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# DESCRIPTION OF THE SERT II SPACECRAFT AND MISSION

by Richard G. Goldman, Guy S. Gurski, and William H. Hawersaat
Lewis Research Center
Cleveland, Ohio

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# DESCRIPTION OF THE SERT II SPACECRAFT AND MISSION

R. G. Goldman, G. S. Gurski, and W. H. Hawersaat
Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio

## Abstract

This paper is a description of the SERT II spacecraft and its mission. The mission purpose and the orbital flight plan to accomplish this purpose will be discussed. The SERT II objectives are to prove the reliability and endurance capability of an ion thruster; develop and refine operational procedures; ascertain operating characteristics in the space environment; and validate ground test results. A mission objective of six months continuous operation of one thruster system is expected to be achieved by a flight program stressing the use and application of reliable, space-proven components and design philosophies.

## Introduction

The SERT II Project was undertaken to substantiate the reliability and endurance capability of an ion thruster in the space environment; to demonstrate the compatibility of a thruster system capable of space missions with its associate spacecraft systems; to develop and refine operational procedures; to ascertain operating characteristics in the space environment; and to validate ground tests. A minimum six-months mission was chosen as a suitable compromise among the life expectancy of flight-proven hardware, a continuous sunlight orbit and operating time on a single thruster long enough to be extrapolated to even longer missions such as would be required in powered flight to the other planets. [1, 2, 3]

The SERT II satellite was launched from Vandenberg AFB on February 3, 1970. This report describes the requirements of the mission, the orbit chosen to meet these requirements, and a general description of the spacecraft including experiments, power system, telemetry, command and attitude control.

## Background

Establishment of the characteristic operating life of the thruster is the primary objective of the SERT II mission because it is characteristic long operating life that transforms the ion thruster from a laboratory device to a useful space thruster system. The test objective of six months was selected as a compromise between future mission requirements and the cost, complexity and organization available.

Also of significant importance is the validation of ground testing provided by a space flight. The more confidence that can be generated in simplified ground testing, the more money can be saved by obviating the necessity of endurance tests in the actual space environment. This saving also reflects in the elimination of effort required in running ion thrusters in large space simulation chambers.

Potential users of ion thrusters must be in possession of the specialized techniques necessary to develop and operate them. The ground work for these specialized techniques has been established through the SERT I and SERT II flight programs. [6, 7, 8, 9, 10, 11]

Finally, the actual operating characteristics of the ion thruster, as affected by the space environment, cannot by accurately determined except in space. [4, 5]

## Mission Requirements

The mission requirements followed from the decision that the smallest thruster which would adequately demonstrate the operating parameters and endurance characteristics of this type of engine was a one kilowatt unit. Taking into account the degradation and spacecraft housekeeping functions, an initial, on-orbit power of 1500 watts was required. To power the thruster continuously requires a solar array flown in a six-month constant sunlight orbit. Sun synchronous orbits are essentially polar orbits. Constant sunlight is achieved by selecting the orbit altitude and inclination such that the oblateness of the Earth, that is, the belt of extra mass distribution in the equatorial region, precesses the orbit plane at approximately one degree per day, i.e., the angular rate at which the Earth moves around the sun. An on-orbit altitude of 1000 km was selected as the result of an optimization of gravity gradient torques, aerodynamic drag, solar cell degradation, required constant sunlight, and launch vehicle limitations.

The use of flight-proven hardware was an important ground rule laid down for the SERT II mission. Table 1 lists components derived from previous missions. In many cases, identical units were purchased from the manufacturer to the original specifications and drawings. In the assembly of the experimental and prototype spacecraft, some actual surplus hardware was obtained from the project concerned. From the list it can be seen that considerable usage was made of previous designs. Although this resulted in significant cost savings, the most important factor gained was confidence in the reliability of the hardware based on its previous flight history.

The Thorad-Agena was the launch vehicle. An advantage of the Agena vehicle is that it could be modified to support the SERT II mission with the large required solar array. The SERT Agena, together with the SSU and S/C, provides excellent moments of inertia for gravity gradient control.

A study indicated that the thermal control system could be obtained by entirely passive means using thermal coating on the outer surfaces of the spacecraft and spacecraft support unit.



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### Flight Sequence

As indicated in the previous section, a 1000 km constant sunlight orbit was desirable to meet mission objectives. The Thorad vehicle was, therefore, launched from the Western Test Range (WTR) which is the most efficient site from which to achieve a polar orientation. The Thorad burn plus first Agena burn put the satellite into an elliptical transfer trajectory to 1000 km where a second burn of the Agena circularized the orbit. Fig. 1 is a diagram of the injection procedure. The Agena vehicle attitude control system oriented the satellite nose down with the thrusters pointed toward earth. The ion thrusters, as operated in orbit, are offset by 10 degrees from the vertical. This offset results in enough tangential thrust to raise or lower the orbit altitude, depending on thruster selected. This altitude change is approximately 100 km during the six-month period. The orbit raising, which directly modifies orbit period, provides a direct measurement of integrated thrust. Real-time thrust was measured by a miniature electrostatic accelerometer.

## Satellite Description

### General

The SERT II configuration is shown in Fig. 2. The satellite is 1. 525 meters in diameter, 7. 92 meters long and weighs 1435 kilograms. Six meters of solar array extend outward on both sides of the Agena aft end equipment rack. As depicted in Fig. 2, the ion thrusters are pointed earthward with the main bulk of the satellite being the empty Agena. Attached to the forward end of the Agena is the spacecraft support unit (SSU). The SSU provides power conditioning and switching and telemetry and command systems. The primary attitude control system components are also in the SSU.

Forward of the SSU is the spacecraft (S/C), which houses the prime ion thruster experiment, together with all associated experiments.

The general weight breakdown for each major assembly follows:

| | Kilograms |
|---|---|
| Spacecraft | 282 |
| Spacecraft Support Unit | 220 |
| Agena Launch Vehicle | 740 |
| Solar Array | 193 |
| | 1435 kg |

### Spacecraft Construction

Identified by the two protruding thrusters, the spacecraft (S/C) provides support and protection for the various experiments. A completely redundant pair of ion thrusters are carried, each with its own power conditioning unit. Each thruster is mounted on gimbals so that the thrust vector can be adjusted on orbit through the center of mass of the satellite, thereby minimizing disturbance torques from this source. Several associated experiments are carried; a space probe to measure spacecraft potentials, motor-driven beam probes which are swept through the ion beam on command to measure the beam potential, surface contamination experiments (solar cells held at +71° C and -43° C) to determine contaminating efflux from the operating thruster, a radio frequency interference experiment (RFI) to measure any noise generated by the ion thruster in selected radio bands (these bands will be used by interplanetary and orbiting spacecraft), and finally, a miniature electrostatic accelerometer (MESA) which measures the very small acceleration of the satellite caused by the thrust of the ion engine.

Additional components satisfy the basic requirements for power switching, conditioning and instrumentation. Spacecraft attitude is measured by horizon scanners. A Backup Acquisition System (BACS) is provided for reacquisition should orientation be lost.

The basic shape of the spacecraft is a right circular cylinder 53. 4 cm high and 149 cm in diameter. Fig. 3 shows the configuration and general construction technique. The component access plates form the skin of the spacecraft. Three of the skins are aluminum rather than the lighter magnesium because of improved compatibility of aluminum with the Z-93 thermal coating used. Magnesium alloy is used for five outer skins to save weight. The power conditioning units are mounted on an aluminum radiator which forms an integral part of the spacecraft structure. The spacecraft component locations are shown in Figs. 4, 6, 7, and 8.

### SSU Construction

The SSU structure is very similar in construction, size, and appearance to the spacecraft, with the power conditioning radiator being the exception. Seven trays are used in the SSU for conventional component mounting whereas the spacecraft requires structural adaptation for supporting the large experiments. Thermal control patterns are similar for both spacecraft and SSU. All component access areas are covered with skins similar to the spacecraft. Three skins have a Z-93 paint and five have perforated aluminum tape. The SSU receives power from the solar array, and with its power system, regulates and controls the housekeeping functions. The thruster power from the array passes through the SSU to the spacecraft power conditioners. The SSU also provides the telemetry and command functions required. It houses in its center section a control moment gyro package which orients the satellite in one axis and provides damping for all axes.

Spacecraft support unit component layout is shown in Figs. 5, 6, 7, and 8.

