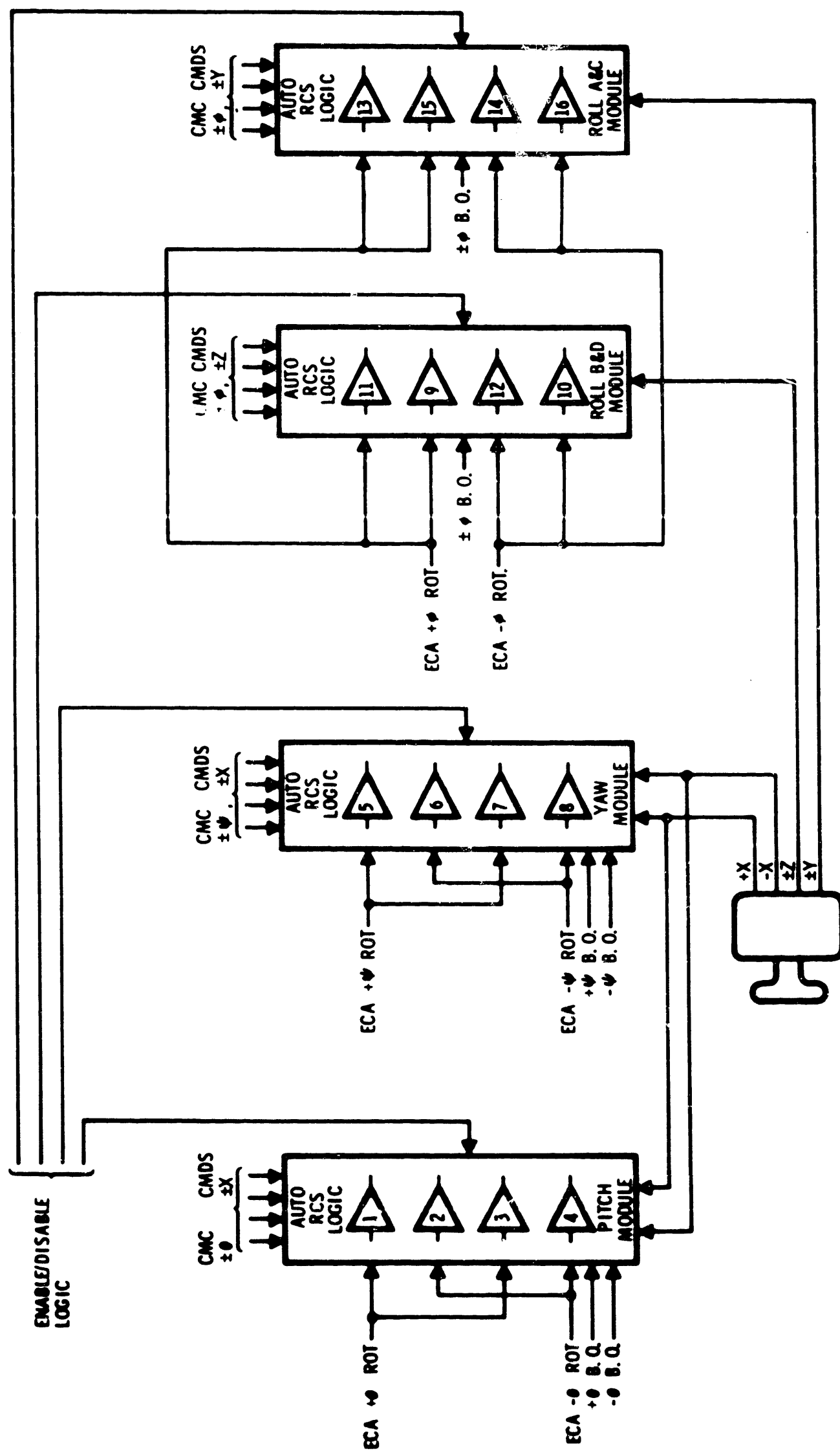
AUTO RCS LOGIC SIG FLOW

FIGURE 5.5

5-20

four discrete inputs from the CMC, one for each solenoid driver. The

ECA provides two discrete signals for rotation control, each input

feeding two drivers. Each Rotation Control provides two discrete

breakout switch signals to each module. Finally, each module receives

two inputs from the Translation Control linear acceleration switches,

each input feeding two solenoid drivers.

The mechanization of each module is identical except for the channel

disable signal. The logic mechanization for the pitch channel is

described in Figure 5.6.

Enable/Disable Logic

As seen in Figure 5.6, the solenoid driver output is connected to a CM

and a SM auto coil (unless the SM has been jettisoned, in which case the

deadface connection is broken). However, only one of the auto coils,

CM or SM, will receive 28VDC, depending upon the position of the RCS

TRNFR switch 2S44. The enable voltage for the auto coil and the solenoid

driver is provided by one of the AUTO RCS SELECT switches 8S17 through

8S32.

The enable/disable logic which controls the command source is provided by

OR gates 5, 6, 7, 8 and 9 and two inverters. The channel enable/disable

signal is provided by gate 7, a logic level 1 representing a disable sig-

nal for all four solenoid drivers. We see that the channel is disabled

by satisfying any one of three possible conditions:

　　1) If a DIRECT ROT CONT PWR switch (1S13 or 1S67) is in the up

AUTO RCS LOGIC
TYPICAL CHANNEL (PITCH)

FIGURE 5.6
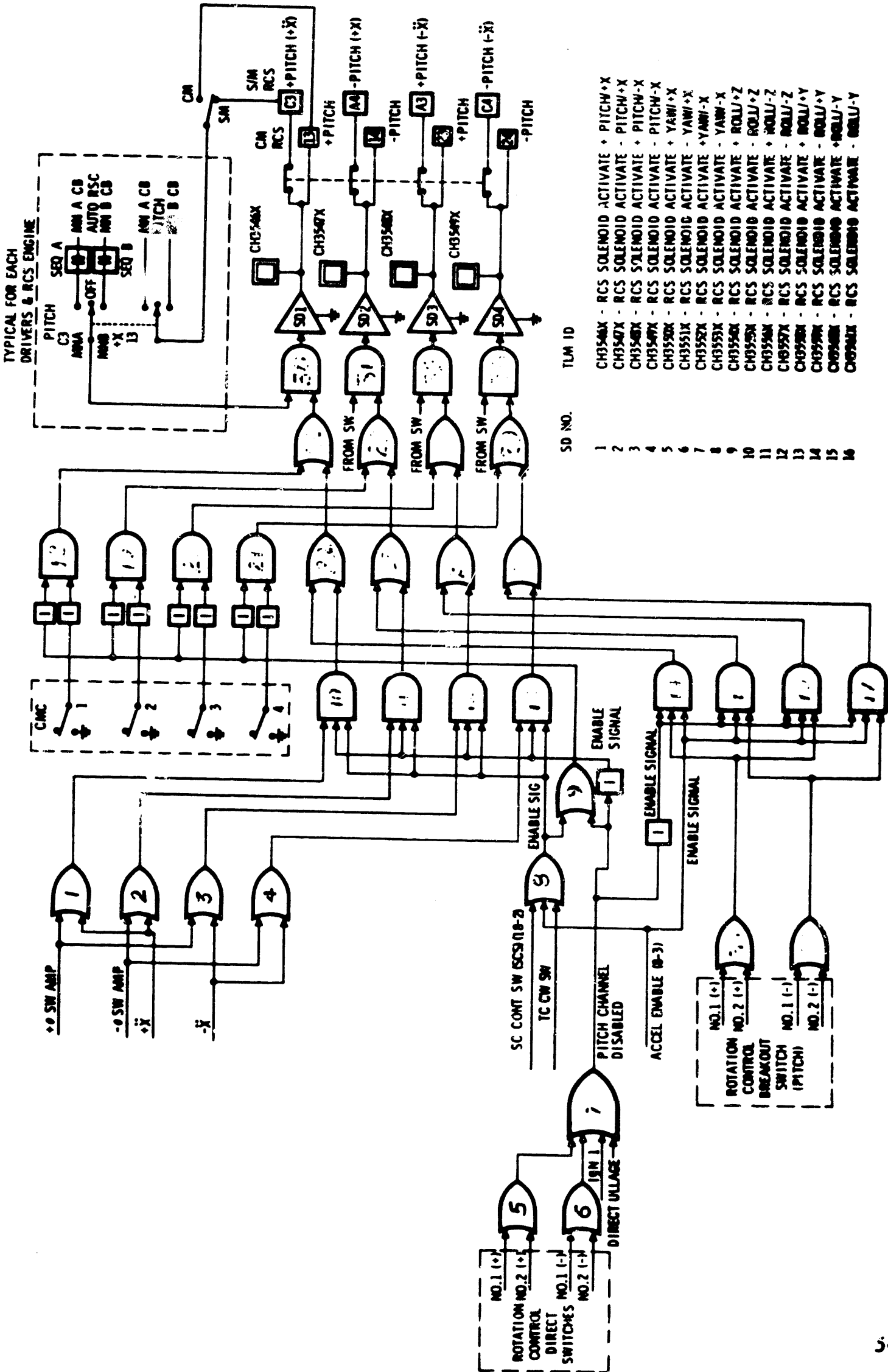
or down position and the corresponding RC + or - direct switches are closed, the corresponding gate 5 or 6 will provide a logic 1 to gate 7.

2) Presence of the IGN 1 signal is sufficient to disable the channel. The IGN 1 signal is provided by the SPS Engine Ignition logic. This signal is present from one second after the SPS engine is commanded ON until one second after it is commanded OFF.

3) Presence of the DIRECT ULLAGE logic signal is sufficient to disable the channel. This signal is a logic 1 whenever the DIRECT ULLAGE button (1S24) is depressed.

The above description is valid for the pitch and yaw channels only. The Roll Channel is identical except that the IGN 1 and DIRECT ULLAGE inputs are removed. This mechanization is required because in a TVC mode the pitch and yaw spacecraft attitude is controlled by gimballing the SPS engine while roll attitude is controlled by the RCS engines.

Gate 8 provides the enable/disable signal for SCS rotational and translational control. A logic 1 enable signal is present when any one of the following conditions is satisfied:

1) the SC CONT switch 1S18 in the SCS position,

2) the Translation Control handle in the CW position, or

3) the MANUAL ATTITUDE switch (1S7, 1S8 or 1S9 corresponding to the channel) in the ACCEL CMD position.

Gate 9 provides the enable/disable signal for control by the CMC. A logic

0 enable signal is present only under one condition; Gates 7 and 8 must provide logic 0 signals.

The Acceleration Mode enable/disable signal is supplied directly from the ACCEL CMD position of the corresponding MANUAL ATTITUDE switch 1S7, 1S8 or 1S9.

SCS Control Mode

The SCS Control Mode is engaged by providing logic 1 signals on the lower two inputs of AND gates 10, 11, 12 and 13. If this condition is satisfied gate 9 will have a logic 1 output resulting in gates 18, 19, 20 and 21 having logic 0 outputs. Also, to be in an SCS Control Mode, the MANUAL ATTITUDE switches 1S7, 1S8 and 1S9 must <u>not</u> be in the ACCEL CMD position. This condition (not ACCEL CMD) disables gates 14, 15, 16 and 17 and they have logic 0 outputs. Now the four solenoid drivers are controlled by, and only by, the ECA and TC outputs. For example, the only way SD1 can be turned on to provide a ground to auto coil C3 or 13 is with a $+\theta$ command from the ECA or a $+\ddot{X}$ command from the TC. For either of these cases gate 1 will be enabled (logic 1 on the output), enabling gates 10, 22, 26 and 30 which turns on SD1. The $+\ddot{X}$ command is a 28 VDC signal provided by the TC $+\ddot{X}$ switch when the TC handle is pushed forward. The $+\theta$ command is a $+15$ VDC signal supplied by the ECA in one of several ways which are described in sections 5.7 and 5.8.

CMC Control Mode

The CMC Control Mode is engaged by providing a logic 0 on the gate 9

output. This provides a logic 1 on the top inputs of AND gates 18, 19, 20 and 21. AND gates 22 through 25 are disabled. Now the four solenoid drivers are controlled by, and only by, the four CMC outputs. SD1 is turned on by providing a ground (logic 0) on CMC output line 1. The inverter provides a logic 1 to the lower input of gate 18 which enables gates 18, 26 and 30.

The type of CMC Control Mode is dictated by the CMC MODE switch 1S19. The CMC recognizes inputs from RC1, RC2 and the TC for each position of 1S19. In the AUTO and HOLD positions commands from the RC breakout switches will result in the CMC generating commands to maintain the S/C rotational rate at a specific level which is selectable by the astronaut through the DSKY. TC commands are also recognized, but the CMC is mechanized for rotation priority. In the FREE position RC commands cause the CMC to generate acceleration commands. The CMC also accepts inputs from the G&N Minimum Impulse Control with the CMC MODE switch in the FREE position.

Acceleration Command Mode

The Acceleration Command mode is engaged in an axis or combination of axes by placing the corresponding MANUAL ATTITUDE switches 1S7, 1S8 and 1S9 in the ACCEL CMD position. Acceleration Command mode selection overrides the SCS and CMC Control modes. Selection of the Acceleration Command mode enables AND gates 10 through 17 and disables 18 through 21. OR gate 34 allows RC1 or RC2 breakout switches to provide logic 1 inputs to AND gates 14 and 16 which turns on gates 22, 24, 26, 28, 30 and 32.

Thus SD1 and SD3 are turned on providing a ground for the + Pitch RCS auto coils. Similarily, the - Pitch auto coils are turned on via gates 35, 15, 17, 27, 29, 31 and 33 and SD2 and SD4.

Translation control is available in the Acceleration Command mode via gates 1, 2, 3, 4, 10, 11, 12, 13 and 22 through 33. Minimum Impulse outputs from the ECA are not disabled when the Acceleration Command mode is selected. However, minimum impulses are generated by the same RC breakout switches that generate the acceleration commands. Hence, the only effect of this parallel mechanization is that acceleration commands of less than 15 millisecond duration are possible only if the ECA power is turned off (and the RC breakout switches are closed less than 15 milliseconds).

SCS ATTITUDE CONTROL

In this section we will consider the SCS capabilities for rotational control of the spacecraft via the RCS reaction jets. Figure 5.7 provides a block diagram description of the SCS Attitude Control mechanization. This drawing shows the basic functions performed in the ECA and the inter-connections with other devices. The ECA outputs, along with command signals from the TC, CMC, RC1 and RC2, are fed to the Auto RCS logic in the RJ/EC. The four basic SCS Attitude Control modes are described in the following paragraphs.

Automatic Control

The SCS automatic attitude control capabilities consist of rate damping
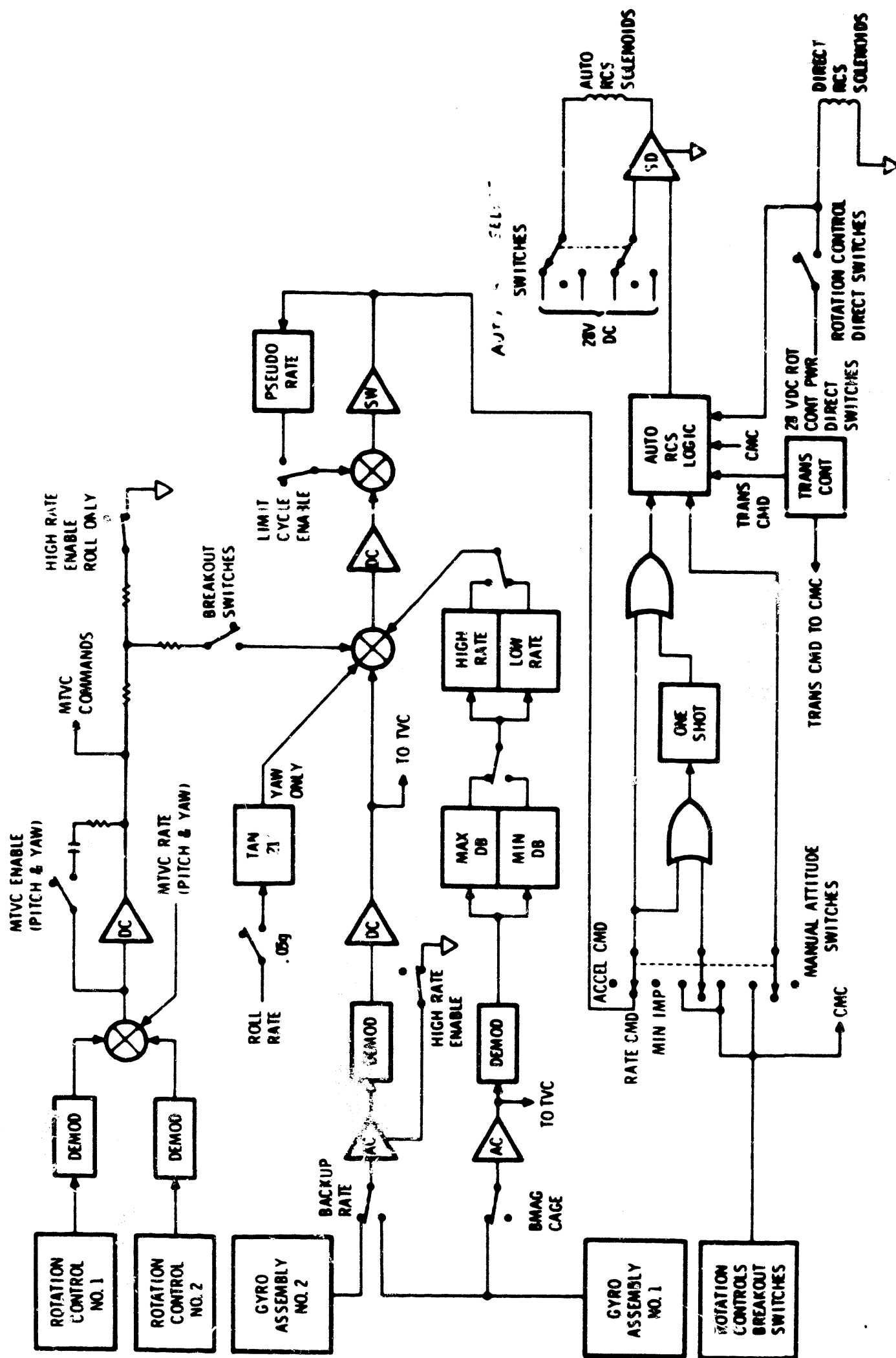
SCS ATTITUDE CONTROL
FUNCTIONAL OPERATION

FIGURE 5.7

5-27

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 spacecraft will maintain a fixed

inertial reference.

Rate Damping Mode

The control signals in the rate damping mode originate in the Gyro

Assemblies. These AC rate signals (amplitude proportional to the rotational

rate about the respective axis and phase dependent upon the direction of

rotation) are amplified, demodulated, and then filtered and amplified again

by the DC amplifier whose output is fed to the summing point of another DC

(summing) amplifier. As indicated in Figure 5.7, this is one of four signal

flow paths into the summing point at the center of the drawing. When the

rate damping mode is mechanized, this is the only signal to the summing

point (with one exception). During the entry phase of the mission (after

.05G), the DC roll rate signal is coupled into the yaw channel to cancel

the so-called "false yaw rate" sensed by the yaw gyro as a result of the

spacecraft roll about the stability axis (rather than the S/C X axis).

Except for this additional input in the yaw axis, the mechanization of the

pitch, yaw and roll rate channels are identical.

The output of the DC summing amplifier is fed to the switching amplifier.

The switching amplifier has two outputs, one for positive rotation commands

and the other for negative. A deadband circuit is mechanized at the

switching amplifier input. In other words, a certain voltage level

must be reached at the switching amplifier input before a rotation

command is generated at the output. When the input exceeds the deadband

level, a command output is generated on one of the output lines, depend-

ing upon the polarity of the input signal.

The magnitude of the rate deadband depends upon the input voltage level

required to turn on the switching amplifier and the scale factor at the

switching amplifier input. Two values of rate deadband are available

as a function of the Control Panel 1 RATE switch. The deadband is $\pm 0.2^{\circ}$/

sec. for the low rate condition. When high rate is selected the AC

amplifier gain is decreased by a factor of 10, increasing the deadband

to $\pm 2.0^{\circ}$/sec.

The switching amplifier output is fed to the Auto RCS logic directly

via the Minimum Impulse circuit. (Notice that the MANUAL ATTITUDE switch(es)

must be in the RATE CMD position to supply these signals to the RJ/EC logic).

The Minimum Impulse (one-shot) circuit ensures that the command signal is

of sufficient duration to "pull-in" both the fuel and oxidizer RCS solenoid

valves.

Attitude Hold Mode

The control signals in the attitude hold mode originate in both GA 1 and

GA 2. GA 2 provides AC rate signals to the rate channels described in

5.7.1.1 GA 1 provides AC attitude error signals to the attitude channels

(signal flow path below the rate channel in Figure 5.7). The mechanizat-
ions of the attitude channels are identical for each axis. The AC
attitude error signal is amplified, demodulated and sent to the attitude
deadband circuit. The deadband magnitude is selected by the Control
Panel 1 ATT DEADBAND switch. This particular circuit provides 0° and
$\pm 4^o$ deadband for the MIN and MAX positions respectively. The attitude
deadband circuit output is scaled by the DC summing Amplifier input
resistors from the attitude channel (represented by the HIGH RATE and
LOW RATE blocks in Figure 5.7). In the LOW rate, MIN deadband configuration,
the scale factor at the DC summing amplifier output (Switching Amplifier
input) is the same for the attitude error signal as for the rate signal.
In other words, the attitude deadband for the LOW rate, MIN deadband
configuration is 0.2^o, switching to high rate decreases the attitude
channel gain by a factor of 20. Hence the attitude error deadband is
increased to 4.0^o. The table below summarizes the rate and attitude
error deadbands as a function of the RATE and ATT DEADBAND switch positions.

SWITCH POSITION		RATE DEADBAND	ATT ERROR DEADBAND
RATE	ATT DEADBAND		
LOW	MIN	$\pm 0.2^o$/Sec.	$\pm 0.2^o$
LOW	MAX	$\pm 0.2^o$/Sec.	$\pm 4.2^o$
HIGH	MIN	$\pm 2.0^o$/Sec.	$\pm 4.0^o$
HIGH	MAX	$\pm 2.0^o$/Sec.	$\pm 8.0^o$

The switching amplifier circuit contains a "Pseudo-rate" feedback circuit which is enabled or disabled as a function of the Control Panel 1 LIMIT CYCLE switch position. The pseudo-rate circuit provides the capability of maintaining low spacecraft rates, while holding the attitude within the deadband limits (limit cycling). The pseudo-rate function is normally useful only to establish low limit cycle rates in an attitude hold mode. However, the pseudo-rate function is enabled, regardless of operating mode, by placing the LIMIT CYCLE switch in the up position. The pseudo-rate circuit is merely a resistor-capacitor integrator which provides negative feedback around the switching amplifier to pulse modulate its output. With pseudo-rate enabled (LIMIT CYCLE switch "ON"), the commands will have the following characterics relative to the switching amplifier input level:

- Input within deadband level ($\pm$) - No RCS Commands.

- Input just outside deadband level ($\pm$) - The RCS Commands are short duration pulses (Minimum Impulse Commands due to the one-shot circuit) with a low repetition rate.

- Increasing input (magnitude) - The pulse width and repetition rate increase until the on-time is equal to the off-time. After this point the pulse width (on-time) continues to increase while the repetition rate decreases.

- Large inputs ($\pm$) - An RCS Command is continuously present. As the input decreases, the operations described above are repeated in the reverse order.

BMAG Cage Logic

The rate and attitude channels of the ECA receive signals from GA1 and GA2 for both the rate damping and attitude hold modes (and also for the manual proportional rate control mode).

The SCS contains two gyro assemblies (GA1 and GA2). Each gyro assembly contains three sensing elements (Body Mounted Attitude Gyros or BMAGs) and the necessary gyro electronics to provide output signals proportional to rate or to attitude displacement in each of the three S/C body axes. The two gyro assemblies are identical. However, GA2 is capable of providing rate information only, while GA1 can be uncaged through external switching and thus provide attitude error (angular displacement) information.

The gyro assemblies thus provide angular displacement and rate information to all three SCS subsystems. Each caged BMAG has three electrically isolated rate outputs (400 hz control signals to the ECA, 400 hz display signals to the EDA, and a DC rate output to the GDC). The BMAGs in GA1, when uncaged, have two electrically isolated angular displacement outputs (400 hz control and display signals). The mode of operation of GA1 and the use of the BMAG outputs are functions of several conditions.

Figure 5.8 shows the functional mechanization of a single channel of the gyro assemblies and the uncage logic for that channel. The BMAG outputs are determined by the BMAG Mode switch. If the switch is in the center position (ATT 1 RATE 2), BMAG 2 supplies 400 hz rate signals to the ECA. If the "uncage logic" is satisfied, relay K1 will be energized and GA1 will supply 400 hz attitude error signals to the ECA.

BMAG 1&2 CAGE AND OUTPUT LOGIC
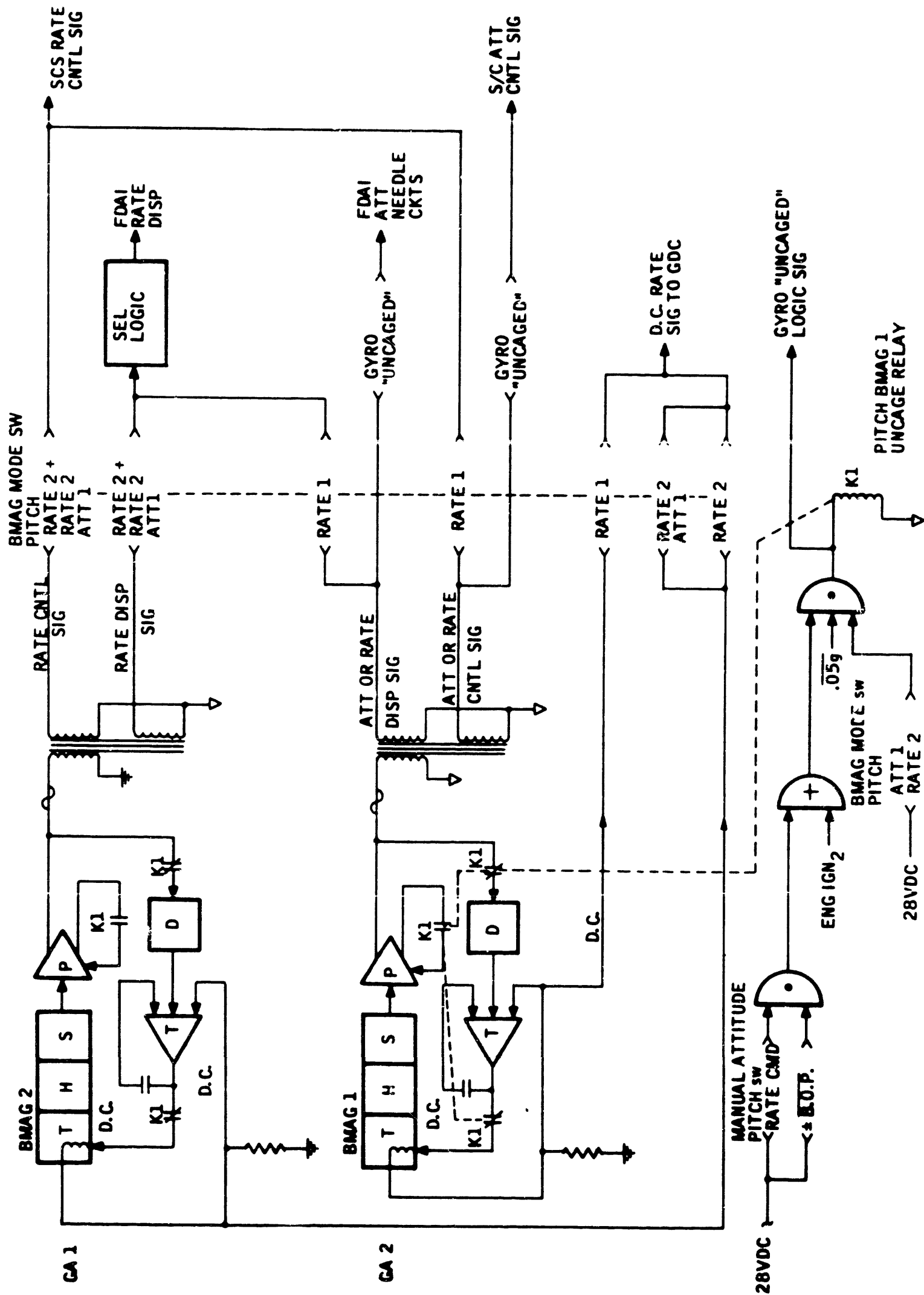
As indicated in Figure 5.8 the conditions necessary to uncage the gyro are given by:

$$(\text{ENTRY} (.05G) - \text{OFF}) - (\text{BMAG MODE} - \text{ATT 1 RATE 2})$$
$$\cdot \left\{ \left[(\text{MANUAL ATTITUDE} - \text{RATE CMD}) \cdot (^{\pm}\text{BO}) \right] + \text{ENG IGN 2} \right\}.$$

If these conditions are not satisfied, the BMAG is caged to prevent large outputs signals from being developed due to angular displacement of the spacecraft about the respective axis.

As indicated by the uncage logic, all BMAGs are caged during the atmospheric entry phase (by placing the ENTRY switch in the .05G position). There is no requirement for attitude hold during entry, thus both gyros in each axis are available for rate information.

5.7.2   Manual Control

The manual attitude control capabilities consist of Direct, Acceleration Command, Minimum Impulse and Proportional Rate Control. The commands listed are initiated by manual inputs using RC1 or RC2. With the exception of Direct, the RC functions command rotation through the RCS auto coils. The manual control loops are illustrated in the block diagram of Figure 5.7.

Proportional Rate Control

The proportional rate capability is available by-axis when 1) the SC CONT switch is in the SCS position or 2) the TC CW switch is engaged by rotating the control handle to the CW position, and the MANUAL ATTITUDE switch(es) is (are) in the RATE CMD position. However, commands are not initiated until RC breakout (BO) switch actuation. Prior to BO switch actuation, the

S/C is under SCS Automatic Control (rate damping or attitude hold) and reverts back to this control when the manual input is removed. The proportional rate control mechanization provides the ability to command a S/C rate directly proportional to the magnitude of RC stick deflection. This capability is obtained by summing the command signal (from the RC transducer) and the S/C actual rate signal (from the caged BMAG). As indicated in Figure 5.7, the transducer outputs from RC1 and RC2 are demodulated, amplified and (if the breakout switch has been actuated) fed to the DC Summing Amplifier. Thus, when the stick is deflected and the error signal exceeds the deadband, the proper RCS auto coils are energized turning on the reaction jets. The jets stay on until the resultant S/C rate is such that the rate signal reduces the switching amp input signal to less than the deadband level. The S/C rate will remain constant until the RC input is changed.

