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THE OBJECTIVES AND DESIGN OF THE INERTIAL REFERENCE UNIT FOR THE LUNAR ORBITER

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Summary

The Lunar Orbiter, a photographic satellite, is one of the family of lunar reconnaissance spacecraft planned for the unmanned exploration of the moon prior to Project Apollo. The spacecraft requires a guidance capability for trajectory changes, for example, midcourse corrections and transfer into lunar orbit, and for camera pointing during photography. The inertial sensing function for guidance is provided by the Inertial Reference Unit.

This paper describes in brief the Lunar Orbiter objectives, mission and spacecraft configuration, and discusses in detail its Inertial Reference Unit including the system configuration, accuracy, and physical design characteristics.

Introduction

The Lunar Orbiter satellite will utilize a film-type camera for obtaining high-resolution photographs of fairly large areas on the moon's surface. The three-axis stabilized spacecraft employs a guidance system using celestial bodies for attitude reference during the extended periods of cruise, and inertial measurements during short periods of maneuvering.

The guidance system uses the Inertial Reference Unit (IRU) for the inertial measurement function. The system controls and measures spacecraft maneuvers required for trajectory changes and photography, and maintains a fixed spatial attitude with respect to the reference celestial bodies while they are occulted by the moon. The IRU measures attitude error and rate and during spacecraft velocity changes, measures linear acceleration.

The basic concept and design of the IRU was initiated and accomplished by the Lunar Orbiter prime contractor, The Boeing Company, and its subcontractor, the Sperry Gyroscope Company. The project is managed by the NASA, Langley Research Center.

The Lunar Orbiter Mission Objectives

The primary Lunar Orbiter mission objective is to obtain photographic information concerning the lunar surface in support of Project Apollo. A secondary objective is to obtain general scientific measurements for the extension of man's knowledge of the moon and its environments both for Apollo and general scientific use. The photography will lead to landmark mapping, geological interpretation of lunar surface characteristics, and eventual selection of Surveyor and Apollo landing sites.[1] Other instrumentation will provide data relative to the lunar gravitational field, high-energy particle flux, and micrometeoroid puncture rates.

The Lunar Orbiter Mission Description

The first Lunar Orbiter is scheduled for launch in 1966, utilizing an Atlas-Agena launch vehicle. The launch techniques and procedures are similar to those used for the Ranger spacecraft (fig. 1). During the launch phase, tracking and guidance will be provided by the Eastern Test Range (ETR) using the guidance capabilities of the launch vehicle. Also as for Ranger, subsequent tracking will be accomplished by the NASA Deep Space Instrumentation Facility (DSIF). Centralized control of the mission will be provided by the NASA Space Flight Operations Facility (SFOF) in Pasadena, California. Launch windows and lunar orbital parameters for the Lunar Orbiter are constrained within any given month by unique target illumination and coverage requirements.

After insertion into the cislunar trajectory (fig. 2), the spacecraft will separate from the launch vehicle, its solar panels and two antennas will unfold, and it will start its search for the sun. After sun lock-on occurs, the spacecraft begins a controlled roll around the sun line mapping the stars that fall within the field of view and sensitivity limits of its star tracker. The mapping data are telemetered to earth via the omnidirectional antenna. A correlation is accomplished and the star Canopus identified. The spacecraft is then commanded to roll to and lock on Canopus. Thus the three axes are stabilized, the solar panels are pointed toward the sun, and celestial base lines are established for referencing the directional antenna and attitude maneuvers throughout the remainder of the mission.

During the cislunar phase midcourse corrections may be made. Upon arrival at the moon, spacecraft velocity changes will establish an initial lunar orbit, with a perilune of 200 Km and an apolune of 1800 Km for a typical mission. After a period of tracking to determine the orbital parameters, a final adjustment is made to position the perilune 46 Km over the target area. Prior to the arrival over the target on each orbital pass, the spacecraft maneuvers to point the onboard camera for picture taking. After film [redacted]o its celestial base lines, the [redacted] and is ready to be read out and transmitted to earth. This is accom[redacted]t system and the telecommunications [redacted] high-gain antenna. All spacecraft events are initiated by the flight programer, utilizing a data storage and timing system, or by real time commands transmitted from the earth.

Spacecraft Configuration

The 850-pound Lunar Orbiter spacecraft (fig. 3) is $5\frac{1}{2}$ feet tall and 5 feet in diameter in its launch configuration. When deployed, the span across the antenna booms and the solar panels are $18\frac{1}{2}$ feet and about 12 feet, respectively. The Inertial Reference Unit is attached to the equipment mounting deck, as are all the major spacecraft components with the exception of the rocket engine and tanks.

Functions of the IRU

During the Lunar Orbiter's attitude maneuver phase, the IRU provides the vehicle's attitude rate to the flight programer, which integrates this information so that the proper commands can be given to the attitude jets to position the vehicle. Positioning is accomplished first along one axis and then along the others. After completing these operations, the vehicle returns to its celestial base lines, performing the maneuver in reverse. During all maneuvers, the maximum error introduced by the inertial reference unit is 1.8 deg/hr at the Lunar Orbiter's typical maneuver rate of 0.5 deg/sec. Thus the unit can introduce only a 0.1-percent attitude maneuver error into the system.

The Lunar Orbiter must maintain tight attitude control during the vehicle thrusting and photography phases. Attitude errors up to a maximum angle of 3 degrees are measured by the IRU. This range is more than sufficient because the vehicle itself is controlled to much smaller angles. The IRU introduces noise of approximately 1 mv. Therefore the attitude control system can apply the attitude error information to achieve good fuel economy during the tight limit cycling required while vehicle thrusting or photography missions are being conducted. During this phase, maximum gyro drift is 0.3 deg/hr.

During the Lunar Orbiter's vehicle thrusting maneuvers, the IRU measures acceleration. The acceleration information is provided to the flight programer, where it is integrated and compared with the desired velocity increment to determine engine cutoff time.