### Launch Vehicle

The Agena is modified to serve as a spacecraft support platform. It has large mass distributions which provide favorable moments of inertia for gravity gradient stabilization. The SERT II Agena is 6. 25 meters long by 1. 525 meters in diameter, with a modified aft equipment rack which supports and thermally protects the solar array.

Once orbit is achieved, the Agena performs several key functions: 1) pointing the spacecraft earthward, 2) deploying the large solar array, 3) enable the SSU battery, 4) dumping of all propellants aboard the vehicle, 5) maintaining proper attitude during this sequence, and finally (upon ground command), switching horizon scanners to spacecraft control for attitude determination.

## Attitude Control

The satellite's orientation, depicted in Fig. 9, is maintained in pitch and roll by gravity gradient and in yaw by gravity gradient augmented by the control moment gyroscopes (CMG). The satellite weighs 1435 kg with a mass distribution similar to a bar bell, i. e., idealized moments of inertia which results in an optimum gravity gradient orientation capability. Moments of inertia are as follows:

| | |
|---|---|
| $I_{pitch}$ | 11000 $kgm^2$ |
| $I_{roll}$ | 8600 $kgm^2$ |
| $I_{yaw}$ | 2750 $kgm^2$ |

Resultant gravity gradient restoring torques for small angles are:

| | |
|---|---|
| $T_{pitch}$ | $2.21 \times 10^{-5}$ kgm/deg |
| $T_{roll}$ | $5.25 \times 10^{-5}$ kgm/deg |
| $T_{yaw}$ | $.69 \times 10^{-5}$ kgm/deg |

These restoring torques maintain the spacecraft orientation stably in conjunction with the CMG's which provide additional stiffness in the yaw axis and the necessary damping in all three axis.

A Backup Reacquisition Control System (BACS) is provided in case a temporary disturbance should cause the satellite to lose orientation. This BACS is a cold gas system which can reorient the spacecraft from disturbance rates up to 5°/sec. Reacquisition is accomplished open loop, by commands timed and executed by ground control.

## Thermal System

The SERT II satellite uses a passive thermal design because the satellite orientation with respect to the sun and earth is relatively constant. Extensive analysis, model tests, full scale tank tests, and experiments with coatings were employed in the thermal control design. This allowed the use of thermal coatings only (Z-93, black paint, polished aluminum and perforated aluminum tape). The thermal control pattern limits solar and earth thermal inputs, and regulates the radiated output so that a satisfactory thermal environment is maintained for all components. The thruster power conditioners are mounted on a large radiator which serves as part of the spacecraft structure. The thermal dissipation of the radiator requires that, when both P/C's are off, radiator strip heaters be turned on to maintain an adequate thermal environment.

## Power System

<u>Solar array</u>. - Rated at 1500 watts of usable power, the solar array is divided into two functional supplies, a 60 volt thruster section (1300 watts) and a 35 volt housekeeping section (200 watts). In the stowed position the array is attached to the rectangular truss work aft of the Agena tank section within the envelope of the Agena-Thorad adapter. Fig. 10 shows the stowed array.

Deployment of the array is accomplished by springs and a scissor mechanism. The array is fixed and so oriented on the orbital vehicle that, when deployed, it lies within the orbital plane facing the sun. Figure 2 shows the array in the deployed position. The array has a length of 6 meters on each side of the Agena and a width of 1.525 meters. It provides an active area of 17.5 square meters.

Thermal control on the array is achieved by the use of tinted white kemacryl on the back of the array, and reflective coating on the cover glass on the front of the array, maintaining the array temperature between 40 and 60° C depending on its orientation.

The array employs a modular construction, the basic module being 36.2 cm by 45.7 cm. Each module is composed of a thin magnesium waffle plate to which are bonded 74 solar cell submodules. The submodules are composed of five individual 2 cm × 2 cm solar cells connected in parallel on a 2 cm × 10 cm metallic plate. The solar cells are 12 mil thick N/P type with a 2 ohm-cm base resistivity and 11 percent efficiency at air mass zero. Radiation and thermal protection is provided by 20 mil thick Corning 7940 cover glass. Six of the basic modules are incorporated into a panel which is 0.76 meters × 1.52 meters. The overall array consists of 90 modules. The panel structure to which the modules are attached is lightweight aluminum.

<u>Power distribution.</u> - The 60 volt section of the solar array provides power through motor driven switches to the ion thruster power conditioning system. The 35 volt section of the array is fed into switching mode regulators (SMR) in the SSU for regulation and distribution. A block diagram of the power system is shown in Fig. 11.

Two SMR's provide regulated 26.5 VDC ± 1 percent. The main SMR supplies power to the spacecraft loads, inverters, battery charger, telemetry system and signal conditioning. The standby SMR supplies power to the command system and transmitters. Diode connection of the main SMR to the standby SMR provides power to the command system and transmitters if the standby SMR fails. Command capability exists to use either SMR to supply the spacecraft and SSU loads and also the capability to bypass both SMR's and operate directly from the solar array.

Redundant 115 VAC 400 cycle inverters operating from the main SMR output provide power to any two of the four control moment gyroscopes in the SSU, to the beam probe actuators and thruster gimbal motors in the spacecraft and to the Agena horizon sensors.

A 40 ampere-hour silver oxide-zinc battery capable of at least five discharge cycles is electrically back biased from the housekeeping solar array and is instantaneously available for emergency conditions. Should the SSU sense an undervoltage condition, the battery comes on line to support essential loads excluding the thrusters and experiments. A battery charger is provided to maintain charge status and recharge the battery should it be required.

Power switching for the telemetry and power systems is provided in the SSU while experiment switching is performed in the spacecraft. The command receivers and decoder cannot be switched off. All components in the spacecraft and SSU are fused. Undervoltage protection is provided to disable all systems except the command system and transmitter should the SMR regulated output drop below 23 VDC for longer than 200 milliseconds. The ion thruster power conditioning is disconnected from the thruster array should the housekeeping array drop below 23 volts for longer than one second.

## Communication System

Telemetry. - The airborne telemetry system, together with the supporting ground station and communication network form a composite system which provides the LeRC Control Center with real-time telemetry and command verification data during periods of ground coverage. In addition, data storage capability via two onboard tape recorders is provided for periods when STADAN coverage is not available.

A simplified block diagram of the airborne telemetry system is shown in Fig. 12. Data from four subcommutators are fed into a multicoder where the analog to digital conversion and time division multiplexing are accomplished. The pulse code modulated (PCM) multicoder output is then fed into a voltage-controlled oscillator (VCO) through a mixer and transmitted at 136 MC. The overall modulation scheme is PCM/FM/PM.

Timing pulses for experiments and transmission of tape recorder playbacks and command verification are also provided.

Redundancy in the telemetry system is provided for the multicoder, VCO's, mixer, tape recorder and transmitter. Although the four subcommutators are not redundant, some key data is on two subcommutators and also data is allocated so that all data from one device is not on one subcommutator.

Command system. - The airborne portion of the command system functions in conjunction with the tracking network to achieve real-time command control from the LeRC Data Control Center. Fig. 13 is a block flow diagram showing primary ground and spacecraft interfaces. Real-time commands are sent, and executed by the operator at the ground station in contact with the spacecraft.

The command system is an amplitude modulated 148 mc system with a 216 command capability. Upon receipt of a command, the receiver aboard the spacecraft demodulates the rf signal and directs the command signal to the decoder where the command received is identified, put in storage and inserted in the telemetry output for transmission to the ground station. At the LeRC Control Center, the command is verified as the correct command and the STADAN operator is instructed to send an execute signal to the spacecraft.

Redundancy is provided in the command system by use of two command receivers operated in parallel and a command decoder consisting of two fully redundant decoder "halves", each capable of decoding 108 commands. All essential commands for the thruster experiment and key housekeeping systems are redundant.

Antenna system. - A circularly polarized turnstile configuration of four monopole antennas with associated diplexers and hybrids is used as the common telemetry command antenna system.

## Experiments

Thruster. - The general configuration of the mercury bombardment ion thruster is shown in Fig. 14. The thruster assembly includes the propellant storage, feed system and neutralizer. (10) Permanent magnets are employed to create the longitudinal magnetic fields. The feed system is a positive pressure type using gas behind a diaphragm to force the mercury from the reservoir. Both the thruster cathode and the neutralizer are plasma discharge devices utilizing hollow cathodes.