The rate commanded by a constant RC stick deflection is a function of the ratio of the command and rate control loop gains. The ratio has two possible values which are controlled by the RATE switch. The nominal rate commanded at maximum stick deflection (soft stop) for both RATE switch positions are shown in the table below.

| RATE Switch | Max. Prop Rate Command (by-axis) | |
Position	Pitch & Yaw	Roll
LOW	0.7 °/Sec.	0.7 °/Sec.
HIGH	7.0 °/Sec.	20 °/Sec.

The proportional rate control capability would not normally be used to provide a constant rate over a long or even a relatively short period of time. Not only would it cause an unnecessary strain on the astronaut, but excessive propellants would be consumed if the stick position is not held constant.

The LIMIT CYCLE switch should be in the OFF position when proportional rate control is being utilized. Performing a proportional rate maneuver with pseudo-rate enabled (LIMIT CYCLE switch ON) requires more RCS propellants than the same maneuver without the pseudo-rate feedback.

Minimum Impulse Control

The minimum impulse control capability is available by-axis when 1) the S/C CONT switch is in the SCS position or 2) the TC CW switch is engaged, and the MANUAL ATTITUDE switch is in the MIN IMP position. A command is initiated by actuating a RC BO switch. The S/C is in free drift in the axis which is under minimum impulse control.

The minimum impulse control mechanization provides the capability of making small changes in the S/C rate. As indicated in Figure 5.7, this capability is provided by feeding the BO switch command to a one-shot circuit. The one-shot provides a command to the RCS auto coils (through the AUTO RCS logic) for a specified length of time (normally 15 milliseconds) each time a BQ switch is closed. Additional minimum impulse commands are obtained by repeated opening and closing of the RC BO switch.

Acceleration Command Control

The acceleration command control capability is available by-axis when the appropriate MANUAL ATTITUDE switch is placed in the ACCEL CMD position. A command is initiated by actuating a RC BO switch. The S/C is in free drift in the axis which is under acceleration command control when no command is present.

The acceleration command mechanization provides continuous firing of the proper RCS jets while a RC command signal is present. As indicated in Figure 5.7, the command signals are sent directly from the RC BO switches to the Auto RCS Logic. The SC CONT and TC CW switches have no function in enabling or disabling the acceleration command control capability.

Direct Rotation Control

The direct rotation control capability is available as backup to every other control capability. A command is initiated by displacing the RC stick to a hard stop. As indicated in Figure 5.7, this action actuates a direct switch which supplies 28VDC to the appropriate RCS direct coils. 28VDC is also routed from the direct switches to the Auto RCS logic to inhibit all auto coil commands in the axis under direct control.

Prior to the direct switch closure, but after BO switch closure, the corresponding spacecraft axis is under the manual control indicated by the MANUAL ATTITUDE switch for that axis. This is true for the MIN IMP and RATE CMD positions only if the S/C is under SCS control (SC CONT switch to the SCS position or TC CW). When under CMC control, the RC BO switches provide signals to the CMC.

5.8 TRANSLATION CONTROL

The translation control capabilities for the Apollo spacecraft are:

- Translation Control (TC) Commands
- Direct Ullage (DU)
- Separation Ullage SIVB (SU)
- SM/CM Separation (SM Jet Cont)

The three latter capabilities are described in Section 5.10.3.

The TC commands are made available by placing the TRANS CONT PWR switch in the up position. TC commands are recognized by the Auto RCS logic during all of the attitude control modes listed in Section 5.7 (unless a channel disable signal is present).

The TC commands can be initiated simultaneously in the three axes. 28VDC discrete signals are fed from the TC switches to the auto RCS logic. The commands have the same priority as the rotation commands from the ECA and RC. Thus, it is possible to fire opposing SM jets when specific TC and rotation commands are present. Translation control is not available after CM/SM separation.

TC commands are also recognized while the S/C control is provided by the G&N system. (When under G&N control, translation commands are shared with rotation commands, the latter having higher priority. Therefore is is possible to have a single jet firing.)

5.9 MECHANIZATION DETAILS

In this section we will consider the mechanization of the ECA in some

detail. This is simply a more detailed description of the functional
mechanization, signal flow paths and circuits that were discussed in
Section 5.7 with reference to Figure 5.7.

5.9.1   Pitch Channel

Figure 5.9 provides a detailed description of the functional mechanization
of the ECA pitch channel. As described previously, the mechanization
consists of three bridge legs which are summed into an analog to digital
converter (Switching amp, Pseudo-rate and Minimum Impulse Circuits).

Rate Leg

Inputs to the rate leg of the ECA pitch channel are 400 hz signals from
the pitch BMAGs in GA1 and GA2. The signals are grounded or fed to an AC
buffer amplifier through solid state switches 3QS28 and 3QS29. When the
BMAG MODE switch is in the ATT 1 RATE 2 or RATE 2 position, the BMAG 2
signal is fed to the AC amplifier. If the BMAG MODE switch is in the
RATE 1 position, the BMAG 1 signal is fed to the AC amplifier. 3QS25 allows
the AC amplifier gain to be changed by a factor of 10. The amplifier has
minimum gain when the RATE switch is in the HIGH position (or when the
IGN2 signal is present, for use in Thrust Vector Control). The amplifier
output is fed to two Demod/switches, one providing demodulation and the
other an open switch. The Demod/switches are controlled by the same
logic as the previously mentioned solid state switches. This mechanization
ensures that the demod is referenced to the same AC bus as the input
signal. The demodulated signal is amplified and filtered by the DC
amplifier and fed to the summing amplifier (and also to the TVC circuitry).

SCS ATTITUDE CONTROL
PITCH CHANNEL

FIGURE 5.9 .

Rate Command Leg

Inputs to the rate command leg of the ECA pitch channel are 400 hz signals from the pitch RLVDTs in RC1 and RC2. Each input is fed to a separate full wave demodulator and passive ripple filter. The demodulator filter output(s) are fed to the DC amplifier. For attitude control, the amplifier is always mechanized as a proportional amplifier. The amplifier output is fed to the solid state switch 3QS32 (and to the TVSA for thrust vector control). Switch 3QS32 passes the signal to the summing amplifier only when a RC pitch BO switch is closed.

Attitude Error Leg

The attitude error leg of the ECA pitch channel is fed by 400 hz signals from the pitch BMAG in GA1. Solid state switch 3QS21 passes the signal to the AC buffer amplifier only when the BMAG is uncaged. The amplifier output is fed to a demod (and to a TVC circuit). The demodulated signal is amplified and filtered by the DC amplifier. The DC amplifier output is fed to the deadband circuit (and to a TVC circuit). The signal deadband is controlled by the ATT DEADBAND switch. From the deadband circuit the signal is fed to the summing amplifier through one of two resistive networks, dependent upon the RATE switch position. The resistive network allows the summing amplifier gain to be changed by a factor of 20 for attitude error signals. Maximum gain is provided with the RATE switch in the LOW position.

Analog to Digital Converter

The pitch channel bridge signals are fed to the DC summing amplifier whose

output feeds the pitch channel analog to digital converter. The converter generates pulse commands as functions of the input signal magnitude and the position of the LIMIT CYCLE switch. The converter consists of the switching amplifier, pseudo rate circuit and minimum impulse generator. The switching amplifier output pulses are fed to the auto RCS logic directly and through the minimum impulse generator when the MANUAL ATTITUDE switch is in the RATE CMD position. When the MANUAL ATTITUDE switch is in the ACCEL CMD or MIN IMP positions, the switching amplifier output is grounded while the minimum impulse circuit accepts signals from the RC1 and RC2 pitch breakout switches and supplies short duration pulses to the auto RCS logic. These circuits are described in detail in paragraph 5.9.5.

5.9.2   <u>Yaw Channel</u>

The mechanization of the yaw channel is identical to that of the pitch channel with one exception. The DC summing amplifier receives an additional input from the roll channel (roll to yaw coupling), when the ENTRY switch is in the .05G position.

The rate command leg of the yaw (and pitch) channels also receive rate signals from GA1 or GA2 for use during Manual Thrust Vector Control. These signals flow paths are always disabled unless the SPS engine is firing (in which case pitch and yaw RJC is disabled).

5.9.3   <u>Roll Channel</u>

Figure 5.10 provides a detailed description of the functional mechanization of the ECA roll channel. The differences between the roll and pitch
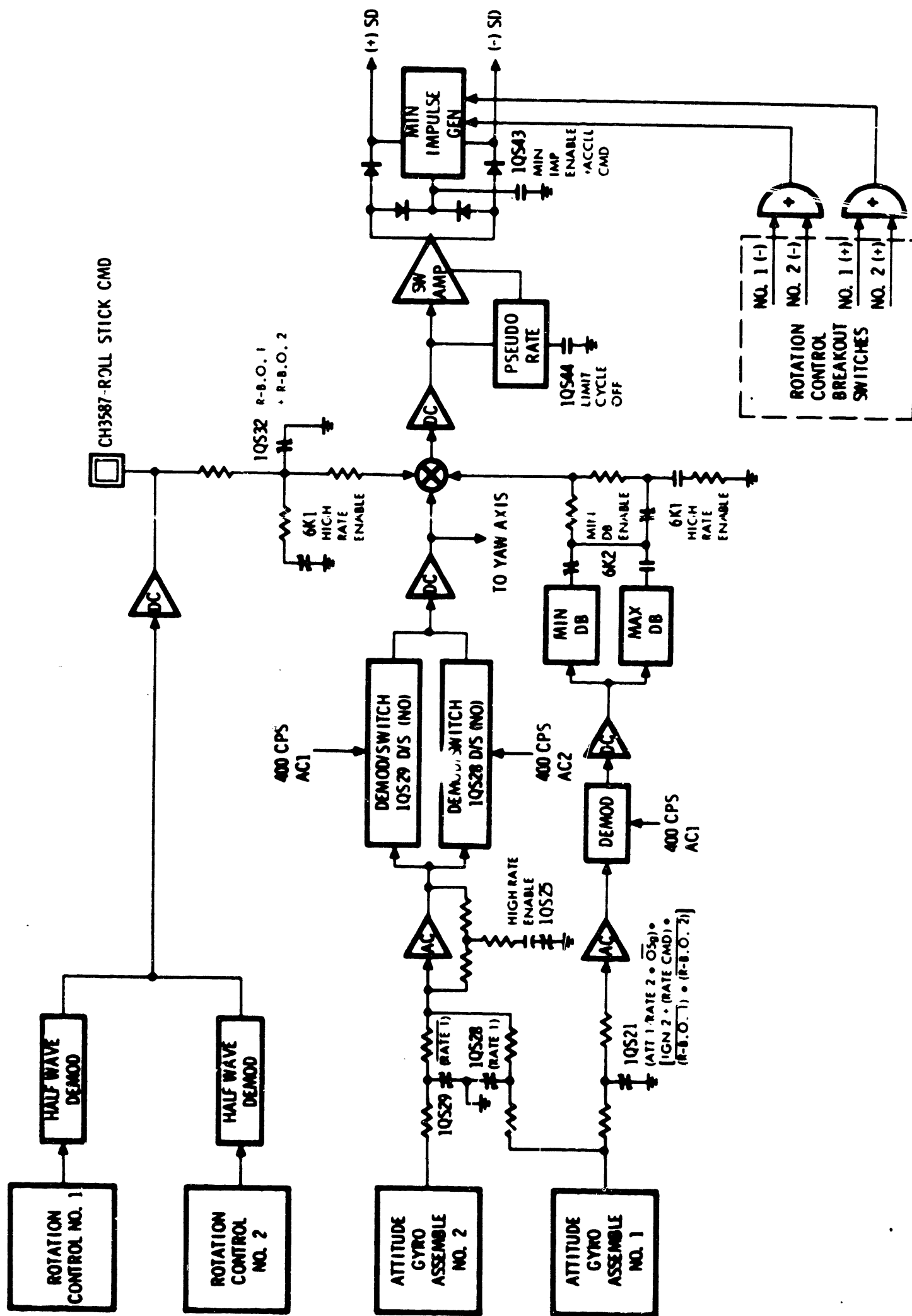
SCS ATTITUDE CONTROL
ROLL CHANNEL

FIGURE 5.10

5-43

channels are described below.

The roll channel does not provide any outputs for thrust vector control since the SPS engine gimbal can be deflected only in the pitch and yaw axes. The rate command signals are demodulated by half-wave (instead of full wave) demodulators and the filtering is accomplished by the DC amplifier circuit.

The rate command DC amplifier output is fed through a voltage divider which provides two values of DC summing amplifier gain for rate command signals as a function of the RATE switch position. The gain is changed by a factor of approximately 3. Maximum gain is provided when the RATE switch is in the HIGH position. Thus the RC command authority increases by a factor of 30 when the RATE switch is placed in the HIGH position. The only other difference between the roll and pitch channels is that the roll DC rate signal is fed to the yaw channel when the ENTRY switch is in the .05G position.

5.9.4   ECA Bridge Circuits

This section describes the mechanization of the different circuits which are used in the rate, rate command and attitude error legs of the bridges in the pitch, yaw and roll ECA channels.

Solid State Switch

Figure 5.11 shows the mechanization of a typical solid state switch. The solid state switch consists of two resistors and a PNP chopper transistor. The switch is turned ON (allowing the input voltage to provide a current
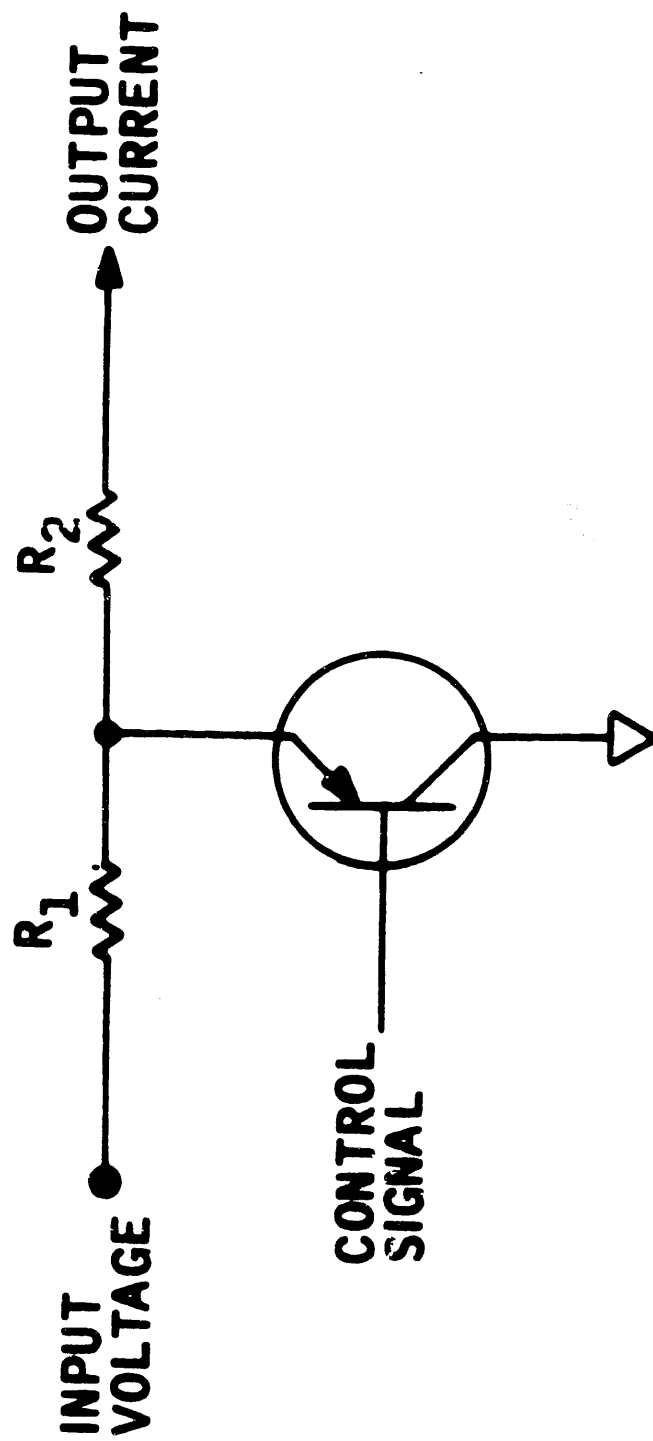
SOLID STATE SWITCH

FIGURE 5.11

through R1 and R2, normally to the summing point of an amplifier) by making the control signal positive, which turns the transistor OFF. The switch is turned OFF by making the control signal slightly negative, which turns the transistor ON (shorting the current through R1 to ground).

Demodulator Circuits

Three types of demodulators are used in the ECA bridge circuits.

Figure 5.12 shows the differential full wave demod which is used in the rate and attitude error legs. The demod consists of four resistors and 2 NPN transistors; it feeds the two inputs (inverting and non-inverting) of a DC operational amplifier. Assume that the input voltage is in phase with the reference voltage. Thus when the input is positive, Q2 is ON and Q1 OFF, providing a positive input on the top line to the DC amp. When the input is negative, Q1 is ON and Q2 OFF, providing a negative input on the bottom line to the DC amp. Hence the unfiltered output of the DC amplifier would be unidirectional pulses. This circuit is made to act like an open switch by providing a slightly positive voltage on both Q1 and Q2 bases, which turns both transistors ON and prevents the signal from reaching the DC amp.

Figure 5.13 shows the half wave demod used in the roll rate command leg. The demod consists of two resistors and a PNP transistor. Assume the input signal is in-phase with the 400 hz reference. When the input is positive, Q1 if OFF and a positive signal is supplied to the DC amp. When the input is negative Q1 is ON and the signal is shorted to ground.
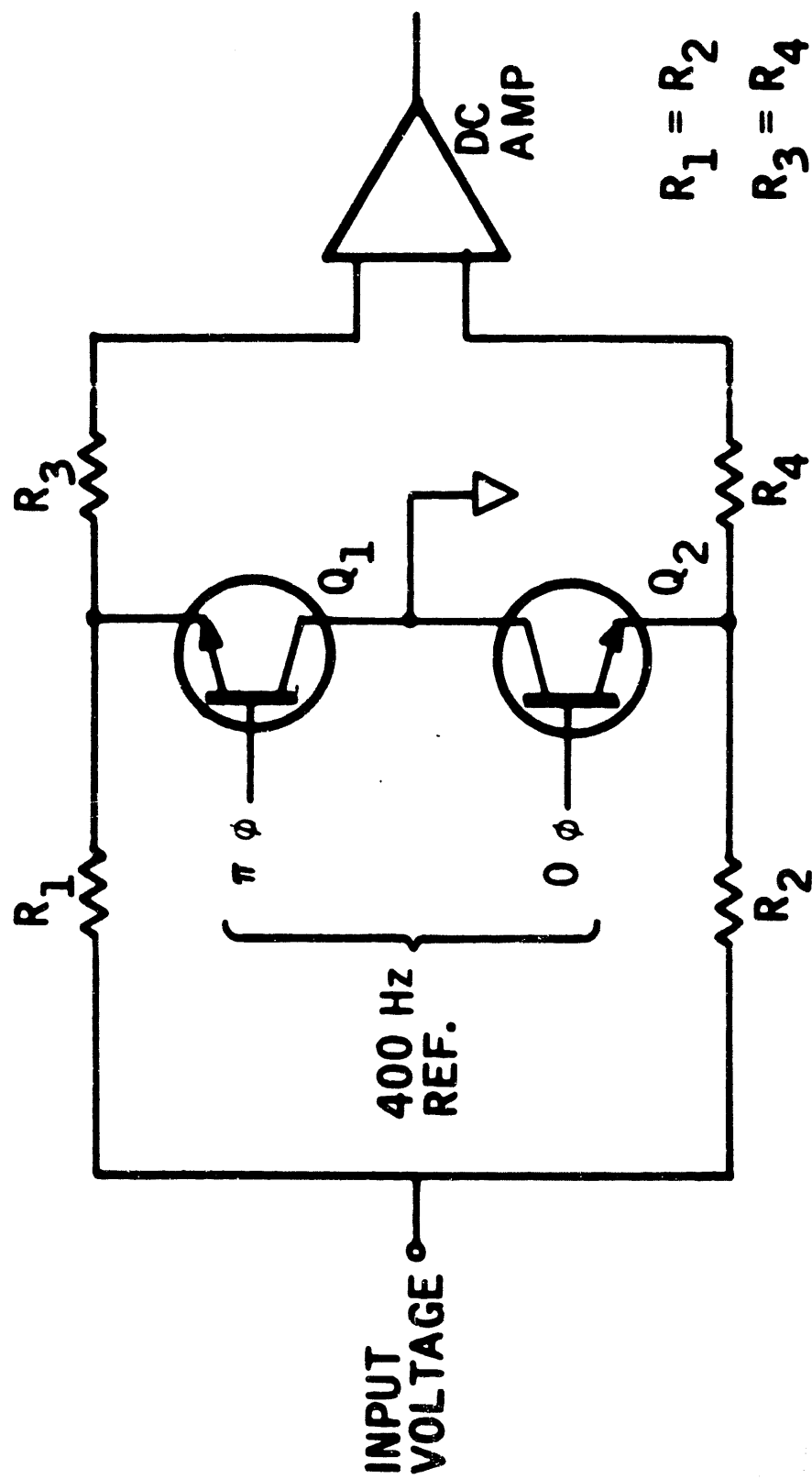
DIFFERENTIAL FULL WAVE DEMOD
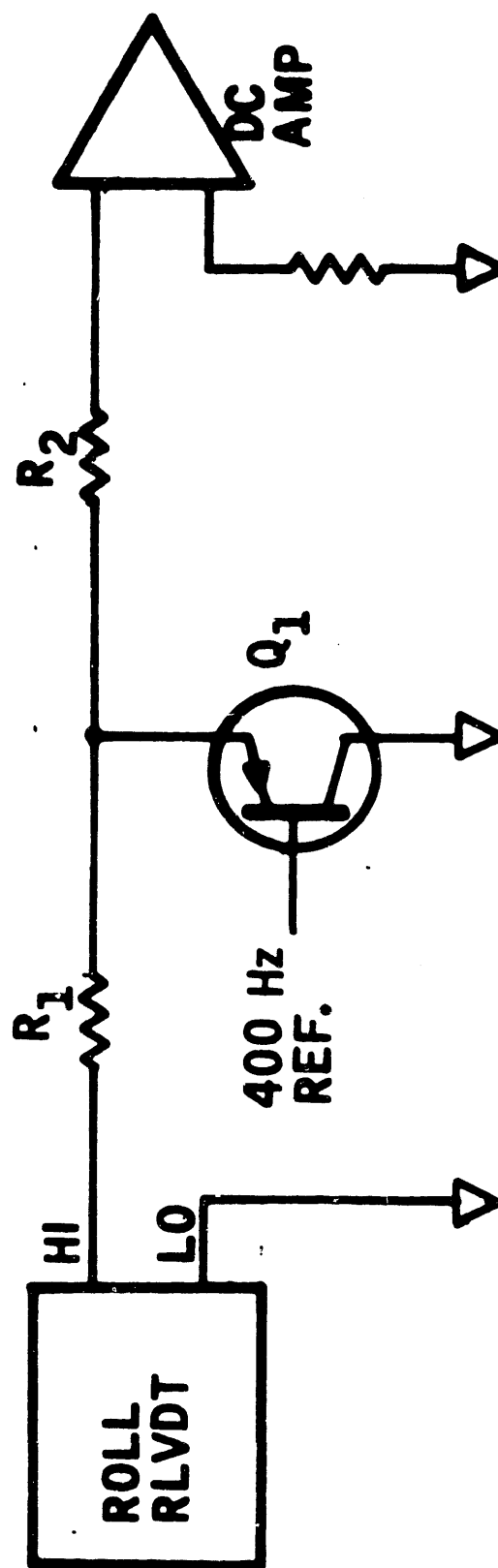

FIGURE 5.12

ROLL RATE COMMAND DEMOD

FIGURE 5.13

Figure 5.14 shows the full wave demod and filter used in the pitch and yaw rate command legs. The demod filter consists of two PNP transistors, three resistors and two tanalum capacitors. Assume the RLVDT output is in-phase with the 400 hz reference. When the input signal is positive, Q1 is ON and Q2 OFF, resulting in the negative signal on the top line being fed through the filter to the DC amp. When the input signal is negative, Q2 is ON and Q1 OFF, resulting in the negative signal on the top line being fed through the filter to the DC amp. Hence the filter receives unidirectional pulses.

Amplifier Circuits

All three legs of each ECA channel bridge utilize amplifier circuits. Figure 5.15 shows the basic forms of amplifier circuits used. Both the AC and DC amplifiers are integrated circuit, differential input, multistage, operational type amplifiers. The output of the AC amplifier is given by the equation:

$$E_O = \left(-\frac{R2}{R1}\right) E_i$$

where E and E are 400 hz signals. The steady state DC response of the DC amplifier is given by the equation:

$$E_O = \left(\frac{R2}{R1}\right)(E_2 - E_1).$$

It is apparent that the steady state response is the same for both amplifiers if E_2 is tied to ground. The dynamic response for the DC amplifier is given by

$$E_O = \frac{R2}{R1}\left(\frac{2}{SR_2C_2+2}\right)(E_2 - E_1)$$

where E_0, E_1 and E_2 are functions of time.
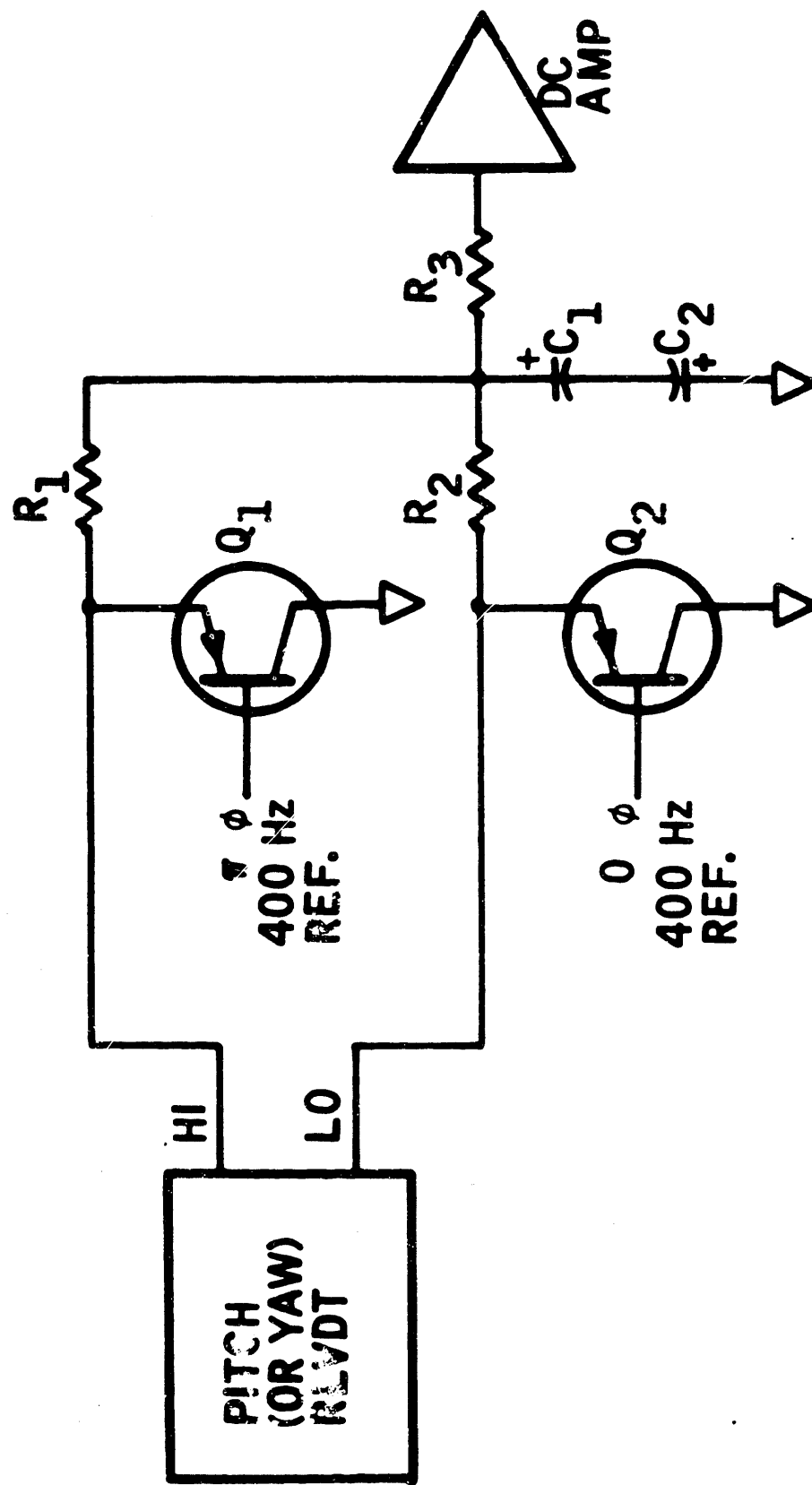
PITCH, YAW RATE COMMAND DEMOD-FILTERS

FIGURE 5.14

AMPLIFIER CIRCUITS
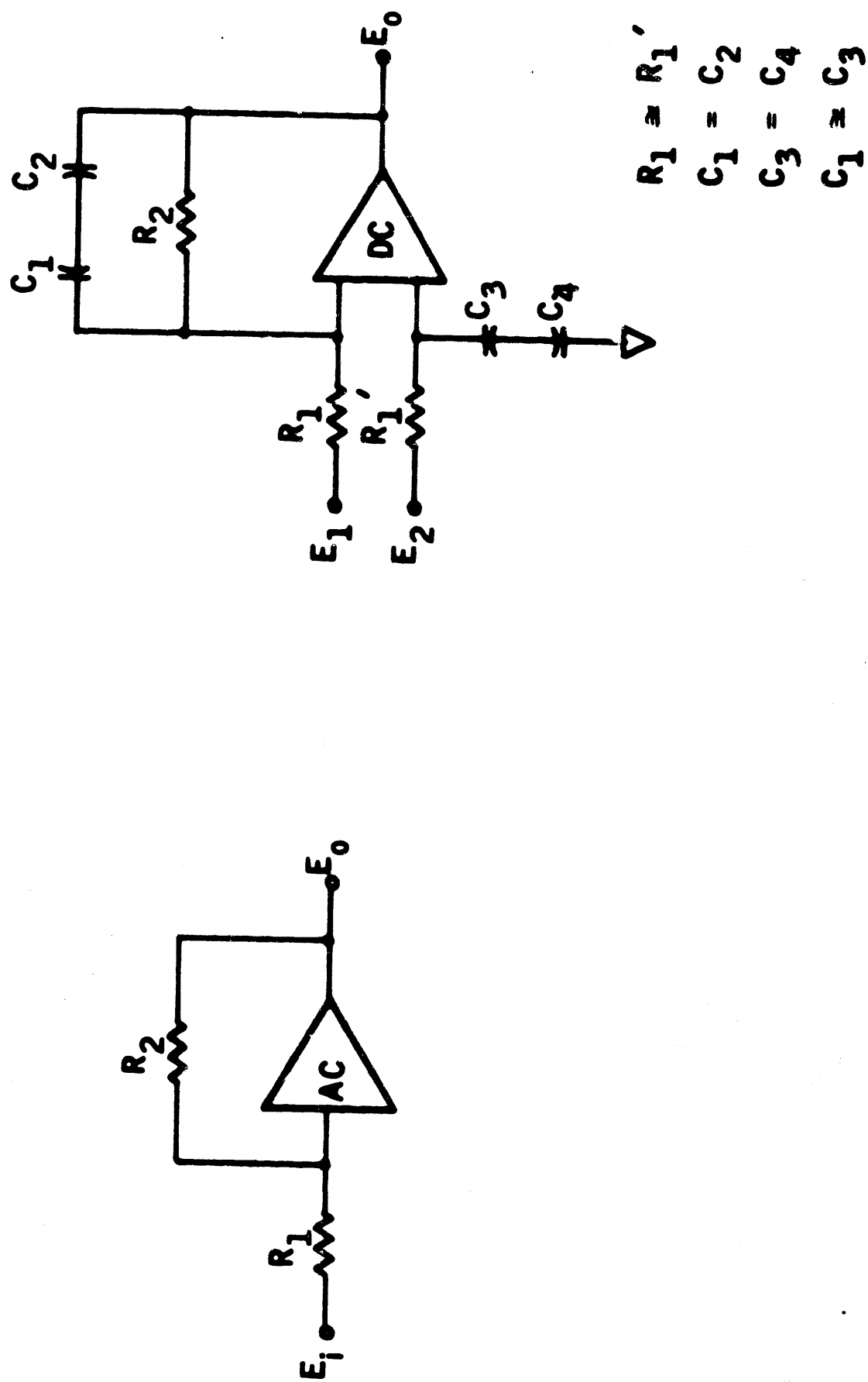

$R_1 = R_1'$
$C_1 = C_2$
$C_3 = C_4$
$C_1 = C_3$

FIGURE 5.15

Attitude Error Deadband - Rate Circuit

The attitude error deadband - rate circuit shown in Figure 5.16 is identical for all three channels. The states of relays K1 and K2 are controlled by the RATE and ATT DEADBAND switches respectively. The relays are shown for the maximum deadband and low rate conditions. For the configuration shown, the DC attitude error signal absolute magnitude must be large enough to forward bias CR1 or CR2 in order to provide a current to the DC summing amplifier. Hence the deadband magnitude is proportional to the ration $\frac{R2}{R1 + R2}$. The deadband is reduced to zero by energizing relay K2. The magnitude of the summing amplifier input current for a specific value of DC attitude error voltage is dependent upon the state of relay K1. For the configuration shown, the input current is determined by R2 plus the parallel combination of R4 and R5. When relay K1 is energized (by placing the RATE switch is the HIGH position) the input current, determined by R2 plus R4, is decreased by a factor of 20. R2'', R3 and R6 are required for impedance matching.

