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SERT C PROJECT STUDY

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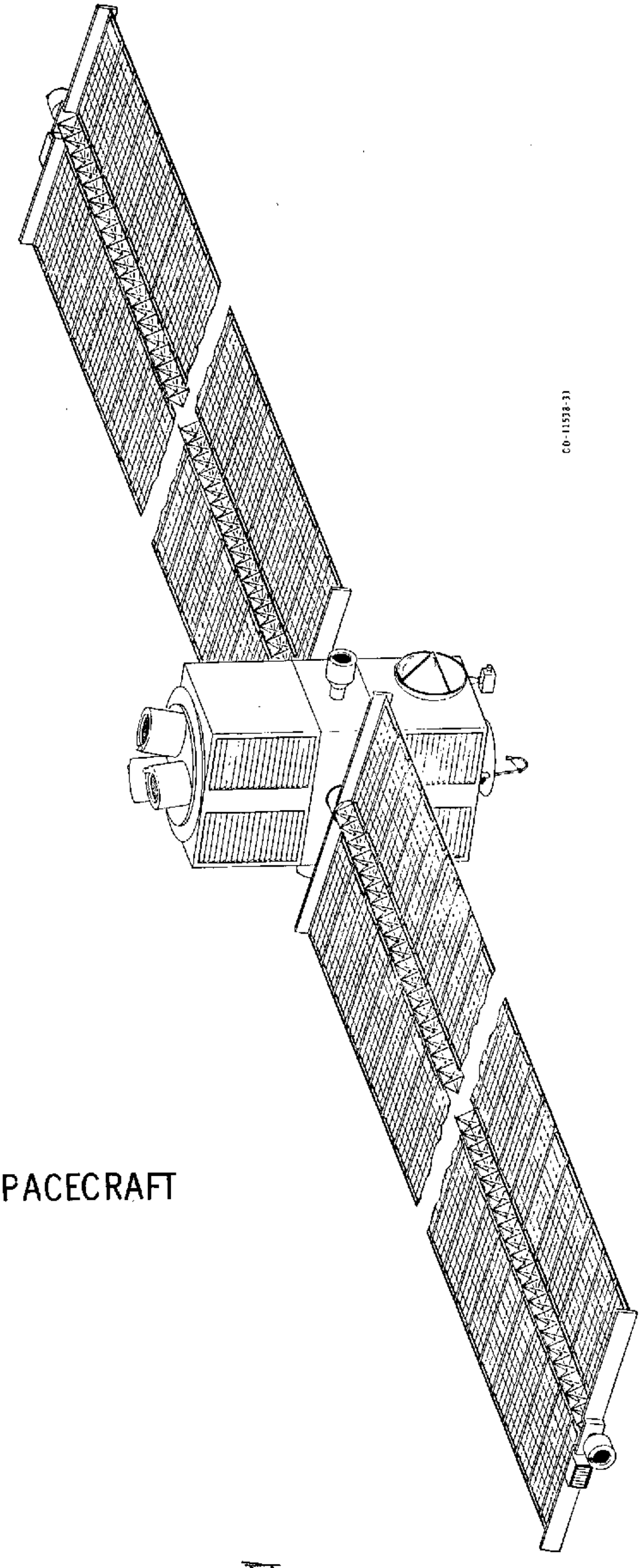

SERT C SPACECRAFT

ABSTRACT

The SERT C (Space Electric Rocket Test - C) project study defines a spacecraft mission that would demonstrate the technology readiness of ion thruster systems for primary propulsion and station keeping applications. As a low cost precursor, SERT C would develop the components and systems required for subsequent Solar Electric Propulsion (SEP) applications. The SERT C mission requirements and preliminary spacecraft and subsystem design are described.

Foreword

This SERT C project study has been prepared by members of the Spacecraft Technology Division who served on the SERT design study team. The study team also included members of the Solar Cell Applications Section and the Electrochemical Technology Section of the Energy Conversion and Materials Science Division. The study team was headed by Elmer H. Davison and Robert C. Finke. The team was assisted by all members of the Spacecraft Technology Division who contributed their advice and information during the course of the project study.

This document has been reviewed by E. H. Davison, Assistant Chief, Spacecraft Technology Division and J. Lazar, Space Propulsion and Power Division, NASA Headquarters. The contributors to the various sections of this report are listed below:

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The contributors to this report would like to acknowledge Linda M. Finke and Pat Miller of the Spacecraft Technology Division for their skill and perseverance in preparing the manuscript of this report.

