# *Pioneer* Mission Support

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*This article describes the design profile of the Pioneer F and G missions. The characteristics of these flights that interface with Deep Space Network tracking and data acquisition support are depicted. A delineation of the mission description and a summary of spacecraft systems and subsystems are given.*

## I. Introduction

Reference 1 contains a summary of the tracking and data system planning activities for the *Pioneer F* and *G* missions. This article begins the description of *Pioneer F* and *G* missions, with special emphasis on characteristics that interface with the tracking and data acquisition functions. DSN planning efforts during the design phase of the spacecraft hardware were centered around the assurance of a functional compatibility between the spacecraft design and the configuration of the DSN Mark III system.

In this article, *Pioneer F* and *G* mission profiles are presented. Spacecraft systems are briefly described and the following subsystems are outlined: electrical power supply, electrical power conditioning, structure, thermal control, propulsion, and attitude control. Future articles will contain: (1) a description of the telecommunications and antenna subsystems; (2) a summary of launch and space flight trajectories, giving tracking and data acquisition constraints; (3) the configuration of the six DSN systems; and (4) the planning activities related to navigational requirements.

## II. *Pioneer F* and *G* Mission Profile

The mission objectives of the *Pioneer F* and *G* spacecraft are to conduct, during the 1972–73 Jovian opportunities, exploratory investigation beyond the orbit of Mars of the interplanetary medium, the nature of the asteroid belt, and the environmental and atmospheric characteristics of the planet Jupiter.

As the earth and Jupiter orbit around the sun, their relative position permits a spacecraft to be launched every 13 months into a Jupiter-bound trajectory with a minimum of launch energy. During optimum conditions, injection velocities of approximately 14 km/s will suffice; a favorable launch period can cover several weeks. During the remainder of the 13-month interval, the velocity requirements will rise to prohibitive values. Among the available launch vehicle configurations, the *Atlas/Centaur*/TE 364 vehicle was chosen for *Pioneer F* and *G* missions. The launch energy generated by this vehicle will make possible the launch of *Pioneer F* during late February or early March 1972, and *Pioneer G* during April 1973. These missions will make possible the initial direct exploration of vast regions of the solar system from 1 to 6 AU from the sun, including the vicinity of the planet Jupiter. The asteroid belt is located in this region between 2 and 3.6 AU from the sun. *Pioneer F* will be the first NASA spacecraft to explore the outer planet environment beyond the orbit of Mars. It will take between 600 and 800 days to travel from the earth to the vicinity of Jupiter, using trajectories compatible with the favorable launch opportunities.

The spacecraft are equipped with field-and-particles and optical-type instruments. Nearly all the time during travel between the earth and Jupiter will be spent in the interplanetary solar-wind environment. The influences of the earth's magnetosphere will cease several hours after launch. The spacecraft will fly through the high-density Jovian magnetosphere to explore the trapped radiation particles of the solar system's largest planet. The general relationship of a typical *Pioneer F* trajectory to the sun, earth asteroid belt, and Jupiter is depicted in Fig. 1.

A simplified mission profile of *Pioneer F* is shown in Fig. 2. The telecommunications range at Jupiter encounter will be approximately 5½ AU and, after the encounter of Jupiter, *Pioneer F* will move toward the outer planets and eventually leave the solar system. In contrast, the *Pioneer G* trajectory profile is such that this spacecraft, after the Jupiter encounter, will stay within the solar system as a solar orbiter.

The *Pioneer F* and *G* missions will require continuous tracking and data acquisition coverage by the 26-m antenna network from liftoff to Jupiter encounter plus 6 months. The flight project also expects that during critical mission events the 64-m antenna stations will be continuously available to accomplish mission objectives. In addition, the project plans to obtain from the 64-m antenna stations at least one horizon-to-horizon tracking pass per week from liftoff to encounter plus 6 months.