5.9.5 ## Analog to Digital Converter Circuit

The purpose of the analog to digital converter circuit is to provide ON/OFF commands to the RCS auto coils via the auto RCS logic as functions of input signals from the ECA bridge or the RC BO switches. A block diagram of the circuit is provided by Figure 5.17.

Switching Amplifier

The switching amplifier receives inputs from the bridge summing amplifier (and the Pseudo-rate circuit if the LIMIT CYCLE switch is ON) and provides discrete command signals to the minimum impulse (one-shot) circuit, pseudo-
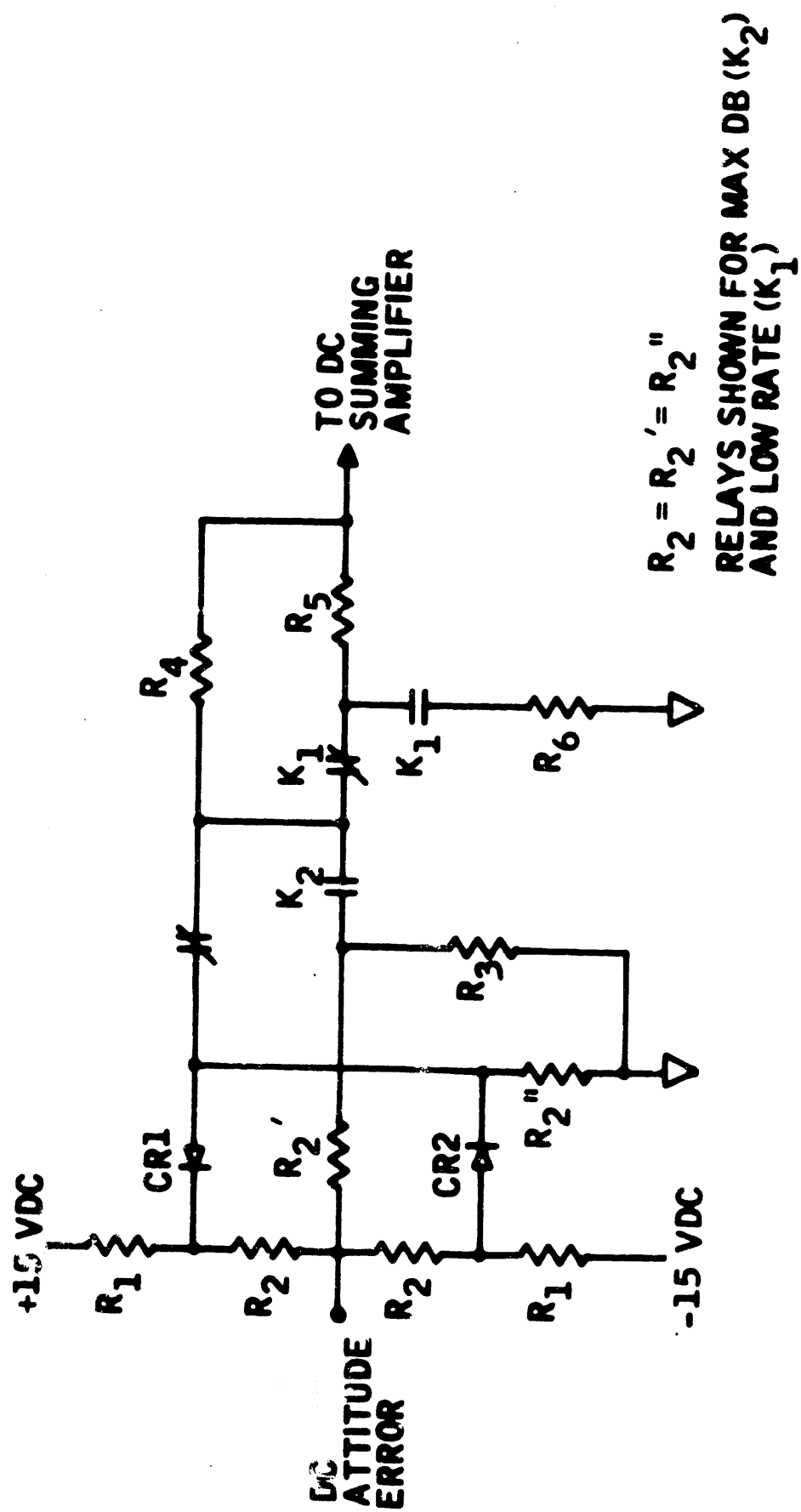
ATTITUDE ERROR DEADBAND - RATE CIRCUIT

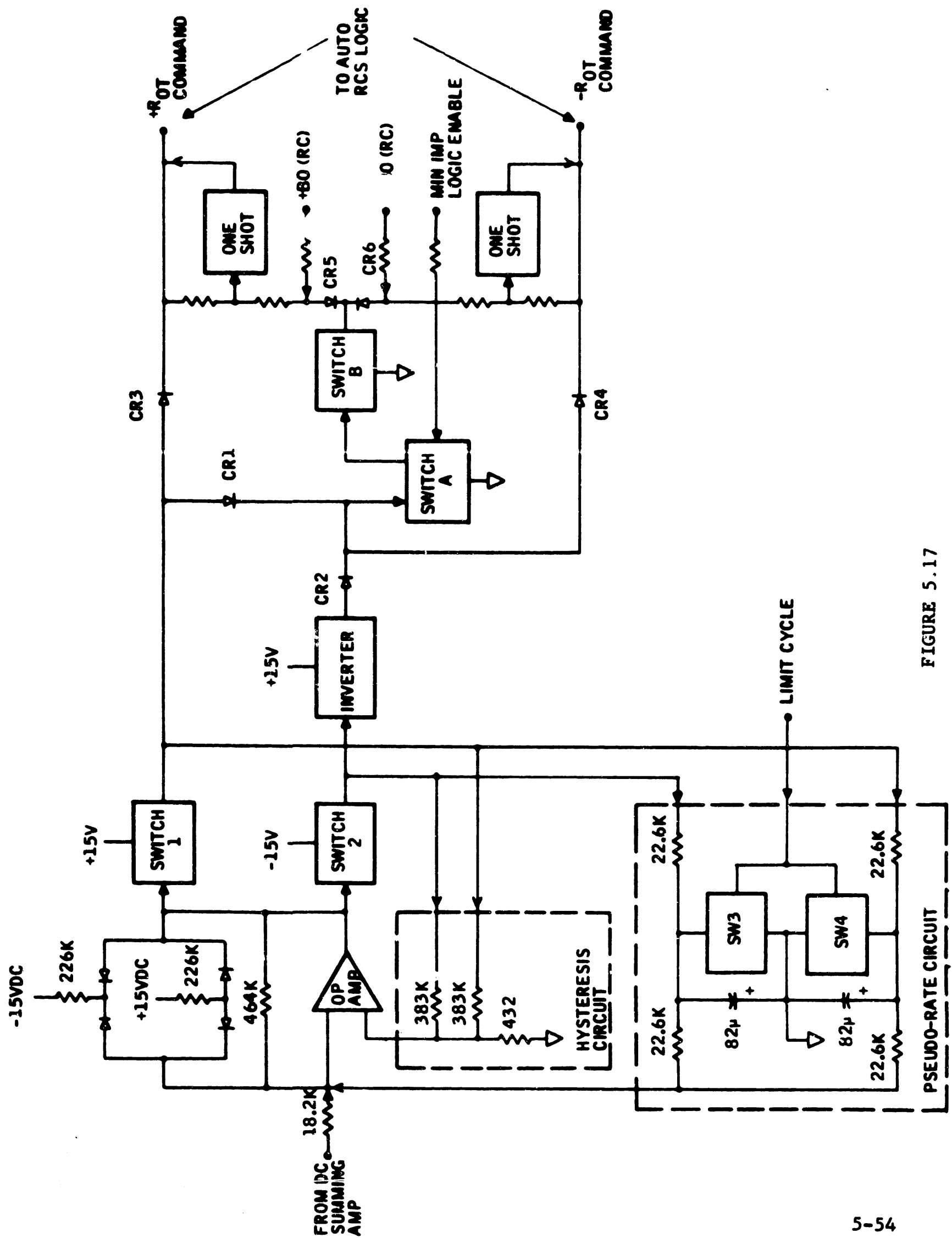FIGURE 5.16

ANALOG TO DIGITAL CONVERTER CIRCUIT

FIGURE 5.17

5-54

rate circuit and the auto RCS logic. The basic elements of the switching amplifier are an operational amplifier, deadband diode bridge circuit, two transistor switches and a hysteresis circuit.

The operational amplifier provides a high gain for the DC input from the summing amplifier. It also provides for a simple deadband and hysteresis mechanization and allows incorporation of the pseudo-rate function for proper limit cycle control.

The deadband diode bridge effectively keeps the operational amplifier gain at zero until the input voltage reaches $\pm$1.2V DC. With the RATE switch in the LOW position this is equivalent to $\pm$0.2^o/sec (rate) and 0.2^o (att error). $\pm$1.2V DC represents $\pm$2.0^o/sec and $\pm$4.0^o with the RATE switch in the HIGH position. When the input voltage exceeds the deadband value, the amplifier gain increases to 25 resulting in fast turn on of SW 1 (+ DC) or SW 2 (- DC). The switch outputs are +15V DC and -15V DC for SW 1 and SW 2 respectively.

The hysteresis network provides positive feedback to the operational amplifier non-inverting input. This feedback ensures that the input must decrease a discrete amount in order to turn the switching amplifier back off. Hence, low level noise can not result in erratic turn on and turn off of the switching amplifier.

Minimum Impulse Circuit

The minimum impulse mechanization consists of two transistor switches and two one-shot circuits. The one-shots receive discrete signal inputs from the switching amplifier output stages or the RC BO switches. The input signal

accepted is controlled by the transistor switches whose states are controlled by the Minimum Impulse Logic Enable signal.

When the MANUAL ATTITUDE switch is in the RATE CMD position SW A is open and SW B is closed. Thus RC commands from the BO switches are grounded through CR5, CR6 and SW B. Positive rate commands from SW 1 are fed through CR3 (directly and through the one-shot) to the auto RCS logic. The one-shot simply ensures that any command to the auto RCS logic is at least of minimum impulse duration. The mechanization for negative rate commands is identical, with +15V DC inputs from the inverter.

When the MANUAL ATTITUDE switch is in the ACCEL CMD or MIN IMPULSE position SW A is closed and SW B ic open. Hence switching amplifier commands are grounded through CR1, CR2 and SW A. RC BO switch commands are now fed to the one-shot circuits which provide minimum impulse commands to the auto RCS logic.

The duration of the minimum impulse command signal is controlled by the 28V DC MN B magnitude (15 millisec nominal). If the DC voltage applied to the solenoid valve coils increases, the valves "pull in" faster. Hence the minimum impulse command duration is decreased with an increase in bus voltage magnitude and vice versa.

Pseudo-Rate Feedback

As indicated in Figure 5.17, the pseudo-rate circuit is merely a passive integrator circuit which provides negative feedback to the operational amplifier. The pseudo-rate feedback is enabled when the LIMIT CYCLE

switch is placed in the "up" position. When the switching amplifier is turned ON, SW 1 or SW 2 provides the "integrator" with a + or - 15V DC input. At the time the sum of the input from the summing amplifier and the "integrator" output gets below the deadband-hysteresis level, the switching amplifier turns OFF. The integrator output then decays until the switching amplifier deadband voltage is again exceeded, at which time the switching amplifier again turns ON.

The following discussion may provide an appreciation for the function of pseudo-rate feedback. The objective is to control the attitude of the spacecraft within a certain deadband by firing the proper SM reaction jets when the S/C attitude gets outside the deadband. For simplicity consider the pitch axis only. Obviously, for maximum conservation of fuel, it is desireable to have the lowest possible rate of drift between the two limits of the deadband and the lowest possible impulse from the reaction jet when a limit is reached. A graphical representation of these desired qualities is shown in Figure 5.18 (a). Since some level of rate must exist, and since the momentum of the spacecraft will extend the displacement beyond the deadbands, a more realistic limit cycle is shown in the phase-plane diagram of Figure 5.18 (b). However, even this limit cycle is difficult to obtain because of the following parameters which cause lags:

. Switching hysteresis of the ON-OFF switch

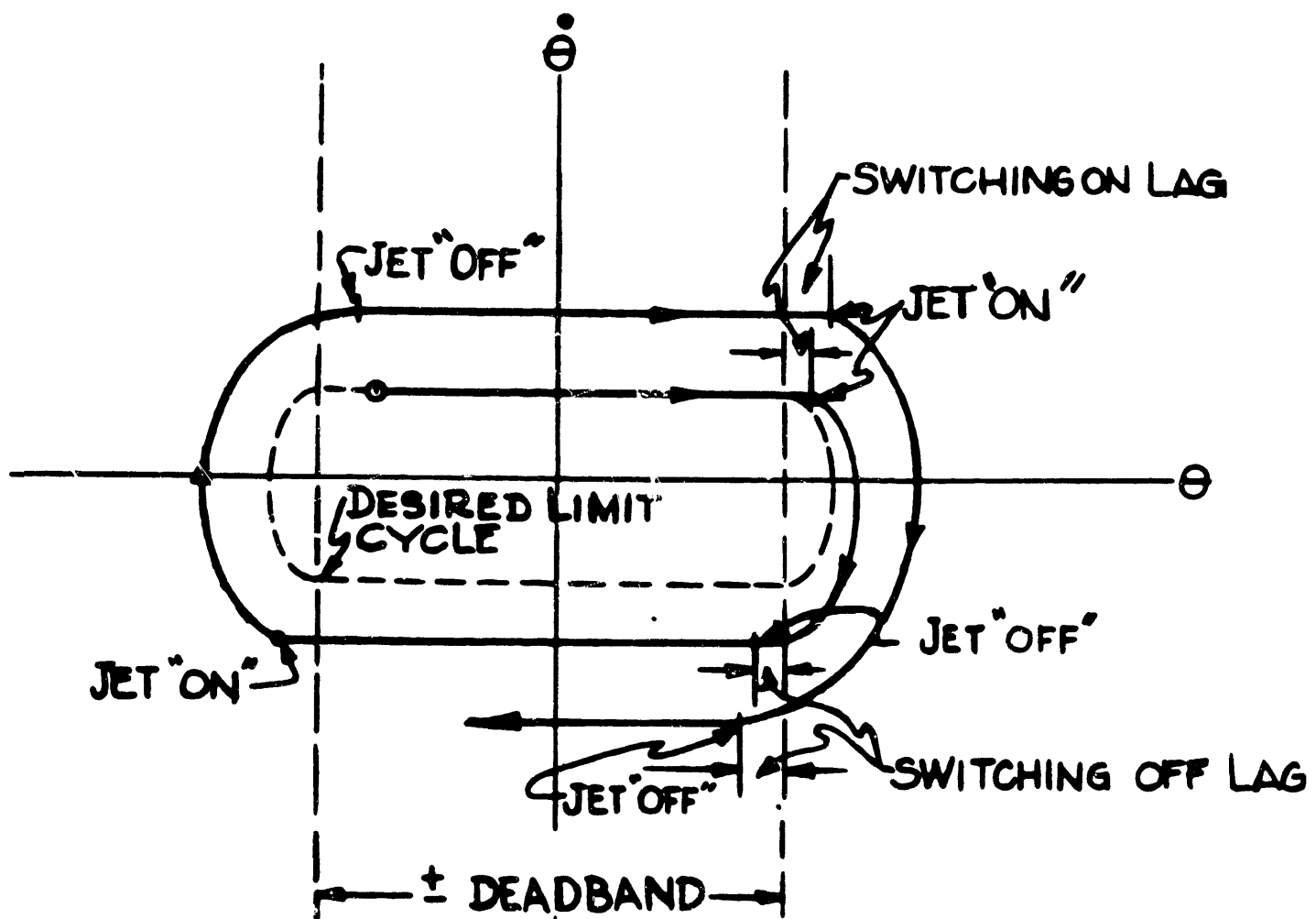. Threshold of the sensor

. Lags due to system time constants

LIMIT CYCLES
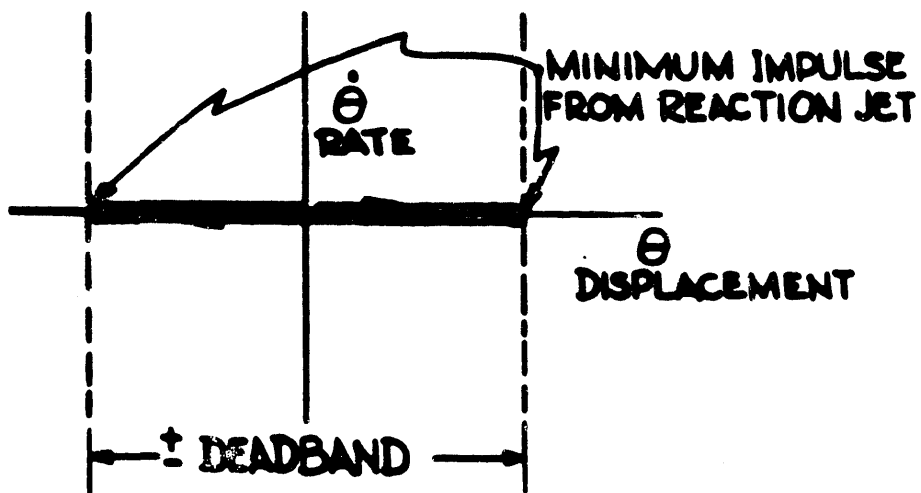
(a) DESIRED IDEAL LIMIT CYCLE
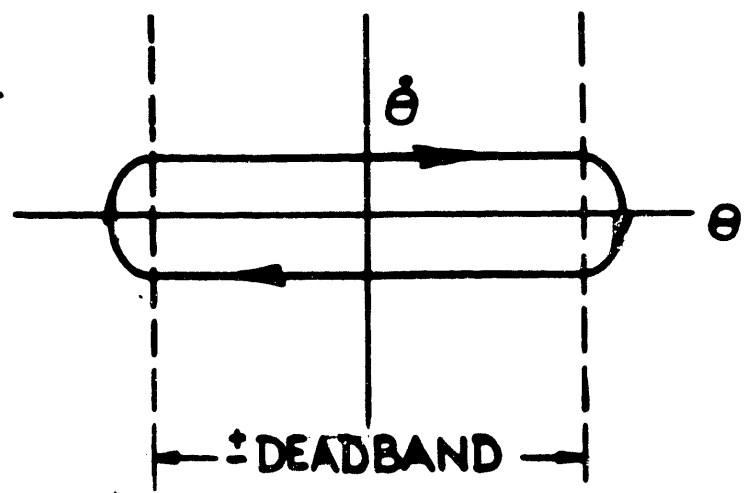
(b) ACTUAL IDEAL LIMIT CYCLE

(c) UNDAMPED LIMIT CYCLE

FIGURE 5.18

. Reaction jet velocity impulse after switch off

. Transport delay in turning a reaction jet ON

Now consider controlling the S/C pitch attitude by sensing the attitude
error and using this signal to control the reaction jets. The phase-plane
diagram would be as shown in Figure 5.18(C). Obviously, in this diagram,
rate and displacement are continuing to build up and the system is unstable.
In our system, the undesirable buildup of rate is controlled by sensing
the vehicle rate and providing a proportional signal as negative feedback
to the switching amplifier. This causes the jets to fire before the actual
attitude deadband is reached and thus stabalizes the system. Figure 5.19
describes such a system.

The previous discussion considered a control system which turned a jet
on when the deadband limit was exceeded and off again when the displacement
returned to within the deadband. It has been found that the effects of most o
of the system lags in such a system can be eliminated by pulsing the jets
on and off when control is required. The pseudo-rate feedback loop around the
the Apollo switching amplifier results in both pulse width and pulse
frequency modulation.

One might conclude that _either_ a rate gyro or a pseudo-rate feedback loop
will stabilize an attitude control system. However it has been found
advantageous to use both. (See Figure 5.20) The rate gyro signal provides
damping due to actual vehicle movement to converge to the limit cycle and
the pseudo-rate signal provides both the anticipation factor and a pulsed
signal to the jets. The combination, used in the Apollo SCS results in
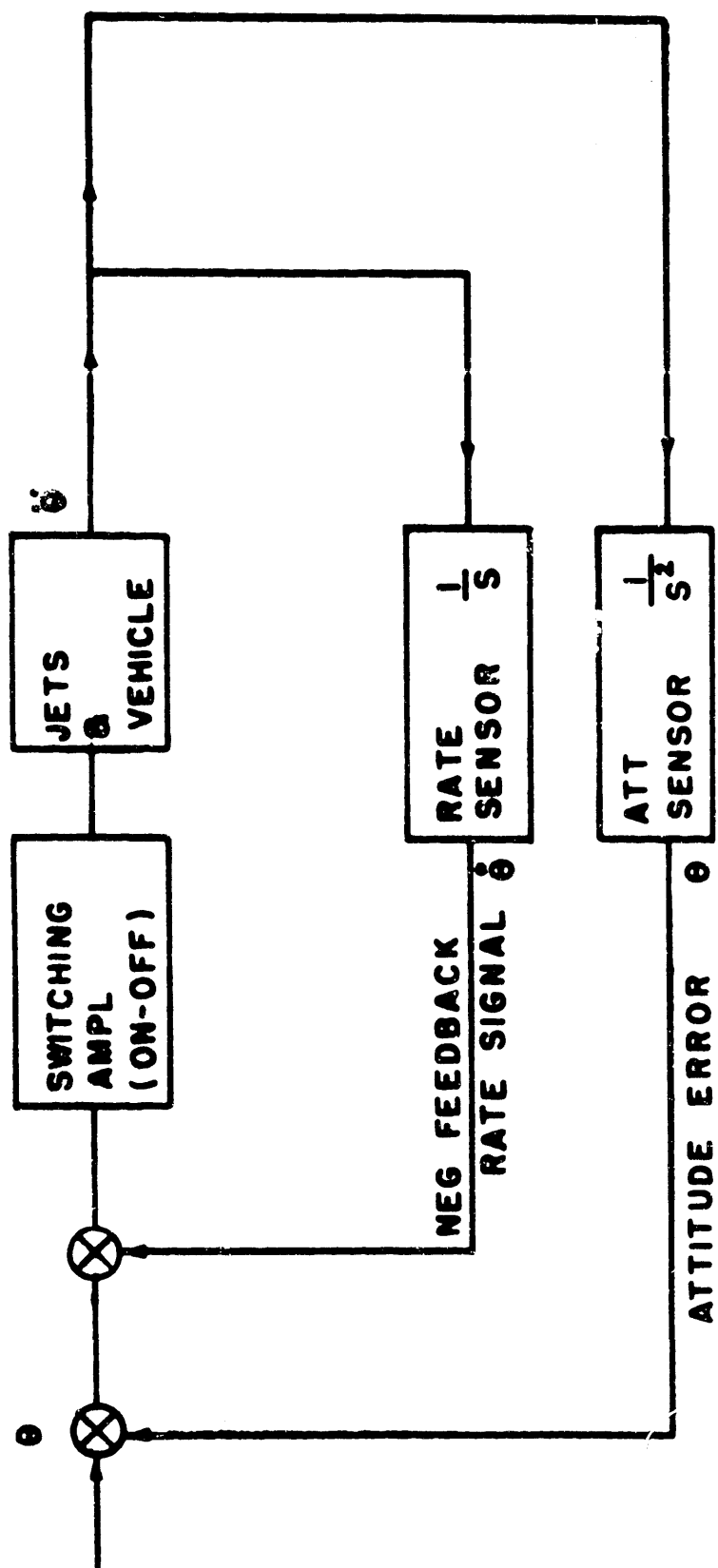
ATTITUDE CONTROL SYSTEM WITH RATE FEEDBACK

FIGURE 5.19

5-60

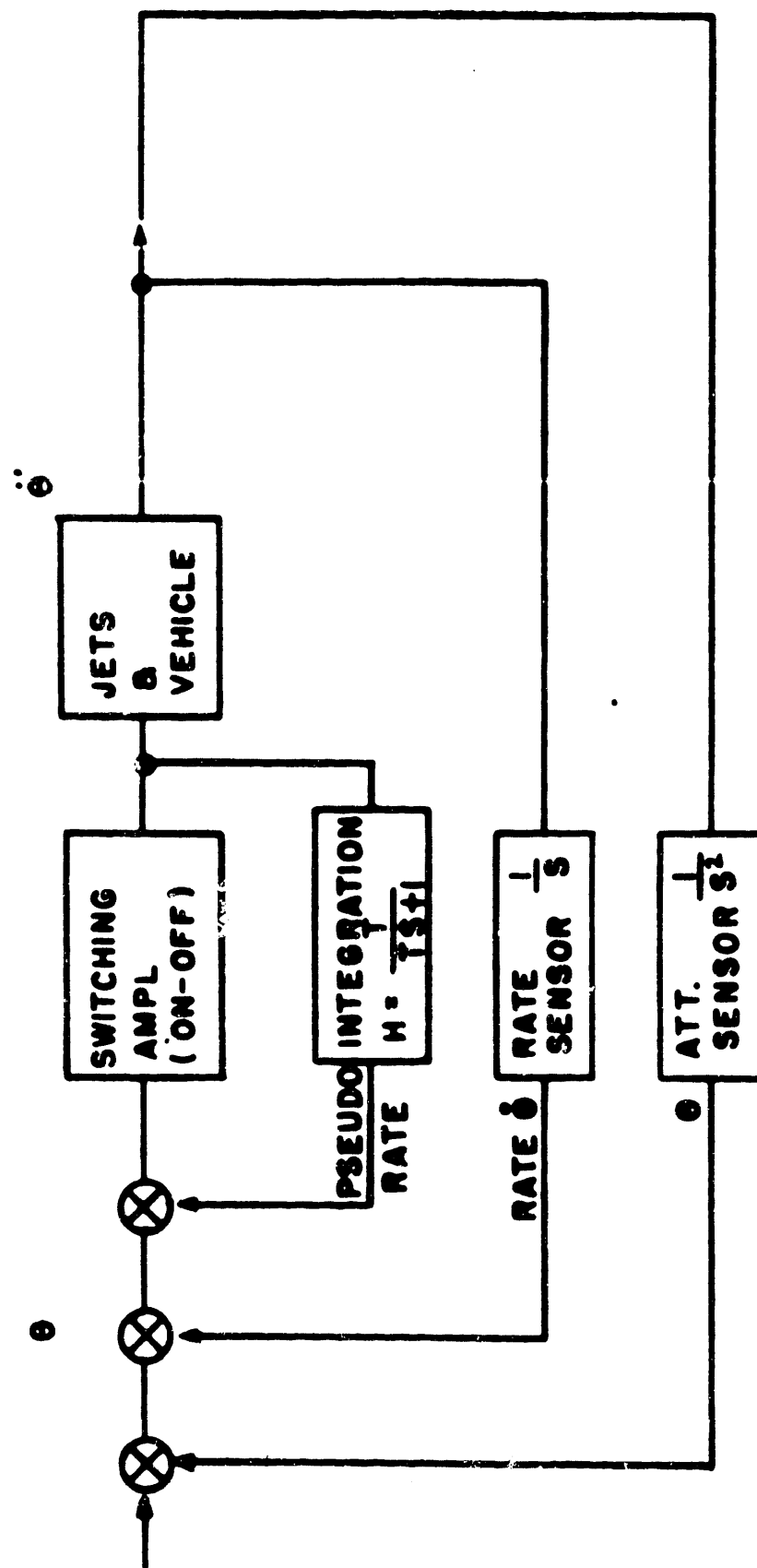# ATTITUDE CONTROL SYSTEM WITH RATE & PSEUDO RATE FEEDBACK

FIGURE 5.20

very low limit cycle excursion time and consequently low fuel consumption.
If desired, the rate gyros may be turned off for electrical power economy
during midcourse drift, using only pseudo-rate for stabilization.

The configuration of the pseudo-rate circuit for the Apollo SCS has been
developed for limit cycle operation. During maneuvers, the effect of the
feedback would be to pulse the jets during the beginning and end of the
commanded maneuver, resulting in an overdamped response and wasted fuel.
To avoid this, the pseudo-rate feedback should be switched out during
manual maneuvers, by placing the LIMIT CYCLE switch in the OFF position.

5.10   POWER DISTRIBUTION FOR REACTION JET CONTROL

This section considers the basic mechanizations for providing power to
the RCS auto and direct coils and to the hardware which controls signals
to these solenoid valve coils.