During the coasting phase, routine attitude control is required. The attitude rate output of the IRU is provided to determine the vehicle's limit cycle rate - the rate at which the vehicle oscillates about its celestial baselines. The maximum error in this rate is 0.7 deg/hr at near-zero rates involved, and noise is approximately 0.2 percent of full-scale voltage. As a result, the unit's rate signal is used by the attitude control system to effect fuel economies during this phase.

Whenever either celestial reference is occulted by the moon, the IRU provides a sun or Canopus memory. Usually, the sun is in shadow one-fourth of the time. Attitude error is provided by the IRU for limit cycling during this operating mode. Here, the major accuracy consideration is the gyro's maximum drift rate of 0.3 deg/hr, which permits rapid reacquisition of the sun or Canopus when that body emerges from the shadow.

The IRU functions in the spacecraft are depicted in figure 4. The mode commands are five binary electrical signals consisting of: power mode (all power on or off); accelerometer mode (accelerometer power on or off); and gyro mode (attitude rate or the attitude error). The gyro mode commands can be applied to the three axes independently. The attitude rate and attitude error outputs are d-c analog signals, but the accelerometer output consists of digital pulses from which the total velocity increment is subsequently derived by counting the pulses. The reference unit furnishes ten signals to the telemetry subsystem: attitude rate and attitude error (each axis), gyro spin motor current (each axis), and gyro operating temperature (average for the three axes).

IRU Design Criteria

The IRU weighs approximately 13 pounds, and measures 7 inches by 10 inches by 7 inches (this includes all inertial components, self-contained electronics, and the electrical interference and noise-filtering equipment required in the Lunar Orbiter). The maximum primary power requirement over the full mission profile is 18.5 w not including the temperature-control functions. Power demand of these functions, which depends on the temperature range of the equipment mounting deck to which the reference unit is attached (85° F to 35° F), varies from almost zero to 0.5 w. Therefore, over the full mission profile, total primary power reaches a maximum of only 28 w - scarcely sufficient power to read by.

Among the design criteria for the unit, reliability received top consideration. Parts and components were selected that had been proved in earlier space or missile applications. The SYG-1000 gyro and the pulsed integrating pendulum (16 PIP), for example, had been used in DOD space and missile programs where they served many thousands of hours. Also, the 16 PIP is man rated by NASA (rated for manned spaceflight) for the Apollo program in both the Command Module and the Lunar Excursion Module. In addition to the gyros and accelerometers, most of the electronic parts have been qualified through previous applications; the minority of parts for which no such experience existed were individually qualified specifically for the IRU. Materials and processes, too, were individually screened for reliability in the design.

After reliability, the next criterion considered was design for low weight and power. As a result of this criterion, the beryllium frame, extremely compact electronics packaging, and a number of low-power circuits were designed.

Finally, the various elements of the space environment required particular attention. Design for survival of launch shock and vibration conditions was achieved through the ruggedness of the inertial components and through the strength of the beryllium frame. The design of the thermal-vacuum path had to be carefully selected to minimize increases in temperature of electronic components (a reliability consideration) and to minimize the electrical heater power required to control the temperature of inertial components.

IRU System Description

The complete IRU is depicted functionally in figure 5, and photographs of the component are shown in figure 6. The major assemblies are its beryllium frame, three SYG-1000 gyros, one 16 PIP, and six electronic modules.

Inertial Components

Three SYG-1000 single-degree-of-freedom, floated, rate integrating gyros are used in the IRU. Each consists of a beryllium float assembly containing a symmetrical spin rate with integral bearings. Its dynasyn pickoff is an air-core differential transformer having no moving iron and, hence, is free from reaction torques and radial forces. The dynasyn torquer is an air-core permanent-magnet device operating on the d'Arsonval principle. The float assembly is neutrally buoyant in a flotation fluid whose non-freezing characteristics provide low-temperature exposure capabilities. The gyro has a pivot-jewel output-axis suspension and a bellows at each end to prevent end-to-end fluid flow; it provides both low-temperature and high-temperature volume compensation.

The 16 PIP, together with pulse torquing electronics, was selected partially because of its low power requirements. The unit has a lightweight, high-strength beryllium case and consists of three major functional subassemblies: The pendulus float or sensitive element, the signal pickoff ducosyn, and the torque generator ducosyn. A ducosyn is a dual-purpose electromagnetic device consisting of two concentric 8-pole stators. The inner stator provides the magnetic suspension while the outer stator serves as the torque motor or signal pickoff. The pendulum float requires no physical connections whatever.

Electronics Subsystem

The electronic circuits of the IRU provide gyro control, accelerometer control, power conditioning, temperature control, and telemetry conditioning.

The gyro electronics are depicted in figure 7 with the associated mode switching and signal conditioning. In the attitude rate mode, the torque-to-balance loop having an overdamped loop characteristic is utilized with a lower frequency of 120 rad/sec. In the attitude error mode, fast response time of 1 msec is achieved in the open-loop gyro. The critical attitude rate output amplifier is noteworthy for its extremely low drift of 0.005 percent of full-scale voltage.

The accelerometer loop is shown in figure 8 with its associated mode switch and signal conditioner. Here, the sense detector is actuated by the 200-cps clock, which is phase coherent with the 4.8-kc pickoff excitation. As a result, direction (or sense) of the pendulum's offset from null is determined during each clock period. This same signal, in the form of excitation of the proper output (plus or minus), switches a precision, fixed-level current source to the torquer in a direction that will restore pendulum null. In effect, pendulum offset is maintained near zero with the precision torque pulses balancing the pendulum acceleration. These torque pulses, therefore, constitute a measure of the acceleration experienced by the pendulum, one pulse for one clock period representing a fixed velocity increment during the period (in this case 0.1 ft/sec). The performance features of the accelerometer loop are that the accuracy depends only (1) on calibration of the 16 PIP to a fixed current level and (2) on the ability of the electronic circuits to maintain the fixed current and the clock timing. The normalization circuit shown in the figure provides for magnetic centering of the floats and for tuning of the torquer coils. The few components controlling the precision current level and requiring temperature control are mounted directly on the 16 PIP to benefit from its controlled temperature.

