GUIDANCE AND CONTROL

The Apollo spacecraft is guided and controlled by two interrelated subsystems. One is the guidance and navigation subsystem. The other is the stabilization and control subsystem.

The two subsystems provide rotational, line-of-flight, and rate-of-speed information. They integrate and interpret this information and convert it into commands for the spacecraft's propulsion subsystems.

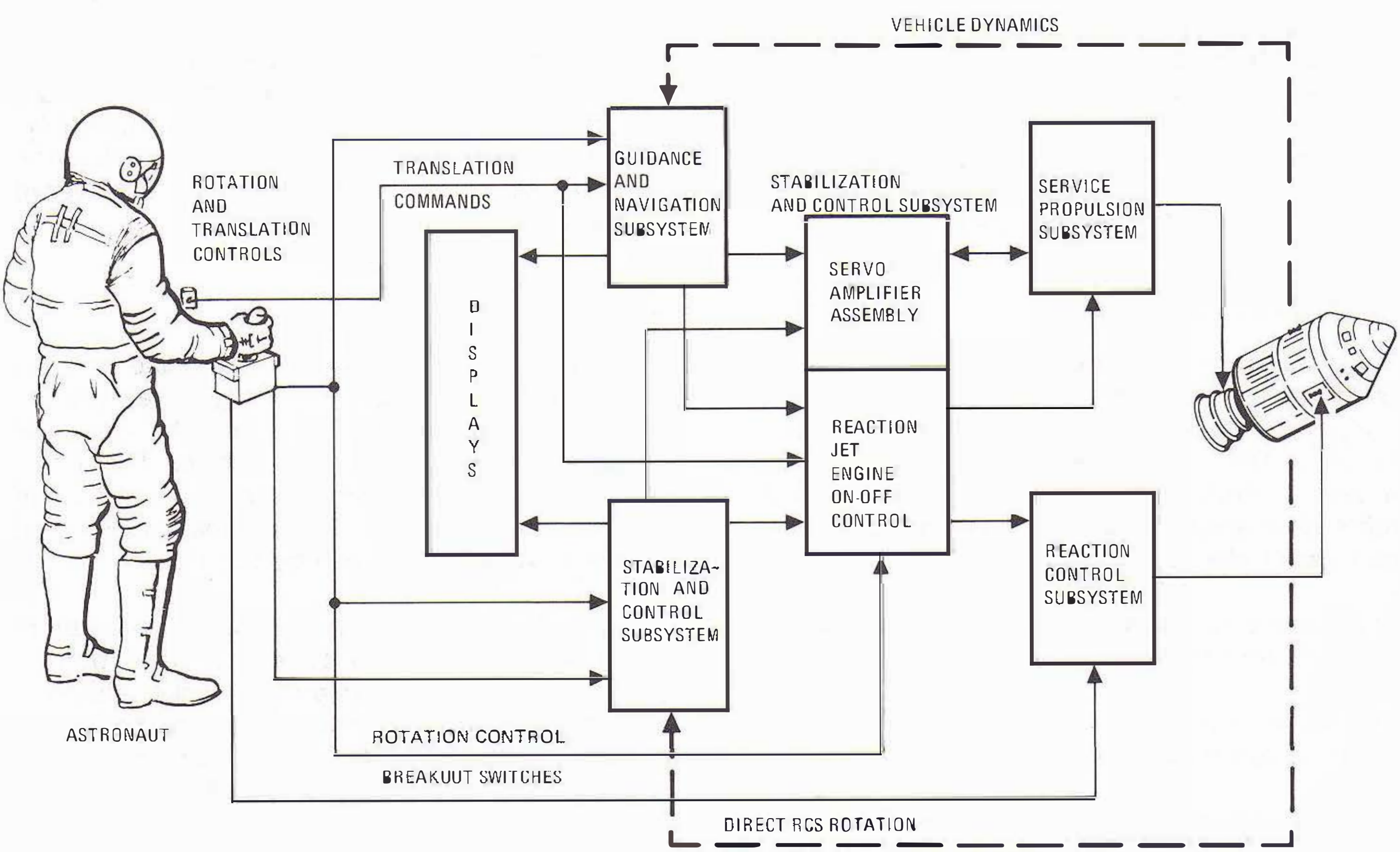
The guidance and navigation subsystem contains three major elements. They are the inertial, optical, and computer subsystems.

The inertial subsystem senses any changes in the velocity and angle of the spacecraft and relays this information to the computer. The computer digests the information and transmits any necessary signals to the spacecraft engines.

The optical subsystem is used to obtain navigation sightings of celestial bodies and landmarks on the earth and moon. It passes this information along to the computer for guidance and control purposes.

The computer subsystem uses information from a number of sources to determine the spacecraft's position and speed and, in automatic operation, to give commands for guidance and control. Data fed into the computer include: telemetry information from the ground regarding velocity, attitude, and position in space; a fixed memory which permanently stores navigation tables, trajectory parameters, programs, and constants; an erasable memory which stores intermediate results of computation, auxiliary program information, and variable data supplied by the guidance and control, and other subsystems of the spacecraft.

The other of the two interrelated subsystems is the stabilization and control subsystem. In general, it operates in these three ways: it determines the spacecraft's attitude (its angular position); it maintains the spacecraft's attitude; it controls the



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Guidance and control functional flow

direction of thrust of the service propulsion engine.

Both of the subsystems are used by the computer in the command module to provide automatic control of the spacecraft. This is done electronically. Manual control of the spacecraft's attitude and thrust is provided mainly through the stabilization and control subsystem equipment.

The space trajectory of Apollo is established essentially by the firing of engines, either its own or those of a launch vehicle. "Flying" Apollo, in the sense that an airplane is flown, is not possible because there is no atmosphere in space. The speed, altitude, and flight path angle of Apollo at the instant the last booster engine cuts off determines the characteristics of the spacecraft's orbit: its apogee (maximum altitude), perigee (minimum altitude), velocity and position at any point, and the time to complete an orbit.

Any additional engine firing automatically changes the flight path in space. To maneuver Apollo, the engines must be fired. The direction, amount, and duration of thrust of the service propulsion engine, for example, causes a change in the shape of the orbit about the earth, tilts the plane in which Apollo orbits the earth, or slows it down to permit entry and return to earth. During the flight to the moon, the firing changes the path to the moon (course correction), or slows the spacecraft at the proper time to permit the moon's gravity to pull Apollo into an orbit about the moon.

SPACECRAFT ATTITUDE

Two indicators on the main display console show the spacecraft's attitude. The indicators show the angle, or angular position, in reference to star sightings. These two indicators are called flight director attitude indicators. They tell the spacecraft's total angle (or position), the attitude errors, and rates of change.

The term total attitude refers to a combination of information from two sources.

One of the sources is the stable platform of the inertial measurement unit.

This provides total attitude information by holding the gimbaled, gyro-stabilized stable platform to a fixed inertial reference (maintained by periodic star sightings). In other words, no matter how the spacecraft moves, the stable platform retains its fixed, star-sighted position.

The second source is a gyro display coupler. This is a device which gives a reading of the spacecraft's actual attitude as compared with an attitude which the crew desires to maintain. A reading of the desired attitude is given by manually dialing roll, pitch, and yaw dials of the attitude set display in the console. The gyro display coupler then shows attitude errors by comparing total attitude information against the manually dialed attitude.

Information about attitude error also is obtained by comparison of the inertial measurement unit's gimbal angles with computer-reference angles. Another source of this information is gyro assembly No. 1, which senses any spacecraft rotation about any of the three axes.

Total attitude information goes to the command module computer as well as to the attitude indicators on the console.