5.10.1   Auto RCS Enabling Power

Enabling power for the auto RCS coils and for the RJ/EC solenoid drivers
is provided via the Control Panel 8 circuit breakers and AUTO RCS SELECT
switches as indicated in Figure 5.21. Power is supplied to the 16 SM
auto coils or the 12 CM auto coils as a function of the RCS TRNFR switch
(2S44) position (deadface connector for the A/C Roll SM auto coils). Power
is supplied directly from the circuit breakers through the appropriate AUTO
RCS SELECT switches and the RCS TRNFR switch (or deadface connector) to the
RCS auto coils. Solenoid driver enabling power is supplied from the circuit
breakers through the RCS latching relay contacts in the MESC and the
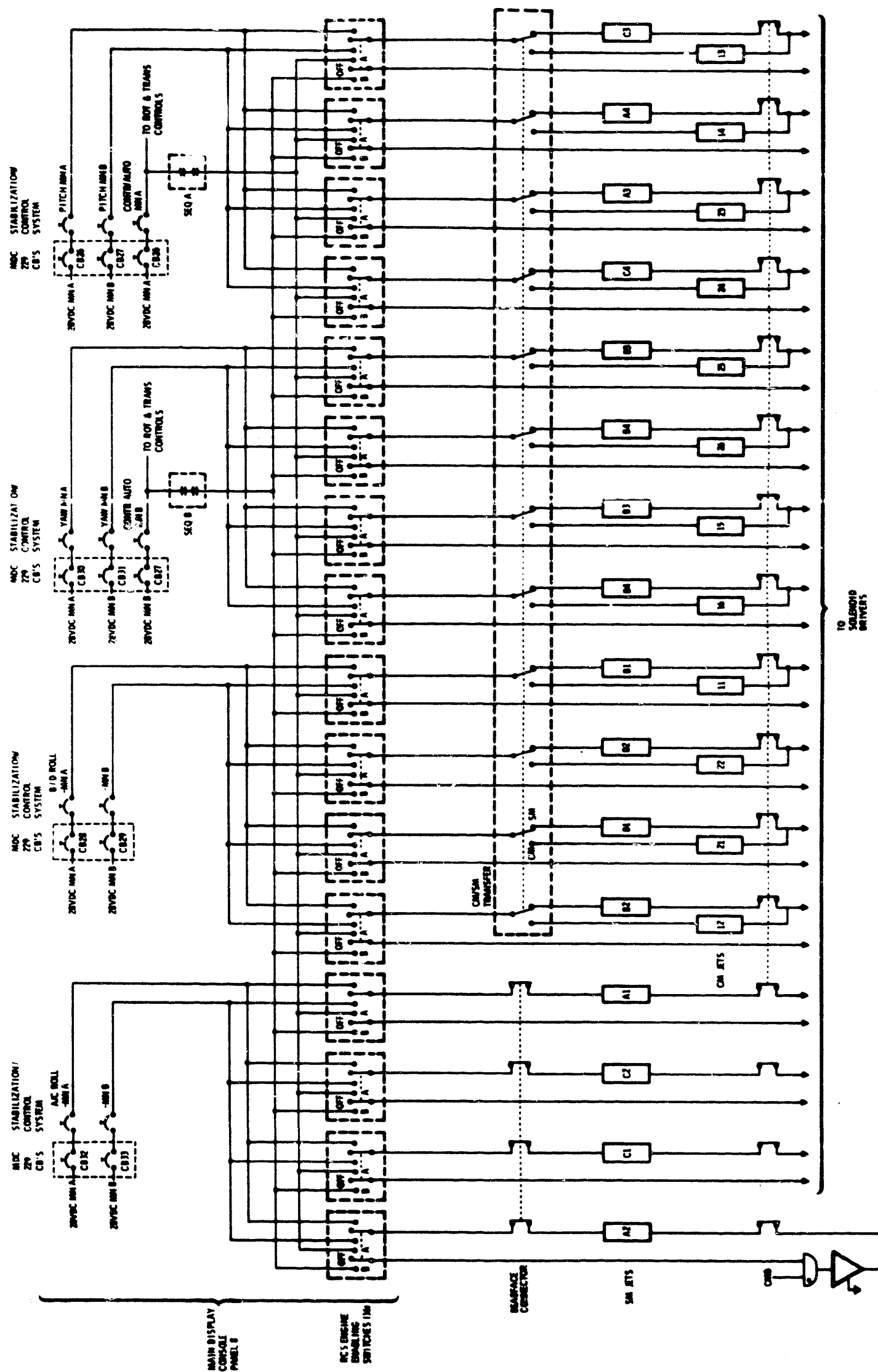appropriate AUTO RCS SELECT switches.

ENABLING POWER–AUTO RCS

FIGURE 5.21

5-63

Figure 5.22 describes the logic which controls the RCS latching relays. The latching relay can be armed or disarmed manually via the spring loaded RCS CMD switch 2S43. The relay can also be controlled automatically as a function of the Abort and Earth Landing System Logic.

5.10.2   Hand Control Power Switching

Figure 5.23 describes the power switching required for operation of the TC, RC1 and RC2. DC power is supplied to the RC BO and direct switches from the circuit breakers through the appropriate ROT CONTR PWR switches (1S64, 1S65, 1S15, 1S67). AC power is supplied to the RC transducers from the ECA through 1S64 and 1S65. The TC translation switches receive DC power from the circuit breakers through the TRANS CONTR PWR switch 1S66 and the RCS TRNFR switch 2S44. The TC NO CW switch receives SCS Logic bus 2 DC power.

5.10.3   Direct RCS Coil Enabling Power

Figure 5.24 describes the methods of turning on the RCS reaction jets via the direct coils. The direct coils of the 12 CM or 16 SM RCS engines can be energized via either or both of the Rotation Control's direct switches. The CM/SM Transfer Control determines whether the CM or SM coils receive the command signals. All of the CM direct coils, except those of the + pitch reaction jets (13 and 23), can be energized from the CM Prop Jet Dump Control.

The SM +X translation (rear facing A4, B4, C3 and D3) reaction jet direct coils can be energized via the DIRECT ULLAGE button or the abort control

RCS LATCHING RELAY LOGIC
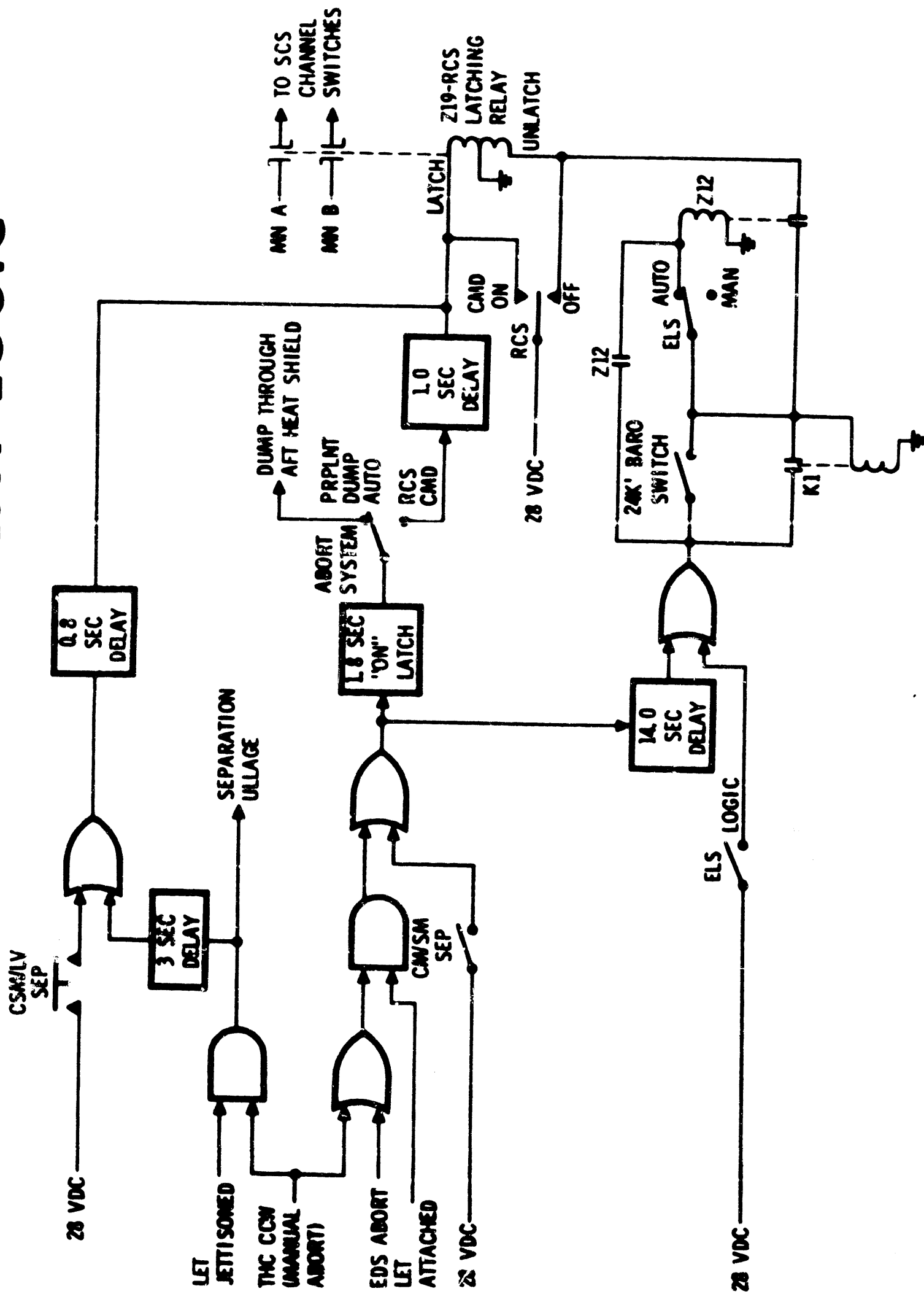
FIGURE 5.22

POWER SWITCHING
ROTATION & TRANSLATION CONTROLS
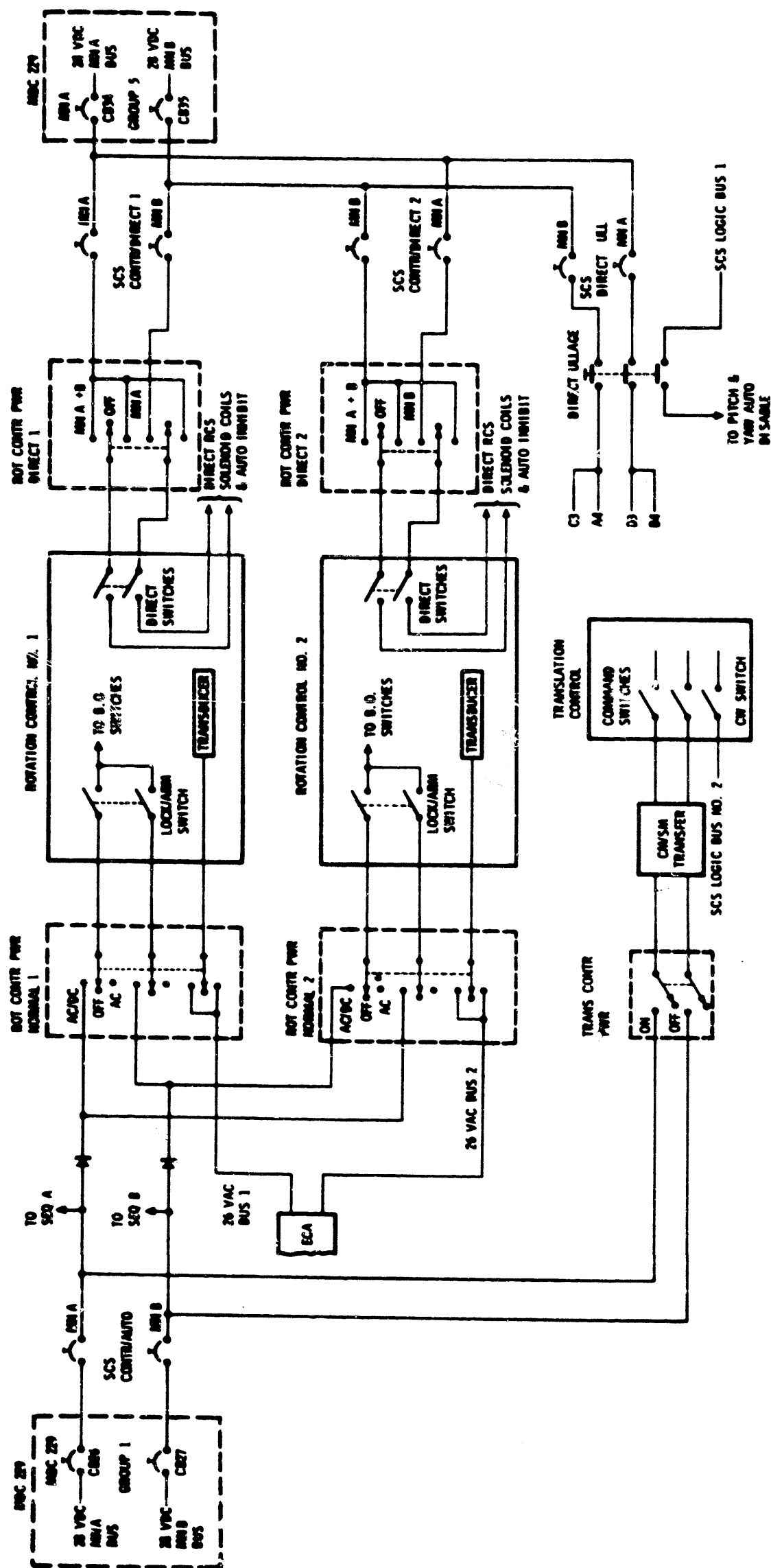
FIGURE 5.23

DIRECT CONTROL LOOP
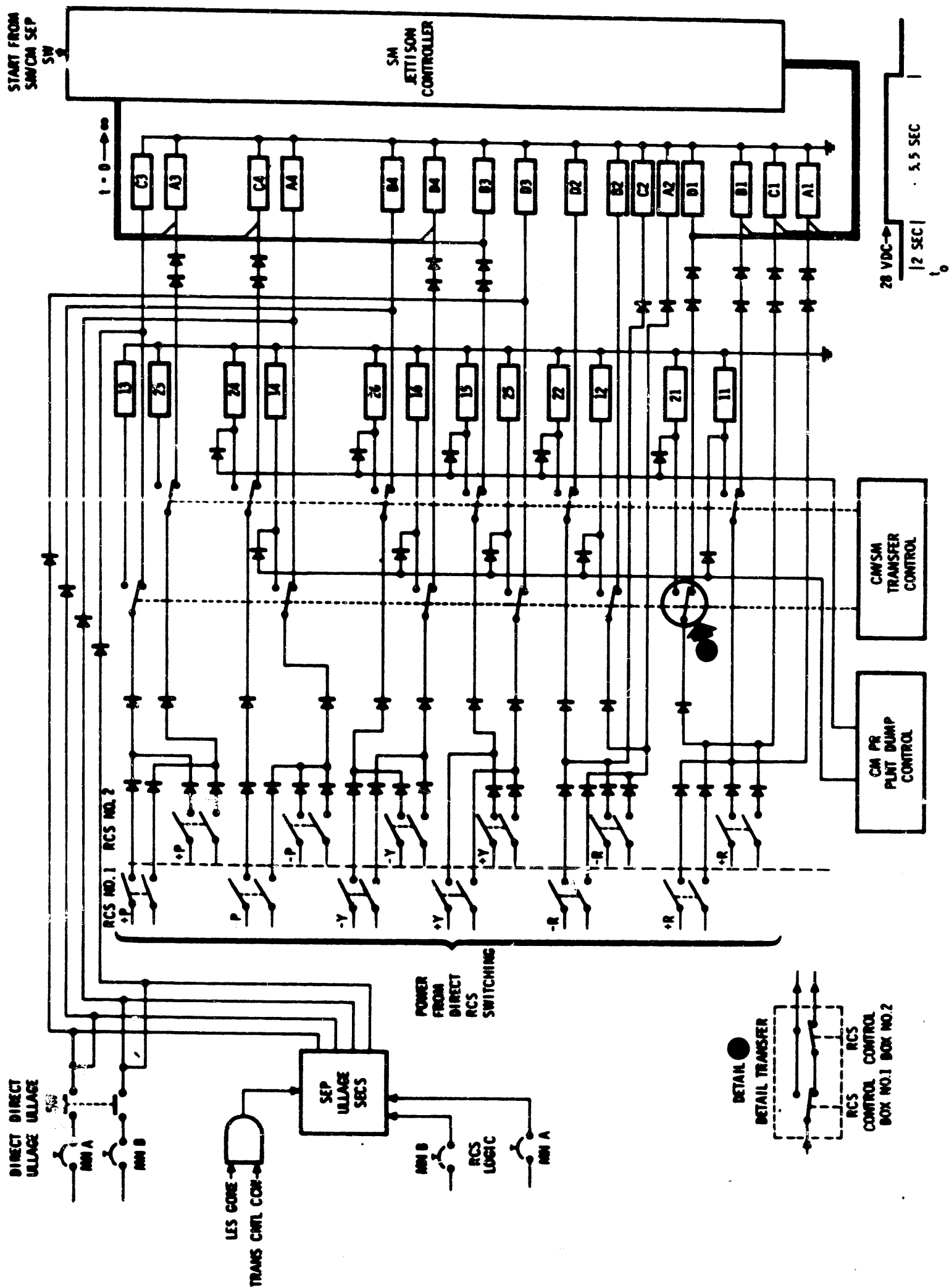
FIGURE 5.24

SECTION 6

THRUST VECTOR CONTROL SUBSYSTEM

6.1   ### INTRODUCTION

This section describes the functions and mechanization of the Thrust
Vector Control (TVC) subsystem. As its name implies, this subsystem
is used to control the thrust vector of the spacecraft during a thrust
maneuver. A thrust maneuver is used to change the inertial velocity of
the spacecraft, such that the free-fall trajectory will carry it to a
specified point in space at a specified time.

Mission Thrust Maneuvers

The Thrust Vector Control subsystem is used for all thrust maneuvers of
the Apollo S/C after translunar injection from earth orbit. For the
nominal mission there will be three mid-course corrections during the
translunar coast period. These corrections will allow the velocity to
be changed by about 300 ft/sec in a total thrust time of about 40 seconds.
When the spacecraft is at the proper point in its trajectory near the
moon, a thrust maneuver must be performed to place the spacecraft in
lunar orbit at an altitude of 80 nautical miles above the lunar surface.
This maneuver will be about 6 minutes in duration to decrease the space-
craft velocity by approximately 3500 ft/sec.

The TVC subsystem is also utilized to place the CSM on the proper trajectory
to allow a safe return to earth of the Command Module and crew (TEI). When
the spacecraft is at the proper point in its lunar orbit, a thrust maneuver

is initiated to place it on the required transearth trajectory. This

maneuver will be less than two minutes in duration, which will increase

the spacecraft velocity by approximately 2800 ft/sec. During the trans-

earth coast period, three mid-course corrections are scheduled to allow

a change in velocity of about 300 ft/sec. The total maneuver time required

for these corrections is on the order of 10 seconds.

Though the TVC subsystem is used for control of the spacecraft for less

than 10 minutes of the total Apollo mission, it should be evident that

it must respond correctly when called upon. The thrust maneuvers performed

are extremely time critical. To ensure proper operation at the required

time a good deal of redundancy is built into the TVC subsystem.

The Basic TVC Problem

The Thrust Vector Control subsystem must contend with a two-fold problem:

1) the spacecraft must be oriented in such a manner during the entire

maneuver, that the change in velocity ($\triangle V$) occurs in the proper direction

and 2) the $\triangle V$ must be initiated at the proper time and terminated when

the desired $\triangle V$ has occurred.

Despite the accuracy of on-board stellar navigation, up-dated by the earth

tracking stations, final achievement of desired spacecraft position and

velocity is always dependent upon control of the vehicle during thrust

vector control phases of the Apollo mission. Operating in a frictionless

and gravity free void, while consuming liquid propellants during a $\triangle V$ burn,

the Apollo spacecraft experiences a moving center of gravity which gives

rise to undesirable moment arms and cross axis accelerations. If uncompensated for, these factors affect the trajectory and become detrimental to the safety of the crew and success of the mission. The Thrust Vector Control subsystem utilizes SCS components which accept commands from either the SCS or the G&N system to minimize trajectory errors. The SCS provides the commands to the SPS servo actuators and to the SPS solenoid valves.

The TVC subsystem is required to position the gimballed SPS engine in pitch and yaw to: 1) ensure that the resultant thrust vector will be aligned through the spacecraft's center of gravity and 2) correct the spacecraft attitude so that the thrust vector will also be in the correct inertial direction. These requirements are complicated by the facts that 1) the actual center of gravity is not always at the calculated location (c.g. uncertainty), 2) the thrust vector is not always along the geometric axis of the SPS engine (thrust misalignment), and 3) the c.g. travels during long burns due to SPS propellant consumption.

The TVC subsystem also provides the ON-OFF command signals to the SPS solenoid valves to initiate and end the thrust maneuver.

6.2   G&C THRUST VECTOR CONTROL

Before getting involved with the detailed mechanization of the Thrust Vector Control subsystem, we will look at the overall configuration. Figure 6.1 is a block diagram which shows the hardware used and the basic signal flow paths necessary to provide the different TVC mechanizations.

G & C THRUST VECTOR CONTROL

FIGURE 6.1

The TVC subsystem provides engine ON/OFF commands to the SPS solenoids
and pitch and yaw control commands to the SPS servo actuators. The
S/C roll attitude is controlled by the RJC subsystem during thrust
maneuvers while pitch and yaw attitude is controlled by positioning
the SPS gimballed engine.

6.2.1   Servo Actuator Control

One of the major functions of the TVC subsystem is to control the position
of the SPS engine. This is accomplished by controlling the clutch engage
current in the SPS servo actuators with the Thrust Vector Position Servo
Amplifier (TVSA). Actuator position and velocity information is fed back
to the servo amplifier to complete the servo loop. The gimbal position
is displayed on the GP/FPI. The actuator position transducers' outputs
are conditioned in the TVSA and EDA to drive the indicators in the GP/FPI.
Command signals for controlling the position of the SPS engine gimbals
originate in either the G&N or SCS system. The TVSA accepts the command
signals from the system selected via control panel switches.

G&N Control

The primary source of thrust vector control command information is the G&N
system. Initializing information is fed to the CMC via telemetry from the
ground and through the Display Keyboards by the crew. The CMC generates
digital TVC command signals as functions of the computer program and the
total attitude and velocity information generated by the IMU. Total attitude
information is fed to the CMC from the IMU resolvers through the Inertial
EDU interface. Changes in velocity along the three IMU axes are sensed by

the IMU accelerometers which provide the information directly to the CMC. The digital TVC commands are converted to analog control signals by the optical CDUs and fed to the TVSA for control of the pitch and yaw servo actuators. Roll attitude control is provided by the CMC through the RJC subsystem.

SCS Automatic Control

The SCS is capable of providing automatic as well as manual control of the SPS servo actuators. Prior to a maneuver, the SPS gimbals are positioned such that SPS thrust will be directed through the calculated spacecraft cg. This positioning is accomplished by setting the GP/FPI thumbwheels to the desired gimbal angles. The resultant gimbal trim command signals are fed to the TVSA. During a thrust maneuver the ECA provides SCS Auto TVC commands to the TVSA to control the SPS servo actuators. The TVC commands generated are functions of the spacecraft rate, attitude error and the difference between manual gimbal trim command and SPS gimbal position. These signals are supplied to the ECA by GA2, GA1 and the TVSA respectively. Roll attitude control is provided by the ECA through the RJC subsystem.

6.?.2 SPS Engine ON/OFF Control

A thrust maneuver is initiated and stopped by providing commands to open and close the SPS solenoid valves which control the flow of propellants to the SPS engine thrust chamber. The command signals are fed to the SPS thrust ON/OFF logic in the RJ/EC for both G&N and SCS control.

G&N Control

For a G&N thrust maneuver the SPS engine is turned on automatically via a discrete command signal from the CMC. The SPS engine is also turned off automatically by the CMC when the desired change in velocity has been accomplished. Pitch and yaw attitude control is transferred from the RJC system to the TVC system during a ΔV burn. The CMC generates an automatic inhibit for the pitch and yaw RCS commands from one second after the engine ignition command is generated until one second after the thrust-off command is provided.

SCS Control

Thrust ON-OFF control is provided via the RJ/EC for all SCS Control modes. Prior to initiating the maneuver, the desired velocity change is set into the EMS ΔV Counter. During the maneuver the EMS accelerometer senses the change in velocity and provides the signal to decrement the counter. An auto thrust off command is generated by the counter when it reaches -0.1 fps. Pitch and yaw attitude control is transferred from the RJC system to the TVC system during a ΔV burn. The SPS Engine ON-OFF logic in the RJ/EC automatically disables the pitch and yaw RCS engines from one second after the engine ignition command is generated until one second after the thrust-off command is provided. If control is transferred from G&N to SCS during a maneuver, the RJ/EC SPS engine ON/OFF logic ensures that the engine remains ON until the ΔV counter reaches -0.1 fps. To initiate a thrust maneuver under SCS control, the control panel 1 THRUST ON switch must be momentarily depressed while an ullage command is present. The engine then remains ON until the ΔV counter reaches -0.1 fps.

TVC RELATED CONTROL PANEL SWITCHES

The following is a brief discussion of the basic functions of the control panel switches as they affect the Thrust Vector Control subsystem. Figure 6.2 and 6.3 depict control panels 1 and 7.

SC CONT (1S18)

The SC CONT switch provides the capability of selecting SCS or CMC control.

The CMC (up) position places the S/C under CMC control when the Translation Control is not rotated CW. The CMC MODE switch must also be in the Auto (up) position in order to perform a G&N controlled $\triangle$V.

The SCS (down) position places the S/C under SCS control. With the TC CW the S/C is under S control, regardless of the SC CONT switch position selected.

SCS TVC - Pitch, Yaw (1S38, 1S39)

The SCS TVC switches are used to select the desired SCS mode of operation by axis.

The Auto (up) position configures the system for SCS Auto TVC if the SC CONT switch is in SCS and the TC is not CW ($\overline{CW}$), or if the SC CONT switch is in CMC and the TC is CW. If the SC CONT switch is in SCS and the TC is CW the system is configured for MTVC with rate damping.

The RATE CMD position configures the system for MTVC with rate damping for all combinations of the SC CONT switch and the TC positions, except SC CONT switch in CMC and TC $\overline{CW}$.

CONTROL PANEL NO.1

FIGURE 6.2

CONTROL PANEL NO. 7
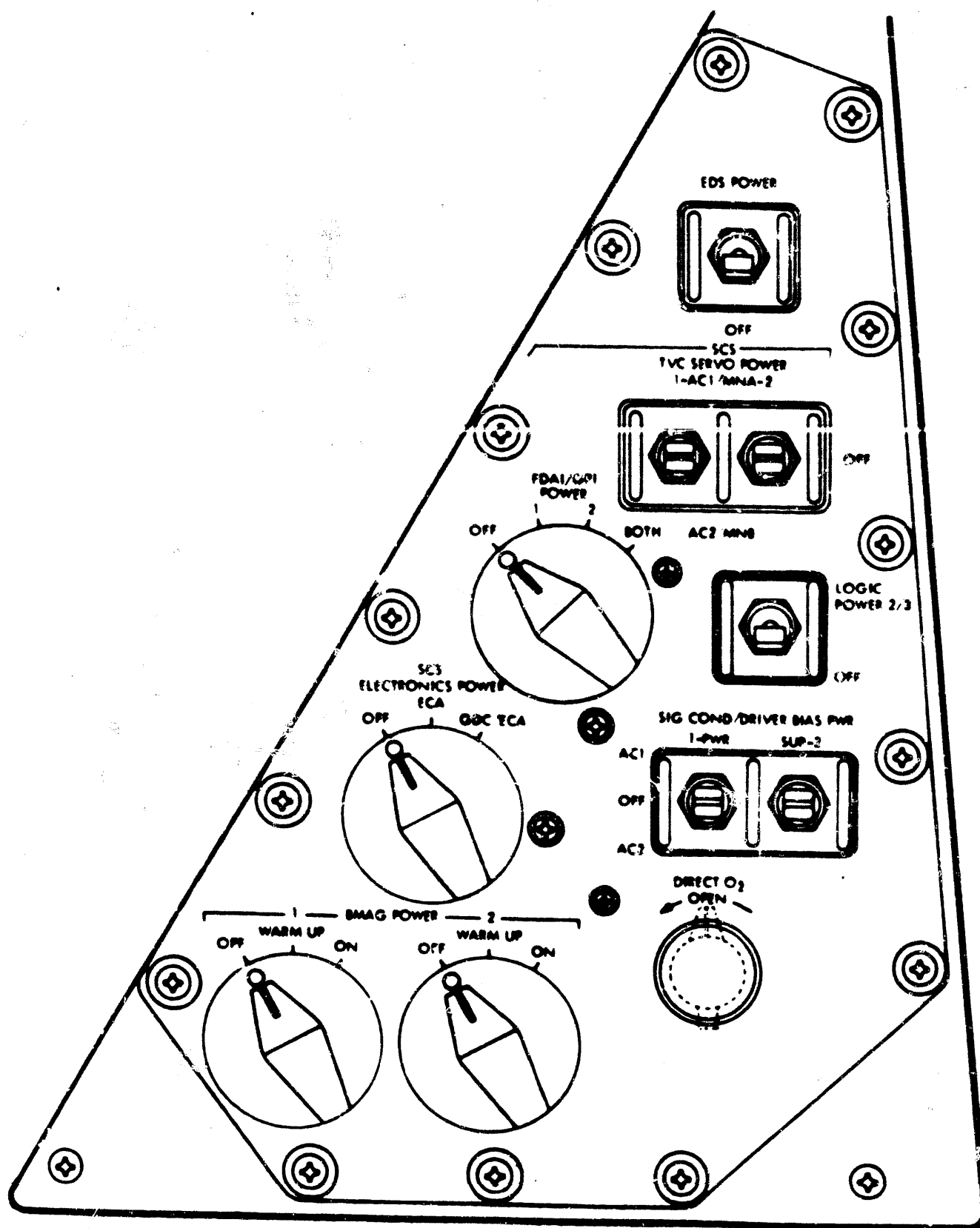
FIGURE 6.3

The ACCEL CMD position configures the system for MTVC without rate damping for the same conditions described above.

TVC GMBL DRIVE - Pitch, Yaw (1S27, 1S28)

The TVC GMBL DRIVE switches permit selection of the primary of redundant servo-positioning loops.

The 1 (up) position selects the primary servo positioning loop with no automatic switchover for a fail-sense signal or TC CW.

The AUTO (center) position selects the primary servo-positioning loop and permits automatic switchover to the redundant loop for a fail-sense signal or for a CW rotation of the TC.

The 2 (down) position selects the redundant servo-positioning loop with no automatic switchover capability.