The thruster requires 1000 watts of power and produces 6.2 millipounds of thrust when operated at the rated 250 milliampere ion beam current and 3000 volts DC net accelerating potential. The design operating point of the thruster is 89 percent power efficiency and 76 percent propellant utilization including neutralizer, at a specific impulse of 4240 seconds. Each of the two ion thruster systems weighs 9 kg plus 15 kg of mercury propellant.

Power conditioning. - The power conditioner(9) takes electrical energy directly from the thruster portion of the solar array and conditions it to meet the requirements of the thruster. Power requirements of the thruster are maintained over a solar array input voltage range of 54 to 75 volts. Average efficiency is 87 percent at 60 volts input and operating a thruster at 250 milliampere beam current.

The packaging configuration utilized for the power conditioning is open-to-vacuum construction as opposed to hermetic sealing. Unencapsulated magnetic components are used, eliminating the need for void-free encapsulation. Each of the two power conditioners weighs 14.8 kg and is $26.6 \times 51 \times 17.8$ cm. The operating temperature range is $0^{\circ}$ C to $+49^{\circ}$ C.

The all solid state transistor type inverters operate at 8 kHz and provide the nine power supply requirements of the thruster including the necessary control loops and telemetry outputs. A neutralizer bias supply which is one of the thruster experiments is also provided. Each of the individual power supplies has the capability to withstand shorts between any output terminals and to continuously operate into a short circuit. Overload shutdown and input voltage undervoltage shutdown are also provided.

Beam probe experiment. - It is required that the thruster beam plasma potential with respect to the spacecraft be measured. Due to the high efflux of the plasma accelerated from the thruster and the consequent wear on the probe filament, a sweeping type emissive probe is required. [7] The probe momentarily samples the beam potential and then returns to a position out of the beam.

When the beam probe is commanded to sweep, the probe electronics are automatically turned on to provide the necessary signal conditioning. The data is transmitted to the ground station in real time.

Each thruster has its own beam probe and actuator mounted on the thruster gimbal ring assembly.

Space probe experiment. - In order to evaluate the spacecraft potential during ion thruster operation, a space probe[7] is required to determine the spacecraft potential with respect to ambient space plasma potential. It must be measured at a point unaffected by the spacecraft plasma sheath and is, therefore, mounted on an isolated 5 ft boom. The boom is designed to deploy into the space plasma on the leading edge of the spacecraft. This separation serves to minimize spacecraft effects.

Surface contamination experiment. - The object of the surface contamination experiment[8] is to determine if contamination exists in the vicinity of operating thrusters. This experiment was formulated to determine the compatibility of solar cells to products generated by electric thrusters.

Each surface contamination experiment (one for each thruster) consists of two cell assemblies, one hot, and one cold. Both are thermally isolated from the structure and located near the ion beam.

The solar cells are located directly facing the sun and produce their own signal (voltage) which is proportional to the intensity of sunlight transmitted through the contaminant coating on the surface of the cell assemblies.

Miniature electrostatic accelerometer experiment. - The object of this experiment is to obtain an accurate measure of the thrust imparted to the spacecraft during the mission. There are four modes of ion thruster operation of particular interest: thruster off, 30 percent beam, 80 percent beam and 100 percent beam. The accelerometer output permits an accurate determination of these levels.

The accelerometer[11] permits instantaneous measurement of the thrust in real-time. This measurement is provided from the accelerometer electronics in both analog (coarse) and digital (accurate) form.

RFI experiment. - The need for identifying the type of RF noise generated by an ion bombardment thruster system has resulted in the RFI experiment. The experiment receives predetermined frequency bands and establishes the power levels at these frequencies. A wideband antenna which views either thruster's ion beam is required to receive these signals. Frequency bands, which have been selected, reflect spacecraft communication systems design for application in future deep space missions. The frequency bands are 300-700 MHz, 1680-1720 MHz, 2090-2130 MHz.

Reflector erosion experiment. - The reflector erosion experiment determines the degradation of a reflector surface in near Earth orbit. The optical degradation caused by the impact of micrometeoroids is measured as a change, during the course of the mission, of the temperature of a polished plate. Since the disk is oriented toward the sun, with a relatively constant thermal input, the temperature shift of the disk isolated from the spacecraft is a direct indication of optical degradation.

## Concluding Remarks

The SERT II program was required to provide a long life test of an electric thruster, obtain operational experience in this new method of propulsion and measure the characteristics of electric thrusters as they affected the design of future electrically propelled spacecraft.

The satellite consists of a spacecraft support unit (SSU) in which are concentrated the telemetry and command functions, AC and DC power conversion equipment and most switching and housekeeping functions as well as control moment gyros. The principal experiments, two ion thrusters, each with its own specialized power conditioning, together with subsidiary experiments to closely monitor the characteristics of the thrusters, are mounted in the spacecraft which bolts to the SSU. The subsidiary experiments comprise beam probes and a space probe, radio frequency interference (RFI), miniature electrostatic accelerometer (MESA), contamination cells and reflector erosion unit (REX).

The Agena launch vehicle provides proper moments of inertia for gravity gradient attitude control and supports the solar array and S/C-SSU combination. The resulting satellite represents a simple, low-cost solution to the basic requirements laid down for the SERT II mission. To date, the spacecraft has performed satisfactorily and preliminary data indicates that all mission objectives will be met.

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TABLE I

List of Components Having Previous Flight History

| Components Used In SERT II | Flight Vehicle |
|---|---|
| DC-AC Inverter | Agena |
| Switching Mode Regulator | LEM |
| Phase Sensitive Demodulators | LEM |
| Battery Charger | Agena |
| Battery | Mariner |
| Command Decoder | ISIS |
| Command Receiver | Pegasus |
| Transmitter | FR-1 |
| Hybrid | Pegasus |
| Diplexer | Pegasus |
| Tape Recorder | Air Force |
| Subcommutator | BIOS |
| Multicoder | BIOS |
| Frequency Division Multiplexer | BIOS |
| Time Code Generator | BIOS |
| DC-DC Converter | Agena |
| BACS | Surveyor or Lunar Orbiter |
| CMG | Agena |
| Solar Array | Air Force |
| MESA Accelerometer | Saturn |
| Horizon Scanner | Agena |

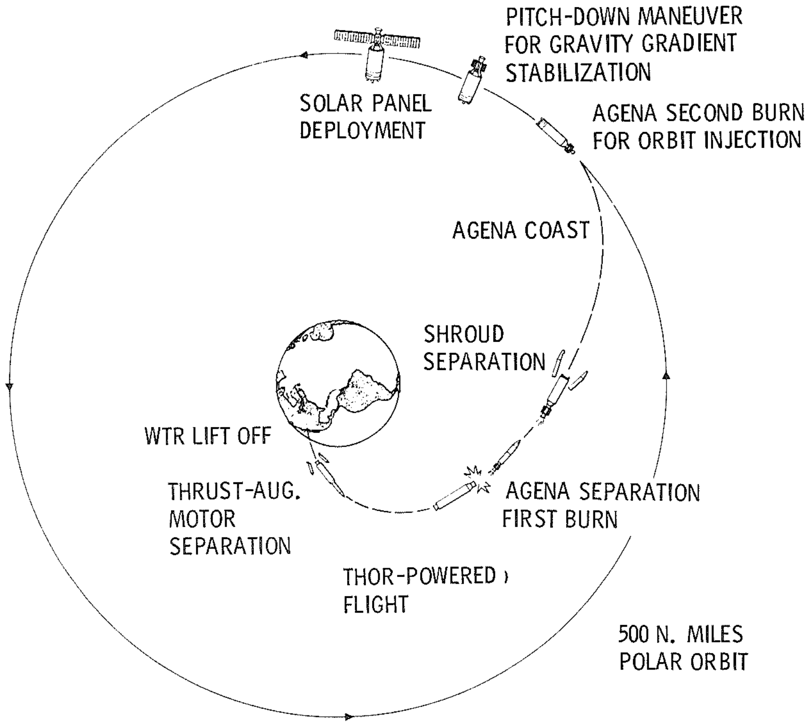


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Figure 1. - Representation of Sert II flight sequence.