The power conditioning regulation technique, devised for low power dissipation with regulation over wide input voltage variations (21 to 32v), is shown in figure 9. Basically, the flux in the final transformer that supplies all direct current and gyro-motor power is regulated against a Zener reference voltage. If this flux is low, an error signal increases the on time of a switch in series with primary power. The increased time that the switch is on results in an increase in the d-c component of primary voltage, thereby raising controlled d-c voltage. This, in turn, increases transformer flux until a proper value is reached, as evidenced by the derived feedback d-c voltage that balances the reference d-c voltage. The power conditioner consists of the regulation circuits, rectifiers and filters for d-c outputs from the inverter, a capacitor-phase, splitting circuit for three-phase gyro-motor outputs from the inverter, a clock, and a precision 4.8-kc pickoff and temperature sensor excitation supply. The clock operates from a 19.2-kc crystal oscillator with digital countdown to derive its outputs.

In the temperature-control circuits pulse-width modulation handles the significant heater power levels at low amplifier dissipation. For added power efficiency, the loads for both driver and output stages are the heater windings themselves.

Amplification and buffering are provided for the attitude rate and attitude error outputs on each axis, gyro-motor current on each axis, and for the sum of the three gyro temperatures.

Transistors are used in the IRU. The digital functions, including the clock countdown and accelerometer logic, contain integrated flat-pack circuits. Switching is accomplished with solid-state elements.

The low power demand of the reference unit results from the high-efficiency switching techniques described for power conditioner regulation and temperature control. The difference between the switching technique of the power conditioner and the conventional series regulator is that in the power conditioner a reactive (dissipationless) voltage drop in the filter following the power switch is substituted for a resistive (and dissipative) voltage drop. The transistor switches here and in the heater amplifiers achieve very good efficiencies. Other examples of power-saving measures in the design are the choices of switching frequencies and the accomplishment of phase-sensitive demodulation and mode switching at low signal level in the gyro electronics.

The electronic packaging was selected for low weight and good thermal conductivity using proved techniques. The aluminum frame of each module (inset fig. 6) supports compact honeycombs that house the electronic parts normally in cordwood formation, as shown in the photo at the far left. Electronic parts are integrated into the honeycombs with a special encapsulant of high thermal conductivity. The soldering is in accordance with NASA procedures for every detail of the joint - including the arrangement, materials, processes, tools, and environment. All together, the modules occupy 243 cubic inches and house 1,600 electronic parts, thus achieving average package density of 6.5 parts per cubic inch, and peak densities that are much higher. The total weight for the six modules is 6.20 pounds. The weight of the electronic parts alone is 3.91 pounds. Thus the payload-to-nonpayload weight ratio is 1.71, which compares favorably with the best of the present-day discrete component packaging technology.

Thermal Design

Conductive heat flow was selected instead of radiation because it resulted in lower temperature rise in the electronic components. At the hard vacuum experienced (10^{-14} mm Hg), conduction is confined to solid material that has been engineered into the design.

The heat sink for the IRU is the spacecraft's equipment mounting deck to which the unit is bolted. In the hard vacuum, all junctions must be very carefully controlled. Although the IRU mounting surface can achieve a 16-microinch finish and 0.0005-inch flatness, the larger light-weight spacecraft frame cannot realistically do this. Therefore, suitable conductivity between the IRU's mounting surface and the spacecraft's frame must be obtained by using indium foil and a large number of mounting bolts (8) at the junction.

The frame design is a compromise between weight and temperature drop. Accordingly, a nominal 1/8-inch thickness is employed with thickening provided in critical spots.

The thermal path from a typical electronic part is through the encapsulant to the honeycomb, then through another thickness of epoxy adhesive that fastens the honeycomb to the aluminum module chassis, and finally through the machined-surface junction between this chassis and the beryllium frame. The adhesive and honeycomb materials were selected for their thermal conductivity. Also, the aluminum module chassis was formed by dip brazing, a method selected for thermal conductivity, and aluminum cross pieces were used liberally in the frame. The results for the electronics can be summarized by the following typical temperature rise, corresponding to an 85° F cold plate:

Path element	Maximum temperature rise, °F
Spacecraft junction . . .	3
Main frame	8
Module junction	3
Module frame	8
Honeycomb	9
Total	31

Control of thermal paths is also required to minimize the power demanded by the temperature controller for the inertial components. The factors that enter into the power problem are inherent power in the controlled element (e.g., gyro-motor power), value of the control temperature, thermal conductance within the controlled element, and all thermal conductances to the heat sink. Tolerances must be allowed for all these factors. As a result the controller's maximum power demand is increased. The IRU tolerance study indicates that reasonable allowances can be budgeted under the conditions existing in manufacture and use, and the resulting increment in heater power demand over the theoretical minimum can be held to about 1 w.

Reliability

In the IRU, the gyros are by far the most complex single item, and, as such they have the dominant effect on system reliability. The mean-time-between-failure (MTBF) assessment made for the SYG-1000 gyro is 25,000 hours based on confidence gained in previous applications. Infant mortality of gyros is eliminated by prescribed burn-in times of at least 200 hours.

Since high-reliability electronic parts, such as those developed for the Minuteman program, are used along with the 16 PIP, the system failure rate is increased only slightly above that of the gyros. In the 30-day Lunar Orbiter mission the gyros, associated electronics, and power-supply electronics are required continuously, while the 16 PIP and its electronics are required for orbital corrections during the first 10 days only. The probability of no failure for the three gyros is 0.915; for all electronics 0.985; and for the accelerometer 0.997. The compound probability of mission success is 0.897.

These probabilities are based on no failures, even noncatastrophic ones such as telemetry items. The high reliabilities for the electronic circuits reflect Minuteman-program experience of failure rates for the standard electronic components in the range of 0.004 to 0.05 failures per million hours. For the approximately 1,600 electronic parts, this leads to the reliability of 0.985. White-room conditions, material control, quality control, source control, and lot control and traceability of specific workmen, machines, and times are specified during manufacture. Screening of parts is also conducted on a 100-percent basic including X-ray, performance test, environmental test, and burn-in.

For these reliability assessments, a severity factor of 140 times was assumed for the launch environment. Negligible failure rate was assumed for solder joints because of stringent NASA soldering procedures. Also, all electronic parts are operated under 60 percent of rating, with 90 percent of the parts under 20 percent of rating.

