Preface

This study guide has been prepared to augment the oral presentation of the Stabilization and Control System courses. The SCS configuration for this document reflects the first manned S/C (AFRM 012) as of the publication date.

Arrangement of the guide corresponds to the instructor's presentation and affords the user the advantage of review of the material accompanied by a printed description, in the originaly presented sequence.

The format contributes directly to recall memory, minimizes the necessity of making notes and the possibility of misinterpretation.

The accounts and descriptions contained herein are intended to apply only to manned Apollo spacecraft. Unmanned airframe and boilerplate articles will, in most cases, have considerably less sophisticated system configurations.

The contents of this document are accurate as of the publication date. This document is intended for training purposes only and is not subject to periodic revision.
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<td>Attitude Control Subsystem (of SCS)</td>
</tr>
<tr>
<td>AGAP</td>
<td>Attitude Gyro Accelerometer Package</td>
</tr>
<tr>
<td>AGC</td>
<td>Apollo Guidance Computer</td>
</tr>
<tr>
<td>AGCU</td>
<td>Attitude Gyro Coupling Unit</td>
</tr>
<tr>
<td>ARCS</td>
<td>Automatic Reaction Control System</td>
</tr>
<tr>
<td>ARS</td>
<td>Attitude Reference Subsystem</td>
</tr>
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<td>AS/GPD</td>
<td>Attitude Set/Gimbal Position Display</td>
</tr>
<tr>
<td>BMAG</td>
<td>Body Mounted Attitude Gyro</td>
</tr>
<tr>
<td>CDU</td>
<td>Coupling Display Unit</td>
</tr>
<tr>
<td>CSS</td>
<td>Control Stick Steering (Maneuver)</td>
</tr>
<tr>
<td>DASAPECA</td>
<td>Display-AGAP-Electronic Control Assembly</td>
</tr>
<tr>
<td>DECA</td>
<td>Display Electronic Control Assembly</td>
</tr>
<tr>
<td>DRECS</td>
<td>Direct Reaction Control System</td>
</tr>
<tr>
<td>ECA</td>
<td>Electronic Control Assembly</td>
</tr>
<tr>
<td>ECA-P</td>
<td>Pitch Electronic Control Assembly</td>
</tr>
<tr>
<td>ECA-R</td>
<td>Roll Electronic Control Assembly</td>
</tr>
<tr>
<td>ECA-Y</td>
<td>Yaw Electronic Control Assembly</td>
</tr>
<tr>
<td>ECA-X</td>
<td>Auxiliary Electronic Control Assembly</td>
</tr>
<tr>
<td>ETI</td>
<td>Elapse Time Indicator</td>
</tr>
<tr>
<td>FDAI</td>
<td>Flight Director Attitude Indicator</td>
</tr>
<tr>
<td>GAN</td>
<td>Guidance &amp; Navigation</td>
</tr>
<tr>
<td>GOSS</td>
<td>Ground Operation Support System</td>
</tr>
<tr>
<td>IMU</td>
<td>Inertial Measuring Unit</td>
</tr>
<tr>
<td>LCL VERT</td>
<td>Local Vertical</td>
</tr>
<tr>
<td>MIN-IMP</td>
<td>Minimum Impulse</td>
</tr>
<tr>
<td>MTVC</td>
<td>Manual Thrust Vector Control</td>
</tr>
<tr>
<td>P</td>
<td>Roll Body Rate</td>
</tr>
<tr>
<td>PHI</td>
<td>Roll</td>
</tr>
<tr>
<td>PSI</td>
<td>Yaw</td>
</tr>
<tr>
<td>THETA</td>
<td>Pitch</td>
</tr>
<tr>
<td>TVC</td>
<td>Thrust Vector Control</td>
</tr>
<tr>
<td>V&lt;sub&gt;C&lt;/sub&gt;</td>
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</tr>
<tr>
<td>V&lt;sub&gt;G&lt;/sub&gt;</td>
<td>Initial Velocity</td>
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1.1 GENERAL

The Stabilization and Control System (SCS) is the first of four integrated systems for spacecraft control. The SCS receives manual inputs from the crew and electrical inputs from the inertial sensors to generate commands that result in rotation and translation maneuvers. Second, the propulsion systems provide all the external forces and mechanical couples which cause the spacecraft to maneuver under direct control of the SCS. Third, man is included because of the flexibility and reliability that he contributes. The crew will obtain information from the on-board Apollo Guidance Computer (AGC), Manned Space Flight Network (MSFN), or displays that will indicate the necessity to initiate one or more of the six basic motions. Fourth, a combination of the Guidance and Navigation System (GNS) and SCS have the capability of complete control over C/M attitude after the required electronics have been properly configured by the astronauts.

1.2 PHYSICAL DESCRIPTION

The SCS consists of controls and displays, located on the main display panel, the inertial sensors, and the electronic control assemblies, located in the lower equipment bay. The SCS component location in the lower equipment bay is shown in Figures 1-1 and 1-2.

The Rate Gyro Package (RGP) containing the three rate gyro's.

The Attitude Gyro and Accelerometer Package (AGAP) containing the pendulous accelerometers and three body mounted attitude gyro's (BWAG).

The Electronic Control Assembly (ECA) for pitch (ECA-P), and yaw ECA (ECA-Y), and roll ECA (ECA-R).

The Auxiliary ECA (ECA-X) containing the attitude gyro coupling unit (AGCU) and the SPS thrust on/off logic.

The display and AGAP ECA (DAGPECA) containing the display and AGAP electronics.

For discussion purposes the SCS may be divided into the following five categories:

- Inertial sensors (3 types)
- Electronic control assemblies (5)
- SCS control panel
- Manual controls (2 types)
- SCS displays (5)
Functional Description

The SCS is the medium by which the astronauts, the Service Propulsion System, the Reaction Control System, and the GAN System are integrated into a closed loop spacecraft control system.

The two closed loops of the SCS are depicted in Figure 1-4. The inner loop indicates the capabilities of the SCS to provide S/C automatic attitude hold, rate stabilization, and thrust vector control with the capability of crew action. The outer loop represents the capability of the GAN System to provide S/C attitude control or thrust vector control through the SCS interface. Rate stabilization, originating in the rate gyro of the SCS for all operating modes, aids in control of S/C attitude. A third loop, implied in the diagram, is crew operation of the manual direct control to perform S/C maneuvers. The SCS displays provide the crew with indication of S/C rates, attitude, and attitude error.

The SCS provides the following capabilities:

A. Rate stabilization during high-altitude abort after S/C separation from the booster, stabilization of the C/M after toned jettison, and rate damping during entry.
B. Attitude control and rate stabilization about three axes in response to either manual or automatic rotation commands.
C. Translation control along three axes in response to manual commands for attitude maneuvers, transposition docking, and automatic attitude control.
D. Service Propulsion System thrust vector control.
E. Manual SPS engine thrust on-off functions with an automatic thrust off from within the Delta V display.
F. Displays for visual indication of S/C rate, attitude, and attitude error.

The system has been divided into the five categories listed under the physical description. The function of each of the categories will be discussed.
1.4 Inertial Sensors

The inertial sensors are discussed according to their operating characteristics and the components used to perform all required functions internal to the sensors.

1.4.1 Rate Gyro Package

Figure 1-5 shows the rate gyro assembly and its mounting base. It consists of three miniature rate gyro's, orthogonally mounted, to sense angular rates about the three spacecraft body axes, and the associated electronics required for the gyro's.

The rate gyro package electronics is a replaceable assembly designated as plug-in electronics (PIE). This assembly consists of a welded module, power transformer, and three SMRD isolation transformers, which converts the three phase, 400 CPS power to 20V, 400 CPS, single phase signal generator excitation, and isolates the gyro SMRD magnets from transient voltages.

The rate gyro package is designed to be mounted in the spacecraft with the mounting plate reference axes oriented to spacecraft body axes within ±1°. Use of the standard tool allows removal of the gyro assembly from the mounting plate or replacement of an individual gyro in the gyro assembly. Cooling of the package is by conduction to the mounting plate which in turn is cooled by contact with liquid cooled cold plate.

The disassembled mounting base, gyro's, and electronics are shown in Figure 1-6. The gyro's are shown in their mounted positions. The alignment pins shown in the mounting base aid in replacement of the gyro's. There are slots in the ring about the gyro's that permits omitting any alignment procedure after the gyro is installed. The connectors will not rate until after the gyro is correctly aligned with respect to the alignment pin during the installation. The gyro's are interchangeable between the three mounting slots for gyro's.

The Apollo rate gyro is a high reliability, miniature component with reliability and ruggedness in its major design goals. The design of this gyro utilizes experience gained in the design of similar components used in other space programs, and features a rugged spring restraint and self-test torquing circuits consisting of a self-test torquing coil and a SMRD completely isolated from the basic gyro circuits and from each other, so that failure of the self-test circuits will not jeopardize performance of the gyro.

1.4.1.1 Performance Characteristics

Some mechanical and electrical performance characteristics of the gyro are:

- Full scale range ±30°/sec
- Max. rate without damage 600°/sec
- Residual null 0.1°/sec (±77°F)
- Threshold 0.02°/sec
- Resolution 0.06°/sec
RATE GYRO PACKAGE

- X
- Z
- Y

ETI

GRABBER-USED TO SLIDE PACKAGE FROM BASE

REMOVES PLUG-IN ELECTRONICS

MOUNTING BASE

TIGHTENS PACKAGE TO BASE

REMOVES GYRO

FIGURE 1-5

RATE GYROS

ELECTRONICS & MOUNTING BASE SEPARATED

ALIGNMENT PINS

MATING CONNECTORS

PITCH GYRO

YAW GYRO

ROLL GYRO

FIGURE 1-6
1.4.1.2 Spin Motor

The spin motor used in this rate gyro is a synchronous, three-phase, hysteresis motor mounted within the gimbal so that temperature and vibrations will not affect performance. The power associated with the Y to Delta stepdown transformer for spin motor excitation is 25 volts, 400 cps, three-phase. The maximum acceleration time for the motor to reach operating speed is 17 seconds.

1.4.1.3 Gimbal Assembly

This assembly includes the spin motor assembly, pickoff rotor, torquer motor, S/WD assembly and quadrilever spring. The pickoff rotor, torquer motor, S/WD assembly and quadrilever spring are supported at one end by a low-friction shock absorber bearing and at the other end by the quadrilever spring.

1.4.1.4 Signal Generator (Pickoff)

The pickoff is a variable reluctance device used to convert gimbal displacement to a proportional electrical output. The stator is mounted to the gimbal. The rotor excitation is obtained from the secondary of the stepdown transformer in the power supply assembly. The signal generator output voltage is applied as a rate input to the pitch, yaw, and roll ECA's and to the BAGAPECA for the dial rate needle function.

1.4.1.5 Spin Motor Rotation Dector

The S/WD is made up of four soft iron discs, silver brazed into the edge of the spin motor at 90° intervals and a pulse generator (S/WD windings) mounted below the torsion spring. As the iron discs pass the S/WD windings, the reluctance of the air gap is changed causing a voltage to be induced in the windings. Thus a typical S/WD output is a series of pulses.

1.4.1.6 Gimbal Torquer

The gimbal torquer is mounted along the output axis. The stator is wound to produce a torque on the rotor located within the stator on the gimbal output axis. The stator winding is made up of pattern and control field coils. Thus, a DC or AC excitation applied to both windings will cause alternate polarization. Reversal of control field excitation will reverse the direction of resulting torque. In order to obtain linear torquing, the pattern winding current is kept constant while the control field current is varied. Therefore, *28 volts DC is used for pattern excitation during bench maintenance or calibration.

1.4.1.7 Quadrilever Spring

This type of spring mount provides torsion restraint with radial support for the gimbal. The four leaves of the spring provide high transverse stiffness, eliminating bearing support around the torsion spring. Thus, gyro threshold and resolution are improved by reducing overall bearing friction.

1.4.2 ATTITUDE GYRO ACCELEROMETER PACKAGE

Figure 1-7 shows the attitude gyro accelerometer package and its mounting base. It consists of three "single degree of freedom" integrating gyro's mounted orthogonally to sense angular rates about the three spacecraft body axes, and a pendulous accelerometer. Mounted on the spacecraft body X axis, to sense positive accelerations imparted to the spacecraft.

The attitude gyro accelerometer package is designed to be mounted in the spacecraft with the mounting plate reference axes oriented to spacecraft body axes within ± 10 arc minutes. The gyro input axes will be aligned within ± 10 arc minutes of the reference axes on the mounting plate, and the accelerometer sensitive axes within ± 10 arc minutes of the X axis of the plate. Use of the standard tool allows removal of the attitude gyro accelerometer assembly from the mounting plate or replacement of an individual gyro or the accelerometer. Cooling of the package is by conduction to the mounting plate which in turn is cooled by contact with a liquid.

1.4.2.1 PERFORMANCE CHARACTERISTICS

SOME MECHANICAL AND ELECTRICAL PERFORMANCE CHARACTERISTICS ARE:

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<th>Specifications</th>
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<tr>
<td>Gyro Input Axis Freedom</td>
<td>± 20° minimum</td>
</tr>
<tr>
<td>Command Rate</td>
<td>± 25°/sec</td>
</tr>
<tr>
<td>Gimbal Freedom</td>
<td>± 4°/sec</td>
</tr>
<tr>
<td>Threshold</td>
<td>± 0.1°</td>
</tr>
<tr>
<td>Acceleration (Gyro Axes)</td>
<td>20°/s</td>
</tr>
<tr>
<td>Accelerometer Pendulum Travel</td>
<td>± 5.0 Mr</td>
</tr>
<tr>
<td>Threshold</td>
<td>1.0 x 10^-6</td>
</tr>
<tr>
<td>Range</td>
<td>0.16 to 25</td>
</tr>
<tr>
<td>Null Torque</td>
<td>1.0 x 10^-6</td>
</tr>
<tr>
<td>Null Uncertainty</td>
<td>2.0 x 10^-6</td>
</tr>
</tbody>
</table>

1.4.3 BODY MOUNTED ATTITUDE GYRO

The following discussion describes the attitude gyro (BAG components shown in Figure 1-9).

Input turning rates about the input axis produce a gyroscopic precession torque about the output (gimbal) axis which is proportional to the product of the input turning rate and the gyroscopic elements angular momentum. Due to laminar shear of the viscous fluid,
**ATTITUDE GYRO ACCELEROMETER PACKAGE**

- ACCELEROMETER
- REMOVES GYRO
- GRABBER - USED TO SLIDE PACKAGE FROM BASE
- TIGHTENS PACKAGE TO BASE

**TYPICAL FLOATED INTEGRATING GYRO**

- PERMANENT MAGNET TORQUER
- JEWEL PIVOT
- GYRO CASE
- SPIN AXIS
- HEATER AND SENSING COILS
- MAGNETIC SHIELDS
- SIGNAL GENERATOR
- OUTPUT AXIS
- ORIFICE DAMPER SPIN MOTOR
- SPIN MOTOR RUNNING DETECTOR COILS
- INPUT AXIS
- SPIN MOTOR RUNNING DETECTOR MAGNET

**FIGURE 1-7**

**FIGURE 1-8**
FLOATATION FLUID BETWEEN THE GIMBAL AND ITS CONCENTRIC CASE, VISCOSUS RESTRAINT TORQUES ARE PRODUCED WHICH ARE PROPORTIONAL TO THE GIMBAL ANGULAR VELOCITY. THE GIMBAL ACCELERATES TO AN ANGULAR RATE AT WHICH THE VISCOSUS RESTRAINT TORQUE BALANCES OUT THE PRECESSIONAL TORQUE. THE GIMBAL IS THUS TURNING AT A RATE PROPORTIONAL TO THE APPLIED INPUT, AND THE GIMBAL DISPLACEMENT IS A TIME INTEGRAL FUNCTION OF THE APPLIED TORQUE. THE SIGNAL GENERATOR PRODUCES AN OUTPUT VOLTAGE WHICH IS A DIRECT FUNCTION OF THE ANGULAR DISPLACEMENT OF THE GIMBAL. COMMAND INPUTS TO THE GYRO TORQUE GENERATOR DEVELOP TORQUE AT THE GIMBAL, AND COMMAND GIMBAL RESPONSE SIMILAR TO THAT DESCRIBED FOR GYRO INPUTS.

1.4.3.1 Spin Motor

The spin motor is a synchronous, three phase, 400 CPS Hysteresis Motor powered by 13.5 Volts, 400 CPS from a supply in the DAGAPCA. The spin motor will obtain a synchronous speed of 24,000 RPM in approximately 25 seconds.

1.4.3.2 Gimbal Assembly

The gimbal assembly consists of the signal generator, torque generator, and eutectic balance pan.

The signal generator consists of a skew wound, 16 pole, laminated stator, and an eight pole laminated rotor which gates the flux linking the stator poles. The stator has four windings, they are: signal generator primary, signal generator secondary, 6 insensitive drift rate primary and 6 insensitive drift rate secondary. The signal generator indicates angular position of the gimbal with respect to the case and is capable of detecting displacements of less than 2 seconds of arc. The 6 insensitive drift rate applies a fixed torque to the output gimbals insensible drift rate to a net value less than 2°/hour during an earth orbit mission after an initial trim to a drift rate of 0.1°/hour.

The torquer generator consists of a highly stable Alnico Delta V permanent magnet assembly mounted to the gyro case, and an encapsulated moving coil mounted on the gimbal assembly along the gyro output axis. Direct or pulsed currents passed through flex leads to the coil will develop a torque about the output axis, enabling the torque generator to hold the gimbal at signal generator null for any input in the range of zero to ± 2°/second.

The eutectic balance pan consists of a small disk wound with a heater and filled with an eutectic alloy. The balance pan is mounted on the output axis allowing dynamic balancing of the gyro after assembly.

1.4.3.4 Orifice Damper

Orifice damping is a technique for providing wide input angular freedoms using relatively low viscosity fluid, kerosene and practical damping gap thickness. The orifice damper (paddle damper) is a combination of two pairs of paddles, one pair extending from the case wall and the other from the gimbal diameter, resulting in a gimbal rotation which is smaller than the input.

1.4.3.5 Temperature Control Circuit

The gyro temperature is controlled by a temperature sensing element and built in heater, which when connected as one arm of a Wheatstone Bridge and amplifier controlling a relay, the temperature of the gyro is held to 170° + 2°F. The case heater consists of two sections, one for warm up and the other for nominal control. When the gyro reaches operating temperature, a relay keyed to the temperature control bridge will switch automatically to the normal heater circuit.

1.4.3.6 Spin Motor Rotation Ductor

The SMOD consists of a pair of self iron discs, silver brazed into the edge of the spin motor, and a pair of pickup coils wound around the outside of the gyro case. As the iron discs pass the pickup coils, the reluctance of the air gap is changed causing a voltage to be induced.

1.4.4 Accelerometer

Input accelerations to the accelerometer cause displacement of the pendulous mass relative to the case, inducing a signal in the moving coil of a differential transformer pickup. This signal is fed into a servo amplifier where it is amplified, demodulated, and fed back as dc torquer current. Torque balance is established in the presence of steady input accelerations so that the feedback current provides a linear measure of acceleration. The following discussion describes the accelerometer components shown in figure 1-7.

1.4.4.1 Pendulous Element

The pendulous element is isolated from the instrument case by means of a pair of fixtures, or hinged pivots. The pivots, free to deflect in any one plane, offer extreme rigidity to deflection in each of the other two orthogonal planes.

1.4.4.2 Signal Generator

The signal generator mounted at the apex of the pendulum frame, providing maximum angular sensitivity, is excited by a 26.8 kc excitation voltage. The secondary is a moving coil pickoff with a ferrite coil for flux concentration which provides the phase sensitive output signals.

1.4.4.3 Torquer

Restoring torques required to maintain a torque balanced condition result from dc current flow through the torquer coils. The torquer design utilizes the principle employed in electrodynamic loud speakers, in which the moving coil is located within a radial flux gap. The torquer utilizes a pair of coils, coaxially positioned on the pendulum frame.
1.4.4.4 Temperature Control

To provide the required viscous damping, the entire instrument housing is filled with a silicone type oil.

The accelerometer can operate over a 20°F temperature variation and maintain a relatively constant scale factor by means of a simple compensator. Because of the accuracy required, a heater and temperature sensor is used to maintain the temperature at 170°F ± 2°F.

1.5 Electronic Control Assemblies

The ECA's are individual chassis with card modules that are replaceable at the factory only because of the humidity proof card connectors. They are mounted on the S/C cold plate to conduct the electronically generated heat away from the packages. Telemetry conditioning equipment is provided in each unit as are front panel connectors for prelaunch checkout.

1.5.1 Pitch, Roll, and Yaw Electronic Control Assemblies

The pitch, roll, and yaw ECA's are similar with the exception that the ECA-R does not contain thrust vector control circuitry. Each ECA performs the following functions:

A. Provide a means to select the modes and configuration of the control system operation from externally supplied switching signals.

B. Accept input control signals and transform them into suitable output commands for the reaction control system engines or to the service module propulsion system engine gimbals.

C. Provide individual voltage sources not available from the spacecraft power system.

The roll, pitch, and yaw ECA's accept control signal commands from the GAN system, the rate gyro package, the rotation and translation controls, and the BNAG's. The control outputs are signals to the service and command module reaction control system engines and the service propulsion system engine (gimbal actuator magnetic clutches). The main switching interfaces are with the SCS control panel, spacecraft separation systems, and power switches.

1.5.2 Auxiliary Electronic Control Assembly

The auxiliary electronic control assembly contains circuitry to mechanize the attitude gyro coupling unit and the service propulsion system engine thrust on/off control function.

The function of the AGCU loop is to convert BNAG attitude outputs to suitable SCS attitude displays. This loop can be initially aligned to the GAN system inertial coordinates (IMU gimbals angles). It also
MUST TRANSFORM ATTITUDE SET DIAL INFORMATION INTO BODY AXES INFORMATION. THE DISPLAY ON THE FDAI ATTITUDE ERROR NEEDLES WILL BE REFERENCED TO 5/C BODY AXES. THE DISPLAY ON THE FDAI ATTITUDE RATE NEEDLES WILL BE REFERENCED TO BODY AXES. IN THE SCS LOCAL VERTICAL MODE, THE AGCU PROVIDES AN ORBITAL RATE TORQUING SIGNAL TO THE BMAG'S TO MAINTAIN 5/C ATTITUDE WITH RESPECT TO A LOCAL VERTICAL.

THE SPS ENGINE ON/OFF CONTROL FUNCTION PROVIDES CIRCUITRY TO ACCEPT AND CONDITION ON/OFF COMMANDS FROM THE SCS OR THE G&M SYSTEM AND OFF COMMANDS FROM THE DELTA V DISPLAY ELECTRONICS. THE OUTPUT FROM THIS CIRCUITRY WILL INITIATE OR TERMINATE SPS ENGINE THRUSTING.

1.5.3 Display and Attitude Gyro Accelerometer Package Electronic Control Assembly (DAGAPECA)

THE DAGAPECA PROVIDES THE ELECTRONICS ASSOCIATED WITH THE SCS DISPLAYS, THE BMAG'S, AND THE ACCELEROMETER.

THE DISPLAY ELECTRONICS PORTION OF THE DAGAPECA PROVIDES CIRCUITRY TO:

A. ACCEPT AND CONDITION TOTAL ATTITUDE SIGNALS FROM THE G&M SYSTEM AND THE AGCU FOR USE IN POSITIONING THE FDAI ATTITUDE BALL.
B. ACCEPT THE CONDITION ATTITUDE ERROR SIGNALS FROM THE G&M SYSTEM AND THE SCS FOR DISPLACING THE FDAI ATTITUDE ERROR INDICATORS.
C. ACCEPT AND CONDITION ANGULAR VELOCITY SIGNALS FROM THE RATE CYPD PACKAGE FOR DISPLACING THE FDAI ANGULAR VELOCITY INDICATORS.
D. ACCEPT AND CONDITION ENGINE GIMBAL POSITION SIGNALS FROM THE SPS ENGINE ACTUATOR POSITION TRANSDUCERS FOR POSITIONING THE GIMBAL POSITION INDICATORS ON THE AS/GPD.
E. ACCEPT THE CONDITION ACCELERATION SIGNALS FROM AGAPECA ELECTRONICS PORTION OF THE DAGAPECA FOR THE DELTA V DISPLAY.

THE AGAPECA ELECTRONICS PORTION OF THE DAGAPECA PROVIDES THE CIRCUITRY TO:

A. ACCEPT AND CONDITION BMAG INPUTS FOR THE AGCU.
B. ACCEPT AND CONDITION AGCU TORQUING COMMANDS TO THE BMAG'S.
C. CONTROL BMAG AND ACCELEROMETER TEMPERATURES.
D. CONTROL AND CONDITION THE ACCELEROMETER RELANCE LOOP AND INPUTS TO THE INTEGRATOR DURING ENTRY.
E. SUPPLY EXCITATION VOLTAGES TO THE BMAG'S AND THE ACCELEROMETER.
F. CONDITION THE BMAG AND ACCELEROMETER OUTPUTS FOR THE TELEMETRY SYSTEMS.

1.6 SCS CONTROL PANEL

THE SCS CONTROL PANEL PROVIDES CONTROL OVER THE FOLLOWING SCS FUNCTIONS:

A. CONTROL MODE SELECTION SWITCHES (4)
B. ATT DEADBAND SELECT SWITCH
C. 0.05G ENTRY SWITCH
D. BACKUP RATE SELECTION SWITCHES (3)
E. CHANNEL ENABLE SWITCHES (4)
F. DIRECT MODE SELECTION
G. LIMIT CYCLE SELECTION

THE MODE CONTROL SWITCHES ALLOW SELECTION OF ONE OF THE FOLLOWING MODES:

A. SCS ATTITUDE CONTROL
B. SCS LOCAL VERTICAL
C. G&M ATTITUDE CONTROL
D. MONITOR
E. G&M DELTA V
F. SCS DELTA V
G. G&M ENTRY
H. SCS ENTRY

THE ATT DEADBAND SWITCH PROVIDES TWO LEVELS OF ATTITUDE DEADBANDS IN ALL THREE ROTATION AXES OF THE S/C.

THE BACKUP RATE SELECTION SWITCHES PROVIDE INDIVIDUAL SELECTION OF A REDUNDANT RATE SIGNAL SOURCE.

THE CHANNEL ENABLE SWITCHES PROVIDE AN ELECTRICAL ENABLE OF S/M OR G/M RCS BY CHANNELS (E.G., PITCH, YAW, AND ROLL JETS). DIRECT MANUAL CONTROL OF THE RCS IS NOT AFFECTED BY THE ENABLE FUNCTION.

1.7 MANUAL CONTROL

THE MANUAL CONTROLS PERMIT SIX DEGREES OF COMMANDABLE VELOCITIES AND FOR ACCELERATION TO PERFORM ALL MANEUVERS ON THE APOLLO MISSION.

1.7.1 Three Axes Rotation Control

THE THREE AXES ROTATION CONTROL IS A HAND OPERATED STICK TYPE ASSEMBLY CONTAINING TRANSDUCERS AND SWITCHES FOR A SIMULTANEOUS CONTROL OF ANGULAR RATES ABOUT THE SPACECRAFT PITCH, ROLL, AND YAW AXES. THE CONTROL IS DETACHABLE FROM THE CREW SEATS FOR USE AT OTHER LOCATIONS WITHIN THE COMMAND MODULE. BREAKOUT SWITCHES ENERGIZE SYNCHRONIZING CIRCUITS WITHIN THE SCS WHEN THE S/C IS TO BE MANUALLY MANEUVERED. THE S/C ANGULAR RATES CAN BE COMMANDED WITH THE CONTROL. DIRECT SWITCHES ARE PROVIDED IN EACH AXIS TO CONTROL THE RCS SOLENOIDS BY BYPASSING THE SCS ELECTRONICS AND ROUTING THE COMMAND DIRECTLY TO THE RCS. TWO ROTATION CONTROLS ARE PROVIDED AND BOTH MAY BE USED SIMULTANEOUSLY IF NECESSARY. A MECHANICAL LOCKING DEVICE IS PROVIDED TO PREVENT INADVERTENT MOVEMENT OF THE CONTROL.
1.7.2 **TRANSLATION CONTROL**

The translation control is hand operated T handle assembly containing switches for control of translational velocities along the spacecraft body axes, crew initiated aborts, and to open the attitude hold control loops and select manual thrust vector control. Translation maneuvers are introduced by the astronaut’s application of force to the T handle in the desired direction of motion. Abort commands and manual command modes are initiated by rotary motion of the T handle. The control initiates translation commands through on/off switches rather than proportional transducers. The translation control contains a locking mechanism to prevent inadvertent stick motion. Two controls are presently provided for the left arm rest of the commander’s and systems engineer’s seats.

1.8 **SCS DISPLAYS**

The SCS displays provide information to the crew for manual inputs.

1.8.1 **FLIGHT DIRECTOR ATTITUDE INDICATOR**

The attitude display portion of the FDAI is a servo driven gimbaled ball with full and continuous bi-directional rotation capabilities in pitch, yaw, and roll. An indication of attitude is provided by observation of the ball markings in relation to two indices, a body axes index and navigation axes index. The ball utilizes a fixed vehicle-moving horizon concept. Signals to drive the ball are processed in the DGA-PECA according to data received from the AGCU in the auxiliary ECA for SCS control modes, and from the IMU gimbal resolvers during GAN control modes. An indication of IMU gimbal angle limitations is provided on the ball to avoid maneuvering the spacecraft through angles which would cause an IMU gimbal lock. Pitch is indicated by great circle lines on the ball, yaw by minor circles lines, and roll by a separate roll “bug” about the periphery of the ball. The ball may be aligned to a new reference through the AGCU loop using the FDAI Align switch on the AS/GPD.

The attitude error portion of the FDAI consists of three “fly-to” needle type indicators that display attitude error in body axes or, during GAN entry, navigation axes. The attitude error signals may originate in the GAN system or the SCS depending on the operational mode. The attitude needles can display the errors between S/C and the command attitudes dialed into the attitude set dials on the AS/GPD. When holding an inertial attitude, the attitude errors displayed represent the S/C drift from the commanded attitude within the deadband limits. The range of displayed attitude errors is changed according to the selected modes.

The angular velocity indicators are three "fly-to" needle type displays hotted about the periphery of the FDAI to indicate angular rotation rates about S/C body axes. The normal signal input to the angular velocity indicators are body rates derived from the SCS rate gyro's. However, when a backup rate source is selected, these indicators display body rates derived from the DG-PECA.

1.8.2 **ATTITUDE SET/GIMBAL POSITION DISPLAY**

The attitude set/gimbal position display provides an indication of SPS engine gimbal yaw and pitch position. Thumbswheels are provided to manually position the engine gimbals prior to initiating a Delta V maneuver. This function is required to approximately align the SPS engine’s thrust vector through the spacecraft’s center of gravity. Gimbal position indicator signals are derived from the SPS engine gimbal position transducers. Commands to these gimbals are processed by the SCS electronic control assemblies.

In addition to the engine gimbal displays and controls, the AS/GPD contains two switches. The ATT SET switch, in conjunction with the dials and thumbswheels, can be used to insert new commanded attitude orientations that will appear as attitude error indications on the FDAI. The FDAI ALIGN switch provides alignment of the AGCU and thus the FDAI ball to a new reference determined by the reading on the pitch, yaw, and roll thumbswheels.

1.8.3 **DELTA V DISPLAY**

The Delta V display provides indications of velocity changes after ascent, and automatic backup and manual control of velocity changes. The Delta V SET switch provides slewing control of the Delta V remaining digital display to preset the desired velocity in ft/sec. As the spacecraft accelerates in the X direction, the Delta V remaining display decreases as a function of the integrated SCS accelerometer signal.