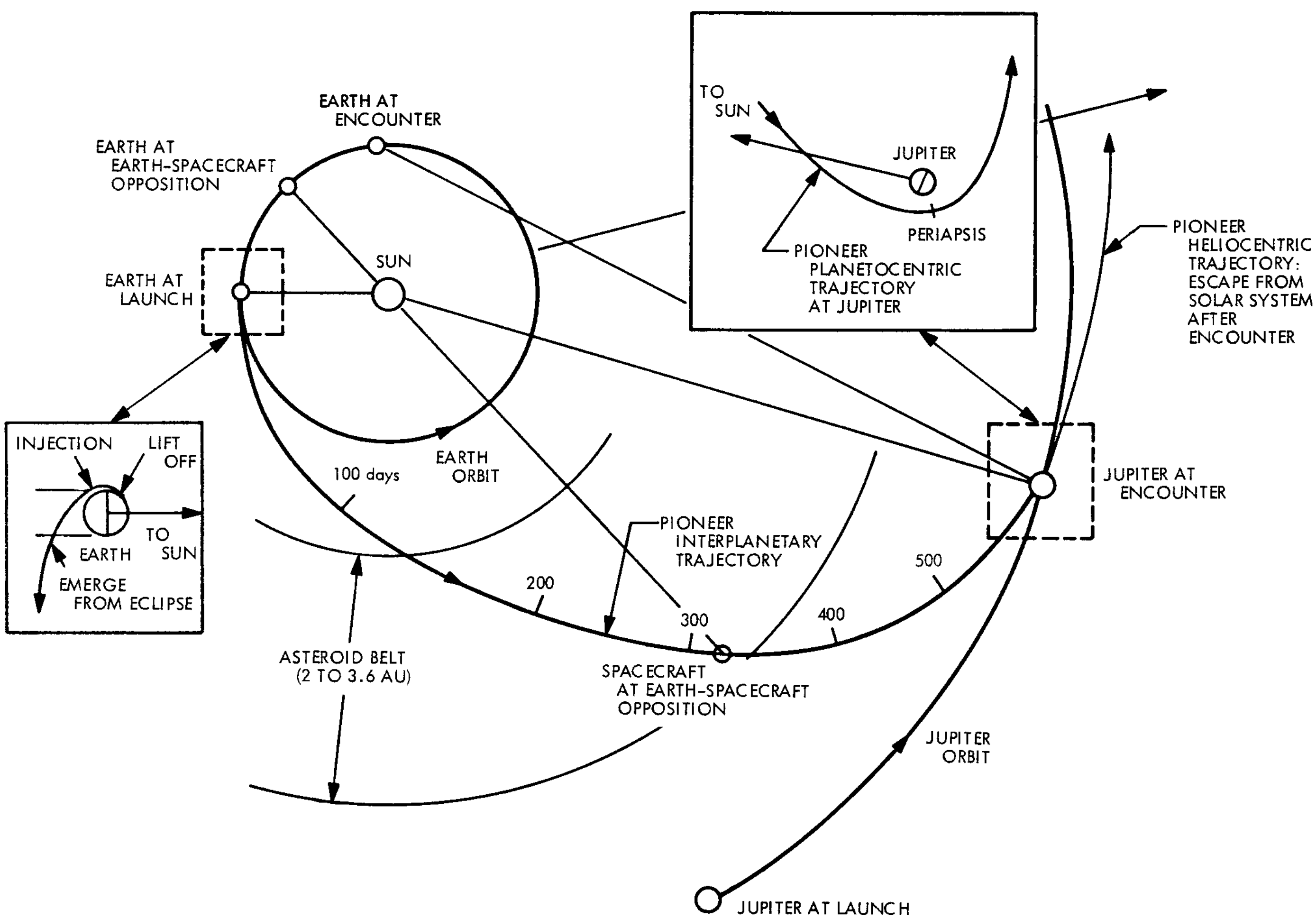



**Fig. 1. Ecliptic projection of a typical *Pioneer F* trajectory**

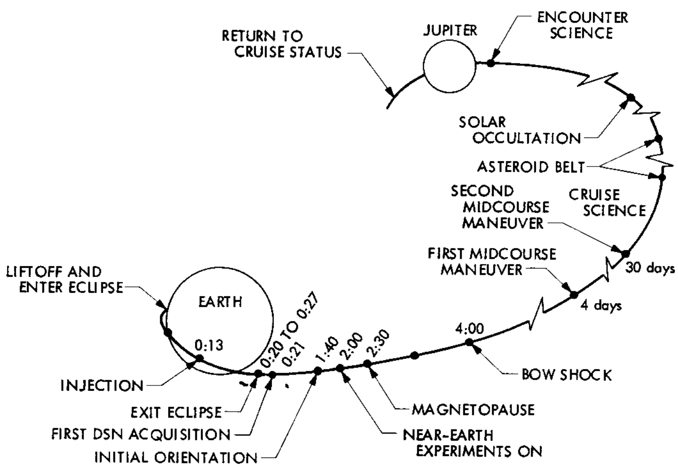



**Fig. 2. *Pioneer F* mission profile**

After the Jovian encounter plus 6 months, the tracking and data system should furnish, as a minimum, daily coverage for both missions until the limit of the receipt of downlink data signal.

It will be a continuous challenge for the DSN to acquire and collect the data stream transmitted by the Jovian missions. Variations of the solar activities will naturally have a continuous impact on the measurements made by the field-and-particles instruments relating to the solar wind and the interplanetary magnetic field.

*Pioneer F* and *G* spacecraft will reach the asteroid belt approximately 6 months after launch and will stay in this belt for more than 6 months. Some astronomers hypothesize that the asteroid belt might be composed of the remnant of a planet which was circling the sun at 2.8 AU. Seventy-five thousand bodies of the asteroid belt are larger than magnitude 20; the largest is Ceres with a diameter of 768 km. The exploration of the asteroid belt can lead to the understanding of the origin and evolution of the solar system; therefore, the data collected by the DSN can have a high scientific significance.

To give a better idea of the challenge of a Jupiter flyby, a few of the important characteristics of the largest solar planet are given: Jupiter's average distance from the sun is 5 AU, which is equivalent to approximately $75 \times 10^7$ km. Its mass is equivalent to 318 times the earth's mass and comprises 70% of all planetary mass. Jupiter's diameter (approximately 71,387 km) is 11 times the earth's diameter. It is remarkable that Jupiter's density is only approximately one-fourth of the earth's density and its fastest rotation period is 9 h and 50 min. In other words, one Jupiter day is almost 10 h long. Jupiter has 12 satellites, 4 of them having a magnitude equivalent to the earth's moon. The four Jupiter moons are called IO, Europa, Ganymede, and Callisto. Jupiter has many varying striped surface features. The red spot is about three times the size of earth. It is assumed that Jupiter's upper atmosphere is composed of hydrogen, helium, amonnia, and methane. It is an intense radio noise source at decametric frequencies. Fortunately, this noise drops considerably toward the S-band frequency range. The typical cold sky system noise temperature of the Goldstone 64-m antenna station is 25°K. If one moves this antenna to the direction of Jupiter, the system noise temperature will increase to approximately 30°K. This increase of 5°K can cause a telemetry signal-to-noise degradation of around 0.5 dB. The dense magnetosphere of this large planet is caused by an extremely high magnetic field, which can be as high as $5 \times 10^{-4}$ T (5 gauss).

To place the spacecraft on a trajectory to the desired target point in the vicinity of Jupiter, one or more midcourse corrections have to be made to compensate for launch vehicle injection velocity vector errors. Flight project plans to perform the first maneuver within the first ten days and the second, if necessary, about 30 days after launch.

The medium and the 2.75-m-diameter high-gain antenna of the *Pioneer F* and *G* spacecraft have to continuously point toward the earth during the entire flight to assure effective data return (Fig. 3). Therefore, precession maneuvers will be conducted throughout the mission as required to maintain an earth-pointing cruise attitude. These maneuvers, homing into the uplink signal transmitted by the DSN toward the spacecraft, will take place only once every 2 or 3 days during the early phase of each mission, and will decrease in frequency to once in 1 or 2 weeks by the time of Jupiter encounter.

Approximately 315 days after launch, the relative position of the spacecraft and earth will place the spacecraft in a superior conjunction configuration versus the sun and the earth. Because the spacecraft is somewhat out of the ecliptic plane, the spacecraft/earth line will not intercept the sun but will come within a few solar radii. In this configuration, the radio beam will be intercepted through the high-density part of the solar corona and will be influenced significantly by the plasma. Because of the closeness of the spacecraft to the sun, DSN antennas will pick up, together with the spacecraft signal, solar high-frequency noise, which will degrade the