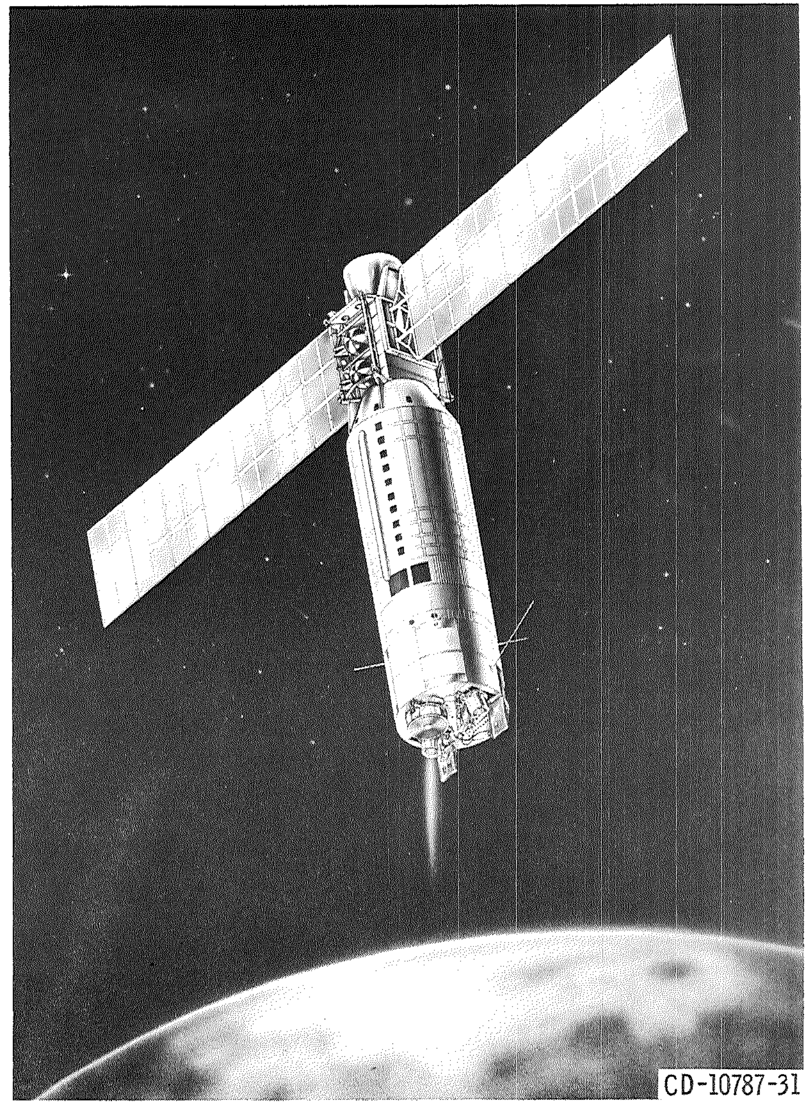


Figure 2. - Sert II in orbit.

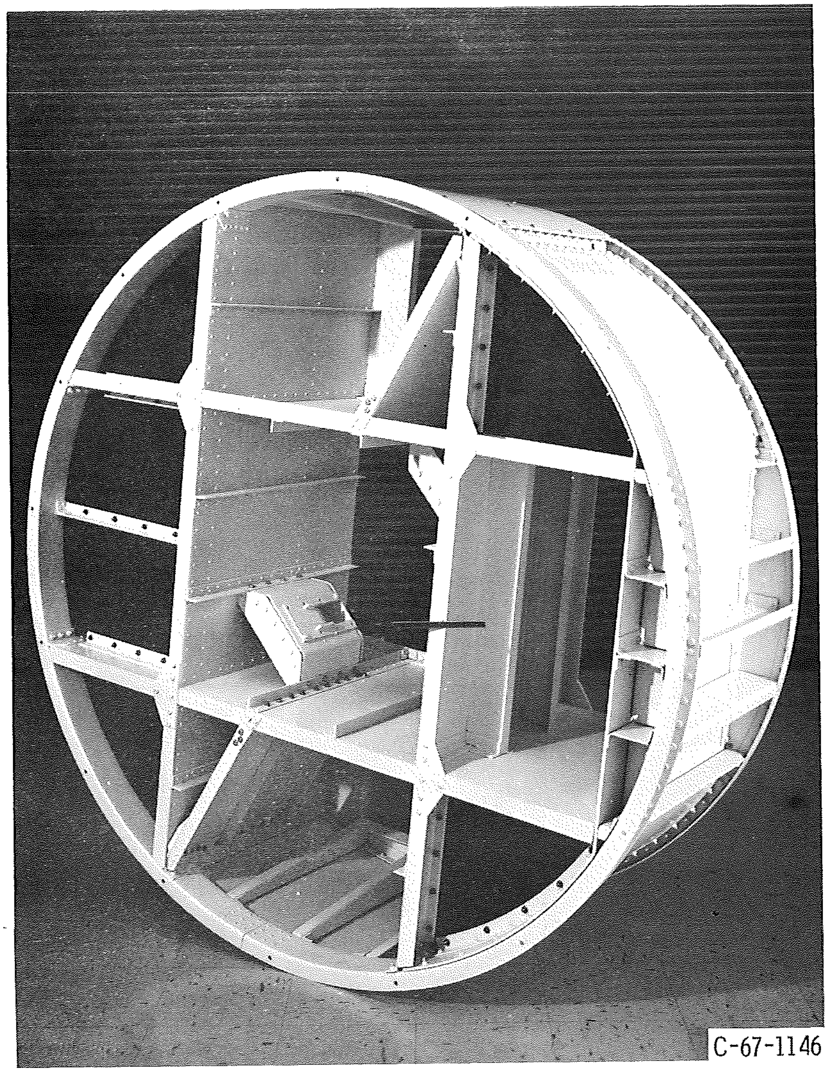


Figure 3. - Spacecraft structure.

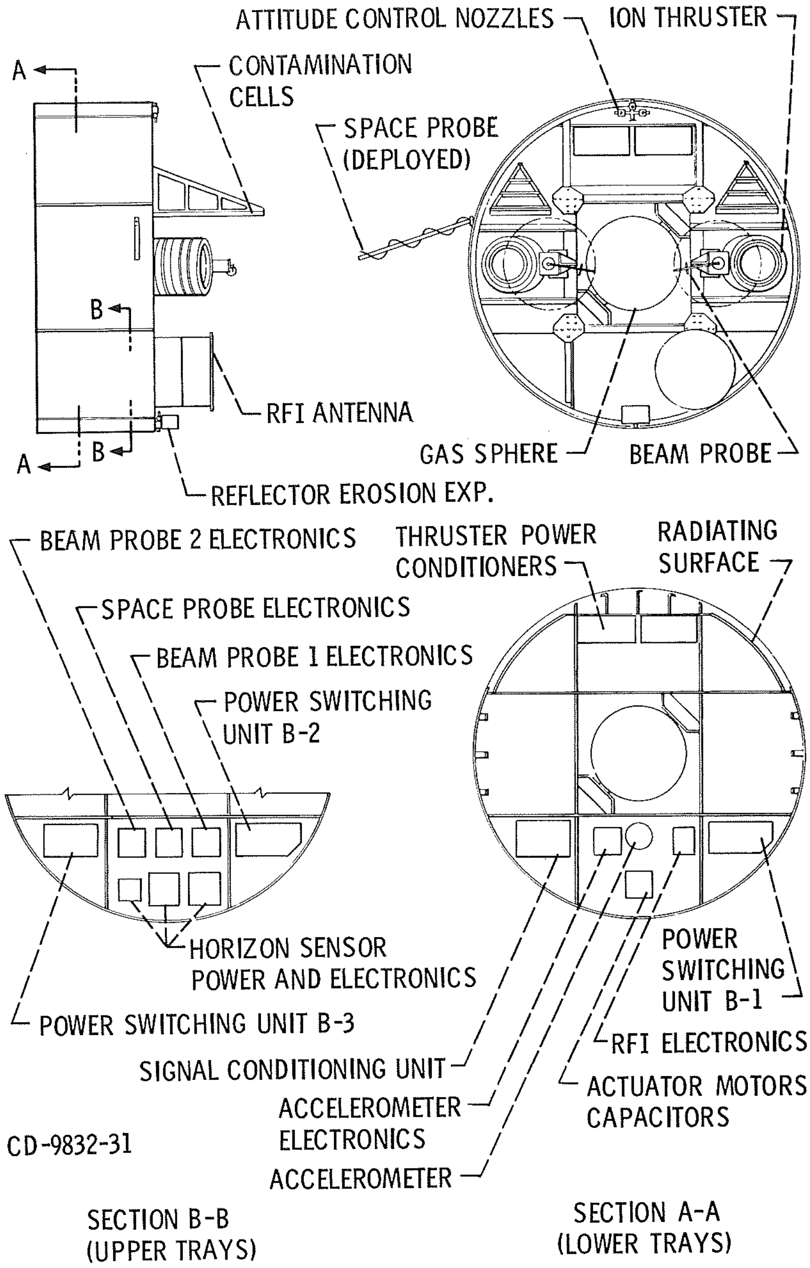


Figure 4. - General arrangement of spacecraft.