The direct vllage switch is used to perform a backup acceleration using the direct solenoid in the RCS prior to ignition of the SPS engine.

The thrust control switch provides arming power to the SPS engine propellant valves for normal ignition of the SPS engine. It also provides a backup thrust off command to the GAN or SCS automatic thrust off command. The switch also provides a direct on command to the SPS engine.

The thrust on switch is illuminated at all times when the SCS or GAN has a thrust on command. It also permits the astronaut to perform the manual input to command thrust on in the SCS Delta V mode. The automatic thrust off command in the SCS Delta V mode is commanded when the Delta V remaining display reading is zero or less.
SECTION II

CONTROLS AND DISPLAYS

2.1 INTRODUCTION

The SCS controls and displays are designed to allow utilization of man in the control loop. He is provided with the capability to monitor and evaluate gross malfunctions for automatic attitude control maneuvers. At his perogative, he can manually fly the spacecraft.

2.2 SCS CONTROL PANEL

The control panel shown in Figure 2-1 contains the toggle switches required for configuration of the SCS into its eight operating modes: control enabling functions, redundant rate gyro selection, direct mode enabling, and attitude performance (deadband, limit cycle, and etc.) selection. The SCS control switches are located on the main display console just forward and above the translation control on the commander's left arm rest. This location permits left hand switching for SCS functions in a fully pressurized space suit.

2.2.1 Mode Selection

The mode selection is controlled by four switches. The first order selection is performed by the GM-SCS switch. It selects the source of control and display information for total attitude display, attitude error display, and control inputs. The three position switch labeled ATTITUDE-MONITOR-ENTRY controls the basic selection of display scale factors, maneuver rate capability, level of rate damping, and enables selection of additional modes in the attitude and entry positions. The Delta V switch will be used to modify the attitude position (GM or SCS) and select the desired Delta V mode (GM or SCS). The LCL VERT switch will modify the SCS and attitude position of the first two switches to generate the correct configuration for maintaining a local vertical attitude while in a earth orbit. The entry position of the three position switch will permit operation of the .05g switch to select the required changes in the SCS configuration to achieve the correct attitude control for the aerodynamic phase of earth entry. There are eight selectable modes as shown in Table 2-1. It lists the mode selected versus the switch position. Figure 2-2 gives the switch operation that is required to provide the mode selection.

<table>
<thead>
<tr>
<th>Mode</th>
<th>SW #1</th>
<th>SW #2</th>
<th>SW #3</th>
<th>SW #4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Monitor</td>
<td>ON</td>
<td>ON</td>
<td>OFF</td>
<td>OFF</td>
</tr>
<tr>
<td>GM</td>
<td>ON</td>
<td>ON</td>
<td>OFF</td>
<td>OFF</td>
</tr>
<tr>
<td>Monitor</td>
<td>OFF</td>
<td>OFF</td>
<td>ON</td>
<td>ON</td>
</tr>
<tr>
<td>SCS</td>
<td>OFF</td>
<td>OFF</td>
<td>OFF</td>
<td>OFF</td>
</tr>
</tbody>
</table>

Table 2-1. Operating Mode vs Selection Switch Positions
FIGURE 2-1

FIGURE 2-2
TABLE 2-1. OPERATING MODE VS SELECTION
Switch Positions (Cont’d)

<table>
<thead>
<tr>
<th>MODE</th>
<th>SW #1</th>
<th>SW #2</th>
<th>SW #3</th>
<th>SW #4</th>
</tr>
</thead>
<tbody>
<tr>
<td>G&amp;N ATT. CONT.</td>
<td>G&amp;N ATTITUDE</td>
<td>OFF</td>
<td>OFF</td>
<td></td>
</tr>
<tr>
<td>G&amp;N DELTA V</td>
<td>G&amp;N ATTITUDE</td>
<td>DELTA V</td>
<td>OFF</td>
<td></td>
</tr>
<tr>
<td>G&amp;N ENTRY</td>
<td>G&amp;N ENTRY</td>
<td>OFF</td>
<td>OFF</td>
<td></td>
</tr>
<tr>
<td>SCS LOCAL VERT</td>
<td>SCS ATTITUDE</td>
<td>OFF</td>
<td>LOC VERT</td>
<td></td>
</tr>
<tr>
<td>SCS ATT. CONT.</td>
<td>SCS ATTITUDE</td>
<td>OFF</td>
<td>OFF</td>
<td></td>
</tr>
<tr>
<td>SCS DELTA V</td>
<td>SCS ATTITUDE</td>
<td>DELTA V</td>
<td>OFF</td>
<td></td>
</tr>
<tr>
<td>SCS ENTRY</td>
<td>SCS ENTRY</td>
<td>OFF</td>
<td>OFF</td>
<td></td>
</tr>
</tbody>
</table>

2.2.1.1 Monitor Mode

The monitor mode of operation will be for the launch phase of the normal Apollo missions. It will provide automatic rate damping of the S/C for the contingencies normally considered in the launch phases if activated by the Sequential Events Control System. It also provides a rate damped mode only when desired in the mission. The displays are controlled by the G&N and the SCS RATE GYROS. Translations along the X, Y, and Z axes are also provided in this mode for the C/M-S/M configuration. Additional use may be incorporated as the exact mechanization for orbit with the SIVB attached to the S/C as defined.

2.2.1.2 G&N Attitude Control Mode

The G&N Attitude Control mode can be used during orbit periods between orbit injection and entry. The guidance system must be functioning during this mode. The functions available include the capability of automatic attitude hold, manual and automatic attitude maneuvers, and translations along the X, Y, or Z axes. However, attitude hold will be the normal function of this mode.

2.2.1.3 G&N Delta V Mode

The G&N Delta V mode is selected when a velocity change, requiring the use of the Service Propulsion System, is to be made and the G&N system is operational. In this mode, the G&N system will control the attitude and velocity changes necessary to correct the S/C trajectory as required. This is the primary mode of control for velocity corrections with the Service Propulsion System. Translation control is provided in this mode except when the SPS engine is thrusting.

2.2.1.4 G&N Entry Mode

The G&N Entry mode is used to perform a programmed automatic entry through the Earth's atmosphere.
to a selected landing site. The program is initiated prior to landing and requires crew action to configure the SCS electronics for entry aerodynamic conditions. The automatic program controls spacecraft range and azimuth by commands to the roll axis electronics of the SCS. The GAN "entry" mode is the primary operating mode for entry. A clockwise rotation of the translation control gives the crew the capability to manually fly the GAN commands.

2.2.1.5 SCS Local Vertical Mode

The SCS Local Vertical Mode is a special case of attitude hold using the SCS. It would be used in orbital periods to hold the S/C at some approximately fixed attitude with respect to the local vertical of the Earth. The S/C will be oriented to the desired attitude and direction, with respect to the orbital path, before initiating this mode of operation. Translation control is available in this mode, but not considered desirable for the normal function this mode performs.

2.2.1.6 SCS Attitude Control Mode

The SCS Attitude Control Mode of operation can be used during the orbital phase of the mission. In this mode, the GAN system need not be functional. The capabilities of this mode are automatic attitude hold, manual attitude maneuvers, and translation along the vehicle body axes. Attitude control will be the normal function of this mode.

2.2.1.7 SCS Delta V Mode

The SCS Delta V Mode of operation performs inertially oriented thrusting maneuvers with the service propulsion system by changing the vehicle attitude to keep the thrust vector orientation inertially fixed. It will also maintain the cross axis velocity within the maneuver accuracy required. This mode of operation can perform, or complete, all thrusting maneuvers in the Apollo mission where inertially fixed thrust vector orientation will conform to the mission trajectory requirements.

The SCS Delta V Mode is a backup to the GAN Delta V Mode for control of the service propulsion system when performing Delta V maneuvers.

An additional thrust vector control function is now incorporated for the manned spacecraft. It provides for manual thrust vector control from the rotation control with the selection of either GAN or SCS Delta V modes and a CH rotation of the translation control. It permits direct control from the rotation control to the servo drive amplifier #2 in the thrust vector control electronics for gimballing the service propulsion engine.

2.2.1.8 SCS Entry

The SCS Entry Mode is a manual backup mode for the GAN Entry Mode. The astronaut can use manual proportional rate control or the manual direct system to "fly" a safe entry. Displays will be provided to evaluate the entry trajectory. The displayed indications provide the cues for attitude control inputs required to perform a safe entry. No range or azimuth control is required in the SCS Entry Mode.

2.2.2 Rate Gyro Selection

Rate gyro selection consists of three, two position toggle switches. The two positions are NORMAL and STBY. In the NORMAL position, the SCS displays and electronics obtain rate input signals from the rate gyro package. The STBY position will convert the selected BAVG (pitch, roll, or yaw) to a rate gyro and insert its signals into the SCS displays and electronics in place of the normal rate gyro for the selected axes (pitch, roll, or yaw). One or all these switches can be placed in the BAVG position, as required, should contingencies arise. The AGCU will not be operational with one BAVG in the caged configuration.

2.2.3 Channel Enable Switches

The channel enable function is desirable in segments of the mission where no strict attitude control is required. The channel enable switch can be placed to the OFF position and remove automatic attitude control and proportional rate maneuver capabilities. Attitude control can be used to set up constant angular velocities about one or more axes, and then by placing the channel enable switches to the OFF position, it will permit the angular velocities to continue when the rotation control is returned to neutral with zero command output. The four channel enable switches are two position toggle switches labeled ABC ROLL, B&D ROLL PITCH, and YAW. Each channel enable switch will control four reaction control engines. Therefore, two switches are required for the roll channel. ABC ROLL will be ineffective after C/M-S/M separation because of a reduction in the number of roll reaction control engines and removal of translation requirements after separation of the C/M-S/M. The channel enable functions will not inhibit the manual direct mode when in the OFF position.

2.2.4 Attitude Deadband Switch

The attitude deadband selection enables conservation of reaction control system fuel and oxidizer, as required on the mission, by selection of the amount of drift permissible in the attitude hold modes. It is a two position toggle switch for selecting a MAX position which permits a nominal movement of ±5° about the desired attitude, and a MIN position which permits a nominal movement of ±0.5° about the desired attitude before signals will be processed through the SCS for reducing the attitude error. The three position switch labeled ATTITUDE, MONITOR, and ENTRY must be in the attitude position to select the MIN positions of the ATT DEADBAND. When the three position switch is in either of the other two positions, only the MAX deadband is provided.

2.2.5 .05g Entry Switch

The .05g Entry Switch enables the necessary gain, display, and cross coupling changes throughout the system required at .05g. It is a two position toggle
2.2.6 Limit Cycle Switch

The LIMIT CYCLE SWITCH IS A TWO POSITION TOGGLE SWITCH TO REMOVE PSEUDO RATE IN ALL THREE AXES MANUALLY. PSEUDO RATE WILL BE REMOVED FOR ROTATION AND TRANSLATION HANDEVERS AUTOMATICALLY WITH THE MANUAL OVERRIDE CAPABILITY. IT ALSO PERMITS REMOVING PSEUDO RATE WHERE NOT DESIRED FOR OTHER SEGMENTS OF THE APOLLO MISSION.

2.2.7 Direct Mode Switch

The DIRECT MODE SWITCH IS A TWO POSITION, DOUBLE POLE, TOGGLE SWITCH WHICH PRODUCES POWER TO THE DIRECT SWITCHES IN THE ROTATION CONTROL. ROTATION CONTROL NO. 1 RECEIVES POWER FROM DC BUS A AND ROTATION CONTROL NO. 2 RECEIVES POWER FROM DC BUS B. THERE CAN BE NO DIRECT ROTATION COMMANDS INITIATED WHEN THE DIRECT MODE SWITCH IS IN THE OFF POSITION. THEREFORE, THE SWITCH WILL BE PLACED IN THE DIRECT MODE POSITION FOR CRITICAL PHASES OF THE MISSION SUCH AS ENTRY.

2.3 Flight Director Attitude Indicator

The Flight Director Attitude Indicator (FDII) COMBINES THREE DISPLAY FUNCTIONS PERTAINING TO SPACECRAFT ORIENTATION. TOTAL ATTITUDE, ATTITUDE ERRORS, AND BODY ANGULAR RATES ARE INDICATED FOR THE PITCH, ROLL, AND YAW AXES. THE FDII IS SHOWN ON FIGURE 2-3.

2.3.1 Total Attitude

Total attitude is displayed on the gimbaled attitude ball. It has 360° of freedom in the pitch, roll, and yaw axes. The ball is marked in pitch from pole to pole similar to longitude markings on the globe, and in yaw from 270° - 0 - 90° by markings placed perpendicular to the pitch markings, similar to the latitude markings on a globe. Roll is marked from 0° to 359° in 10° increments in a CCW direction about the case surrounding the ball.

The attitude ball is positioned from two sources of data (Figure 2-4). In the monitor node and all GNS modes, total attitude is obtained from the guidance and navigation system (GNS) gimbal angle outputs. Therefore, the attitude ball is an H/SM gimballed angle repeater when the GNS is providing the attitude information. There are two indices on the face of the instrument. The symbol represents the body axes orientation with respect to the GNS stable element. The symbol represents the navigation axes displacement from the GNS stable element. The two symbols are displaced by approximately 35° in the vertical or pitch direction because of the angular offset between the body and navigation axes of the spacecraft.

The navigation axes symbol and red circular pole markings on the yaw axis of the ball serve as a gimbaled lock indicator when the GNS is operating. The crew must ensure that the navigation axes symbol does not enter the

---

**Figure 2-3**

- **Legend:**
  - + POLARITY RATE - ATTITUDE ERROR
  - - POLARITY RATE - ATTITUDE ERROR

- **Key Points:**
  1. Attitude error = Attitude desired - Actual attitude
  2. The ball is of the inside-out convention
  3. Dulator angle convention is pitch, yaw, roll
  4. The ball attitude shown is pitch 35°, yaw 90°, and roll 360° with respect to the navigation base index.

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2-7
RED CIRCULAR POLE (GIMBAL LOCK) MARKING ON THE YAW AXIS. THE ATTITUDE BALL THEN SERVES TWO FUNCTIONS IN THE G&N CONTROL MODES; AS AN IMU GIMBAL ANGLE REPEATER, AND AS AN INDICATOR TO JUDGE MANEUVERS AND TO AVOID GIMBAL LOCK CONDITION OF THE IMU PLATFORM GIMBALS.

THE SECOND SOURCE OF DISPLAY INFORMATION IS OBTAINED FROM THE ATTITUDE GYRO COUPLING UNIT (AGCU). THE THREE SHAFT ANGLE RESOLVERS IN THE AGCU PROVIDE INFORMATION FOR ATTITUDE BALL POSITIONING COMPAREABLE TO THE IMU GIMBAL INFORMATION. WHERE THE IMU INFORMATION REPRESENTS ONLY THE ANGULAR DISPLACEMENT BETWEEN THE IMU STABLE ELEMENT AND THE SPACECRAFT. THE AGCU INFORMATION PROVIDES A STRAIGHTDOWN BACKUP INERTIAL REFERENCE SYSTEM AND CAN BE DIRECTED TO PRESENT INFORMATION THAT IS IDENTICAL TO IMU INFORMATION, OR REPRESENT AN ENTIRELY DIFFERENT INERTIAL REFERENCE SYSTEM. THEREFORE, IT WILL BE IMPORTANT TO BE CONSCIOUS OF THE INFORMATION SOURCE WHEN EVALUATING DISPLAYED ATTITUDE ORIENTATION.

2.3.2 ATTITUDE ERRORS


### Table 2-2. Operating Mode vs Attitude Error Full Scale Deflection Value

<table>
<thead>
<tr>
<th>Modes</th>
<th>Roll</th>
<th>Pitch</th>
<th>Yaw</th>
</tr>
</thead>
<tbody>
<tr>
<td>MONITOR</td>
<td>±25°</td>
<td>±25°</td>
<td>±25°</td>
</tr>
<tr>
<td>SCS &amp; G&amp;N ENTRY</td>
<td>±25°</td>
<td>±25°</td>
<td>±25°</td>
</tr>
<tr>
<td>ALL OTHER MODES</td>
<td>±5°</td>
<td>±5°</td>
<td>±5°</td>
</tr>
</tbody>
</table>

THE ATTITUDE ERROR INFORMATION IS DERIVED FROM THE COUPLING DISPLAY UNIT (CDU) IN THE G&N MODES OF OPERATIONS AND FROM THE GAD'S IN THE SCS MODES OF OPERATIONS (FIGURE 2-4). ATTITUDE SET AND FDAI ALIGN ARE TWO SPECIAL CASES, WHERE THE ATTITUDE ERROR IS DERIVED FROM THE AGCU FOR PERFORMING MANEUVERS, OR ALIGNING THE BACKUP INERTIAL REFERENCE SYSTEM TO A NEW OR DESIRED ATTITUDE WITH RESPECT TO A GIVEN INERTIAL REFERENCE. THE CREW WILL BE REQUIRED TO UNDERSTAND THE MODES AND REACTION OF THE ATTITUDE ERROR DISPLAYS FOR EACH MODE TO PROPERLY INTERPRET THE ERROR INDICATOR.

2.3.3 ATTITUDE RATES

THE ATTITUDE RATES ARE DISPLAYED ABOUT THE FACE OF THE FDAI WITH ROLL AT THE TOP, PITCH ON THE RIGHT SIDE, AND YAW AT THE BOTTOM. THE MAGNITUDE OF THE RATE DISPLAY SCALES ARE NOT MARKED BECAUSE FULL
SCALE DEFLECTION VALUE WILL CHANGE ACCORDING TO OPERATING MODE OF THE SCS. THE FULL SCALE DEFLECTION VALUE FOR EACH OPERATING MODE IS GIVEN IN TABLE 2-3.

<table>
<thead>
<tr>
<th>MODES</th>
<th>ROLL</th>
<th>PITCH</th>
<th>YAW</th>
</tr>
</thead>
<tbody>
<tr>
<td>MONITOR</td>
<td>$\pm 25^\circ$/SEC</td>
<td>$\pm 5^\circ$/SEC</td>
<td>$\pm 5^\circ$/SEC</td>
</tr>
<tr>
<td>SCS LCL VERT</td>
<td>$\pm 1^\circ$/SEC</td>
<td>$\pm 1^\circ$/SEC</td>
<td>$\pm 1^\circ$/SEC</td>
</tr>
<tr>
<td>SCS &amp; GAN ATT</td>
<td>$\pm 25^\circ$/SEC</td>
<td>$\pm 25^\circ$/SEC</td>
<td>$\pm 25^\circ$/SEC</td>
</tr>
<tr>
<td>SCS &amp; GAN DELTA</td>
<td>$\pm 25^\circ$/SEC</td>
<td>$\pm 25^\circ$/SEC</td>
<td>$\pm 25^\circ$/SEC</td>
</tr>
<tr>
<td>SCS &amp; GAN ENTRY</td>
<td>$\pm 25^\circ$/SEC</td>
<td>$\pm 25^\circ$/SEC</td>
<td>$\pm 25^\circ$/SEC</td>
</tr>
</tbody>
</table>

The rates displayed are true S/C rotation rates displayed on a "FLY-TO" rate needle and will normally be obtained from spring restrained rate gyros for all operating modes (Figure 2-4). A second source of rate information is obtained from the Body Mounted Attitude Gyros (BMG) when they are caged for rate gyro operations and the rate gyro selection switch for the appropriate channel (ROLL, PITCH, or YAW) is in the BMG position.

In this configuration, rate damping has priority over attitude hold functions because it is removed, and since attitude errors will not be available from the BMG's, attitude rates will be displayed.

2.4 ATTITUDE SET/GIMBAL POSITION DISPLAY

The Attitude Set/Gimbal Position Display (Figure 2-5) provides two distinct SCS functions. The Gimbal Position section provides visual display and manual control of the SPS engine gimbal angles. The Attitude Set section is used to set inertial reference commands into the Attitude Gyro Coupling Unit.

2.4.1 Gimbal Position Display

The Gimbal Position Display provides an accurate indication of the present SPS engine gimbal angle so that the engine thrust vector can be positioned to act through the center of gravity of the spacecraft. The pitch gimbal indicator has a range of $90^\circ$ about zero, measured in the X-Y plane. The inherent offset center of gravity in the S/C causes the gimbal null or center point in the Yaw Axis to be $45^\circ$.

2.4.2 Gimbal Position Trim Controls

The Gimbal Position Control Thumbwheels are used to position the SPS engine gimbals prior to all thrusting phases. The pitch and yaw gimbal position controls generate the position commands to the respective gimbal positionning loops. The trim control will be adjusted prior to each SPS firing and the proper engine position verified from the visual display. The initial positioning error should be small to prevent a moment arm from...
Causing undesirable rotational movements about the S/C center of gravity. The alignment error should be less than an approximate value of 1°. The thumbwheels are marked with gimbal angle values. The angle marked on the thumbwheel represents the approximate command gimbal trim angle. They represent a gross comparison of the display reading and cannot be used as a comparator for display accuracy without additional, more precise data.

2.4.3 Attitude Set Controls and Indicators

The three attitude set thumbwheels and adjacent indicators provide manual updating capabilities of the AGU. The AGU resolver shaft angle is compared in the indicator with the shaft position reference of the indicator resolver. The resultant differential signal may be used to align the AGU and FDAI, or the differential signal can be displayed on the attitude error indicators for manually maneuvering the S/C to the new desired attitude.

2.4.4 Attitude Set Switch

The two positions of the ATT SET switch are ATT SET and OFF. The ATT SET position provides an AGU signal to the attitude error indicators that will display the difference between the S/C attitude indicated on the FDAI and the desired attitude set on the thumbwheels of the AS/GRD. This will permit the crew to align, or update, the AGU to any reference frame or determine S/C attitude displacement from a chosen reference frame.

2.4.5 FDAI Align

The FDAI ALIGN switch is a push-to-make switch used to initiate alignment of the AGU. The switch must be depressed until alignment is accomplished.

2.5 Delta V Display

The Delta V display, as shown on Figure 2-6, is used to monitor the thrusting velocity imparted to the S/C by the service propulsion system and yaw translations. This panel enables the crew to set in the required velocity correction to be attained by thrusting, perform manual backup to automatic engine fire command, manual backup to the automatic thrust off command, and provides a redundant direct ullage capability.

2.5.1 Delta V Remaining

The Delta V remaining indicator will display the velocity correction (in feet per second) to be attained by thrusting with the service propulsion system during a Delta V maneuver. It is a digital indicator which is stepped in ½ feet per second increments. The indicating range is from -1,000 (99,000) to 12,999 feet per second. In a thrusting sequence, an automatic thrust off command should be generated by the...
SCS or GaN, depending upon the operating mode. When the Delta V remaining indicator reaches 30,000 feet per second, it generates a thrust-off command, and the indicator will continue to count in a minus direction tabulating the velocity imparted to the S/C by the residual thrust from the Service Propulsion System.

2.5.2 Delta V Set

The Delta V set switch is a blue command switch to drive the Delta V remaining indicator. It is the only method by which the Delta V remaining indicator can be set or changed without accelerating. The Delta V set switch has two commandable drive rates. The soft stop in each direction represents a drive rate of 1 2 feet per second per second. The second the switch is depressed. The hard stop, which is beyond the soft stop, will command a drive rate of 205 feet per second per second. The second the switch is depressed.

2.5.3 Thrust Control Switch

The thrust control switch is a lever-lock, three-position toggle switch. The center position is off. To select either of the other two positions, the enlarged end of the switch must be displaced laterally from the display panel before the switch can be moved up or down. The position is normal. In this position, the Apollo guidance computer or the thrust on command can energize the fuel and oxidizer solenoids in the service propulsion system providing all other logic steps have been completed. The down or direct on position energizes the SPS solenoids in the service propulsion system by passing all electronics and logic circuits. The thrust control switch toggles back to the off position with no restrictions.

2.5.4 Thrust On Switch

The thrust on switch is a momentary contact type switch for manual thrust on logic command to the service propulsion system engine in the SCS Delta V mode. Illumination of the thrust on switch provides an indication of service propulsion engine ignition commanded by the thrust control logic. The switch will remain illuminated at all times when a logic thrust on command exists and will be extinguished when the logic thrust off command is detected. The thrust on command cannot originate from the Apollo guidance computer or the momentary thrust on switch.

2.5.5 Direct Ullage

The direct ullage switch will provide a direct backup to the translation control normally used to initiate an ullage maneuver (positive X translation) prior to SPS engine ignition. If used, the vehicle will translate with no attitude hold control until SPS engine ignition.

2.6 Rotation and Translation Controls

In addition to the capability of commanding specific attitudes through the keyboard of the guidance computer, the astronauts are able to manually command specific attitudes or attitude change rates during the mission. Translation control is required for docking practice in earth orbit and backup for de-orbit with complete SPS failures. A pair of manual rotation and translation control are provided for the astronauts use where required in the command module.

All rotation controls can be removed from the seats but have a permanent cable connecting them to the control system or directly to the reaction jets (manual direct control).

All controls have a mechanical lock to prevent inadvertent initiation of commands during re-location or storage. A visual cue is provided to ascertain the status of the lock.

2.6.1 Rotation Control

The rotation controls (Figure 2-7) are designed for right hand use and can generate inputs in either proportional rate or direct commands to the SCS attitude control system for the roll, pitch, and yaw channels simultaneously or individually. Manual direct operation of one axis will not affect proportional rate operation in the other two axes.

The command input axes are shown in Figure 2-7 with the offset comfort angles for right hand use. The direct command switches are closed approximately one degree short of full travel in each axis. The roll axis of control has soft stops incorporated before the direct switches are closed. The soft stops prevent inadvertent direct commands from interrupting manual proportional rate commands during manual entry control. There are no soft stops in the pitch and yaw axes of the rotation controls.

The rotation controls are held in the center or neutral position by spring force gradients. The angular displacement is directly related to the displacement force applied. Breakout switches are provided in each control axis to sync the SCS for manual control inputs from the rotation control.

2.6.2 Translation Control

Translation control is provided through the S/M RCS and, therefore, is available prior to S/M/C/M separation only. The translation control (Figure 2-8) is a "U" handle configuration with up-down, right-left, and in-out motions providing multiple or singular axes translations simultaneously. Translation command can be initiated simultaneously that will result in a composite maneuver. The positive X translation will be called an ullage maneuver when performed prior to a firing of the SPS engine.

The counterclockwise and clockwise switches will be used to provide switching inputs to the SCS and master events sequence controller.
2.6.3 Attitude Impulse Control

The ATTITUDE IMPULSE control (Figure 2-9) at the Navigator's station is used for small precise maneuvers when aligning the sextant for navigation sightings. The attitude impulse control signal generation is internal to the SCS but activated by external switches.

The ATTITUDE IMPULSE ENABLE switch activates the required circuits in the SCS to permit minimum impulse operation. The SCS must be in the GAN or SCS Attitude Control modes before the Attitude Impulse Enable will be solenoid held in the on position.

2.7 Associated Control Panels

The SCS requires switching inputs from additional control panels to perform its required operations. The additional control panels are within reach of the commander's control station. The control panels will contain switching functions other than for the SCS. Therefore, only the SCS switching functions will be discussed.

2.7.1 Left Hand Circuit Breaker Panel

The left hand circuit breaker panel (Figure 2-10) contains one switch and 18 circuit breakers that operate directly into the SCS system. The circuit breaker for yaw, pitch, and roll provides DC power to the solenoids for the Reaction Control System. The DC power will be used only upon a command from the SCS attitude control subsystem.

G&N Optics Control Panel

100 Series

Figure 2-9
The SCS system power is divided into Group 1 and 2. Group 1 will power the circuits required for the GAN operating modes of the SCS. Group 2 will provide all the required SCS power to perform the manual thrust vector control mode of operation and in so doing provides the additional power required for the normal SCS operating modes of the SCS.

Group 1 circuit breakers A and B determine the route of the power through the SCS control panel and mission sequencer for RCS enable. Power will be routed through diodes from both buses simultaneously. Group 1 circuit breaker A will provide DC power to translation control #1 and Group 1 circuit breaker B will provide power to translation control #2 for commanding translation maneuvers. All AC power is routed through Group 1 power switches. Group 1 circuit breakers AC and AC 2 and the remaining DC power is provided into the SCS by the Group 1 power switches located on the SCS power switches.

Group 2 circuit breakers A and B will provide the switching power for manual and backup functions in the SCS. Group 1 circuit breaker A provides power direct to the breakout switches in the rotation control #1 and the clockwise switch in the translation control #1. Group 1 circuit breaker B provides power direct to the breakout switches in rotation control #2 and clockwise switch in translation control #2. The inputs from AC 1 and AC 2 circuit breakers and the remaining DC power will be selectable by Group 2 of the SCS power switches on the SCS power switches.

The direct circuit breaker A and B provide all the required DC power to actuate the direct modes of operation. DC circuit breaker A will provide 28 VDC to the direct mode switch for rotation control #1, direct uillage switch for uillage capability from the two aft pointing pitch jets (Engine #1 and #2) and thrust control switch to energize the Service Propulsion System coil set #1 for the thrust on the Service Propulsion engine. DC circuit breaker B will provide 28 VDC to the direct mode switch for rotation control #2, direct uillage switch for uillage capability from the turn aft pointing yaw jets (Engine #5 and #6) and thrust control switch to energize the Service Propulsion System coil set #2 for the thrust on of the Service Propulsion engine.

The direct functions in the SCS provide power for control with no electronics between the DC buses and the coils energized.

The GAN sync switch is a two position toggle switch that is operational only in the GAN attitude control mode of operation. In the ON position, and with the GAN operational, the GAN sync circuit permits the breakout switch in the rotation control to place the COU'S in a follow mode. When the maneuver is completed, the new attitude will be accepted as the new command attitude, thus providing a manual maneuver capability in the GAN attitude control mode.
2.7.2 SCS Power Switches

The SCS power switches (Figure 2-11) are divided into group 1 and 2. The switches will control all switchable power required for the SCS functions. They control AC and DC power wherever required. The AC1 position of each switch will select AC Bus I and MN Bus A for DC power if required. The AC2 position of each switch will select AC Bus #2 and MN Bus B for DC power inputs if required.

2.7.2.1 SCS Group 1 Power Switches

The SCS Group 1 power switches provide the SCS power required to perform all necessary control functions in the GAN operating modes of the SCS.

The PARTIAL SCS power switch is a three position rotary switch. The three positions are AC1, OFF, and AC2 in a clockwise sequence. The switch provides AC & DC power to the non-thrust vector control rate and attitude electronics, Attitude Gyro Coupling Unit, and miscellaneous electronics required for attitude control.

The RATE GYRO power switch is a three position rotary switch. The three positions are AC1, OFF, and AC2 in a clockwise sequence. It provides 3 phase AC power to the rate gyro package and reference frequency for the rate display demodulator. The two SCS switches covered thus far will provide sufficient SCS power for attitude control with appropriate GAN attitude error input signals.

The TVC #1 power switch is a three position rotary switch. The three positions are AC1, OFF, and AC2 in a clockwise sequence. It provides AC and DC power to the thrust vector control electronics for the pitch and yaw channels, X-axis accelerometer, delta V display electronics, and thrust on-off electronics.

2.7.2.2 SCS Group 2 Power Switches

The SCS Group 2 power switches provide power to the segments of the SCS system that permit manual crew inputs into the SCS control and all required power for manual thrust vector control.

