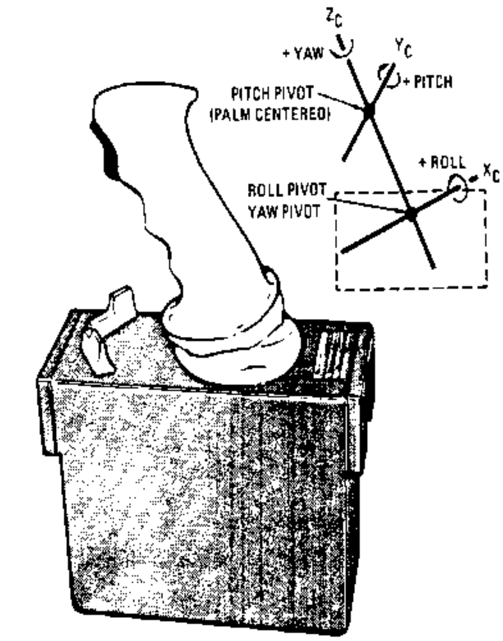


P-241 *Rotation control*

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

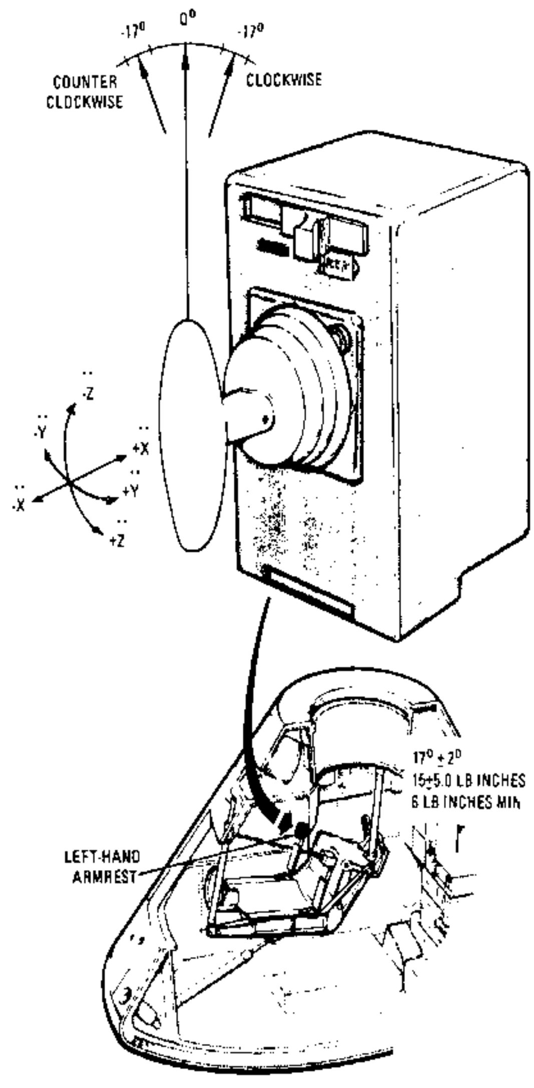
3. Direct Switches – A switch closure occurs whenever the control is moved a nominal 11 degrees from its null position (hardstops limit control movement to ±11.5 degrees from null in all axes). Separate switches are provided in each axis and for each direction of rotation. These switches are enabled 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.

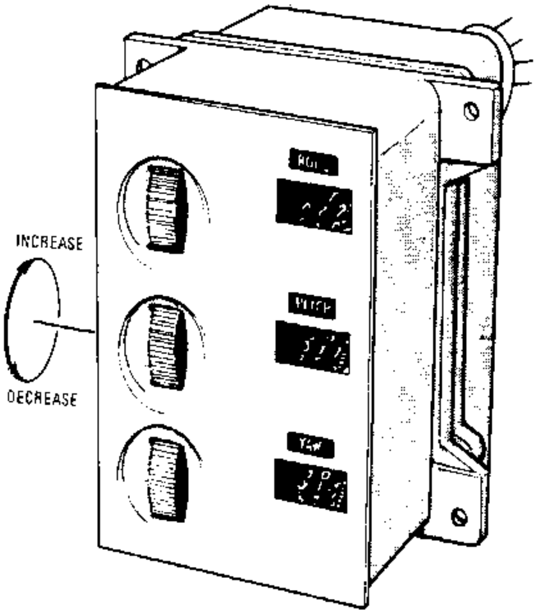
The translation control gives the astronauts a means of accelerating the spacecraft in either direction along any of the three axes. The control is mounted with its axes approximately parallel to those of the spacecraft. Redundant switches close for each direction of control displacement. These switches 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*



P-243

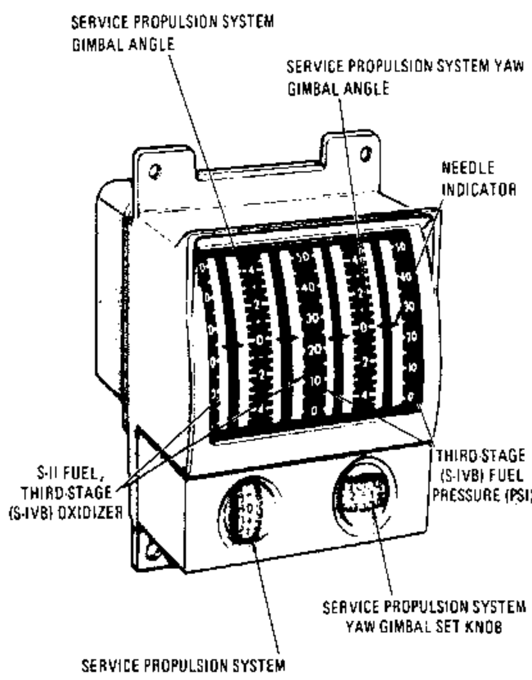
Attitude set display

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.



P-244

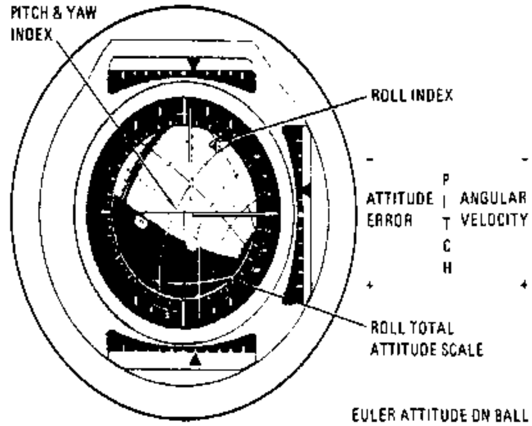
Fuel pressure/gimbal position indicator

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 ≤ 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.