SERT C

TABLE OF CONTENTS

1.0 PROJECT PROPOSAL

1.1 Introduction

The development of space technology in the United States has seen a steady upward progression in spacecraft size, weight and power requirements. There has also been a comparable increase in cost as might be anticipated. This progression has reached the point of requiring a new generation or class of spacecraft. A few basic technology limitations are dictating the characteristics of this new generation of spacecraft just as they determined the characteristics of their predecessors. In general, these limitations are launch vehicle payload weight and volume capabilities, long-term satellite attitude stability and/or station keeping requirements for synchronous orbit spacecraft and power sources for satellites. Presently, launch vehicle payload weight limitations are not a problem although the cost of alleviating payload weight restraints still determines whether certain missions are practical. The cost of meeting payload weight requirements of some missions is so high as to preclude their accomplishment with available national resources. Some missions could be accomplished, however, with large electric thrusters (30 millipounds of thrust or greater) providing a primary propulsive force for a stage launched into orbit by conventional launch vehicles or as shuttle payloads. At present, however, electric thrusters represent a relatively new untried mode of propulsion. They provide a low thrust of high specific impulse which is applied over a long duration to achieve a delta change in vehicle velocity. By contrast, a velocity change is achieved with a conventional launch vehicle by application of a much higher thrust for a short duration. Many studies have enumerated the benefits to be gained by the use of electric thrusters for certain applications. These benefits can not be described in detail in this proposal. However, it will suffice to note that the high specific impulse of electric thrusters used as a primary propulsion device has been shown to translate into either; (1) greater useful payload weight in orbit or (2) higher inertial velocities for a spacecraft and/or (3) lower mission cost. It is worth noting in the above regard that the energy for the electric thrusters is derived from solar arrays. Hence, in a sense, this energy is free since in space solar cells can convert solar energy directly into electrical energy.

Smaller versions of electric thrusters can also be used to advantage by spacecraft for attitude control and/or station keeping at synchronous attitude as described subsequently. Long-term satellite attitude stability was initially obtained by employing spin stabilization. This stability approach was employed by spacecraft at synchronous altitude in particular, primarily because such spacecraft are used in long duration missions and it was necessary to minimize the propellant required for stability. However, spin stability in conjunction with the use of solar cells as a power source and the restrictions on payload volume seriously restricted the power available to a satellite. Since payload volume limitations are an inherent restriction and solar cells appear to be the power source for the foreseeable future, spacecraft designers are developing lightweight deployable solar arrays to meet increased satellite power requirements. Such arrays cannot be used with spin stabilization techniques, but require three axis stabilized spacecraft with articulated joints between the solar array and spacecraft center body. In addition, as space technology has advanced the reliability and life of subsystems has increased markedly. As a consequence, lifetimes of 5 to 10 years are routinely planned for synchronous orbit spacecraft. Such lifetimes mean that station keeping requirements have assumed much more importance in the sense that propellant weight for the station keeping function becomes a major portion of the spacecraft weight.

The above developments are leading to a new generation of satellites which are in an early development stage. Their evolution however, will require flight demonstration of new technology items such as lightweight deployable arrays, efficient transfer of power across articulated joints, etc. Since the inherent stability of a spinning body can no longer be employed and larger satellite areas result in greater disturbance torques, new attitude control techniques must be demonstrated. The longer mission durations also require new approaches to provide more efficient station keeping techniques. In both instances, the new techniques require thrust devices with high specific impulse to avoid the requirement of allocating an appreciable percentage of satellite weight to propellant for attitude control and station keeping purposes. Cold gas or chemical systems are not suitable for attitude control and station keeping because they have inherent specific impulse limitations that impose large propellant weight requirements on satellites. Electric propulsion devices, on the other hand, can provide at least an order of magnitude increase in specific impulse. For this reason, NASA has actively pursued their development as satellite thrust system devices.

Electric propulsion thrusters in the 1.0 to 30 millipound thrust range have been under development since 1958. The specific impulse of the most extensively developed systems, the Kaufman mercury bombardment ion thruster, is in the 3000 sec range, high enough for the longest planned satellite missions. Over 100,000 hours of development tests have been performed including thruster life tests up to 9715 hours and component life tests exceeding 13,000 hours on these thrusters. In addition, two flight tests (SERT I and SERT II) have been conducted with these thrusters. With this development history behind the mercury bombardment ion thrusters, it appears appropriate to apply their unique capability to the next generation of synchronous satellites and as primary propulsion devices.

The SERT C study described herein addresses the applications of ion thrusters and other technology to a new generation of spacecraft. Because the proposed SERT C satellite represents a new generation of technology to be employed in spacecraft, it is appropriate at this stage of development to demonstrate the technology and concepts required in such a way as to standardize and lower the cost of future satellites.

The spiral-out mission proposed is an effective low cost means for demonstration of the new technology incorporated in SERT-C. It allows a class of spacecraft (9 kW solar array) of interest to subsequent missions to be launched with a low cost Thor/Delta vehicle. Further, the spiral-out mission provides, in near-earth orbits, the flight durations and operational sequences applicable to subsequent missions.

1.2 Mission Objectives

The SERT C mission has three broad basic objectives. These are:

1) to demonstrate technology readiness of electric thruster systems for primary propulsion and station keeping applications,
2) to demonstrate guidance, control and operational techniques to effect trajectory shaping, rendezvous and station keeping with Solar Electric Propulsion (SEP) spacecraft

3) to develop via a low cost precursor mission, the components and systems required for subsequent SEP applications.

A discussion of the way SERT C supports the above objectives follows.