The ROTATION CONTROL power switch is a three position toggle switch. The three positions are: AC1 OFF, and AC2 from top to bottom. In the AC1 position, phase A power is supplied from AC Bus I to the rotation control transducer transformers located in the pitch and yaw electronics control assemblies. The transformer in the pitch electronics control assembly supplies excitation to the transducer in rotation control #1, and the transformer in the yaw electronics control assembly is for the transducer of rotation control #2. It also supplies reference frequency to the rotation control demodulator in the roll, pitch and yaw electronics control assemblies. The AC2 position provides phase A power to the same circuits from AC Bus 2.

The BANG power switch is a three position rotary switch. The three positions are AC1, OFF, and AC2 in a clockwise sequence. It provides AC and DC.
power to the body mounted attitude gyros for temperature control, spin motor operation, and required electronics for normal operation. It also provides reference frequency for the manual thrust vector control rate demulators.

The TVC 2 Power switch is a three position rotary switch. The three positions are AC 1, OFF, and AC 2 in a clockwise sequence. It provides AC power to all elements of the pitch and yaw electronics control assembly used for the manual thrust vector control mode not activated by other switches in SCS group 2 power.

2.7.3 Crew Safety Control Panel

The Crew Safety Control Panel (Figure 2-12) provides switches that control functions for the SCS and Reaction Control Systems. There are five switches that provide an input directly or indirectly for normal checkout and control functions.

2.7.3.1 Abort System Oxidizer Dump Auto Switch

The OX Dump Auto Switch controls circuits for automatic dumping of the oxidizer from lift off until approximately 42 seconds have elapsed. After the 42 seconds have elapsed from lift off, it will be manually placed in the RCS CM AUTO position. In the down position, there will be one enabling input for activation of the inputs to the reaction control system solenoids at abort or separation by the Sequential Events Control System (SECS).

2.7.3.2 Reaction Control System CMD Switch

The Reaction Control System CMD switch is a three position toggle switch spring loaded to the center position. It is a backup control to the automatic enabling of the reaction control system solenoids driven in the SCS except for SIVB separation. The center position of the toggle switch permits the SCS to enable the reaction control system solenoid drivers in the SCS when the OX Dump Auto is in the RCS CMD AUTO position.

In the normal separation procedure from the SIVB or at other times in the mission as required, the CMD switch enables the reaction control system solenoid drivers in the SCS when momentarily placed in the OFF position. The CMD switch also disables the reaction control system solenoid drivers in the SCS when momentarily placed in the OFF position.

2.7.3.3 Reaction Control System C/M Press Switch

The C/M Press is a spring loaded to OFF toggle switch. Momentary positioning to the UP position will pressurize the C/M Reaction Control System. The switch will fire the helium pyrotechnic isolation valves, thus pressurizing the C/M Propellant tank.

2.7.3.4 Reaction Control System Transfer C/I Switch

The Transfer C/I/S/I Switch functions as a backup control to the normally automatic transfer of the reaction control systems after C/I/S/I Separation. This switch also allows the checking of both reaction

Figure 2-12

2-26
CONTROL SYSTEMS PRIOR TO LAUNCH WITHOUT STAGING THE VEHICLE. THE CENTER POSITION OF THE SWITCH IS NORMAL FOR AUTOMATIC SWITCH OVER.

The Reaction Control System C/M-S/M SEP switch is a two position toggle switch spring loaded to the OFF position. Momentary positioning to the UP position will provide normal initiation of the C/M-S/M Separation and generate all related events in Attitude Control change over to the C/M. It also will initiate retrograde of the S/M after separation.

2.7.4 Service Propulsion Gimbal Actuator Motors

Located on the Emergency Detection and Gimbal Drive Control panel are the six switches associated with the thrust vector control subsystem (Figure 2-13). These switches serve to actuate the gimbal drive motors of the SPS engine gimbals and arming of the SPS rocket engine.

The four switches on the left hand side of the panel are for Gimbal Actuator Motor control. They are three position switches; the up position start, the center position on, and the down position off. The start position is momentary or spring loaded. The two switches in this group on the left hand end are for control of the motors in the pitch actuator. The two on the right hand end of the group are for control of the motors in the yaw actuator. The number 1 indicates the primary motor in the yaw and pitch actuators respectively. The number 2 denotes the secondary, or backup, motor of the redundant electrical system. Both motors will be operating for Delta V maneuvers to provide immediate switch over in event of failure in the primary motors or selection of a manual thrust vector control mode. The two switches at the extreme right end of the six switches are identified Inject Pre-Valves. These are two position switches with the up position on and the down position off. These switches provide control to the arming valves of the SPS rocket engine.

2.7.5 Master Caution Panels

The master caution lights associated with the SCS provide warning of failures not easily detected from displays. There are two panels of caution lights and the SCS receives warning indications from each. One panel is on the upper left of center on the main control panel and the second is on the upper right of center.

2.7.5.1 Agap Temperature Failure (Figure 2-14)

The three Biw heaters will keep the damping fluid within given temperature limits of 170 ± 2°F after initial warm up. Should a heater fail in one of the three Biw's, other sources would be needed to isolate the failed heater. Operating the attitude gyro's when they are not at their proper operating temperature could, in extreme cases, cause catastrophic failures. However, normally degraded system performance will be detected by observing the various attitude displays before the gyro's reach these extreme operating temperatures.
2.7.5.2 SPS Rough EGO (Figure 2-14)

The SPS Rough External Combustion Light is illuminated by the latch relay of the Flight Combustion Stability Monitor. To extinguish the light after illumination occurs, the FCSM power switch must be placed to the RESET position. When the light is illuminated, thrusting with the SPS engine will be terminated unless an override is used.

2.7.5.3 Gimbal Drive Failure Lights

The two gimbal drive failure caution lights are on the right hand master caution panel (Figure 2-15). One caution light is for the pitch gimbal failure light and the other is the yaw gimbal failure light. Illumination of either caution light will indicate a failure of the primary gimbal drive motor in the axis of control corresponding to the caution lights. The caution lights will indicate an under or over current condition for the indicated primary motor. There is an automatic switch to remove power from the faulty motor and the same signal will cause the SCS to be switched to its redundant electronics for drive commands to the redundant DC motor clutches for gimbal positioning commands.

2.7.6 Panel 2

Panel 2 provides controls to perform FDI rate self test, FDI illumination, switching for the FCSM, and the entry "H" meter.

2.7.6.1 FDI illumination

The brightness control selects the level of illumination desired on the FDI attitude ball. It augments the cabin lighting to reduce shadows under the indices and improve readout capability.

2.7.6.2 FDI rate self test

The FDI self test switch inserts an input to the rate electronics causing a deflection of the rate needles to 4/5 full scale negative rate in each axis. It also removes the rate gyro input if backup rate is not selected. If backup rate is not selected, if backup rate is selected, the rate self test input is supplied with the mag rate signal and a summation of the two signals would determine rate needle deflection.

2.7.6.3 FCSM switching

There are two switches for control. The ON-OFF-RESET switch controls power to the FCSM. In the ON position, the FCSM will perform its normal function of terminating thrust of the SPS engine for a rough combustion. Several conditions could combine and cause a momentary rough combustion that a complete shutdown and re-start would correct. The reset position is required to extinguish the SPS rough EGO light and set the FCSM for monitoring the SPS engine combustion upon restart. The switch must be in the ON position for all monitoring functions.
The AUTO-OVERRIDE SWITCH PROVIDES A BY-PASS OF THE FCSM FOR MALFUNCTIONS OR CREW CRITICAL FUNCTIONS. The AUTO position permits the FCSM to perform its normal functions and would be the normal position. The OVERRIDE position would by-pass the FCSM function and prevent SPS engine shutdown for rough combustion except for manual crew action with other SCS switches.

2.7.6.4 $^g_\alpha$ METER

The $^g_\alpha$ METER is a mechanical device that provides a continuous $^g_\alpha$ force indication. It has a readout scale marking of $-5$ to $+15$ $^g_\alpha$. The same $^g_\alpha$ meter is used in the Gemini. It contains three readout needles. The top needle is the $^g_\alpha$ force readout. The lower two needles will provide a record of the largest negative and positive $^g_\alpha$ applied since last reset. The reset control switch will return both memory needles to zero if the actual $^g_\alpha$ level at that time is zero. If not the $^g_\alpha$ force applied will determine which needle does not return to zero but stops at the displacement position of the $^g_\alpha$ needle.

The instrument face is mounted at a 17° offset from the instrument panel to provide for alignment of the sensor with the spacecraft's $X^1$ body axis. This rotates the $^g_\alpha$ readout down toward the astronaut and should reduce readout parallel error.

2-33
3.1 INTRODUCTION

The ability of the astronaut to aid in the determination of spacecraft attitude within inertial space may well be a deciding factor contributing to the success or failure of any mission involving flight from one inertial body to another. In Apollo, two systems are provided to assist the astronaut in this task, see Figure 3-1. Each system is mechanized with a computer, an inertial reference device, and a readout device. The primary system is the Guidance and Navigation System (G&N). The guidance and navigation function of this system is mechanized within two separate subsystems; the optics subsystem, and the inertial subsystem. The optics provide a variable line of sight telescope and a fixed and variable line of sight sextant. When used in conjunction with the Apollo Guidance Computer (AGC) and the optics-coupling display unit (CDU), this subsystem can be used to determine spacecraft inertial attitude and position. Once inertial attitude has been determined, the inertial subsystem may be aligned to hold a reference attitude and provide inertially referenced delta velocity obtained from accelerometers mounted on a three gimballed inertial, or stable platform. The backup attitude reference subsystem (ARS) is mechanized within the SCS. This device can provide and display on the Flight Director Attitude Indicator (FDI) all the attitude data normally available from a three gimballed platform, but it is configured as a monitor rather than a command system, consequently no automatic maneuver capability is provided in the SCS systems. Precise manual maneuvers may be accomplished through proper utilization of the SCS controls and displays. The ARS utilizes an orthogonal set of single degree of freedom gyroscopes to sense changes in vehicle attitude about the body axes of the spacecraft and generate equivalent attitude error signals. These error signals are sent to the Flight Director Attitude Indicator (FDI) for display, and to the ARS where, by means of an electro/mechanical process, they are transformed into directional sines and cosines and gyro torquing signals. The directional sines and cosines are returned to the FDI and associated electronics, where they are transformed into an Euler angle readout which is equivalent to the inertial angle output of the G&N stable platform. The gyroscopes and electro/mechanics of the ARS are mechanized into a strapdown inertial reference package which can provide all the attitude data normally available from a gimballed platform. The chart, Figure 3-1, compares the gimballed reference system and the strapdown reference system.

3.2 PURPOSE

The purpose of either reference system of the Apollo G&N is to relate spacecraft attitude to the

![Diagram of Apollo Attitude Reference Systems](image)

**Figure 3-1**

<table>
<thead>
<tr>
<th>Basic Requirements</th>
<th>G&amp;N System Devices</th>
<th>Strap Down Inertial Reference</th>
</tr>
</thead>
<tbody>
<tr>
<td>Computer</td>
<td>AGC-AGC Keyboard</td>
<td>AGCU-ECA-ASIGPD</td>
</tr>
<tr>
<td>Inertial Reference Device</td>
<td>IMU</td>
<td>DGAPE-CA-AGCU</td>
</tr>
<tr>
<td>Readout Device</td>
<td>CDU AGC Keyboard Display</td>
<td>FDI-ASIGPD</td>
</tr>
</tbody>
</table>

SCS-206A
COORDINATE SYSTEM OF AN ARBITRARILY SELECTED FIXED REFERENCE FRAME. A MATHEMATICAL SOLUTION TO THIS RELATIONSHIP WAS DEVELOPED BY THE SWISS MATHEMATICIAN, LEONHARD EULER, WHO SUMMED SUCCESSIVE ROTATIONS ABOUT ANY AXIS OF A RIGID BODY USING A THREE BY THREE MATRIX. SOLVING THIS MATRIX AFTER ANY SINGLE ROTATION OR SERIES OF ROTATIONS YIELDS THE "EULER ANGLE" ATTITUDE OF THE BODY, AS DISPLAYED ON THE FDAI, WITH RESPECT TO THE COORDINATES OF A FIXED FRAME. THE EULER ANGLE ATTITUDE OF THE BODY AFTER A SERIES OF MANEUVERS IS DEPENDENT NOT ONLY UPON THE DIRECTION AND MAGNITUDE OF BODY ROTATIONS, BUT ALSO UPON THE SEQUENCE OF ROTATION. THE IMPORTANCE OF ROTATIONAL SEQUENCE IS ILLUSTRATED BY FIGURE 3-2 AND BY THE FOLLOWING DESCRIPTION:

**Position A** - Shows the attitude of the rigid body when it is aligned to the fixed reference frame.

**Position B** - Shows the alignment of the rigid body after a +90° roll with respect to the body or reference axes.

**Position C** - Depicts the body attitude at the completion of a -90° yaw about the reference axes or a +90° body pitch rotation.

Now for positions D, E, and F of Figure 3-2, he will reset the body to the initial position and perform the maneuver again, keeping the direction and magnitude of rotation the same but changing the sequence of rotation.

**Position D** - Shows the position of origin and is identical to Position A.

**Position E** - Shows the body after a -90° yaw about the body or reference axes.

**Position F** - Shows the final position of the body at the completion of a +90° body pitch.

The finite Euler attitude of the body in each example is not relevant to this discussion even though this attitude could be determined by observing the final position of the body with respect to the reference axes. What is important is that the body has a different attitude at the completion of each maneuver sequence and that only the maneuver sequence has been changed, thus illustrating the necessity of an Euler angle display (such as the FDAI) of spacecraft attitude to facilitate manual spacecraft maneuvers. A figure, similar to Figure 3-2, could be drawn to show that rotational sequencing is as important for body axes maneuvers as it is for maneuvers about the reference axes; i.e., roll, pitch vs. roll, then pitch.
3.2.1 Apollo Reference Axes

The problems associated with maintaining a reference attitude in an arbitrarily selected fixed reference frame is in reality more complicated than it appears in Figure 3-2. The complication is due to the zero reference of the Navigation Base Triad. This triad is oriented so that its X and Z axes are rotated 35° with respect to the spacecraft body X and Z axes. The Y axis of both triads are parallel. The relationship of these axes to the spacecraft is shown in Figure 3-3. Fortunately, the electronic mechanism (which will be discussed later) required to solve the axial misalignment is not unduly complicated and does provide the capability of rotating the spacecraft through 360° about the yaw body axis without going into gimbal lock. Gimbal lock is encountered when the middle and outer gimbals of the Hun become aligned. The GAN provides a logic signal to the Master Caution Lights to alert the crew to an impending gimbal lock condition. In addition, the FBAI displays the critical gimbal lock zone by means of two red 30° circles, on the surface of the FBAI dial, centered on the Y axis. The optics base zero reference is also shown in Figure 3-3. The angles generated by the shaft and trunion are utilized in aligning the GAN and ARS.

3.3 Flight Director Attitude Indicator

The FBAI provides the crew with a visual link to the outside world and the relationship of this world to spacecraft rotational maneuvers resulting from manual or automatic commands. In addition, it provides an excellent means, to the trained operator, of detecting malfunctions in the electro/mechanical systems of the SCS and GAN. The device simultaneously displays eleven independent cues for referencing the body and navigation axes to an inertially fixed reference. These indications are roll, pitch, and yaw, total attitude, command attitude error, and command rate. (The term command refers to the directional relationship between indicator movement and control stick deflection required to null the indicator.)

3.3.1 FBAI Total Attitude

The total attitude readout is obtained from a spherical display, or ball, which has three servo driven gimbals that are sequenced pitch inside, yaw middle, and roll outside. These gimbals have full continuous bi-directional rotational capability. The sphere, see Figure 3-4, is marked with great circle lines and small circle lines which correspond to the longitude and latitude lines of a globe. There are twelve great circle lines and five small circle lines. The angular distance between each of the lines is 30°. The sphere provides indications from 0° to 90° and 270° to 0° pitch and yaw total attitude may be read from either the body axis index, a miniature airplane (△) shape, or the navigation axes index, a cross hair ( ⊙ ). If the instrument were to indicate 0° pitch and 0° yaw, the 0° pitch line would be tangent to the horizontal cross hair of the nav index and

INTERRELATED APOLLO REFERENCE FRAMES OR TRIADS
THE 0° YAW LINE WOULD BE DIRECTLY UNDER THE VERTICAL CROSS HAIR OF EACH INDEX. ROLL EULER ATTITUDE IS INDICATED BY THE ROLL BUG (AN INDICATOR NEEDLE DRIVEN BY THE ROLL GIMBAL THROUGH A GEAR MECHANISM). AS THE VEHICLE ROTATES POSITIVELY OR CLOCKWISE, THE ROLL BUG IS DRIVEN COUNTERCLOCKWISE. THE EULER ROLL ANGLE IS OBTAINED FROM A BEZEL RING. THE BEZEL RING IS MARKED IN 10° INCREMENTS WHICH INCREASE POSITIVELY IN A COUNTERCLOCKWISE DIRECTION FROM THE TOP CENTER OF THE BEZEL.

3.3.1.4 FDI Total Attitude Display Philosophy


3.3.2 FDI Attitude Error and Rate Displays

In addition to total attitude, the FDI also displays attitude error and rate of change in attitude. The attitude error display needles use the center point between the Body and Navigation Indices as zero or null point for all three display needles. The attitude error displayed by these needles is command signal required to null the needle, thus correcting the error. In a command error configuration, the

3.4 GUIDANCE AND NAVIGATION SYSTEM MECHANIZATION

The Guidance and Navigation System maintains the inertial or fixed reference frame through the use of a stable platform. This device takes advantage of the inertial properties of gyroscopes mounted on an inner block, gyros maintain a nonrotating reference frame by driving isolation gimbals. This, in effect, allows the spacecraft to move or rotate about the inner block. Accelerometers, also mounted on the inner block, keep track of and null any inertial displacement of the reference frame by a process of double integration. Gyro mechanization is depicted in Figure 3-1. It shows two distinct operating loops or subsystems within the GNS; the optics subsystem and the navigation subsystem. The optics are automatically or manually controlled, and, with the computer, may be used by the astronaut in aligning the stable platform and the ARS. The navigation section platform will maintain the fixed reference frame and automatically control spacecraft attitude and thrust vector attitude by providing a command error signal to the SCS electronics. CGN command errors are derived by taking the error between the shaft angles of the inertial measuring unit (IMU) gimbals and the shaft angles of the coupling display unit (CDU). These errors may be generated either by the computer driving the CDU shafts or by the spacecraft maneuvering about the CDU indicated attitude. In the first case, the computer will command a particular vehicle attitude by driving the CDU shafts to a new position. This causes the CDU shaft angle and IMU gimbal angle to be in disagreement and generates an attitude error which is sent to the SCS, where it is displayed, and processed to energize the appropriate set of reaction jets to correct the error. As the IMU gimbal position approaches the CDU resolver position, the magnitude of error is reduced to zero and the vehicle is stabilized at the new attitude. Since the reaction jets are nearly aligned with the spacecraft body axes, the command which cause them to energize must be referenced to the body axes of the spacecraft during non-atmospheric flight. CGN command errors are transformed to the body axes coordinate system through a fixed resolver in the roll and yaw error channels of the coupling display unit. This resolver is bypassed during CN entry mode so that commands, sent to the reaction jets and displayed, are referenced to the navigation axes rather than the body axes. This

IS DONE TO ACCOMMODATE THE PHYSICAL OFFSET OF THE SPACECRAFT ROTATIONAL AXES DURING ATMOSPHERIC FLIGHT.

3.5 ATTITUDE REFERENCE SUBSYSTEM MECHANIZATION

The Attitude Reference Subsystem, of the stabilization and Control System, computes and maintains a stable reference frame for the spacecraft. In this case, the reference is maintained by a strapdown package which provides three single-degree-of-freedom wide-angle integrating gyros (QG240) mechanized into a rate loop which maintains an extremely low level output (less than approximately 0.15° attitude error) while the vehicle is being maneuvered, unless the rotational velocities exceed the ARS limits of 5 deg/second, in pitch and yaw and 20 deg/second in roll. The system is capable of providing and displaying a short term attitude reference for the spacecraft. Three signal loops are used; control, information, and logic. These signal loops are mechanized within six electronic control assemblies of the ARS, figure 3-5. These assemblies are:

A. Rate Gyro Package (RGP)
B. Attitude Gyro Accelerometer Package (AGP)
C. Attitude Set/Gimbal Position Display (AGP/DP)
D. Flight Director Attitude Indicator (FDI)
E. Display Attitude Gyro Accelerometer Package Electronic Control Assembly (DECA)
F. Attitude Gyro Coupling Unit (ACU)

3.5.1 CONTROL LOOP

Control loop accuracy is the most critical factor affecting the successful operation of the ARS. For ease of understanding, the control loop may be made analogous to the operation of a servo loop in which a sensor detects a change in attitude and generates an equivalent electrical signal. This signal is ultimately returned, after processing, to null or zero the sensor. A single axis mechanization of the control loop is represented in Figure 3-6. This diagram shows signal processing within the ARS control loop. Changes in spacecraft attitude are detected by the gyros. The resultant signal is converted to the navigation axes reference frame and subsequently converted into a Euler angle rate, which is demodulated and amplified before being applied to the level detectors. If the error signal is large, the forward level detector will switch when the sensed level becomes greater than the deadband. Switching the level detector causes the stepper motor to step 0.2 degrees and sends a torque signal back through the reverse transformation (Euler to Navigation to Body) to torque the gyro to null, or at least below the level detector switching point. The stepper motor and torqueing circuits are controlled by a timer which makes pulse timing periods of 40 ms in pitch and yaw and 10 ms in roll for each .2° change in attitude.
3.5.2 **INFORMATION LOOP**

The information loop, Figure 3-5, of the ARS provides total attitude, attitude error, and rate information for the Flight Director Attitude Indicator. Total attitude and attitude error information may be obtained from either the Guidance and Navigation System of the ARS. Rate is always sensed by either the rate gyros or the attitude gyros in backup rate mode. The term information loop is used because the ARS display electronics are used only for signal processing and gain control required for display of the input signals.

3.5.3 **Logic**

The logic signals required for configuring the ARS are derived from four sources: AS/GPD, SCS Control Panel, GAN Control Panel and the Control Stick. The only switch on the SCS Control Panel which does not affect ARS configuration is the ATT DEADBAND switch.

3.6 **ARS FUNCTIONAL DESCRIPTION**

ARS mechanization and the GAN interface are represented in a simplified functional schematic, Figure 3-7. All the functions and modes of the ARS may be presented using this diagram. The system is mechanized within six units of the SCS. These units have all been discussed in Section One of this study guide; however, a few devices, the DASAPEC, AGCU, and ATTITUDE SET SECTION of the AS/GPD require some further explanation to provide continuity for the discussion of ARS modes of operation.

3.6.1 **Display and Attitude Gyro Accelerometer Package Electronic Control Assembly (DASAPEC)**

The DASAPEC is functionally divided into two separate but integral units which may be operated independently. The first of these units, the Attitude Gyro Accelerometer Package Electronic Control Assembly (AGAPEC), contains the electronics required to operate the attitude gyro and interface with the Attitude Gyro Coupling Unit (AGCU). The second unit, the Display Electronic Control Assembly (DECA), provides interface electronics between the SCS sensors and their corresponding displays. These units are mechanized as follows:

**A. AGAPEC**

<table>
<thead>
<tr>
<th>Function</th>
<th>Purpose</th>
</tr>
</thead>
<tbody>
<tr>
<td>1. Gyro Preamp</td>
<td>Isolation and amplification of attitude gyro outputs.</td>
</tr>
<tr>
<td>2. Body to Navigation Axis</td>
<td>The input summing to this buffer amplifier is scaled to provide the first amplifier conversion in the forward loop by combining roll and yaw signals to</td>
</tr>
</tbody>
</table>

**ATTITUDE REFERENCE SUBSYSTEM**

FDI/AGCU ALIGN

![Diagram of Attitude Reference Subsystem](image-url)
TABLE 3-1. RELAY SWITCHING FUNCTIONS

<table>
<thead>
<tr>
<th>RELAY</th>
<th>PURPOSE</th>
</tr>
</thead>
<tbody>
<tr>
<td>K1ABC*</td>
<td>Applies torquing signals to the attitude gyros.</td>
</tr>
<tr>
<td>K2</td>
<td>Applies attitude gyro outputs to the AGCU.</td>
</tr>
<tr>
<td>K3</td>
<td>Applies AGCU torquing signals to the attitude gyros.</td>
</tr>
<tr>
<td>K4ABC</td>
<td>Applies attitude gyro outputs to the FDI.</td>
</tr>
<tr>
<td>K5</td>
<td>Energized during attitude set to provide an electrical signal equivalent to the difference angle between the AGCU resolver shafts position and the attitude set dial resolver shaft position.</td>
</tr>
<tr>
<td>K6ABC</td>
<td>Applies gyro torquer amp output to the FDI command rate error needles (in the selected axis).</td>
</tr>
<tr>
<td>K7</td>
<td>Applies G&amp;N attitude error signals to the FDI command attitude error needles.</td>
</tr>
<tr>
<td>K8</td>
<td>Applies AGCU generated directional sine and cosine signals to the FDI roll and roll bus.</td>
</tr>
<tr>
<td>K9</td>
<td>Applies G&amp;N generated directional sine and cosine signals to the FDI roll and roll bus.</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>RELAY</th>
<th>PURPOSE</th>
</tr>
</thead>
<tbody>
<tr>
<td>K10</td>
<td>Provides AGCU shaft angle attitude set dial differential signal to realign the AGCU and FDI.</td>
</tr>
<tr>
<td>K11</td>
<td>Applies an orbital rate signal for local vertical operation.</td>
</tr>
<tr>
<td>K12ABC</td>
<td>Provides casing signals to put the attitude gyro in a rate mode individually or together depending upon the selected mode.</td>
</tr>
<tr>
<td>K13ABC</td>
<td>K13 and K14 have contacts which operate independently with corresponding backup rate switch. When a K13 contact is opened during backup rate, the corresponding K14 contact is opened.</td>
</tr>
<tr>
<td>K15ABC</td>
<td>These are used independently in conjunction with backup rate selection in a single channel.</td>
</tr>
<tr>
<td>K16ABC</td>
<td>They switch the gyro pre-amp output from the attitude demodulator to the rate demodulator in the selected channel or in all three channels.</td>
</tr>
</tbody>
</table>

*ABC Subscript denotes relay contacts which may be energized independently according to configuration requirements.

A. AGAPECA - CONTINUED

FUNCTION | PURPOSE
---|---
Obtain $\psi$ and $\varphi$ errors. The pitch buffer amp requires no conversion.

3. Attitude Torquing Amplifier Provides a torque to the body axes of the AGCU to the body axes error signal for use in the FDI attitude error display.

B. DCEA

1. Navigation Axes Conversion To convert navigation axes attitude error signals derived from the AGCU to the body axes error for use in the FDI attitude error display.

2. Range Adjust and Amplification Provides the necessary gain change to allow a variance in full scale magnitudes of attitude error and rate displays of FDI.

3.6.2 ATTITUDE GYRO COUPLING UNIT (AGCU)

The Attitude Gyro Coupling Unit (AGCU) is the primary control device for the attitude reference subsystem. It performs the major part of all the strapdown inertial reference computations and, in addition, provides the transformation matrices required to display attitude and attitude error of the spacecraft. The primary elements of the AGCU are in the attitude gyro torquing loop, or control loop. They perform the following functions.

FUNCTION | PURPOSE
---|---
A. Navigation Resolves pitch and yaw attitude error to derive pitch and yaw rates.

B. AGCU Amp and Level Detector Generates a pulse train that is equivalent to the total change in $\varphi$/C attitude.

C. AGCU Stepper Motor Control Provides bi-directional control of resolver shaft angle.
3.7 ATTITUDE SET/GIMBAL POSITION DISPLAY (AS/GPD)

The attitude set portion of the Attitude Set/Gimbal Position Display (AS/GPD) provides the capability for realignment of the AGCU and FDI in the AGCU and FDI to compensate for long term drift or loss of the SCS reference attitude. It also enables adjustments in S/C attitude using the rotation control and the FDI command at attitude error needles. The functions provided by the Attitude Set portion of the AS/GPD are:

**Function** | **Purpose**
--- | ---
A. | ATTITUDE SET INDICATORS AND THUMBWHEELS: Display and change the reference attitude of the ARS for alignment or maneuvering purpose.
B. | ATTITUDE SET SWITCH: Provides means of displaying the body axes equivalent of the Euler angle error between the attitude displayed on the Attitude Set Dials and the angle of the resolver shafts in the AGCU, on the FDI command attitude error needles.
C. | FDI ALIGN SWITCH: Aligns the AGCU resolvers to the attitude displayed on the attitude set dials.

3.8 ARS CONFIGURATION

There are some seventeen unique configurations of the ARS. These configuration changes are generally accomplished through the mode select switches, however, other separate functions such as minimum impulse, control stick steering, channel disable, and others also affect ARS configuration. Each of the preceding paragraphs discusses a different configuration of the ARS. Table 3-1 explains the function performed by each of the relays.

3.8.1 FDI/AGCU ALIGNMENT

FDI/AGCU alignment allows the astronaut to realign or update the Attitude Reference Subsystem to compensate for system drift or to align the attitude reference subsystem to any chosen reference frame, either celestial or IMU. Figure 3-7 shows the relays required by FDI/AGCU alignment. FDI/AGCU alignment is accomplished by first determining the displacement in Euler angles of the spacecraft navigation axes triad with respect to a known reference frame. These Euler angles are then manually dialed into the attitude set indicators on the Attitude Set/Gimbal Position Display. Depressing the FDI ALIGN switch will cause the AGCU to realign its resolvers with respect to the AS/GPD. During all SCS modes, the FDI attitude ball will align itself so that pitch and yaw Euler angles, displayed on the AS/GPD indicators, will be displayed by the roll attitude indicator or bug when the switch is depressed. Alignment accuracy and good functional test of the AGCU resolvers and drive circuits may be demonstrated by placing the ATT SET switch in the ON position before depressing the FDI ALIGN switch. The attitude error needle should displace and then null as FDI alignment is completed. Alignment may be accomplished during SCS attitude control modes but only the AGCU will be aligned. The FDI will continue to indicate null gimbal angles and the attitude set function may not be used. Alignment may be accomplished by pressing the FDI ALIGN switch at any time during any mode providing the applicable power switches are in the ON position. Alignment should not normally be accomplished unless the system is deadweight within the minimum +0.5° dead and in minimum impulse. At the completion of alignment, the difference between SCS ARS Euler angles and the Euler angles of the S/C NAV axes, with respect to a chosen reference frame, shall be less than ±0.25° excluding built-in mechanical errors or operator errors.

3.8.2 ATTITUDE SET

The ATT SET position of the ATT SET switch provides a signal from the AGCU to the command ATTITUDE ERROR NEEDLES to indicate the direction and magnitudes (when less than ±5°) of any angular error between the attitude existing in the AGCU and a desired attitude manually set in the Attitude Set/ Gimbal Position Display. Figure 3-8 shows the interface switching for attitude set.
ATTITUDE SET is provided to allow accurate manual rotational maneuvering to facilitate alignment for delta V, GAN coarse alignment, navigation sightings, or reorientation to the initial spacecraft attitude after a series of manual rotational control inputs. This switch allows the astronaut to monitor the accuracy of FDI/ACCU alignment during SCS control modes. The attitude set function may be used during all SCS modes prior to manual 0.05g switching. It may not be used during any GAN mode. Using attitude set, the vehicle may be manually maneuvered, using the rotation control, to the attitude indicated on the Attitude Set/Gimbal Position Display with an accuracy of ± 0.71°, (not including mechanical alignment errors in systems other than the GAN). This is accomplished by nulling the FDI/ACCU command attitude error display.