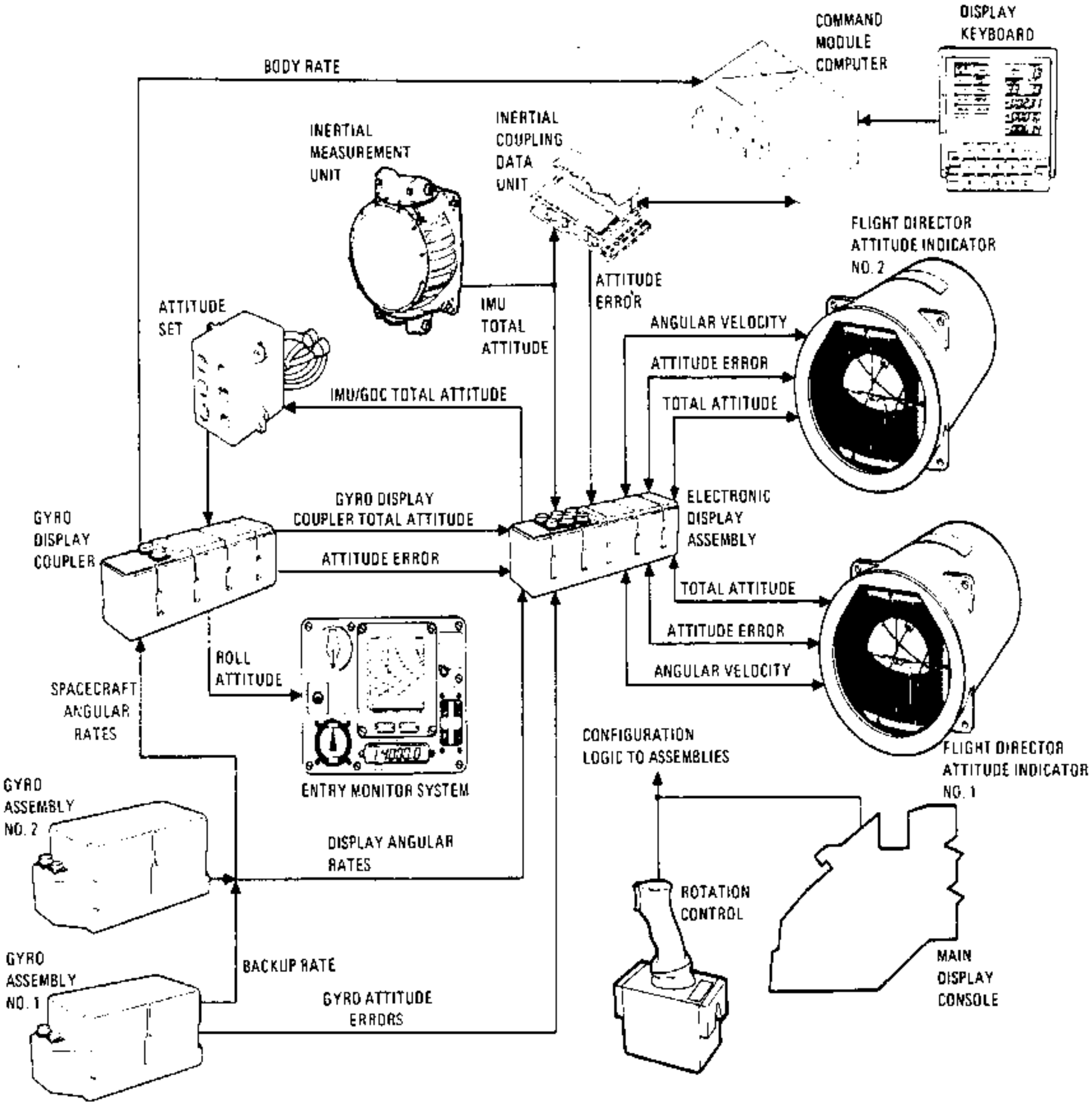
In the Euler configuration, the coupler accepts pitch, yaw, and roll dc rate signals from either gyro assembly and transforms them to Euler angles to be displayed on either attitude ball. The coupler Euler angles also are sent to the attitude set control panel to provide an Euler angle error, which is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

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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.

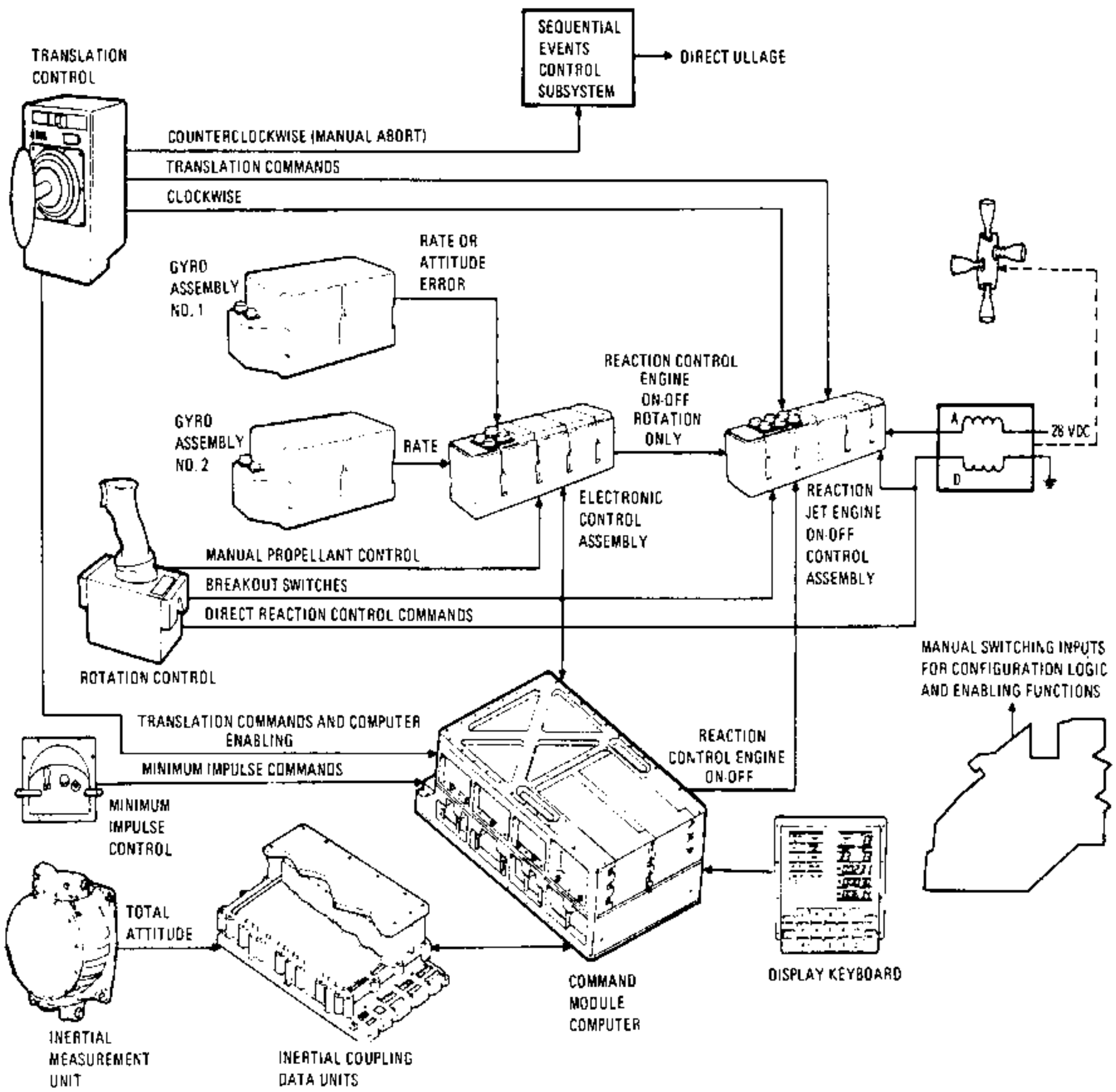
Automatic control involves rate damping and attitude hold. In rate damping large spacecraft rates are reduced to a small range (rate deadband) and held within the range. In attitude hold, angular deviations about the body axes are kept within certain limits (attitude deadband). If attitude hold is selected 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



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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.