Thruster Systems - Ion thrusters have been under development by the LeRC and other organizations for a number of years, yet their practical application remains to be demonstrated. The thrusters under development fall into two classes. Large thrusters having thrust levels of the order of 30 millipounds and small thrusters having thrust levels of the order of 1.0 millipounds. The large thrusters are suitable for Solar Electric Propulsion (SEP) missions wherein they are used to spiral spacecraft from low earth orbits to synchronous altitude or escape velocities and for interplanetary missions. Such use provides significant increases in useful payload weight, lowers overall mission cost and/or permits missions that cannot be accomplished by other means. Small thrusters are used for spacecraft attitude control and station keeping at synchronous altitude. Their use for these purposes for spacecraft that must operate for many years with precise control provides significant increases in useful payload weight and can lower overall mission cost.

SERT C will provide a practical in-line demonstration of both classes of thrusters in applications having great significance to future NASA and commercial spacecraft. The large thrusters will be used to spiral SERT C from the launch vehicle injection orbit of approximately 3000 km altitude to synchronous altitude and remove the injection orbit inclination. At synchronous altitude these thrusters will be used to accomplish rendezvous with another satellite. Such applications have never been demonstrated and would represent a low cost precursor to such missions as SEP.

ATS-F will fly small ion thrusters that will be used for demonstration of north-south station keeping and attitude control at synchronous altitude. This initial experiment will be valuable in promoting the use of ion engines. However, the demonstration is short-term and experimental in nature and, therefore, will not provide the practical demonstration of long-term operation of small ion thrusters required for future spacecraft. In contrast, SERT C will demonstrate the practicality of using small ion thrusters as in-line functional elements of both an attitude control system during a spiral-out mission, and the attitude control and station keeping system for long duration missions at synchronous altitude. Thus, the successful performance of SERT C will greatly enhance the practical application of ion thrusters to future spacecraft.

Guidance and Control - Ion thrusters represent a new mode of propulsion. A low thrust of high specific impulse is applied over a long duration to achieve a delta change in vehicle velocity which, by contrast, is achieved with a launch vehicle by application of a high thrust for a short duration. As a consequence, new thrust vector control and operational procedures are required. SERT C will develop and demonstrate these procedures and make them available for subsequent missions. These procedures will make use of the capability of ion thrusters to trade mission time against payload weight in planning and executing a mission.

SERT C will be used to demonstrate the use of an ion thruster subsystem to station walk the spacecraft from one synchronous orbit longitude to another. The longitude adjustment and inclination change maneuvering capability provided by the ion thruster subsystem will allow a rendezvous mission to be performed.

Ground tracking stations will command the ion thruster subsystem to relocate the spacecraft into proximity with an existing synchronous satellite. Then, on-board radar will be exercised to locate and pinpoint this satellite's position, allowing SERT C to be positioned close enough to perform a visual inspection by means of an on-board television system.

Ion thrusters will provide a very fine degree of control, allowing a close approach for a minute inspection of the satellite. The orbits of SERT C and the satellite will be closely matched to provide ample time for a thorough inspection of the satellite.

Precursor Systems - Several applications of large thrusters as propulsive devices are being pursued by NASA. The SEP effort by JPL, MSFC and LeRC has been under way for several years. Shuttle Tug applications have been proposed by the MSFC. These efforts will commit major NASA resources when they become approved projects. Their cost will be much greater than the estimated cost of SERT C. Under these circumstances, SERT C will serve as a low cost precursor that will provide vital engineering data needed for the successful execution of future major NASA projects.

There exists also a real requirement to demonstrate using small ion thrusters, a low cost precision station keeping and attitude control capability for a synchronous orbit spacecraft having sun oriented solar arrays and an earth oriented center body. Such a configuration and capability has many advantages over existing synchronous orbit spacecraft. A sun oriented solar array minimizes the solar cell area required to perform mission functions and hence reduces the spacecraft size and cost. Precise station keeping of the earth oriented center body greatly simplifies the ground communication equipment required to make use of the synchronous orbit platform. The simplification of ground equipment and attendant lower cost will greatly expand the range of domestic and foreign user organizations that can afford a synchronous orbit platform. The Communications Technology Satellite (CTS) will provide a first step in the required development of the new generation of spacecraft, but its planned station keeping capability (± 1 deg. north-south, ± 0.2 deg. east-west) and attitude control (± 0.1 deg. in pitch and roll, 1.1 deg. in yaw) can be improved upon significantly. The proposed SERT C would provide a demonstration of this improved capability with ± 0.05 deg. north-south, 0.10 deg. east-west, 0.08 deg. pitch and roll, and 0.2 deg. yaw. In addition, SERT C would represent a NASA technology development readily available to U.S. industry and government users.

A major item of cost in developing and constructing a precisely stabilized spacecraft is associated with the attitude sensing systems. Most three-axis stabilized spacecraft employ either star trackers, interferometers (not yet flight tested) or both for precise sensing of spacecraft attitude. Both systems are extremely expensive and the interferometer systems require ground station support to function. The proposed SERT C sensing system eliminates the need for either a star tracking or interferometer system and relies on inexpensive rate integrating gyros and sun sensors for the critical yaw sensing requirements. Demonstrations of the performance and reliability of the SERT C system would permit cost reductions in future synchronous orbit spacecraft.