Conclusion

The IRU design represents a significant advance in the miniaturization of high-performance inertial equipment of its type, without departing from the fundamental precept of using only components and processes that have been proved in missile and space vehicles.

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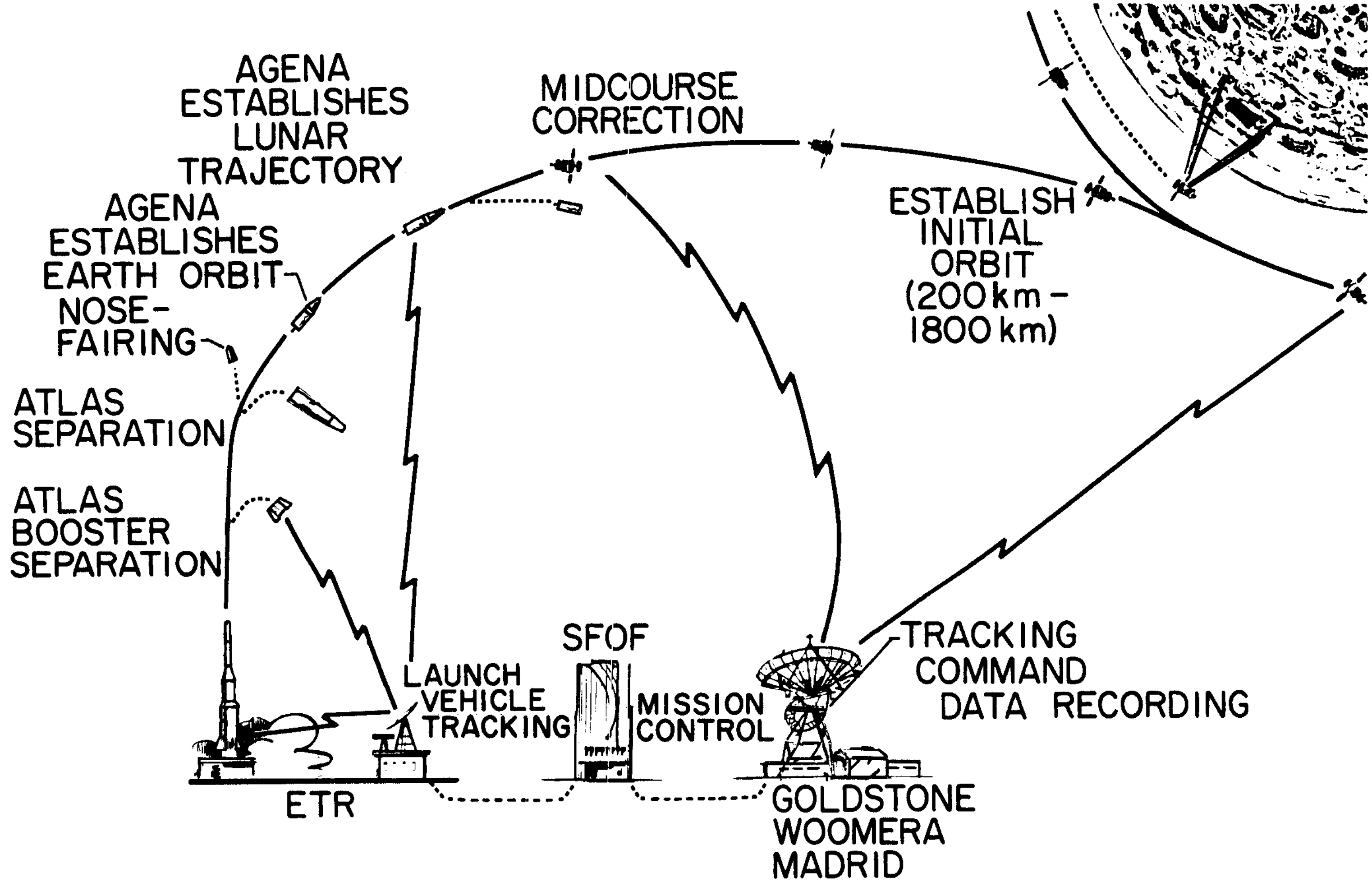
2. Taback, I., and Brummer, E. A., "The Lunar Orbiter." Paper presented at the AIAA, Unmanned Spacecraft Meeting, Los Angeles, California, March 1-3, 1965.

3. Whitcomb, E. W., "Inertial Components." Sperry Engineering Review, vol. 17, no. 1 (Spring 1964), pp. 6-12.



NASA

Figure 1.- Mission operations.