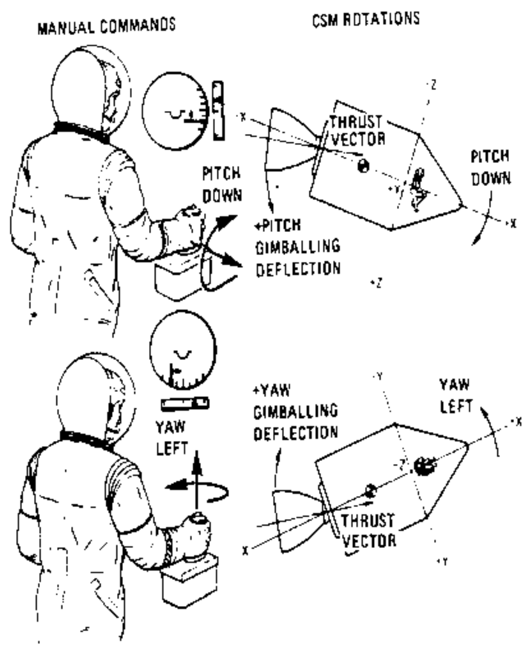
Commands from the translation control can be initiated simultaneously in the three axes and appear as logic inputs to the automatic reaction control subsystem logic. The logic signals are obtained from switch closures in the control. Translation control is not available after CM/SM separation.

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

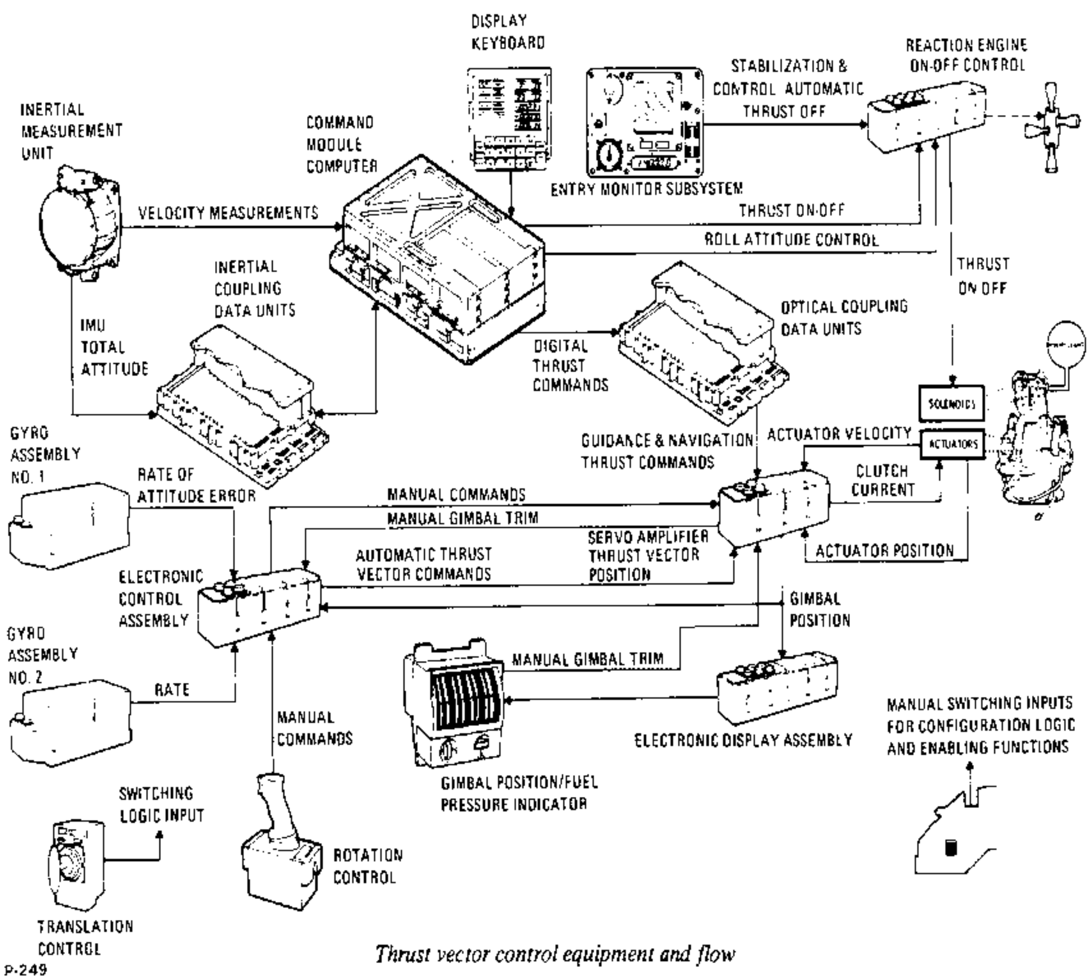


P-248 *Gimballing of service propulsion engine* aligned through the vehicle center of gravity and the error at both summing points is a constant-zero. The integrator input error is zero when gimbal position minus gimbal trim is equal to the negative of the attitude excursion sensed by gyro assembly No. 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.

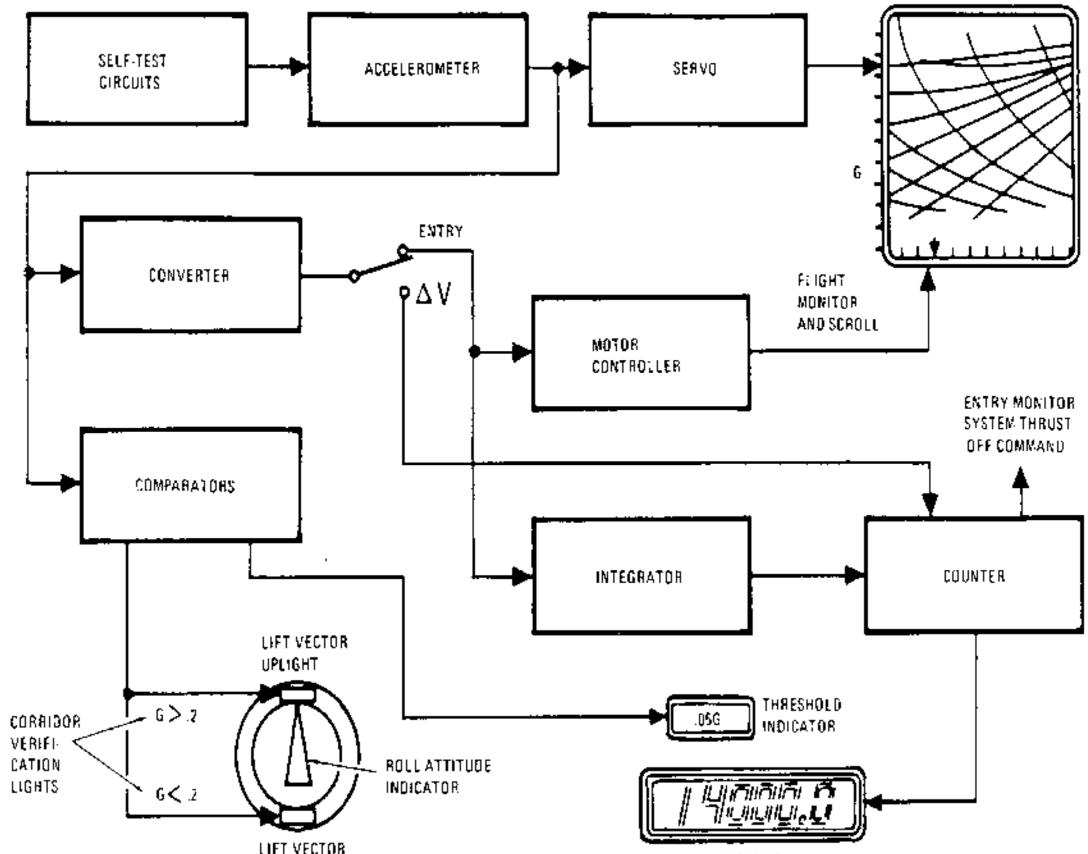
The roll attitude indicator displays the lift vector position throughout entry. During entry, stability axis roll attitude is supplied to the indicator by the gyro display coupler. There are no degree markings on the display, but the equivalent readout will be zero when the indicator points toward the 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