3.8.3 Control Stick Steering (CSS), Minimum Impulse, and Channel Disable

These sub-modes provide the crew with manual control of spacecraft attitude using proportional or direct reaction jet control. Figure 3-10 functionally shows the data flow during SCS attitude control and any of the above sub-modes. Control stick steering mode is energized by moving the rotation control stick out of detent. This action closes break-out switches in the rotation control to provide logic inputs to the subsystems of the stabilization and control system. At the completion of a manual maneuver during SCS modes the SCS will hold the spacecraft in its new attitude within the selected deadband. In monitor mode the system will be rate stabilized having no attitude hold capability. Control stick steering (direct or proportional control) may be used to control spacecraft attitude while in any of the following modes:

- SCS Attitude Control
- SCS Delta V (before engine ignition)
- GAN Attitude Control (with the GAN sync switch in the SYNC position)
- SCS Entry Monitor

Minimum impulse is used during spacecraft fine alignment, which has been previously discussed. The ARS is configured so that it may follow the maneuver keeping the gyros nulled and maintaining a valid attitude reference.

Channel Disable may be used to allow the astronaut to make long term minimum propellant consumption maneuvers. The rotation control is used to initiate a maneuver then the channel is turned off and the stick is released, the vehicle continues to rotate until the channel is again enabled.

During GAN entry, GAN Delta V, and GAN attitude control (with the GAN sync switch in the OFF position) the spacecraft attitude may be changed using direct or proportional control, but the GAN attitude signals will automatically maneuver the spacecraft.
ATTITUDE REFERENCE SUBSYSTEM
CONTROL STICK STEERING (CSS) = ATTITUDE SET "ON"

3.8.4 Backup Rate

The backup rate, sub-mode of the Attitude Reference Subsystem, figure 3-12, electronically cages any or all of the attitude gyro's so that they may be used in place of the rate gyro's to provide rate stabilization of the spacecraft. Placing an attitude gyro in backup rate mode will disable the backup reference capability of the AGCU. The selection of backup rate in any channel applies only the rate output of the selected gyro to the Attitude Control System for rate stabilization of the spacecraft, however, all the attitude gyro's will be caged electronically if a maneuver is performed while any gyro is caged. The backup reference capability of the ARS will be lost until the gyro's are uncaged and the AGCU/FGA1 is reinitialized.

During the GBN Delta V and SCS Delta V modes, the selection of a single channel of backup rate will electronically cage only the selected gyro or gyro's and apply their preamp output to the AGCU. The rate display will be driven by torquing amp outputs of the selected gyro's instead of preamp outputs. The selection of backup rate in two channels will apply the outputs of the gyro's in those channels to the Attitude Control System and FGA1. Backup rate capability is present in any channel during all SCS or GBN control modes.

3.8.5 Monitor Mode

The Monitor mode provides a means of monitoring spacecraft attitude utilizing the Guidance and Navigation System as a reference. Manual proportional control of spacecraft attitude and rate stabilization are available using the S/M reaction control system after SIVB staging. Monitor mode will be utilized primarily during launch and any time only a rate damping mode is desired. Figure 3-13 functionally shows the data flow during Monitor mode.

Crew utilization of information displayed on the Flight Director attitude indicator will depend primarily upon the program being utilized in the Apollo Guidance Computer. During Monitor mode, either the launch polynomial or fine align programs could be made available to the AGCU.

3.8.5.1 Launch Polynomial

If the launch polynomial is inserted in the Apollo Guidance Computer, the FGA1 Attitude Error Needles will display body axes attitude errors up to the maximum indicator reading of 15.0° in pitch and yaw, and 25° in roll. The attitude errors are generated by the angular difference between MU Gimbal angles and computer-computed attitude. The attitude ball and roll bug will indicate MU Gimbal angles and
3.8.5.2 Fine Align

If this configuration is used, the attitude error display will indicate IMU gimbals/spacecraft misalignment, which can be nulled by changing S/C attitude. S/C attitude, as displayed on the ball after fine alignment has been completed, would be referenced to the vehicle true inertial attitude.

3.8.5.3 Attitude Reference Subsystem Backup

The Attitude Reference System may be aligned to agree with initial conditions set up in the G&N program. During launch or translunar injection, any change in vehicle attitude will be sensed by the attitude gyro's. Since the display electronics and AGC have closed the control loop, the AGC will compute the total change in S/C attitude and record the change in attitude in the resolver shafts of the AGC. If an SCS mode were selected during launch, the attitude ball will show until the attitude (pitch and yaw) indicated under the NAV INDEX, and opposite the roll bug, will be a composite of the SCS initial alignment attitude plus any change in attitude caused by launch or translunar injection computer programs.

3.8.5.4 Monitor Mode Utilization

The monitor mode is a means of comparing the launch automatic commands and vehicle attitude. Therefore, during monitor mode, the SCS or G&N system have no means of controlling spacecraft attitude, except that the S/M RCS is available with the direct manual switch in the ON position and deflection of the rotation control to the hard stops. No other mode should be selected during the normal launch phase. Accuracy of the F&I during monitor mode will be dependent upon the accuracy of the G&N system. The attitude reference subsystem will follow G&N commanded changes in attitude. Its accuracy will be dependent upon the amount of maneuvering and the elapsed time between alignments.

3.8.6 G&N Attitude Control - G&N Entry Figure 3-16, 17

The selection of either G&N Attitude Control or G&N Entry modes will allow the guidance and navigation system to position the spacecraft to a new attitude and drive the F&I using automatic, semi-automatic, or manual commands to the stabilization and control system. These commands are:

A. Changing spacecraft attitude in response to commands from an AGC automatic program.

B. Crew commanded changes in spacecraft attitude using the AGC keyboard as an input device.

ATTITUDE REFERENCE SUBSYSTEM

G&N CONTROL MODES-FDAI/AGCU ALIGN

Figure 3-16
C. Spacecraft attitude changes commanded manually by placing the GAN system in CDU manual and repositioning the CDU set knobs, or by placing the GAN sync switch in the sync position and using the rotation control. (The latter type of manual command may only be used during GAN attitude control mode).

D. GAN Manual Entry

The FDI total attitude indicators are positioned by signals from the IMU which represent IMU gimbal angles. The command attitude error needles represent the magnitude of the difference between IMU gimbal angles and CDU position. If the CDUs are repositioned by the computer, an electrical signal would be generated to drive the command attitude error needles and to cause the C/M or S/M reaction jets to fire, maneuvering the spacecraft to null the generated error signal and repositioning the total attitude indicators so that when the maneuver is completed, information presented under the offset axes index and roll bug will agree with the information displayed on the CDU dials. As the maneuver is being accomplished, ARS attitude gyros sense the change in spacecraft attitude and cause the AGCU resolver shafts to be repositioned and update the attitude reference subsystem.

3.8.6.1 GAN to SCS Mode Change

If an SCS control mode is selected during the GAN maneuver, or after its completion, the FDI attitude indicators will slew until the information displayed under the NAV axes index is a composite of the aligned attitude, plus or minus the magnitude of all maneuvers accomplished during the GAN control mode.

3.8.6.2 Limitations

The FDI/AGCU may be aligned to the information displayed on the attitude set dials during any GAN control mode but the attitude indicators will not indicate the alignment process and the ATT SET switch will not be functional. No manual control stick maneuvers should be attempted in the GAN automatic entry mode. Automatic commands are subject to the GAN system accuracies and to within the 0.2 or 2.0 degree/second rate and the selected attitude deadbands. Manual commands are subject to the accuracy to which the FDI may be read.

3.8.7 GAN Delta V Mode

The GAN Delta V mode of operation, as shown in Figure 3-18, automatically commands S/C motion by controlling SPS engine ignition and gimballs, to control pitch, yaw, and roll attitude, and maintain thrust vector direction. The roll reaction jets are used to maintain roll attitude. The ARS portion of the SCS will provide backup attitude errors from the attitude gyros and backup attitude from the AGCU resolvers. The FDI will be driven by the GAN system and will indicate IMU gimbal angles on the attitude displays and the CDU body axes steering commands on the command attitude error needles. Rotational velocity will be derived from the
SCS Rate Gyro Package.

3.8.7.1 Operation

During a normal operating sequence, the vehicle will change attitude after SPS thrust on to correct for movement of the center of gravity, and to allow realignment of the thrust vector to an inertial thrusting plane. As vehicle attitude is varied, the attitude and attitude error displays on the FCAI will drive to indicate the magnitude of attitude change or error and the new vehicle attitude. Assume the IMU X axis is aligned in the thrusting direction and along the X body axis, and the CG is located on the X axis at the Y and Z coordinate intersection. The attitude indicated by the FCAI would be 0° pitch, 0° yaw, 0° roll (indicated at the body axes symbol), and zero attitude error and rotational velocity. The following sequence of events would occur if the CG were to suddenly move 2° in the X2 direction.

Note: The following explanation of the sequence illustrated in figure 3-19 is intended to provide a step by step analogy of a TVC operational sequence in a single axis.

3.8.7.2 Position 1

Initial conditions will be with the thrust vector through the CG and aligned on the X axis and

SELECTED THRUSTING PLANE, AS LISTED BELOW.

<table>
<thead>
<tr>
<th>INDICATOR</th>
<th>SOURCE</th>
<th>(PITCH)</th>
</tr>
</thead>
<tbody>
<tr>
<td>FCAI ATTITUDE</td>
<td>IMU</td>
<td>0°</td>
</tr>
<tr>
<td>FCAI ATTITUDE ERROR</td>
<td>CDU</td>
<td>0°</td>
</tr>
<tr>
<td>CDU ANGLE</td>
<td>CDU</td>
<td>0°</td>
</tr>
<tr>
<td>GIMBAL POSITION</td>
<td>Pitch Gimbal</td>
<td>0°</td>
</tr>
</tbody>
</table>

3.8.7.3 Position 2

The CG shifts 2° in the X2 direction and the vehicle begins a negative pitch rotation because the thrust vector is no longer through the center of gravity. The pitch display indications will begin to change, the pitch attitude error needle will deflect upward opposite the direction of vehicle rotation, the pitch rate needle will deflect downward in the direction of vehicle rotation, and the pitch gimbal position display will deflect upwards as steering signals applied to the TVC electronics change the SPS engine gimbal angle in the positive direction.

3.8.7.4 Position 3

The gimbal has now deflected sufficiently to bring the thrust vector through the CG and the vehicle stops rotating and continues accelerating in the thrust vector direction which is away from the selected thrusting plane. The SCS indicators could read as follows:
INDICATOR                SOURCE         (PITCH)
FDAI ATTITUDE            IMU             350°
FDAI ATTITUDE ERROR      CDU             -2° (UP)
CDU ANGLE                CDU             0°
GIMBAL POSITION          ENGINE          +4°  GIMBAL

3.8.7.5 Position 4

The IMU senses vehicle accelerations along the Z axis and provides this data to the Apollo Guidance Computer. The ASC performs a double integration on the acceleration signal and repositions the CDU shafts to generate additional attitude errors to drive the SPS pitch gimbal farther up, CG positive, and directs the thrust vector below the new CG location. At this time, the SCS indicator would display the readings shown below.

INDICATOR                SOURCE         (PITCH)
FDAI ATTITUDE            IMU             350°
FDAI ATTITUDE ERROR      CDU             3° (UP)
CDU ANGLE                CDU             001°
GIMBAL POSITION          ENGINE          +3°  GIMBAL

3.8.7.6 Position 5

The moment arm created by directing the thrust vector below the CG will rotate the vehicle towards the selected thrusting plane, causing the attitude error needles to begin to null as the vehicle rotates and the SPS pitch gimbal position indication will begin to move back towards the CG.

3.8.7.7 Position 6

The vehicle will continue to rotate until the thrust vector is directed parallel to the selected thrusting plane and aligned through the CG. The vehicle will then stabilize at this attitude until the Z axis acceleration has been nulled. If a coincidence angle between the thrust vector and selected thrusting plane is assumed to be four degrees, then the SCS indicators will display the information shown below.

INDICATOR                SOURCE         (PITCH)
FDAI ATTITUDE            IMU             6°
FDAI ATTITUDE ERROR      CDU             +2° (UP)
CDU ANGLE                CDU             8°
GIMBAL POSITION          ENGINE          +2°  GIMBAL

3.8.7.8 Position 7

When the ASC determines (using the Z accelerometer) that the vehicle is approaching zero Z axis velocity, it will reposition the CDU resolver shafts and generate attitude error that will drive the SPS gimbal to position the thrust vector above the center of gravity and rotate the vehicle downward.

3-36
3.8.7.9 Position 8

The vehicle is now stabilized with the thrust vector through the CG and parallel to the selected thrusting plane. The SCS indicators will now display information as shown below.

<table>
<thead>
<tr>
<th>Indicator</th>
<th>Source</th>
<th>(Pitch)</th>
</tr>
</thead>
<tbody>
<tr>
<td>FDI Attitude</td>
<td>HLU</td>
<td>2°</td>
</tr>
<tr>
<td>FDI Attitude Error</td>
<td>CDU</td>
<td>2° (IN)</td>
</tr>
<tr>
<td>CDU Angle</td>
<td>CDU</td>
<td>4°</td>
</tr>
<tr>
<td>Gimbal Position</td>
<td>Gimbal</td>
<td>2°</td>
</tr>
</tbody>
</table>

3.8.7.10 Selection of SCS Delta V Mode

If an SCS mode is selected during a G&N Delta V maneuver, the attitude ball will slew to the attitude indicated on the Attitude Set Dials of the AS/G&N. The attitude error needles will indicate the magnitude of the differential angle between the desired thrusting plane and the body X axis. The rate needles display the magnitude of rotation up to 3°/second. If the switch over was accomplished with the vehicle stabilized as in paragraph 3.8.7.7, the FDI Attitude Error would be the same magnitude as shown, but the polarity would be opposite.

3.9 SCS Attitude Control, SCS Delta V, and SCS Entry .05g’s (Figure 3-20)

During these modes, the information loop of the G&N is operating. The Attitude Display is fixed, and the attitude error and rate needles will indicate changes in vehicle attitude due to drift, and the angular velocity at which the vehicle is drifting. The overall operation of the G&N is similar to comparable G&N modes except that the Attitude Display Information is derived from the AMS rather than the G&N. Minimum Impulse, Channel Disable, Attitude Set, FDI Align, Control Stick Steering, and Backup Rate Switches may be used in any of the primary modes; however, each switch will alter the configuration to reflect the selected sub-modes. All of the above sub-modes, except FDI Align, Attitude Set and Backup Rate are directly connected with maneuvering the vehicle and will cause the AMS command or torquing loop to be energized so that the AMS may tract vehicle maneuvers. In summary, if the command or torquing loop is energized, the FDI ball moves to indicate vehicle inertial attitude and the attitude error needles indicate the difference between the indicated attitude and the body axes attitude error of the spacecraft. Selection of backup rate causes caging of the appropriate Attitude Gyro to

ATTITUDE REFERENCE SUBSYSTEM

SCS Attitude Control - SCS Delta V - SCS Entry .05g

Figure 3-20

![Diagram of Attitude Reference Subsystem](image-url)
ANGULAR VELOCITY FOR THAT CHANNEL AND DISABLES THE
TORQUING LOOPS OF THE AGCU.

3.10 SCS LOCAL VERTICAL

This mode utilizes an amplifier and voltage divider
with a switch for selecting nominal earth
orbit rate for a circular orbit with a pitch rate of 4.1°/minute.
Orbit rate may be selected prior to launch, according
to the mission to be performed. The local vertical
signal provides, in conjunction with the reverse
transformation circuit of the AGCU, an all attitude local vertical
capability. Figure 3-22 shows signal flow
through the AGU during local vertical, beginning in the
AGCU local vertical signal source. The error signal
is applied to the pitch gyro torquing circuit and torques
only the pitch attitude gyro. If the vehicle is maneuvered
in roll or yaw, after initial alignment, the
signal previously applied only to the pitch gyro will
be applied to all three gyro's, providing them with
sufficient torquing current to maintain a selected
local vertical attitude. In short, the torquing signal
induces a precision rate into the gyro's to compensate
for the apparent precession of the orbiting spacecraft.
Displayed attitude error will be a composite of the
local vertical signal and vehicle drift. These errors
should never exceed the selected SCS deadband.

3.11 FDIATTITUDE BALL POSITIONING LOOP

Figure 3-24 illustrates the gimbal drive
circuitry for a single channel of the FDI. Identical
circuitry is used in each of the other two channels.
Simply stated, the FDI gimbals are positioned through
a closed loop servo system which receives excitation
power and control inputs from either of two sources,
GAN or SCS. The logic networks shown on the left of
the diagram are the means of selecting the power and
signal source. The logic networks depicted on this
diagram are not fully mechanized, however, they are
functionally correct.

The GAN OR GATE AND RELAY DRIVER 1 CONTROL
RELAYS 1A3K1, K3, K4, K5, K6, 1A3K1, AND 1B3K1.
These relays select the signal and power source for
the amplifier demodulator, modulator, servo motor, and
velocity generator, and are energized only during GAN
control modes. The excitation frequency of the velocity
generator, amplifier demodulator, modulator and
motor is selected by contacts of 1A2K4, K5 and K6.
Contacts of 1A3K1 provide the proper input scaling for
GAN or AGCU input signals.

RELAYS 1A3K3 AND K5 CONTROL GAN SIGNALS TO
THE FDI. WHEN THEY ARE ON, GAN SIGNALS DRIVE THE
BALL. THE AGCU PROVIDES DIRECTIONAL SINES AND COSINES
POSITION THE BALL WHEN RELAYS 1A3K4 AND K6 ARE NOT
ENERGIZED.

DURING CERTAIN SCS MODES AND SUB MODS THE
FDI IS REPOSITIONED TO FOLLOW SPACECRAFT DRIFT. LOGIC
SIGNALS, GENERATED BY MODE SELECTION CAUSE RELAY
2B3K6 TO BE ENERGIZED, CONTACTS OF 2B3K6 CLOSE TO
ENERGIZE RELAY 1A2K4. IF SCS ATTITUDE CONTROL OR
LOCAL VERTICAL MODES ARE ENERGIZED, RELAYS 1A2K4, 5
AND 5 WILL ENERGIZE AND APPLY THE SELECTED REFERENCE FREQUENCY TO THE BALL POSITIONING LOOP.

SIGNAL FLOW IS THE SAME FOR EITHER GAN OR SCS NODES OF OPERATION. AN INPUT SIGNAL IS APPLIED TO THE STATOR OF A RESOLVER. ITS ROTOR IS IN TURN POSITIONED BY THE GIMBAL MOTOR. WHEN THE INPUT SIGNAL CHANGES VALUE, AN ERROR IS SENSED BY THE ROTOR. THIS ERROR IS SCALED, THEN AMPLIFIED AND DEMODULATED. THE DC ERROR IS THEN REMODULATED ON A 400 CPS CARRIER AND APPLIED TO AN INPUT TO THE SERVO AMPLIFIER. THE SERVO AMPLIFIER DRIVES THE MOTOR, WHICH REPOSITIONS THE ROTOR OF THE INPUT RESOLVER TO NULL THE INPUT SIGNAL. THE MOTOR ALSO DRIVES A VELOCITY GENERATOR TO PROVIDE A RATE DAMPING SIGNAL TO PREVENT SERVO OVERSHOOT AND ALSO CONTROLS THE BALL SLEW RATE TO A MAXIMUM OF 50°/SECOND.

3.12 ATTITUDE ERROR INDICATOR ELECTRONICS

THE ATTITUDE ERROR DISPLAY OF THE FDAI WILL INDICATE VARIATIONS BETWEEN ACTUAL AND DESIRED ATTITUDES FROM THE GAN SYSTEM OR THE SCS DEPENDING UPON THE NODE SELECTED.

FIGURE 3-25 SHOWS THE REFERENCE FREQUENCY SOURCES, THE SIGNAL INPUT METER DRIVE, AND LOGIC SWITCHING. THE REFERENCE IS USED TO DEMODULATE THE INCOMING ERROR SIGNAL WHICH MAY BE EITHER 800 OR 400 CPS. FOR THE GAN MODES, RELAYS 2IA3K3 AND 2IA3K3 WILL BE ENERGIZED TO SELECT THE 800 CYCLE REFERENCE FREQUENCY FROM THE GAN SYSTEM. WHEN DE-ENERGIZED, THESE RELAYS APPLY AN SCS 400 CPS REFERENCE FREQUENCY. THE RELAYS ARE ENERGIZED BY SELECTING A GAN CONTROL MODE. RELAYS 2IA3K3 AND 22A3K3, CONTROLLED BY THE SAME LOGIC INPUTS, ARE FOR ERROR SIGNAL SELECTION AND INHIBITING THE ATTITUDE SET FUNCTION DURING GAN NODES RESPECTIVELY. RELAY 2IA3K3 IS ENERGIZED TO REMOVE BNA G ATTITUDE ERROR INPUTS, WHEN A GAN OPERATING MODE IS SELECTED. THE LOGIC SHOWS THAT RELAY 2IA3K3 WILL BE ENERGIZED TO OBTAIN GAN BODY AXES ERROR INFORMATION FOR ALL OPERATING MODES EXCEPT GAN ENTRY. RELAY 22A3K2 IS ENERGIZED TO SELECT STABILITY AXES ERROR SIGNALS WITH GAN ENTRY AND 0.5° SELECTED. 0.5° LOGIC IS OBTAINED WHEN RELAY 22A3K6 IS DE-ENERGIZED. GAN SIGNALS ARE SCALING FOR MONITOR MODE BY RELAY 2IA3K1.

THE SCS ERROR SIGNALS ARE DERIVED FROM THE BNA G'S OR THE AGCU DEPENDING UPON THE CONDITION OF RELAYS 2IA3K4 AND 2IA3K2. WITH NO INPUTS TO THE CONTROL LOGIC, RELAY K2 WILL BE DE-ENERGIZED AND ALLOW ERROR SIGNALS TO ENTER THE DISPLAY FROM THE BNA G. WHEN THE ATT SET SWITCH, ON THE AS/SP PANEL, IS PLACED IN THE ON POSITION, RELAYS K2 AND K4 WILL BE ENERGIZED TO REMOVE THE BNA G ERROR SIGNALS AND INSERT NAV AXES ERROR SIGNALS FROM THE AGCU. THE ELECTRONICS AND SUMMING NETWORKS ARE USED TO CONVERT THE NAVIGATION AXES INFORMATION INTO BODY AXES ERRORS. THE ATTITUDE SET FUNCTION IS INHIBITED IF A GAN NODE IS SELECTED. THIS IS ACCOMPLISHED BY ENERGIZING RELAYS 22A3K4 OR 22A3K3.
WHEN THE BMG IS SWITCHED TO A RATE BACKUP MODE, THE LOGIC REMOVES THE BMG INPUT TO THE ERROR NEEDLE AND CONNECTS IT INTO THE RATE DISPLAY BY ENERGIZING RELAYS 21AX3K5 AND 21AX3K4. THE YAW ATTITUDE ERROR NEEDLE IS INOPERATIVE DURING YAW BACKUP RATE.

3.13 RATE INDICATOR ELECTRONICS

The rate indicator is supplied with rate information by signals derived within the SCS. Figure 3-26 shows the inputs for rate signals, power circuits, and the logic necessary to perform the required switching.

Power requirement, other than the DC power required to perform logic switching and for amplifier operation, is a 400 cycle reference frequency used to demodulate the incoming rate signals. Signal polarity is determined by the phase relationship between rate signal and the reference.

The two sources of reference frequency are the RATE GYRO POWER switch and the PARTIAL SCS POWER switch. Normal operation will select the reference frequency from the RATE GYRO POWER switch with RELAY K1 DE-ENERGIZED. Selection of any one of the three BMG's for backup rate operation will energize RELAY K1, which selects the reference frequency for the demodulator from the PARTIAL SCS POWER switch. This provides a capability of removing power from the RATE GYRO PACKAGE and using the BMG's as RATE GYROS FOR DISPLAY AND CONTROL PURPOSES.


Cross coupling is required between the roll and yaw axes during the entry phase of the mission, after -9 deg. sensing, to provide navigation axes rate damping for control of the C/M during entry. The cross coupling is performed by energizing RELAY K2.

RELAY K3 AND K6 ARE USED TO CHANGE AMPLIFIER DEMODULATOR GAIN AND THUS ACHIEVE THE DESIRED SCALE FACTOR ON THE RATE NEEDLES. THE RATE NEEDLES HAVE THREE FULL SCALE DEFLECTION VALUES: ± 1°/SECOND WITH K3 DE-ENERGIZED, ± 5°/SECOND WITH RELAY K6 ENERGIZED, AND ± 25°/SECOND WITH RELAYS K3 AND K6 ENERGIZED.

RELAY K2 IS ENERGIZED AFTER .056'S TO PROVIDE IMPEDANCE MATCHING FOR ENTRY ROLL TO YAW COUPLING.
4.1 INTRODUCTION

The Attitude Control Subsystem provides control of spacecraft motion. Referring to Figure 4-1, one sees that this control is accomplished by means of the reaction jets on either the Command Module or the Service Module. These jets are neither gimbaled nor throttleable. Consequently, commands to the Reaction Control System from the Attitude Control Subsystem consist simply of on-off signals. The reaction jets on the S/M are used for control prior to C/H-S/M separation, and the C/H reaction jets are used after separation.

The control functions of the Attitude Control Subsystem may be divided into four capabilities:

A. Rotational maneuvers, which are required for navigation sightings, alignment of the vehicle prior to Delta V's, antenna orientations for communication purposes, lift vector control during re-entry, etc.

B. Translational maneuvers, which are required for transposition, docking, and ullage.

C. Attitude hold, which is required in situations such as navigation sightings, docking, preparation for Delta V's, ullage, lift vector control during re-entry, etc.

D. Rate damping only, as opposed to attitude hold. This capability is required during re-entry and during an abort which occurs prior to LES jettison but after the canards on the LES have ceased to be effective due to rarified atmosphere.

4.2 CAPABILITIES OF THE ATTITUDE CONTROL SUBSYSTEM

The Attitude Control Subsystem provides the above mentioned capabilities by utilizing inputs from various sources. The astronaut has manual inputs by means of the translation and rotation controls, switches on the control panel, and the direct ullage switch on the Delta V display panel. These manual inputs are used for normal mission procedures as well as backup for normally automatic functions. Two inertial sensor packages, the Attitude GyroAccelerometer Package (AGAP) and the Rate Gyro Package (RGP), provide rate and attitude error signal inputs. These signals are used for attitude hold, rate damping, and manual rotational control. The manner in which they are used for these functions will be described in a later part of this section. The GAN system provides attitude error signals for attitude hold, and command rate signals for rotational maneuvers. The command rate signals from the GAN system.

ATTITUDE CONTROL SUBSYSTEM FUNCTIONS

PURPOSE: CONTROL OF VEHICLE MOTION

MEANS:
1. SM REACTION CONTROL SYSTEM
2. CM REACTION CONTROL SYSTEM

CAPABILITIES:
1. ROTATIONAL MANEUVERS
2. TRANSLATIONAL MANEUVERS
3. ATTITUDE HOLD
4. RATE DAMPING ONLY

FIGURE 4-1

SCS-212
Can be commanded manually as well as automatically. Finally, the attitude impulse controller on the gantry optics control panel provides manual inputs for commanding very small angular accelerations.

The attitude control electronics, which consists of three EGA's (pitch, yaw, and roll), accepts these inputs and gives output signals to the proper reaction jets to turn them on or off. The configuration of the attitude control electronics is dependent upon various mode and logic signals from the control panel, control sticks, and direct attitude switch. These signals change gains and signal paths within the attitude control electronics to modify the capabilities of the various control inputs.

A closer look (ref. figure 4-2) will now be taken at these inputs to the attitude control subsystem, together with the modes during which these functions are available. In paragraph 4.3 (mechanization of the attitude control subsystem), the switching and gain changes which take place in the attitude control electronics, as a result of the various logic inputs, will be covered on a block diagram level.

4.2.1 Rotational Maneuvers

Rotational maneuvers can be classified as manual or automatic. There are four ways for commanding rotations manually and one automatically. The four manual means will be described first, then the one automatic capability.

4.2.1.1 Manual Proportional Rotation

Either one or two rotation controls can be used for rotating the vehicle about one or more of the three body-referenced axes simultaneously. This capability (as well as direct rotation), is called control stick steering (CSS). The rotational rate achieved in any axis will be proportional to the magnitude of the stick displacement in that axis. However, the relationship between stick displacement and vehicle rate is a non-linear proportionality rather than a direct proportionality. The proportional control is achieved by utilizing the rate output of the rate gyro's (or BWAG's in backup rate) to cancel the transducer output of the control stick. During manual proportional rotation, the three BWAG's are kept caged through the AGCU and the attitude error inputs to the ACS electronics are inhibited. When the control stick is rotated approximately 2.5° from its neutral position in any axis, a CSS breakaway switch is activated, giving a CSS logic signal to the EGA's that accomplishes the casing and inhibiting functions mentioned above.

The maximum rates which can be commanded by this type of control are approximately:

A. 17 deg/sec during entry modes (roll axis)
B. 5 deg/sec during entry modes (pitch and yaw axes)
B. 0.65 deg/sec during all other modes (pitch, yaw, and roll axes)
CSS is available during all modes.

4.2.1.2 Direct or Emergency Rotation

The solenoid valves controlling the reaction jets contain two sets of coils, automatic and direct. Energizing either coil in a solenoid valve will operate the valve and turn on the jet. Direct rotation, direct ullage, and separation ullage use the direct coils. All other control functions use the automatic coils.

Direct rotation can be commanded about one or more axes simultaneously by moving the rotation control all the way to its hard stops in the appropriate axis. A switch is closed applying the direct command to the direct coils of the RCS about 1° before the stick hits the hard stops. This control is non-proportional since the jets keep firing as long as the direct switches are closed. There is no feedback signal to cancel this direct command. The three BNA's are also caged and the attitude error inputs to the EC's inhibited during direct rotation by the signal from the CSS breakout switches.

Direct control is available at any time throughout the mission, providing only that the direct circuit breakers are in, the direct mode switch is on, and the necessary power circuits as well as the direct coils of the RCS are functioning properly.

4.2.1.3 Attitude Impulse

The attitude impulse switch operates in three axes similar to the rotation control except that it gives only one pulse to the reaction jets when it is displaced. This enables the astronaut to command very low accelerations, and consequently achieve low angular rates of the vehicle.

Attitude impulse control is available only in SCS attitude control or GAN attitude control mode.

4.2.1.4 Manual Positioning of the CDU's

This method will not be the normal way for commanding rotations, but it is included since it is nevertheless possible. There are three CDU shafts in the GAN system which can be rotated by hand or automatically by the Apollo Guidance Computer. When one or more shafts are rotated by hand, using either the three slew switches and the thumbwheels which are provided on the CDU's, or the AGC keyboard, a command signal (proportional to the difference between the CDU shaft angle and the corresponding IMU gimbals angle) is generated which commands a vehicle rate about the proper axes. The command signals from the GAN system pass through a limiter circuit in the SCS which limits the maximum rates which can be commanded by the GAN system to the same values for maximum possible rates using manual proportional rotation.

4.2.1.5 Automatic Positioning of the CDU's

This control is similar to manual positioning of the CDU’s except that the Apollo Guidance Computer (AGC) rotates the CDU's automatically. This function would be used during the GAN entry mode. That is, the AGC would have an entry trajectory programmed into it, and at the proper times would automatically rotate the CDU's properly control the lift vector and hence the profile of the re-entry trajectory.