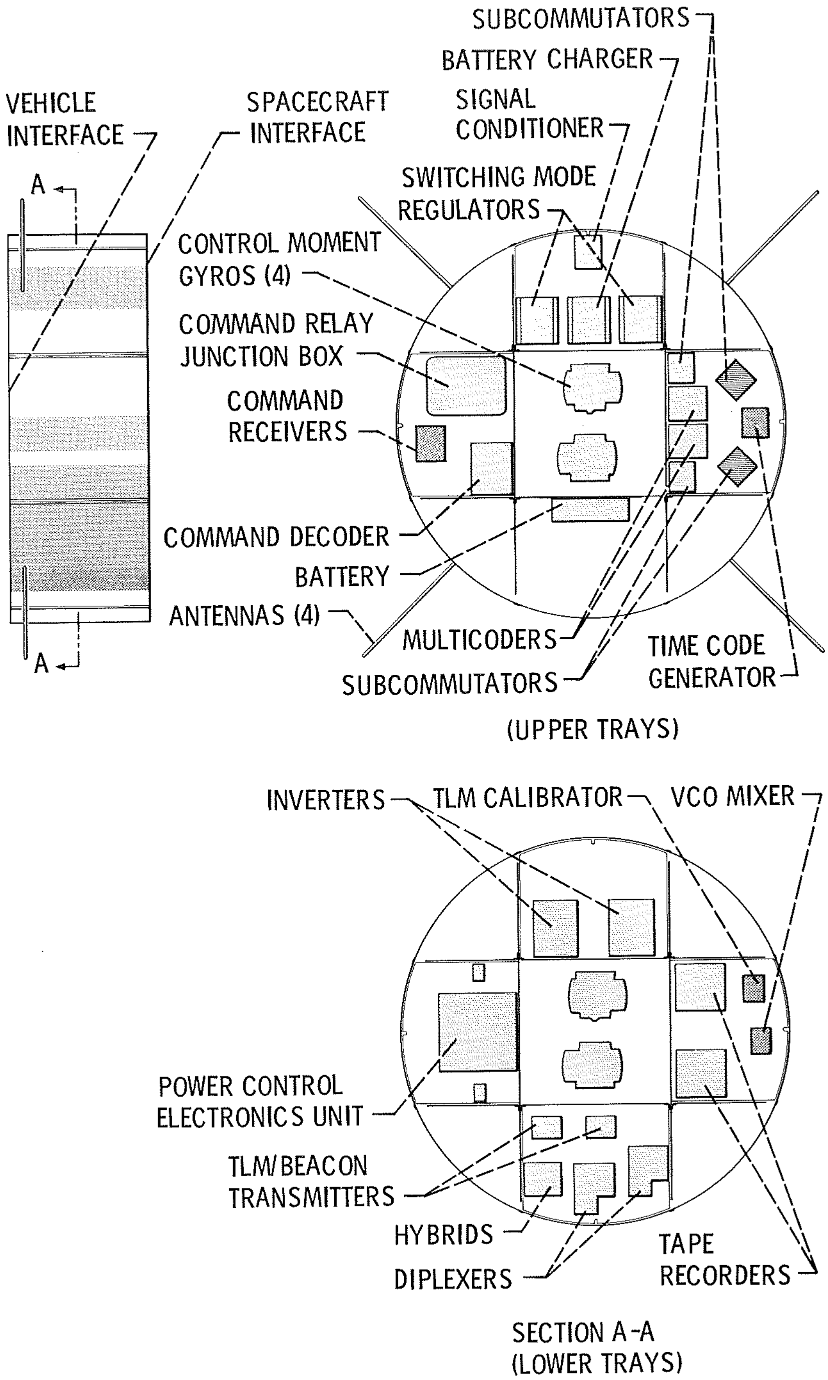


Figure 5. - General arrangement of spacecraft support unit (SSU).

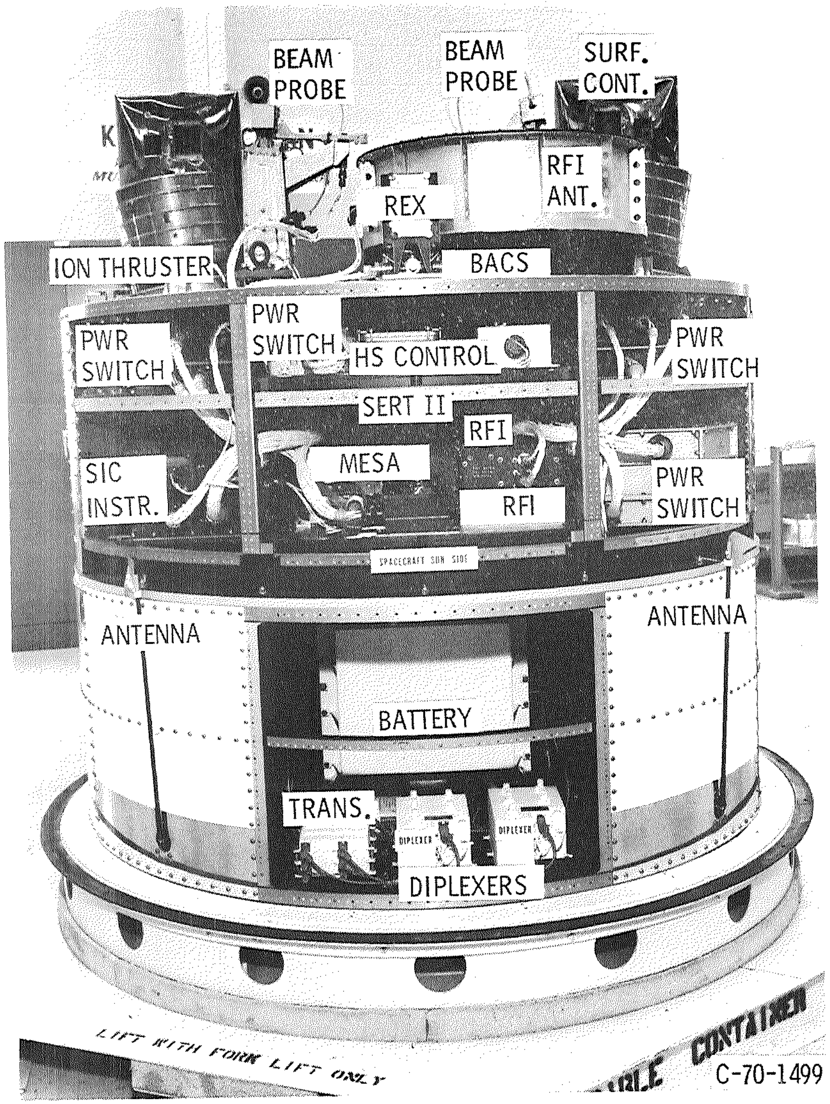


Figure 6. - S/C and SSU (Bay 8).