Availability of a solar array orientation mechanism (SAOM) and power/signal transfer slip ring system is crucial to the development of spacecraft with solar arrays that must rotate relative to the center body to remain sun oriented. Systems have been under development for a number of years within NASA, DOD and industry that circumvent the problems of existing systems. For example, liquid metal slip rings (LMSR) have been developed at the LeRC and under contract that solve the wear debris problem and eliminate the stiction problem associated with conventional slip rings. This technology must be demonstrated in an actual spacecraft, however, before it will find acceptance and receive general application to future spacecraft. This LMSR technology as well as the latest SAOM technology would be demonstrated on the proposed SERT C.

Solar arrays using silicon solar cells are essentially the exclusive source of electrical power for earth satellites. However, the cells have basic limitations associated with them in that:

1) the cell's power is developed at low voltage, approximately 0.5 volts per cell;
2) the cell's power degrades with time in the radiation environment of space; and
3) the cell's power output is sensitive to cell temperature variations.

As a result of the above limitations, the cells have only been used to date as a raw source of low voltage power. Unfortunately, this low voltage power does not meet most spacecraft electrical load requirements. As a consequence, this low voltage power must be conditioned using complex power conditioning equipment.

This power conditioning equipment is also heavy and its inefficiency results in significant spacecraft thermal, structural and other design problems as noted in the body of this proposal. For many electrical loads, this power conditioning equipment can be eliminated by using directly regulated solar array power conditioning techniques pioneered by the LeRC. These techniques consist essentially of configuring solar cells in the required series/ parallel groupings such that the electrical load voltage and current requirements are satisfied. Power variations in the cells due to temperature and radiation effects are compensated for by shorting switches on the array that can be used to control the power output of the cell grouping in a controlled fashion. Protection against cell or interconnect failures is provided by diodes in parallel with cell groupings. These advanced power control techniques would be demonstrated on SERT C for appropriate thruster electrical loads. Such a demonstration would provide a significant step forward in spacecraft power system technology.

An advanced solar power system will be demonstrated. The higher efficiency violet solar cells will be used to reduce solar array size and weight. The multi-kilowatt deployable array will represent a substantial increase over present power levels. This array will make use of advanced Astromast deployment boom techniques, cell mounting techniques and blanket fabrication techniques. Major portions of array power will be developed at voltages up to 400 volts to improve overall power system efficiency and reduce its weight. Other advanced power components such as batteries and power conditioners will be demonstrated. The totality of these advanced components and techniques will represent a major advance in solar power systems.

A considerable concern to SEP missions is the magnitude of the ion thruster/experiment interactions and thruster Hg efflux contamination levels of experiments and spacecraft components. These interactions and contamination levels are not easily predicted because they are of small magnitude and too complex to analyze properly. Also, in some instances, long time periods would be required to detect measurable effects. SERT C will provide an in situ opportunity to evaluate these effects and provide engineering and experiment design data for subsequent missions.

Other Benefits - The spacecraft will provide a platform for other experiments. These experiments would be defined based on proposals received after project approval. Obvious candidates are advanced solar cells, array fabrication techniques and follow-on SPHINX experiments. Some scientific experiments could be accommodated and it has already been proposed by Dr. R. G. Cruddace of the University of California that SERT C carry an extreme ultraviolet telescope experiment.

However, all proposed experiments must be scrutinized carefully to determine acceptability. They should in no way impact the primary purposes of SERT C and would be carried on a non-interference, weight permitting basis. Further, the cost associated with such experiments and their integration into the spacecraft would have to be borne by the experimenter.

A significant benefit to be obtained from the SERT C development is that NASA will have an electric thruster propulsion module or stage technology available for a wide variety of applications. The ability of the SERT C stage to change orbit parameters such as altitude and inclination by applying low thrust at high specific impulse over long periods of time will open up a new operating dimension for propulsion vehicles. Thus, SERT C will represent a versatile propulsion stage that will expand NASA performance capability significantly. For some missions it can also significantly lower total mission cost.

Another major benefit to be obtained from the development of SERT C is that NASA would have available for future applications a means of delivering large high power spacecraft to synchronous orbit cheaply. Such spacecraft could be placed in orbit in the future by their own launch vehicles or could be launched as shuttle/tug payloads.

As noted in the introduction, a new generation of spacecraft is developing. The various technologies required by this new generation of spacecraft have been developed, but have not been combined into flight proven spacecraft. In executing a SERT C Project, flight proof of the concepts and technologies required by the next generation of spacecraft will be accomplished.

1.3 Baseline Spacecraft Configuration

The SERT C baseline configuration is shown in figures 1.3.1 and 1.3.2. Figure 1.3.3 shows the nominal attitude of the spacecraft during the orbit raising and on orbit phases of the mission.