4.2.2 Translation Maneuvers

There are three ways for commanding translations manually, and none automatically. These are described below.

4.2.2.1 Normal Translation Using the Translation Control

Either one of two translation controls can be used for commanding translational accelerations along one or more of the body referenced axes simultaneously. The change in velocity which is achieved is non-proportional since the hand control merely closes switches which turn on jets giving a constant acceleration until the hand control is released or the fuel is expended.

This control is available during all modes until C/M-S/M separation, at which time it is unavailable for the rest of the mission.

4.2.2.2 Direct Ullage

By means of the direct ullage button on the Delta V display, a positive X axis acceleration can be commanded. This command uses the direct coils of the RCS and is available any time, providing the direct circuit breakers are in and the direct coils of the RCS are functioning properly.

4.2.2.3 Separation Ullage

A counterclockwise rotation of the translation control handle into a detent position (closing a switch) which commands either an SPS abort or SIV-B separation, will also command an ullage maneuver, using the direct coils of the RCS. This ullage is semi-automatic. If the hand control is rotated counterclockwise and returned to neutral, a 2.5 second ullage will result. If the controller is kept twisted for 2.5 seconds or more, the ullage will last until the controller is returned to neutral.
This function is available after the LES system has been jettisoned, the Master Event Sequence Controller (MESC) circuit breakers are in, and the MESC Logic Arm Switch is armed.

4.2.2.4 Additional Functions of the Translation Control

In addition to commanding translations, the translation control has a few other capabilities which will be described at this point. During the launch phase of a mission, a counterclockwise twist of the hand control will initiate abort sequences when the launch escape system is attached, and will cause SIV-B separation if the LES has been jettisoned.

Furthermore, during any mode, if the translation control is rotated clockwise, closing a switch, a CSS logic signal is generated which removes the attitude error and command rate signals (from either the GAN or SCS system) from the CSS electronics in all three axes, and also causes the BAOG's through the AGCU. This enables the astronaut to manually control the vehicle during the GAN entry mode, which normally provides automatic attitude control (in the roll axis) by providing the CSS with a command rate signal. If the SCS is in either GAN or SCS Delta V mode, the clockwise twist of the translation control also initiates manual thrust vector control. In addition to removing the attitude error signals from the CSS electronics and causing the BAOG's through the backup rate loop, both the clockwise and counterclockwise rotations of the translation control handle move the handle into a detent position which holds the handle in that position.

4.2.3 Attitude Hold

The spacecraft can hold a particular orientation in space about all three body-referenced axes. It will oscillate approximately ± 3° or ± 0.5° (depending on whether the SCS is in the maximum or minimum deadband configuration) in each axis about the particular orientation which it is holding. If the SCS is in either GAN or SCS entry modes, attitude hold will use the maximum deadband regardless of the position of the deadband select switch. There is no attitude hold capability in monitor mode.

The attitude hold capability is mechanized by using the sum of attitude error signals and rate signals. Whenever the spacecraft rotates away from its reference position by an amount exceeding the deadband value, the sum of the rate and attitude error signals turn on the proper jets to rotate the vehicle back within its deadband.

The attitude error signal for a particular axis is proportional to the magnitude of the angular displacement of the spacecraft (in that axis) from the orientation in space at which the attitude reference systems are nullled. The attitude error signals are generated by the three BAOG's in the uncaged configuration (in SCS attitude control, local vertical, Delta V, and entry modes), or by the difference between the angular positions of the three CDU shafts and the three gimbal axes of the IMU (in GAN attitude control, Delta V, or entry modes) in the GAN system.

The rate signal, in a particular axis, is proportional to the angular velocity of the spacecraft about that axis. The rate signals are generated by the rate gyro's (or the BAOG's in a backup rate configuration), whether in SCS or GAN control modes. The rate signals are added to the attitude error signals to provide for stable oscillations about the reference orientation.

Attitude hold about all three axes is automatically in effect during all modes except

A. Monitor mode
B. SCS entry mode after the .05g switch is turned on
C. GAN entry mode after the .05g switch is turned on; however, in this mode, it is still in effect about the body-offset roll axis

An additional comment should be made concerning attitude hold during the local vertical mode in which the spacecraft holds approximately a constant attitude with respect to the earth. This means that its actual attitude with respect to an inertially fixed reference frame must be continuously changing. This is accomplished by torquing the BAOG's at a fixed rate, such that the inertial attitude of the spacecraft, defined by the BAOG's null position, is kept changing in the proper manner.

4.2.4 Rate Damping Only

In addition to the ability of the SCS for providing attitude hold and maneuvering capabilities, the SCS can also provide rate damping only. That is, although rate damping is used for stability of oscillations during attitude hold, rate damping can also be used to limit the rotational rates of the spacecraft even when not holding a particular attitude. This capability, as has been previously indicated, would be used during aborts and re-entry. The outputs of either the rate gyro's or the BAOG's (during backup rate) are used to mechanize this capability. During monitor mode, (which would probably be used for aborts utilizing the Launch Escape System) if vehicle rates in any axis exceed ±2 degrees/sec., the outputs of the rate gyro's will turn on the reaction jets to decrease vehicle rates below ±2 degree/sec. deadband. During either SCS or GAN entry modes, the rate deadband is ±2 degrees/sec.

At this point, a few words will be said on the backup rate function. The three backup rate switches on the control panel can only be turned on or off BAOG's, turning them into rate gyro's, and applying their outputs, which would now be proportional to
VEHICLE RATES, TO THE SCS ELECTRONICS. IF BACKUP RATE IS CHOSEN IN ANY ONE CHANNEL, THE OTHER TWO BWAG’S ARE CAGED IF THE SPACECRAFT IS BEING MANEUVERED BY MEANS OF CSS OR ATTITUDE IMPULSE CONTROL. FURTHERMORE, THE OTHER 2 BWAG’S ARE ALSO CAGED WHEN ONE BACKUP RATE SWITCH IS PLACED ON IF:

a. The SCS is in monitor, local vertical, or gan attitude control, or gan entry modes.
b. The SCS is in gan entry mode and the .05g switch is on.
c. The SCS is in gan entry mode and either the pitch, yaw, or both roll channels have been disabled by means of the channel enable switches.

Rate damping only is available in gan entry mode. SCS entry mode after the .05g switch is turned on, gan entry mode (pitch and yaw channels) after the .05g switch is turned on, and anytime the translation control handle is rotated clockwise, generating a css logic signal.

4.3 MECHANIZATION OF THE ATTITUDE CONTROL SUBSYSTEM

The first part of this section has described the various capabilities of the attitude control subsystem and the modes during which they would be available. The remainder of this section will be devoted to describing (mostly at a block diagram level) the actual mechanization of these capabilities.

Figures 4-3 and 4-4 schematically present the general signal flow and functions associated with the SCS attitude control system. In the block diagrams, the signal flow proceeds from left to right. Except for roll to yaw coupling after 0.05g in the yaw channel (upper left hand corner of the diagram) and corresponding jet numbers for the pitch channel (jets 1 thru 4), the pitch and yaw channels are basically identical. Therefore, one diagram has been adapted to describe the typical functioning of either channel. Since figure 4-3 utilizes jets 5, 6, 7, and 8, it represents the action of the yaw channel.

Figure 4-4 represents the typical roll channel. Since its function is similar to pitch and yaw on the general signal flow level, only important differences will be pointed out in the description of that channel.

Most of the electronics which mechanize the block diagrams, given in figures 4-3 and 4-4, for the attitude control subsystem, are contained within the roll, pitch, and yaw ECA’s. The inputs to these channels include reaction jet control signals (or SPS pitch and yaw gimbal control signals during delta V thrusting) at the output. Logically, the inputs will be covered first, with a transition through the signal flow and end in a discussion of the reaction jet function. Most of the inputs to the attitude control channel appear...
ATTITUDE CONTROL SUBSYSTEM

(ROLL CHANNEL)

ON THE LEFT HAND SIDE OF THE DRAWING. THEY ARE:

A. ATTITUDE ERRORS FROM THE UNCASED AGAP GYROS.
B. ATTITUDE ERROR OR RATE COMMANDS FROM THE GAS SYSTEM.
C. PROPORTIONAL RATE COMMANDS FROM EITHER OF THE TWO ROTATION CONTROLS.
D. VEHICLE BODY RATES FROM THE SCS RATE GYROS OR FROM THE CASED AGAP GYROS DURING BACKUP RATE.

OTHER COMMAND INPUTS TO THE SIGNAL FLOW ARE:

A. ATTITUDE IMPULSE FROM THE ATTITUDE IMPULSE CONTROLLER (LOWER CENTER OF DRAWING).
B. TRANSLATION COMMANDS FROM THE TWO TRANSLATION CONTROLS (TOP CENTER OF DRAWING).
C. DIRECT COMMANDS FROM THE DIRECT SWITCHES OF THE ROTATION CONTROLS (LOWER RIGHT OF DRAWING).
D. SEPARATION ULLAGE FROM THE TWO TRANSLATION CONTROLS VIA THE MISSION SEQUENCER, WHEN COMMANDING A SERVICE MODULE ABORT OR S1VB SEPARATION) AND DIRECT ULLAGE FROM THE DELTA V DISPLAY (LOWER RIGHT OF DRAWING).


4.3.1 ATTITUDE HOLD

ATTITUDE OF THE SPACECRAFT CAN BE HELD BY THE UNCASED SCS ATTITUDE GYROS (DMAG's) IN THE SCS MODES OR BY GAS INPUTS IN GAS MODES. IN EITHER CASE, THE ATTITUDE ERROR SIGNAL IS APPLIED TO ITS RESPECTIVE DEMODULATOR. THE OUTPUTS OF THE DEMODULATORS (DEMOD) ARE APPLIED TO A COMMON DC AMPLIFIER VIA RELAYS K2 AND K9, WHICH SELECT GAS ATTITUDE ERRORS IN GAS MODES, AND SCS ATTITUDE ERRORS IN SCS MODES.

DEADBAND SELECTED, THE ERROR IS APPLIED TO THE COMMAND RATE LIMITER. THIS CIRCUIT LIMITS THE APPLIED ATTITUDE ERROR SIGNAL TO AN EQUIVALENT RATE SIGNAL OUTPUT OF APPROXIMATELY .65°/SEC. FOR NON-ENTRY MODES AND 5°/SEC. FOR ENTRY MODES. THE OUTPUT FROM THE COMMAND RATE LIMITER IS REMOVED (GROUNDED) BY RELAY K4, WHEN THE .050 Switch IS TURNED ON, PROVIDING THE SCS IS IN EITHER SCS OR GAN ENTRY MODES. THEN, AFTER .050 Switching, PITCH AND YAW ATTITUDE CONTROL OF THE S/C ARE REMOVED WHETHER IN A GAN OR SCS ENTRY MODE. HOWEVER, AERODYNAMIC FORCES ENCOUNTERED AFTER .050 WILL CONTROL THE ATTITUDE OF THE VEHICLE IN PITCH AND YAW, WITH RESPECT TO APPROXIMATELY A 22° OFFSET ORIENTATION OF THE SPACECRAFT (WEATHER-DOCKED IN THE AIRSTREAM). THE COMMAND RATE LIMITER OUTPUT IS ALSO REMOVED BY A CSS SIGNAL WHICH CAN BE GENERATED EITHER BY CSS ITSELF OR A CLOTHING TWIST OF THE TRANSLATION CONTROL HANDLE.

FROM THE COMMAND RATE LIMITER THE ERROR SIGNAL BECOMES AN INPUT TO THE SUMMING POINT FOR THE NEXT DC AMPLIFIER. AT THIS SUMMING POINT, THE ATTITUDE ERROR SIGNAL SUMS WITH THE RATE GYRO OUTPUT (OR DRAK DURING BACKUP RATE WHEN IN A GAN MODE). IF THE VEHICLE IS HOLDING AN ATTITUDE WITHIN THE SELECTED DEADBAND, THERE WILL PROBABLY BE NO APPRECIABLE RATE SIGNAL FROM THE RATE GYRO BECAUSE OF THE LOW VEHICLE DRIFT RATE. WITH RESPECT TO A RATE SIGNAL, RELAY K5 CHANGES THE GAIN OF THIS AMPLIFIER BY A FACTOR OF TEN FOR ENTRY MODES. THE REASONS FOR THIS IS THAT THE TRANSLATION AMPLIFIER HAS A TRIGGERING THRESHOLD OF 2 VOLTS, WHICH CORRESPONDS TO A RATE INPUT OF .2°/SEC. DURING NON-ENTRY MODES. BY DECREASING THE GAIN OF THE SUMMING AMPLIFIER BY 1/10 DURING ENTRY MODES, THE RATE DEADBAND IS INCREASED TO 2°/SEC., WHICH RESULTS IN A FUEL SAVINGS. HOWEVER, WITH RESPECT TO OUR ATTITUDE ERROR SIGNAL, RELAY K5 DECREASES THE GAIN OF THIS AMPLIFIER BY 1/5 DURING ENTRY MODES, SUCH THAT THE ATTITUDE ERROR DEADBAND IS STILL APPROXIMATELY 5°.

THE OUTPUT OF THE SUMMING AMPLIFIER, UPON BEING APPLIED TO THE SWINGING AMPLIFIER, TURNS IT ON WHEN THE 2 VOLT THRESHOLD IS EXCEEDED. DUE TO POSITIVE FEEDBACK WITHIN THE SWINGING AMPLIFIER, IT TURNS FULL ON, GIVING AN OUTPUT OF APPROXIMATELY 20 VDC. THE SWINGING AMPLIFIER TRANSFORMS THE INPUT SIGNAL TO A SERIES OF WIDTH AND FREQUENCY MODULATED PULSES DUE TO A PSEUDO-RATE FEEDBACK CIRCUIT. THESE PULSES ARE INPUTS TO THE JET SELECTION LOGIC WHICH DETERMINES WHICH JETS ARE TO FIRE TO OBTAIN DESIRED ATTITUDE DIRECTION. A COMPREHENSIVE COVERAGE OF THE JET SELECTION LOGIC (WITH OR WITHOUT A TRANSLATION INPUT) IS COVERED IN FIGURE 4-7 AND WILL NOT BE COVERED IN THE GENERAL SIGNAL FLOW DESCRIPTION. THE PSEUDO-RATE SIGNAL GENERATES A VARIABLE JET ON TIME BY VARYING BOTH PULSE WIDTH AND REPETITION RATE AS A FUNCTION OF SIGNAL MAGNITUDE. TOX, FOR A LARGE INPUT SIGNAL THE OUTPUT PULSE (OR PULSES) WILL CAUSE A JET TO FIRE MORE OFTEN AND FOR A LONGER PERIOD OF TIME. AS THE ERROR SIGNAL DECREASES, PULSE WIDTH AND REPETITION RATE DECREASE, THUS CREATING A MINIMUM IMPULSE CONDITION TO BRING THE S/C CLOSE TO A ZERO RATE CONDITION.

THE DIAGRAM SHOWS THAT THE SWINGING AMPLIFIER HAS TWO MODES OF OPERATION; WITH OR WITHOUT PSEUDO-RATE FEEDBACK. THE PSEUDO-RATE FEEDBACK LOOP IS

ALWAYS EMPLOYED EXCEPT DURING CSS, TRANSLATION MANEUVERS, OR WHEN THE LIMIT CYCLE SWITCH ON THE CONTROL PANEL IS IN THE OFF POSITION. THIS INHIBITING FUNCTION FOR PSEUDO-RATE IS ACTIVATED BY THE OUTPUT OF THE CSS BREAKOUT SWITCHES ON THE ROTATION CONTROL DURING CSS. DURING TRANSLATIONAL MANEUVERS, IT IS THE OUTPUT OF THE TRANSLATION CONTROL WHERE THIS PSEUDO-RATE ACTION OFFERS NO REAL FUEL SAVINGS DURING CSS, TRANSLATION MANEUVERS, OR ENTRY MODES AFTER .050, AND CAN CONSEQUENTLY BE REMOVED DURING THESE TIMES.

WITH PSEUDO-RATE FEEDBACK OPEN, THE SWINGING AMPLIFIER OUTPUT REMAINS AT APPROXIMATELY 420 VDC AS LONG AS THE INPUT IS ABOVE ± 2 VDC.


FROM THE JET SELECTION LOGIC, A 428 VDC SIGNAL IS SENT TO THE APPROPRIATE JET DRIVER AMPLIFIERS. WHEN THE JET DRIVER FOR A PARTICULAR SOLENOID COIL IS TURNED ON BY THE 28 VDC SIGNAL FROM THE JET SELECTION LOGIC, A GROUND IS SUPPLIED TO ONE SIDE OF THE SOLENOID COIL (WHICH HAS 28 VDC APPLIED TO THE OTHER SIDE), CAUSING CURRENT TO FLOW IN IT, THUS OPENING THE SOLENOID VALVE AND TURNING ON THE JET. SINCE IT TAKES APPROXIMATELY 7 MILI-SECONDS FOR A FUEL VALVE AND 9 MILI-SECONDS FOR AN OXIDIZER VALVE (AND POSSIBLY UP TO ABOUT 12 MILI-SECONDS FOR THESE VALVES) TO OPEN, THE CURRENT THROUGH THE COIL SHOULD LAST AT LEAST THIS LONG TO AVOID FOILING OF THE JET. THIS MEANS, CONSEQUENTLY, THAT A LATCHING CIRCUIT IS NEEDED FOR THE JET DRIVER, BECAUSE THE OUTPUT OF THE JET SELECTION LOGIC IS A SERIES OF PULSES WHOSE DURATION COULD BE LESS THAN 10 MILI-SECONDS FOR SMALL INPUTS TO THE SWINGING AMPLIFIER. THIS LATCHING CIRCUIT IS CALLED THE MINIMUM IMPULSE FEEDBACK CIRCUIT AND SHOULD NOT BE CONFUSED WITH THE MANUAL ATTITUDE IMPULSE CONTROL LOCATED IN THE GAN SYSTEM. THE LATCHING CIRCUIT KEEPS THE JET DRIVER ON FOR A MINIMUM OF 18 ± 1 MILI-SECONDS, THUS ASSURING THAT EACH SINGLE PULSE OUTPUT OF THE SWINGING AMPLIFIER WILL TURN ON THE APPROPRIATE JETS. FURTHERMORE, THIS LATCHING CIRCUIT IS Always IN EFFECT, WHETHER IN AN ATTITUDE HOLD CONFIGURATION OR NOT.

4.3.2 Rotational Manuevers

Rotational maneuvers can be commanded by means of manual proportion control, rotation of the CDU's in the GAN system, direct rotation, or attitude impulse.

4.3.2.1 Manual Proportional Rotation

For manual proportional rotation, the AC output of the control stick transducer, which is proportional in magnitude to the magnitude of the stick displacement, is demodulated to a positive or negative DC voltage, depending upon whether a positive or negative rotation is commanded. The signal then passes...
through relay K10, (which is closed whenever the SPS engine is not firing) to a voltage divider network, after which it is summed with the rate signal. The voltage divider network causes a voltage drop such that the signal from a maximum stick displacement is cancelled by a rate output equal to about .55°/sec. During non-entry modes, and 5°/sec during entry modes. The sum of the stick and rate outputs are amplified, and if greater than 2 volts at the input to the switching amplifier, turn it on, thus activating the appropriate jets to change the vehicle rate. When the rate is large enough, the rate signal will cancel the control stick signal and turn off the jets, leaving the vehicle rotating at this rate. When the stick is moved to its neutral position, the rate signal will slow the vehicle down within the rate deadband.

4.3.2.2 Rotation Of the CDU’s

When the CDU’s (in the GAN system) are rotated, they generate signals proportional to the differences between their angular position and the angular positions of the IMU signals. These signals are converted from the Euler angle reference body-axes coordinates and fed into the RCS where they are demodulated to a dc voltage. If the RCS is in an entry mode, the signals are converted from an Euler angle reference to the body-offset coordinates, and then demodulated. If the RCS is in a GAN mode, relay K2 then allows the signal for the yaw channel to be amplified, fed through the deadband circuit and command rate limiter, after which it is summed with the rate signal.

The command rate limiter puts a maximum limit on the magnitude of the output signal from the deadband circuit which is summed with the rate signal. For non-entry modes, the limiting is such that the maximum output of this circuit can be cancelled by approximately a .05 deg/sec rate signal. During entry modes this limiting is to 5 deg/sec. Thus, a GAN command signal which is less than the maximum value for the limiter will command a proportional vehicle rate in a similar fashion to manual proportional control. For a GAN command signal larger than the maximum output of the limiter circuit, the vehicle will only achieve the limiting rates given above.

4.3.2.3 Direct Rotation

Direct rotation can be accomplished anytime that the direct circuit breakers are in and the direct mode switch on. By rotating the rotation control to within about 1 degree of the hard stop, a switch is closed applying 24 VDC via cable connection to the direct coils of the proper jet. In the roll channel, there are soft stops just before the direct switches on the rotation control. However, the pitch and yaw axes of the rotation controls do not have these soft stops.

4.3.2.4 Attitude Impulse

The attitude impulse controller operates in three axes similar to the rotation control, except that its output is not a proportional output but a plus or
4.3.4 Rate Damping Only

Rate damping only is accomplished during GAN entry (pitch and yaw channels) or SCS entry (all three channels) modes by grounding the attitude error signal with relay K4 when the OSW switch is turned on. During any mode, rotating the translation control handle clockwise will generate a CSS logic signal which also grounds the attitude error signal by means of relay K4, thus giving rate damping only. Rate damping only is accomplished during monitor mode by inhibiting the attitude error signal, from either the GAN 8MAG or the SCS system, by means of relays K2 and K3. The rate signal can come either from the rate gyro or, if in backup rate, from the cage 8MAG. After the OSW switch is turned on, the roll to yaw coupling circuit is used, because after .05 sec the 8CG will experience roll maneuvers around the stability roll axis rather than the body roll axis. Consequently, the body axis oriented yaw rate gyro will sense a component of this roll action. The output of the yaw rate gyro would fire yaw jets in response to the sensed rate. Thus, by coupling an out-of-phase component of the body roll rate (tangent 20°) into the yaw channel, the sensed body yaw component of the stability roll maneuver is nullified to prevent undesirable yaw jet firing.

4.3.5 RCS Enabling Functions

It can be seen in Figure 4–3 that the jet drivers and solenoid coils have 28 VDC enabling power inputs. These inputs do not turn the jets on, but are necessary for enabling the jets to be turned on by a command signal. The availability of the various attitude control subsystem capabilities (such as manual proportional rotation, attitude hold, etc.) during the different modes has previously been mentioned. Nevertheless, although a particular capability may be present during a given mode, its availability is also dependent upon whether the jet drivers and solenoid coils are enabled. The jets can be disabled by disabling either the coils themselves, the jet drivers, or both.

Considering just the coils, the MAN A circuit breaker for the yaw axis must be in and the yaw channel enable switch on for the number 7 and 8 jets to be enabled. The MAN B circuit breaker enables the number 5 and 6 jets. Actually, these numbers apply to the service module only. On the command module, the MAN A circuit breaker enables jets 5 and 6, whereas the MAN B circuit breaker enables jets 6 and 7. In the pitch axis, there is no difference in the numbering for the G/N and S/N jets.

Considering the jet drivers, there are some relays and a switch controlling the enabling of the yaw jet drivers. These are the RCS latching relays, the channel enable switch, and relay K7.

4.3.5.1 RCS Latching Relays

The RCS enabling relays within the master event sequence controller, must be closed if the jets are to be enabled. Up until 42 seconds after launch, the RCS latching relays cannot be closed. After this time, they can be activated in several ways.

If an abort is commanded prior to LES JETTISON, these relays are closed 1 second after commanding the abort. They are disabled 15 seconds later if, at that time, the command module is at an altitude less than approximately 24,000 feet. However, if at that time the command module is higher than approximately 24,000 feet, the relays stay enabled until the vehicle descends below 24,000 feet, at which time they are disabled. After prior to LES JETTISON can be commanded automatically by the emergency detection system (EDS) or manually by twisting the translation control handle counterclockwise.

After LES JETTISON, a counterclockwise twist of the translation control handle will command SIVB separation. Following a 2.5 second delay, this counterclockwise twist will also enable the RCS latching relays. In the above cases, as well as the LES abort, the OX CAB-RCS CMD (ref. Figure 6-12) switch must be in the RCS CMD position in order that the RCS latching relays be closed either by the EDS or translation control counterclockwise switch signals.

Providing that 42 seconds have elapsed following launch, the RCS latching relays can be closed immediately at any time by momentarily placing the RCS CMD switch (directly below the OX CAB-RCS CMD switch) to the ON position. Furthermore, they can be disabled at any time by placing this switch in the OFF position.
Finally, during re-entry from either a SM abort or normal mission, the RCS latching relays are opened, disabling the yaw drives, at about the 24,000 foot altitude.

4.3.5.1 Channel Enable Switch and Relay K7

In order that the yaw jet drives be enabled, the yaw channel switch must be on and relay K7 closed. Relay K7 is open, disabling the yaw jets during SPS engine firing (Actually the disabling occurs approximately 1 second after engine ignition, and lasts until about 1 second after the engine off signal is given.), direct voltage (or separation voltage), or direct yaw rotation.

4.3.6 Roll Channel Signal Flow

Since the roll signal flow (Figure 4-4) is basically the same as the previously described pitch and yaw channel, only the important differences will be listed. They are:

A. Maximum manual proportional or automatic roll rate command is approximately 0.5°/sec. for non-entry modes and 1°/sec. for entry modes.

B. The output of the roll command limiter is not grounded at 0.56 unless in the SGR entry mode. In GRN entry mode, roll attitude can be automatically controlled from the GRN system by having the AOC automatically rotate the CDU's generating a roll rate command.

C. The output of the rotation control is not removed in the roll channel during SPS engine firing.

D. The RCS is not disabled in the roll channel during direct voltage or SPS engine firing.

E. For attitude impulse control in the roll channel, either the AOC or B&D channel should be disabled with the proper channel enable switch. If this were not done, four jets instead of two would fire for a given roll attitude impulse command, thus decreasing the ability for commanding a small rotational acceleration about the roll axis.

4.3.7 Translation and Rotation Control Outputs

Figure 4-5 shows the outputs of the number 1 rotation and translation controls. The transducers on the rotation control which are used for proportional rate control receive AC bus 1 or 2 power depending upon the position of the rotation control switch. The direct switches and the CSS breakout switches on rotation control number 1 receive DC bus A power, whereas rotation control number 2 receives DC bus B power. The direct switches are used for direct rotation control in all three body axes of the vehicle. The CSS breakout signal is used for caging the BWAG's, removing
PSEUDO-PACE, AND INHIBITING THE ATTITUDE ERROR INPUTS TO THE ATTITUDE CONTROL SUBSYSTEM.

On the translation control, the clockwise switch generates a CSS signal which accomplishes the same function as the breakout switches on the rotation control. In addition, if the SCS is in a Delta V mode, it also initiates manual thrust vector control. Both the clockwise switch and the translation control switches on translation control number 1 receive dc bus A power whereas control number 2 receives dc bus B. However, the counterclockwise switch which provides the abort or SIV-B separation command uses both dc busses on each of the two controls. Finally, a push to talk switch is provided on the T handle. This switch simply provides a ground for activating relays within the Communications System.

4.3.8 G&N Sync Function

The sync function is necessary to prevent the S/C from rotating back to the initial attitude at the end of a CSS maneuver while in G&N attitude control mode. This action normally would occur because of the attitude error generated as the spacecraft is maneuvered about the G&N stable platform (IMU). During the maneuver the follow-up action of the sync function however, will keep this attitude error signal at zero. The S/C will, upon removal of the CSS function, hold the new attitude within the limits of the selected deadbands.

The details of the synchronizing action are shown in figure 4-6. The signal affected by the sync function is labeled, ATTITUDE ERROR SIGNALS. This is the G&N attitude error signal processed by the SCS electronics to control attitude during G&N modes. In a G&N attitude hold condition, this output signal represents the difference angle (attitude error) between the IMU gimbal angle and the CDU shaft angle. When these angles are equal, a zero attitude error condition exists. While in an attitude hold condition, the spacecraft rotates about the stable platform (which maintains a constant inertial orientation) causing the IMU gimbal angles to change. Since the CDU shaft angles remain stationary, an error signal is generated which, when it exceeds the selected attitude deadband, will cause the SCS to fire the appropriate reaction jets to rotate the spacecraft and reduce this error below the deadband limits. During CSS, without a synchronizing function, an attitude error would be generated as the spacecraft were maneuvered. Reaction jets would not fire as this error were built up because the attitude error signals are inhibited in the SCS during the CSS maneuver. When the CSS command is removed at the end of the maneuver, the attitude error would be present and the SCS would then fire jets as long as the attitude error is greater than the selected deadband. Only when the spacecraft is again returned to an attitude matching the initial CDU-IMU condition would the error signals be eliminated.

In figure 4-6, relays K-1 and K-4 control the signal flow paths during the G&N sync operation. K-1 is energized by fulfilling the two input requirements to its associated and gate. These two conditions are: the G&N ATT CONT mode must be selected; and a

G&N SYNC FUNCTIONAL DIAGRAM

IMU

GIMBAL ANGLE

RT

ATTITUDE ERROR SIGNALS TO ECA’S AND FDAI

G/N SYNC SW

INCRCNC

AGC

ATTITUDE COMMANDS

G/N SYNC SW

INPUT BITS

K1

RD

CSS

G/N ATT CONT
CSS SIGNAL INITIATED (CSS BREAKOUT SWITCHES CLOSED OR TRANSLATION CONTROL HANDLE ROTATED COUNTERCLOCKWISE). THEN IF THE GN SYNC SWITCH IS IN THE ON POSITION, A GROUND IS PROVIDED THROUGH THE ENERGIZED CONTACTS OF K-1 TO ENERGIZE K-4. WITH K-4 ENERGIZED, ANY OUTPUT ERROR SIGNAL PRESENT FROM THE CDU RESOLVERS IS APPLIED TO THE SERVO AMPLIFIER TO DRIVE THE CDU MOTOR AND NULL THE ERROR. Thus, as an error is generated between the IMU GIMBAL ANGLES AND THE CDU SHAFT POSITIONS DURING THE MANEUVER, THIS SAME ERROR WILL DRIVE THE CDU TO KEEP THE ATTITUDE SIGNAL AT ZERO. WHEN THE CSS SIGNAL IS REMOVED, K-1 IS DE-ENERGIZED OPENING THE CIRCUIT TO DE-ENERGIZE K-4. THE CDU IS CONSISTENTLY ELECTRICALLY LOCKED (WITH A ZERO ATTITUDE ERROR SIGNAL) AT THE TIME OF CSS REMOVAL. UNDER THESE CONDITIONS, ANY VEHICLE ATTITUDE CHANGE (DRIFT) WOULD AGAIN GENERATE AN ATTITUDE ERROR SIGNAL FROM THE CDU'S TO HOLD THE S/C WITHIN THE SELEEDED DEADBANDS AT THE NEW ATTITUDE. THE GN SYNC FUNCTION IS ONLY POSSIBLE IN THE GN ATTITUDE CONTROL MODE. CONSEQUENTLY, A CSS MANEUVER IN ANY OTHER GN MODE WILL RESULT IN THE S/C RETURNING TO THE INITIAL ATTITUDE UPON TERMINATION OF THE CSS SIGNAL.

4.3.9 JET DRIVER CONTROL LOOP

NOW THAT THE CAPABILITIES OF THE ATTITUDE CONTROL SUBSYSTEM HAVE BEEN DESCRIBED, TOGETHER WITH THE MANNER IN WHICH THESE CAPABILITIES ARE MECHANIZED, A MORE DETAILED DESCRIPTION OF THE JET SELECTION LOGIC AND ITS OUTPUTS TO THE RCS WILL BE GIVEN. THE MECHANIZATION OF THE RCS ENABLING CIRCUITRY WILL ALSO BE SHOWN.