LV/SPS IND (1S53)

The LV/SPS IND switch provides the capability of selecting the information displayed on the GP/FPI. The SII/SIVB (up) position allows display of the SII fuel and SIVB fuel and oxidizer tank pressures on the indicator. The GPI (down) position allows display of the pitch and yaw SPS engine gimbal positions.

△V/CG (1S54)

The △V/CG switch configures the SCS Auto TVC electronics to provide the proper dynamic response for the two S/C configurations, LM/CSM or CSM.

$\triangle$V THRUST - A, B (1S26, 1S59)

The $\triangle$V THRUST switches provide enable-disable capability for SPS engine ignition via the two redundant sets of SPS solenoid valves. The enable function is provided with either or both switches in the NORMAL (up) position. The switch identification (A and B) indicate the DC power source used as well as the solenoid valve controlled, primary and secondary respectively. The $\triangle$V THRUST switches also provide 28V DC power to the FCSM electronics.

FCSM - SPS A, SPS B (1S61, 1S62)

The FCSM (Flight Combustion Stability Monitor) switches provide an additional automatic disable control over SPS engine ignition in case of an unstable engine firing condition. The two switches are used in conjunction with their respective $\triangle$V THRUST A and B switches. The automatic disable capability is engaged in the up (SPS A or SPS B) position. The down (RESET/OVERRIDE) position provides the capability for resetting as well as overriding the automatic disable function.

THRUST ON (1S25)

The THRUST ON push button switch provides the manual start capability for an SCS $\triangle$V. The SPS engine is started by momentarily depressing the THRUST ON button while a +X acceleration (ullage) maneuver is being performed. The ullage maneuver is commanded via the TC or by depressing the DIRECT ULLAGE pushbutton (1S24).

SPS THRUST (1S23)

The SPS THRUST switch provides the capability for directly enabling (but not necessarily disabling) the SPS engine. The NORMAL (down) position does not provide any command signal. The DIRECT ON (up) position provides the 28VDC ground to directly energize the propellant valves (providing either or both of the ΔV THRUST switches are positioned to NORMAL). Manaul thrust-off function is obtained by setting both ΔV THRUST switches to the OFF position.

ROT CONT PWR - NORMAL - 1,2 (1S64, 1S65)

The ROT CONT PWR - NORMAL switches are designed to allow checking out the MTVC control loop prior to SPS engine ignition while maintaining attitude hold with the RJC subsystem. This capability is provided thru RC1 and RC2 by placing the respective power switches in the AC (down) position. The Rotation Controls are enabled for MTVC commands with the switches in either the AC/DC or AC positions.

BMAG MODE - Pitch, Yaw (1S21, 1S22)

The Pitch and Yaw BMAG MODE switches affect the SCS TVC mechanization. For SCS AUTO TVC, the BMAG MODE switches must be in the ATT 1 RATE 2 (center) positions. For MTVC with rate damping the switch position determines which GA provides the rate information. GA1 rate signals are used with the switch in the RATE 1 (down) position.

TVC SERVO POWER - 1,2 (7S8, 7S9)

The TVC SERVO POWER switches provide independent power switching capability

to the primary and redundant TVSA circuits. Power is removed with the switches in the OFF (center) position. The up position provides AC bus 1 and DC MN A power, while the down position provides AC bus 2 and DC MN B power to the respective channels.

FDAI/GPI POWER (7S3)

The FDAI/GPI POWER switch provides control of power to the EDA for energizing the redundant Gimbal Position Indicator Circuits. Power can be removed or supplied to the No. 1, No. 2 or both sets of indicator electronics.

SCS ELECTRONICS POWER (7S5)

Power to the ECA is controlled by the SCS ELECTRONICS POWER switch. AC bus 1 and bus 2 power is supplied to the ECA with the switch in either the ECA or ECA/GDC positions. Bus 1 power is used for the SCS AUTO TVC circuits while bus 2 supplies the MTVC circuits.

6.4   ### SCS TVC CHARACTERISTICS

The spacecraft attitude is controlled during a delta V by positioning the engine gimbals for pitch and yaw control while maintaining roll attitude with the RJC subsystem. The SCS electronics can be configured to accept Gyro Assembly inputs for automatic control (SCS Auto TVC) or Rotation Control (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.

The following paragraphs present the characteristics of the SCS/TVC configurations. A logic 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.

A simplified TVC signal flow diagram is shown in figure 6.4. Functional enabling switches are used in the drawing for reference. The logic table (figure 6.5), relates the functional switching and Control Panel 1 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 Control 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 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 engages a gimbal extend actuator gear, the other engages a retract actuator gear. 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
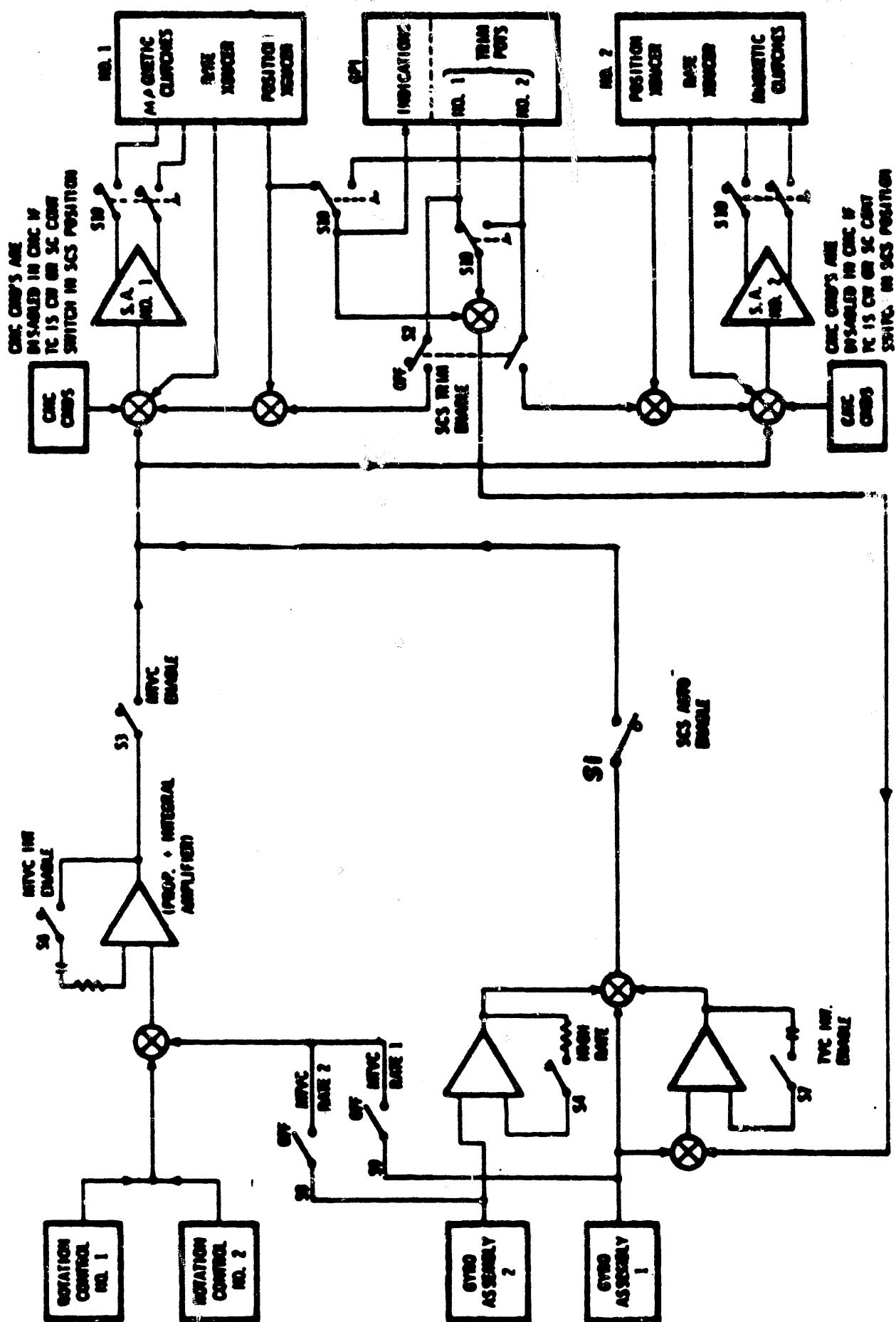
THRUST VECTOR CONTROL - SIGNAL FLOW

FIGURE 6.4

6-16

THRUST VECTOR CONTROL - SWITCHING

PANEL SWITCHING AND LOGIC FOR ENABLING FUNCTIONAL SWITCH POSITION

SYSTEM CONFIGURATION	FUNCT SW./POSITION		SC CONT (1S18)		BMAG MODE (1S2 OR 1S23)			SCS TVC (1S38 OR 1S39)			TVC GMBL DR (1S27 OR 1S28)			ELATION CONTROL		FAIL SENSE (YP OR YY)		IGM 2
			SCS	CMC	ATT 1/RATE 2	RATE 2	RATE 1	AUTO	RATE CMD	ACCEL CMD	1	AUTO	2	CW	CW	FS	FS	
SCS AUTO TVC	S1	SCS AUTO ENABLE	X		(2)			X						X				X
	S2	SCS TRIM ENABLE		X				X							X			X
	S4	HIGH RATE		X											X			X
	S7	TVC INT. ENABLE	X	X											X			X
															X			X
COMMON FUNCTION	S3	MTVC ENABLE	X			X OR X	X OR X		X OR X					X				X
			X	X		X OR X	X OR X		X OR X					X				X
M T V C	S6	MTVC INT. ENABLE		X	SWITCH POSITIONS ARE SAME AS FOR S3 BUT LOGIC REQUIRES 'IGM 2' SIGNAL													
RATE CMD	S8	MTVC RATE 2	X			X OR X	X OR X	X	X					X				X
	OR		X	X		X OR X	X OR X	X	X					X				X
	S9	MTVC RATE 1		X	SWITCH POSITIONS ARE SAME AS FOR S8 EXCEPT BMAG MODE SWITCH MUST BE IN RATE 1 POSITION													
ACCEL CMD	S11 AND S9 OFF									X								
SERVO NO. 1	S10	1									X	X		X		X		
											X	X		X		X		
SERVO NO. 2	S10	2										X	X	X		X	X	

FIGURE 6.5

through functional switch S10. 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 signal will enable
transfer to servo No. 2 in both axes, whereas, the FS signal will enable
transfer only in the axis where it is present. The fail sense signal is
generated in the motor excitation circuitry of servo actuator No. 1 when
an overcurrent is sensed or if motor No. 1 is not started. The transfer
logic described is included in the logic table (figure 6.5).

6.4.1   SCS Auto TVC

The SCS auto configuration is selected as shown in Figure 6.5. In this
configuration spacecraft angular rates and attitude errors are sensed by
the BMAG packages. Attitude error, gimbal position, and gimbal trim
signals are summed at the input to an integrator amplifier. The integrator
output is then summed with rate, attitude error, gimbal position, and
gimbal rate at the servo amplifier input (figure 6.4).

Steady-state operation is obtained when the gimbal is positioned such
that the thrust vector is aligned through the vehicle center of gravity
and the error at both summing points is a constant-zero. The integrator
input error is zero when gimbal position minus gimbal trim is equal to the
negative of the attitude excursion sensed by BMAG 1. This is the spacecraft
/gimbal orientation necessary to obtain and maintain the desired thrust
direction. Thus, transients, due to cg uncertainty errors or cg shifts

during thrusting, are forced by the integrator to have the necessary steady-state solution. However, final pointing vector errors will be incurred due to the quadrature accelerations induced during the transient phases.

Pre-thrust gimbal trim is accomplished by manually turning the trim wheels on the gimbal position indicat. (GPI) to obtain the desired indicator readout. The trim wheel in each axis is mechanically connected to two potentiometers. As shown in figure 6.4, one potentiometer is associated with servo No. 1 and the second with servo No. 2. It is desirable to trim before an SCS delta V, to minimize the transient duration time and the accompanying quadrature accelerations. It is also desirable to properly set the trim wheels 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.

6.4.2   Manual Thrust Vector Control

The electronics can be configured for MTVC by enabling functional switches S3 and S6 as shown in figure 6.5. Selecting MTVC before a delta V would be accomplished with the first panel configuration. The two remaining panel configurations would be used to transfer to MTVC from either the SCS AUTO or CMC control configuration, during a delta V, by rotating the Translation Control clockwise. Control can be transferred from CMC to either the RATE CMD or the ACCEL CMD (or SCS AUTO) configuration, or from SCS AUTO to the RATE CMD configuration. The configurations are selected by axis.

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 reset.

b. Make corrections with the RC about its reset position, rather than holding a large displacement.

c. With no manual input, SC rate is damped out in the RATE CMD configuration.

The selection between the RATE CMD and ACCEL CMD configurations is made through functional switches S8 and S9. If either switch is enabled, as called out in the logic table, then the RATE CMD configuration is established. 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 both functional switches.

The RATE CMD configuration is analogous to the proportional rate capability described in the RJC subsystem (paragraph 5.7.2) 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 of
the spacecraft will develop. The rate gyro, through the proportional
amplifier, 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 the cg). However,
due to the integrator, the rate is driven to zero. When a RC input (manual)
is present, a steady-state vehicle rate will be established such 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 spacecraft attitude and
thrust vector position.

6.4.3   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.

A block diagram of the ignition control circuitry is shown in figure 6.6.

SPS ENGINE ON-OFF LOGIC
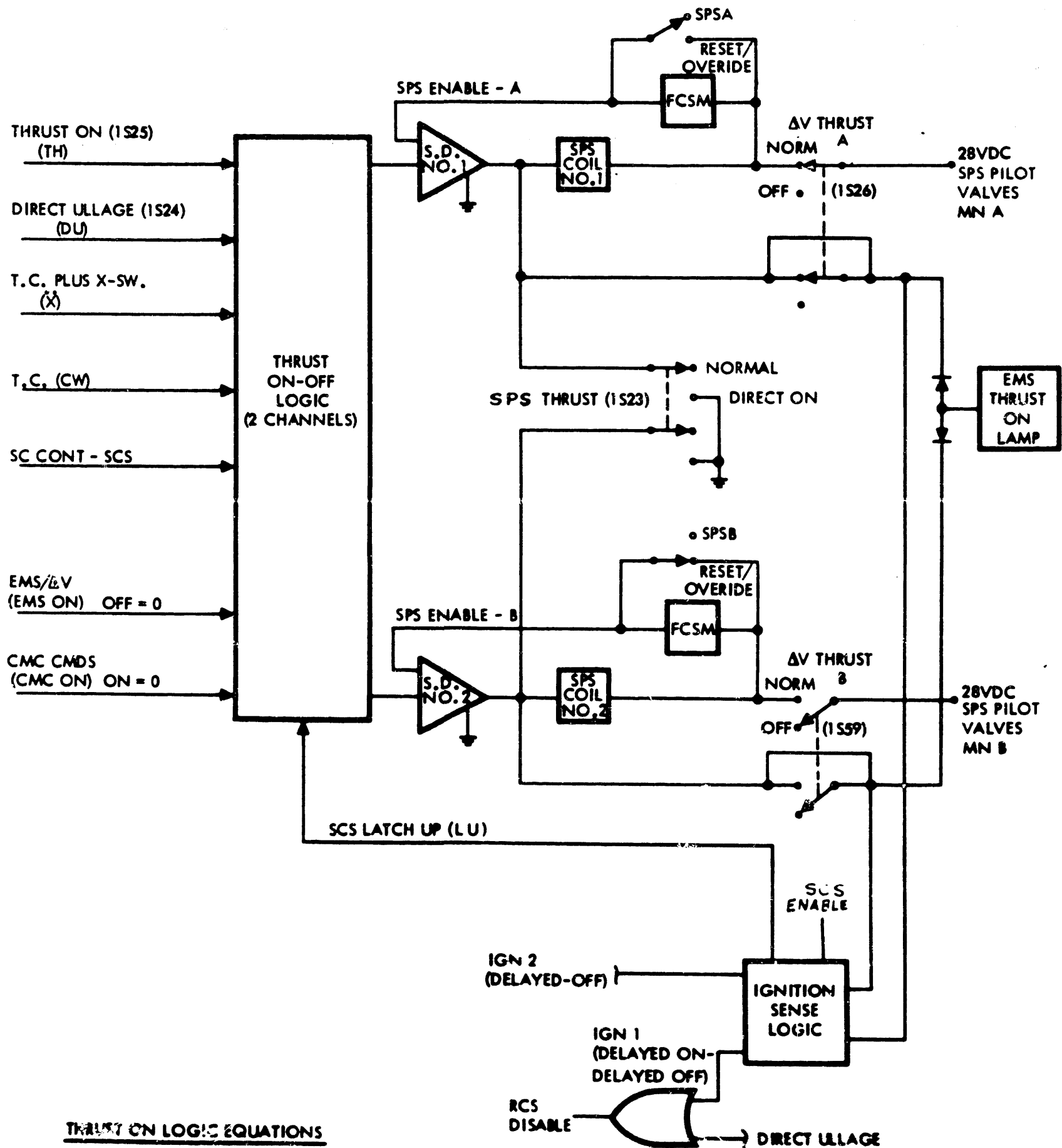
THRUST ON LOGIC EQUATIONS

1. CMC THRUST ON = $\overline{SCS} \cdot \overline{CW} \cdot$ (CMC ON)

2. SCS THRUST ON = $\left[TH \cdot (DU + \overset{..}{X}) + LU \right] \cdot (SCS + CW) \cdot$ (EMS ON)

3. CMC TO SCS = $LU \cdot CW \cdot$ (EMS ON) TRANSFER

FIGURE 6.6

Redundant d-c power is supplied to redundant SPS coils and solenoid drivers as shown in the figure. The $\triangle$V THRUST (A and B) and FCSM (SPS A and SPS B) may be positioned either as shown, or in the opposite positions, or both SPS coils may be enabled. (Figure 6.7 provides a detailed drawing of the logic mechanization).

With the switch positions shown in figure 6.6, 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 placing the $\triangle$V THRUST A (or B, if used) 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 6.6 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 Control is not clockwise ($\overline{CW}$). When the CMC changes the logic signal from 0 to a 1, thrust-off is commanded.

For the SCS control configuration the SC CONT sw must be in the SCS position or the TC handle clockwise (CW). A thrust-on enabling signal is obtained from the EMS/$\triangle$V display. Thrust-on is then commanded by commanding a +X-axis acceleration and pressing the THRUST ON pushbutton. When the ground to the SPS coil has been sensed by the ignition sense logic, the
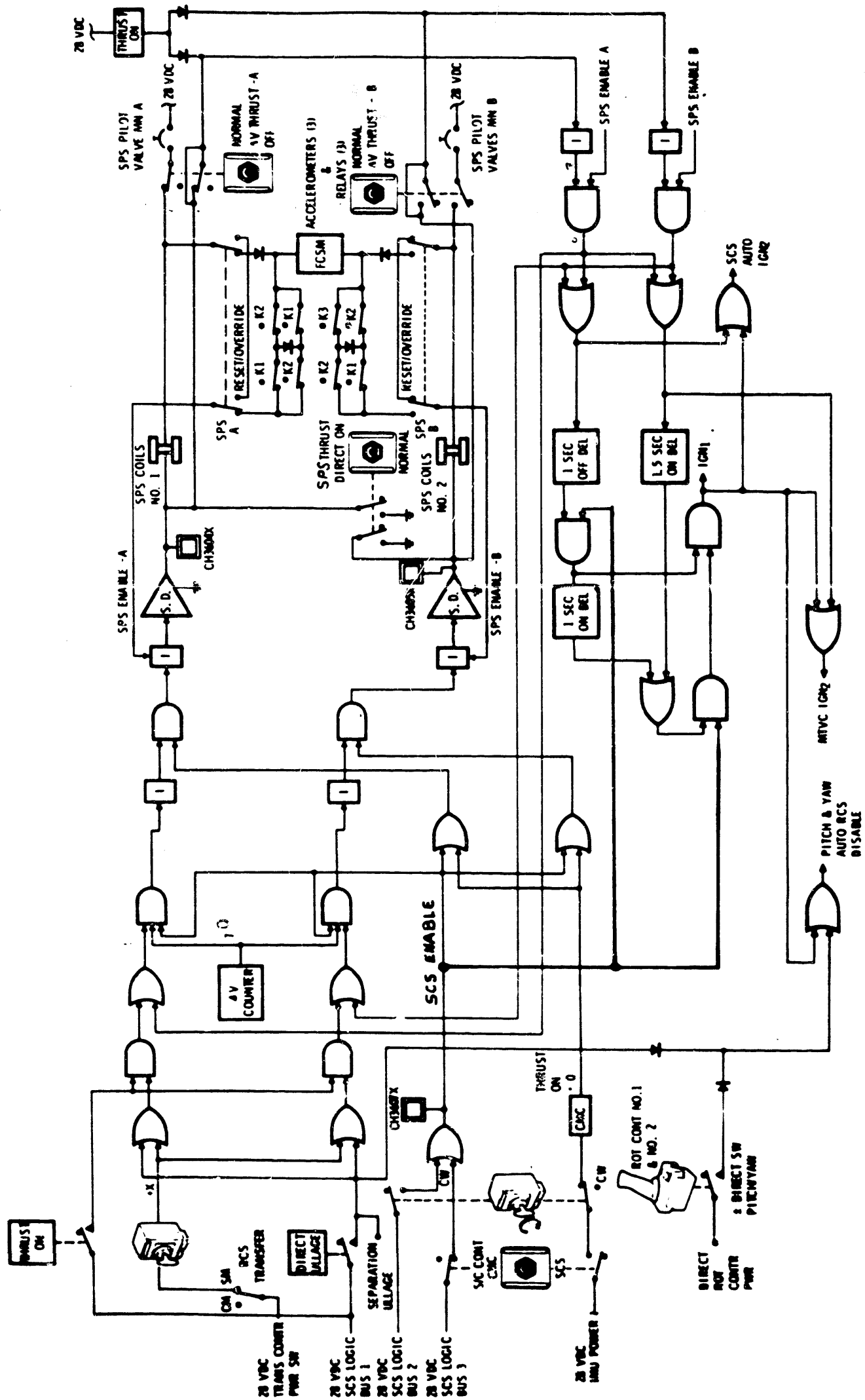
SPS ENGINE ON-OFF LOGIC

FIGURE 6.7

THRUST ON and +X-axis commands can be removed and engine ignition will be maintained by the SCS LATCH UP signal. When the $\triangle$V counter on the Entry Monitor System (EMS) display reads less than 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 rotated 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 $\triangle$V THRUST (A and B) switches to the OFF position. However, this will affect the operation of the ignition sense logic circuitry described later.

The +X command logic signal necessary to enable thrust-on in the SCS configuration, can be obtained from either the DIRECT ULLAGE switch or the TC +X contacts. The differences 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. The TC +X command allows attitude hold to 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 sensed by the ignition sense logic, which generates signals fro disabling the RCS and configuring the SCS electronics for thrust vector control.

The RCS disabling signal, IGN 1, 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. The signal is generated at the same time the ground is switched to the SPS coil, but is not removed until one second after the ground is removed. The delayed OFF enables the TVC electronics to maintain spacecraft control during thrust decay. The delayed OFF function is lost if the backup thrust-off method ($\triangle$V THRUST A and B to OFF) is used or if the FCSM interrupts ignition.

6.5   <u>SERVO ACTUATOR CONTROL MECHANIZATION</u>

Section 6.4 described the basic SCS TVC characteristics through the use of a functional block diagram and a TVC switching table. In this section we will consider a more detailed block diagram description of the TVSA and ECA electronics used to provide control signals to the SPS servo actuator.

6.5.1   <u>Servo Electronics Signal Flow</u>

The mechanization of the servo electronics is identical for the pitch and yaw axes. Figure 6.8 provides a block diagram description of the mechanization for a single axis. Each axis has two completely redundant servo control loops which control the coil current of the respective No. 1 and No. 2 gimbal drive clutches. Command inputs are accepted by the DC Summing Amplifier which drives the servo amplifier. The servo amplifier output consists of a differential current to the clutch coils. In the absence of a command signal (null input to the DC amplifier), a small quiescent

SPS SERVO ELECTRONICS (PITCH)

FIGURE 6.8

6-27
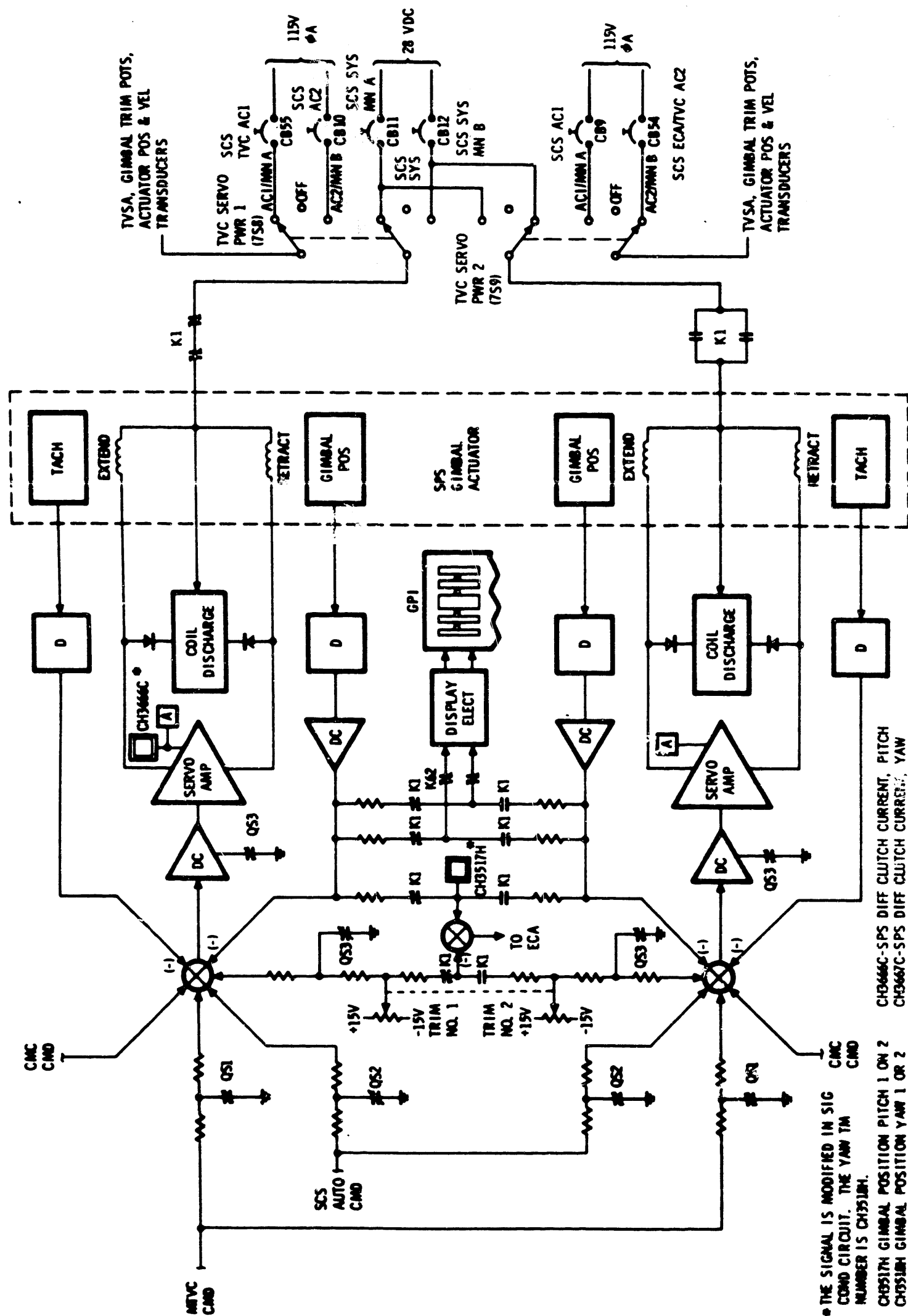
current flows through both the extend and retract clutch coils, holding the gimbal in its null position. As the summing amplifier input increases, the magnitude of the current in one of the clutch coils (depending on the input signal polarity) increases. This results in an extension or retraction of the actuator, altering the gimbal position. The gimbal actuator position and rate transducers supply 400 hz signals proportional to gimbal position and rate respectively. The demodulated position and rate signals are fed back to the DC summing amplifier to complete the basic servo loop. The position feedback circuit contains a buffer amplifier to provide sufficient drive for the gimbal position indicator and SCS Auto TVC circuits. Separate K1 relay contacts are used to supply the gimbal position signal to the redundant GPI meters and to the ECA from the active servo loop.

The redundant servo loops are enabled via the two TVC SERVO POWER switches. AC power is fed to the redundant TVSA power supplies, demods and the gimbal actuator position and rate transducers. The TVSA power supplies feed the gimbal trim pots. The extend and retract clutches are enabled by 28 VDC power through the K1 relay contacts. (The contacts of separate relays are connected as shown to ensure normal operation in the event either one of the relays fails.)

The command inputs to the servo loops are supplied by one of three sources, depending upon the mode selected. The SCS commands are enabled and disabled by redundant solid state switches QS1, QS2 and QS3. The CMC commands are enabled and disabled within the CMC, and thus are not affected by the SCS logic mechanization.

CMC Control

QS1, QS2 and QS3 are shown in the proper states for CMC control via the
No. 1 gimbal drive mechanization. Prior to SPS engine ignition the CMC
trims the gimbal to the proper position by providing a constant DC voltage
on the CMC CMD line. The servo loop responds to this command, driving the
gimbal until the position feedback signal is of sufficient amplitude to
null the summing amplifier input. The gimbal will be held in this trim
position due to the quiescent current through the extend and retract
clutch coils. The gimbal position signal is fed thru the two NC K1 relay
contacts to the redundant EDA circuits from driving the GPI. (Gimbal
trim command signals inserted vai the gimbal trim thumbwheels have no
effect on the gimbal position when the CMC control mode is selected.
However, the gimbal trim adjustments must be made prior to engine ignition
to ensure proper control if an SCS mode is selected during the burn.)
The CMC controls the gimbal position during the entire burn via the CMC
CMD input to the servo loop.

SCS Control

Selection of any SCS Control mode enables the redundant QS3 switches.
This allows positioning of the SPS gimbals via the gimbal trim thumbwheels.
The servo loop functions in the same manner for all control modes; the
only difference is in the source of command signals.

Selection of an SCS MTVC control mode enables the QS1 switches. The ECA
electronics is mechanized to provide DC command signals proportional to
RC stick displacement prior to engine ignition. Thus it is possible to

checkout the MTVC mechanization as well as the redundant servo loops by exercizing the Rotation Control in pitch and yaw, and observing the response on the GPI.

QS2 is enabled only when both the SCS Auto TVC mode has been selected and the IGN 2 logic signal is present. The IGN 2 signal is present from the time of engine ignition until one second after the thrust-off command is provided. The one second delay ensures proper gimbal control until thrust has decayed. The gimbal trim command and the gimbal position signal are supplied to the SCS Auto TVC integrator in the ECA.