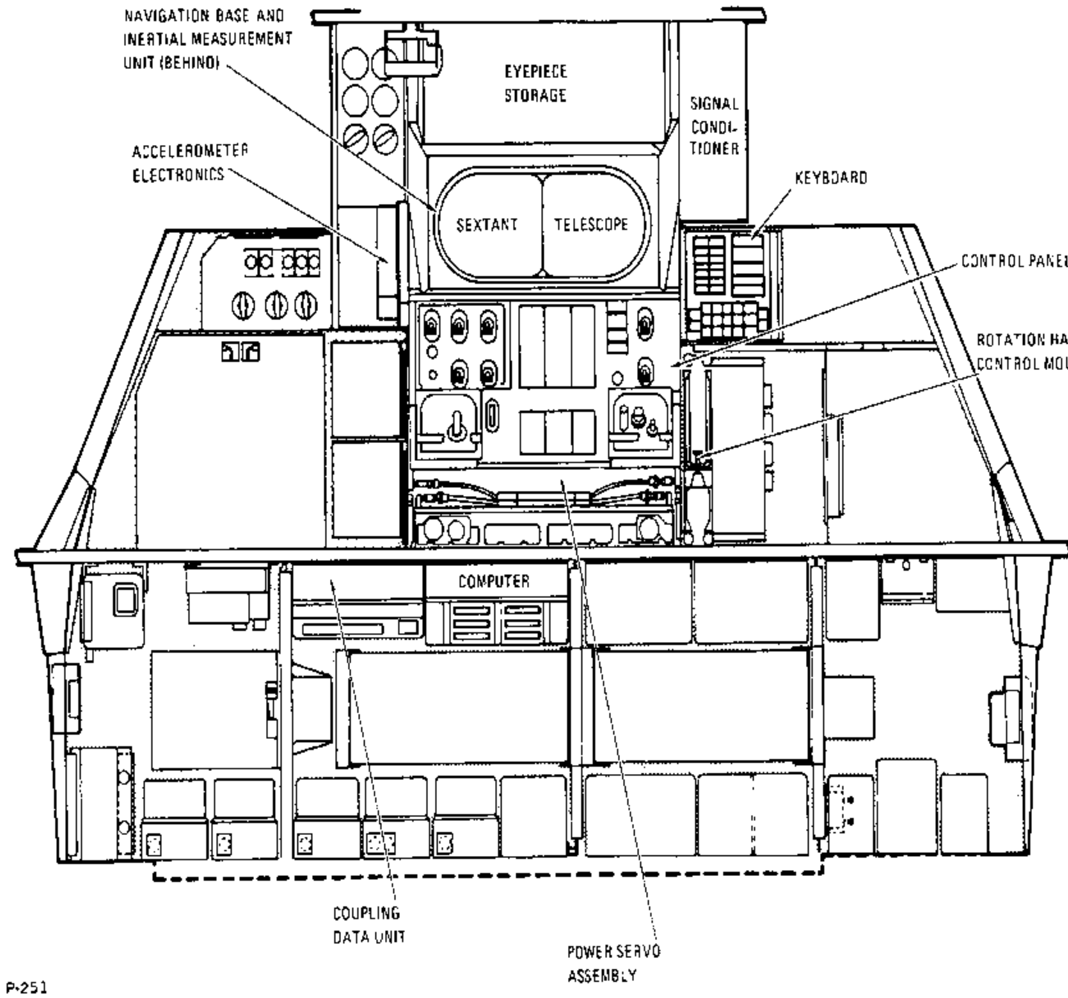
the velocity remaining to be gained or lost. The latter display can have a range of 14,000 to -1000 feet per second in tenths of a foot per second. The desired velocity change for all service propulsion engine thrusting maneuvers is set in the panel and the display will count up or down. During thrust 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.

GUIDANCE AND NAVIGATION



Location of guidance and navigation equipment in lower equipment bay

The guidance and navigation subsystem gives astronauts the ability to navigate the spacecraft on a required course through space. It can be operated either semi-automatically or manually and performs the basic functions of guidance and navigation -- similar to the navigation of an airplane or of a ship at sea. It can also be updated by the ground via telemetry.

While an airplane or ship at sea is concerned with two-dimensional navigation (it is always on or near the surface of the earth), Apollo is faced with exacting three-dimensional navigation as it speeds through deep space. Sightings from the spacecraft of stars and pre-determined landmarks on the earth and moon are used to establish the location and path of the spacecraft in space. The guidance and navigation system is used in conjunction with the stabilization and control, service propulsion, reaction control, electrical power, environmental control, and telecommunications subsystems.

Massachusetts Institute of Technology, an associate contractor to NASA's Manned Spacecraft Center, is responsible for development and design of the subsystem. AC Electronics Division of General Motors is responsible for its production, operation, and integration. The guidance computer is manufactured by the Raytheon Co. and the optics are produced by Kollsman Instrument Co.

There are three main elements in the guidance and navigation subsystem. The inertial guidance subsystem measures changes in the spacecraft position and velocity and assists in generating steering commands. The optical subsystem is used to take precision navigational sightings and provide the computer with measured angles between the stars and landmarks. The computer subsystem consists of a digital computer which takes data from the inertial guidance and optical subsystems and calculates spacecraft position, velocity, and steering commands.

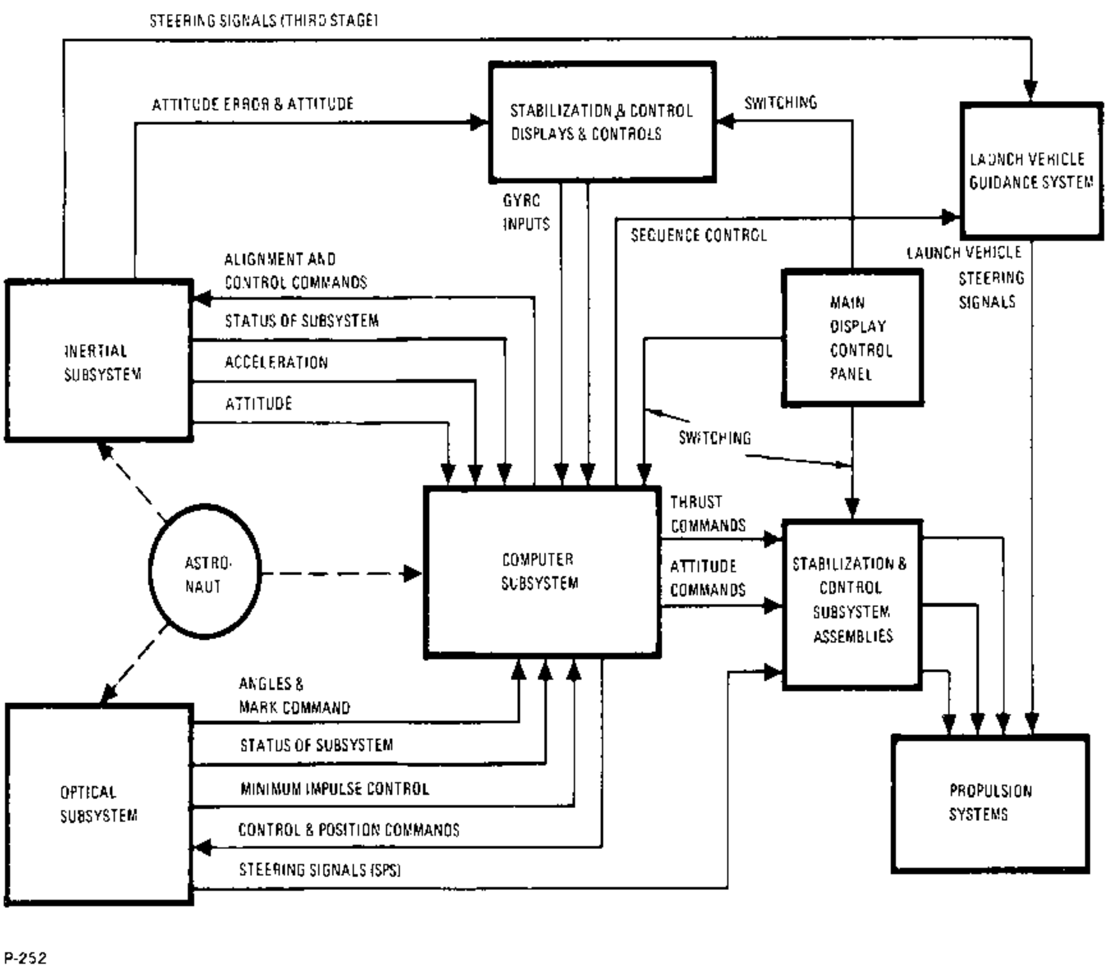
The components of the guidance and navigation subsystem, including the primary controls and displays, are located in the lower equipment bay of the command module. The main display console contains the switches and displays necessary for the astronauts to control the spacecraft while in their couches.