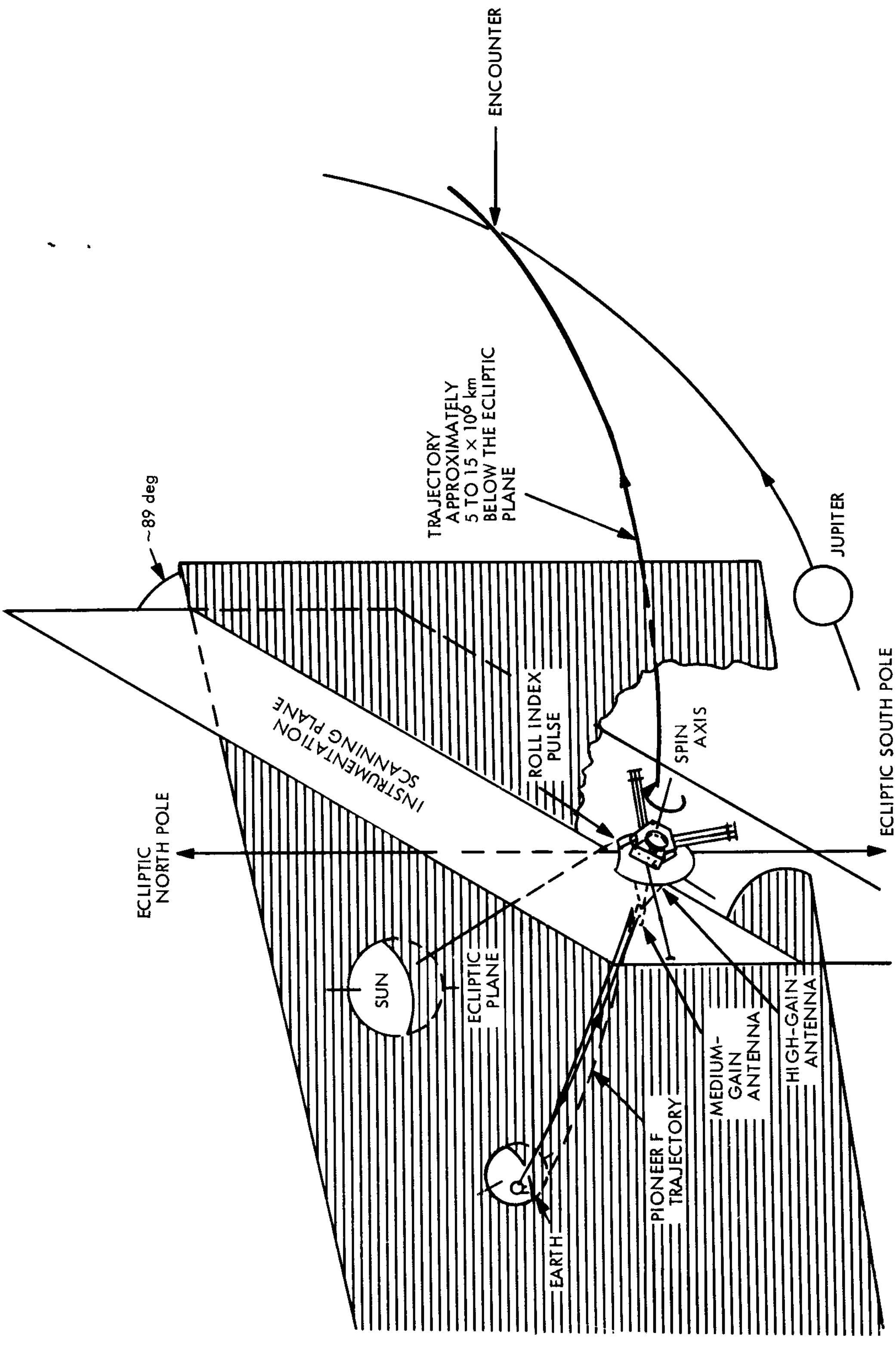



**Fig. 3. *Pioneer F* earth/Jupiter transfer trajectory**

received signal-to-noise ratio and can cut down considerably the quality and usefulness of the spacecraft/earth telecommunication link. This condition will exist for approximately 1 or 2 weeks.

To optimize the scientific value of the data return during the Jovian encounter, the flight project has selected specific flyby trajectories. The objectives of the encounter trajectory for *Pioneer F* are:

(1) To penetrate the Jupiter radiation belt.

(2) To provide good viewing conditions for Jupiter before periapsis.

(3) To obtain a short occultation for the spacecraft by Jupiter (less than 1 h).

(4) To provide a radius of closest approach to the center of Jupiter between 2 and 3 Jupiter radii.

The typical trajectories will approach Jupiter in a counterclockwise direction, as viewed from the celestial North Pole, will meet the planet slightly south of the ecliptic, pass around the planet in a counterclockwise direction to an attitude between 1½ and 2 Jupiter radii above the surface of the planet, and will exit slightly north of the planet. The postencounter trajectory will have an inclination to an ecliptic of less than 5 deg and the spacecraft will slowly escape the solar system. An additional 3 years will be required to cross Saturn's solar orbit.

The *Pioneer G* trajectory and sequence of events up to Jupiter encounter will be similar to that of *Pioneer F*.

During the voyage from earth to Jupiter, the spacecraft has to travel through a hostile environment, which can directly or indirectly affect the flight hardware and the operational characteristics of the spacecraft transponder/antenna system and can also constrain data acquisition and collection by the DSN. Special efforts will be made to study and analyze all possible constraints and to plan a configuration that will operate properly with these constraints and assure the best data return. A brief summary of some of the space hazards the spacecraft can be exposed to follows:

(1) As the spacecraft rushes through the earth's Van Allen belt, the instruments will detect vast intensity variations of high-energy protons and electrons. During this early deep space phase, the DSN must assist the Mission Operations Team in transmission of numerous commands and must collect all telemetry data.

(2) As the spacecraft travels through the asteroid belt, onboard sensors will attempt to detect meteoroids. During this phase, there is a very small probability that the spacecraft may collide with one of the meteoroids. This can cause an abrupt change in the spacecraft attitude and a loss of the spacecraft high-gain antenna radio link. In this event, the DSN must attempt to acquire spacecraft signals radiated by the medium-gain or omni-antennas.

(3) The electrons, protons, and magnetic field in the Jovian belt have a much higher intensity and flux density than in the Van Allen belt. It is possible that the crystal oscillators of the spacecraft transponder will slightly detune as the spacecraft traverses the Jovian belt. The high-intensity Jovian magnetosphere can also change the polarization ellipticity of the spacecraft signal radiated toward earth. In addition, the spacecraft flying in the close vicinity of Jupiter undergoes an abrupt velocity change which causes, within 2 or 3 h, a considerable change in the doppler shift; this shift has to be tracked by the spacecraft receiver and by the DSN.

## III. Spacecraft Systems

The *Pioneer F* and *G* spacecraft are carrying, as primary electrical power sources, radioisotope thermoelectric generators (RTGs) provided by a contractor of the Atomic Energy Commission. The spacecraft has been designed by the TRW Systems Group on contract with the NASA/Ames Research Center. The production and testing of the spacecraft will also be performed by the same vendor. The basic design requirements of the *Pioneer F* and *G* spacecraft can be summarized as follows:

(1) Be compatible with the *Atlas* SLV3C/*Centaur*/PE-364-4 launch vehicle.

(2) Be compatible with the Mark III DSN system.

(3) Provide a thermally controlled compartment to house the scientific instruments.

(4) Provide a telecommunications data system to sample readings from the instrumentation and transmit the information to earth.

(5) Provide a system to permit changes in operating modes of onboard equipment on command from earth.

(6) Provide a magnetically clean and an electromagnetic interference-free spacecraft.

(7) Operate in space for more than 2 years and for distances beyond the orbit of Jupiter.