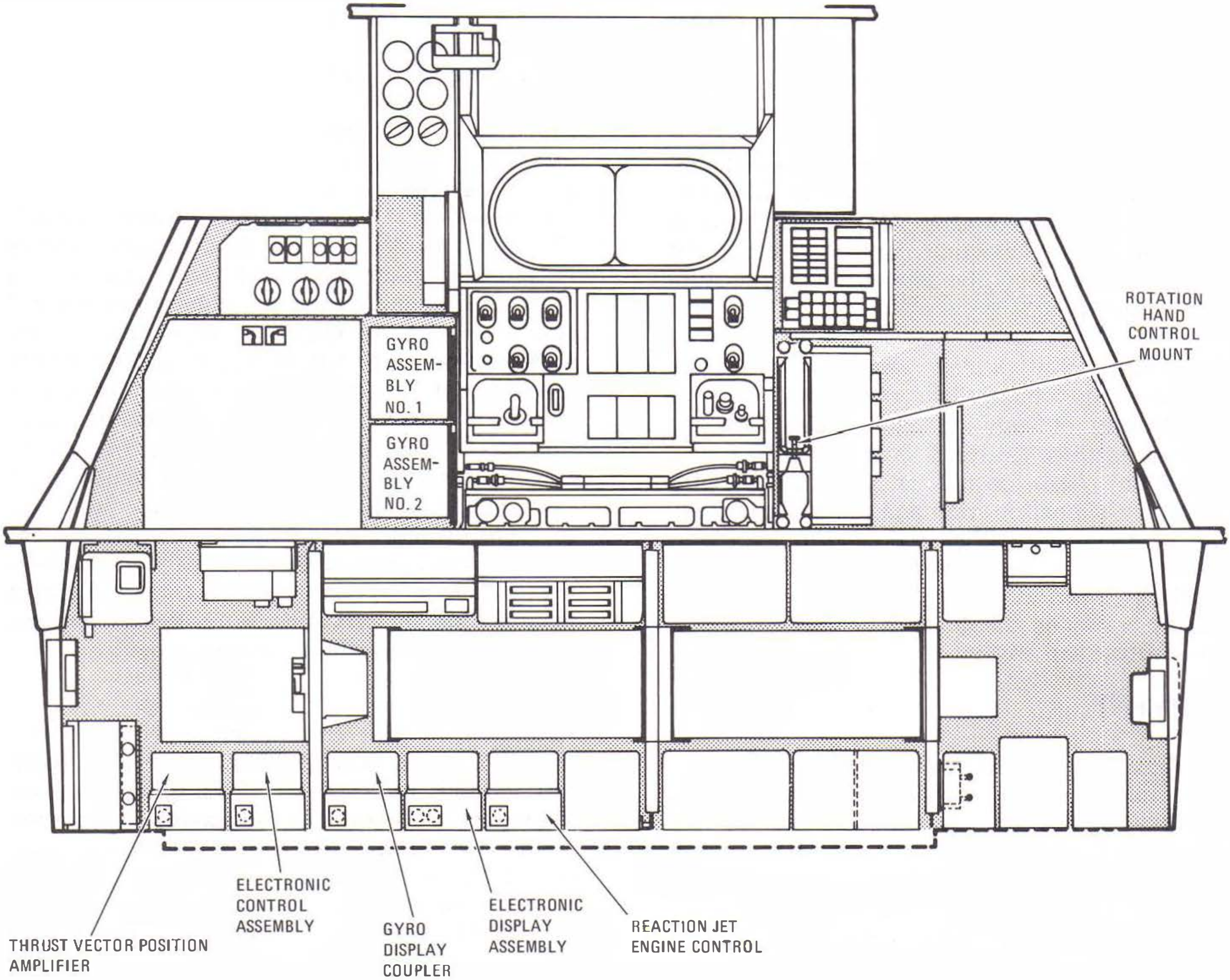
Attitude control of the spacecraft is provided for the purpose of maintaining a certain angle or for changing it during maneuver. If a specific attitude or orientation is desired, attitude error signals are sent to the reaction jet engine control assembly. Then the proper reaction jet fires in the direction necessary to return the spacecraft to the desired position.

THRUST CONTROL

The computer in the command module provides primary control of thrust. The flight crew pre-sets thrusting and spacecraft data into the computer by means of the display keyboard. The forthcoming commands include time and duration of thrust. Accelerometers sense the amount of change in velocity obtained by the thrust.

Thrust direction control is required because of center of gravity shifts caused by depletion of propellants in service propulsion tanks. This control is accomplished through electro-mechanical actuators which position the service propulsion engine, which rides on gimbals. Automatic control commands may originate in either the guidance and navigation subsystem or the stabilization and control subsystem. There is also provision for manual controls.

STABILIZATION AND CONTROL



P-239 *Location of main components of stabilization and control subsystem*

The stabilization and control subsystem provides control and monitoring of the spacecraft's attitude, and control of the firing direction or thrust vector of the service propulsion engine. It also serves as a backup system for inertial reference. The subsystem may be operated either automatically or manually. It is produced by Honeywell, Inc., Minneapolis, Minn.

All the control functions of this subsystem are backups to the guidance and navigation subsystem.

This subsystem is divided into three basic elements. It provides attitude reference, attitude control, and thrust direction control.

An electronic control assembly is used for attitude control and thrust direction control. This assembly has the circuits required for utilizing the rate and attitude error sensors. It also has the manual commands necessary to maintain backup stabilization and control in all axes (pitch, yaw, and roll).

An electronic display assembly provides the logic for establishing signal sources to be displayed and displays to be used. This assembly also provides for monitoring, isolation, and signal conditioning for telemetry of display signals.

There are three hand controls. One is for line-of-flight thrusting. Two are rotation controls for angular thrusting. The control for line-of-flight (or rectilinear motion) of the spacecraft also enables the crew to initiate abort during launch. It can also be used to transfer spacecraft control from the guidance and navigation subsystem to the stabilization and control subsystem.

The rotation control gives the crew manual control of spacecraft rotation in either direction about its three control axes. Also, it may be used for manual thrust direction control in pitch and yaw conditions when the service propulsion subsystem is thrusting.

The flight data display associated with the stabilization and control subsystem includes the flight director attitude indicators. They show the spacecraft attitude, attitude error, and angular rate information. Also located on the main display console is a gimbal position and fuel pressure indicator which shows either the service propulsion engine pitch and yaw gimbal angles or the booster propellant pressures. This provides a means for manually trimming the gimbals of the service propulsion subsystem engine.

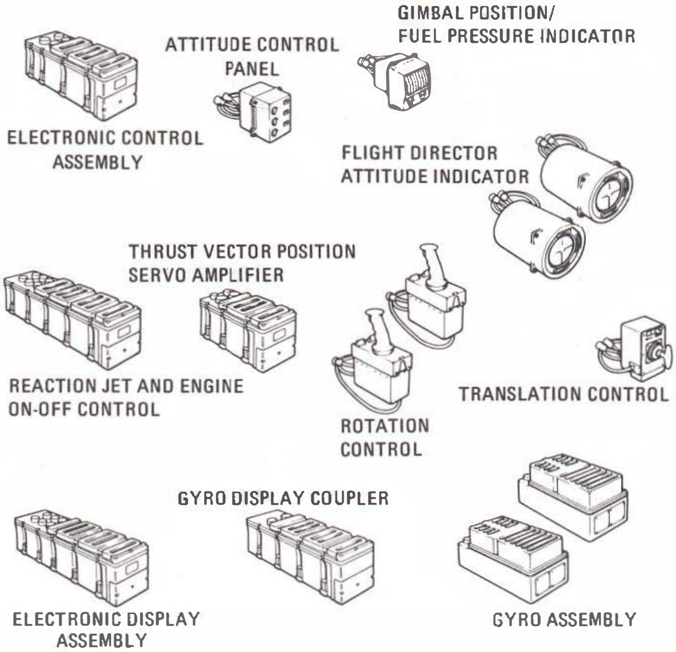
EQUIPMENT

Gyro Assemblies (Honeywell, Inc., Minneapolis) — Two units, each 6 by 7 by 14-1/4 inches weighing 22 pounds 7 ounces, are on the left-hand side of the navigation station in the lower equipment bay. Each contains three body-mounted attitude gyros mounted along body axes to sense attitude displacement. Components of each assembly are contained in a welded aluminum alloy enclosure designed to provide vacuum sealing and electrical continuity. In the attitude-hold mode, any displacement causes one gyro assembly to signal the reaction control system jets to restore the original attitude. The signal is displayed for the crew on the flight director attitude indicator. The other gyro assembly provides signals for display of body rates on either or both flight director attitude indicators, for rate damping, and for computation of inertial attitude changes.

Rotation Controls (Honeywell) — Two identical three-axes rotation controllers with control sticks provide proportional rate command signals for attitude or manual thrust vector maneuvers. Motion of the controls is analogous to rotation of the spacecraft about its axes. Direct angular acceleration command signals are provided when the direct reaction control system switch is actuated and the control is displaced to full stop position. Controls can be mounted on left and right couches and at the navigation position in the lower equipment bay. Each control is in a 3 by 5.5 by 7.26-inch metal box with a 5.25-inch aluminum stick handle. Each weighs 7 pounds, including cable and connector. A trigger-type push-to-talk switch is in the stick handle. There are redundant locking devices on each control. The control unit includes six breakout switches to provide on-off command and signals to the command module computer, stabilization control system minimum impulses, acceleration commands, caging of attitude gyros, and to enable proportional rate commands in the electronics; three transducers to command spacecraft rotation rates during proportional rate control and to command service propulsion engine gimbal position in pitch and yaw during manual thrust vector control; and direct switches in each axis for each direction of rotation. Direct switches produce acceleration commands through the direct solenoids on the reaction control system engines, bypassing stabilization control system electronics.