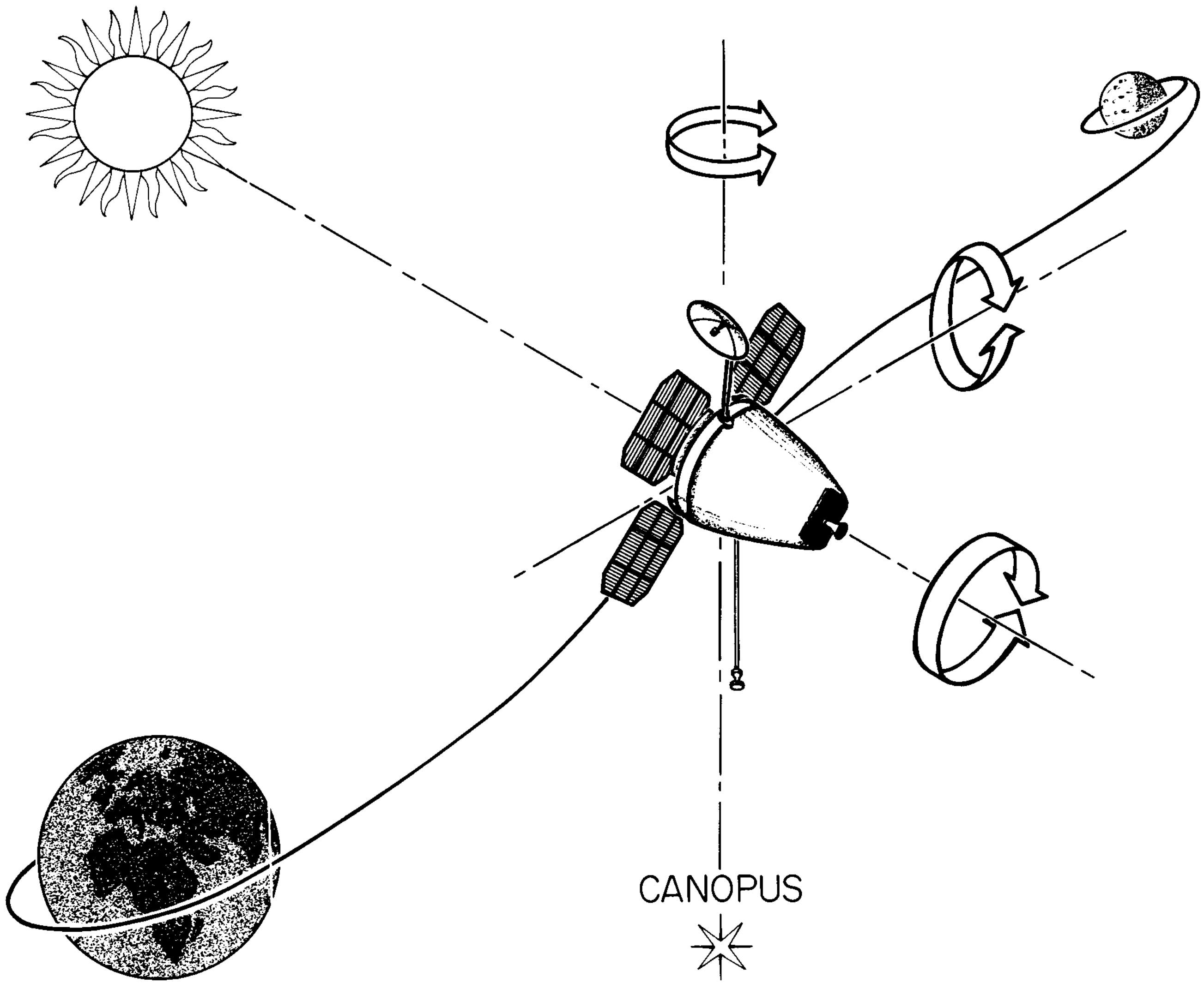


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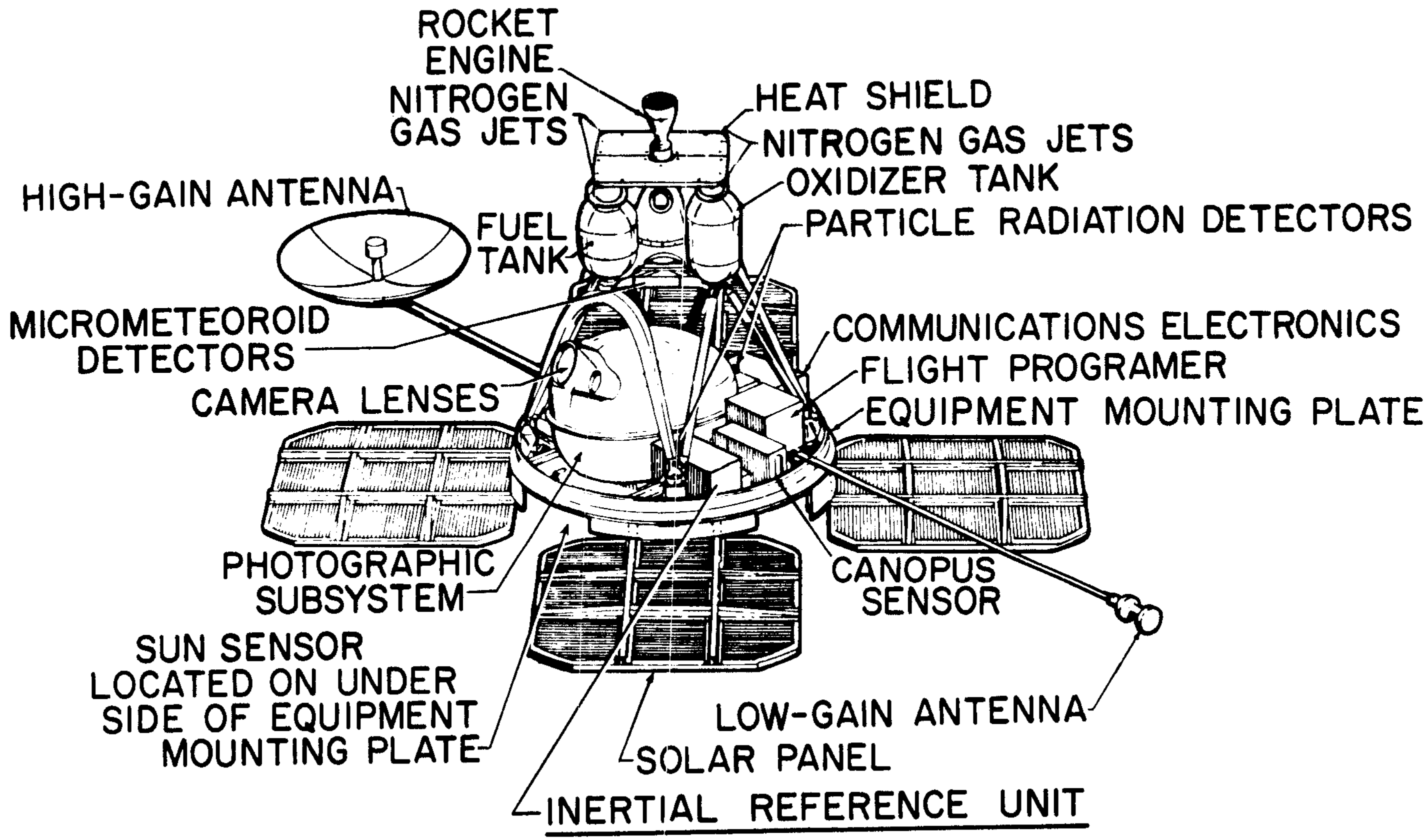
Figure 2.- Celestial base lines.