INERTIAL GUIDANCE SUBSYSTEM

This subsystem measures changes in the spacecraft position, assists in generating steering commands, and measures spacecraft velocity changes. Its instruments sense changes in velocity and attitude in a manner similar to the balance system in the human ear.

The main part of this subsystem is an inertial measurement unit mounted on a navigation base. The key part of this mechanism is a device called a stable platform, suspended on gimbals which allow it to incline freely regardless of spacecraft position. It is aligned to star direction references and retains this alignment regardless of the rotational movement of the spacecraft and thus provides a reference against which the movements of the spacecraft can be measured.

Mounted on this stable platform are the actual sensors: three accelerometers and three gyroscopes. The gyros are used to keep the stable platform fixed with respect to some point in space. The accelerometers sense any change in the speed of the vehicle -- forward, backward, up, down, sideways.



Data flow in guidance and navigation subsystem

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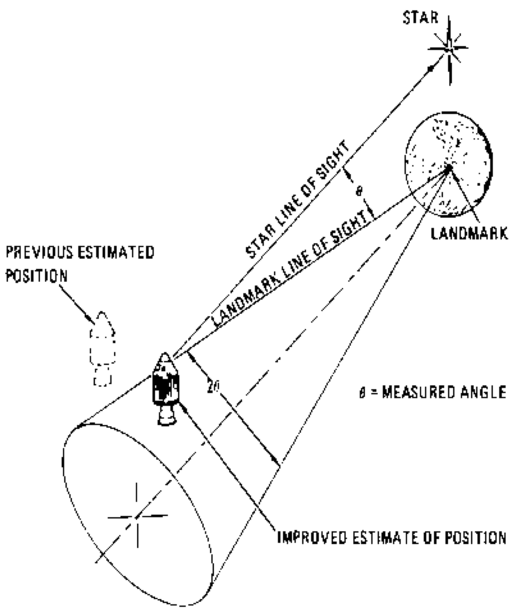


P-253 *Determination of midcourse position*

Any change from the pre-determined flight path or attitude generates electronic signals which result in firing of the reaction control subsystem engines (for attitude change only) or positioning of the service propulsion engine for flight path changes. The service propulsion engine is fired on signal from the computer or astronauts.

Devices known as resolvers, mounted on the gimbal axes, measure how far the spacecraft has rotated with respect to the stable platform. These measurements are transmitted to the guidance computer. While the gyros are used to maintain the spacecraft in a required attitude, the resolvers are used primarily to orient the spacecraft when firing the service propulsion engine.

The inertial guidance subsystem is controlled automatically by the guidance computer through crew selection of a computer program.

OPTICAL SUBSYSTEMS

This subsystem is used by the astronauts to take navigational sightings of the stars and earth or moon landmarks. It consists of a navigation base, a telescope, a sextant, and equipment to permit operation with the computer and inertial guidance subsystems.

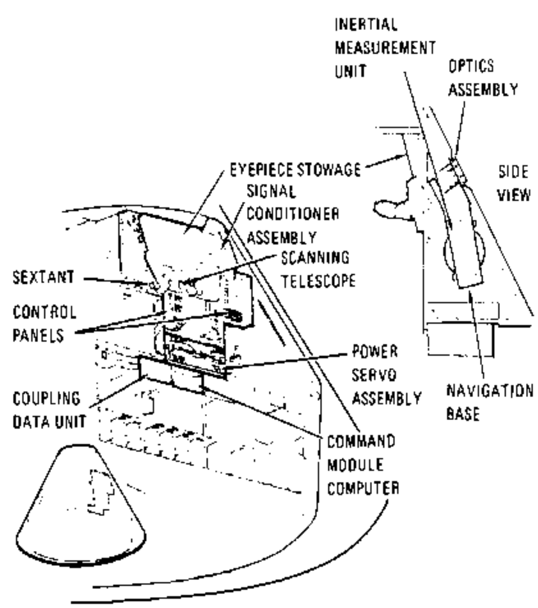
The telescope and sextant can be operated independently but generally are used together to obtain precision navigational sightings. The telescope has a 60-degree field of view with no magnification and is used to obtain coarse sightings of the stars or landmarks. Because of the telescope's limited field of view, controls are provided so the astronaut can maneuver the entire spacecraft to point the instrument in the general desired direction. The sextant is used to take the precision sightings and has a much smaller field of view (1.8 degrees) but has a magnification of 28.

These two instruments are used to measure the angle between two targets such as stars and earth or moon landmarks. The telescope and then the sextant are manipulated by the astronaut to line up a sighting and enter the reading into the computer.

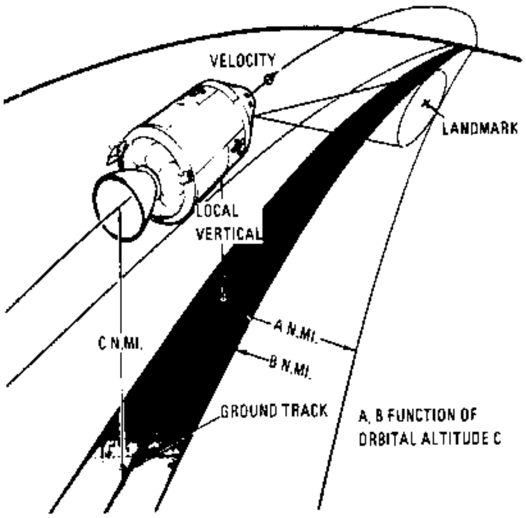
COMPUTER SUBSYSTEM

This subsystem consists of a digital computer which stores and uses signals from the inertial guidance subsystem and sightings by the optical subsystem with other data. This highly sophisticated computer uses this data to calculate necessary corrections to maintain course. The computer memory contains 38,912 words, or pieces of guidance data, and is divided into non-erasable and erasable sections. The non-erasable memory contains all the basic data necessary to achieve the round trip to the moon. The erasable memory is used by the astronauts when performing the various guidance and navigation computations.

The major functions of the computer are to



P-254 *Optical equipment installation*



P-255 *Orbital tracking of landmark*

calculate spacecraft position, velocity, and steering data; to calculate the signals for the main engine and attitude control jets necessary to keep the spacecraft on the required flight path and attitude; to position the stable platform in the inertial measurement unit as defined by precise optical measurements; to position the optical unit to celestial objects; to monitor the guidance and navigation subsystems for failure indications; and to supply information to the display and control panel.

The display of information, the results of computations, and the control of the computer by the astronauts is accomplished with two display and keyboard panels. One is located in the lower equipment bay and the other (a duplicate) is on the main display console. Lights on the panels also indicate the detection of malfunctions in guidance and navigation systems.