Translation Control (Honeywell) — This control with a T-handle stick provides a means of accelerating along one or more of the spacecraft axes. Motion of the control is analogous to translation motion of the spacecraft. The metal box measures 3.78 by 6.3 by 3.8 inches with a 3-inch rubber boot and aluminum T-handle. It weighs 5 pounds 11 ounces. The control is mounted with its axis approximately parallel to those of the spacecraft. Redundant switches close for each direction of control displacement. Clockwise switches transfer spacecraft control from the command module computer to the stabilization control system. One of the switches supplies 28 volts dc to a portion of the primary guidance navigation and control system. Redundant counterclockwise switches provide for initiation of manual abort during the launch phase. A discrete signal from switch closure is fed to the master events sequence controller, which initiates other abort functions.

Flight Director Attitude Indicator (Honeywell) — There are two indicators, each in a metal box 6.9 inches in diameter and 9.33 inches long weighing 9 pounds. They have glass cover plates. One is on Panel 1 in front of the commander



P-240 *Major stabilization and control components*

and the other on a panel to the right and directly above it, just over the display and keyboard. The display is a ball with circles showing pitch attitude, yaw attitude, and roll attitude, and three scales — one above and below the ball and a third to the right of it — showing pitch, yaw, and roll rate with needles showing pitch, yaw, and roll error. Attitude and attitude error signals to the displays are supplied by the stabilization and control system or the guidance and navigation system. Rate signals to the displays are supplied by the stabilization and control system only.

Attitude Set Control Panel (Honeywell) — This 3.5 by 4.8 by 4.4-inch panel weighs 3 pounds 6 ounces and is in the lower left-hand corner of the main display panel facing the commander. It has three windows to display roll, pitch, and yaw in degrees and three thumbwheels to the left of the windows. The panel provides the means for manually inserting desired attitude information into the stabilization control system in the form of three angles. It receives signals that represent the actual attitude of the spacecraft relative to an arbitrary inertial (fixed) reference frame. Output signals are provided, which represent the attitude error or the difference between the actual and desired total attitude of the spacecraft. These signals can be used to drive the attitude error needles on a flight director attitude indicator, thereby providing the astronaut with a visual indication of the spacecraft attitude error and to align the gyro display coupler to a fixed reference frame.

Gimbal Position and Fuel Pressure Indicator (Honeywell) — The indicator is in a 4.25 by 4.5 by 4.82-inch metal box with a single window showing four meter movements. The unit weighs 2 pounds 14 ounces and is directly below the flight director attitude indicator in front of the commander. It contains redundant indicators for both the pitch and yaw channels. During boost phases it displays second-stage fuel pressure on the redundant pitch indicators and third-stage fuel pressure on the two yaw indicators. The gimbal position indicator consists of two dual servometric meter movements mounted within a common hermetically sealed case. For a stabilization control system velocity change mode, manual service propulsion system engine gimbal trim capability is provided. Desired gimbal trim angles are set in with the pitch and yaw trim thumbwheels.

DETAILED DESCRIPTION

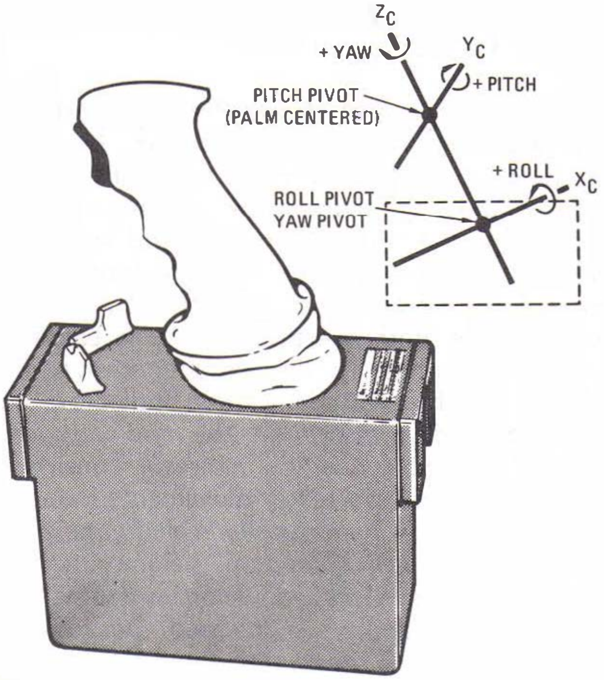
CONTROLS AND DISPLAYS

Stabilization and control subsystem controls and displays consist of two rotation controls, a translation control, an attitude set control panel, a gimbal position and fuel pressure indicator, two flight director attitude indicators, and two gyro assemblies.

Two identical rotation controls give the astronauts control of the spacecraft's rotation in either direction around all three axes. The controls are connected in parallel so that they operate in a redundant fashion without switching. Each axis of control performs three functions:

1. Breakout Switches — A switch closure occurs whenever the control is moved 1.5 degrees from its rest position. Separate switches are provided in each axis and for each direction of rotation. These six switches are used to provide command signals to the CM computer, stabilization and control minimum impulses, acceleration commands, attitude gyro caging, and proportional rate commands in the electronics.

2. Transducers — These produce alternating current signals proportional to the rotation control displacement from the null position which are used to command spacecraft rotation rates



P-241 *Rotation control*

and to command service propulsion engine gimbal position in pitch and yaw during manual thrust vector control. All three transducers can be used simultaneously.

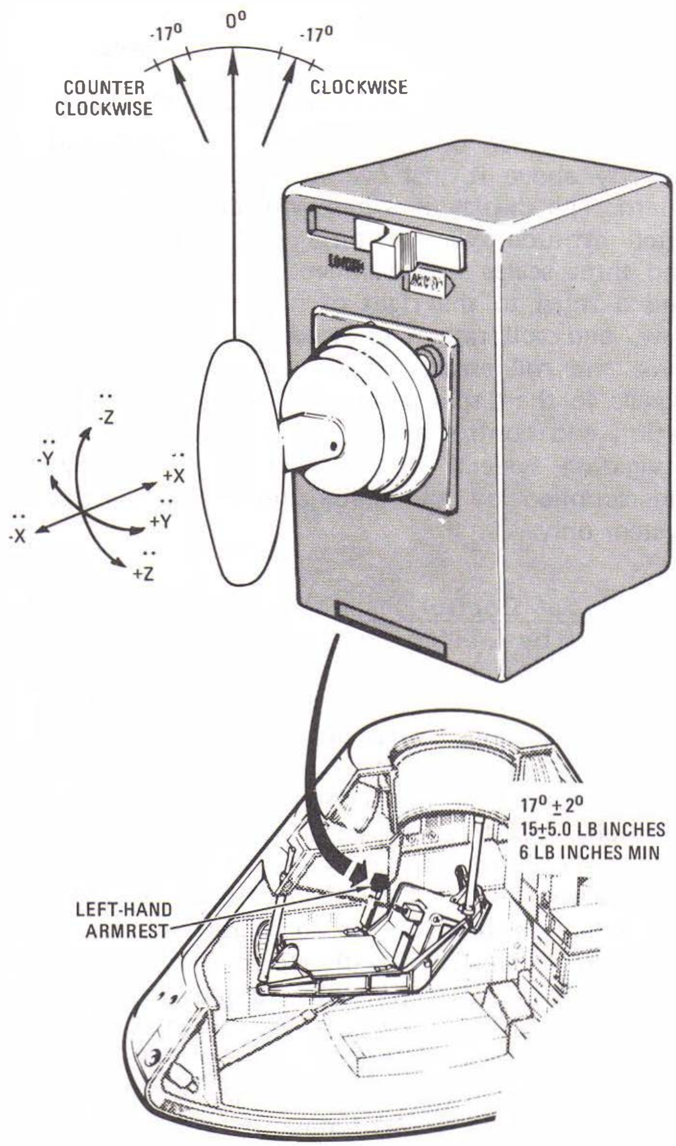
3. Direct Switches – A switch closure occurs whenever the control is moved a nominal 11 degrees from its null position (hardstops limit control movement to ±11.5 degrees from null in all axes). Separate switches are provided in each axis and for each direction of rotation. These switches are enabled by placing the DIRECT RCS switch to ON. Direct switch closure will produce acceleration commands through the direct solenoids on the RCS engines. All SCS electronics are bypassed.

The rotation control has a tapered female dovetail on each end of the housing which mates with mounting brackets on the couch armrests and at the navigation station in the lower equipment bay. When attached to the armrests, the input axes are approximately parallel with spacecraft body axes. The control is spring-loaded to null in all axes. A trigger-type push-to-talk switch also is located in the control grip. Redundant locking devices are provided on each control.