To make the telecommunications subsystem of the spacecraft compatible with the Mark III DSN system, specific capabilities must exist. The spacecraft transponder must operate at predetermined DSN S-band frequencies and should have the capability to operate in a coherent two-way mode having a fixed ratio between the uplink and downlink frequencies. This capability is necessary to extract at the deep space stations accurate doppler measurements to determine the accurate spacecraft velocity relative to earth. These measurements are necessary for trajectory determination and for the generation of predictions required by the deep space stations to make an effective acquisition of the data link possible. The spacecraft must also be equipped with an auxiliary onboard oscillator to provide a downlink during times when the deep space stations have not established a two-way uplink/downlink telecommunications configuration.

The *Pioneer* Project Office of NASA/Ames Research Center plans to control every element of the mission system by a project documentation system. *Pioneer F* and *G* Project Specification PC-210, Spacecraft and Related Requirements, describes the functional requirements that the space vehicles must meet to achieve the primary and secondary mission objectives. This key document describes the expected environment through which the spacecraft will pass, and the required characteristics and performance of the spacecraft systems and subsystems. In addition, it specifies the reliability, quality assurance, and test programs to be conducted at the subsystem and system levels and the requirements for the activities at the launch complex to prepare the spacecraft and scientific instruments for launch. This document references the *Pioneer F* and *G* specification PC-222, Spacecraft/DSIF Interface Specification. A subset of this document is PC-222.02, which defines the characteristics of the equipment of the deep space stations pertinent to the spacecraft and the requirements imposed on the spacecraft by the DSN. The contents of this document were collected by the Project office from the DSN Standard Practice, 810-5, Revision A, Change 1, Deep Space Network/Flight Project Interface Design Handbook. DSN developed this handbook for use by all flight projects as a standard source of advanced technical information that describes the DSN interfaces with the Project office in the fields of telecommunications, data processing, and simulation. Since the *Pioneer* project uses the contents of this interface design handbook, assurance can be made that, on a functional level, the project interfaces will be compatible with the DSN configuration. To validate the functional compatibility between spacecraft design and the Mark III DSN system, plans are underway to make RF and data system compatibility tests between *Pioneer F* and *G* flight hardware at the Compatibility Test Area located at JPL. A verification of this compatibility will be made a few days prior to launch at DSS 71, located at Cape Kennedy, during the final checkout of the spacecraft and launch vehicle.

Based on the contractual functional design requirements of the spacecraft and its telecommunications subsystem, the spacecraft contractor, under the guidance of the *Pioneer* Project office, has developed the flight hardware.

The basic geometry of the *Pioneer F* and *G* spacecraft is shown in Figs. 4 and 5, and a perspective view is given in Fig. 6. The spacecraft essentially has two thermally controlled equipment compartments, one hexagonally shaped and containing spacecraft equipment and the other an appendage containing scientific instruments. Forward of the equipment compartments is a 2.75-m-diameter parabolic reflector, which is a basic component of the spacecraft high-gain antenna. Mounted on three struts forward of the reflector are the medium-gain antenna and the feed for the high-gain parabolic reflector. Electric power is supplied by four RTGs mounted in pairs on two radially deployable trusses. The generators are in a stowed position for launch, next to the equipment compartment and under the reflector of the high-gain antenna. The angle between the support trusses for the generators is 120 deg. In the deployed position, the generators extend well beyond the perimeter of the reflector to reduce the radiation environment within the equipment compartments and to reduce their magnetic influence on the magnetometer. This latter instrument is located on the end of a long folding boom, which, in the deployed condition, extends radially from the instrument side of the equipment compartment.

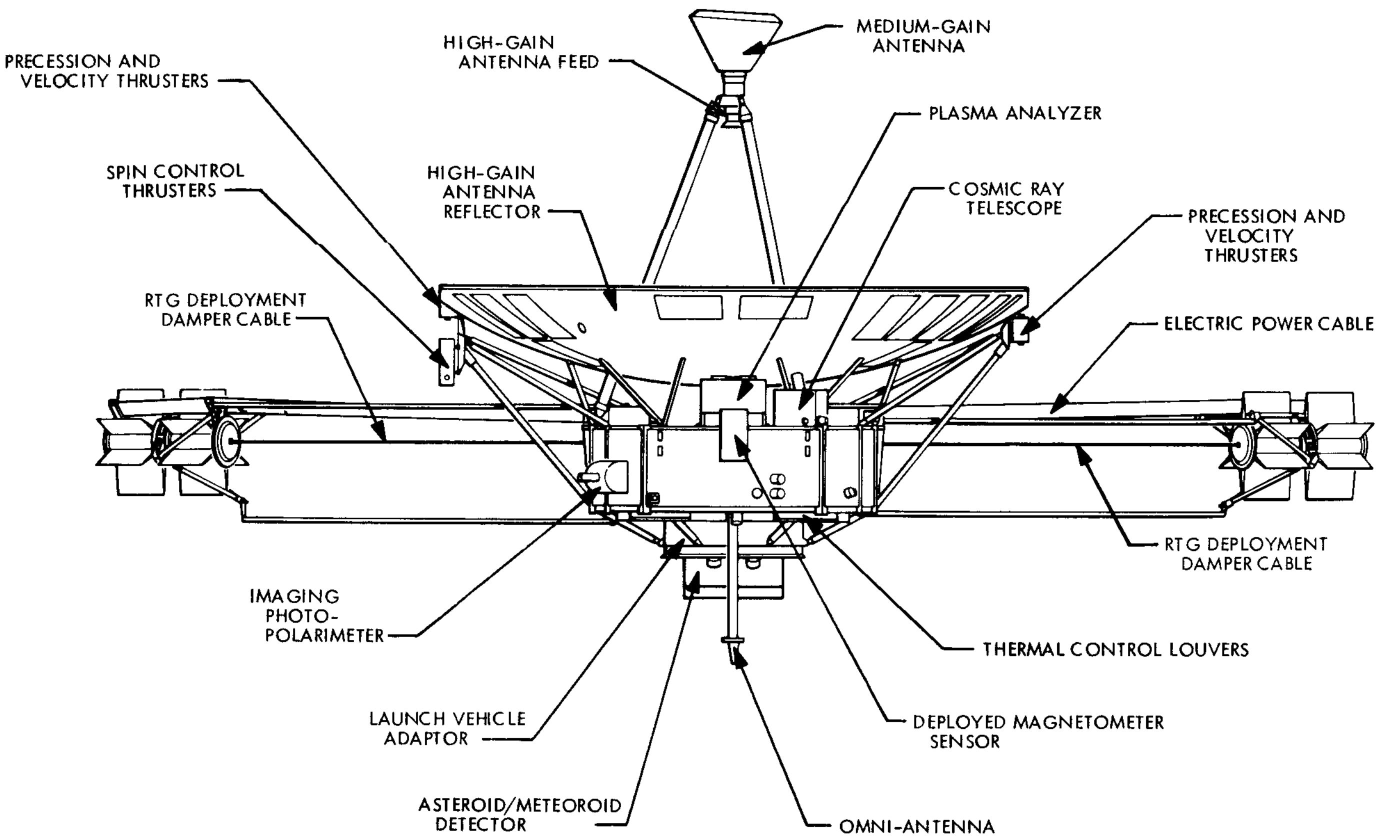



**Fig. 4. *Pioneer F* and *G* spacecraft (side view)**

Numerous viewing apertures are provided in the equipment compartments, as required by the scientific instruments. Mounts, external to the equipment compartment, have been provided for the meteoroid and asteroid instrumentation.