6.5.2   ## SCS TVC Electronics Signal Flow

The mechanization of the SCS TVC electronics is identical for the pitch and yaw axes. Figure 6.9 provides a block diagram description of the basic elements of a single axis. Portions of the ECA circuitry shown are used for both TVC and attitude control via the Reaction Jet Control sub-system. This is acceptable since the pitch and yaw reaction jets are disabled during SPS engine ignition. The SCS TVC electronics provides the MTVC CMD and SCS AUTO CMD signals to the servo loop summing amplifiers discussed in the previous section.

SCS Auto TVC

Recall that the SCS Auto CMD signal is not made available to the servo loop until the SPS engine ignition signal has been provided. The SCS Auto CMD signal is supplied by the DC amplifier which receives inputs from one of two circuits, depending upon the S/C Configuration. The gain and dynamic
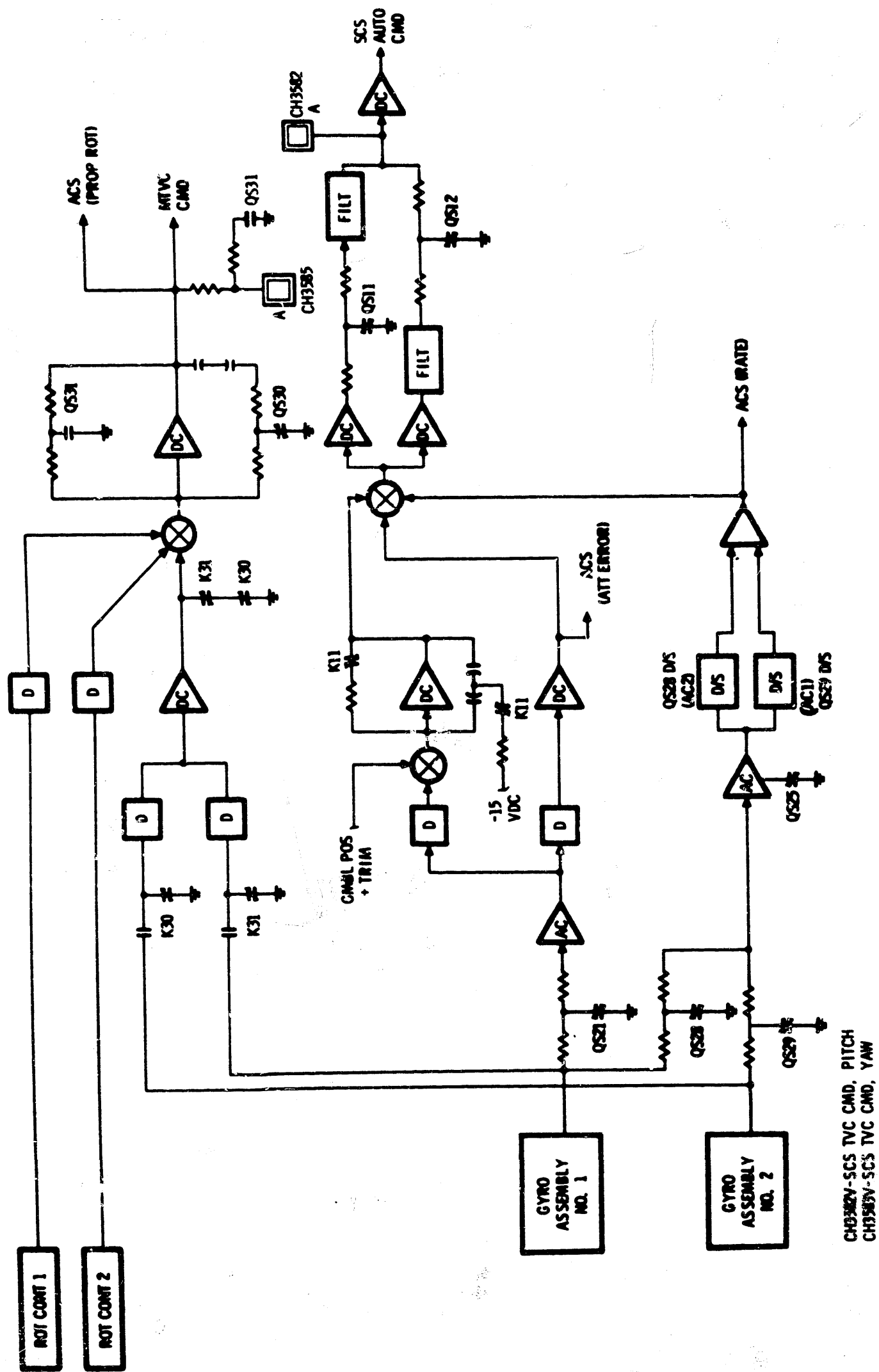
SCS TVC ELECTRONICS (PITCH)

FIGURE 6.9

6-31

response required of the TVC electronics depends upon whether the LM is attached to the CSM. QS11 enables the circuit used for the CSM configuration. QS12 enables the LM/CSM circuit. The latter contains a notch filter to damp the low frequency oscillations due to body-bending of the vehicle. These two circuits receive identical inputs. To ensure proper SCS Auto TVC control, S/C rate and attitude error and gimbal position error information must be provided.

GA2 provides the 400 hz rate signal thru QS29 to the AC amplifier. The amplifier gain is decreased by enabling QS25 via the IGN 2 signal to provide the proper rate scale factor for TVC operation. (Prior to engine ignition this amplifier is configured to provide low rate deadband attitude control via the RJC subsystem). The amplifier output signal is demodulated, amplified and fed to the two DC amplifier summing points.

GA1 provides the 400 hz attitude error signal thru QS21 to the AC amplifier. The amplifier output is fed to the two demods. The lower demod output is amplified to provide the properly scaled attitude error signal to the two DC amplifier summing points. The upper demod output is summed with the gimbal error signal and fed to the TVC integrator. Prior to engine ignition, the TVC integrator is disabled. A forming voltage is supplied to the tantalum (polarized) capacitors during this time to ensure proper operation during the thrust maneuver.

During the thrust period the forming voltage is removed and the integrator is enabled by energizing relay K11. The integrator output is fed to the

two DC amplifier summing points. (See section 6.5.4 for an intuitive
discussion of the operation of the SCS Auto TVC mode).

MTVC

The MTVC electronics consists of the "MTVC Integrator", a DC rate amplifier,
4 demods, 2 relays and 2 solid state switches. For MTVC/ACCEL CMD (MTVC
without rate damping), rate inputs are disabled by relays K30 and K31.
The "MTVC Integrator" is configured as a proportional amplifier until engine
ignition (IGN 2 logic signal), at which time Q830 and Q831 are enabled,
making it a proportional plus integral amplifier. Command signals consist
of 400 hz RC transducer outputs which are demodulated and fed to the "MTVC
Integrator", whose output is the MTVC CMD signal.

For the MTVC/RATE CMD mode (MTVC with rate damping) the DC rate signal is
summed with the RC command signal at the "MTVC Integrator" input. For
this mode either K30 or K31 is energized at engine ignition, depending upon
which rate source is selected via the BMAG MODE switch. The 400 hz GA
output is then demodulated, amplified and fed to the "MTVC Integrator".

6.5.3   TVC Logic Mechanization

The signals which control the relays and solid state switches of the TVSA
and ECA mechanizations are generated by the logic circuits in those devices.
The logic circuits accept signals from the Control Panel 1 switches, TC CW
switch, SPS Engine ON/OFF Logic (IGN 2) and the Servo Actuator (Motor 1 Fail
Sense). Rather than provide the detail of the actual logic mechanization,
the conditions required to enable the relays and solid state switches of
Figures 6.8 and 6.9 are identified below.

K1 = (GIM DR 2) + GIM DR AUTO (FS + CW)

QS1 = (SCS   CW) + MTVC (SCS + CW)

QS2 = (IGN 2   AUTO TVC) (SCS $\overline{CW}$ + CMC   CW)

QS3 = SCS + CW

QS21 = BMAG UNCAGE

QS28 = RATE 1

QS29 = (RATE 2) + ATT 1 RATE 2)

QS25 = (HIGH RATE) + (IGN 2)

K11 = (IGN 2) (SCS + CW)

QS11 = VCG - LM/CSM

QS12 = VCG - CSM

QS30 = (IGN 2)   (SCS + CW) (RATE CMD + ACCEL CMD) + (SCS) (CW)

QS31 = (IGN 2)   (SCS + CW) (RATE CMD + ACCEL CMD) + (SCS) (CW)

K30 = (IGN 2) (QS29)   (RATE CMD) (SCS + CW) + (AUTO TVC) (SCS) (CW)

K31 = (IGN 2) (RATE 1)   (RATE CMD) (SCS + CW) + (AUTO TVC) (SCS) (CW)

6.5.4   An Intuitive Discussion of SCS Auto TVC Operation

This discussion describes the action of the TVC integrator during a delta V controlled by the SCS Auto TVC mechanization. It is an intuitive discussion of the problem, giving a pictorial representation of the vehicle and gimbal motions and the signal polarity and voltage changes.

During non-thrust periods, the BMAG signal normally opposes an attitude change (through the RCS). Thus, during reaction jet control, if the vehicle pitches up, the pitch BMAG signal commands the negative pitch engines ON to correct the attitude error. If the vehicle drifts left, the yaw BMAG

signal similarly turns on the yaw right jets. Initially (during SPS gimbal control), if the vehicle pitches down, the pitch BMAG similarly causes the pitch gimbal to move so as to produce a vehicle pitch up correction.

For example, during a long ΔV maneuver, in order to maintain an inertially fixed thrust vector, it is necessary to generate and maintain a pitch up gimbal command after the vehicle has actually been rotated to a pitch up attitude. The integrator generates this command by producing a pitch up signal and thus maintains a pitch up SPS gimbal deflection. The following discussion will develop the above conclusions.

The following ground rules and relationships are assumed:

1. The discussion is limited to the pitch axis only, since yaw and pitch SCS AUTO TVC functions are identical.

2. Initially, the center of gravity is assumed to be on the "X" axis of the vehicle, and the SPS pitch gimbal angle (trim) is zero degrees.

3. The rate gyro input is not included in this discussion, since its function is merely to provide the vehicle rate damping signal to the servo loop. The TVC integrator acts only on the position feedback and BMAG attitude signals.

4. SPS gimbal rate is not included in this discussion either. Its purpose is to provide gimbal rate damping for the servo loop.

5. Prior to thrusting, the pitch BMAG output is zero. A nose down vehicle attitude change produces a positive (+) BMAG output, and a nose up attitude change produces a negative (-) BMAG output.

6. "Nose up gimbal" means the SPS gimbal has driven in a direction to cause the vehicle to pitch in attitude.

7. A negative (-) input to the servo amplifier produces a nose up gimbal drive, and a positive (+) input produces a nose down gimbal drive.

8. Nose up gimbal results in a negative (-) position feedback signal at the output of the position feedback DC amp, and a nose down gimbal results in positive (+) position feedback signal.

Discussion

During long ΔV's (TEI), the center of gravity moves due to SPS propellant consumption. In order to maintain an inertially fixed thrust vector (i.e., to continue thrusting in the same inertial direction), two basic requirements must be satisfied:

1. The resultant thrust vector must be oriented through the vehicle cg.

2. The vehicle pitch attitude must change if the thrust line from the gimbal hinge point thru the vehicle cg is to remain parallel to the original flight path.

Under these conditions, the cross axis acceleration is driven to zero, and the remaining cross axis velocity will be within trajectory requirements. These end items, illustrated in Figure 6.10, are achieved by the SCS AUTO TVC loop during a SPS burn.

Thus, tracking the cg with the thrust vector is not the total solution to the problem, nor is it desireable to simply hold vehicle attitude with SPS

S/C ATTITUDE & GIMBAL ANGLES FOR AN SCS AUTO TVC MANEUVER

INITIAL CONFIGURATION
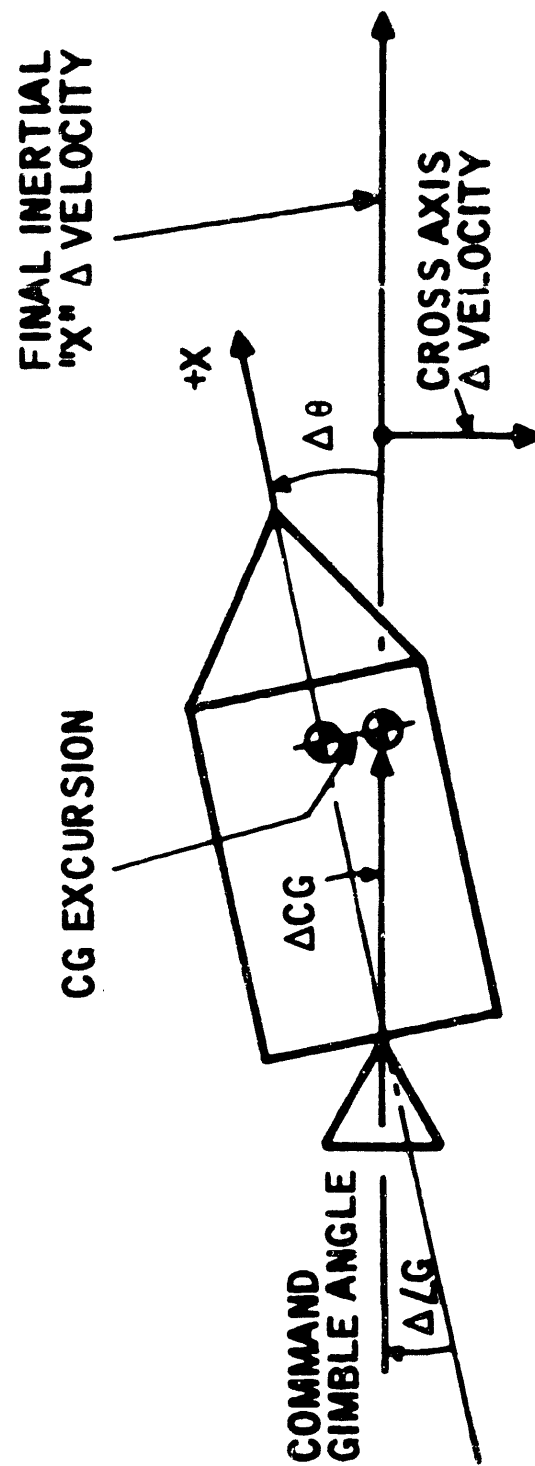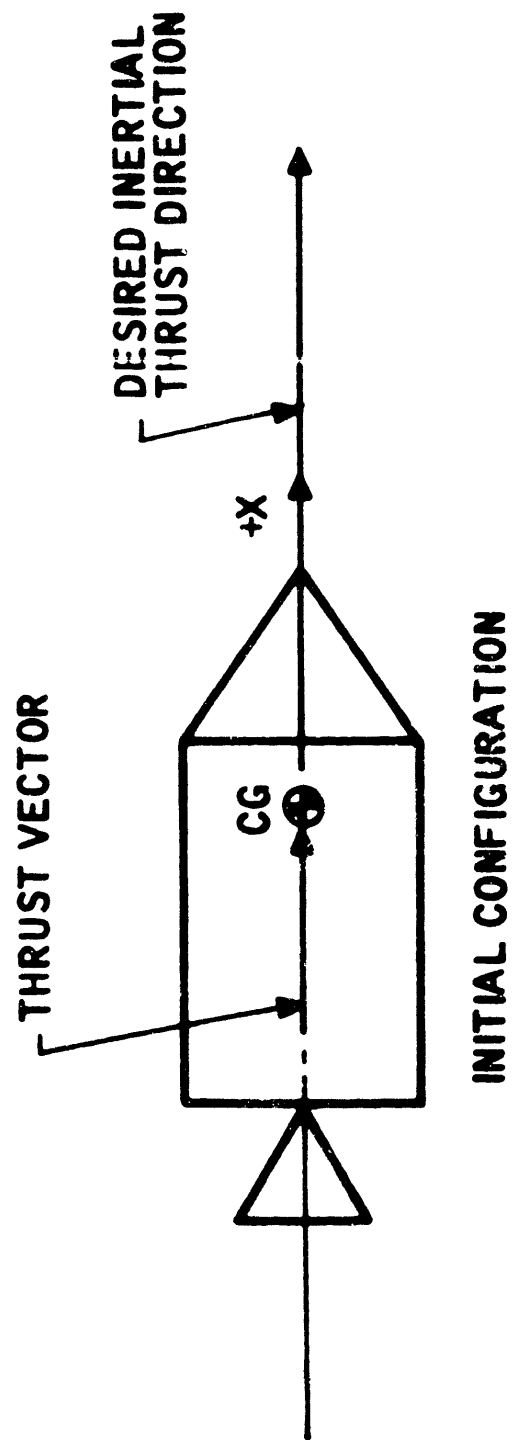
$\Delta CG = \Delta\theta = \Delta\angle G$

FINAL CONFIGURATION

FIGURE 6.10

gimbal control. Both would result in an unwanted change in direction of the inertial thrust vector and subsequent trajectory.

Figure 6.11 illustrates a block diagram of the pitch axis SCS AUTO TVC mechanization. The polarities shown are those resulting from a downward movement of the S/C cg (see step 3). The sequence of events in terms of S/C attitude changes and the TVC loop operation is described in the following eight steps. The S/C attitude and gimbal angle for each step is given in Figure 6.12.

(1) Initially, the cg is on the X-axis and thrust is directed thru the cg. Thus $\Delta\theta$ and $\Delta\angle G$ are zero and all voltages on the block diagram are at null.

(2) The cg moves downward, resulting in a vehicle nose down moment arm. The vehicle thus begins to pitch nose down.

(3) The BMAG develops a (+) output; the gimbal begins to drive to command nose up; polarities develop as shown in Figure 6.11. The integrator input is (-) due to the demodulated BMAG and gimbal position feedback signals. The integrator output is (+) and sums with the (+) DC amp output. The gimbal therefore continues driving in the vehicle nose up command direction.

(4) The thrust vector is now below the cg, creating a vehicle nose up moment arm. Thus the vehicle stops pitching down and starts pitching nose up. The BMAG output goes less (+), but the gimbal continues to drive nose up.
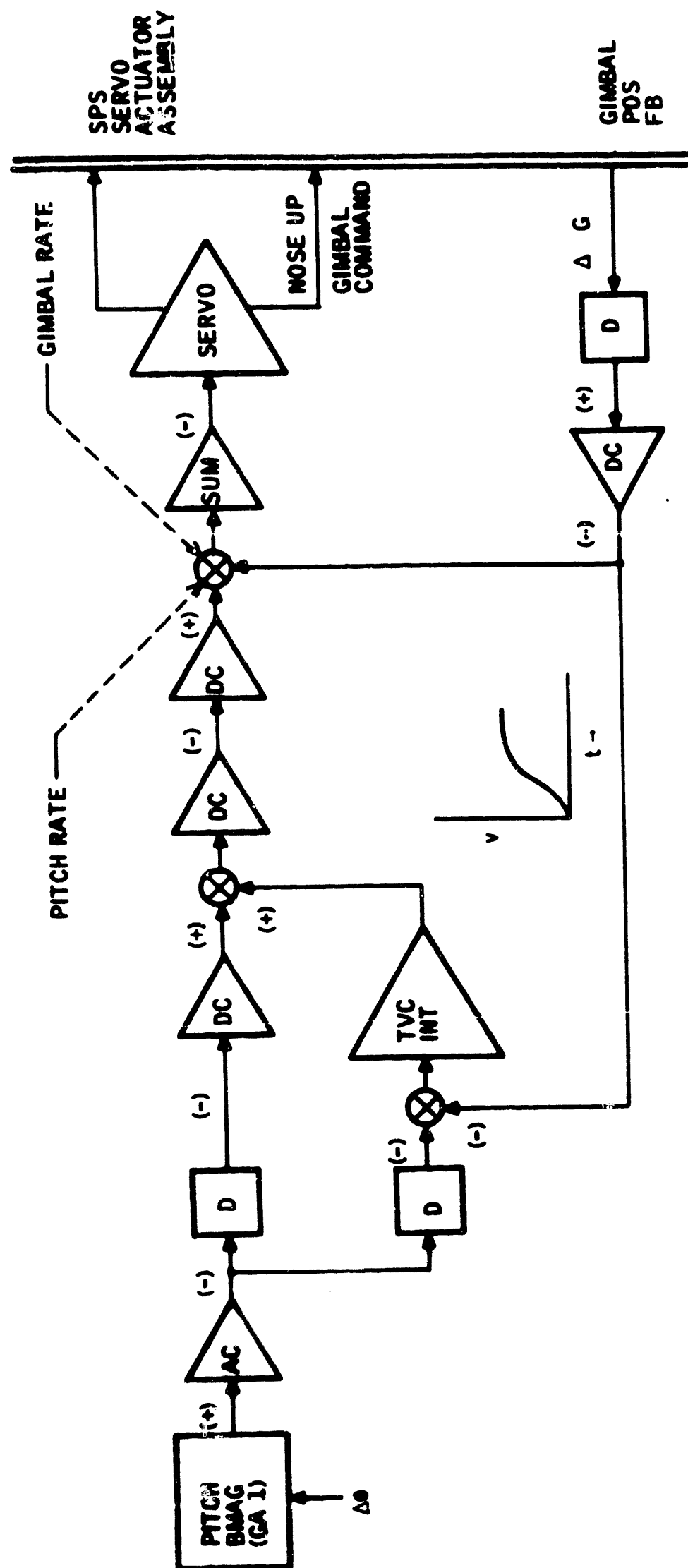
SCS AUTO TVC BLOCK DIAGRAM

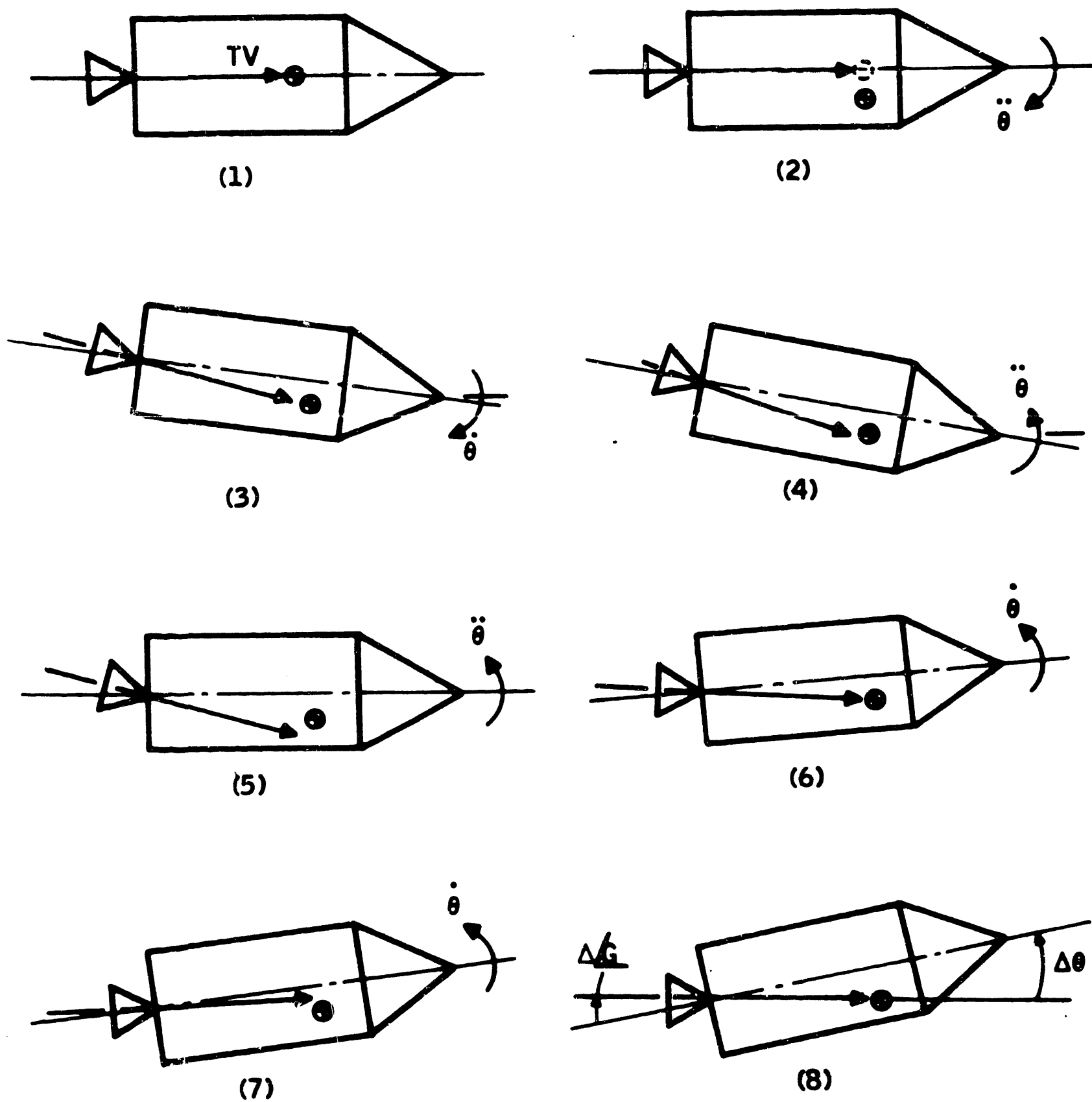FIGURE 6.11

6-39

S/C & GIMBAL MOTIONS DUE TO CG SHIFT

FIGURE 6.12

(5) The BMAG output goes to zero when the original pitch attitude is achieved, but the integrator is still receiving a (-) GPFB signal and gimbal continues to drive nose up. Thus the vehicle continues to pitch up.

(6) As the vehicle pitches beyond the initial reference attitude, the BMAG output becomes (-), the AC amp output goes (+) and subtracts from GPFB (-) signal at integrator input.

(7) While the vehicle is still pitching up, the gimbal is driven to provide the line of thrust above the cg. This reduces the vehicle pitch rate, which goes to zero as the gimbal is positioned to provide thrust thru the cg.

(8) The gimbal motion stops when the GPFB (-) input to the TVC summing amp is nulled by the (+) signal generated by BMAG (-) output and the integrator (+) output, the latter signal predominating. The integrator output stops increasing when GPFB (-) signal cancels the demodulated (+) BMAG signal at the integrator input.

The end result is that the vehicle is in a nose up attitude and the SPS gimbal is also in a nose up command position. Note that the gimbal has been in a nose up position throughout almost all of the maneuver but the vehicle went through a nose down and then nose up attitude change. When the vehicle initially pitched down, the (+) BMAG output produced a nose up gimbal drive. The integrator output level then kept the gimbal from moving to a nose down position as the vehicle pitched back up through the initial reference attitude.

A (+) signal was needed to cancel a (-) position feedback signal to hold the gimbal in a nose up spacecraft attitude position. The vehicle attitude resulted in establishing the thrust vector parallel to the original inertial flight path. However, since the thrust vector was never above the desired thrust direction, a vehicle cross axis velocity still remains. Actually no SCS sensor is available to measure this velocity, so no corresponding commands can be generated to drive the cross axis velocity to zero. If the cross axis velocity is minimal, the trajectory errors can be defined within an accuracy cone to meet the Apollo mission requirements.

6.5.5   Gimbal Position Display Mechanization

The electronics used to control the GP/FPI indicators is contained in the EDA. Each of the four indicators is controlled by a separate circuit. Figure 6.13 is a block diagram which describes the mechanization of the indicator control circuits. The drive circuit for an indicator must be capable of accepting a LV tank pressure signal or a gimbal position signal and providing the proper control signal to the servometric drive motor for each. A zero psi tank pressure signal is 0 VDC as is a zero degree gimbal angle signal. However the pointer must be driven to the bottom of the scale for the former and to mid-scale for the latter null signal. This is accomplished by providing a different zero reference bias voltage to the motor driver for the tank pressure and gimbal position indication modes.

The motor driver command signal is received through the closed K62 relay contacts. The motor driver provides an output current to the servometric drive motor proportional to the magnitude of its input error signal. (The
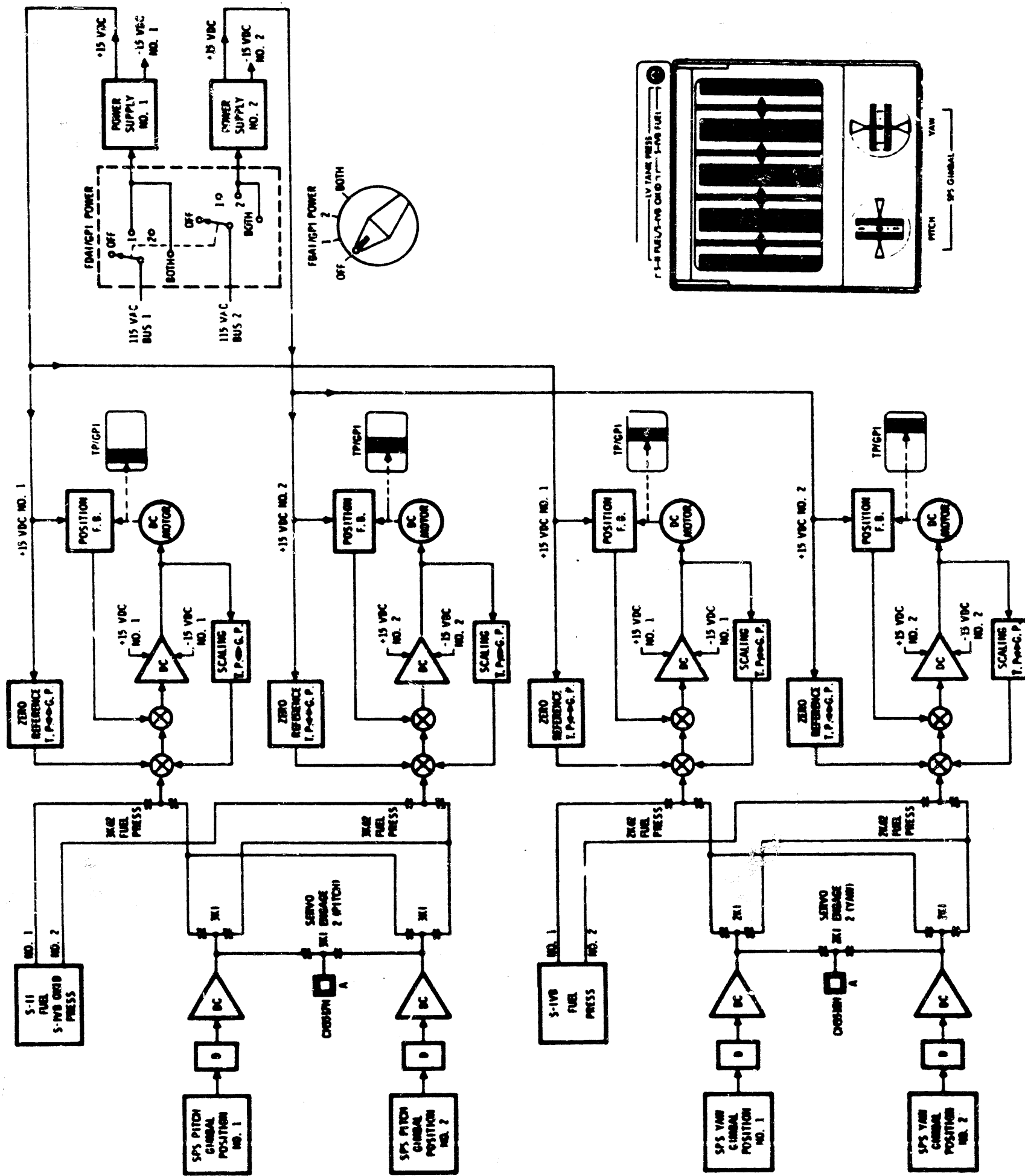
GPI SIGNAL FLOW

FIGURE 6.13

gain and dynamic response of the motor driver is altered as a function of the display mode.) The indicator pointer and the feedback potentiometer wiper are positioned by the drive motor. The wiper voltage is fed back to the motor driver summing point. The motor is driven until the voltage at the summing point is nulled.