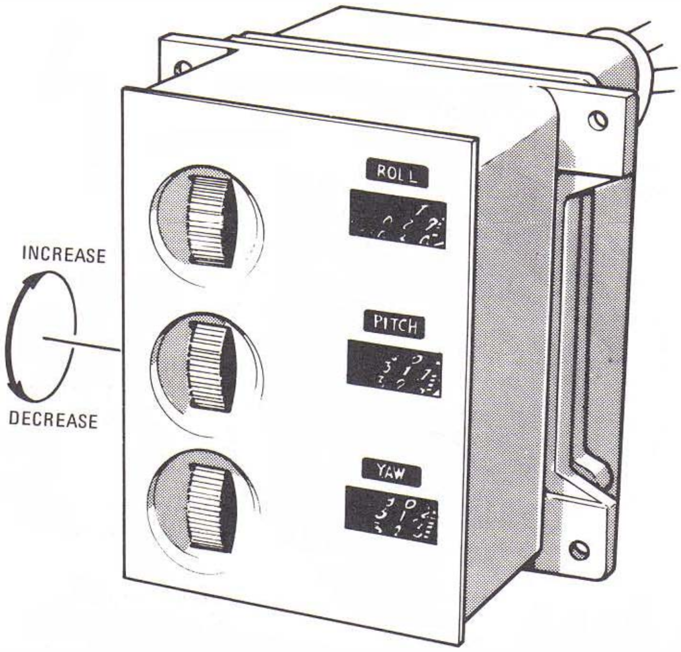
The translation control gives the astronauts a means of accelerating the spacecraft in either direction along any of the three axes. The control is mounted with its axes approximately parallel to those of the spacecraft. Redundant switches close for each direction of control displacement. These switches command the CM computer and the reaction jet and engine control. A mechanical lock inhibits the commands.

The controls T-handle may be rotated in either direction about the centerline of the shaft on which it is mounted. Hardstops for these rotations are ±17 degrees from null with detent positions encountered at a nominal ±12 degrees. In the detent position the hand can be removed and the T-handle will not return to null.

The clockwise switches will transfer spacecraft control from the CM computer to the stabilization



P-242 *Translation control*



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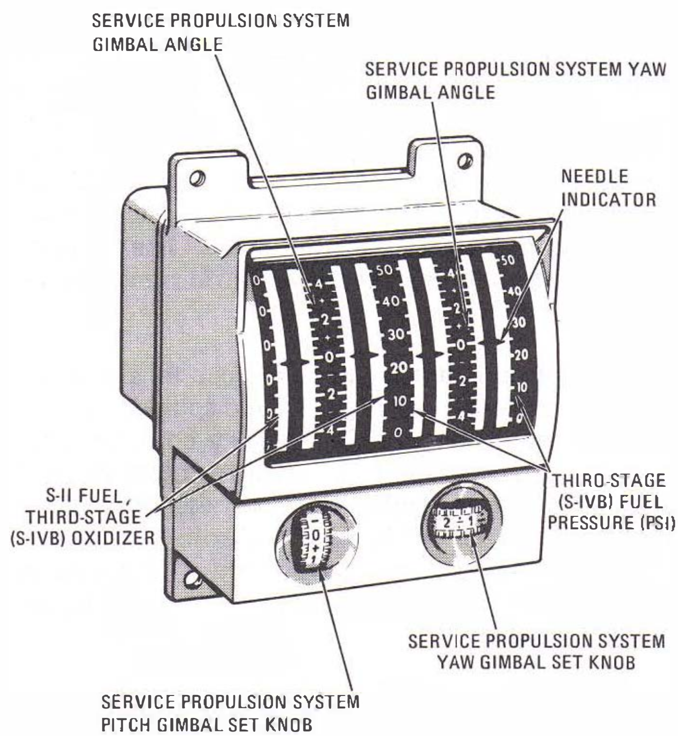
Attitude set display

and control subsystem. It may also transfer control between certain submodes within the stabilization and control subsystem. The redundant counterclockwise switches initiate abort during the launch phase. A discrete signal from switch closure is fed to the master events sequence controller which initiates other abort functions.

The attitude set control panel provides thumbwheels to position resolvers for each of the three axes. The resolvers are mechanically linked with indicators to provide a readout of the dialed angles. The signals to these attitude set resolvers are from the inertial measurement unit or the gyro display coupler.

The panel counters indicate resolver angles in degrees, and allow continuous rotation from 000 through 359 to 000 without reversing direction. There are graduation marks every 0.2 degree. Pitch and roll are marked continuously between 0 and 359.8 degrees. Yaw is marked continuously from 0 to 90 degrees and from 270 to 359.8 degrees; it is also marked with 0.2-degree graduation marks from 270 to 0 to 90 degrees and is numbered at 180 degrees. Readings increase for an upward rotation of the thumbwheels. One revolution of the thumbwheel produces a 20-degree change in the resolver angle and a corresponding 20-degree change in the counter reading.

The gimbal position and fuel pressure indicator contains redundant indicators for both the pitch and yaw channels. During the boost phases the indicators display fuel and oxidizer pressures for the Saturn second and third stages. Second stage fuel pressure (or third stage oxidizer pressure depending on the launch vehicle configuration) is on the redundant pitch indicators while third stage fuel pressure is on the two yaw indicators. The gimbal position indicator consists of two dual servometric meter movements mounted within a common hermetically sealed case. Thumbwheels enable crewmen to set service propulsion engine gimbal angler for stabilization and control subsystem velocity change maneuvers. Desired gimbal trim angles are set in with the pitch and yaw trim thumbwheels. The indicator displays service propulsion engine position relative to actuator null and not body axes. The range of the engine pitch and yaw gimbal displays is ±4.5 degrees. This range is graduated with marks at each 0.5 degree and reference numerals at each 2-degree division. The range of the fuel pressure scale is 0 to 50 psi with graduations at each 5-psi division and reference numerals at each 10-psi division.



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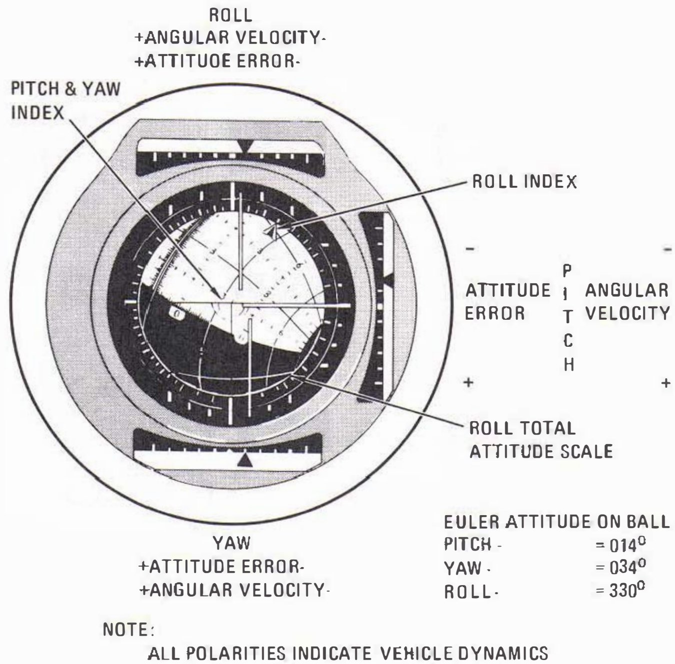
Fuel pressure/gimbal position indicator

The body rate (roll, yaw, or pitch) displayed on either or both flight director attitude indicators is derived from the body-mounted attitude gyros. Positive angular rates are indicated by a downward displacement of the pitch rate needle and by leftward displacement of the yaw and roll rate needles. The angular rate displacements are related to the direction of motion required by the rotation control to reduce the indicated rates to zero. The angular rate scales are marked with graduations at null and full range, and at 1/5, 2/5, 3/5, and 4/5 of full range. Full-scale deflection ranges are obtained with the FDAI SCALE switch and are 1, 5, and 10 degrees per second in pitch and yaw, and 1, 5, and 50 degrees per second in roll. Servometric meter movements are used for the three rate indicator needles.

The indicator's attitude error needles show the difference between the actual and desired spacecraft attitude. Positive attitude error is indicated by a downward displacement of the pitch error needle, and by a leftward displacement of the yaw and roll error needles. The attitude error needle displacements also are related to the direction of motion required by the rotation control to reduce the error to zero. The ranges of the error needles are 5 degrees or 50 degrees for full-scale roll error, and 5 degrees or 15 degrees for pitch and yaw error. The error scale factors are selected by a switch that also establishes the rate scales. The pitch and yaw attitude error scales contain graduation marks at null and full scale, and at 1/3 and 2/3 of full scale. The roll attitude scale contains marks at null, 1/2, and full scale. The attitude error indicators also use servometric meter movements.

Spacecraft orientation with respect to a selected inertial reference frame is also displayed on the attitude indicator ball. This display contains three servo control loops that are used to rotate the ball about three independent axes. These axes correspond to inertial pitch, yaw, and roll. The control loops can accept inputs from either the inertial measurement unit gimbal resolvers or gyro display coupler resolvers.

Pitch attitude is represented on the ball by great semicircles. The semicircle displayed under the inverted wing symbol is the inertial pitch at the time of readout. The two semicircles that make up a great circle correspond to pitch attitudes of θ and θ+180 degrees.



P-245 *Flight director attitude indicator*

Yaw attitude is represented by minor circles. The display readout is similar to the pitch readout. Yaw attitude circles are restricted to the intervals of 270 to 360 degrees (0) and 0 (360) to 90 degrees.