SEQUENCE OF OPERATIONS

Operation of the guidance and navigation subsystem in specific flight phases, control modes, and critical maneuvers during a typical lunar mission is described here briefly.

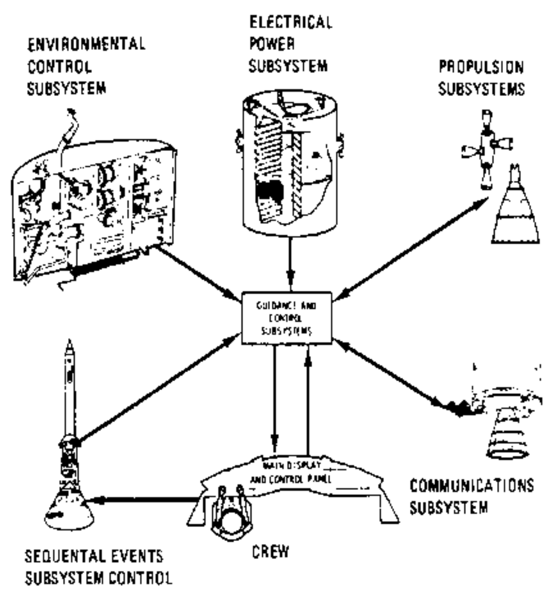
To conserve electrical power and fuel, the subsystem is activated only about 20 percent of the time, for specific sightings, alignments, and engine-firing maneuvers. Each time the guidance and navigation subsystem is activated the stable platform must be aligned with respect to a predetermined reference. Before launch the platform is aligned with respect to the earth and during flight it is aligned to the stars.

Launch and Translunar Injection – Guidance of the launch vehicle is monitored during the ascent from earth and firing of the third stage which sends the spacecraft on a trajectory to the moon. At launch, the inertial measurement unit is switched from an earth reference to a space reference frame. The crew then uses the guidance and navigation subsystem to monitor the spacecraft flight profile.

Earth and Lunar Orbit – During these phases, the crew checks the spacecraft position and orbital path with optical sightings. The crew takes these sightings by identifying and tracking landmarks with the optical instruments. The computer records optical sighting data, spacecraft attitude, and the time of the optical sighting.

Midcourse Navigation – This may be performed several times during the translunar and transearth mission phases. Starlandmark sightings are taken and the computer records the angles and time of sighting and determines spacecraft position and velocity.

Course Correction – The spacecraft's course must be corrected during both the translunar and transearth journeys. The computer, after calculating spacecraft position and velocity from navigation sightings, determines the need for a course



Guidance and control relationship to other subsystems

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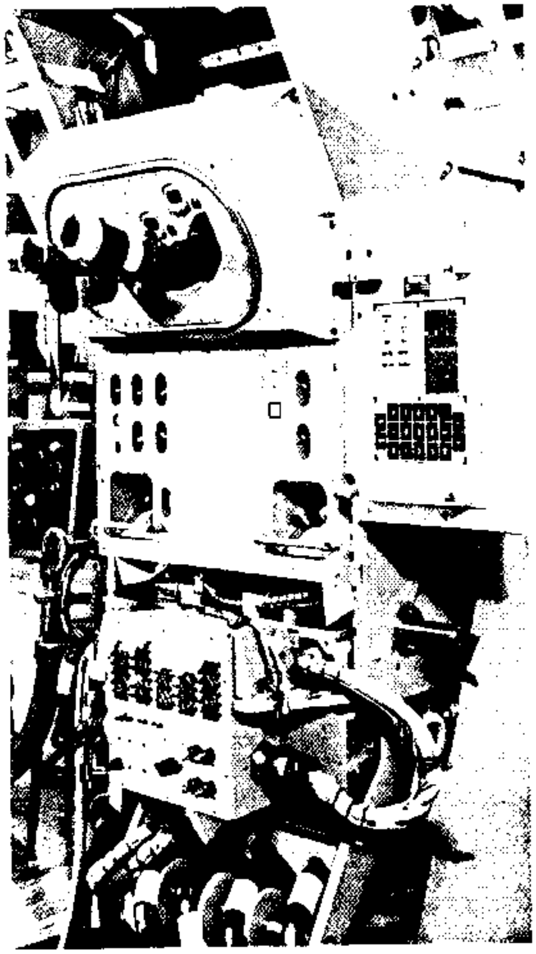

P-257 *Guidance and navigation equipment in test fixture simulating spacecraft installation*

correction by comparing the actual to the required trajectory. If a course correction is necessary, the computer calculates the time of firing and velocity change needed, repositions the spacecraft, and controls the initiation and duration of thrusting of the service propulsion engine.

Lunar Injection and Return to Earth – The guidance and navigation subsystem places the spacecraft into the attitude required for firing the service propulsion engine and controls the time of thrusting required.

Entry – The guidance and navigation subsystem controls the flight path of the command module during entry. The computer determines the proper trajectory and steers the command module by rolling it. This changes the lifting force acting on the command module and thereby varies its trajectory.

EQUIPMENT

The guidance and navigation system occupies a space 4 feet high, 2 feet deep, and about 3 feet across the top and 2-1/2 feet across the bottom. It is in the command module lower equipment bay.

Inertial Measuring Unit (AC Electronics Division of General Motors) – This ball-shaped unit has a diameter of 12.6 inches and weighs 42.5 pounds. It consists of three gimbals of which the inner gimbal is the stable member, with three gyroscopes and three accelerometers, all can-shaped and mounted onto the stabilized inner member structure, or platform. The gimbals are connected to each other by drive motors and angle resolvers. The unit is pressurized in dry air for good heat transfer. When in operation, the unit requires 217 watts at 28 volts dc. It maintains an inertially referenced coordinate system for spacecraft attitude control and measurement and maintains three accelerometers in this coordinate system for accurate measurement of spacecraft velocity changes.

Navigation Base (AC Electronics) – This 27-by-22-by-4.5-inch unit weighs 17.4 pounds and is made of riveted and bonded anodized preformed sheet aluminum alloys. It is filled with polyurethane foam. It is a rigid supporting structure for the inertial measuring unit and optical equipment.

Power and Servo Assembly (AC Electronics) – It is 2.75 inches high, 23.1 inches wide, and 22.6 inches deep and weighs 49.4 pounds. It contains 37 modules, most of the electronic modules for the inertial and optical subsystem servo loops and power supplies.

Coupling Data Unit (AC Electronics) – It is 20 by 11.3 by 5.5 inches and weighs 36.5 pounds. This sealed unit contains modular packaged solid-state electronics necessary to provide five separate coupling data unit channels for use with inner, middle, and outer inertial measuring unit gimbal resolvers and the shaft and trunnion resolvers of the optical subsystem. It also contains failure detection circuitry for inertial subsystem and optical subsystem. It provides analog-to-digital conversion of the inertial measuring unit gimbal angles and optical subsystem trunnion and shaft angles, and digital-to-analog conversion of computer-derived data to control inertial subsystem and optical subsystem modes of operation using discretes issued by the computer. It also controls service propulsion system engine gimbaling, attitude error display, and, as a backup mode, it controls the attitude of the launch's third stage.

Inertial Reference Integrating Gyros (AC Electronics) – Three gyros are mounted mutually perpendicular to each other on the stable platform of the inertial measuring unit. Each is 2-1/2 inches in diameter. They sense displacement of the inertial measuring unit's stable platform and generate error signals. They are pressurized in an atmosphere of helium to provide good heat transfer.

Accelerometers (AC Electronics) – There are three pulse-integrating pendulous accelerometers mounted perpendicular to each other on the stable member of the inertial measuring unit. Each is can-shaped with a 1.6-inch diameter. They measure velocity changes along all axes of the three-axis inertial measuring unit.