THE OUTPUT OF THE SWITCHING AMPLIFIERS IS EITHER A PULSED OR STEADY 20 VDC SIGNAL ON ONE OF TWO LINES; ONE FOR POSITIVE AND THE OTHER FOR NEGATIVE ROTATIONAL COMMANDS. EACH OF THE TWO OUTPUT LINES OF THE SWITCHING AMPLIFIERS IS SPLIT, ONE BRANCH GOING THROUGH AN INVERTER TO GIVE A "NOT ROTATION" LOGIC SIGNAL. THE ROTATION COMMAND OUTPUTS OF THE SWITCHING AMPLIFIERS ARE THEN FED INTO THE SOLID STATE JET SELECTION LOGIC ALONG WITH THE OUTPUTS OF THE TRANSLATION CONTROLS. AFTER C/M - S/M SEPARATION, HOWEVER,

FIGURE 4-7
There can no longer be any input signals to the jet selection logic from the translation controls, since the DC power is removed from the translation controls at separation. Prior to separation, therefore, the translation and rotation commands are applied to AND gates and OR gates (within the jet selection logic) as necessary to energize the appropriate reaction jet drivers.

Since the translation commands and the rotation commands may be applied simultaneously, it is necessary to establish a priority system. Table 4-2 provides the basis for the priority system for two simultaneous commands according to the reaction jet numbering system shown in Figure 4-8 for the C/M-S/M. Notice that for translation maneuvers in the Y and Z direction, two jets per axis are used. In the X direction, four jets are used to provide additional thrust for ullage. The jet selection logic provides a means for maintaining attitude control during translation without allowing two opposing jets to be fired. Rotation maneuvers in the roll axis require either two or four jets to be fired depending upon whether all four or only two S/M clusters are operative (AEC or B&D). The selection of the operative cluster pairs is accomplished by the channel enable switches on the control panel, as shown in Figure 4-4. After the C/M is separated from the S/M, the switch controlling AEC has no function and only the B&D roll switch can disable the C/M roll jets.

The enabling functions for the jet drivers are indicated in Figure 4-7. All reaction jet driver amplifiers are normally disabled by the RCS latching relays until SIV-B separation, at which time these relays are closed. They are opened again during re-entry upon reaching the 24,000 foot altitude. They can also be closed during launch if an abort occurs, as was described previously. The RCS C/M switch (located directly below the go/keep switch) on the crew safety control panel provides manual backup capability for this switching, if necessary.

During thrusting of the SPS engine, relays K-1 and K-2 are energized to remove power from the jet selection logic driver amplifiers in the yaw and pitch axes since control of the spacecraft's attitude is accomplished with the gimbaled engine. As has been mentioned previously, these relays are energized about 1 second after engine ignition and de-energized about 1 second after the engine off signal is given. However, roll jets will remain operable throughout the thrust period.

Relays K-1 and K-2 are also energized with a direct (or separation) ullage signal (UE) to disable the automatic pitch and yaw RCS. Since the direct ullage command is not processed through the automatic RCS jet selection logic, opposing jets could be fired by the automatic RCS if that system were not disabled.
ENGINE LOCATION

If the direct reaction control switches on the rotation control are actuated, relays K-1, K-2, or K-3 (or all three) may be energized, and the power to the appropriate jet driver amplifiers is removed by axis, provided that the direct mode switch has been placed in the on position.

At C/M-S/M separation, control of the spacecraft is transferred to the C/M reaction jets by applying 28 volts dc and jet selection commands to the C/M jet solenoids rather than the S/M jet solenoids. This transfer is accomplished by SEP Sw A and SEP Sw B. Roll Jets 13, 14, 15, and 16 are deadfaced since these roll jets are not present on the C/M. A detailed discussion of the separation switches (A and B) is developed under control loop heading.

4.3.10 Direct Control Loop

A general description of the direct reaction jet control has already been given. A more detailed description of its mechanization will now be given, together with a description of the C/M-S/M separation switching functions and the fuel dump and burn function.

The direct control loop (Figure 4-9) provides manual direct rotational control by using the direct fuel and oxidizer solenoids of the C/M and S/M reaction jets. The direct control loop also provides the capability to perform a direct ullage maneuver by using the X axis direct solenoids of the S/M reaction jets. No backup translation maneuver is provided through the direct control loop except in the X axis as noted above.

After arming the direct rotation capability using the direct mode switch, angular acceleration commands may be applied to the spacecraft. Since the commands are direct to the reaction jets with no rate feedback as in manual proportional rotation, the spacecraft accelerates as long as the direct switches in the rotation control are closed. After the control is released and it returns to neutral, the spacecraft will continue to rotate until an opposite rate command is applied by the astronauts. If the jet driver control loop were also operative, the rate gyro would cause the appropriate reaction jets to fire until rotational rates were within the deadband.

Direct ullage is initiated by the direct ullage pushbutton on the Delta V display. This switch bypasses the rotation control switches and applies 28 volts dc directly to the S/M X direct solenoid coil. Present separation switching allows the direct ullage commands to be applied to C/M jets 1, 2, 6, and 7 if the direct ullage pushbutton is pressed after separation from the S/M. Since no ullage maneuver is possible or necessary after C/M-S/M separation, this capability should create no apparent problem. Separation ullage can be commanded by a counterclockwise rotation of the translation control handle commanding either SIV-B separation for a service module abort.
DIRECT CONTROL LOOP

(AFTER THE LES TOWER HAS BEEN JETTISONED), OR S-IVB SEPARATION FOLLOWING ORBIT ACQUISITION. THIS VEHICLE IS SEMI-AUTOMATIC IN THAT IT WOULD LAST 2.5 SECONDS IF THE CONTROL HANDLE WERE TWISTED TO THE CCW POSITION AND THEN RETURNED TO THE NEUTRAL POSITION. IF THE CONTROL WERE KEPT ROTATED TO THE CCW POSITION FOR MORE THAN 2.5 SECONDS, THE VEHICLE WOULD LAST UNTIL THE CONTROL IS RETURNED TO NEUTRAL OR UNTIL 12 SECONDS PASSED.


COINCIDENT WITH THE C/M-S/M SEPARATION ASSOCIATED WITH MOTORS A AND B ON THE C/M, IS THE C/M-S/M SEPARATION SIGNAL APPLIED TO THE SEPARATION SEQUENCER UNIT IN THE S/M. UPON RECEIPT OF THE C/M-S/M SEPARATION SIGNAL (T=0), THE SEQUENCER APPLIES A CONTINUOUS FIRE SIGNAL TO S/M JETS 3, 4, 7, AND 8. SINCE THESE ARE THE +X TRANSLATION JETS, THEIR ACTION WILL CAUSE THE S/M TO TRANSLATE AWAY FROM THE C/M. THESE JETS CONTINUE TO FIRE AFTER SEPARATION UNTIL THEIR PROPELLANT SUPPLY IS EXHAUSTED OR THE JETS HAVE BURNED OUT. TWO SECONDS AFTER THE +X COMMAND HAS BEEN GENERATED, A SIX SECOND FIRE SIGNAL IS APPLIED TO THE POSITIVE ROLL JETS (9, 11, 13, AND 15) BY THE SEQUENCER.

A FINAL TIE-IN TO THE DIRECT RCS IS THAT OF THE FUEL DUMP BURN FUNCTION. ACTIVATING THE FUEL DUMP SWITCH WILL CAUSE TWO MOTOR DRIVER SWITCHES TO BE CLOSED. AGAIN THE A AND B MOTORS ARE ASSOCIATED WITH THE SWITCHING ACTION AFFECTING THE A AND B PROPELLANT SYSTEMS OF THE C/M. A DECISION TO RID THE C/M OF EXCESS RCS FUEL AFTER SCS DEACTIVATION, CAUSES THE FUEL TO BE BURNED OUT OF ALL C/M JETS EXCEPT JETS 1 AND 3. SINCE THESE ARE POSITIVE (+) PITCH JETS THAT POINT UPWARD TOWARDS THE APEX OF THE C/M, THEIR BURN ENVELOPE COULD ENDANGER THE PARACHUTES AND ASSOCIATED SHROUD LINES.
SECTION V
THRUST VECTOR CONTROL

5.1 INTRODUCTION

Since the moon and the earth are moving in orbits, it is necessary to launch the Apollo spacecraft so that the spacecraft and the moon arrive at a pre-selected point in space at the same time. It is conceivable to obtain errors in the trans-lunar injection trajectory so that the preselected point could be missed. A ground-based IBM 7090 computer precomputes the precise trajectory which should be followed, based on a four body equa-tion of motion that takes into account the effect of the gravitational field of the earth, moon, and sun on the vehicle’s motion.

The mid-course navigation system (MNS) takes over after the spacecraft is injected into the chosen trajectory. The MNS computer calculates the degree of deviation from the reference trajectory; and in determining the deviation, the computer will calculate the required velocity correction to establish a new trajectory that will rendezvous at a preselected point in space. Successful achievement of the calculated trajecto-ry is dependent upon control of the vehicle thrust vector during these velocity correction phases.

The Apollo spacecraft, operating in a frictionless void, experiences a varying center of gravity caused by the consumption of expendables. The varying center of gravity causes undesirable moment arms and side velocities during the accelerating phases; and if uncompensated, they become detrimental to the success of the mission and safety of the crew.

5.1.1 THRUST ALIGNMENT

The thrust vector control subsystem is required to position the gimbaled SPS engine. It main-tains the engine thrust through the spacecraft center of gravity, and holds the spacecraft attitude so that the thrust vector will be in the desired direction.

Positioning the thrust vector through the center of gravity is complicated by the amount of expend-ables consumed during large thrusting maneuvers (lunar orbit injection and trans-earth injection). The total travel of the spacecraft center of gravity (Figure 5-1 and 5-2) is approximately 200 inches along the x-axis, 12 inches along the y-axis and 7 inches along the z-axis. The spacecraft center of gravity travel during the large thrusting maneuvers causes undesired lateral accelerations and residual lateral velocities; and they combine with initial alignment errors of the thrust vector and pointing errors in the two inertial sensors (IMU and SCS ARS) to decrease the accuracy of the thrusting maneuver in most incidents.
The thrust vector control system consists of two axes (pitch and yaw) control of the gimbaled SPS engine and roll control provided by the reaction jets when roll attitude errors or angular rates exceed the specified deadband. The thrust vector control systems ability to command a maximum gimbal displacement of 25° in yaw about a null offset of 94° in the spacecraft X-Y plane, and 35° from zero in the spacecraft X-Z plane, allows the spacecraft to thrust through the center of gravity and in the desired direction.

The mechanical travel of each actuator is less than the maximum commandable by the TVC subsystem. The yaw axis is a nominal 7° ± 5° with an additional 1° for snubbing and the pitch axis is a nominal 30° ± 5° with an additional 1° for snubbing. The spacecraft and gimbal elasticity should absorb the extreme travel conditions that permit stability for large error signals that could occur in aborts. The normal TVC commanded gimbal angles should be slightly less than the nominal gimbal travel in each axis.

5.1.2 Primary Delta V Mode

The primary Delta V mode (thrusting) will be the GAN Delta V mode which provides the most accurate trajectory corrections. The IMU in the GAN system has three accelerometers which permit continuous trajectory corrections while thrusting to compensate for any lateral accelerations or residual lateral velocities. The GAN computer integrates the lateral accelerations and provides steering commands to the gimbaled SPS engine for trajectory corrections. When calculating the velocity required to correct the trajectory, the GAN computer considers the gravitational accelerations effecting the spacecraft; therefore, the actual thrusting velocity will be of a different magnitude than the required velocity change and result in a curved trajectory.

5.1.3 Backup Delta V Mode

The backup Delta V mode will be the SCS Delta V mode which has the ability to perform inertially oriented thrusting maneuvers by controlling the SPS thrust vector. The normal action for a malfunction in the Delta V mode would be to terminate thrust -- investigate and re-align to complete the Delta V maneuver. However, there are limitations and restrictions that require individual consideration. The backup mode can be selected to complete the Delta V maneuver where it provides greatest crew safety in an uncertain situation when sufficient information is available.

5.1.4 Manual Delta V Mode

The manual thrust vector control operating mode provides a secondary backup control of the gimbaled SPS engine. The astronaut will provide command inputs from the Rotation Control to position the gimbaled SPS engine for this operating mode. For Block I missions, manual thrust vector control will be used.
5.2 THRUST VECTOR CONTROL

The thrust vector control functional diagram (Figure 5-3) illustrates the functional capabilities of the Q/N Delta V mode, SCS Delta V mode, and the manual thrust vector control mode.

5.2.1 Q/N DELTA V MODE

For the Q/N Delta V mode, the inputs from the Q/N system position the FDI ATTITUDE BALL and ATTITUDE errors for display through relay Ki. Relay Kb will also be closed to provide command inputs for the ATTITUDE CONTROL ELECTRONICS and the THRUST VECTOR CONTROL ELECTRONICS. Prior to THRUST ON, the inputs will be used to hold the S/C in all three axes of ATTITUDE CONTROL. After THRUST ON is commanded, ROLL ATTITUDE ERROR ONLY will be recognized and acted upon by the ATTITUDE CONTROL ELECTRONICS FOR HOLDING THE SELECTED ROLL ATTITUDE. Relay Kii will close for THRUST ON TO SUPPLY PITCH AND YAW RATE and ATTITUDE ERROR signals into the gimbal POSITIONING ELECTRONICS TO POSITION THE GIMBLED SPS ENGINE for control of the SPS THRUST VECTOR. Relay Kio will open, removing all proportional rate inputs from the ATTITUDE CONTROL ELECTRONICS for PITCH and YAW. The ATTITUDE ERROR input from the Q/N SYSTEM will provide the required change in the SPS GIMBAL POSITION which will ROTATE THE SPACECRAFT TO COMPENSATE FOR THE
CENTER OF GRAVITY TRAVEL WHILE THRUSTING. WHEN THIS
ATTITUDE ERROR IS USED TO "FLY" A CURVED TRAJECTORY,
THE ATTITUDE ERROR INPUTS WILL REFLECT A STEERING COM-
MAND IN ADDITION TO THE ATTITUDE ERROR REQUIRED TO
COMPENSATE FOR THE CENTER OF GRAVITY TRAVEL. THEREFORE,
THE GAN SYSTEM WILL BE REQUIRED TO PERFORM THE CURVED
SPS THRUSTING MANEUVER.

THE GIMBAL POSITIONING ELECTRONICS AND GIMBAL
ACTUATORS PROVIDE A CLOSED LOOP SERVO THAT WILL BE
TRIMMED PRIOR TO ALL THRUSTING PHASES. THE GIMBAL
POSITION DISPLAY AND CONTROL KNOBS WILL BE USED TO TRIM
THE CLOSED SERVO LOOP FOR THRUSTING THROUGH THE CENTER
OF GRAVITY AT THRUST INITIATION. ALL INPUTS FROM THE
GAN SYSTEM WILL MOVE THE ACTUATOR ABOUT THE INITIAL
TRIM POINT. INITIAL TRIM CONDITIONS ARE ALWAYS MANU-
ALLY SET BY THE ASTRONAUTS FOR ALL OPERATING MODES. THE
SPS ENGINE GIMBALS ARE ALIGNED TO THE S/C BODY AXIS
FOR THE ASCENT PHASE.

THE GAN DELTA V MODE AFFECTS THE BODY MOUNTED
ATTITUDE GYROS TO COMPLETE THE SCS BACKUP CAPABILITY.
THE GYROS WILL BE UNCASED UPON SELECTION OF THE GAN
DELTA V MODE SO THE GAN AND SCS WILL HAVE THE SAME INER-
TIAL REFERENCE NULL FOR A MINIMUM TRANSIENT SWITCH OVER
IF REQUIRED WHILE THRUSTING.

5.2.2 SCS DELTA V MODE

IN THE SCS DELTA V MODE, THE FDDI ATTITUDE
BALL WILL BE POSITIONED FROM THE ATITUDE GYRO COUPLING
UNIT THROUGH RELAY K5 IF A MANUAL MANEUVER IS COMMANDED
PRIOR TO ENGINE IGNITION. THE ATTITUDE ERRORS WILL BE
SUPPLIED BY THE BODY MOUNTED ATTITUDE GYROS THROUGH
RELAY K12 WITH THE ATTITUDE SET SWITCH IN THE OFF
POSITION. RELAY K6 WOULD BE CLOSED AND K12 OPEN WHEN MAN-
EUVERING TO THE THRUSTING ATTITUDE USING THE COMMAND
ATTITUDE ERROR OUTPUT OF THE ATTITUDE GYRO COUPLING
UNIT. THE ATTITUDE SET SWITCH WOULD BE IN THE ATT SET
POSITION TO UTILIZE THIS CAPABILITY BUT IT WOULD BE
PLACED TO THE OFF POSITION PRIOR TO THRUST INITIATION.

THE ATTITUDE RATE AND ERROR SIGNALS WOULD BE
PROVIDED TO THE THRUST VECTOR CONTROL ELECTRONICS
THROUGH RELAY K7. THE THRUST VECTOR CONTROL ELECTRON-
ICS WILL BE CONFIGURED SUCH THAT THE THRUST VECTOR WILL
REMAIN INERTIALLY ORIENTATED FOR A DELTA V MANEUVER
BECAUSE OF THE ATTITUDE ERROR INPUT SIGNAL AND THE
INTEGRATION PERFORMED ON THE ATTITUDE ERROR AND POSITION
FEEDBACK SIGNALS IN THE THRUST VECTOR CONTROL ELECTRON-
ICS.

THE SAME SWITCHING FOR RELAYS K11 AND K10
WILL OCCUR AT THRUST INITIATION IN THE SCS DELTA V
MODE AS OCCURRED IN THE GAN DELTA V MODE. ATTITUDE
HOLD WILL REVERT TO ROLL ONLY, AS BEFORE WITH SIGNALS
PROVIDED TO THE JET SELECTION LOGIC IN THE ATTITUDE
CONTROL ELECTRONICS.

5.2.3 MANUAL THRUST VECTOR CONTROL MODE

THE MANUAL THRUST VECTOR CONTROL (MTVC) MODE
IS NOT A SELECTABLE MODE FROM THE SCS CONTROL PANEL.
IT CAN BE ENTERED ONLY IF THE CONTROL PANEL HAS A
DELTA V MODE SELECTED. WITH A DELTA V MODE SELECTED,
THE CLOCKWISE SWITCH ON THE TRANSLATION CONTROL WILL
ON COMMAND IF THE LOGIC NETWORK HAS BEEN PROPERLY EN-
ABLED. ON RECEIPT OF THE GAN THRUST ON COMMAND, THE
SCS ELECTRONICS WILL CAUSE THE SPS ENGINE SOLENOID
VALUES TO ENERGIZE, THEREBY INITIATING A VELOCITY CHANGE
MANEUVER. AFTER THE DESIRED VELOCITY HAS BEEN
OBTAINED, THE GUIDANCE COMPUTER WILL REMOVE THE THRUST
ON COMMAND WHICH WILL BE INTERPRETED BY THE THRUST ON-
OFF LOGIC AS A THRUST OFF COMMAND. THEREFORE, IF THE
PROPER CREW PROCEDURES ARE FOLLOWED, THE GAN SYSTEM
CAN AUTOMATICALLY TURN THE SPS ENGINE ON AND OFF.

5.3 THRUST CONTROL

THRUST CONTROL MEASURES AND CONTROLS THE
MAGNITUDE OF VELOCITY IMPARTED TO THE S/C BY THE SERVICE
PROPULSION SYSTEM. WITH PROPER SEQUENCE, THRUST FROM THE
SERVICE PROPULSION SYSTEM CAN BE APPLIED OR REMOVED
IN THREE METHODS AS SHOWN IN THE LOWER SECTION OF
FIGURE 5-3.

5.3.1 GAN THRUST ON-OFF

THE APOLLO GUIDANCE COMPUTER, AFTER PROPER
SEQUENCES AND PROGRAMS, CAN SUPPLY A THRUST ON COMMAND
TO THE SERVICE PROPULSION SYSTEM THRUST ON-OFF ELECTRON-
ICS. THE THRUST ON-OFF LOGIC WILL ACCEPT A GAN THRUST
5-7

5.3.2 SCS THRUST ON-OFF

THE SCS NORMALLY PROVIDES BACKUP CONTROLS AND
DISPLAYS TO MONITOR OR REPLACE, IF NECESSARY, GAN
THRUST ON-OFF COMMANDS TO THE SPS ENGINE. THE
MONITORING FUNCTION IS ACCOMPLISHED BY THE DELTA V
REMAINING INDICATOR AND ILLUMINATION OF THE THRUST ON
BUTTON. THE COMMAND WILL BE ACCEPTED BY THE SPS THRUST
ON-OFF LOGIC. IF THE PROPER SEQUENCE OF EVENTS HAVE
OCCURRED, THE SPS ENGINE SOLENOIDS WILL BE ENERGIZED TO
START A THRUSTING MANEUVER. WHEN THE DESIRED AMOUNT OF
VELOCITY HAS BEEN IMPARTED TO THE S/C, THE DELTA V
REMAINING INDICATOR WILL READ ZERO AND INITIATE AN
AUTOMATIC THRUST OFF COMMAND. THE SPS ENGINE SOLENOIDS
CAN NOT DISTINGUISH BETWEEN THE SYSTEMS ORIGINATING
THE THRUST ON COMMAND, BUT THE THRUST CONTROL SWITCH
MUST BE IN THE NORMAL POSITION BEFORE THE SOLENOIDS
CAN BE ENERGIZED BY THE THRUST CONTROL LOGIC.
THE NORMAL AND DIRECT-ON SIGNAL PATH PROVIDES 28
VDC FOR ARMING THE SPS ENGINE.

5-8
Solenoids from the thrust control switch. The DIRECT ON position also provides a ground to energize the solenoids. The DIRECT OFF position provides a backup thrust OFF command capability to both automatic thrust OFF commands and is the normal OFF for the DIRECT ON position.

5.3.3 Direct Thrust On-Off

The direct thrust on-off function provides a path to energize the SPS engine solenoids with no electronics. It requires manual action to unlock the switch and provide the direct thrust on command, and manual action to perform the thrust off command. The direct on will provide the power and ground connection required to energize the SPS engine solenoids.

5.4 Thrust Vector Control Operation

A typical thrust vector control channel is shown in figure 5-4. It is drawn for the yaw channel. The pitch channel would be the same.

5.4.1 Servo Positioning

The servo positioning loop consists of velocity position feedback loops to control rate and displacement of the actuator. The servo positioning is a closed servo loop with an external open loop trim. The open loop trim is composed of a manual gimbal position control and gimbal position display. Manual crew action provides loop closure for the initial SPS engine gimbal alignment. The servo positioning loop must be active to hold the SPS engine gimbal stationary and prevent external moments from inflicting motion. The mechanical action will withstand small forces, but the magnetic clutches provide the required holding forces. The magnetic clutches will have a quiescent current output of equal level for CW and CCW rotations of the gear trains in the static conditions. For input positioning commands, a differential current output will be generated to produce correct gimbal drive for the gimbal actuators. Torque output from the gimbal actuation will be proportional to the differential current level.

Manual crew action is always required to position the SPS engine gimbal prior to each thrusting of the SPS engine. The closed loop servo positioning actually provides the initial stability for thrust buildup.

The actuator assembly used by the servo positioning loop, figures 5-5 and 5-6, consists of two DC motors, four electro-magnetic clutches, two rate generators, and four linear position transducers for interfacing with the TVC subsystem. The actuator assembly also contains all the mechanical gearing and linkage required to position the SPS engine gimbal.

The upper section of figure 5-6 contains one motor, two clutches, and a rate generator which are identified as system 1. The lower section contains an identical set that is identified as system 2 within the actuator. System 2 is positioned by the redundant servo positioning electronics.

Figure 5-4
The clutch housing rotates as a function of motor speed and the TVC subsystem energizes the electromagnetic clutches to transmit required torque to the center shaft by action of the magnetic particles in the clutch. The rate generator and position transducer provides the electrical feedback for the closed servo loop. Table 5-1 defines gimbal display, thrust vector pointing, and actuator movement as related to trim command input.

5.4.1.1 Redundant Servo Positioning

Each channel of the thrust vector control has redundant servo positioning electronics and motors. The primary electronics and motor will normally be used to position the SPS engine gimbals. Power distribution to the motors and electronics is designed to avoid single point failures in power distribution from disabling the servo positioning loop.

There are three methods of placing the redundant electronics and motor (M2) into operation. First, with both M1 and M2 operating, there is an automatic switch over from M1 to M2 should motor current exceed the high or low specified tolerances. Second, if M1 is not activated, the switch over will occur and M2 will generate the controlling torques. Third, the manual thrust vector control operating mode will automatically select M2 and the redundant electronics to generate the controlling torques. It is assumed that SCS power switches for TVC 1 and 2 are procedurally turned on and connected to different buses for TVC operations.

The TVC monitor signal is generated by the over-under current sensing circuit shown in Figure 5-7 to provide the automatic switch-over for actuator current drain malfunctions. The motor control switch is activated by the actuator motor switch on panel 1. When the switch is placed to the START position, the motor switch will be driven to the energized position for the DC actuator motor. The motor switch contact opens and will not send a fail sense signal to the SCS. When the panel switch is released, it will drop to the OFF position and apply power to the over-under current sensing circuit. If an over or under current condition should occur, one SCR will illuminate the yaw gimbal drive fail master caution light and the other will start the motor switch to remove the DC actuator motor from its DC bus. When the motor switch is positioned to FULL OFF, the switch contact closes and provides 20 volts to the SCS for a TVC monitor signal, which automatically causes the SCS to select the redundant gimbal positioning loop. The master caution light will remain illuminated until the panel switch is positioned to OFF.

Motor No. 2 has an identical over-under current sensing circuit to protect the DC bus. It has no signal outputs to the SCS or master caution lights in event of an over or under current malfunction; therefore, no switching or warning circuits are provided.

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GIMBALING OF THE SERVICE PROPULSION ENGINE

<table>
<thead>
<tr>
<th>SIC Motion Desired by the Crew</th>
<th>Throttle Wheel Positioned on Panel Assembly to A</th>
<th>Engine Nozzle Moves Towards the</th>
<th>The Thrust Vector of the Engine Moves Towards the</th>
<th>Gimbal Position Indicator on Panel Assembly to A</th>
<th>Actuator RPM</th>
</tr>
</thead>
<tbody>
<tr>
<td>SIC Nose Down (pitch)</td>
<td>Numerical (UP)</td>
<td>+ Z Axis of the SIC</td>
<td>- Z Axis of the SIC</td>
<td>+ Numerical (pitch)</td>
<td>Pitch Extends</td>
</tr>
<tr>
<td>SIC Nose Up (pitch)</td>
<td>Numerical (DOWN)</td>
<td>- Z Axis of the SIC</td>
<td>+ Z Axis of the SIC</td>
<td>- Numerical (pitch)</td>
<td>Pitch Retracts</td>
</tr>
<tr>
<td>SIC Right (yaw)</td>
<td>Numerical (Right)</td>
<td>+ Y Axis of the SIC</td>
<td>- Y Axis of the SIC</td>
<td>+ Numerical (yaw)</td>
<td>Yaw Extends</td>
</tr>
<tr>
<td>SIC Left (yaw)</td>
<td>Numerical (Left)</td>
<td>- Y Axis of the SIC</td>
<td>+ Y Axis of the SIC</td>
<td>- Numerical (yaw)</td>
<td>Yaw Retracts</td>
</tr>
</tbody>
</table>

The actuator null position in yaw is 0° offset from the SIC X axis due to the engine mounting pads in the service package being cantilevered (block 1 vehicles). The thrust vector is 4° to the +Y axis of the SIC and the engine nozzle is 4° to the -Y axis of the SIC.

As a result of the gimbal position indicator on panel assembly A it will indicate A-4° when the actuators are in the null position.

To parallel the thrust vector and the engine nozzle with the X axis of the SIC, the yaw throttle wheel on panel assembly A must be positioned to 0° and as a result the yaw position indicator on panel assembly A will indicate 0°.

In this case, the yaw actuator is no longer in its null position in respect to the gimbal ring.

The thrust vector and the engine nozzle are now parallel to the X axis of the SIC.

The actuator null position in pitch is 0° (block 1 vehicles).

---

TABLE 5-1

P-109A ( + )
5.4.1.2 Servo Amplifier Limiting

Current limiters are incorporated around the servo amplifier for protection of the amplifiers. The current limiter permits a linear output up to 540 mA and then changes the output slope by gain control until a maximum output of 350 mA is reached. The output slope becomes zero and will have no change in the output current for an increased command signal input.

5.4.1.3 Relay Switching

Operation of the primary gimbal position loop requires no relay switching. All relays are drawn in the normally de-energized state with the function required to energize a relay adjacent to that relay. The primary position loop requires that one gimbal actuator motor be on to generate controlling torques and the TVC I power switch be on to provide amplifier operation before the SPS gimbal can be held stationary. The redundant actuator motor and TVC electronics must be on to provide redundant SPS gimbal hold ability.

5.4.2 Gimbal Position Trim

The gimbal position trim input is for manually positioning the SPS engine gimbals prior to each Delta V maneuver. It is processed through two DC amplifiers before biasing the gimbal positioning loop. The gimbal position display provides a readout of the SPS engine gimbal for verification of the trim position.

There are two thrust vector gimbal position trim pots connected to the control knob. One supplies the normal TVC trim requirements and the second provides a trim control for the manual thrust vector control mode. The gimbal position trim control biases the servon position loop for the desired displacement at engine ignition. Inputs from the inertial sensors in the SCS or GN Delta V mode and from the rotation control in the manual thrust vector control mode will position the SPS engine gimbal about the trim position. The position limiter will control the magnitude of the total gimbal command input to the servo amplifier.

5.4.3 Command Signal Inputs

There are three inertial sensors that can provide command signals into the servo positioning loop for normal Delta V maneuvers. The rotation control inputs will be used as command inputs only for the manual thrust vector control mode. The inertial sensor inputs will not become active inputs until the thrust on command is generated from the SPS thrust control logic to energize relay KG. Except for manual thrust vector control, the only command input active before the thrust on (engine ignition) signal is generated will be the gimbal position trim.

5.4.3.1 Rate Gyro Input

For the GN and SCS Delta V mode, the yaw rate gyro will provide the required stability to oppose any sudden angular motion. The rate gyro signal is processed through body bending filters to provide rate stability, but an attitude error or steering command input is required to correct for the center of gravity.
TRAVEL AND THRUST VECTOR MISALIGNMENT WITH RESPECT TO THE CENTER OF GRAVITY. An alternate source of rate can be obtained from the BMAG if it is placed in a backup rate condition (BUC) and energizes relays K4 and K8. Loss of the attitude error output capability from the BMAG would also remove the capabilities of the SCS Delta V mode.

5.4.3.2 GAN INPUT COMMANDS

The GAN inputs represent attitude error inputs to the thrust vector control electronics. However, the inputs will be a composite of steering commands for trajectory corrections and attitude changes required to compensate for CG travel. The input commands from the GAN system will cause the SCS gimbal to be displaced, creating the necessary lever arms and moments for changes in the S/C attitude. The steering commands permit the GAN system to perform curved trajectories.

The GAN input is activated through relay K1 when the GAN Delta V mode is selected. Therefore, after the thrust on commanded by the SPS thrust on/off electronics, relay K9 will close, inserting rate and attitude signals into the TVE electronics. The attitude error limiter prevents the GAN or SCS command inputs from exceeding ±8° in the pitch and yaw axes. The integrator is not used with the GAN Delta V mode selected.