The spacecraft is spin-stabilized with a spin rate somewhat below 5 rpm. The launch vehicle's spin-up system brings the spacecraft, together with the third stage of the launch vehicle, up to a rotational speed of around 60 rpm. After the third-stage burnout and separation, the spin rate will be slowed down from 60 to 20 rpm by the use of automatically fired thrusters. Finally, deployment of the RTGs and the magnetometer will further de-spin the spacecraft to a nominal spin rate of 4.8 rpm. The objective of this spin stabilization is to stabilize spacecraft attitude. Spin-axis precession maneuvers will be applied during the mission to orient the spin axis of the spacecraft to the earth and thus illuminate the earth with the directional beams of the medium-gain and high-gain antennas.

Because the geocentric longitude of the spacecraft varies over a wide range during the mission, the spacecraft has a monopropellant attitude-control system which is capable of precessing the spin axis and maintaining the earth-pointing attitude on command. The initial reorientation to point the spacecraft spin axis toward the earth will occur within a few hours after launch. The spacecraft is equipped with a complete telemetry and data handling system which will generate a data stream containing the output of the scientific instruments and spacecraft equipment measurements. A system for receiving the modulating and distributing command instructions received from earth will provide flexibility in operation of the scientific instruments and the spacecraft. Spacecraft equipment will deliver conditioned power to the scientific instruments and also supply the instruments with appropriate timing and orientation indexing signals for control of measurements and data accumulation.

## IV. Spacecraft Subsystems

### A. Electrical Power Supply

The primary electrical power source consists of four Snap-19 radioisotope thermoelectric generators (RTGs).

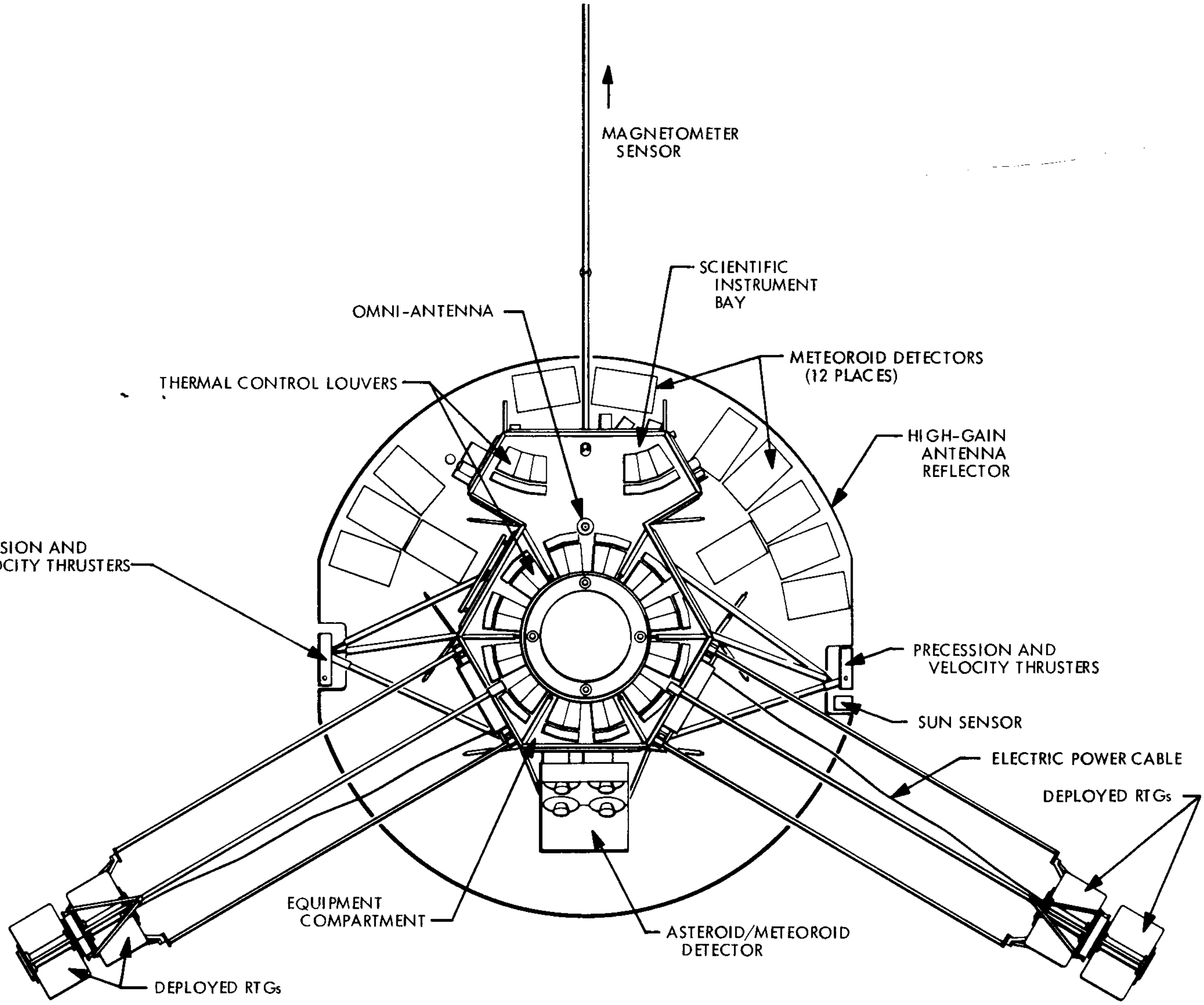



**Fig. 5. *Pioneer F* and *G* spacecraft (bottom view)**

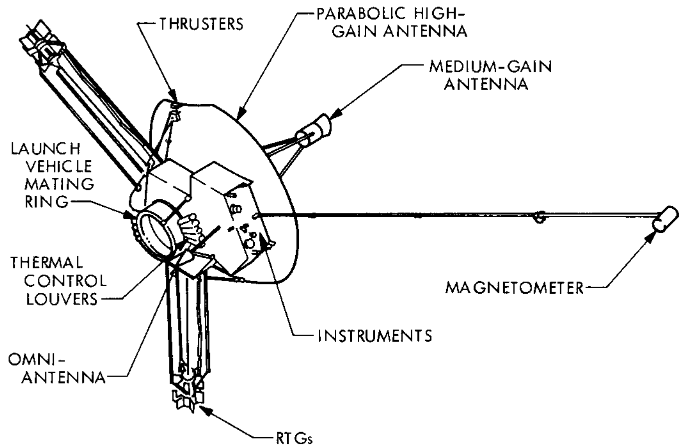



**Fig. 6. *Pioneer F* and *G* spacecraft (perspective view)**

The four units are in pairs in a tandem arrangement on the end of two supporting trusses. Provisions are made for retracting the RTGs to fit within the nose fairing of the launch vehicle for launch and for deploying them radially to the extended position after separation from the launch vehicle. Figure 7 shows the spacecraft mounted in the nose fairing and as assembled for the launch phase of the mission.