Sextant (Kollsman Instrument Co., Syosset, N.Y.) – This is a highly accurate dual line-of-sight electro-optical instrument with 28X magnification and 1.8 degrees field of view. It can sight two celestial targets simultaneously and measure the angle between them with 10 arc seconds accuracy to determine the position of the spacecraft. It is mounted on the navigation base. One line of sight is fixed along the shaft axis normal to the local conical surface of the spacecraft. It is positioned by changes in the spacecraft attitude. The other line of sight has two degrees of rotational freedom about the shaft axis (plus or minus 270 degrees) and trunnion axis (minus 5 to plus 50 degrees). The variation about the trunnion axis is represented by movement of an indexing mirror.

This tiny pendulum is heart of an accelerometer, a key component in guidance and navigation subsystem

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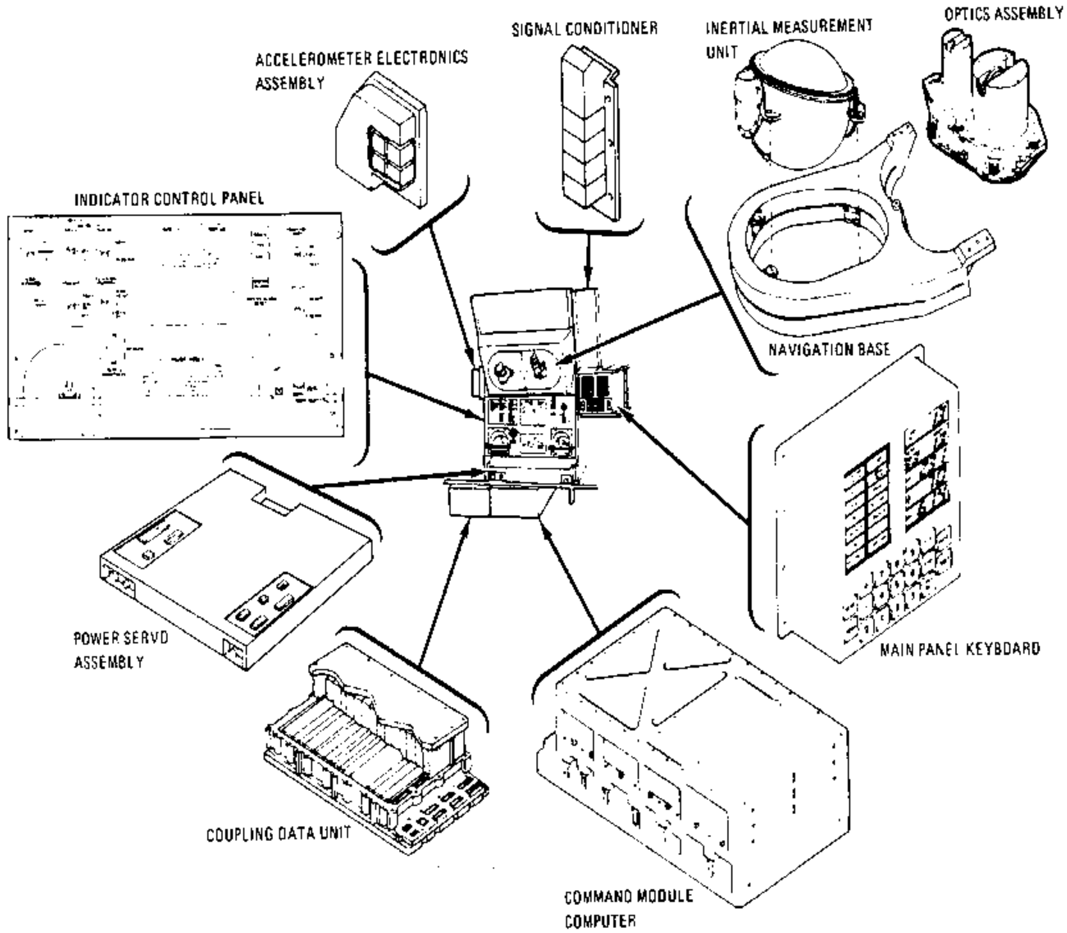
Scanning Telescope (Kollsman) – It is a single line-of-sight, refracting-type, 1X magnification instrument with 60-degree instantaneous field of view. It is similar to theodolite (surveyor's instrument used to measure horizontal and vertical angles). It has a double-dove prism mounted in the head assembly. The operating power for the telescope, sextant, and associated electronic equipment is 94.5 watts at 28 volts dc. The telescope has two axes of rotational freedom, which are normally slaved to the sextant axis. The wide field of view is used for general celestial viewing and recognition of target bodies. It is also used to track landmark points during earth and lunar orbits.

Computer (Raytheon Co.) – This is 24 by 12.5 by 6 inches and weighs 70.1 pounds with six memory modules. It consumes 70 watts of power at 28 volts dc during normal operation. It is a digital computer with fixed and erasable memory. The erasable memory has a capacity of 2048 words; fixed memory has a capacity of 36,864 words. The fixed memory contains programs, routines, constants, star and landmark coordinates, and other pertinent data. The computer solves guidance and navigation problems, provides control information to optical and inertial subsystems as well as other spacecraft systems, provides pertinent information to astronauts and the ground on request, provides means by which astronauts or ground control can directly communicate with the primary guidance navigation control system, provides direct on-off control for reaction control jets and service propulsion engines, and monitors its own operation and other primary guidance navigation control system operations.

Display Keyboard (Raytheon Co.) – This 8-by-8-by-7-inch panel weighs 17.5 pounds. It is made up of a keyboard, power supply, a decoder relay matrix, status and caution circuits, and displays. It is a 21-digit character display and a 16-button keyboard through which crewmen can



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Major guidance and navigation subsystem equipment

communicate in a coded numerical language. Crewman inserts data and commands the computer by punching numbers on the keyboard. They are then displayed to him in electroluminescent counter-type readout windows. The computer communicates with the crewman by displaying numbers in the same windows. When the computer requests the crewman to take some action, numbers flash to attract attention.

Signal Conditioner Assembly – This 3-by 5.7-by-14.3-inch unit weighs 5.8 pounds. It contains encapsulated electronic circuitry to condition primary guidance navigation control system signals so that they are acceptable to the spacecraft telemetry system.

DETAILED DESCRIPTION

INERTIAL SUBSYSTEM

The inertial subsystem provides a space stabilized inertial reference from which velocity and attitude changes can be sensed. It is composed of the inertial measurement unit, the navigation base, parts of the power and servo assembly, parts of the control and display panels, and parts of the coupling data unit.