During the orbit raising phase, the center body face containing the 3 - 30 cm ion thrusters is directed nominally west, and the face containing the high gain communications antenna and the rendezvous TV camera is oriented nominally east. The two faces from which the solar arrays deploy are oriented north and south, so the axis of array rotation is nominally north-south. The face directed toward earth contains a single 8 cm ion thruster, while the face containing the 8 cm thruster and the radar antenna will be directed away from earth. Once synchronous orbit is attained, the spacecraft is rotated 90° so that the high gain communications antenna faces earth and the two body mounted 8 cm thrusters face east and west. As shown in figure 1.3.3 the faces from which the solar arrays are deployed face north and south for both mission phases, and were chosen as the thermal control surfaces. Note also that an 8 cm thruster and its power conditioning equipment are mounted on the outboard tip of each solar array. The location of the TV camera was chosen so that a solar array could be observed.

The SERT C spacecraft, the Delta payload envelope, and the 5724 attach fitting are shown in the launch configuration in figure 1.3.4. The 5724 attach fitting is 24 inches high, and approximately 57 inches in diameter and weighs 83 pounds. It is capable of supporting a 2500 pound spacecraft. The spacecraft is fastened to the attach fitting by means of a Marmon type clamp. The center of gravity of the spacecraft will be 56 inches above the separation plane. The center of gravity limitation is based on spacecraft size, mass, and flexibility. The launch vehicle control system requires that the spacecraft natural frequency be above 15 hertz. The SERT C spacecraft will be approximately 1800 pounds, which is well below the 2500 pound adapter limit, and will have a natural frequency above 15 hertz.

The center body size (3.3' x 4.5' x 9') was determined from requirements imposed by the Delta dynamic shroud, the stowed solar array configuration, and spacecraft thermal constraints. The 9 foot dimension was chosen to give maximum width to the split blanket, solar arrays (9' x 53' when deployed) while holding spacecraft c.g. to 56 inches above the separation plane. The 3.3 foot dimension was chosen to allow both solar arrays, the deployment mechanisms (Astromasts), the orientation mechanisms and the two 8 cm thrusters to be in line within the shroud. The 4.5 foot dimension gives the required north-south face area to dissipate expected heat loads.

During launch the solar arrays are held in the stowed position by a pressure plate and strap mechanism. Solar array deployment is initiated by explosively releasing the strap mechanism. A bar mechanism is then used to elevate the back pallet of the array away from the spacecraft to increase the thermal view factor. The Astromasts complete the deployment of the split blanket arrays.

A bar mechanism will also be used to elevate the radar antenna assembly from the spacecraft. This will allow radar coverage of $\pm 90^\circ$ in two axes. The structure will be fabricated from aluminum structural and sheet metal parts wherever possible. Riveting will be the standard method of joining the structural members. The cover plates will be screwed to the structure. They will be aluminum sheet or honeycomb depending on whether they are used as a cover or as a heat radiating surface with components mounted on the inner face. Figure 1.3.4 also shows the locations of the various subsystems. Table 1.3.1 presents the current subsystem weight estimates.

Note from figure 1.3.3 that rotation of the spacecraft about an earth radius vector is referred to as yaw motion, rotation about an axis perpendicular to the orbit plane is referred to as pitch motion, and rotation about an axis in the orbit plane and perpendicular to the earth radius vector is referred to as roll motion. These designations will hold in subsequent sections of this proposal for both on-orbit and orbit-raising spacecraft attitudes.

In defining subsystem requirements or mission capabilities conservative assumptions were made regarding hardware components. The primary assumption in this regard was that conversion efficiency for solar array power to thruster power was 85 percent. Such an assumption results in conservative (longer) spiral out times, conservative (larger) thermal rejection area requirements, conservative solar array area requirements and conservative component weights.

TABLE 1.3.1 - SERT C SUBSYSTEM WEIGHT ESTIMATES

3 30-cm Thrusters	51 lbs
Thruster Array Platform	10
Propellant Tanks (30 cm and body mounted 8 cm)	30
Hg Propellant (30 cm-thrusters)	300
3 30-cm Power Processors	105
Cabling from 30-cm Thruster to Power Processors	15
4 8-cm Thrusters	20
Propellant Tanks (2 array mounted 8-cm thrusters)	6
Hg Propellant (attitude control and station keeping)	46
4 8-cm Power Processors	35
Attitude Control	94
Structure	355
Thermal System	125
Solar Array (blanket and deployment mechanism)	340
Solar Array Orientation Mechanism	24
Power System and Harness	95
Battery ($Ni-H_2$)	14
Battery (Ni-Cd)	45
Telemetry and Command	45
Inspection Television	28
Rendezvous Radar	76
Computer	14
TOTAL	1873 lbs