Each RTG unit will provide almost 40 W of electrical power during the early part of the mission and about 30 W 5 years after launch. It is expected that the power requirements of the spacecraft and scientific instruments will not exceed the capability of three of the RTG units at the time of Jupiter encounter. Therefore, the probability of having adequate power available at encounter is very high. The net weight of each RTG is around 13.6 kg.

### B. Electrical Power Conditioning

The objective of the electrical power subsystem is to distribute and condition the power received from the RTGs to the spacecraft equipment and the scientific instruments. To meet the requirements of the power loads, the dc output of each RTG goes into a separate inverter. The 2.5-kHz squarewave output of the four inverters is fed into an ac bus. Most of the ac power is rectified and filtered to supply the main dc bus, which is shunt-regulated to 28 V $\pm 1\%$ by dumping excess power through an external shunt radiator. The regulation of the 28-V dc bus is reflected back through the ac bus and through the fixed-ratio inverters, fixing the RTGs operating voltage at 4.2 V. A battery automatically carries any temporary overloads and is recharged automatically when excess power is available. The scientific instruments and the traveling-wave-tube power amplifier of the transmitter section of the transponder receive power from the main dc bus. Most of the other spacecraft loads are supplied from the central transformer–rectifier–filter unit, which receives power from the ac bus and provides various dc output voltages.

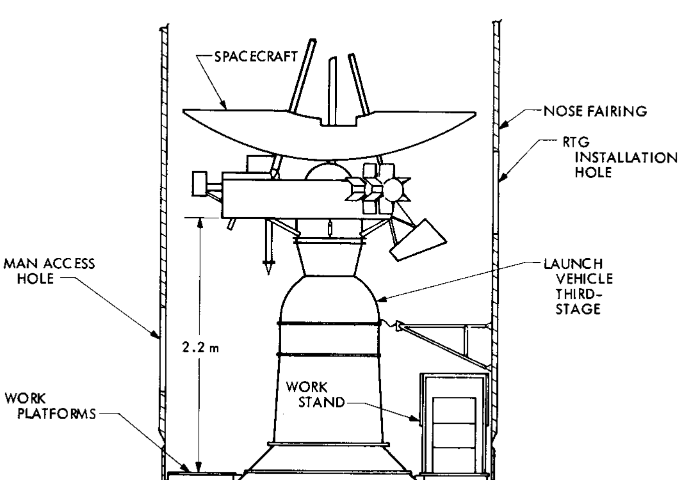



**Fig. 7. Folded *Pioneer F* and *G* spacecraft mounted on launch vehicle within nose fairing**

### C. Structure

The hexagonal equipment compartment is the basic structural element of the spacecraft. It supports the high-gain antenna on its forward end, attaches the launch vehicle by the launch-vehicle mating ring at its aft end, supports the scientific instrument compartment, accommodates the major portion of other subsystem assemblies, and provides for mounting of various external components, including the RTGs and the magnetometer boom. Rigid tubular truss work attached to the framework of the equipment compartment supports the parabolic reflector of the high-gain antenna, the high-gain antenna feed, the medium-gain antenna, the three thruster clusters, the attachment for the deployable booms for the RTGs, and the launch-vehicle mating ring. The reflector of the high-gain antenna is made from an aluminum honeycomb sandwich.

### D. Thermal Control

The thermal control subsystem provides the required thermal environment for the spacecraft components and scientific instruments during all phases of the flight. The objective of the thermal control subsystem is to maintain, in the vicinity of the scientific instruments, temperatures between $-18$ and $+38$ C and to keep the spacecraft equipment at temperatures required for satisfactory operation. Heating effects caused by the RTGs (stowed within the nose fairing), the third-stage motor casing, and the jet plume must be accommodated. The large variations in solar intensity and relative direction during interplanetary flight, and the loss of heat from the equipment compartments through sensor apertures, are compensated through louvers and special installation.

### E. Propulsion and Attitude Control

During the major portion of the flight, earth-pointing attitude is necessary for the narrow beam of the high-gain antenna of the spacecraft to illuminate the earth and to maintain effective communications. Figure 8 depicts

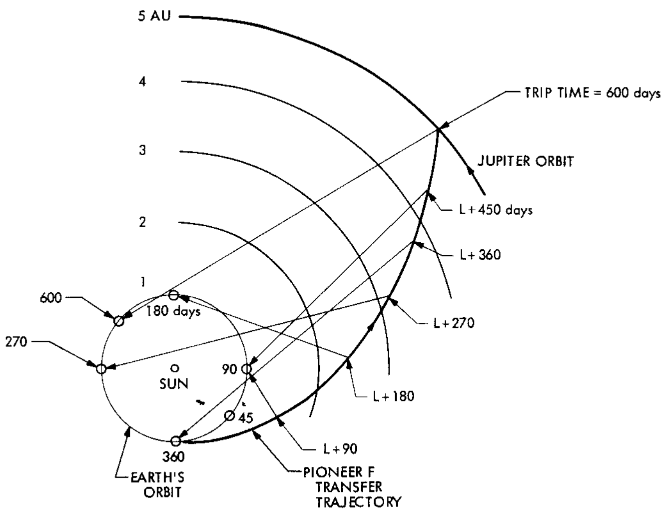



**Fig. 8. Relative position of *Pioneer F* spacecraft spin and antenna axis toward earth**

the relative position of the spacecraft's spin and antenna axis versus the earth. It can be seen that the geocentric longitude of the spacecraft varies continuously throughout the mission and, therefore, it is necessary to make numerous attitude adjustments. Since the spacecraft high-gain antenna has a half-power beamwidth of approximately 3.5 deg, it is anticipated that more than 200 spin-axis orientation maneuvers will be necessary to compensate for the relative movement of the spacecraft versus earth, and also for the precession caused by solar pressure, which is 0.2 deg per day during the early part of the mission. In addition, to provide the planned encounter trajectory at Jupiter, some adjustments of the velocity vector may be required during the interplanetary flight. To make the velocity vector adjustments possible, thrust must be generated in a particular direction; therefore, with thrustors in a fixed relationship to the spacecraft, reorientations of the spacecraft will also be necessary.