The K62 relays are shown in the state required to display gimbal position. The K1 relays' state is shown for the primary servo loop selected. Thus the No. 1 pitch gimbal transducer provides the command signal for the 1 and 2 pitch indicators. The 1 and 2 yaw indicators are driven by the No. 1 yaw gimbal transducer output.

Power is supplied to the 1 and 2 indicator electronics by separate power supplies. The power supplies receive excitation through the FDAI/GPI POWER switch as indicated in Figure 6.13.
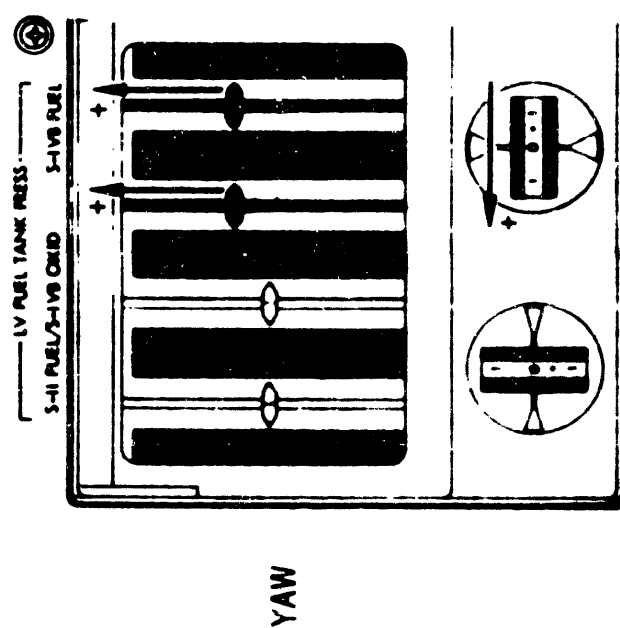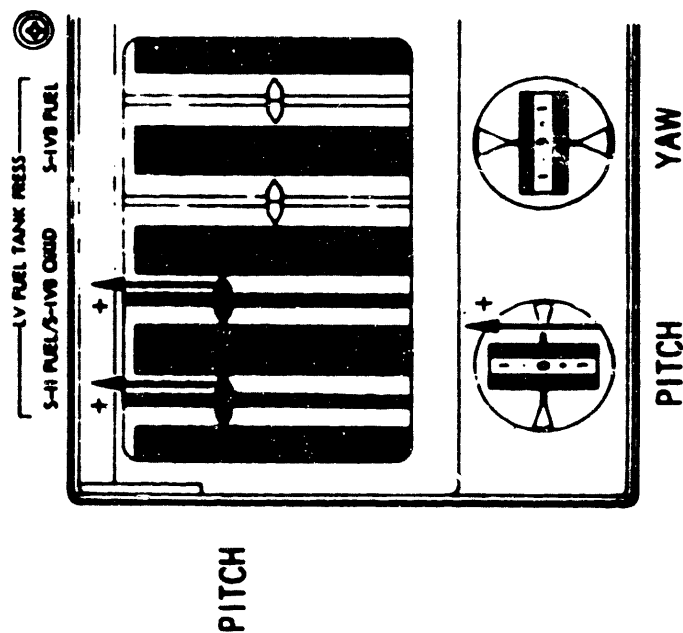
6.5.6   **Gimbal Command, Motion & Display Relationships**

The null position of the SPS engine thrust direction is offset from the spacecraft + X axis. The pitch actuator null position is +2° thrust vector to the + Z axis. While the yaw actuator null position is +1° thrust vector to the + Y axis. The gimbal position of the pitch and yaw gimbals relative to their null positions.

Figure 6.14 describes the relationships between gimbal commands, motions and displays and the resultant spacecraft motion and FDAI displays. Rotating the pitch thumbwheel upward causes the thrust vector to move toward the -Z axis (which would result in a pitch down command if the SPS engine were ON).

SPS GIMBALLING

CSM ROTATIONS

PITCH DOWN

+X

-Z

+Y

+Z

T.V.

-X

+PITCH GIMBALLING DEFLECTION

+

YAW LEFT

+X

-Y

-Z

+Y

T.V.

-X

+YAW GIMBALLING DEFLECTION

MTVC COMMANDS

PITCH DOWN

YAW LEFT

GPI AND TRIM THUMBWHEELS

LV FUEL TANK PRESS
S-II FUEL/S-IVB OXID   S-IVB FUEL

PITCH

YAW

PITCH

LV FUEL TANK PRESS
S-II FUEL/S-IVB OXID   S-IVB FUEL

YAW

FIGURE 6.14

6-45
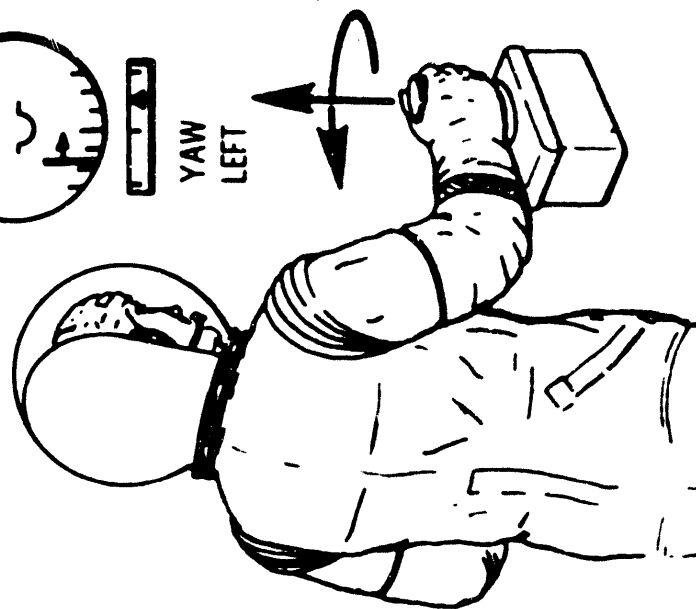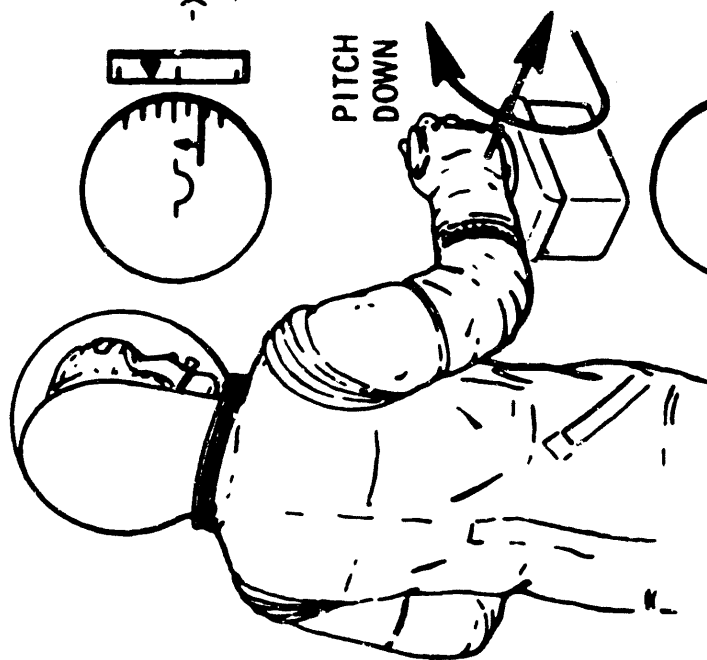
Positioning the pitch thumbwheel to +2 causes the pitch actuator to be displaced 2° from null, yielding a pitch gimbal display of +2. Selection of an SCS MTVC mode allows the astronaut to position the SPS engine 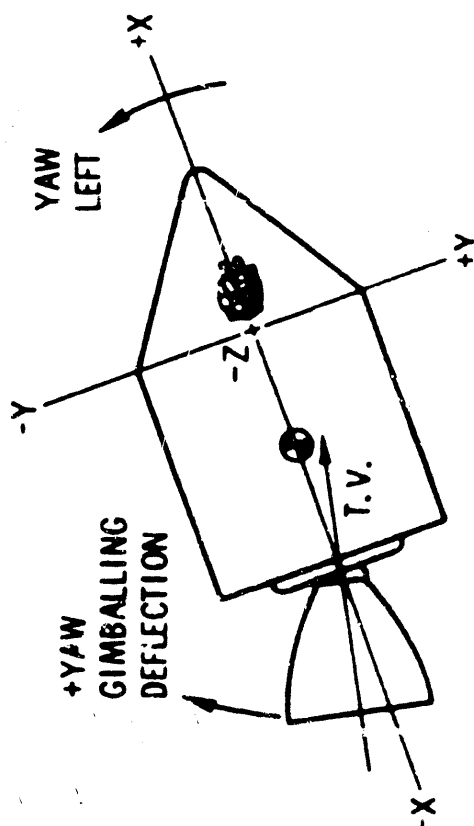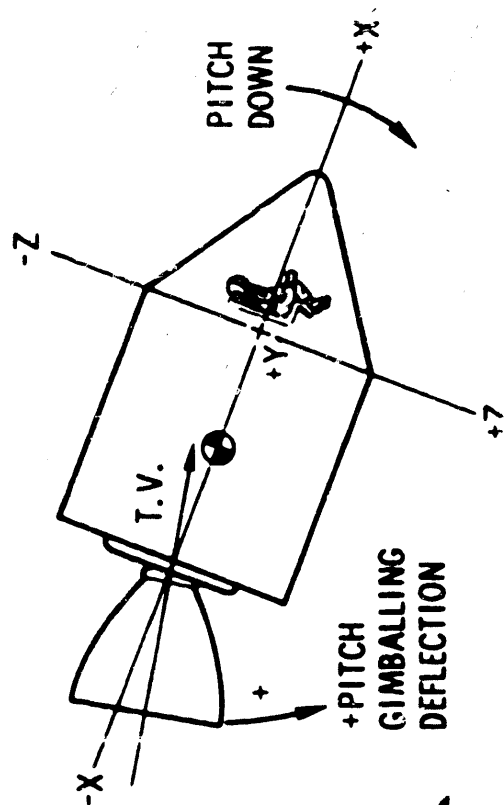gimbals (with or without engine ignition) via the RC. He can insert a pitch down command by rotating the RC stick forward. This results in a + pitch gimbal deflection and the position of the gimbal will be displayed on the GPI. If the spacecraft is actually pitching down, the FDAI rate indicator will display a positive rate (pointer above the null position) while the attitude error indicator will move in the positive (upward) direction.

As indicated in Figure 6.14, positive yaw gimbal trim commands are inserted by rotating the thumbwheel toward the left. This causes gimbal displacement, moving the thrust vector toward the + Y axis (which results in a yaw left command if the SPS engine is ON). Positive yaw gimbal displacement results in upward movement of the yaw GPI indicators. Counterclockwise rotation of the RC stick generates a + yaw gimbal command when an SCS MTVC mode is selected. If the spacecraft is actually yawing left, the FDAI rate indicator will be to the right of null and the attitude error indicator will be moving toward the right.

SECTION 7

POWER DISTRIBUTION

7.1   **SCS DEVICE POWER**

The SCS sensors and electronic assemblies receive AC and DC power through

the control panel No. 7 power switches as shown in Figure 7.1. Figure

7.2 shows the configuration of control panel No. 7. The SCS circuit

breakers on control panel No. 8 (Figure 7.3) supply electrical power to

the power switches.

7.2   **HAND CONTROL POWER**

The distribution of power to the Rotation Controls and the Translation

Control is shown in Figure 7.4. The five hand control power switches

are located on control panel No. 1.

7.3   **SCS LOGIC BUS POWER**

The four SCS logic busses receive 28V DC MN A and MN B power through circuit

breakers on control panel No. 8 as shown in Figure 7.5. Power to logic

busses 2 and 3 is controlled by the LOGIC POWER 2/3 switch on control panel

No. 7. The control panel No. 1 switches which are powered by the individual

logic busses are identified on Figure 7.5.

SCS SWITCHED POWER DISTRIBUTION

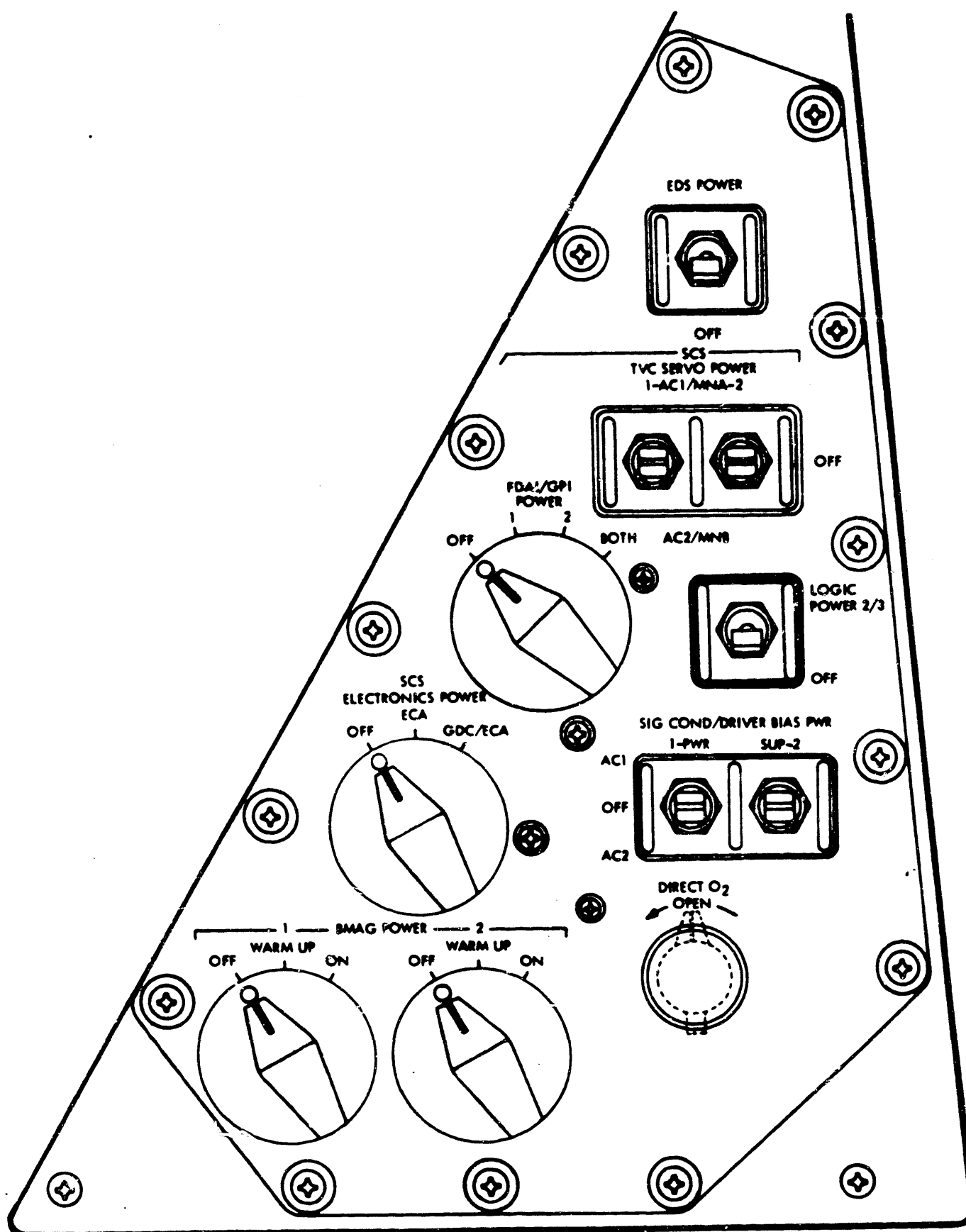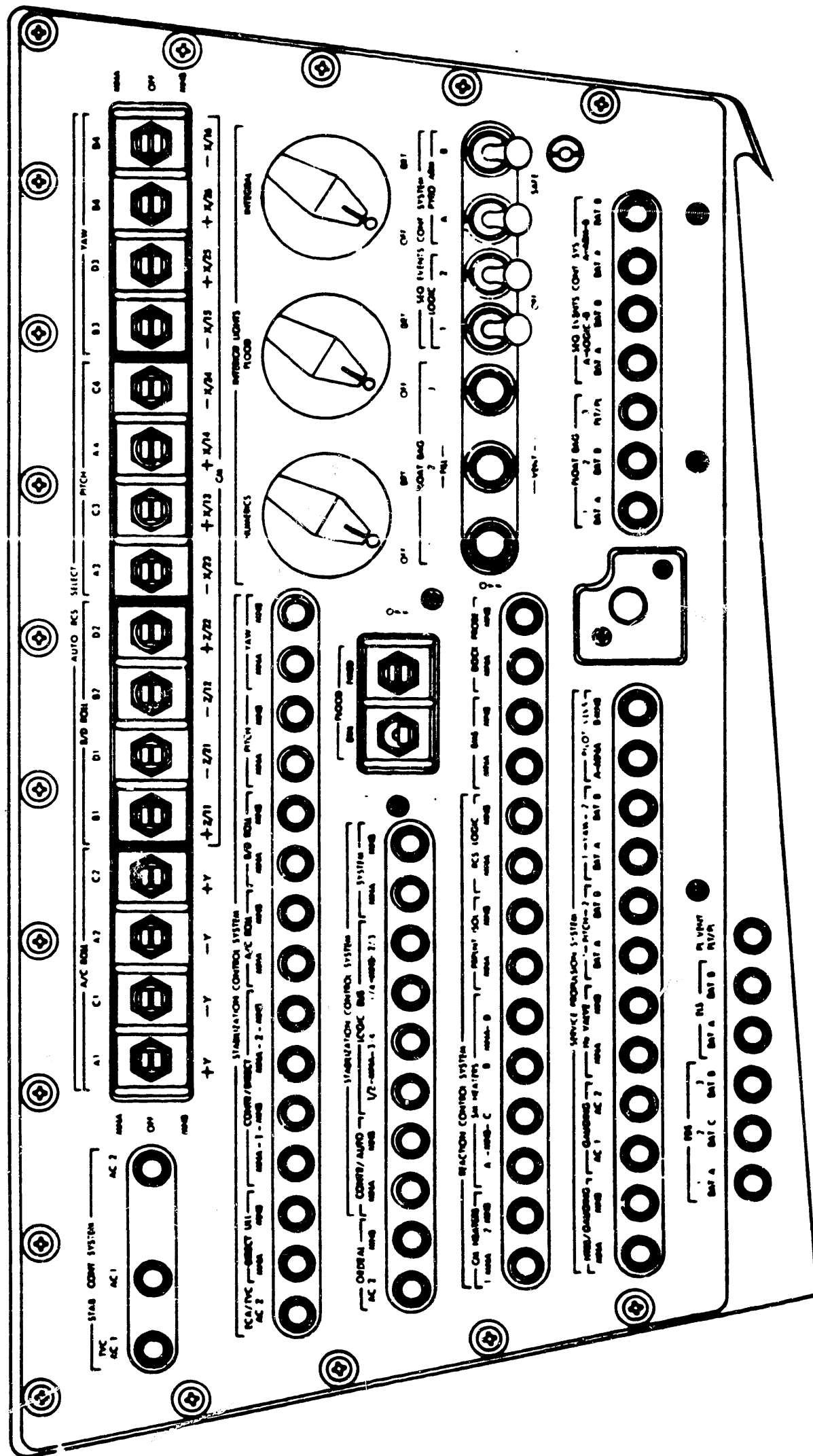FIGURE 7.1

7-2

CONTROL PANEL NO. 7

FIGURE 7.2

CONTROL PANEL NO. 8

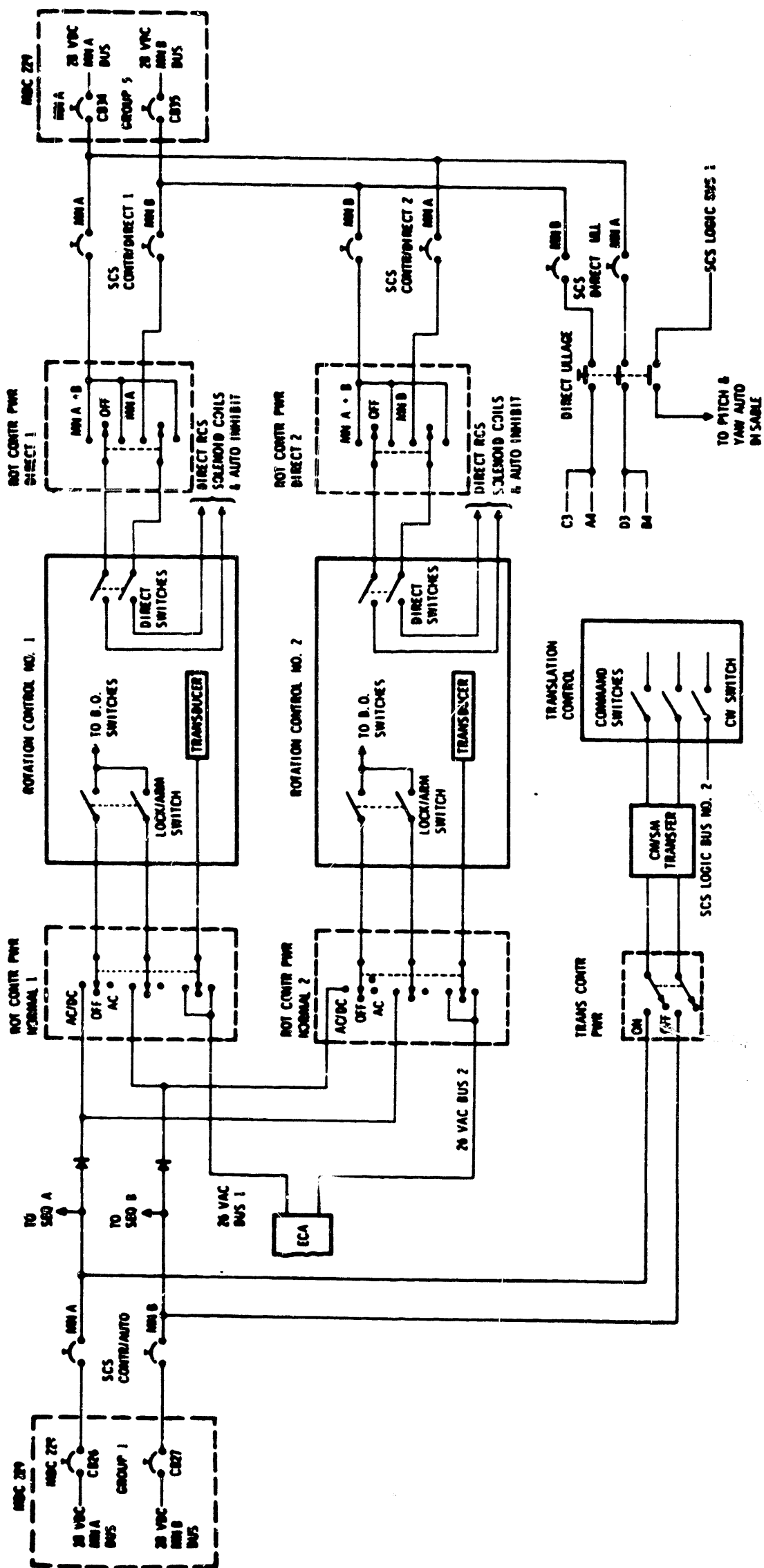FIGURE 7.3

POWER SWITCHING
ROTATION & TRANSLATION CONTROLS
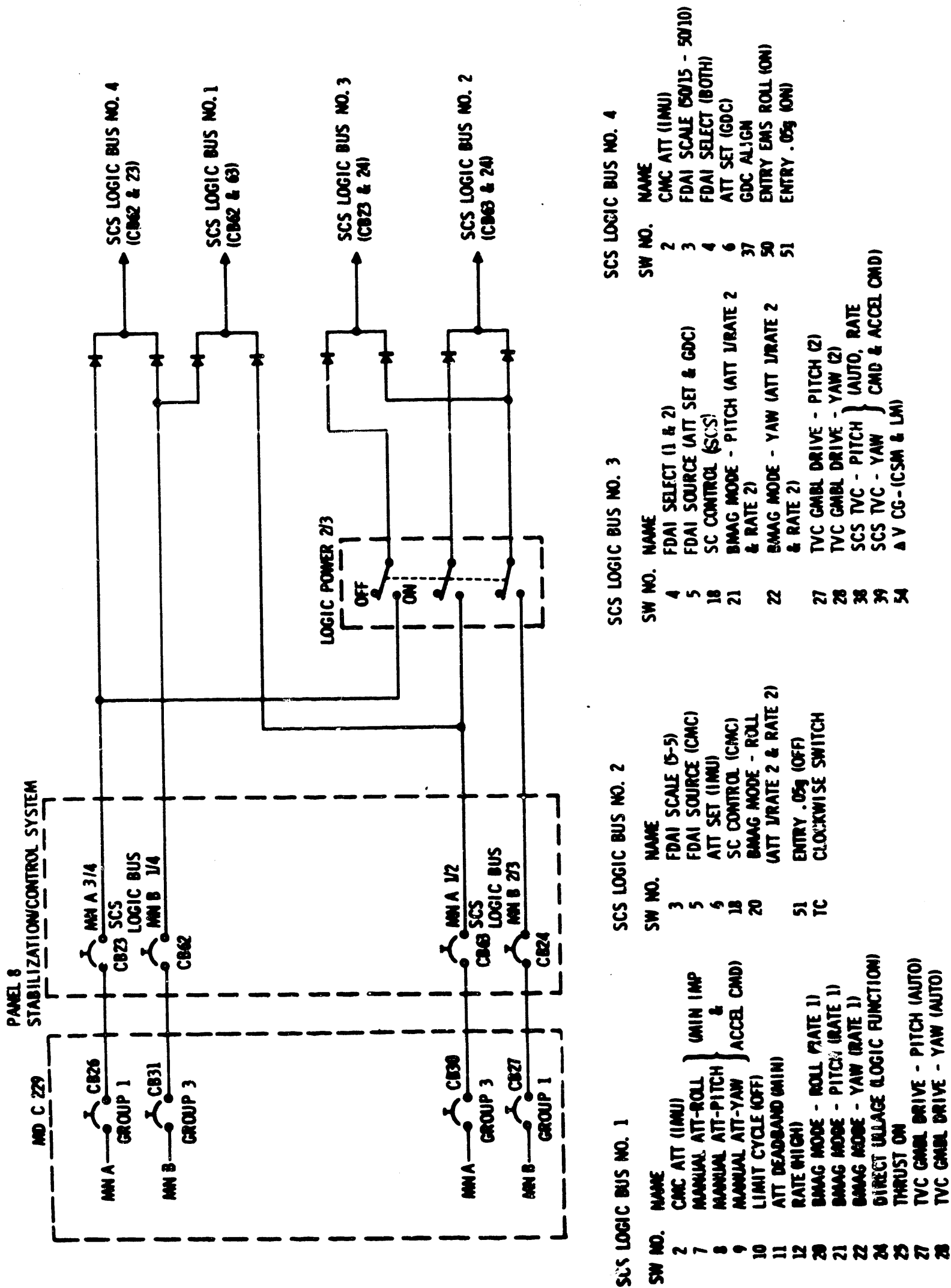

FIGURE 7.4

7-5

SCS LOGIC BUS POWER DISTRIBUTION

MD C 229

MN A — CB26 GROUP 1
MN B — CB31 GROUP 3
MN A — CB30 GROUP 3
MN B — CB27 GROUP 1

PANEL 8
STABILIZATION/CONTROL SYSTEM

MN A 3/4 — CB23 SCS LOGIC BUS
MN B 1/4 — CB62

MN A 1/2 — CB63 SCS LOGIC BUS
MN B 2/3 — CB24

LOGIC POWER 2/3
OFF / ON

SCS LOGIC BUS NO. 4 (CB62 & 23)
SCS LOGIC BUS NO. 1 (CB62 & 63)
SCS LOGIC BUS NO. 3 (CB23 & 24)
SCS LOGIC BUS NO. 2 (CB63 & 24)

SCS LOGIC BUS NO. 1

SW NO.	NAME
2	CMC ATT (IMU)
7	MANUAL ATT-ROLL (MIN IMP
8	MANUAL ATT-PITCH &
?	MANUAL ATT-YAW ACCEL CMD)
9	LIMIT CYCLE (OFF)
10	ATT DEADBAND (MIN)
11	RATE (HIGH)
12	
20	BMAG MODE - ROLL (RATE 1)
21	BMAG MODE - PITCH (RATE 1)
22	BMAG MODE - YAW (RATE 1)
24	DIRECT ULLAGE (LOGIC FUNCTION)
25	THRUST ON
27	TVC GMBL DRIVE - PITCH (AUTO)
28	TVC GMBL DRIVE - YAW (AUTO)

SCS LOGIC BUS NO. 2

SW NO.	NAME
3	FDAI SCALE (5-5)
5	FDAI SOURCE (CMC)
18	ATT SET (IMU)
20	SC CONTROL (CMC)
	BMAG MODE - ROLL (ATT 1/RATE 2 & RATE 2)
51	ENTRY .05g (OFF)
TC	CLOCKWISE SWITCH

SCS LOGIC BUS NO. 3

SW NO.	NAME
4	FDAI SELECT (1 & 2)
5	FDAI SOURCE (ATT SET & GDC)
18	SC CONTROL (SCS)
21	BMAG MODE - PITCH (ATT 1/RATE 2 & RATE 2)
22	BMAG MODE - YAW (ATT 1/RATE 2 & RATE 2)
27	TVC GMBL DRIVE - PITCH (2)
28	TVC GMBL DRIVE - YAW (2)
38	SCS TVC - PITCH (AUTO, RATE
39	SCS TVC - YAW CMD & ACCEL CMD)
54	ΔV CG - (CSM & LM)

SCS LOGIC BUS NO. 4

SW NO.	NAME
2	CMC ATT (IMU)
3	FDAI SCALE (50/15 - 50/10)
4	FDAI SELECT (BOTH)
6	ATT SET (GDC)
37	GDC ALIGN
50	ENTRY EMS ROLL (ON)
51	ENTRY .05g (ON)

FIGURE 7.5

7-6

## GG248 Miniature Integrating Gyro Theory of Operation

The MIG is a floated, single-degree-of-freedom, rate integrating gyro. It is effectively a torque summing computer. Input turning rates are converted by the gyroscopic element into gimbal torques. These torques are time integrated through the viscous fluid into gimbal displacements which are read off directly by a displacement type pickoff. The major elements of the MIG are the gyro wheel, gimbal assembly, signal generator and torque generator. See Figure A.1.