Roll attitude is the angle between the wing symbol and the pitch attitude circle. The roll attitude is more accurately displayed on a scale attached to the indicator mounting under a pointer attached to the roll (ball) axis.

The last digits of the circle markings are omitted. Thus, for example, 3 corresponds to 30, and 33 corresponds to 330. The ball is symmetrically marked about the 0-degree yaw and 0/180-degree pitch circles. Marks at 1-degree increments are provided along the entire yaw 0-degree circle. The pitch 180-degree semicircles have the same marking increments as the 0-degree semicircle. Numerals along the 300- and 60-degree yaw circles are spaced 60-pitch degrees apart. Numerals along the 30-degree yaw circle are spaced 30-pitch degrees apart. Red areas of the ball indicate gimbal lock.

Each gyro assembly contains three body-mounted attitude gyros mounted so that the input axis of one gyro is parallel to one of the spacecraft body axes. Gyro assembly No. 2 provides signals for display of body rates on either or both attitude indicators; rate damping for stabilization and control subsystem control configurations (excluding acceleration command and minimum impulse), and computation of inertial attitude changes.

The gyro assembly gyros provide signals equivalent to body attitude errors (deviations). These signals can be used for attitude control or display on the attitude indicator. The gyros also can provide backup rate signals for the functions of gyro assembly No. 2.

The Block II stabilization and control subsystem uses a switching concept as opposed to the "mode select" switching used in Block I Apollo spacecraft. Functional switching means the manual operation of a number of independent switches to configure the subsystem for various mission functions (e.g., course correction, velocity changes, entry, etc.). Mode switching would, for example, use one switch labeled "entry" to accomplish automatically all the necessary system gain changes, etc., for that mission phase. Mode selection simplifies crew tasks but limits system flexibility. Functional switching offers flexibility in selection of subsystem elements and allows part of a failed signal path to be turned off without affecting the total signal source.

There are two types of controls selectable from the main display console: stabilization and control subsystem or CM computer. The computer is the primary method of control and the stabilization and control subsystem is the backup. Attitude control is obtained from the reaction control engines and thrust vector control from the service propulsion engine.

ATTITUDE REFERENCE

The gyro display coupler provides signals to either of the attitude indicators for display of spacecraft total attitude and attitude errors. Angular velocity for display will always be supplied from either of the two gyro assemblies. The spacecraft total attitude display requires a connection between either gyro assembly and the attitude ball. This combination provides a backup attitude reference system for accurate display of spacecraft position relative to a given set of reference axes. Spacecraft attitude errors can be developed using the attitude set control panel connection between the gyro display coupler and the attitude indicator error needles. This combination provides a means of aligning the attitude reference system to a fixed reference while monitoring the alignment process on the error needles; it can also be used in conjunction with manual maneuvering of the spacecraft.

The gyro display coupler can be operated in the following configurations:

Alignment — Provides a means of aligning the coupler to a given reference.

Euler — Computes total inertial attitude and body attitude error from body rate signal inputs.

Non-Euler — Computes digital body rate signals from dc body rate signal inputs.

Entry $\leq$ 0.05 G — Computes roll axis attitude about the entry roll stability axis from body rate signal inputs.

The alignment function is used to align the coupler Euler angles (shafts) to the desired inertial reference selected by the attitude-set thumbwheels (resolvers). This is done by interfacing the coupler resolvers with the ASCP resolvers in each axis to generate error signals that are proportional to the difference between the resolver angles. These error signals are fed back to the gyro display coupler to drive the resolver angular difference to zero. During this operation all other functions for the coupler are inhibited.

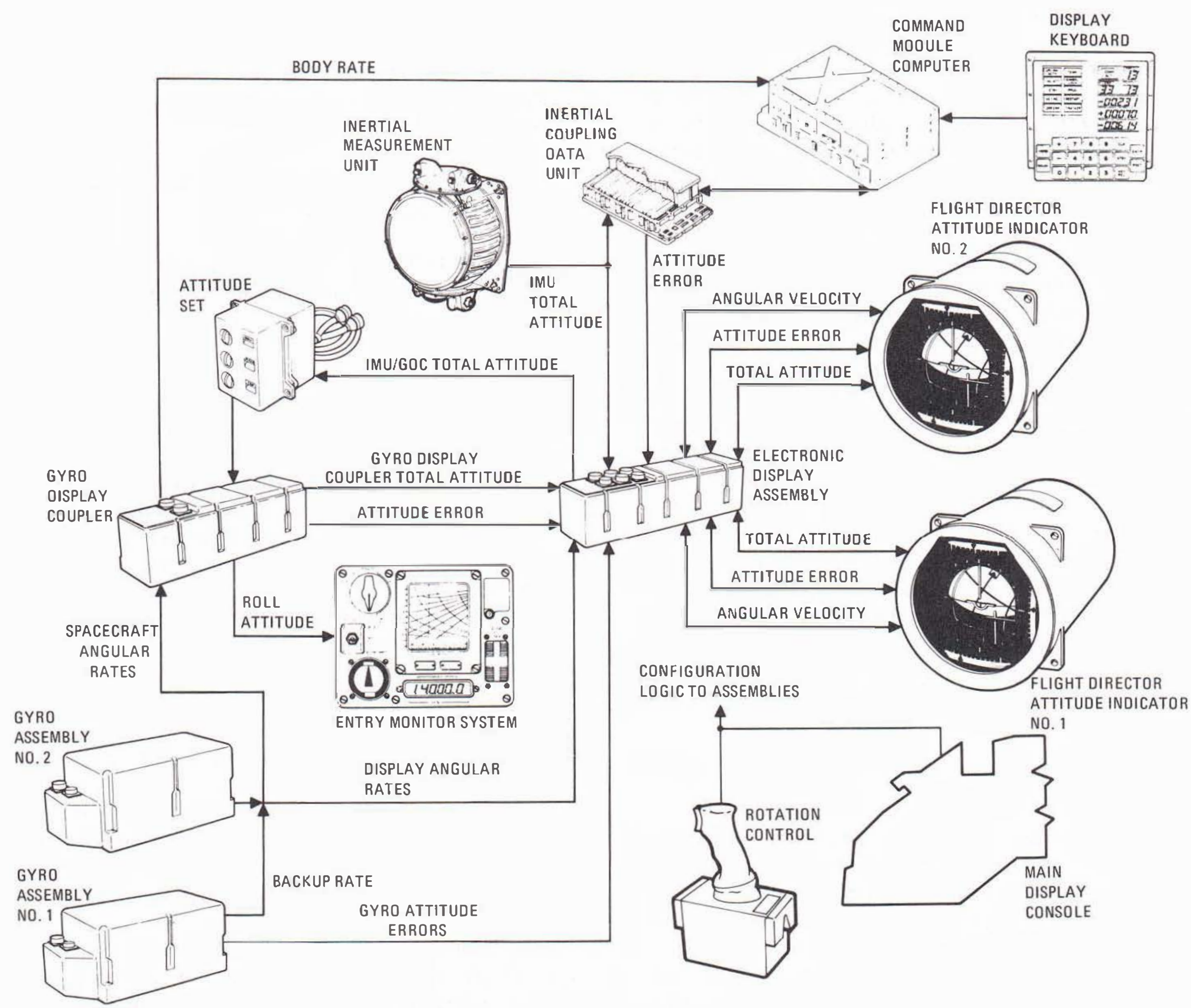
In the Euler configuration, the coupler accepts pitch, yaw, and roll dc rate signals from either gyro assembly and transforms them to Euler angles to be displayed on either attitude ball. The coupler Euler angles also are sent to the attitude set control panel to provide an Euler angle error, which is transformed to body angle errors for display on either attitude error indicator.

Non-Euler pitch, yaw, and roll dc body rate signals from either gyro assembly are converted to digital body rate signals and sent to the CM computer. Power is removed from both attitude ball-drive circuits when this configuration is selected.

In the entry ≤ 0.05 G operation, the coupler accepts yaw and roll dc body rate signals from either gyro assembly and computes roll attitude with respect to the stability axis to drive the roll stability indicator on the entry monitor system or from gyro assembly No. 1 and computes roll attitude with respect to the stability axis to drive either attitude ball in roll only.

The purpose of displaying total attitude, attitude error, and rate is to monitor the spacecraft and system functions. Since a single source can supply more than one type of information, the mission function required at a particular time will normally dictate what type of information is required from the source.

The flight director attitude indicator may be modified by an orbital rate display-earth and lunar unit. This unit is inserted electrically in the pitch channel between the electronic display assembly and attitude indicator to provide a local vertical display in the pitch axis. Local vertical is attitude with respect to the body (earth or moon) the spacecraft is orbiting. Controls on the unit permit selection of earth or lunar orbits and orbital attitude adjustment.