Changes in the spin rate will also be required. After injection, the rate will be on the order of 60 rpm and must be reduced to about 20 rpm before the deployment of the magnetometer and RTGs. As a result of the deployment, the spin rate is reduced to between 4 and 5 rpm (nominal: 4.8 rpm). In addition, to make possible attitude and velocity changes, small changes in the spin rate have to be corrected to maintain the rate of spin within the required limits.

Changes in attitude, velocity, and spin rate during the interplanetary flight will be accomplished by monopropellant hydrazine thrusters. Thrust will be provided by exothermic decomposition of the hydrazine in a catalyst bed and extension of the gas through a nozzle. Figure 9 depicts the location of the attitude-control spin control, velocity, and precession thrusters at the edge of the parabolic high-gain antenna structure. These thrusters can be operated in pairs. Each ~~cluster~~ contains a forward-facing nozzle and a rearward-facing nozzle. Two forward or two rearward nozzles will be used for velocity adjustment and opposite-facing nozzles will be fired for attitude changes, causing precession of the spin axis. Spin-rate changes will be accomplished by tangentially aligned nozzles thrusting with and against the spin.

A sun or star sensor (Fig. 10) provides reference signals necessary to time the thrust pulses for precession of the spin axis in a desired direction (Fig. 11). Attitude changes can be accomplished "open loop" by ground command, or "closed loop" by homing the spacecraft on the S-band uplink signal radiated by a deep space station toward the spacecraft. The duration of thrust for velocity and spin-rate changes is established by ground calculations. This information is transmitted to the spacecraft, where it is stored for execution on command. Similarly, for an "open loop" reorientation, the direction and amount of precession desired can be transmitted to the spacecraft via the command link and stored for execution on command. This storage information can be combined to perform a precession–velocity change–precession sequence with suitable time intervals in the sequence and to provide a completely automated velocity vector adjustment with return to a selected spacecraft orientation.

A closed-loop decision maneuver will be used regularly for accurate realignment of the spin axis of the spacecraft toward earth. A medium-gain and a high-gain spacecraft antenna will be used, respectively, for course and fine homing on the uplink and telecommunications signal. For the closed-loop maneuver, the axis of the medium-gain antenna and the feed of the high-gain antenna are offset from the spin axis and provide an amplitude-modulated signal when the spin axis is not aligned with the earth. The so-called CONSCAN subsystem processes this signal and fires the precession thrusters to establish the required precession and orient the spin axis toward earth.

The hydrazine will be supplied to the thrusters through appropriate lines and valves from a single, spherical pressurized bladder-tank, which is located in the center of the spacecraft equipment compartment. Electrical and small radioisotope heaters will be used to keep the plumbing of the hydrazine system above 2°C, which temperature keeps the fuel in a liquid state.

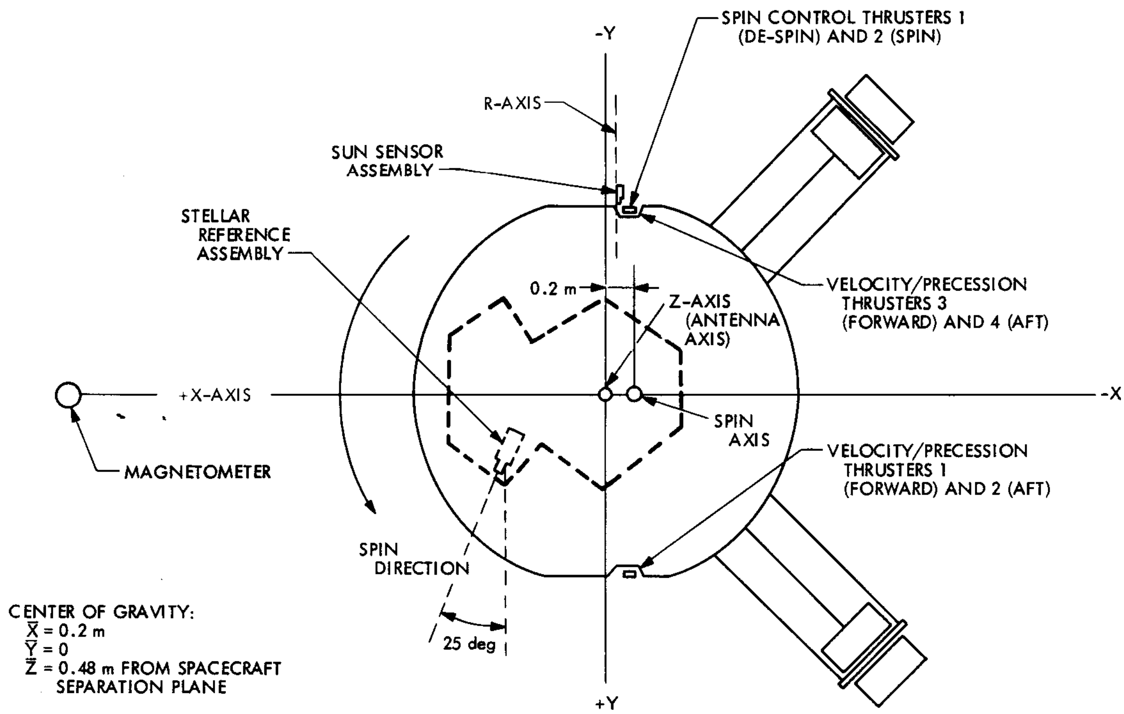
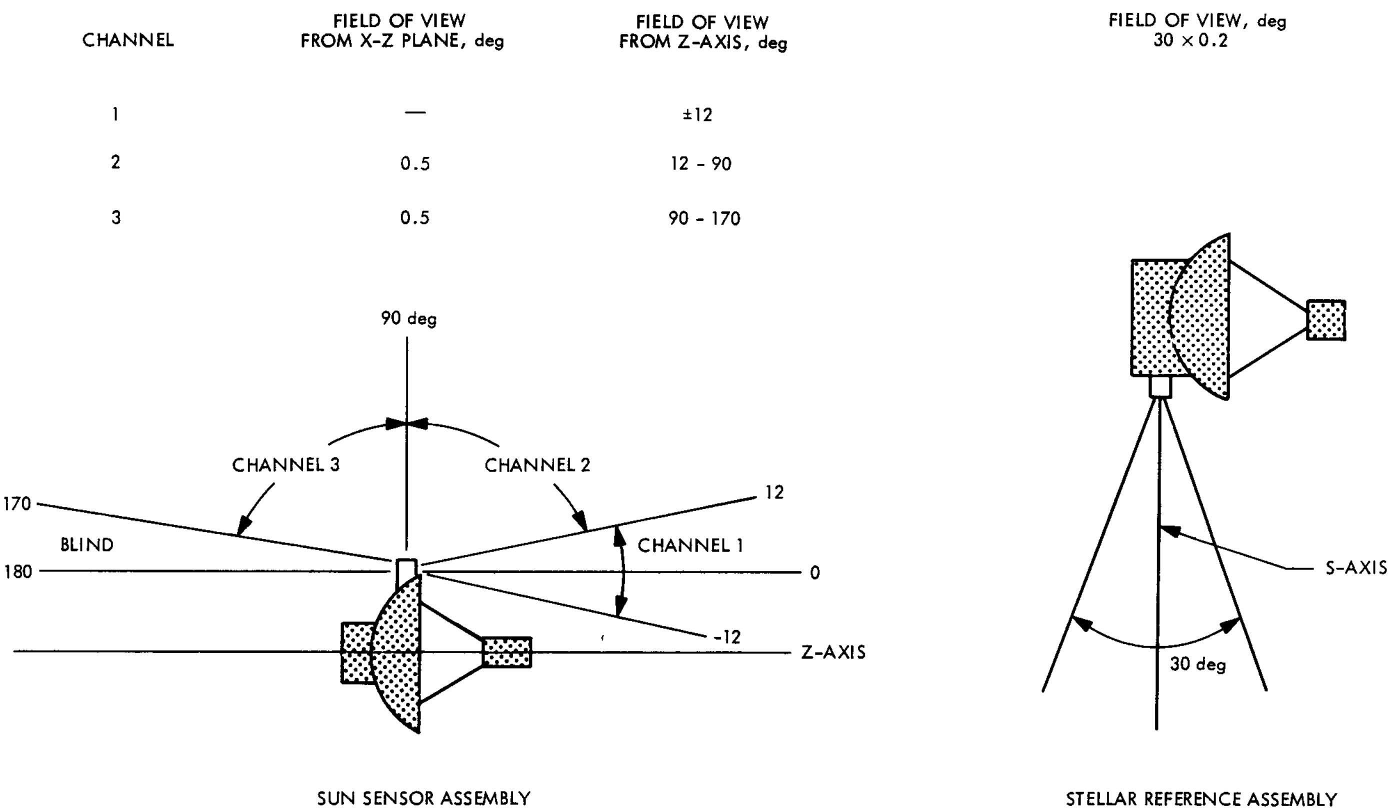