NASA

Figure 3.- Spacecraft configuration.

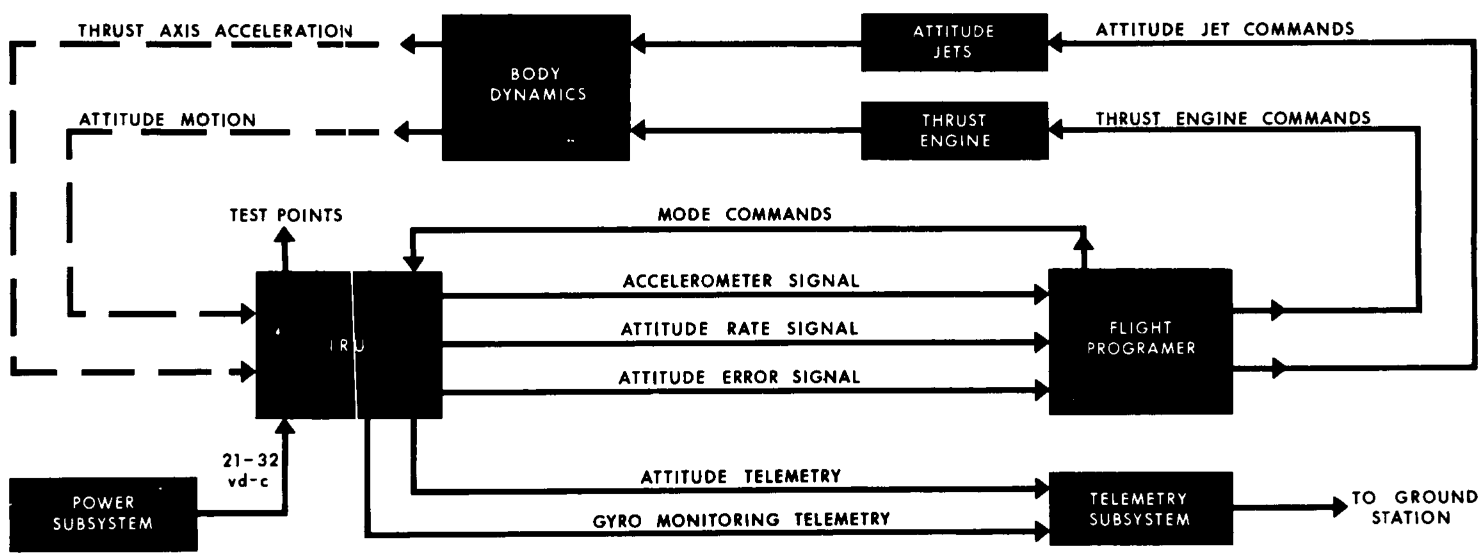


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Figure 4.- Attitude control functions.

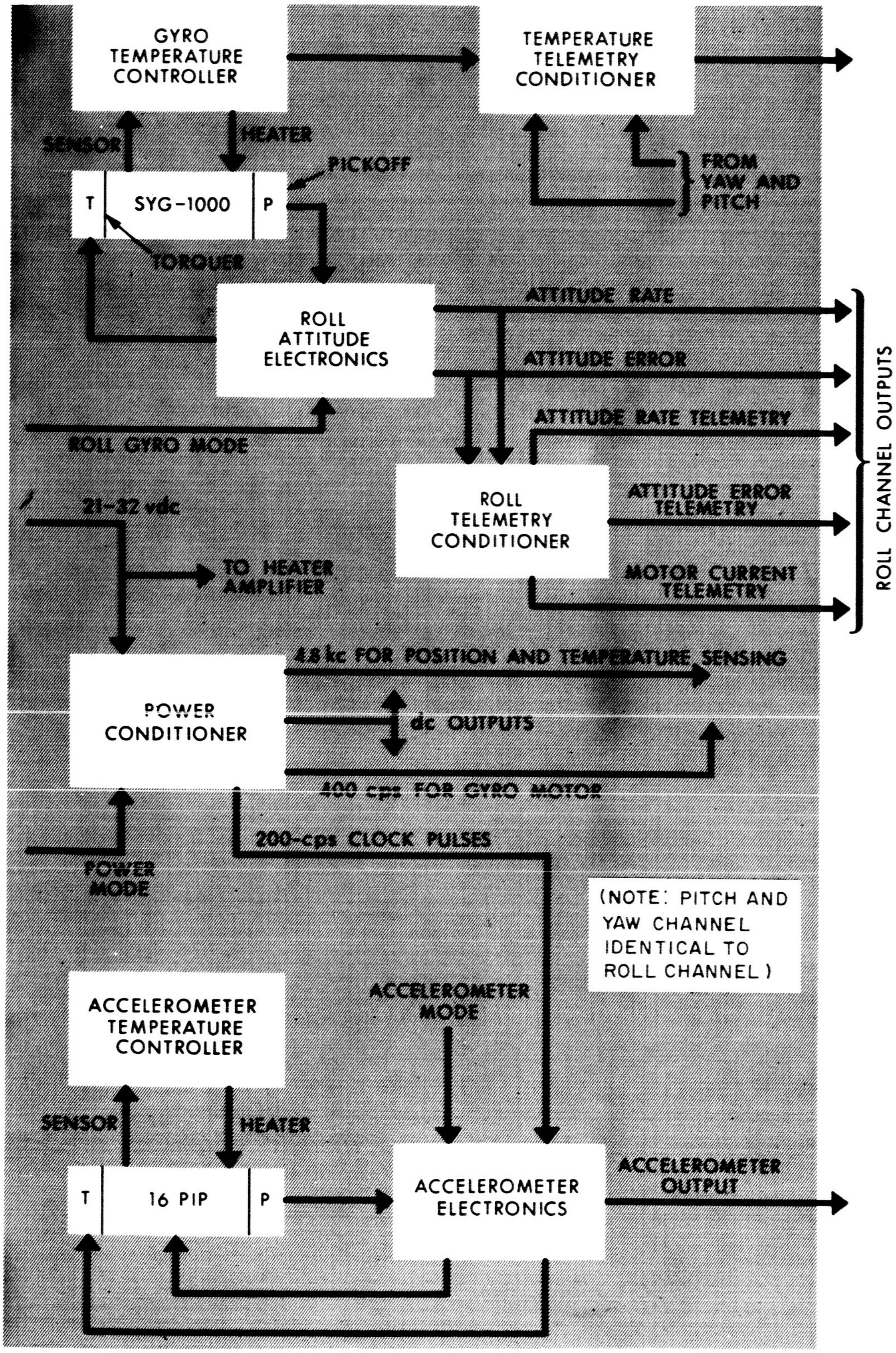

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Figure 5.- IRU functional diagram.