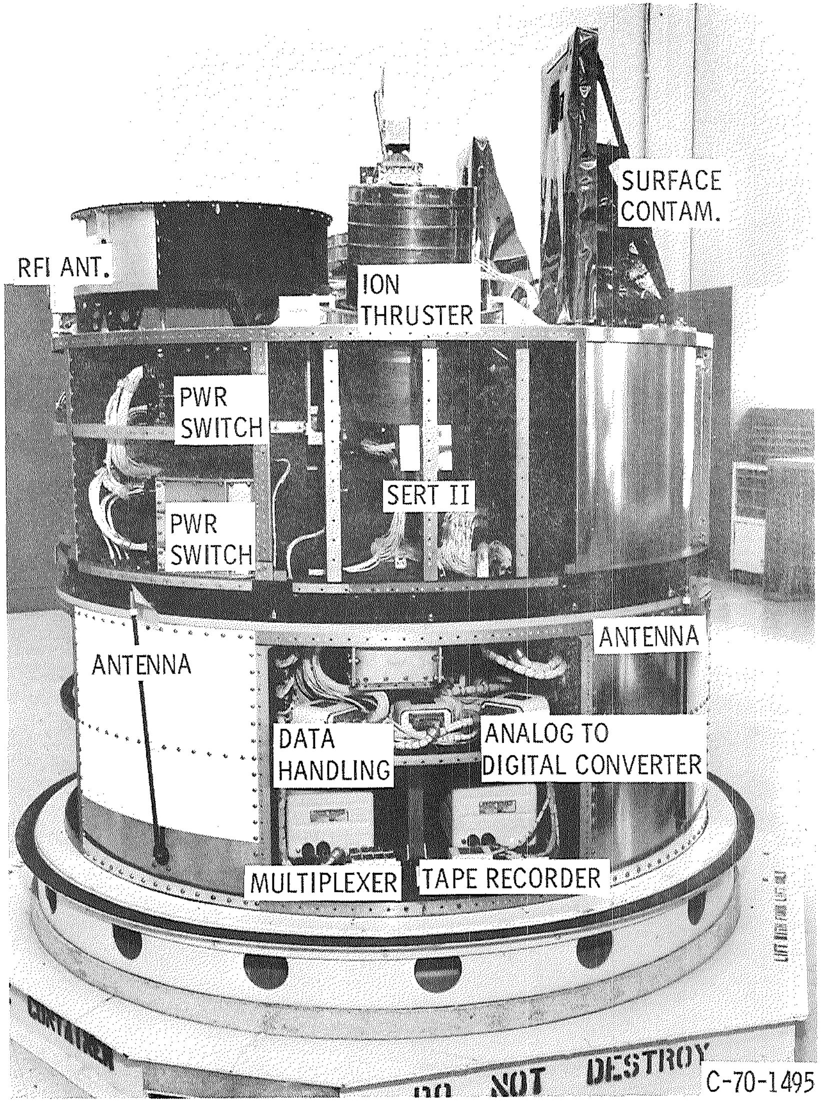


Figure 7. - S/C and SSU (Bay 2).

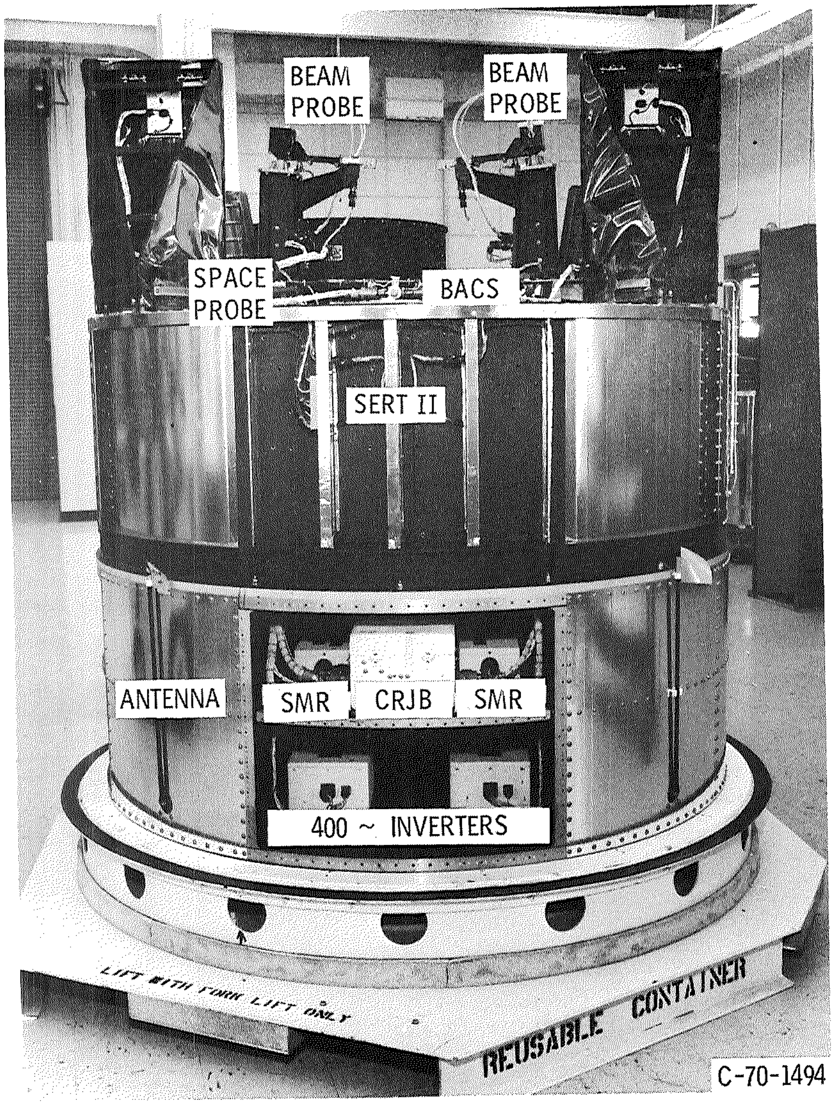


Figure 8. - S/C and SSU (Bay 4).

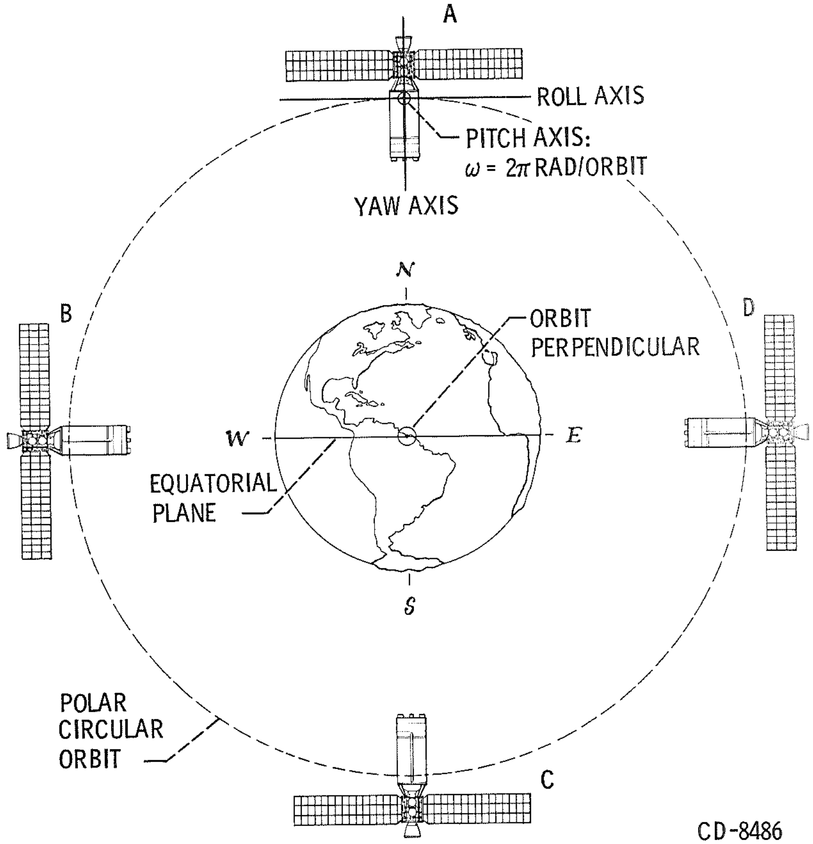


Figure 9. - SERT II circling Earth N-E-S-W orientation.