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Attitude reference equipment and flow

ATTITUDE CONTROL

The attitude and translation control portion of the stabilization and control subsystem uses the reaction control subsystem. The reaction control subsystem and stabilization and control equipment are described only in relation to attitude control. Stabilization and control equipment used for attitude control includes:

1. Gyro assembly No. 1, where three body-mounted attitude gyros provide pitch, yaw, and roll attitude error signals for use when automatic attitude hold is desired.

2. Gyro assembly No. 2, whose three body-mounted attitude gyros provide pitch, yaw, and roll rate damping for automatic control and proportional rate maneuvering.

3. The rotational controllers which enable crewmen to control the spacecraft attitude simultaneously in three axes.

4. The translation controller which enables the crew to command simultaneous accelerations along all three spacecraft axes and is also used to initiate several transfer commands.

5. The electronics control assembly which contains the electronics used for automatic, proportional rate, and minimum impulse capabilities.

6. The reaction jet engine control which contains the automatic reaction control subsystem logic and the solenoid drivers that provide commands to the automatic coils of the reaction control engines.

The reaction control subsystem provides the rotation control torques and translational thrusts for all attitude control functions. The reaction control engine is operated by applying excitation to a pair (fuel and oxidizer) of solenoid coils. Each engine has two pairs of solenoid coils, one automatic and the other direct.

Commands to the reaction control engines are initiated by switching a ground through the solenoid driver to the low side of the automatic coils. The solenoid drivers receive commands from the automatic reaction control subsystem logic circuitry contained in the reaction jet engine control. The automatic reaction control subsystem logic activates the command source selected and commands the solenoid drivers necessary to perform the desired attitude control function. The logic receives reaction control commands from the CM computer (for rotation and translation), the electronics control assembly (rotation for either automatic, proportional rate, or minimum impulse control), the rotation controls (for continuous rotational acceleration), and the translation control (for translational acceleration).

Commands to the direct coils have priority over those to the automatic coils. The direct coils receive commands from the rotation control when the "Direct RCS" switch is actuated and the control is deflected 11 degrees about one or more of its axes. The direct coils are used to command an ullage maneuver before a service propulsion engine firing of normal ullage methods are not available; this is controlled through a pushbutton on the main display console. The master events sequence controller also can command an ullage maneuver to enable separation of the CSM from the third stage. Direct coils on the SM reaction control engines are activated by the SM jettison controller for SM-CM separation. CM direct coils are activated by a switch on the main display console for reaction control propellant dumping during the final descent on the main parachutes.

The stabilization and control subsystem can be placed in various configurations for attitude control, depending on crew selection of control panel switches. The configuration desired is selected independently for each axis. Both automatic and manual control can be selected.

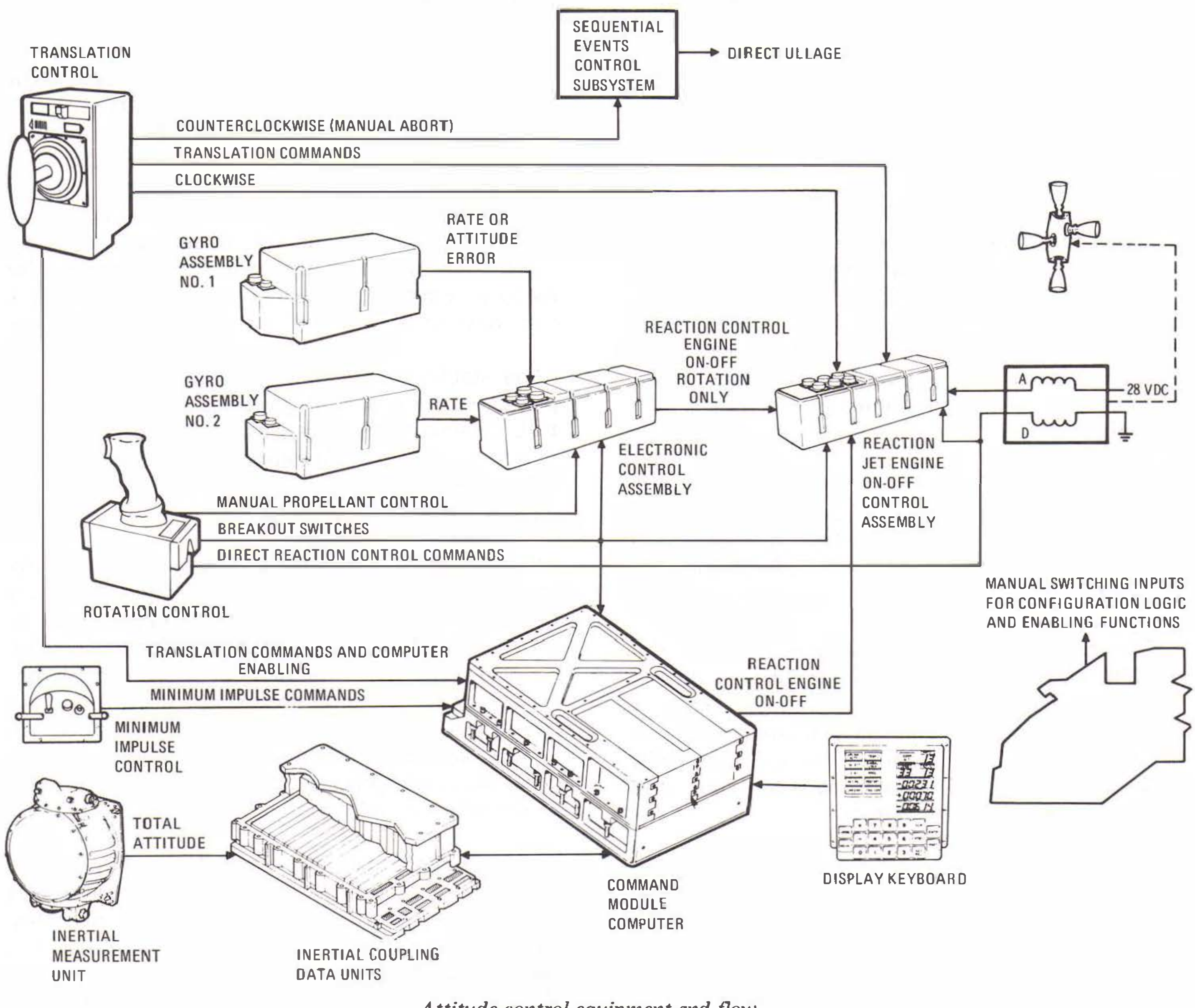
Automatic control involves rate damping and attitude hold. In rate damping large spacecraft rates are reduced to a small range (rate deadband) and held within the range. In attitude hold, angular deviations about the body axes are kept within certain limits (attitude deadband). If attitude hold is selected in pitch, yaw, and roll, the control can be defined as maintaining a fixed inertial reference. Rate damping is used in the mechanization of the attitude hold configuration.

Attitude hold uses the control signals provided by the body-mounted attitude gyros. These signals are summed in the electronics control assembly switching amplifier. The switching amplifier has two output terminals that provide commands to the automatic reaction control subsystem logic: one terminal provides positive rotation commands and the other negative commands. If the magnitude of the signal input is smaller than a specific value, neither output is obtained. The input level required to obtain an output is referred to as the switching amplifier deadband. This can be interpreted as a rate deadband or an attitude (minimum) deadband. The deadband limits are a function of the control loop gains which depend on the position of the "Rate" switch: low is ±0.2 degree per second and high is ± 2 degrees per second. An additional deadband can be selected for attitude control: ± 0.2 to 4.2 degrees in low and ±4 to 8 degrees in high.

When the summation of rate and attitude signals exceeds the switching amplifier deadband, a rotation command is sent to the reaction control subsystem and the engines are automatically fired for the duration needed to correct the deviation.

Manual attitude control involves proportional rate, minimum-impulse, acceleration command, and direct control. These commands are initiated by operation of either rotation control. With the exception of direct, the rotation control commands go through the reaction control subsystem automatic coils.

Proportional rate provides the ability to command a spacecraft rate that is directly related to



Attitude control equipment and flow

the amount of rotation control stick deflection. This capability is obtained by summing the control's transducer output with the body-mounted attitude gyro signal in the electronics control assembly; when the stick is deflected, an error is developed at the switching amplifier input that results in an acceleration command. The command is present until the gyro signal is large enough to reduce the error to less than the deadband. The spacecraft will then coast at a constant rate until the rotation control input is removed.

Minimum impulse provides the ability to make small changes in the spacecraft rate. When minimum impulse is enabled in an axis, the output of the switching amplifier in that axis is inhibited. Thus, the spacecraft (attitude) is in free drift in the axis where minimum impulse is enabled if direct control is not being used. Minimum impulse is commanded by the rotation control breakout switch. When minimum impulse is selected in the roll axis, one-half of the roll solenoid drivers are inhibited for minimum impulse commands. A roll minimum impulse command is executed by two reaction control engines unless one of the "Channel Roll" switches is turned off, which reduces the command to a single engine.