Commanded by the SPS thrust on/off logic which energizes relay K6 if the SCS Delta V mode is selected. Relay K6 converts the amplifier to an integrator at engine ignition to prevent undesirable integrated outputs upon insertion into the thrust vector control electronics. When the SPS thrust on command closes relay K16, the integrator output is the first integral of the signal input. All amplifiers in the thrust vector control subsystem invert the polarity of the DC signal level.

5.4.3.5 MANUAL THRUST VECTOR CONTROL INPUTS

Selecting the MTVC mode removes all normal GAN or SCS inputs to the servo amplifiers. Relays K10, K11, K13, K14, and K5 will be energized. Relay K10 removes the normal inputs from the GAN and SCS, relay K11 inserts the MTVC commands into the gimbal position electronics. Relay K13 removes the servo drive amplifiers' output from the magnetic clutches in the primary gimbal positioning loop. Relay K14 connects to the output of the redundant servo drive amplifier to the redundant magnetic clutches. Relay K5 inserts the rate caged BMAG output into the MTVC electronics to provide the required rate damping for MTVC operation.

The MTVC mode will be used primarily to attenuate rates created by malfunctions in the SCS or GAN Delta V modes. The thrusting maneuver would be terminated after the rates are reduced and the RCS would completely stabilize the S/C with the SPS engine off. It would require excessive RCS propellants to stabilize the S/C with excessive angular velocities created by the SPS engine.

5.4.4 SCS AND GAN DELTA V MODES

In the GAN Delta V mode of operation, the astronaut will use the gimbal position trim controls to position the SPS gimbal engine such that the thrust vector will be through the CG in the thrusting direction. At SPS engine ignition the rate gyro and attitude error signal will become active inputs to control the SPS thrust vector. The attitude error inputs are processed through the error limiter which controls the magnitude of attitude error input. The error limited restricts the command attitude error input to approximately ±8°. In the GAN Delta V mode, the total spacecraft attitude change can be greater but the outer steering will always send attitude errors of ±8° or less to accomplish the attitude change. For the instantaneous condition, the attitude error will reflect the attitude change required because of CG travel while thrusting.

The SCS Delta V mode will use the gimbal position trim controls to perform the initial alignment, as in the GAN Delta V mode. The yaw BMAG and thrust vector control integrator will approximate instantaneous fixed angle thrust vector control. Assume zero initial error and then an attitude error is detected by the BMAG as a result of CG travel. The attitude error signal input will initiate corrective motion of the SPS engine gimbal. The position feedback

The GAN Delta V mode does not require an integrator similar to the SCS because the three accelerometers on the inertial measurement unit in the GAN system provide data to the Apollo Guidance Computer so that solutions to steering equations will incorporate the required corrections. There should be no residual lateral accelerations or velocities in the GAN Delta V mode. However, the accelerometer sensitivity will be a contributing factor.

5.5 SPS THRUST CONTROL LOGIC

The SPS thrust control logic, figure 5-8, provides the necessary on-off control for the SPS engine. Its primary function is to activate the two sets of fuel and oxidizer solenoid valves and indicate visually that thrust is in progress. The logic operation is dependent upon external inputs and switches to generate the thrust on/off commands and display signals.

5.5.1 Logic Inputs

The inputs are derived from the Delta V display, translation controls, display section of the Display/AGPS ECA, Apollo Guidance Computer, and the SCS control panel. Each unit has one or more discrete inputs to the SPS thrust control logic.

5.5.1.1 Delta V Display

The Delta V display provides three switching inputs. The switches are thrust control switch, thrust on switch, and direct voltage switch. The thrust

SPPS THRUST CONTROL LOGIC

Figure 5-8
Control switch has three positions. From top to bottom they are: NORMAL, OFF, and DIRECT ON. The NORMAL position provides the normal arming of the SPS engine solenoid valves to supply a Gn automatic or direct on command to activate the solenoids. The automatic off commands will be generated from the Apollo Guidance Computer or the SCS when the desired amount of velocity is obtained. The OFF position will remove electrical power from the SPS engine solenoid valves and provide a backup direct thrust off command to terminate or prevent initiating of thrusting with the SPS engine at any time during the mission. The DIRECT ON position of the Thrust Control Switch permits energizing the SPS engine solenoid valves in the event of a malfunction in the SPS thrust control logic. This position of the thrust control switch requires manual use of the direct thrust off to terminate the thrusting maneuver with the SPS engine. It will be used in event of malfunctions only.

The Thrust Control Switch provides power from the DC buses through the direct circuit breakers. SPS solenoid set 1 will be connected to Mn bus A and set 2 will be connected to Mn bus B when the Thrust Control Switch is in the NORMAL or DIRECT position. The normal position switches high side power only and the direct position switches high and low sides of the power as shown in Figure 5-8.

The THRUST ON switch is a momentary contact switch for manually initiating SPS engine ignition in the SCS Delta V mode. The switch has a talk back light that will be illuminated at all times when the Gn computer or the THUST ON switch is used to energize the SPS engine solenoid valves. When the THUST ON switch is used to initiate SPS engine thrust, the automatic off command will be derived from within the SCS.

The direct upage switch is a backup method of performing an upage prior to ignition of the SPS engine. It inhibits pitch and yaw attitude hold; therefore, the initial thrusting attitude is degraded. It would be used only if malfunctions prevented use of the +x translation capability of the translation controls. The direct upage switch will also provide the logic input which enables the Gn system to automatically ignite the SPS engine. The input is received at OR gate 3.

5.5.1.2 Translation Controls

The translation controls are the primary method of performing the required manual upage prior to ignition of the SPS engine. They provide the upage at a varying level because of interruptions that may occur for maintaining pitch and yaw attitude hold. This action reduces initial error input to the thrust vector control electronics upon ignition of the SPS engine. The logic input is provided into OR gate 3 to permit automatic ignition of the SPS engine.

5.5.1.3 Display/GAP ECA

The input from the DECA is generated in the display section of the Display/GAP ECA. The generation of this signal is a direct function of the Delta V remaining display. If the display is reading a value greater than zero, a logic one will provide an enabling logic signal to AND gates 1 and 3 in the SPS thrust control logic. When the Delta V remaining display reads zero or less, a logic zero will provide a logic inhibit into AND gates 1 and 3. Therefore, to generate an SPS engine ignition from the SCS, the logic one must be received from the DECA and absence of this logic one will inhibit or terminate thrusting of the SPS engine.

5.5.1.4 Gn System

The Gn system provides a thrust on-off command from the Apollo Guidance Computer. The thrust on command is represented by the presence of a 102.4 KC signal. The absence of the 102.4 KC signal represents a thrust off command. The SPS thrust control logic has a pulse train converter to convert the 102.4 KC signal into a logic switching signal. The output of the pulse train converter will be inhibited at inhibit AND gate 2 if the thrust on switch, on the Delta V display, is used to generate the SPS engine ignition signal. The logic input from the pulse train converter will be passed through inhibit AND gate 2 if SCS has not commanded a thrust on and enable AND gates 2 and 4.

5.5.1.5 SCS Control Panel

The mode select inputs from the SCS control panel are provided to the thrust control logic through the flight combustion stability monitor (FCSM). After the logic signals are passed through the FCSM, they provide enabling signals to the thrust control logic. The Gn Delta V logic signal provides an input to OR gate 1 and AND gates 2 and 4. The SCS Delta V logic signal is provided to OR gate 1 only. The output from OR gate 1 provides inputs to NAND gates 1 through 4, and AND gate 1. To perform a normal thrusting maneuver with the SPS engine, one of the Delta V modes must be selected to enable inputs to the thrust control logic.

5.5.1.6 Flight Combustion Stability Monitor

The FCSM provides an automatic logic thrust off command for rough computation of the SPS engine. Rough combustion would be a function of SPS operation, but the interface with SCS is required, for proper thrust off to insure thrust will not be commanded on again without complete shutdown. The FCSM latching relay opens the mode select logic input so that all logic holding circuits will drop out. A full thrust on logic sequence must be followed before the SPS engine is turned on again. To reset the FCSM relays, the power switch on panel 2 must be placed to the reset position which energizes the unlatch coils and enables the logic path.
5.5.2 SPS Thrust Control Logic Operation

5.5.2.1 GAN Delta V Mode

The GAN Delta V mode of operation will require AND gate 2 to be completely enabled. Its required inputs will be ullage maneuver from OR gate 3 and the holding input after AND gate 2 is enabled, which is all processed through NAND gates 1 and 2. GAN Delta V mode selected from the SCS control panel, and the pulse train converter output of logic one to inhibit AND gate 2. If the SCS thrust ON command has not been commanded, the gate will not be inhibited and a logic one output is generated for AND gate 2. The three inputs will enable AND gate 2 and provide the logic one input to inverter 3 which enables AND gate 6. Inverter 4 will provide a logic one signal to the solenoid drivers at the two sets of SPS fuel and oxidizer solenoids. The output of the pulse train converter will be removed and disable AND gate 2 to terminate a GAN Delta V maneuver.

5.5.2.2 SCS Delta V Mode

The SCS Delta V mode requires enabling AND gate 1 to initiate thrusting with the SPS engine. AND gate 1 has three enabling inputs: the input from OR gate 1 indicating a Delta V mode selected, the output from the DECA indicating a value greater than zero setting on the DELTA V REMAINING display, and the input from the THRUST ON switch desiring thrust to be initiated after it is processed through NAND gates 3 and 4, insuring a Delta V mode selected. The output from AND gate 1 is inverted by inverter 2 and disables AND gate 6, which is inverted by inverter 4 to enable the solenoid drivers. Termination of the SCS Delta V maneuver will be a logic zero from the DECA indicating zero velocity to be gained on the Delta V REMAINING display which disables AND gate 1. The output from AND gate 1 operates into inverter 2, which enables AND gate 6, and the signal is inverted by inverter 4 to de-energize the solenoid drivers and terminate thrusting.

5.5.2.3 Hybrid Mode

The Hybrid mode is a special case where the output of OR gate 1 enables the Delta V mode input to AND gate 1, and with the requirement of setting the DELTA V REMAINING display for monitoring all GAN Delta V maneuvers, the logic output from the DECA will provide an enable input for AND gate 1. If the automatic thrust ON command does not arrive from the Apollo Guidance Computer, the thrust ON switch can be depressed and initiate the engine ignition command. Therefore, the GAN will control the trajectory and the SCS total.

5.5.2.4 Direct Mode

The direct operating mode bypasses all electronics and supplies 28 VDC and ground circuits direct from the thrust control switch. The direct operating mode also required manual crew action to initiate and terminate thrust from the SPS engines.

5.5.2.5 Backup Thrust Off

If the automatic thrust OFF command fails for the GAN, SCS, or Hybrid modes, the thrust control switch will be used to terminate the thrusting maneuver. The backup thrust OFF operation will terminate a thrusting maneuver at all times. It removes the 28 VDC from the solenoids regardless of the SPS thrust control logic conditions.

5.5.2.6 Thrust in Progress Indication

In addition to the operating conditions discussed, the SPS thrust control logic will provide illumination of the talkback light in the thrust ON switch. To illuminate the light in a GAN Delta V mode, and AND gate 1 must be enabled, it has two inputs which are GAN Delta V mode selected and the presence of a logic one output from inhibit AND gate 2, with AND gate 4 enabled, OR gate 2 will trigger the lamp driver and illuminate the thrust ON light. To illuminate the light in the SCS Delta V mode, AND gate 3 must be enabled. The two enabling signals to AND gate 3 are from the THRUST ON switch and DECA. When AND gate 3 is enabled, it will also satisfy OR gate 2 and trigger the lamp driver. The SPS thrust control logic is interlocked so that AND gates 3 and 4 will never be enabled simultaneously because of the THRUST ON switch inhibited to the pulse train converter. Therefore, the THRUST ON light will be illuminated at all times when the thrust is on for normal operation. If the direct function is used to initiate thrusting, it will BY-PASS all electronics and not illuminate the thrust ON light.

5.5.3 SPS Thrust Control Logic Outputs

The output from the SPS thrust control logic performs a function other than energize the SPS solenoid valves and relays K1 and K2. It provides the logic switches required to reconfigure the SCS relays for proper thrust vector control. The thrust control switch provides both inputs to OR gate 5, if both DC buses are operational. Either DC bus will enable OR gate 5 to provide the upper input to inhibit AND gate 1. AND gate 5 provides the controlling input to the inhibit AND gate 1, when the solenoid drivers are not energized and the thrust control switch is in normal, both inputs to AND gate 5 are true. The true input will maintain a logic false output from the
INHIBIT AND GATE 1 BECAUSE OF THE INVERSION ON THE INPUT OF THE GATE. WHEN EITHER SPS SOLENOID SET IS ACTIVATED BY A SOLENOID DRIVER OR BY THE GROUND CONTACTS OF THE DIRECT ON SWITCH, AND GATE 5 WILL BE DISABLED BY THE GROUND OR FALSE INPUT. THE OUTPUT WILL BE INVERTED BY THE INHIBIT AND GATE 1 INPUT TO ENABLE AN OUTPUT TO THE TIME DELAY. THE TIME DELAY IS REQUIRED TO PERMIT THRUST BUILDUP FROM THE SPS ENGINE BEFORE THE VEHICLE MANEUVER IS TERMINATED; HOWEVER, THE PITCH AND YAW ATTITUDE ERROR INPUTS ARE INSERTED INTO THE TVC ELECTRONICS IMMEDIATELY. THIS ACTION ASSURES POSITIVE G FORCES THROUGHOUT THRUST INITIATION OF THE SPS ENGINE AND GIMBALING CONTROL OF THE SPS ENGINE FOR THE ENTIRE THRUST BUILD UP. THE TIME DELAY IS ALSO ACTIVE AFTER INHIBIT AND GATE 1 IS DISABLED BY ENABLING AND GATE 5, AND GATE 5 IS ENABLED WHEN BOTH SETS OF SPS SOLENOID VALVES ARE DE-ENERGIZED. THE TIME DELAY IS ACTIVE FOR BOTH OUTPUTS WHEN THE SIGNAL IS REMOVED TO PERMIT CONTINUED SPS ENGINE GIMBAL CONTROL WHILE THE RESIDUAL THRUST IS PRESENT AND INHIBIT RCS OPERATION IN PITCH AND YAW UNTIL RESIDUAL THRUST IS REDUCED. THIS REDUCES THE TUMBLING INDUCED AT THRUST TERMINATION. THE TIME DELAY IS APPROXIMATELY ONE SECOND FOR APPLICATION AND REMOVAL OF THE ENGINE IGNITION SIGNAL.

RELAYS K1 AND K2 ARE TO AUTOMATICALLY ENABLE FUNCTIONS IN THE SPS COMMENSURATE WITH A THRUST ON COMMAND. THE POWER INPUTS TO THE FUEL QUANTITY GAGING SYSTEM IS THE CONTROLLING INPUT WITH NO MANUAL OVERRIDE FOR POWER INPUTS. THE HELIUM PRESSURIZATION IS INTERFACED WITH ENABLING SWITCHES IN THE SPS AND REQUIRE THEY BE POSITION CORRECTLY TO COMPLETE AUTOMATIC PRESSURIZATION. THERE ARE MANUAL OVERRIDE SWITCHES TO PERFORM HELIUM PRESSURIZATION IF REQUIRED FOR A SPS THRUST ON AS REQUIRED FOR OTHER SPS OPERATIONS.

5.6 DELTA V DISPLAY

THE DELTA V DISPLAY PROVIDES VISUAL INDICATIONS AND CONTROL DIRECTLY ASSOCIATED WITH THRUST VECTOR CONTROL. FIGURE 5-9 IS DIRECTLY ASSOCIATED WITH MEASURING THE ACTUAL VELOCITY imparted TO THE SPACECRAFT THAT WILL MODIFY THE S/C TRAJECTORY.

5.6.1 OUTPUTS

THERE ARE THREE OUTPUTS ASSOCIATED WITH CONTROLLING THE DELTA V. THEY ARE: (1) SPS THRUST LOGIC ON-OFF COMMAND, (2) DELTA V REMAINING DISPLAY READOUT, AND (3) DELTA V TELEMETRY OUTPUT.

5.6.1.1 THRUST OFF COMMAND

THE SPS THRUST CONTROL LOGIC ON-OFF COMMAND TERMINATES THE THRUSTING FROM THE SPS ENGINE WHEN THE VELOCITY TO BE GAINED IS ZERO. THIS FUNCTION WILL HAVE A VALID INPUT TO THE SPS THRUST CONTROL LOGIC IN THE AUXILIARY ECA FOR THE S/C DELTA V MODE WHEN THE THRUST CONTROL SWITCH, ON THE DELTA V DISPLAY, IS IN THE NORMAL POSITION.
5.6.1.2 Delta V Remaining Display

The Delta V Remaining display provides a visual indication to the crew of the instantaneous Delta V Remaining. It also provides a comparison for the thrust termination point at zero so the crew can provide a manual backup to the GAN and SCS automatic off commands previously discussed in the SPS thrust control logic discussion. The Delta V Remaining display also measures the tail-off velocity imparted by the SPS engine so that it can be inserted where required in computations.

5.6.1.3 Telemetry

The Delta V telemetry output provides ground based monitors with a gross indication of the velocity imparted to the spacecraft for the Delta V maneuver. It has no crew visibility in the spacecraft.

5.6.2 Delta V Display Operation

To operate or perform initial set of the Delta V remaining display, the TVC 1 POWER switch must be placed on. The application of power will set FF #1 and inhibit operation of the counter in the Delta V remaining display when the Boolean notation + Slew is true. The Boolean notation will be false when the Delta V SET switch is displaced from neutral. This permits setting the Delta V counter to any desired reading within its display capability. The + Delta V set voltage will pass through the Boolean notation + slew when it becomes true by displacing the Delta V SET switch from neutral. The function of inhibiting the drive for the Delta V counter prevents any stray signals from changing the Delta V counter prior to starting the thrusting with the ullage maneuver. The ullage maneuvers initiated by the translation controls or direct ullage button will pass through the OR gate and reset FF #1 and enable the Delta V counter drive for monitoring the changes in velocity. To inhibit the Delta V counter again, the power switch must be turned off, then on to set FF #1.

The accelerometer output to drive the Delta V counter will be a voltage level proportional to the instantaneous acceleration detected by the accelerometer. The voltage level input will be amplified and passed to the level detectors to determine forward or reverse rotation for the Delta V counter motor. The signal will then be gated through the AND gates with the 4 KC reference frequency for synchronization. The pulse shaper will have a free running 4 KC output if the clock input fails, but it will not have the accuracy of the clock reference frequency. The output from AND gates 1 and 2 will trigger the correct one shot and generate a stepping signal to the Delta V counter motor logic and provide the necessary feedback to the amplifier. Flip flops 2 and 3 will be used to step the Delta V counter motor as directed by the motor logic. As the Delta V counter counts to zero while thrusting, the coincidence detector switch will close, initiating the thrust-off sequence for the SCS Delta V mode.
6.1 INTRODUCTION

The Electrical Power System supplies the SCS with 28 VDC from two separate buses: Main Bus A and Main Bus B. In addition, it supplies 115 VOLT 3 PHASE, 400 CPS power from a Y-connected, 4 wire source which also uses two separate buses: Bus 1 and Bus 2. The two buses for AC and DC power distribution provide redundancy for critical SCS functions in case of partial power failure, shorts, etc. In the following sections, the two DC buses will be referred to simply as Mn A and Mn B. The two AC buses will be called AC 1 and AC 2.

Considering Figure 6-1, it can be seen that this power is supplied to the SCS through 18 circuit breakers located on the Left Hand Circuit Breaker Panel. From these circuit breakers, the power is distributed to the SCS components through six power switches on the Sequence Controller and SCS Power Panel. These six switches are subdivided into two groups of three switches each, called Group 1 and Group 2. The Group 2 power switches supply power to those SCS components which provide the backup (manual TVC) capability for controlling the spacecraft during the critical periods when the SPS engine is firing. The Group 1 power switches supply power to the rest of the SCS.

All six power switches act independently of each other and individually select either DC power from Mn A and AC power from AC 1, DC power from Mn B and AC power from AC 2, or can simply be off. That is, a single power switch selects either:

A. Mn A and AC 1
B. Mn B and AC 2
C. Off

Some of the 28 VDC power is distributed to the SCS directly from the circuit breakers without being routed through any of the six SCS power switches. The power distribution will be subdivided into the following three sections:

A. 28 VDC power which does not go through any of the power switches on the Sequence Controller and SCS Power Panel.
B. 28 VDC and 115 VAC power which goes through the three Group 1 SCS power switches on the above mentioned panel.
C. 28 VDC and 115 VAC power going through the three Group 2 power switches.

Figure 6-1

SCS POWER DISTRIBUTION

FUNCTIONAL DIAGRAM
A more detailed discussion will now be given of the power distribution to the SCS, in which each of the above three categories will be treated separately.

6.2 28 VDC Non-Switched Power

Referring to Figure 6-2, it can be seen that the 28 VDC non-switched power is distributed to the SCS through 14 of the 18 circuit breakers on the left hand circuit breaker panel. These 14 circuit breakers are labeled: Yaw, Pitch, A&C Roll, B&D Roll, Group 1, Direct Cont., and Group 2. This power is called non-switched power since its availability to the SCS is independent of the state of the six SCS power switches. The power distribution from these circuit breakers will now be considered according to their functions.

6.2.1 Reaction Jet Circuit Breakers

The Yaw, Pitch, A&C Roll, and B&D Roll circuit breakers apply 28 volts to the high side of the automatic coils for the indicated jets in the reaction control system. The jets can be activated by the SCS jet drivers, which supply a ground when the jet should be on. The power from the circuit breakers goes through the channel enable switches on the control panel, which are labeled Yaw, Pitch, A&C Roll, and B&D Roll. Any of the four channels can be enabled separately by means of these switches.

6.2.2 Group 1 Circuit Breakers

The two group 1 DC circuit breakers supply 28 volts to the reaction jet drivers, translation controllers, and control panel. They also supply DC power through two of the SCS power switches.

The power to the reaction jet drivers goes through a set of relays in the two master event sequence controllers (which are not part of the SCS), through the channel enable switches on the control panel, through a set of relays in the SCS, and finally to the jet drivers. This power does not turn on the drivers. It is merely an enable signal. Thus, it can be seen that the channel enable switches on the control panel disable a given channel both by removing the power from the high side of the coils, and also by disabling the jet drivers for that particular channel. Notice that in any given channel, half the jets get their power from bus A and half from bus B. This means that if power on one bus is lost, disabling half the jets, the rest of the jets on the remaining bus are sufficient to provide positive or negative rotation in all three axes.

The power, which is used for commanding normal translation maneuvers by means of the translation controls, is routed through the C/M-S/M separation switch, so that translation control is deactivated after C/M-S/M separation. Prior to this separation, however,

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**NON-SWITCHED 28VDC POWER DISTRIBUTION**

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**Figure 6-2**
The outputs of the two controllers are routed to the jet select logic to provide 3 axis translation capability. Again, the two power buses, each feeding one translation control, provide redundant power for translational maneuvers in case of a power failure on one bus.

Two isolation diodes feed the power from both group 1 DC buses to the indicated switches on the control panel. A more detailed drawing of the control panel is shown in Figure 2-2.

Finally, it can be seen that DC power from the group 1 circuit breakers is also routed to the SCS power switches. The distribution from these switches will be covered in the section dealing with the group 1 power switches.

6.2.3 Group 2 Circuit Breakers

In reference to Figure 6-2, it can be seen that the two group 2 DC circuit breakers supply power to the clockwise switches on the two translation controls, to the breakout switches on the rotation controls, and to the mag power switch. The distribution from the mag power switch will be covered in the section dealing with the group 2 power switches.

6.2.4 Direct Control Circuit Breakers

The direct control circuit breakers are used to provide two redundant sources of power for direct voltage, direct rotation, and the SPS solenoid valves. Either one of the two buses is sufficient to provide a 9V voltage maneuver, three-axis rotational control, and turn-on in SPS engine.

Manual direct rotation can be used anytime the direct control circuit breakers are in and the direct mode switch on. Direct voltage is available whenever the circuit breakers are in. As a backup capability, the SPS engine could be turned on simply by putting the thrust control switch in the direct on position. Thus, it can be seen that the power from direct control circuit breakers provides a backup means for the critical maneuvers of voltage, rotation, and SPS engine thrust on. These backup capabilities are independent of the SCS power switches and would be available even if the rest of the SCS were to lose power.

6.3 Group 1 Power

There are three SCS group 1 power switches which supply 28 VDC and 115 VAC power to the SCS. These three switches are shown in Figure 6-3 and are labeled:

A. Partial SCS Power
B. SCS Power
C. Rate Gyro Power

Each switch selects either the combinations of AC 1 and Nn A or AC 2 and Nn B, or off. The power distribution from these switches will be discussed one at a time.

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**GROUP 1 POWER DISTRIBUTION**

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**FIGURE 6-3**
6.3.1 Partial SCS Power

The PARTIAL SCS POWER SWITCH distributes power to seven units in the SCS: the roll, pitch, yaw, auxiliary, and display/DAGAP ECA’s, as well as the Delta V indicator and attitude set/gimbal position display. Each of the five ECA’s contains rectifiers and voltage regulators for generating DC voltages from the 115 VAC supply. Unless it is indicated otherwise on Figure 6-3, the various voltages are generated using all three phases of the 3-phase source. Furthermore, the symbol (straight line with a circle) opposite most of the voltage supplies indicates a test point one of the three test connectors on the face of each ECA. These test points can be used to check the indicated voltage level. The indicated voltages are DC unless explicitly labeled AC.

The DC voltages generated in the pitch and yaw ECA’s are used to power the control electronics within those units. The roll ECA generates power for its control electronics as well as the power for the signal conditioning circuits which are used in telemetering SCS information to the ground stations.

The power supply in the auxiliary ECA is used for the electronics in the ASU and in the SPS primary electronics. The FD1 and gimbal position display electronics receive their power from the DAGAP ECA.

6.3.2 TVC No. 1 Power

The TVC No. 1 POWER SWITCH feeds 115 VAC to the pitch and yaw TVC power supplies as well as 28 VDC and 115 VAC to a power supply in the DAGAP ECA. The pitch TVC power supply is used for operating the pitch SPS gimbal drive mechanism, which consists of a servo amplifiers, gimbal position and rate transducers, demodulators, and gimbal trim potentiometers. Similarly, the yaw power supply feeds power to the yaw channel.

The power supply in the DAGAP ECA generates the voltages necessary for operating the accelerometer and Delta V indicator electronics.

6.3.3 Rate Gyro Power

The rate gyro power switch routes AC power only to the rate gyro package as well as to the yaw, pitch, roll, and DAGAP EA’s.

All three phases of the 115 VAC are used in the rate gyro package for operating the spin motor and signal generator. Phase A only is used in the four ECA’s for the demodulator references.

6.4 Group 2 Power

The Group 2 switches supply power to those components of the SCS which are necessary for manual thrust vector control. The three Group 2 switches are shown in Figure 6-4 and are labeled:

A. DAGAP Power  
B. TVC No. 2 Power  
C. Rotation Control Power

6.4.1 DAGAP Power

This switch supplies 115 VAC and 28 VDC to the roll, pitch, yaw, and DAGAP ECA’s. The voltages generated in the DAGAP ECA are used for operating the BAG’s together with the torque amplifiers and pre-amplifiers in the AGAP. 115 VAC, phase A power is also supplied to the yaw, pitch, and roll ECA’s for demodulator references.

6.4.2 TVC No. 2 Power

This switch supplies only 115 VAC power to the redundant set of TVC power supplies in the pitch and yaw ECA’s. There are two sets of these power supplies for redundancy. One set receives power from the TVC 1 POWER switch, and has been discussed previously in the Group 1 power section. The second set receives its power from the TVC 2 POWER switch. This second set of TVC power supplies in the pitch and yaw ECA’s is used for operating a second gimbal drive mechanism which consists of servo amplifiers, summing amplifiers, gimbal position and rate transducers, demodulators, and gimbal trim potentiometers. If an inproper gimbal drive clutch current is sensed in the TVC 1 mechanism during an SPS engine firing, there is an automatic switchover to the TVC 2 mechanism. This switchover is accomplished in both channels if a manual thrust vector control signal is given by twisting the translation control handle clockwise.

6.4.3 Rotation Control Power

From this switch, 115 VAC (phase A only) power is distributed to the roll, pitch, and yaw ECA’s. 26 VAC is generated in the pitch and yaw ECA’s for exciting the rotation control transducers. In addition, 115 VAC is supplied for the demodulator references.

6.5 Summary

The SCS receives its power through 18 circuit breakers. Some of the power is distributed to the SCS through six power switches, and some is distributed to the SCS directly. Various capabilities of the SCS, consequently, are dependent upon the state of the six SCS power switches. Manual direct rotation, direct up, and SPS engine on can be accomplished with all six power switches. The two inertial sensor packages (Rate Gyros and BAG’s) receive their power from the rate gyro power switch and the DAGAP power switch. The rotation controls receive power, for manual proportional rotation, from a rotation control power switch. The two redundant systems for controlling the SPS engine gimbal’s receive their power from the TVC 1 POWER switch and the TVC 2 POWER switch. One might say that the PARTIAL SCS POWER SWITCH distributes power to
GROUP 2 POWER DISTRIBUTION

THOSE PARTS OF THE SCS WHICH NEITHER RECEIVE POWER DIRECTLY NOR RECEIVE POWER FROM THE OTHER FIVE POWER SWITCHES.

KEEPING IN MIND WHICH COMPONENTS OF THE SCS RECEIVE POWER FROM WHICH SWITCHES, TABLE 6-1 CAN BE SEEN TO BE A SYNDOPSIS OF SYSTEM CAPABILITIES IN SO FAR AS THEY ARE DEPENDENT UPON THE STATE OF THE SIX SCS POWER SWITCHES.
<table>
<thead>
<tr>
<th>SCS POWER VS SYSTEM CAPABILITY</th>
<th>PARTIAL SCS POWER</th>
<th>RATE GYRO POWER</th>
<th>BMAG POWER</th>
<th>R0T CON POWER</th>
<th>TVC NO. 1</th>
<th>TVC NO. 2</th>
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<tbody>
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<td>BACKUP RATE</td>
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⚠️ IF BACKUP RATE HAS BEEN SELECTED IN ANY AXIS. IF BUR HAS BEEN SELECTED IN ALL 3 AXES, RATE GYRO POWER NEED NOT BE ON.
⚠️ PARTIAL SCS POWER MUST BE ON IF ΔV SWITCH IS IN "NORMAL" POSITION. HOWEVER, ENG CAN BE TURNED ON EVEN IF ALL POWER SWITCHES ARE OFF, BY PLACING ΔV SWITCH IN DIRECT "ON" POSITION.
⚠️ TVC NO. 2, RATHER THAN TVC NO. 1, MUST BE ON IF SYSTEM IS IN MAN. TVC. OR SERVO DRIVE NO. 1 HAS BEEN DISABLED.
⚠️ SCS ΔV MODE ONLY. HOWEVER, IN GAIN ΔV MODE, BMAG POWER MUST BE ON IF BACKUP RATE HAS BEEN SELECTED IN ANY AXIS. IN GAIN ΔV MODE, IF BUR HAS BEEN SELECTED IN ALL 3 AXES, RATE GYRO POWER NEED NOT BE ON.
⚠️ SCS MODES ONLY. ALSO GAIN MODES IF BACKUP RATE HAS BEEN SELECTED IN ANY CHANNEL.
⚠️ SCS MODES ONLY, PROVIDED THAT BACKUP RATE HAS NOT BEEN SELECTED.
⚠️ NECESSARY FOR ENTRY. NOT ABSOLUTELY NECESSARY FOR NON ENTRY.