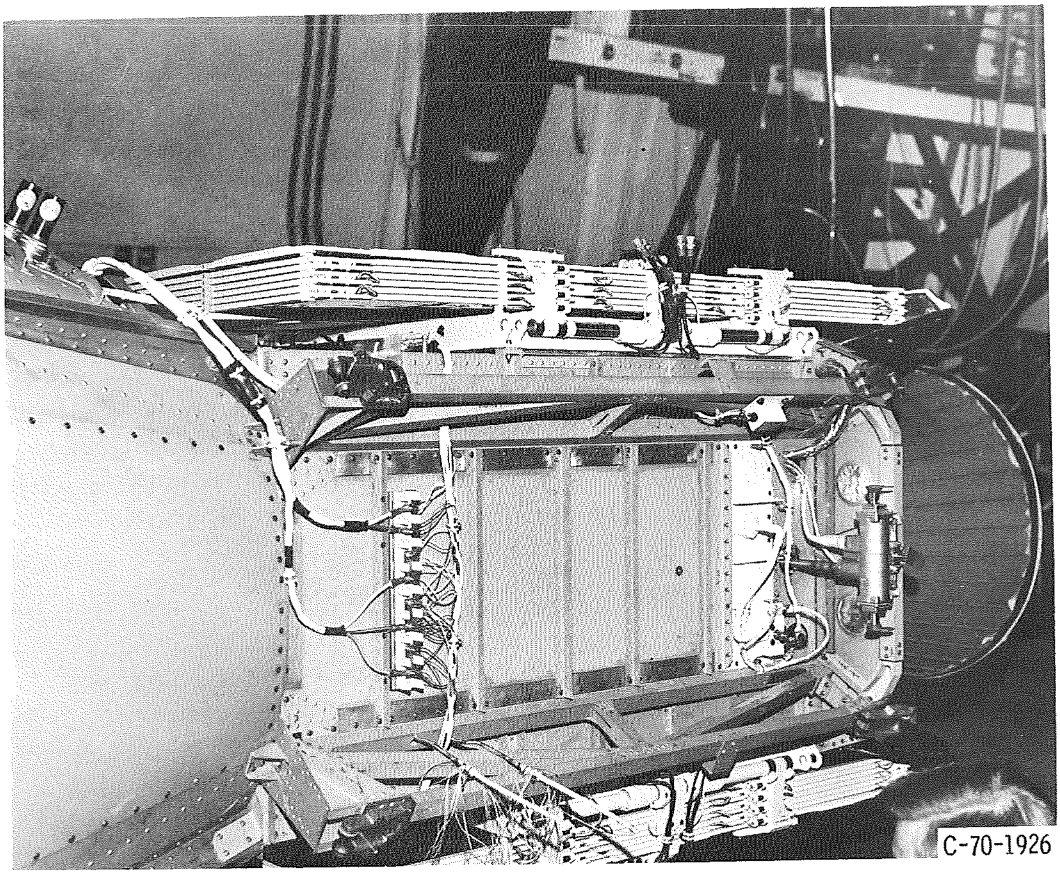


Figure 10. - Stowed solar array.

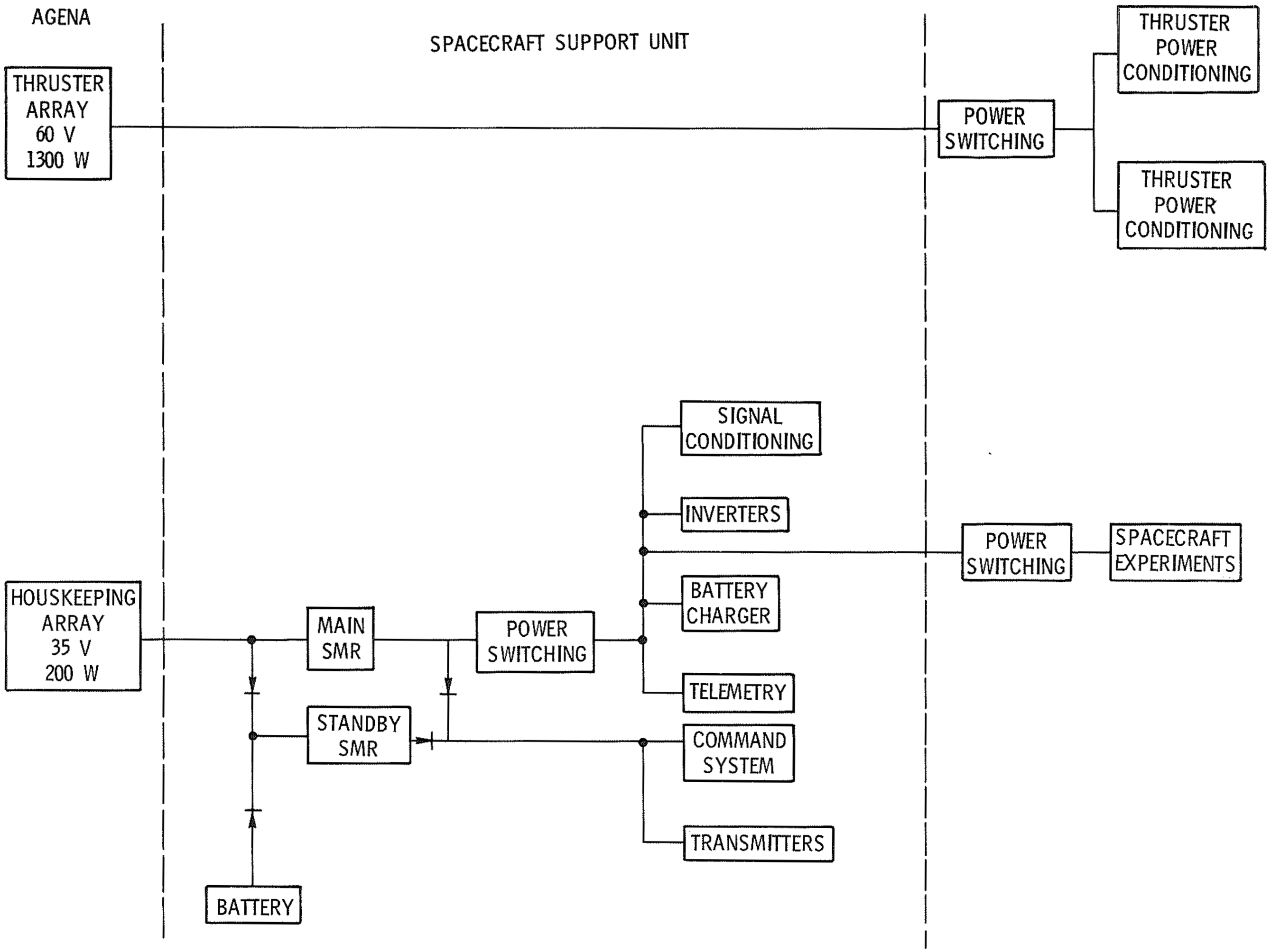


Figure 11. - SERT II power system.