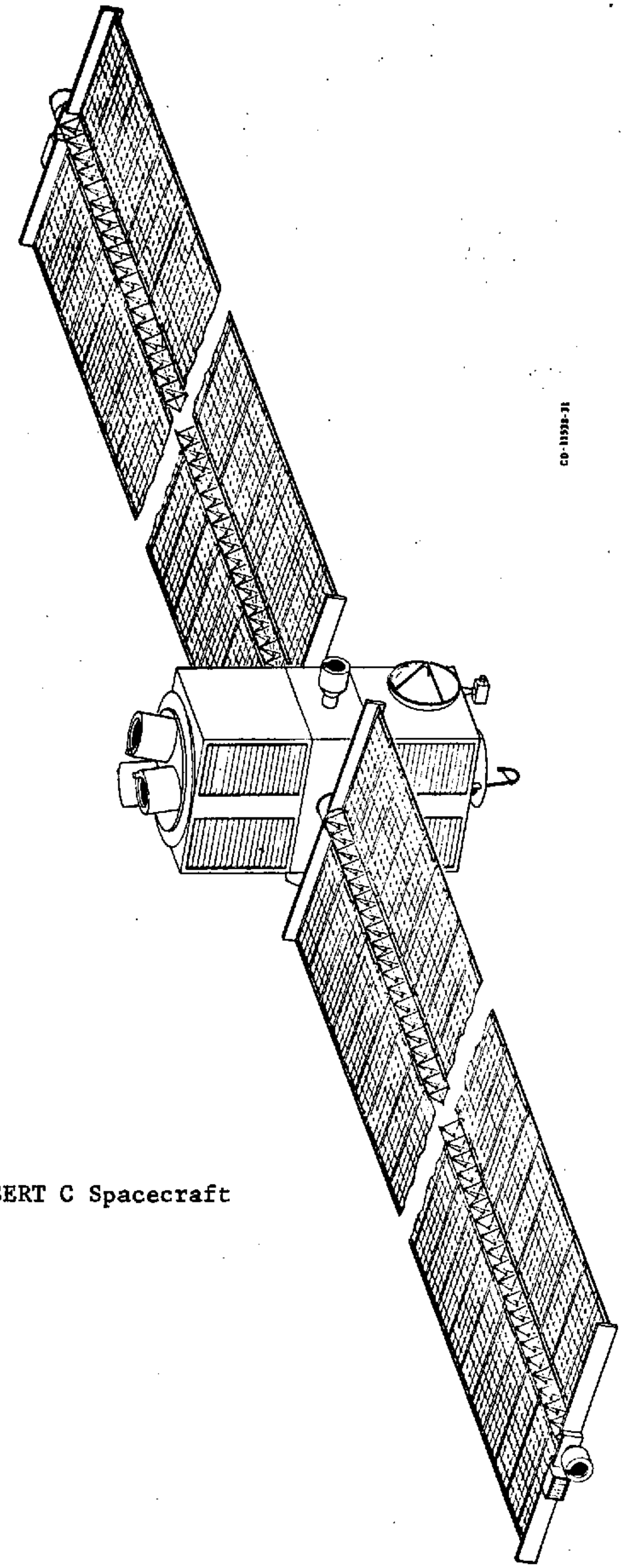


Figure 1.3.1 - SERT C Spacecraft

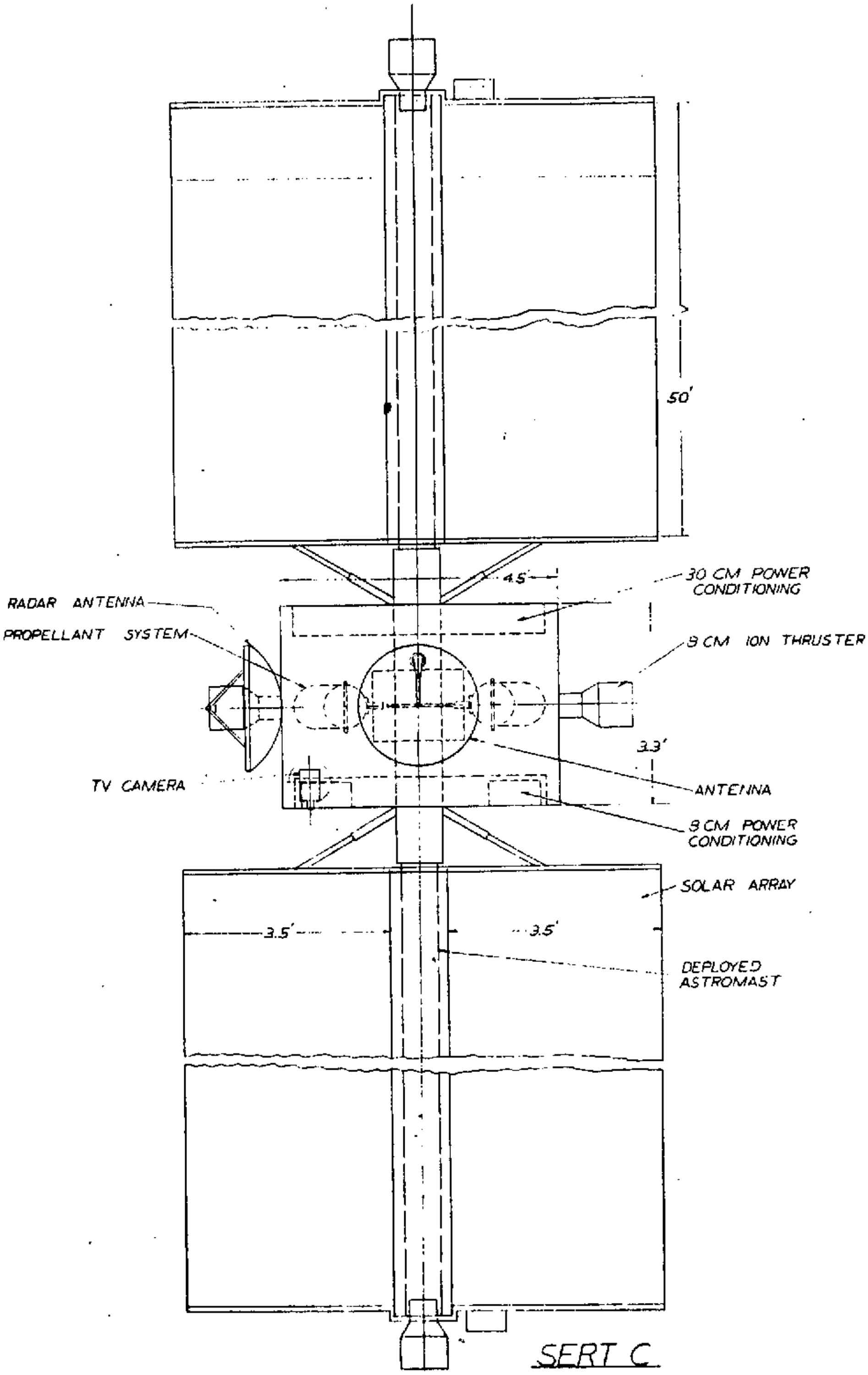


Figure 1.3.2 - SERT C Deployed Configuration

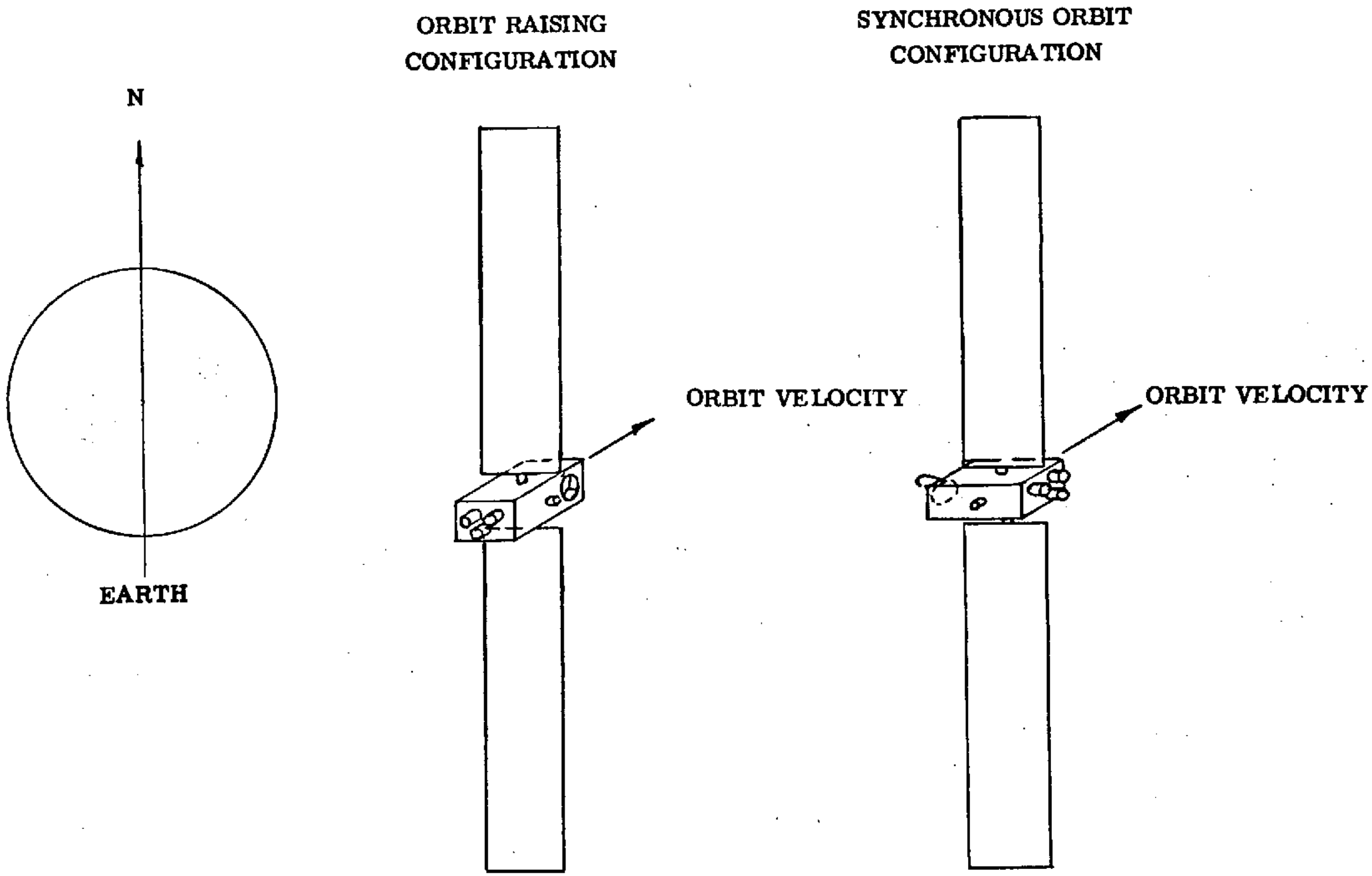