**Fig. 9. Attitude-control equipment locations (view looking aft, from spacecraft to booster)**

| CHANNEL | FIELD OF VIEW FROM X-Z PLANE, deg | FIELD OF VIEW FROM Z-AXIS, deg |
|---|---|---|
| 1 | — | ±12 |
| 2 | 0.5 | 12 - 90 |
| 3 | 0.5 | 90 - 170 |




SUN SENSOR ASSEMBLY

STELLAR REFERENCE ASSEMBLY

**Fig. 10. Sun and star sensors**

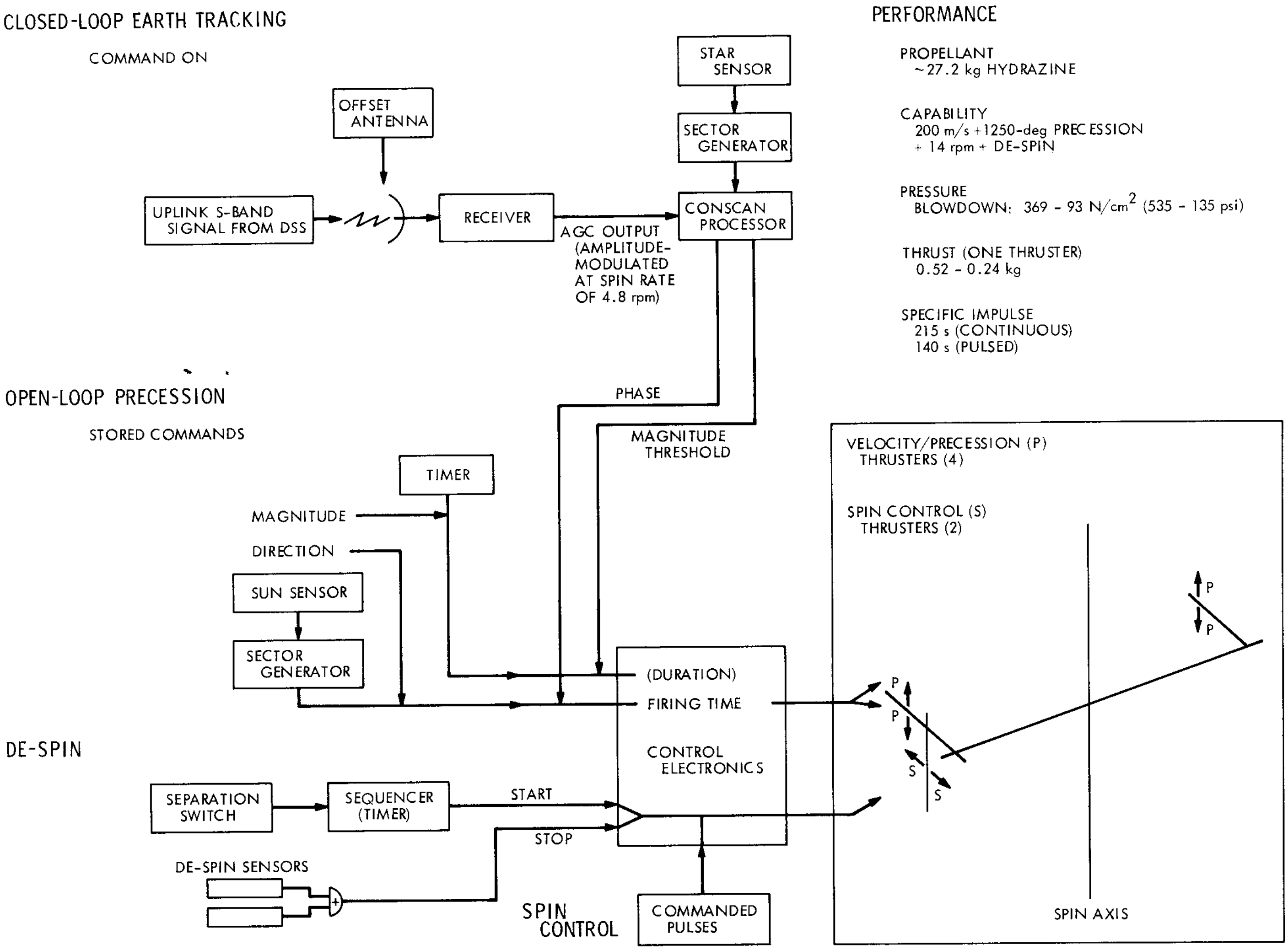



**Fig. 11. Attitude, velocity, and spin control (attitude-control subsystem)**

## Reference


1. Siegmeth, A. J., "*Pioneer* Mission Support," in *The Deep Space Network*, Space Programs Summary 37-66, Vol. II, pp. 4–11. Jet Propulsion Laboratory, Pasadena, Calif., Nov. 30, 1970.