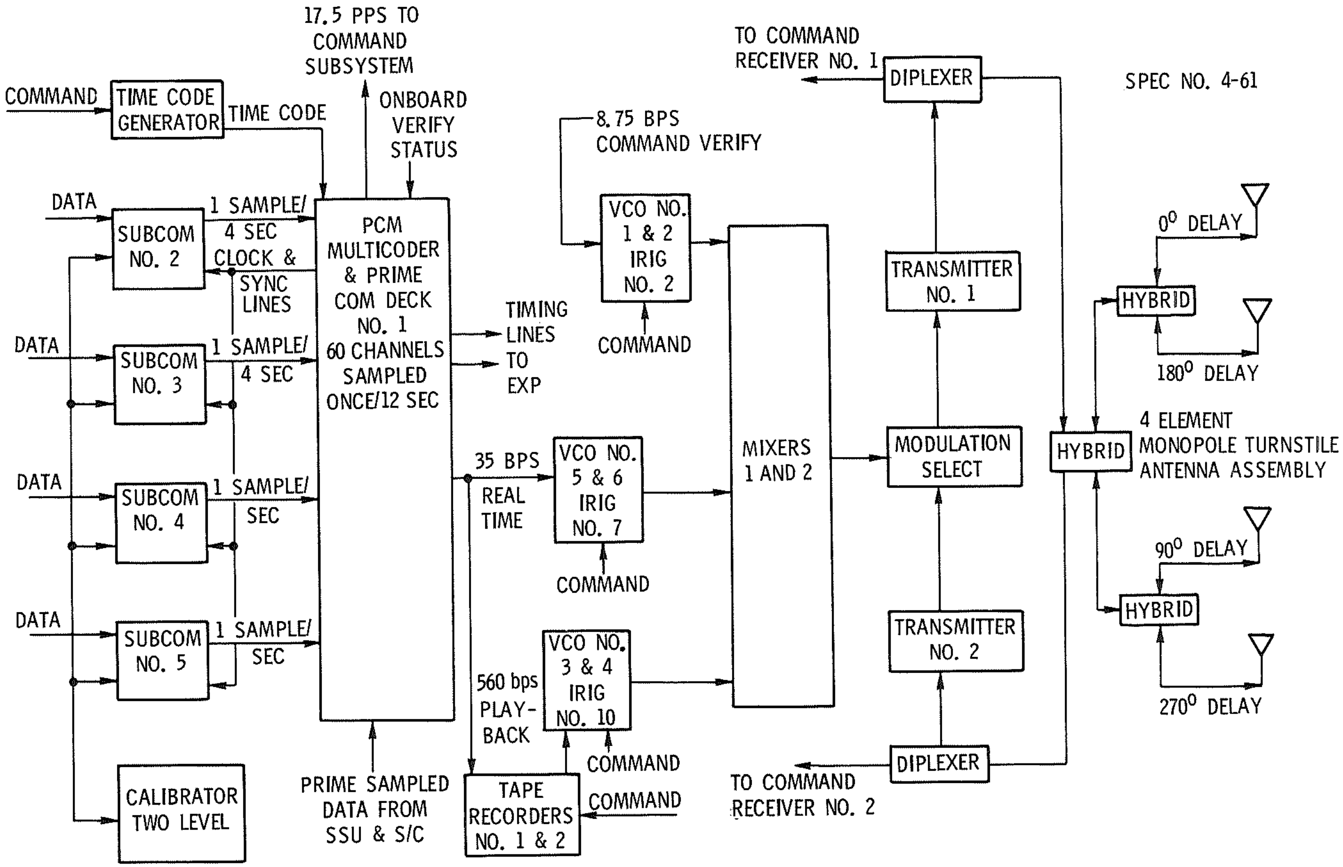


Figure 12. - Airborne telemetry subsystem.

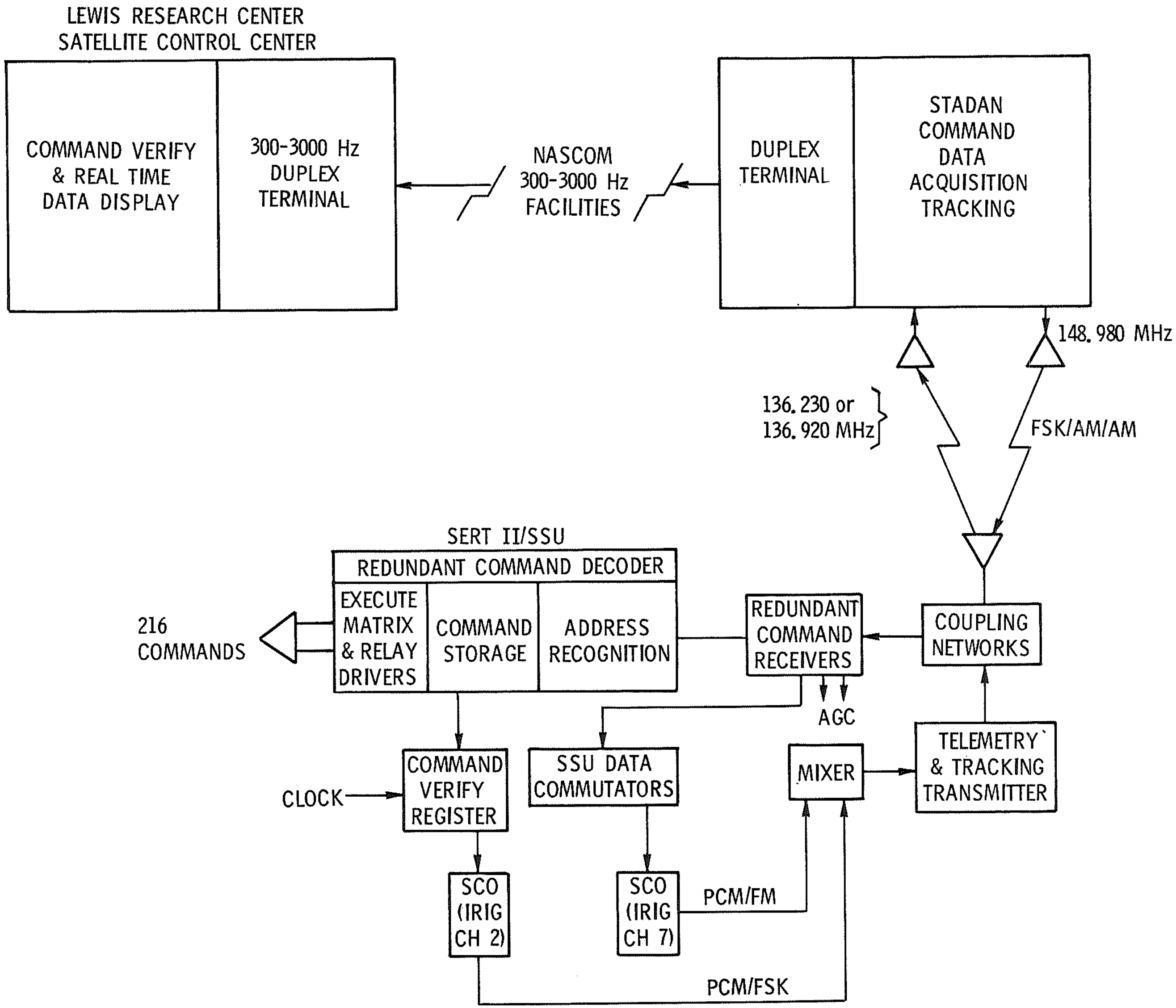


Figure 13. - Command system flow diagram.

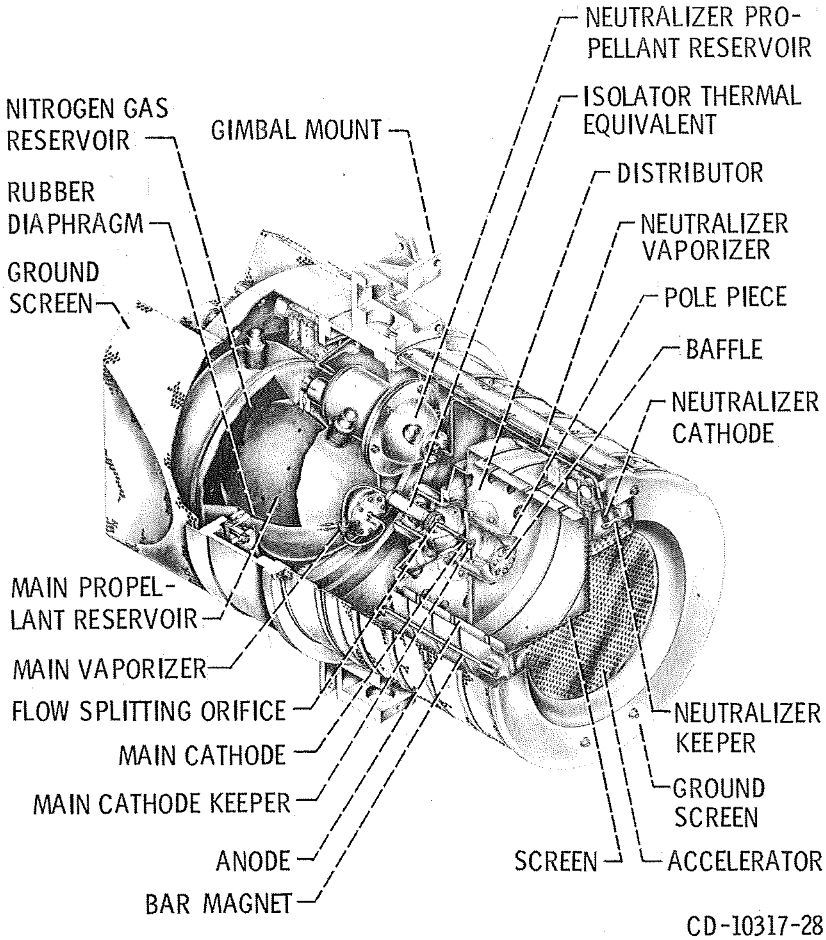


Figure 14. - SERT-II thruster.