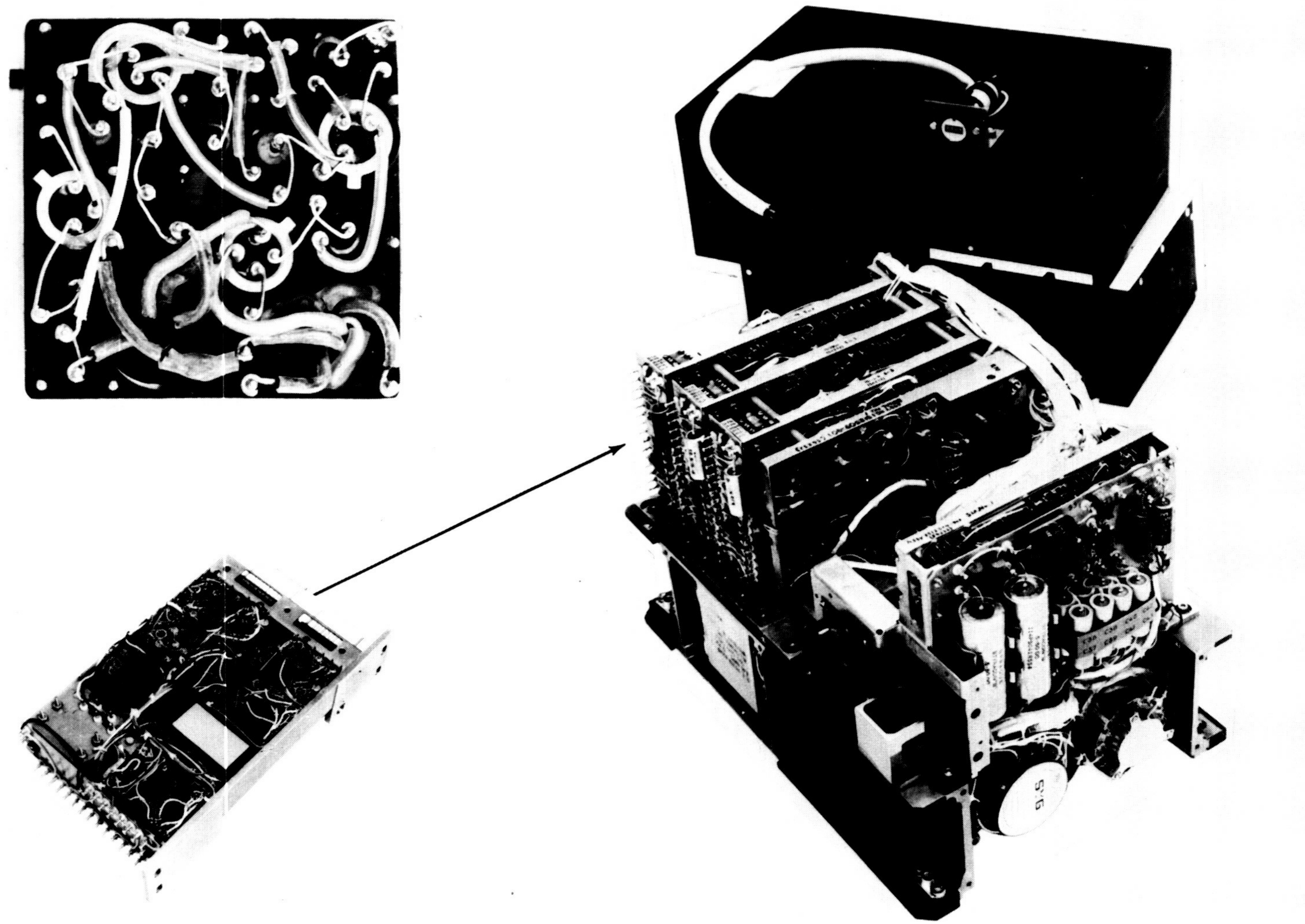

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Figure 6.- IRU assembly.