When acceleration command is activated and a breakout switch is closed, continuous commands are sent to the appropriate reaction control subsystem automatic coils. The selection of acceleration command in an axis inhibits all other inputs to the automatic reaction control subsystem logic for that axis. This differs from minimum impulse selection in that translation control commands are available during minimum impulse control.

Direct rotation control is available as backup to any other control, including CM computer when the "Direct RCS" switch is on. When the rotation control stick is deflected to hard stop, a direct switch is closed and the voltage is routed to the direct coils on the appropriate reaction control engines. The control's direct switch also routes a signal to the automatic reaction control subsystem logic that inhibits all automatic coil commands in the axis under direct control.

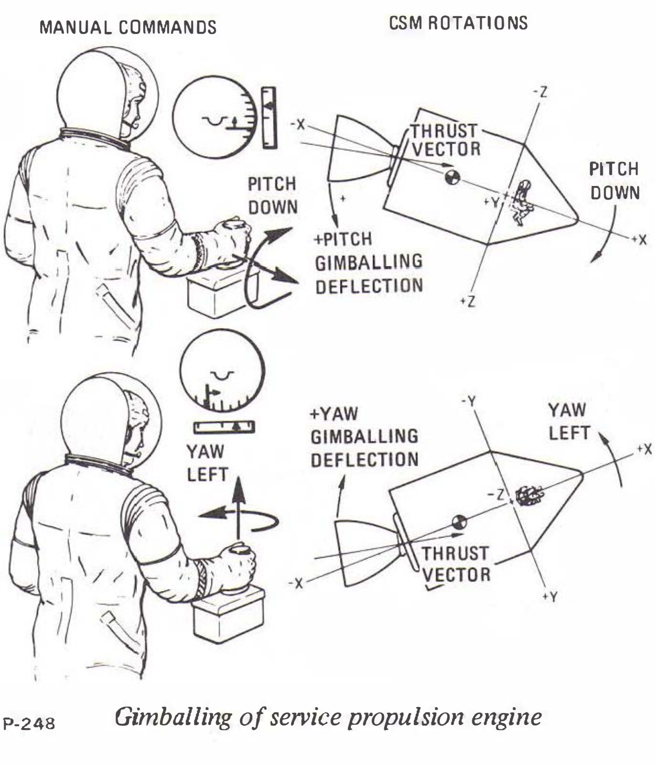
Commands from the translation control can be initiated simultaneously in the three axes and appear as logic inputs to the automatic reaction control subsystem logic. The logic signals are obtained from switch closures in the control. Translation control is not available after CM/SM separation.

THRUST VECTOR CONTROL

Spacecraft attitude is controlled during a velocity change by positioning the engine gimbals for pitch and yaw control while maintaining roll attitude with the attitude control subsystem. The stabilization and control electronics can be configured to accept attitude sensor signals for automatic control or rotation control signals for manual control. Manual thrust vector control can be selected to use vehicle rate feedback signals summed with the manual signals. A different configuration can be selected for each axis; for example, one axis can be controlled manually while the other is controlled automatically.

In automatic thrust vector control, spacecraft angular rates and attitude errors are sensed by the body-mounted attitude gyros. Attitude error, gimbal position, and gimbal trim signals are summed at the input to an integrator amplifier. The integrator output is then summed with rate, attitude error, gimbal position, and gimbal rate at the servo amplifier input.

Steady-state operation is obtained when the gimbal is positioned so that the thrust vector is

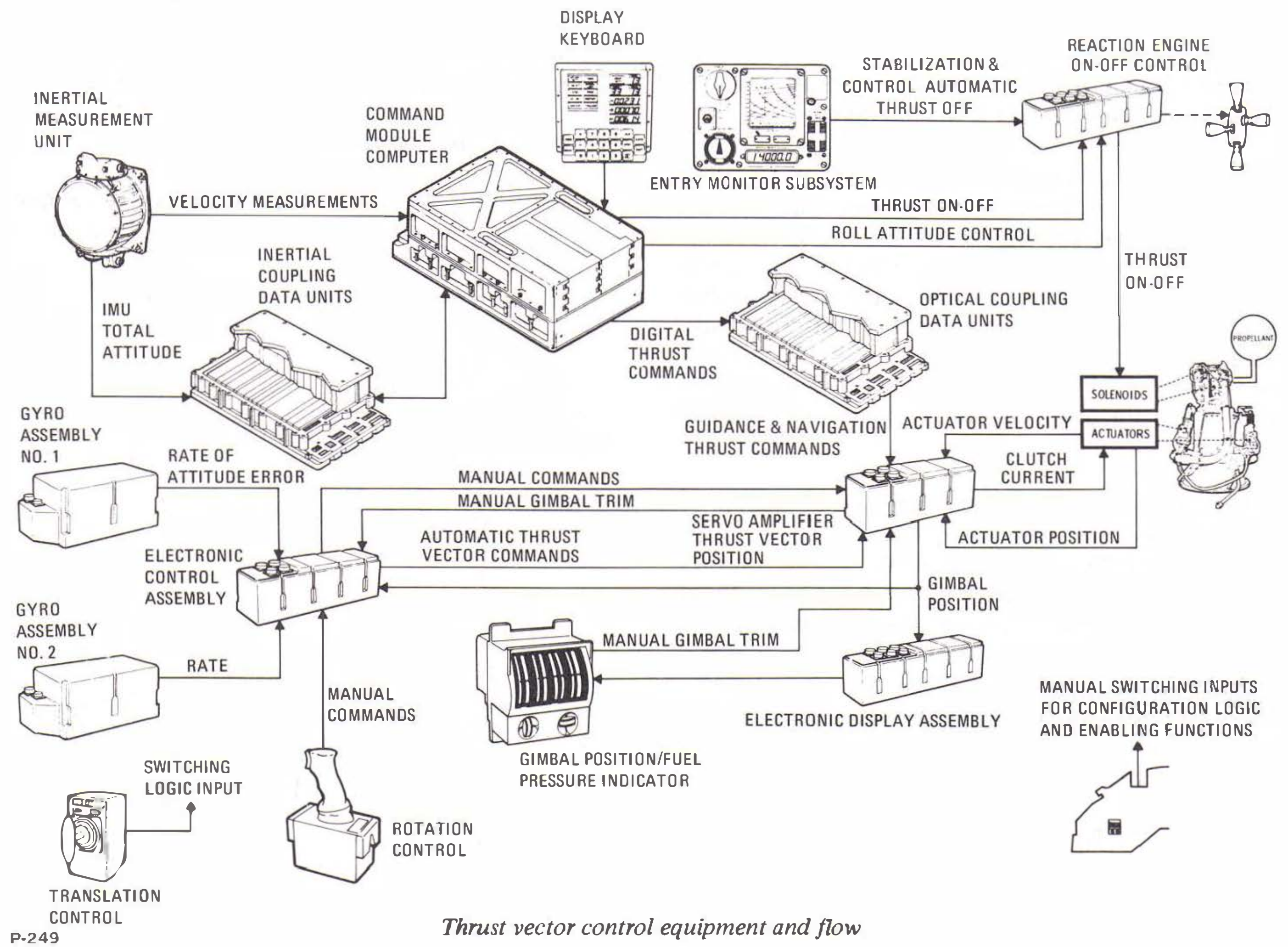


P-248 *Gimballing of service propulsion engine* aligned through the vehicle center of gravity and the error at both summing points is a constant-zero. The integrator input error is zero when gimbal position minus gimbal trim is equal to the negative of the attitude excursion sensed by gyro assembly No. I. This is the spacecraft/gimbal orientation necessary to obtain and maintain the desired thrust direction. Transients due to center-of-gravity uncertainty errors or shifts during thrusting are forced by the integrator to have the necessary steady-state solution. However, final pointing vector errors will be incurred because of the quadrature accelerations induced during the transient phases. Errors also will result from amplifier gain and component inaccuracies.

The gimbals are trimmed before thrust by turning the trim wheels on the gimbal position indicator. The trim wheel in each axis is mechanically connected to two potentiometers connected with the gimbal servomechanisms. It is desirable to trim before velocity change to minimize the transient duration time and the accompanying quadrature acceleration. The trim wheels also are set before a velocity change controlled by the CM computer so that the stabilization and control subsystem can relocate the desired thrust direction if a transfer is required after engine ignition.

In manual thrust vector control, the signal from the rotation control is sent to a proportional plus integral amplifier. This circuit maintains a gimbal deflection after the rotation control is returned to rest and makes corrections with the control about its rest position, rather than holding a large displacement. Depending on how switches are set, it also can damp out spacecraft rate.