Gyro Wheel - The basic element of the MIG (and all gyroscopic instruments) is a wheel whose mass is distributed to provide a specific moment of inertia (I) about its axis of rotation, called the spin axis (SA). The wheel is actually the rotor of a synchronous motor. By applying an excitation voltage to the motor stator (located inside the rotor) the wheel is forced to rotate at a constant angular rate (W_s). The spinning wheel has an angular momentum (H) which is the product of its moment of inertia and its angular velocity.

$$H = IW_s$$

H is a vector quantity (i.e. it has magnitude and direction). The direction of the angular momentum is along the spin axis. A gyro functions by obeying the law of Conservation of Momentum. The spinning wheel is inherently stable. It will continue to rotate with the spin axis rigidly fixed in space unless a torque is applied to tilt the spin axis. The angular momentum is a measure of this stability. In the GG248 MIG the moment of inertia is

GG 248 CUTAWAY VIEW
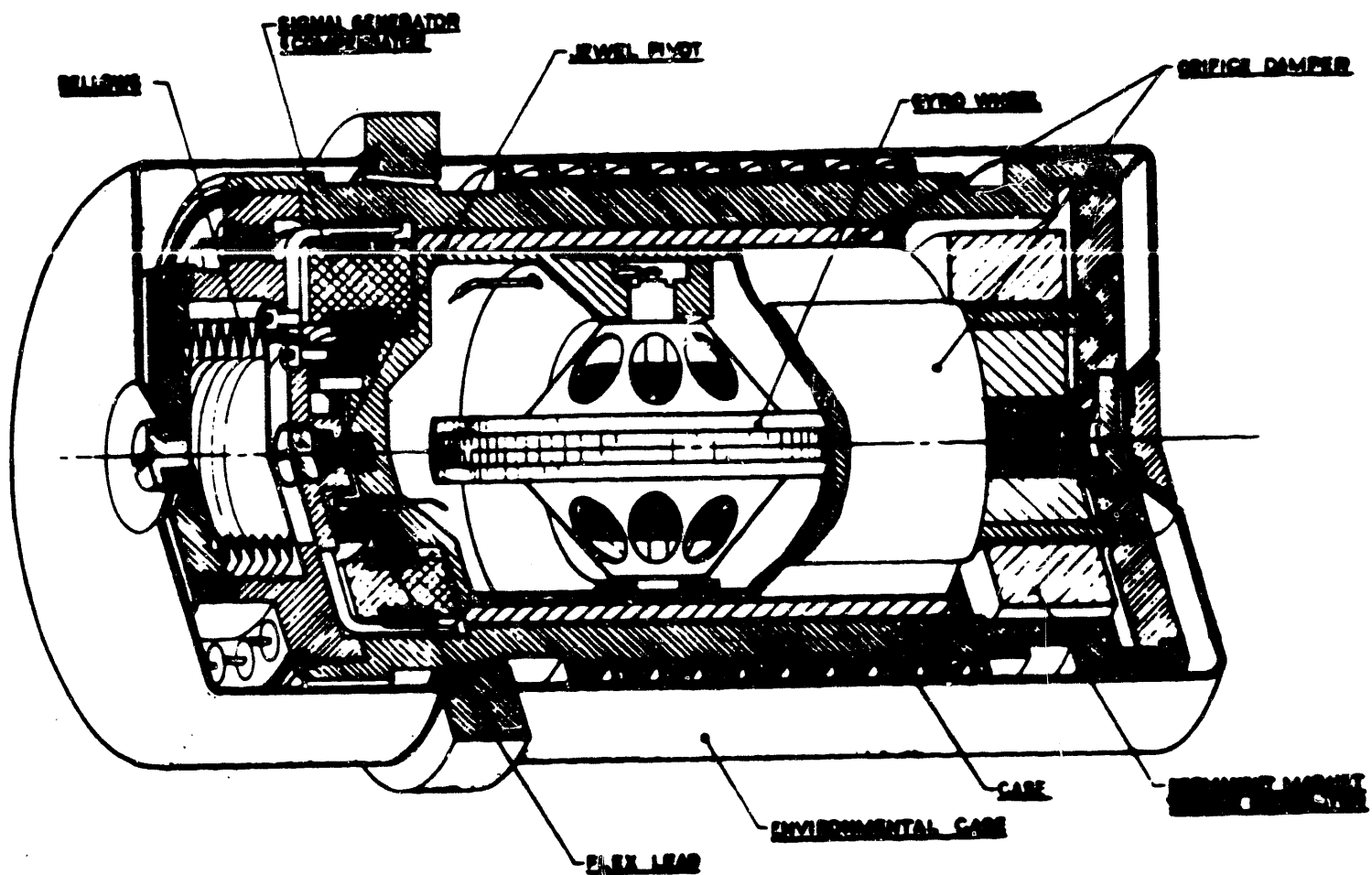
NOTE: The environmental case is not used on the Apollo Block II GG248 gyro.

FIGURE A.1

about 40 gm-cm^2. The angular velocity is 24,000 rpm or about 2500 radians per second. Hence the angular momentum of the spinning gyro wheel is 1 x 10^5 gm-cm^2/second.

Gimbal Assembly - The gyro wheel rotates in the inert gas atmosphere of the sealed cylindrical gimbal assembly. The gimbal cylinder is floated in a fluid which surrounds and supports it. Pivots and jewels support the entire gimbal weight during assembly prior to filling with fluid. With full loading and no lubrication, these pivots and jewels cause less than 10 dyne-cm of stiction (compared to a minimum stiction of 20 dyne-cm with conventional ball bearings). The fluid used has a density within 1% of the gimbal assembly average density at the operating temperature of the gyro. This reduces the stiction to the order of 0.01 dyne-cm. (0.01 dyne-cm is about one billionth of a foot pound, or the level of torque developed by shining a flashlight on a sheet of paper hinged at one edge). The basic function of the pivot supports is to define the output axis (OA), i.e. to prevent any motion of the gimbal cylinder relative to the case except rotations about the axis defined by the pivots. Figure A.2 is a simplified drawing showing the relationship between the gyro wheel, gimbal and gyro case as well as the spin, input and output axes. With the gyro wheel rotating on its shaft in the direction shown, the positive spin axis is defined as indicated by the arrow. (use right hand rule - wrap fingers around the wheel in the direction of rotation; the thumb then points along the positive spin axis.) The gimbal cylinder containing the gyro wheel is free to rotate only about the axis defined by the gimbal pivots, the output
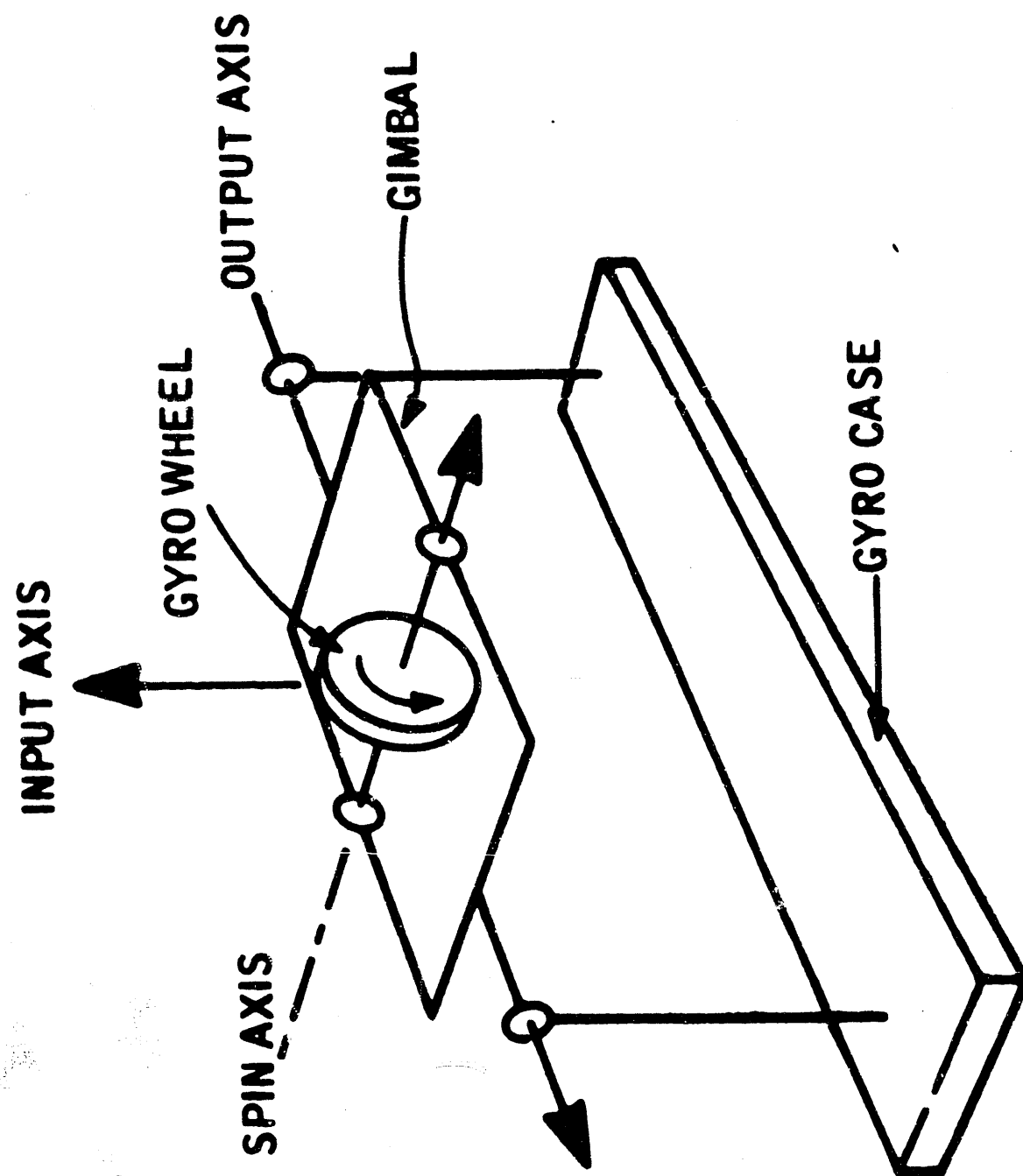
BASIC GYRO ELEMENTS

OUTPUT AXIS

GIMBAL

GYRO WHEEL

INPUT AXIS

SPIN AXIS

GYRO CASE

FIGURE A.2

A1-4

axis (OA). The positive output axis is defined arbitrarily, but determines the direction of the positive input axis. It is obvious that the spin axis is orthogonal to the output axis (axis of the gimbal cylinder). The input axis (IA) is orthogonal to both the OA and SA and is also defined by the right hand rule: Grasp the gimbal cylinder with the right hand such that the thumb points along the +OA and rotate the gimbal in the direction of the fingers 90 degrees; the +SA axis then points in the direction of the original +IA. (Note however that the actual IA is displaced with the SA since the three axes are always orthogonal). Now if the gyro case is rotated CW about the IA, the spin axis is displaced and the angular momentum (H) of the spinning wheel is disturbed. Recall (1) that H is a vector quantity and (2) the law of conservation of angular momentum. By rotating the gyro case about the input axis we are actually inserting a component of angular momentum along the IA. To conserve angular momentum the wheel should spin along an axis between, and in the plane of, the SA and IA. Thus a positive rotation of the gyro case about the input axis results in a gyroscopic torque on the gimbal causing a positive rotation about the output axis. This displaced rotation of the gimbal is called gyroscopic precession. It should be apparent that the gyroscopic precession rate is directly proportional to the input rate and also the angular momentum of the wheel (H). We will now consider the restraining torque which controls the precession rate.

Viscous Damper - In addition to its high-density property, the fluid between the gimbal assembly and the case of the MIG is highly viscous. Consequently,

as the gimbal precesses in the fluid, shearing forces are developed at the gimbal surface proportional to the gimbal angular velocity. The fluid thus provides viscous damping of the gimbal motion. The symbol used to represent viscous damping is "C". C is measured in the same units as angular momentum (H) and in the GG248 has a magnitude of 460,000 gm-cm^2/sec at the normal gyro operating temperature. The precession rate of the gimbal (Wg) is equal to the input rate ($\dot{\theta}$) multiplied by the ratio of the gyro wheel angular momentum to the viscous damping on the gimbal assembly:

$$Wg = \frac{H}{C} \dot{\theta};$$

for the GG248 MIG $\frac{H}{C}$ is equal to 0.217. Thus if the gyro case is rotated about the input axis at a rate of 1°/sec for 10 seconds, the gimbal will be displaced 2.17 degrees. Gimbal displacement is limited to $\pm$4.4^o by mechanical stops.

Signal Generator - Now if we can measure the displacement of the gimbal from its null position, we can tell how many degrees the gyro case was rotated about the input axis. This capability is provided by the signal generator (SG) which is a rotor-stator arrangement providing an inductive pickoff. The SG primary is excited with a 400 hz voltage; the secondary provides a 400 hz voltage whose magnitude is proportional to the gimbal displacement and phase depends on the direction of displacement. The signal generator thus provides an electrical signal which describes the position of the gimbal cylinder relative to the output axis.

Torque Generator - The basic gyro elements previously described provide the capability for measuring changes in the gyro position about its input axis.

The torque generator is an electromagnetic device that, in response to an external command signal, will change the angular position of the gyro gimbal. The torque generator is used to cage (move and lock the spin axis to some desired orientation) the gyro before or after maneuvers to establish or re-establish a reference. Command signals originating at the signal generator can be applied to the torque generator to effectively cancel gimbal precession and thus measure input _rates_ accurately without permitting excessive gimbal displacement.

Functional Operation - The functional operation of the MIG can be described with the aid of the simplified line drawing of Figure A.3. The four essential functional parts are the gyroscopic element (the spinning gyro wheel and the gimbal assembly), the viscous damper (C), the signal generator (SG) and the torque generator (TG). The viscous damper is shown connected to the output axis such that as the gyro wheel precesses a viscous torque is developed. As explained previously, the precession rate of the gimbal assembly is directly proportional to the rotational rate of the gyro case and the ratio of the gyro wheel angular momentum (H) to the amount of viscous damping (C). The signal generator (SG) provides a 400 hz output voltage proportional to the angle of gimbal rotation. The torque generator (TG) provides the capability to torque the gimbal about the output axis inproportion to the magnitude of an externally applied current.

Supplementary Parts - The major supplementary parts of the MIG are the pivots and jewels, heaters, temperature sensors, bellows and flex leads. The pivot and jewel are shown in Figure A.3 with a gap between them to stress the fact

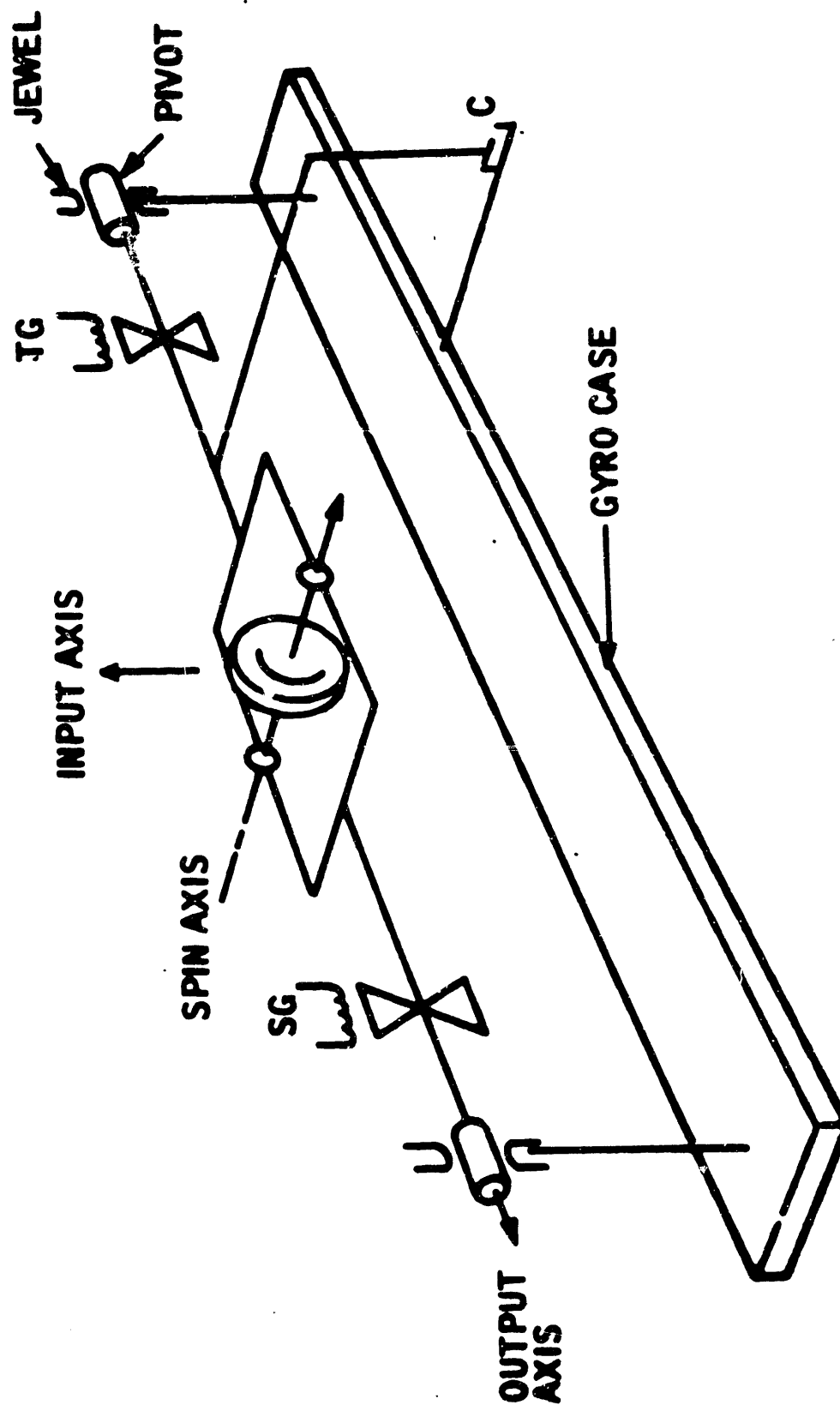# SIMPLIFIED LINE DRAWING OF MIG GYRO

FIGURE A.3

that support is provided by the fluid and not by the pivot and jewel. The nominal clearance between the pivot and the jewel is about 100 micro-inches.

The fluid used for damping and floatation has temperature sensitive density and viscosity characteristics. The GG248 is thus operated at the optimum temperature of 170°F to obtain the proper density and viscosity values. A wire temperature sensing element and a heater wire are wrapped around the barrel of the gyro. The sensing element is connected into the fourth leg of a bridge of a temperature control amplifier which controls power to the gyro heater wire. The gyro temperature is thus controlled to a small fraction of a degree.

A bellows is used to provide a constant positive pressure inside the gyro when the gyro is at operating temperature. It also compensates for the change in fluid volume as the ambient temperature in which the gyro may be stored goes below the operating temperature.

Flex leads are used to deliver power to the spin motor inside the gimbal. The flex leads are small (about .0003 by .004 inches in cross section) with minimum elastic restraint, so that minimal forces are applied to the gimbal from the flex leads. They are enclosed in baffles for protection from forces which occur when the fluid freezes.

APPENDIX 2

Resolver Representation

The input-output relationships of a resolver can be shown by means of a simple diagram. Figure A.4 depicts a resolver schematic and its equivalent representation.

Independent signals A and B are applied to the stator windings S1 - S3 and S2 - S4 respectively. Outputs C and D are taken from the rotor windings R1 - R3 and R2 - R4 respectively. The schematic depicts the resolver shaft at zero degrees and assumes a positive shaft rotation moves the windings in a clockwise direction. The equations describing the outputs C and D are thus:
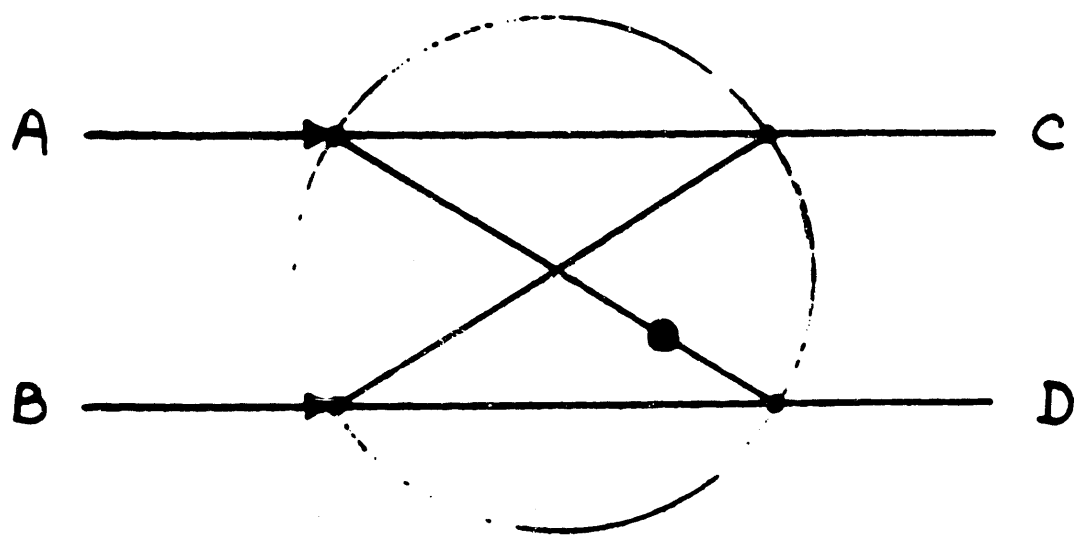
$$C = A \cos \alpha + B \sin \alpha$$

$$\text{and} \quad D = -A \sin \alpha + B \cos \alpha .$$

The equivalent resolver representation describes the same input-output relationship by using the horizontal lines to represent $\cos \alpha$ and the diagonal lines the $\sin \alpha$ term.

A -1 multiplier is indicated by a dot on the applicable line.
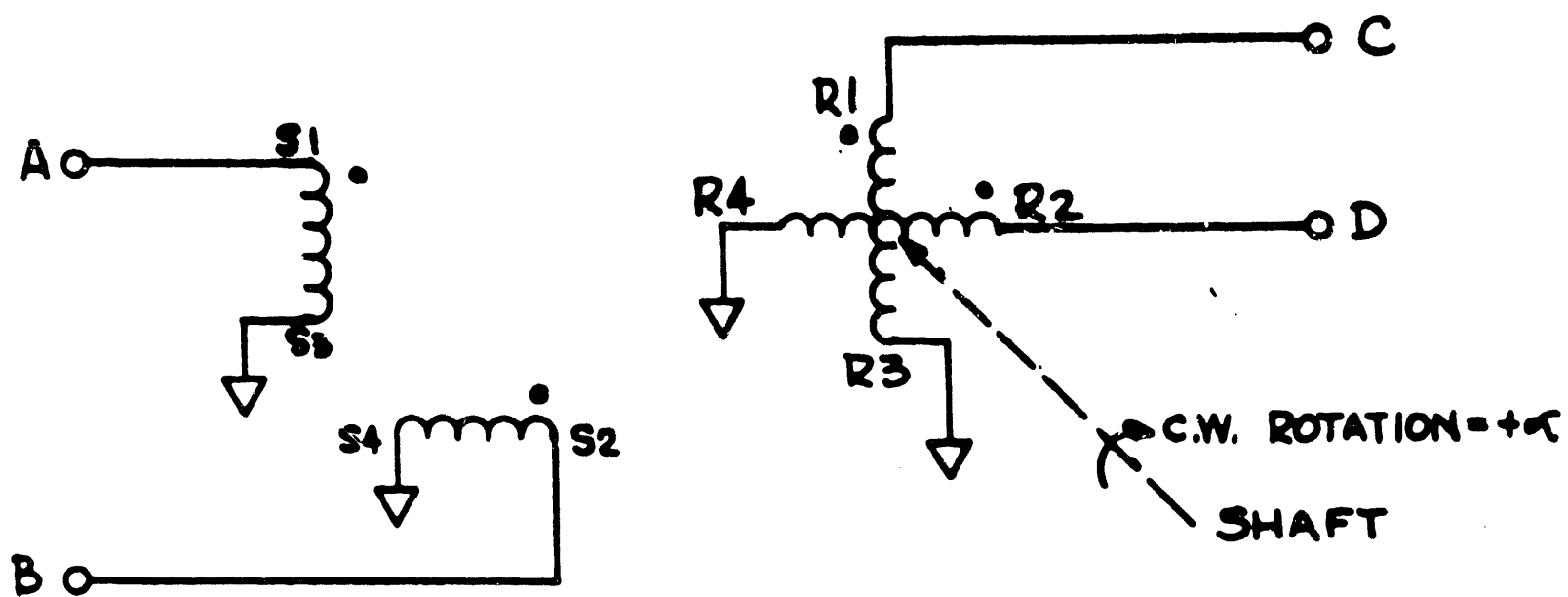
RESOLVER SCHEMATIC

RESOLVER EQUIVALENT REPRESENTATION

FIGURE A.4

APPENDIX 3

ROLL TO YAW COUPLING DURING ENTRY

Yaw Stability Rate Display

During entry the SCS is required to display angular rates about the S/C pitch and roll body axes and the yaw stability axis. It is desireable to display yaw stability rate since roll maneuvers during entry will be accomplished about the roll stability axis which is offset by an angle (nominally 21°) from the roll body axis. Hence a roll maneuver results in yaw and roll body rates. The SCS rate sensors are body mounted gyros and thus provide signals proportional to rates about the body axes. To prevent the sensed yaw body rate from being displayed during a roll maneuver, a modified roll rate signal is coupled into the yaw channel. Figure A.5 identifies the S/C roll body (X_B), roll stability (X_S) and yaw body (Z_B) axes. For a given positive roll stability rate ($\dot{\phi}_S$) the roll gyro will sense $\dot{\phi}_B$ and the yaw gyro will sense $\dot{\psi}_B$ (a negative yaw body rate). From the diagram it is evident that:

$$\left| \dot{\psi}_B \right| = \dot{\phi}_S \sin \alpha$$

and

$$\dot{\phi}_S = \dot{\phi}_B \cos \alpha.$$

Therefore

$$\left| \dot{\psi}_B \right| = \dot{\phi}_B \tan \alpha.$$

Thus the sensed yaw body rate signal is effectively cancelled by summing the roll body rate signal modified by the constant $\tan \alpha$ into the yaw channel.
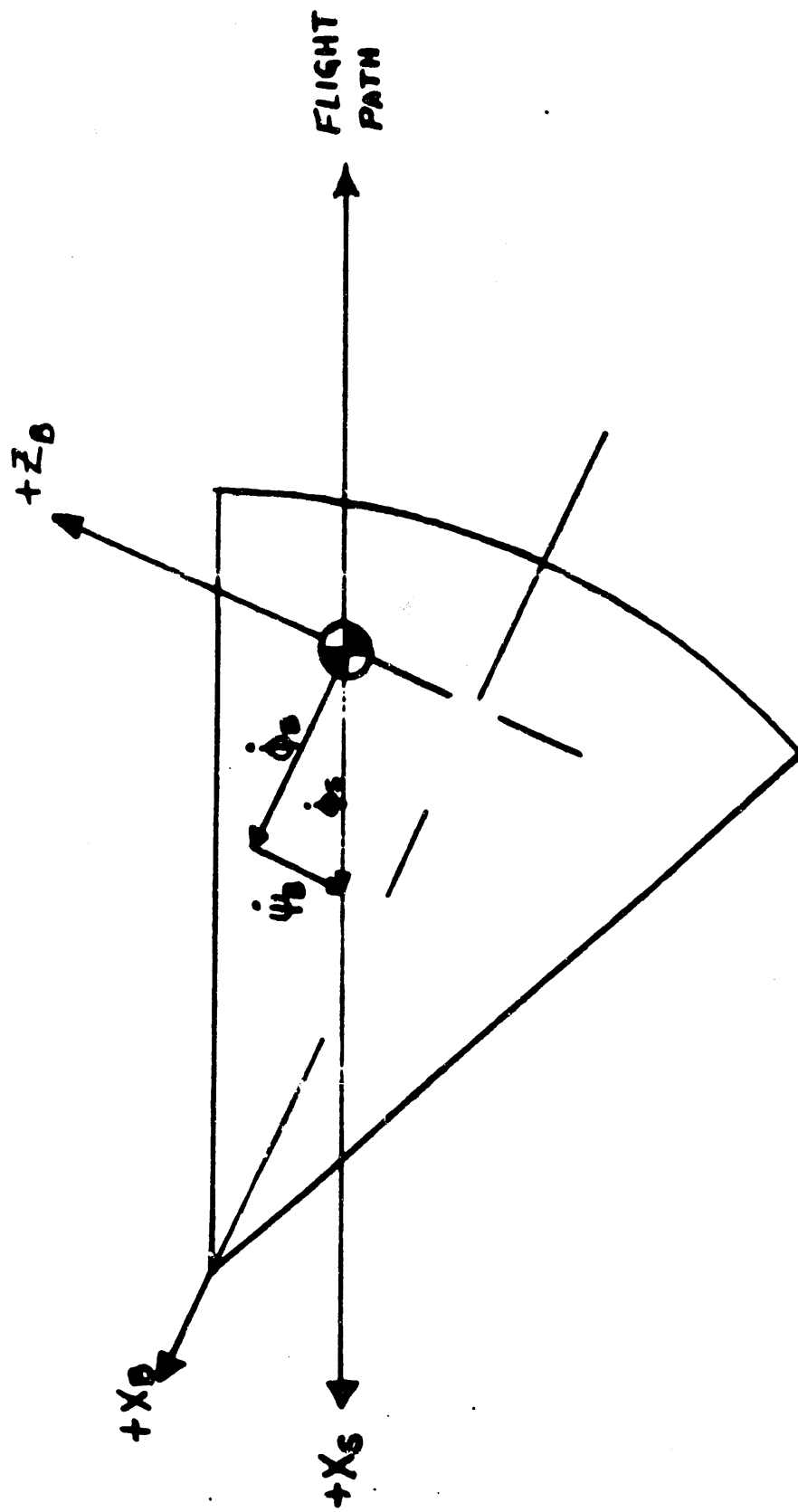
EARTH ENTRY ORIENTATION

FLIGHT PATH

$+Z_B$

$+X_B$

$+X_G$

FIGURE A.5

Reaction Jet Control

As mentioned above, changes in roll attitude during entry will occur about the roll stability axis. The sensed yaw rate signal during a roll maneuver would cause yaw reaction jets to fire if the rate exceeded the deadband value. Consequently, roll to yaw coupling is provided in the reaction jet control mechanization during entry also.