TABLE 6-1

6-11
7.1 INTRODUCTION

This section contains a description of each control mode used in performing the operations required of the crew and of the spacecraft systems. Each mode is discussed to a functional block diagram level as a summary of the SCS and as an integration of the material contained in the previous sections. This summation allows the instructor to delve into all three major subsystems of the SCS as required to explain control mode utilization.

7.2 MONITOR MODE

The Monitor Mode provides display information for the astronaut to monitor boost vehicle stability in the ascent phase of the Apollo mission. Other information will be displayed in gross values. This mode also provides rate damped control when and if required throughout the mission. Figure 7-1 is used to depict the Monitor mode of operation.

7.2.1 Normal Ascent

For discussion purposes the Monitor mode can be broken into four sections: Attitude Control, Attitude Reference, Thrust Vector Control, Delta V Accelerometer, and Display Section. During a normal ascent, all sections of the Monitor mode will be active. The attitude control section will be performing no attitude control function with the boost vehicle attached to the S/C. The continuity path would be broken at K12, preventing command signals from reaching the Reaction Control System.

The backup attitude reference would not be providing attitude to the FDI, but it would be operational and following vehicle attitude changes. Relay K13 and K14 would be closed to complete the continuity path required for driving the AGC and torquing the thrusters.

The display of total attitude and attitude error are provided from the AGM. The total attitude will indicate the changes in launch vehicle attitude as it changes from launch pad alignment. It will be supplied by the IMU through relay K9 for display. The attitude error displayed will represent deviation from the desired trajectory for the first stage booster, but they provide no useful data after the SIVB is ignited for final boost into orbit. A different guidance technique is used in the SIVB guidance package than in the AGM; therefore, rather large errors could exist between the two systems with each.
Having the correct guidance calculations for the technique used, the SINS is primary; therefore, the GNN attitude error supplied through relay K1 will be of little use in a normal ascent. The display scale values are given in Table 2-2 of Section 11.

The attitude rates are provided to the FDI through relay K6 from the SCS rate gyros. The rates displayed represents C/M angular rates. It is not the true rate of the launch vehicle because of body bending and other contributing factors. The display scale values are given in Table 2-3 of Section 11.

The X axis accelerometer has no mechanical casing capability but it will be energized during the ascent phase to provide an abort delta V capability. However, it will not have an actual display unless an SPS abort is initiated. The accelerometer has a normal rate of 0.01 g in the X axis and will withstand over twice the g forces experienced in ascent.

The thrust vector control electronics will be activated to hold the SPS engine stationary throughout ascent to prevent damage by it swinging into the high gain antenna or other objects surrounding the SPS engine and to provide TVC for SPS aborts. TVC 1 power switch would be on one set of buses (AC & DC), and TVC 2 power switch would be on the redundant buses to preclude a power failure from letting the SPS engine swing free.

7.2.2 Aborts

There are several types of aborts and events that occur depending upon where in the ascent phase the abort is initiated. The SCS will have no active control inputs if an abort is initiated within 42 seconds or less after lift off. For aborts where the time lapse is greater than 42 seconds after lift off, the SCS will be required to produce control signals.

7.2.2.1 LES Abort

When a LES abort is initiated requiring SCS operation, the SCS will be enabled 1 second after the launch escape motor is ignited. Relay K23 will be closed and K22 will select the C/M position. Relay K14 will normally be closed in the monitor mode, and the rate gyros will provide the rate input signals to the SCS electronics to automatically rate damp the SCS with K23 closed and K22 in the C/M position. The exact use of the SCS other than rate damping is dependent upon altitude, velocity, if canards are retained, and etc. In any event, the SCS would be disabled at 24,000 feet by relay K23 in the descent for landing.

7.2.2.2 SPS Abort

All aborts after the launch escape tower is jettisoned will be SPS aborts. The aborts will be initiated by the CCN switch in the translation control.

The switches closure will cause the NESC to command a +K translation with the direct coils of the S/M RCS. Relay K26 and K28 will be connected to C/M or S/M positions at all times. They will be in the S/M position until C/H-S/M separation is commanded or the C/M-S/M transfer switch is activated. The direct +K translation provides the tension necessary to separate the command service module combination from the boost vehicle. Various lengths of direct +K translation can be obtained from the separation sequence depending upon the operation of the CCN switch in the translation control. Maximum direct +K translation will be 12 seconds.

A Delta V node would be selected and initiated before 12 seconds has elapsed to perform a Delta V with the SPS engine. It would be performed with no re-alignment of the SPS engine thrust vector from the normal hold position for ascent. The abort could be on injection into earth orbit or selection of a down range landing site. The primary objective will be to get a safe separation distance between the C/M-S/M and the boost vehicle.

7.3 GNN Attitude Control Mode

The GNN Attitude Control Mode will provide an inertially stabilized attitude. It will utilize attitude information from the IMU. A block diagram of this mode is shown in figure 7-2.

The GNN Attitude Control Mode of operation can be used to control S/C attitude in earth orbit operation. Since the GNN system must be completely operational, it represents a power penalty to use this mode extensively. The GNN Attitude Control Mode basically consists of the Attitude Reference System and the Attitude Control System.

The Attitude Reference System consists primarily of the GNN system and its inputs to the FDI. The SCS portion of the Attitude Reference System will be in a follow condition by closing relays K11 and K12 to insure that when the SCS Attitude Control Mode is selected, the AGCU will have the same information as the IMU if the AGCU is initially aligned to agree with the GNN alignment. This will prevent changes to the FDI ball indications at the time of mode changeover. The attitude control electronics section for the GNN Attitude Control Mode will provide stabilization of the spacecraft to enable holding a specific attitude with respect to the inertial reference frame. Attitude errors from the GNN system, through relays K1 and K2, will be used by the SCS electronics to provide attitude corrections. A deadband set up in the SCS control electronics will prevent attitude correction signals from being sent to the RCS unless the errors exceed the selected deadband. Manual selection of the deadband will be based on the functional requirements using the two-position switch (MAX or MIN) located on the SCS control panel. For example, during navigational sightings minimum deadband would be required. The attitude will be maintained within the selected deadband by using the attitude error signal as a command maneuver input to the SCS. The FDI total attitude indicator.
will be driven by the IMU. The attitude error indicators are driven by the CDU outputs for each channel to display the difference between the IMU gimbal position and the guidance computer commanded position. The RSP within the SCS will drive the attitude rate indicators with a scale range of ± 1°/sec.

7.3.1 Attitude Maneuver

The astronaut can use the AOC keyboard to insert commands which will result in commanded attitude changes from the AOC. The SCS will follow the attitude error signal inputs from the CDU in response to AOC inputs for maneuvering to the commanded attitude. A manual maneuver capability exists with the GAN SYNC switch. This switch, when coupled with the rotation control breakout switch, will cause the CDU's to follow S/C attitude changes. The GAN SYNC switch also prevents the AOC from commanding a new position to the CDU when the rotation control is returned to neutral. Returning the GAN SYNC switch to the OFF position will return the CDU's to AOC control.

7.3.2 Translation Maneuvers

Translations can only be commanded manually through use of the translation control or the direct ullage switch. The normal attitude hold function will provide attitude control except for translations commanded with the direct ullage switch. Translation is provided by the S/M RCS and therefore cannot be accomplished after C/J-3/S/J separation.

7.3.3 Minimum Impulse Maneuvers

When performing navigational sightings it is necessary to perform maneuvers of minute rotation rates. The attitude impulse control has been provided for this purpose. By switches at the navigator's station, the command and stabilization signals can be disabled, effecting a free drift condition. The attitude impulse can provide rotation commands in one or more axes simultaneously to the SCS electronics. The command inputs will generate a minimum impulse from the selected pair of RCS engines for each switch closure. The output commands will energize the RCS engine solenoids for 18 ± 4 ms. A minimum impulse from the RCS will provide an angular body rate of about 3 arc min/second. The attitude impulse control can be used only in the GAN attitude control and SCS attitude control modes.

7.4 SCS Attitude Control & SCS Local Vertical Modes

The SCS attitude control and SCS local vertical modes (Figure 7-3) are similar in mechanism, but they perform different control functions. The SCS local vertical mode will maintain the S/C at approximately a constant attitude with respect to the earth's surface while in earth orbit. It is not a precise attitude due to the open loop control that is mechanized and other effects. The SCS attitude control
MODE WILL HOLD AN INERTIAL REFERENCE ATTITUDE IN EARTH ORBIT. BOTH OPERATING MODES INCORPORATE THE CAPABILITY OF PERFORMING MANUAl MANEUVERS USING THE ROTATION CONTROLS.

7.4.1 SCS ATTITUDE CONTROL MODE

IN THE SCS ATTITUDE CONTROL MODE THE FOLLOWING RELAYS WILL BE CLOSED: K3 APPLYING TOTAL ATTITUDE TO THE FDAI FOR TOTAL ATTITUDE DRIVE DURING MANEUVER, K8 FOR DISPLAY OF BODY RATES ON THE FDAI, K10 TO DISPLAY ATTITUDE ERRORS ON THE FDAI, K13 APPLYING ATTITUDE ERRORS TO THE SCS ELECTRONICS, K14 TO APPLY BODY RATES TO THE SCS ELECTRONICS, AND K23 FOR THRUST ON-OFF COMMANDS TO THE S/M RCS.

7.4.1.1 FDAI/AGCU ALIGN

AS IT BECOMES NECESSARY TO ALIGN THE AGCU AND FDAI IN THE APOLLO MISSION, THE CREW MEMBERS WILL DETERMINE THE S/C’S ATTITUDE FROM NAVIGATION SIGHTINGS AND GROUND BASE COMMUNICATIONS. THE THREE INERTIAL ANGLES OBTAINED ARE THEN DIALED INTO THE ATTITUDE SET DIALS AND THE FDAI ALIGN BUTTON, ON THE ATTITUDE SET/ GIMBAL POSITION PANEL, IS DEPRESSED. THIS CAUSES RELAY K6 TO CLOSE AND THE AGCU WILL DRIVE TO THE NEW ATTITUDE, THEREBY REPOSITIONING THE FDAI. ATTITUDE ERROR SIGNALS TO THE AGCU ELECTRONICS WILL NOT BE CHANGED NOR WILL THE BANG OUTPUT SIGNALS.

7.4.1.2 MANUAL ATTITUDE MANEUVERS


A TEMPORARY (SHORT TIME AND SMALL ANGLES) CHANGE IN ATTITUDE MAY BE ACCOMPLISHED SIMPLY BY MOVING THE ROTATION CONTROL. THE BANG’S WILL BE CAGED BY THE BREAKOUT SWITCH CLOSURE AND UNCASED WHEN THE

THE ATTITUDE ERROR OBTAINED FROM AGCU WITH THE ATT SET SWITCH IN THE ATT SET POSITION WILL NOT PERFORM ANY AUTOMATIC CONTROL FUNCTIONS. IT IS FOR DISPLAY ONLY. THE SCS CHANNEL ENABLE SWITCHES ON THE SCS CONTROL PANEL MAY BE USED TO DEACTIVATE ANY CHANNEL IF A FREE DRIFT IS DESIRED. DEACTIVATION OF THE CHANNELS BY THE SCS CHANNEL ENABLE SWITCHES WILL OPEN RELAY K23, INHIBITING THE RCS AND CLOSE RELAYS K11 AND K12, CAGING THE SWING IN ALL AXES. THE ANGLE THROUGH WHICH THE S/C DRIFT IS CALCULATED AND CONTINUOUSLY DISPLAYED ON THE FDI ATTITUDE BALL IF THE RATES ARE NOT TOO LOW. DEACTIVATION OF ONE CHANNEL REMOVES ATTITUDE HOLD IN THE OTHER TWO FOR THE SCS ATTITUDE CONTROL. THIS FREE DRIFT CAN BE USED IN THE SCS ATTITUDE CONTROL MODE IN A SIMILAR MANNER EXCEPT COMPLETE ATTITUDE HOLD IS NOT LOST WHEN ONE CHANNEL IS DEACTIVATED. WHEN FREE DRIFT IS DESIRED FOR EXTENDED PERIODS OF TIME, THE RCS CAN BE TURNED OFF FOR POWER CONSERVATION. RCS PROPELLANTS ARE ALSO CONSERVED IN A FREE DRIFT CONDITION.

7.4.2 SCS LOCAL VERTICAL MODE

THE SCS LOCAL VERTICAL MODE CLOSES ONE ADDITIONAL RELAY ON FIGURE 7-1. IT IS A RELAY K4 WHICH PROVIDES THE ORBIT RATE SOURCE. THE ORBIT RATE SOURCE IS A PRESET LEVEL AND WILL BE MOST ACCURATE FOR THE PREDICTED ORBIT PRIOR TO LAUNCH. FOR THE 1000 KM CIRCULAR ORBIT, THE TORQUING RATE WILL BE APPROXIMATELY 246°/HR.

THE LOCAL VERTICAL MODE CAN MAINTAIN THE APPROXIMATE ATTITUDE WITH RESPECT TO THE EARTH'S SURFACE ONLY WHEN THE AGCU IS ALIGNED TO REPRESENT THE S/C ATTITUDE WITH RESPECT TO THE EARTH SURFACE WHEN THE MODE IS SELECTED. AFTER PROPER ALIGNMENT, BEFORE ENTERING THE SCS LOCAL VERTICAL MODE, THE ROTATION CONTROL CAN BE USED TO MODIFY THE LOCAL VERTICAL ATTITUDE THEREBY PERMITTING MANUAL ATTITUDE MANEUVERS IN THIS MODE. ALL THREE CHANNELS OF ATTITUDE CONTROL MUST BE ENABLED BY THE SCS CHANNEL ENABLE SWITCHES BEFORE THIS MODE CAN BE USED.

7.5 GAN DELTA V MODE

THE GAN DELTA V MODE IS THE PRIME MODE OF OPERATION FOR VELOCITY CHANGES IMPARTED TO THE S/C BY THE SPS ENGINE. IF THE VELOCITY TO BE ACQUIRED IS LESS THAN THE MINIMUM IMPOSSIBLE OF THE SPS ENGINE, IT WOULD NOT BE USED. FOR EARTH ORBIT MISSIONS, ALL DELTA V'S SHOULD EXCEED THE MINIMUM IMPULSE OF THE SPS ENGINE. FIGURE 7-4 SHOWS A BLOCK DIAGRAM OF THE DELTA-V AS IT IS CONFIGURED FOR THE GAN DELTA V MODE.

7.5.1 RELAY CONFIGURATION

WHILE IN THE GAN DELTA V MODE AND PRIOR TO THRUST INITIATION, THE SCS WILL PERFORM AS IN A GAN ATTITUDE CONTROL MODE. THAT IS, RELAYS K1 AND K2 APPLYING ATTITUDE ERROR TO THE FDI AND SCS ELECTRONICS, K3 APPLYING TOTAL ATTITUDE TO THE FDI, K8 AND K14 BODY RATES TO THE FDI AND SCS ELECTRONICS, AND K23 THRUST-ON-OFF COMMANDS TO THE S/M RCS. AFTER SPS IGNITION, THE SCS WILL CONTINUE TO MAINTAIN THE ATTITUDE BY POSITIONING THE RCS RAYS THROUGH RELAY K24 IN PITCH AND YAW, AND ROLL ATTITUDE THROUGH RELAY K23 TO THE S/M RCS.

7.5.2 OPERATION

ATTITUDE INFORMATION FROM THE IMU IS DISPLAYED ON THE ATTITUDE BALL. ATTITUDE ERRORS FROM THE CDU'S ARE SHOWN ON THE ATTITUDE ERROR NEEDLES (± 5° SCALE) AND ACTED UPON BY THE SCS TO HOLD OR CHANGE ATTITUDE IN RESPONSE TO COMPUTER COMMANDS. RATES ARE DISPLAYED WITH A SCALE RANGE OF ± 5°/SECOND. IN THIS CONTROL MODE, THE DELTA V DISPLAY IS USED TO MONITOR THE AUTOMATIC VELOCITY CONTROL EXERCISE BY THE COMPUTER. THE DELTA V DISPLAY WILL BE SET BASED UPON THE INFORMATION OBTAINED FROM THE GAN SYSTEM. THE DELTA V DISPLAY WILL BE ACTIVATED BY TVC 1 POWER SWITCH. THE DELTA V SET SELLS SHOT WILL BE USED TO SET THE DELTA V MANEUVER WILL BE SET IN ON THE EVENT CLOCK. TO PERFORM THE DELTA V MANEUVER, THE SPACECRAFT WILL BE POSITIONED SO THAT THE SPS THRUST VECTORS WILL BE THROUGH THE CENTER OF GRAVITY IN THE DESIRED DIRECTION OF THE VELOCITY CHANGE. THE ATTITUDE MANEUVER CAN BE PERFORMED MANUALLY OR AUTOMATICALLY BY A PROCEDURE OPTION. THE DELTA V MODE IS SELECTED, THE MINIMUM DEADBAND WILL BE SELECTED IF IT HAS NOT BEEN PREVIOUSLY SELECTED. AT SOME DEFINITE TIME BEFORE THE DELTA V MANEUVER IS TO START, THE FOUR GIMBAL ACTUATOR MOTORS WILL BE TURNED ON SEQUENTIALLY FROM THE EMERGENCY DETECTION AND GIMBAL DRIVE CONTROL PANEL.

THE GIMBAL TRIM ANGLES REQUIRED FOR THE DELTA V MANEUVER WILL BE OBTAINED FROM THE BOARD CHARTS OF CONSUMABLES REMAINING (PROPELLANTS, CRYOGENICS AND
The trim angles will be set in by use of the gimbal position trim controls as required prior to each Delta V maneuver. The gimbal trim angles are verified for each axis (pitch and yaw) on the gimbal position display. The TVC 1 and 2 power switches would be set as indicated in Paragraph 7.2.1, and the thrust control switch on the Delta V display will be placed to the normal position.

The astronaut now will watch the event time clock countdown to zero. At some time prior to T=0, he initiates the ullage maneuver by commanding a 4° translation with the translation control. Moving the translation control to the stop will close relay K20 and apply commands to the SCS electronics. The AGC will initiate an engine fire command through relay K16 to the SCS electronics at T=0. This signal will be indicated by a light behind the thrust-on button on the Delta V display and also applied to the SPS engine through relay K25. Upon sensing the ignition of the SPS engine, the S/C selection logic will be inhibited and terminate the 4° translation approximately 1 second after engine ignition. Pitch and yaw attitude error and rate signals are inserted into the thrust vector control electronics at engine ignition. Attitude control of the S/C is accomplished in the pitch and yaw axes by gimbal control while thrusting. The roll attitude will be controlled by the roll reaction jets at all times. The astronaut will select the maximum deadband shortly after SPS engine ignition. The crew members will monitor S/C attitude control on the FDA.

And the velocity change on the Delta V display. When the Delta V remaining indicator indicates zero, the AGC should remove the thrust on command, and illumination of the thrust on push button should cease. If thrusting does not terminate, the thrust control switch, on the Delta V display, will be placed in the off position to perform emergency cut off of the SPS engine. Ignition of the engine is prevented with the thrust control switch in the off position. With removal of the thrust on signal, pitch and yaw control reverts back to SCS and RCS for attitude control approximately one second after the thrust on command is removed. The Delta V remaining indicator will measure the tail off velocity imparted to the S/C by the residual thrust from the SPS engine. This value would be inserted, where required, for trajectory calculations. Other operating modes can now be selected and the SPS actuator motors, TVC electronics, etc., be turned off if not required for the operating mode selected.

7.6 SCS Delta V Mode

SCS Delta V is a backup mode of operation for velocity changes imparted to the S/C by the SPS engine. The same criteria (magnitude and direction of velocity change) would determine if an SCS Delta V mode should be used, as with the G & N Delta V mode. Figure 7-5 is a block diagram showing the SCS configuration during an SCS Delta V mode.
7.6.1 Relay Configuration

Before SPS engine ignition, the SCS Delta V Mode will perform the same function as an SCS Attitude Control Mode by maintaining present attitude within the selected deadband. That is, relays K10 and K13 applying attitude errors to the FDI and SCS electronics, K5 total attitude to the FDI, K8 and K14 body rates to the FDI and SCS electronics, and K23 applying thrust ON/OFF commands to the SPS RCS. After SPS ignition, the SCS will continue to maintain the attitude by positioning the SPS gimbal through relay K26 in Pitch and Yaw, and roll attitude through relay K23 to the SPS RCS.

7.6.2 Operation

The astronaut will procedurally set the Delta V remaining indicator, position the SPS gimbal to the correct gimbal position trim angle, turn on the gimbal actuator motors, set time on event timer clock, and set the SCS TVC power switch on separate bus combination as required for the SCS Delta V Mode, and the thrust control switch on the Delta V display in the normal position. With the correct S/C attitude, the astronaut would initiate a 4x translation (ULLAGE) a predetermined amount of time before T=0 for SPS ignition. At T=0 with the ULLAGE in process, the thrust ON button on the Delta V display would be depressed to initiate the SPS thrust ON command. There is no interlock to prevent the manual thrust ON command from igniting the SPS engine without an ULLAGE. The time delay for inhibiting the RCS electronics is identical to the GAN Delta V Mode.

The Delta V remaining indicator will start counting down at the initiation of the ullage maneuver and the attitude error and rates will be displayed on the FDI. The attitude ball will not be driven in the SCS Delta V Modes except for manual attitude maneuver with the RCS.

Relay K18 closure represents depressing the thrust ON button and K35 represents the thrust control logic ON command. The astronaut will monitor the attitude error and rate displays on the FDI for any malfunctions that may occur and be displayed, and the Delta V remaining indicator to see that a thrust OFF command is generated automatically when the Delta V remaining indicator reaches zero. The thrust ON button will be illuminated at thrust ON and extinguished at thrust OFF as it was for the GAN Delta V Mode. Relay K19 represents the automatic thrust OFF command issued by the Delta V remaining indicator.

7.7 Manual Thrust Vector Control Mode

The Manual Thrust Vector Control (MTVC) mode is not selectable from the SCS control panel, but is selected by a full 360° rotation of the T handle on the thrust vector control with the SCS in a GAN or SCS Delta V Mode. The BAG's will be caged into backup rate mode and inserted into the MTVC electronics. This is accomplished by opening relay K14 and closing K15; however, the actual rate input path would be
Routed through a separate circuit, shown in Figure 5-2. The attitude error output is removed by opening relay K3. Other switching occurring in the SCS electronics removes all normal attitude error and rate signals from the servo amplifier and switches the input rate signals into the redundant TVC electronics. The output of the rotation control transducers is switched into the TVC electronics to provide rate damped manual control of the SPS signals.

The rate damped or attitude out mode removes the attitude error inputs to all three axes but will not remove the attitude and attitude error signals from the FIAI in the GAN Delta V, GAN Attitude control, and GAN Entry Modes. Therefore, rotation control commands can be used to fly GAN programmed attitude maneuvers. For the SCS Local Vertical, Attitude Control, and Entry Modes, Translation Control GAN switch closure will remove the attitude hold capability of each and the attitude ball will be driven to follow S/C rotations.

7.8 GAN Entry Mode

The GAN Entry Mode is the primary mode of control for entry into the earth’s atmosphere. This mode provides an automatic attitude hold and maneuver capability prior to, and throughout the aerodynamic entry. A block diagram of the SCS configuration in the GAN Entry Mode is shown in Figure 7-6.

In all previous modes the rate commands to the SCS have been limited approximately 0.5°/second. Selection of either Entry Mode changes this rate limit to 17°/second in the roll axis. The pitch and yaw rate commands are limited to 5°/second. Selection of either Entry Mode will not normally occur until after service module separation. The rate display scale range is switched to ± 25°/second in roll and ± 5°/second in pitch and yaw. The deadband setting for commands will normally be set to maximum. A rate deadband of ± 2°/second is incorporated by selecting either Entry Mode.

Attitude error information comes from the GAN system through relays K1 and K2 to the SCS electronics and FIAI for attitude control and display prior to 0.05° switching. Also, relays K6 and K4 connect body rates to the FIAI and SCS electronics and relay K5 applies total attitude to the FIAI. At 0.05°, the 0.05° ENTRY switch, on the SCS control panel, is placed in the 0.05° ENTRY position. At this time, switching in the SCS electronics causes the pitch and yaw channels to perform rate stabilization only.

As a backup to the automatic control capabilities through the SCS, the CH switch in the translation control can be used to remove attitude errors from the ACS. The astronauts may then use the rotation control to fly out command attitude errors displayed on the

FIGURE 7-6
FDAI FROM THE GAN SYSTEM. AFTER .05S ENTRY SWITCHING AND AERODYNAMIC BUILDUP, THE ROTATION CONTROL MAY BE USED TO EFFECTIVELY COMMAND ROLL ATTITUDE CHANGES. PITCH AND YAW COMMANDS ARE OPORED BY AERODYNAMIC FORCES. SELECTION OF THE GAN ENTRY MODE WILL CAUSE THE BMAGS THROUGH THE AGCU TO PROVIDE BACKUP ATTITUDE REFERENCE IF REQUIRED.

COMMANDS MAY ALSO BE ENTERED BY WAY OF THE AGC KEYBOARD AS A BACKUP TO THE GAN SYSTEM TOOK IN THE ENTRY PHASE WHEN THE PILOT CAN STILL SEE AND REACH THE COMPUTER DISPLAY. THE COMPUTER CAN DISPLAY ENTRY ANGLE, REQUIRED PULL-OUT ANGLE, AND OTHER ENTRY DATA. IF A MALFUNCTION OCCURS IN THE AUTOMATIC ENTRY, THE ASTRONAUT WOULD MANUALLY TAKE OVER AND FLY A CONTROLLED G LEVEL ENTRY IN THE SCS ENTRY MODE.

7-9 SCS ENTRY MODE

SCS Entry Mode is a backup to the GAN Entry Mode and will not be used unless a malfunction occurs in the GAN System (Figure 7-7).

Total Attitude Information is Applied to the FDAI Through Relay K5 From the AGCU When Maneuvering Prior to .05E, and All the Time After .05E. The BMAGs Provide Attitude Errors Through Relays K10 and K13 to the FDAI and SCS Electronics, Until .05E.

When the .05E Entry Switch is Placed in the .05E Entry Position, the BMAG Attitude Errors are Removed From the FDAI and SCS Electronics by Opening Relays K10 and K13. The BMAGs Are Caged Through Relays K11 and K12 to the AGCU. From This Point On, the FDAI Will Not Have an Attitude Error Indication. The Attitude Control Electronics Will Not Receive Attitude Error Inputs. Therefore, the S/C Will Not Have Automatic Attitude Control But Will Be Rate Stabilized. The Rate Gyro Signals Through the Control Electronics Will Oppose Any Angular Motion of the S/C. This Will Not Eliminate Any Attitude Error, But It Will Reduce the Rates to 2°/Second. The FDAI Will Continue to Indicate S/C Total Attitude and the Crew Will Initiate the Necessary Manual Roll Maneuvers.

After .05E, If Any One of the BMAGs Is Placed in a Backup Rate Condition (Relays K15 and K9 Close and K8 and K14 Open), the Other Two BMAGs Will Also Assume a Backup Rate Condition. This Will Also Remove Total Attitude Information From the FDAI Because the AGCU Cannot Operate with One BMAG in a Backup Rate Configuration. However, Only the BMAG Selected by the Rate Gyro Selection Switch Will Have an Input to the SCS for Rate Stabilization.

The Astronauts Will Control the Lift Vector Direction by Using the Rotation Control to Command the Desired S/C Roll Attitude. The Pointing Direction of Lift Vector (Up, Down, Etc.) Will Be Used to Control the G Level for Entry. Lift Vector Up Will Decrease the G Level, and Lift Vector Down Will Increase the G Level.
**7-10** RELAY CLOSING CONDITIONS FOR SCS-436A

The relay closures on SCS-436A are to represent functional signal inputs. They do not represent actual system relays.

K1 is closed by mode selection to provide attitude error for the FDI attitude error needle when monitor, G&N attitude control, G&N delta V, or G&N entry mode is selected.

K2 is closed by mode selection logic to provide attitude error into the SCS electronics when G&N attitude control, G&N delta V or G&N entry modes are selected except for attitude impulse enabled in the G&N attitude control mode or the G&N entry mode. Roll attitude error will pass through relay K2 after .05g entry switching is performed.

K3 is closed by mode selection to provide total attitude to the FDI attitude ball if the monitor, G&N attitude control, G&N delta V, or G&N entry mode is selected.

K4 is closed by mode selection to provide a torquing voltage through AGCU for the BMG's in the SCS local vertical mode.

K5 is closed by mode selection to provide total attitude to the FDI attitude ball from the AGCU total angle resolvers. Closure is initiated by selection of SCS local vertical, SCS attitude control, SCS delta V or SCS entry mode.

K6 is closed by placing the ATT SET switch to the ATT SET position. It provides inertial attitude error to the AGCU from a comparison of the total attitude shaft resolver in the AGCU and the attitude set dials. This relay would also be used during FDI align. These errors would be driven to zero as the AGCU is driven by the error input and the alignment process is completed.

K7 is closed by mode selection logic in all normal operating modes. Placing a BMG in a backup rate condition will inhibit rate gyro input to the channel selected and supply rate information from the caged BMG.

K8 is closed by axis as the rate selection switches on the SCS control panel are switched, placing the BMG's in a backup rate configuration. If relay K9 is closed for one channel, K8 would be open for the same channel.

K10 is closed by mode selection logic in all four SCS operating modes with the following exception: ATT SET switch to ATT SET position, after .05g entry switching, and as the rate backup configuration for each channel.

K11 is closed by mode selection logic from the monitor, G&N attitude control, and G&N entry mode. It is also closed for SCS local vertical, rotation control break-out switch (CSS) in all four SCS modes, one or more channels disabled for RCS control on the SCS control panel in the SCS modes, attitude impulse enabled in SCS attitude control mode, and .05g entry switches in the SCS local vertical.

K12 is closed at all times when K11 is closed except for SCS local vertical.

K13 is closed by mode selection logic with selection of any one of the four SCS modes except for the following conditions: Rotation control break-out switch closed; one or more control channels disabled; attitude impulse enabled in the SCS attitude control mode; and .05g entry switching in the SCS entry mode.

K14 is closed by mode selection logic in all operating modes except when a BMG is caged for backup rate operation per channel or attitude impulse enable in the G&N or SCS attitude control modes.

K15 is closed by placing the rate gyro control switches on the SCS control panel to the BMG position per channel of control.

K16 is closed when the AGCU initiates a SPS thrust on command.

K17 is closed when the attitude impulse control is enabled in the G&N or SCS attitude control modes.

K18 is closed when the THRUST ON button is depressed on the Delta V display with the SCS or G&N delta V modes selected.

K19 is closed when the delta V remaining indicator generates a thrust off command to the SCS electronics.

K20 is closed when a translation command is initiated after booster separation and before C/M-S/N separation.

K21 is closed when a rotation command is initiated to the SCS electronics, except for pitch and yaw while the SPS engine is thrusting.

K22 will be closed to the SM position while the S/M is attached and will transfer to the CM position after separation.

K23 is closed by the MESC at adapter or abort separation to enable the solenoid drivers.

K24 is closed when the TVC electronics is gimballing the SPS actuator.
K25 is closed when the SPS thrust ON-OFF electronics has a thrust ON commanded.

K26 operates identically to K22.