There are two manual thrust vector control configurations: rate command and acceleration command. Rate command is similar to the proportional rate control in the attitude control subsystem except there is no deadband. The thrust



Thrust vector control equipment and flow vector is under body-mounted attitude gyro control. If there is an initial gimbal center-of-gravity misalignment, an angular acceleration will develop. The gyros, through the proportional gain, will drive the gimbal in the direction necessary to cancel this acceleration. The rate feedback is inhibited in acceleration command; the rotation control input must be properly trimmed to position the thrust vector through the center of gravity. This drives the rotational acceleration to zero but additional adjustments are necessary to cancel residual rates and obtain the desired attitude and positioning vector.

ENTRY MONITOR SYSTEM

The entry monitor system provides a visual display of automatic guidance navigation and control system entry and velocity change maneuvers. It also provides sufficient display data to permit manual entry in case of guidance and control malfunctions and automatic velocity cutoff commands for the stabilization and control subsystem when controlling the service propulsion engine. The velocity display also can be used to cut off thrust manually if the automatic commands malfunction.

The system provides five displays used to monitor an automatic entry or perform a manual entry: threshold indicator, roll attitude indicator, corridor verification indicators, range displays, and the flight monitor.

The threshold indicator, labeled .05G, displays sensed deceleration. The altitude at which this indicator is illuminated depends on entry angle (velocity vector with respect to local horizontal), the magnitude of the velocity vector geographic location and heading, and atmospheric conditions. Bias comparator circuits and timers are used to activate this indicator. The signal used to illuminate the indicator is also used in the system to start the corridor evaluation timer, scroll velocity drive, and range-to-go circuits.

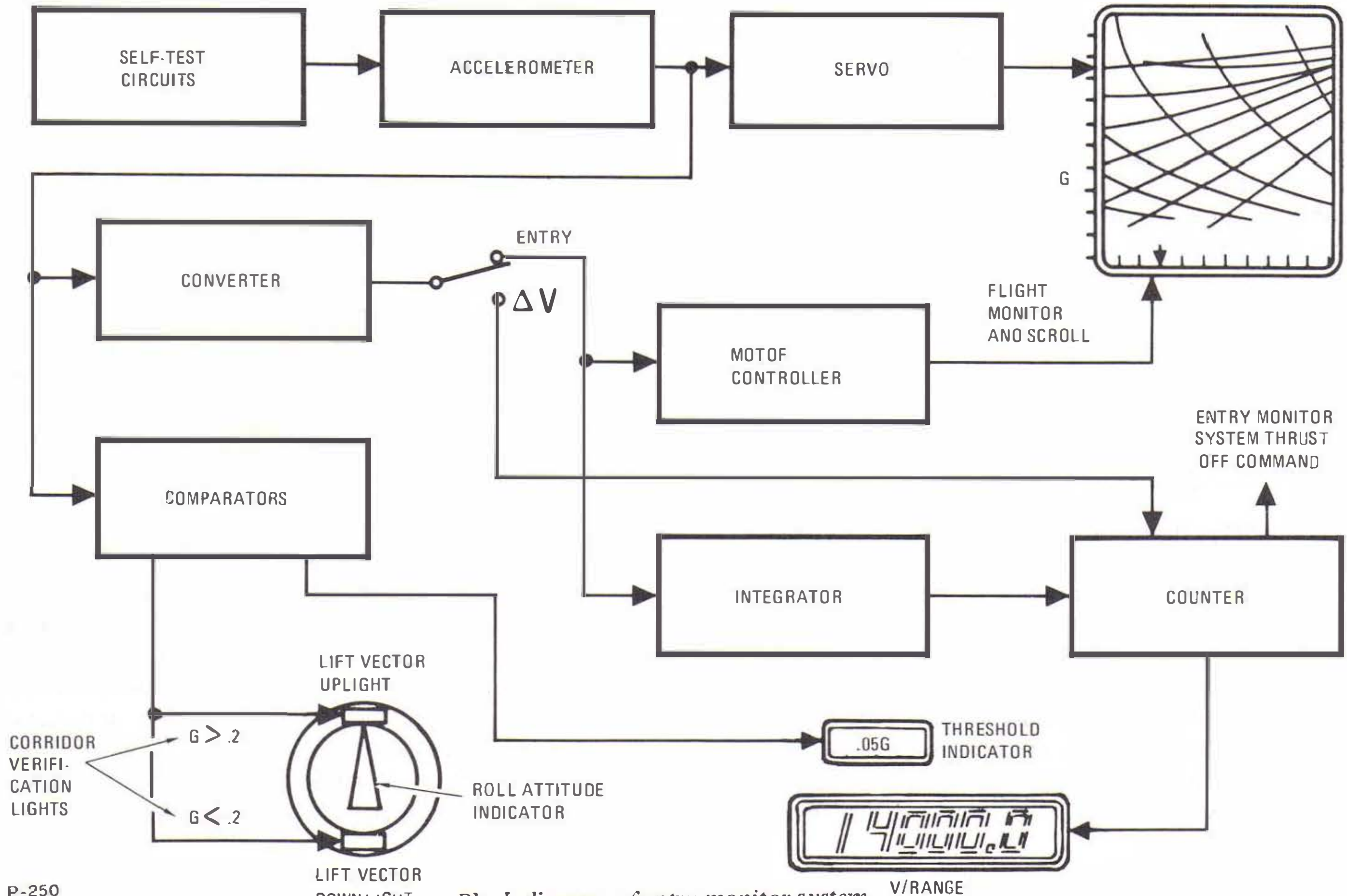
The roll attitude indicator displays the lift vector position throughout entry. During entry, stability axis roll attitude is supplied to the indicator by the gyro display coupler. There are no degree markings on the display, but the equivalent readout will be zero when the indicator points toward the top of the control panel and increases up to 360 in a counterclockwise direction.

The corridor verification indicators are located on the roll attitude indicator. They consist of two lights which indicate the necessity for lift vector up or down for a controlled entry. (The indicators are valid only for spacecraft entering at velocities and angles that will be used on the return from the moon.) The corridor comparison test is performed approximately 10 seconds after the .05G indicator is illuminated. The lift vector up light (top) indicates greater than approximately 0.2G. The lift vector down light (bottom) indicates less than approximately 0.2G. An entry angle is the angle displacement of the CM velocity vector with respect to local horizontal at 0.05G. The magnitude of the entry angles that determines the capture and undershoot boundaries depend on the CM lift-to-drag ratio. Entry angle less than the capture boundary will result in noncapture regardless of lift orientation. Noncapture would result in an elliptical orbit which will re-enter when perigee is again approached. The critical nature of this would depend on CM consumables: power, control propellant, lift support, etc. The CM and crew would undergo excessive G force (greater than 10G's) with an entry angle greater than the undershoot boundary, regardless of lift orientation.

The range display is an electronic readout of inertial flight path distance in nautical miles to predicted splashdown after 0.05G. The predicted range will be obtained from the guidance and control subsystems or ground stations and inserted into the range display before entry. The range display also shows velocity in feet per second during service propulsion engine thrusting.

The flight monitor provides an entry trace of total G level versus inertial velocity. A Mylar scroll has printed guide lines which provide monitor (or control) information during aerodynamic entry. The entry trace is generated by driving a scribe in a vertical direction as a function of G level, while the Mylar scroll is driven from right to left proportional to the CM inertial velocity change. Monitor and control information for safe entry and range potential can be observed by comparing the slope of entry trace to the slope of the nearest guide lines.

In addition to entry functions, the entry monitor system provides outputs related to delta velocity maneuvers during either service propulsion or reaction control engine thrusting. Displays include a lamp which lights any time the service propulsion engine fires and a counter which shows



Block diagram of entry monitor system

the velocity remaining to be gained or lost. The latter display can have a range of 14,000 to -1000 feet per second in tenths of a foot per second. The desired velocity change for all service propulsion engine thrusting maneuvers is set in the panel and the display will count up or down. During thrust controlled by the stabilization and control subsystem, the entry monitor system automatically turns off the service propulsion engine when the display reads minus values.

The Mylar scroll in the entry monitor systems flight monitor has ground and flight test patterns together with four entry patterns. Each entry pattern is preceded by two identical flight test patterns and entry instructions that are used to verify operation of the system's entry circuits and set initial conditions. Each entry pattern contains velocity increments from 37,000 to 4,000 feet per second as well as entry guidelines. The entry guidelines are called G on-set or G off-set and range potential lines. During entry the scribe trace should not become parallel to either the nearest G on-set or G off-set lines. If the slope of the entry trace becomes more negative than the nearest G on-set line, the CM should be oriented so that a positive lift vector orientation (lift vector up) exists to prevent excessive G buildup. If the entry trace slope becomes more positive than the nearest G off-set line, the CM should be oriented to produce negative lift (lift vector down). The G on-set and off-set lines are designed to allow a 2-second crew response time and a 180-degree roll maneuver if the entry trace becomes parallel to the target of the nearest guideline.

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

The vertical line on the scroll corresponds to where the CM velocity becomes suborbital; that is, where the velocity has been reduced to less than that required to maintain orbit. The full positive lift profile line represents the steady-state minimum-G entry profile for an entry.