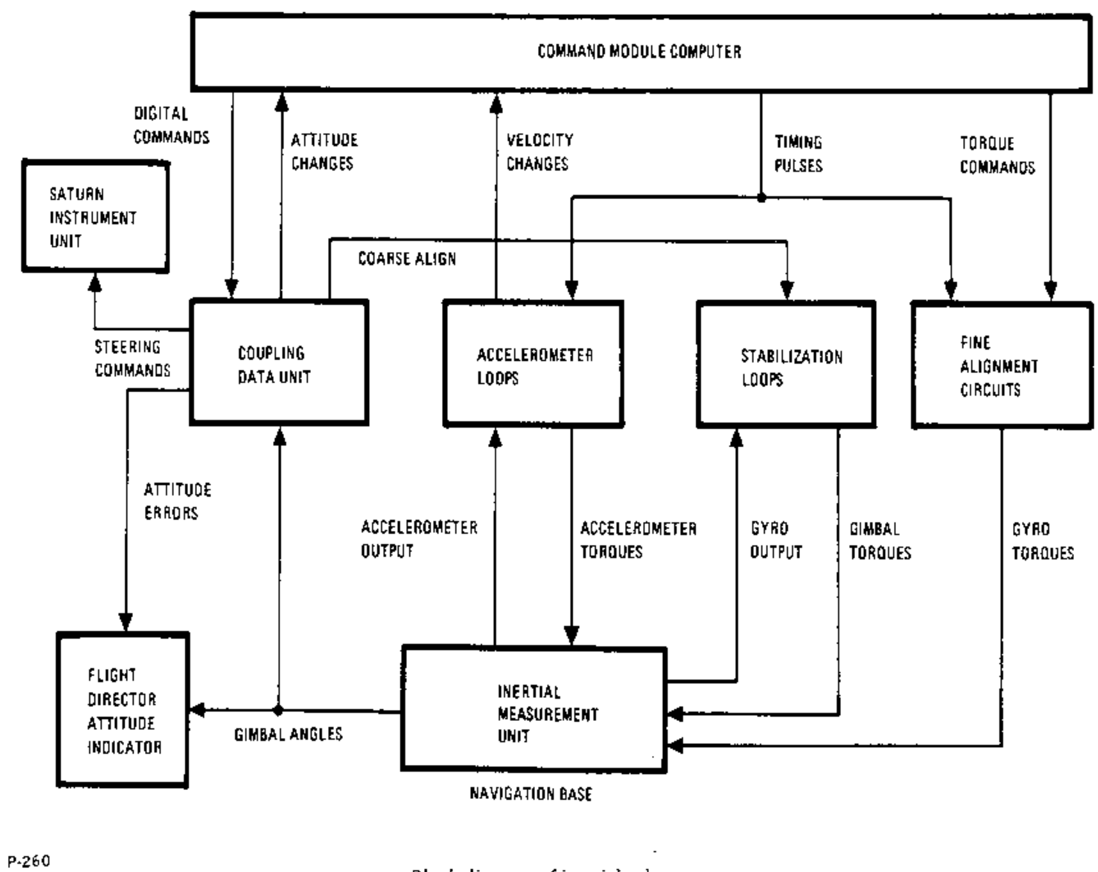
The navigation base is the rigid supporting structure on which the inertial measurement unit and optical instruments are mounted. It is manufactured and installed to close tolerances to provide accurate alignment of the equipment mounted on it. It also provides shock mounting for the inertial measurement unit and optics.

The inertial measurement unit is the main unit of the inertial subsystem. It is a three-degree-of-freedom stabilized platform assembly, containing three inertial reference integrating gyros, and three pulsed-integrating pendulous accelerometers. The stable member is machined from a solid block of beryllium with holes bored for mounting the accelerometers and gyros.

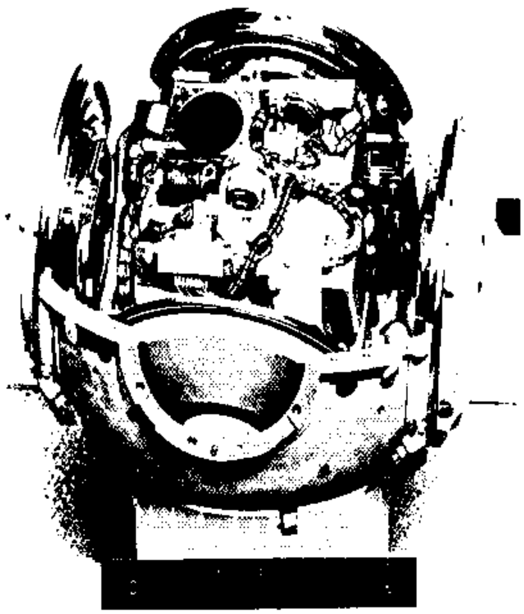
The stable platform attitude is maintained by the gyros, stabilization loop electronics, and gimbal torque motors. Any angular displacement of the stable platform is sensed by the gyros which generate error signals. These signals are resolved and amplified at the inertial measurement unit and applied to stabilization loop electronics. The resultant signal is conditioned and applied to the gimbal torque motors, which restores the desired attitude.

The stable platform provides a space-referenced mount for the three accelerometers, which sense velocity changes. The accelerometers are mounted orthogonally (each at right angles to the other two) to sense the velocity changes along all three axes. Any translational force experienced by the spacecraft causes an acceleration or deceleration which is sensed by one or more accelerometers. Each generates an output signal proportional to the magnitude and direction of velocity change. This signal, in the form of a pulse train, is sent to the computer which uses it to update the velocity information.

Temperature is controlled by a thermostatic system that maintains the gyro and accelerometer temperatures within their required limits during inertial measurement unit standby and operating modes. Heat is applied by end-mount heaters on the inertial components, stable member heaters, and a temperature control anticipatory heater. Heat is removed by convection, conduction, and radiation. The natural convection used during inertial measurement unit standby modes is changed to blower-controlled, forced convection during operating modes. Inertial measurement unit internal pressure is normally between 3.5 and 15 psia, enabling the required forced convection. To aid in removing heat, a water-glycol solution passes through coolant



Block diagram of inertial subsystem



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Cutaway of inertial measurement unit

passages in the outer case of the inertial measurement unit. Therefore, heat flow is from the stable member to the case and coolant. The temperature control system consists of the temperature control circuit, the blower control circuit, and the temperature alarm circuit.

The coupling data unit is an all-electronic device used to convert and transfer signals among the inertial, computer, and optical subsystems and between the computer and various controls and displays. Each of the five channels has two sections: one acting as an analog-to-digital converter and the other as a digital-to-analog converter. These five channels are: one each for the inner, middle, and outer inertial measurement unit gimbals, and one each for the shaft and trunnion optical axes. Each channel converts inertial or optical gimbal angles from a resolver analog form to a digital form, supplies the computer with this information, and converts digital signals from the computer to either 800-cycle per second or direct current analog signals. The coupling data unit also controls the modes of the inertial and optical subsystems through logical manipulations with the computer.

The power and servo assembly provides a central collection point for most of the guidance and navigation subsystem power supplies, amplifiers, and other electronic components. It is located in the lower equipment bay directly beneath the inertial measurement unit. It consists of 42 modules mounted to a header assembly. Connectors and harnessing are integral to the construction of the header assembly, and guidance and navigation harness branches are brought out from the power servo assembly header. A thin cover plate is mounted on the assembly to provide a hermetic seal for the interior. During flight this permits pressurization of the assembly to remain at 15 psi. Connectors are available for measuring signals at various system test points.

OPTICAL SUBSYSTEMS

The optical subsystem is used to take precise optical sightings of celestial bodies and landmarks. These sightings are used to align the inertial measurement unit and to determine the position of the spacecraft. The system includes the navigational base, two of the five coupling and data units, parts of the power and servo assembly, controls and displays, and the optics, which include the scanning telescope and the sextant.

The optics consist of the telescope and the sextant mounted in two protruding tubular sections of the optical base assembly. The scanning telescope and sextant line of sight may be offset depending on the mode of operation.

The sextant is a highly accurate optical instrument capable of measuring the included angle between two targets and the direction of any single target with respect to the navigation base. Angular sighting between two targets is made through a fixed beam splitter and a movable mirror located in the sextant head. The sextant lens provides 1.8-degree true field-of-view with 28X magnification. The movable mirror is capable of sighting a target to 57 degrees line of sight from the shaft axis. This is reduced to approximately 45 degrees when installed in the spacecraft, however, because of interference from structure. The mechanical accuracy of the trunnion axis is twice that of the line-of-sight requirement due to mirror reflection which doubles any angular displacement in trunnion axis.

The scanning telescope is similar to a theodolite in its ability to measure elevation and azimuth angles of a single target accurately using an established reference. The lenses provide 60-degree true