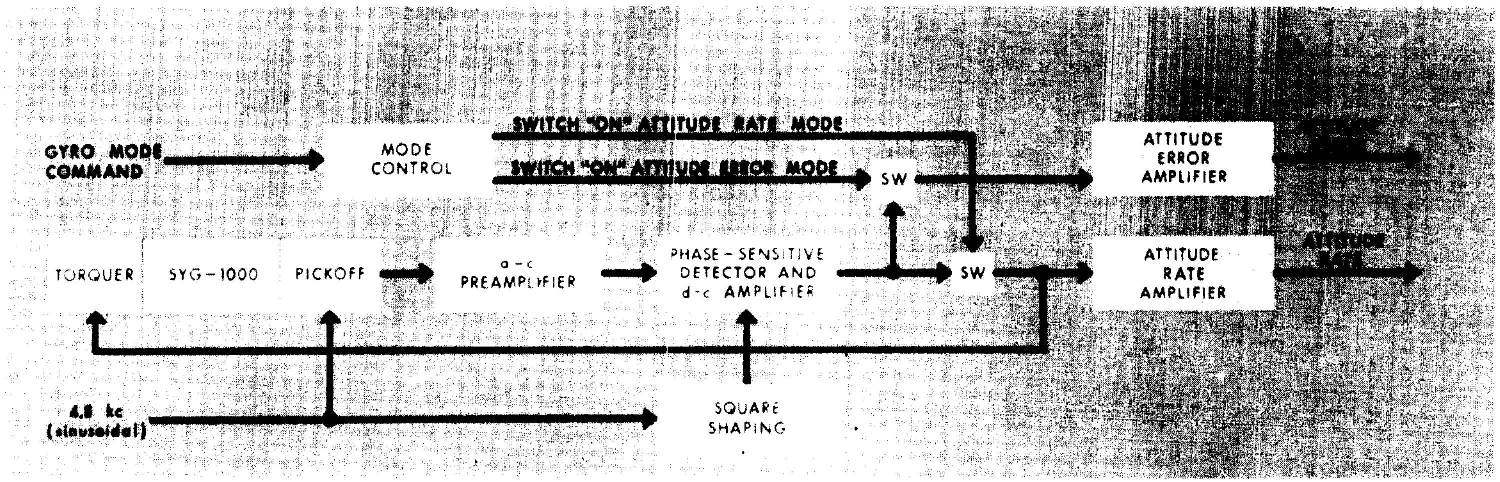

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Figure 7.- Gyro electronics block diagram.

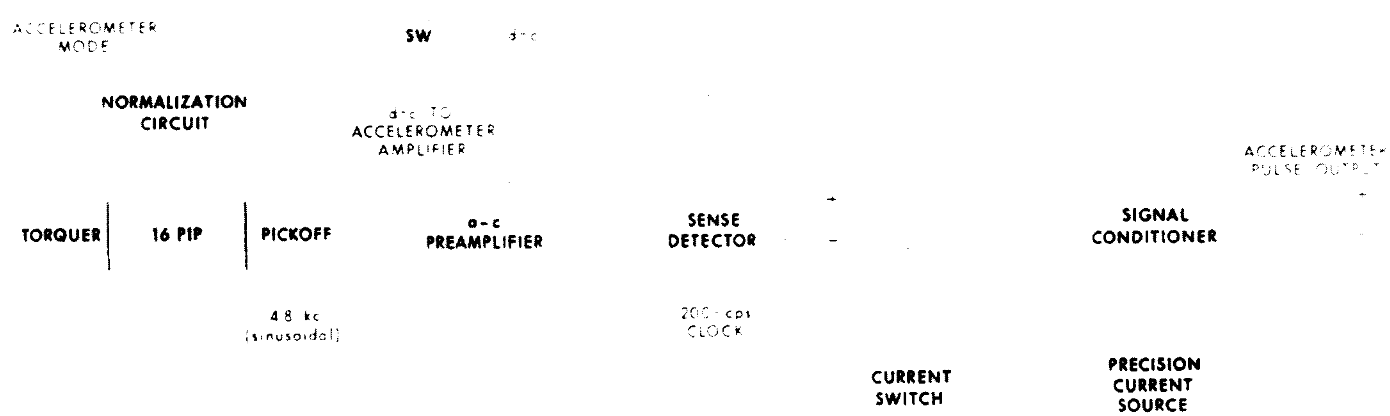

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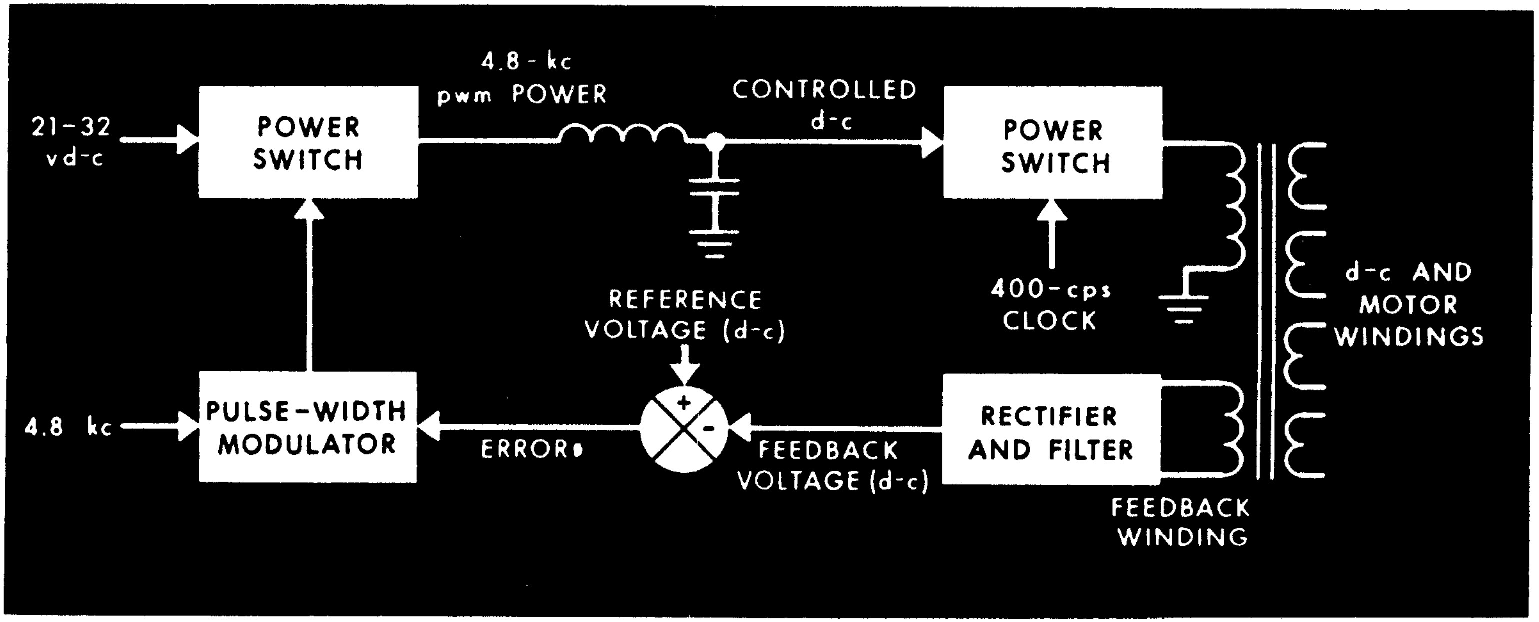
Figure 8.- Accelerometer electronics block diagram.



NASA

Figure 9.- Power conditioner electronics block diagram.

NASA-Langley, 1965