Figure 1.3.3 - SERT C

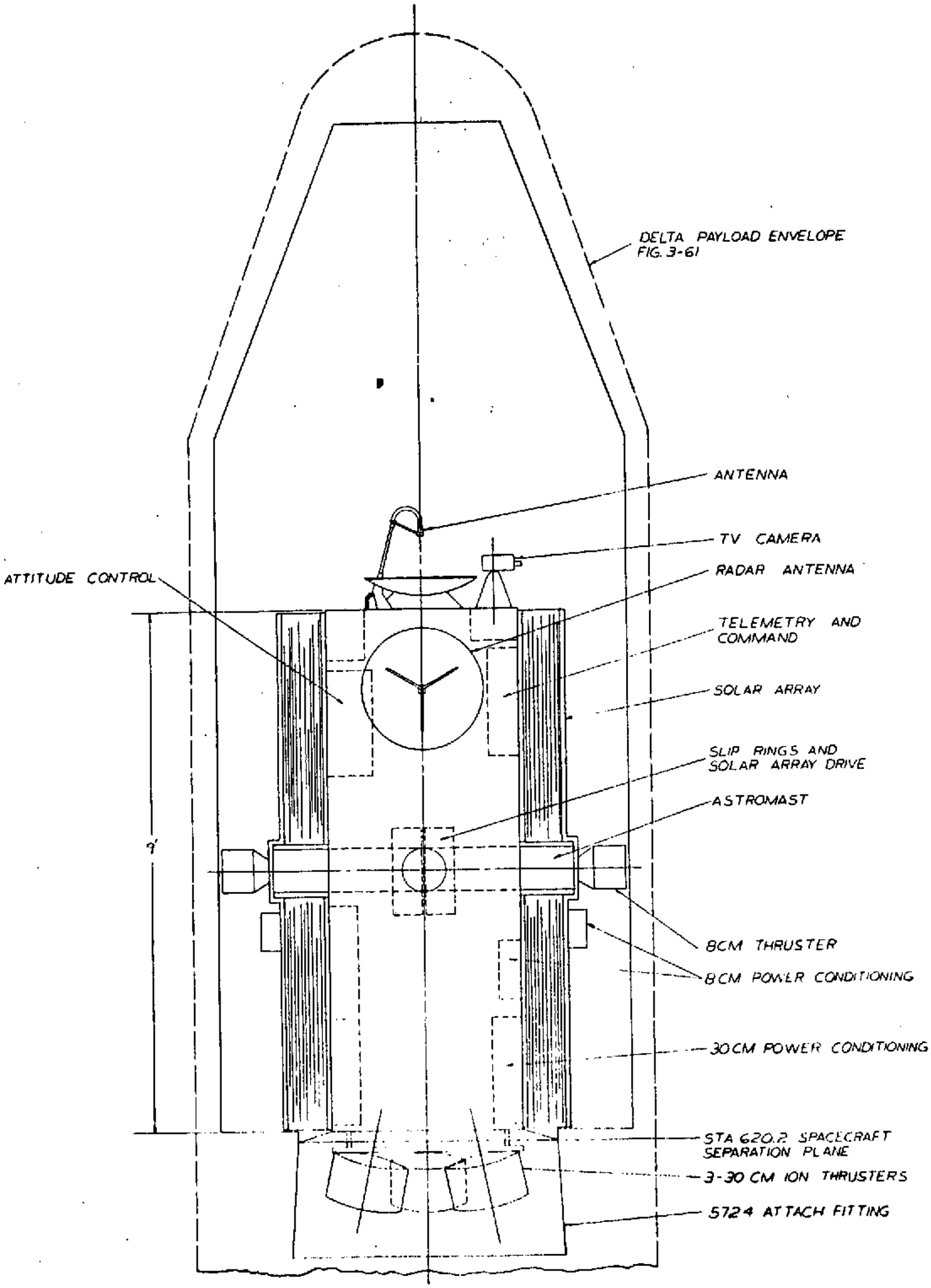


Figure 1.3.4 - SERT C Stowed Configuration

1.4 System/Subsystem Trade-Off Studies

The baseline spacecraft description identified in the previous section was selected from four configuration options considered during the project study. The configurations were compared on the basis of the accomplishment of the mission objectives, simplicity of the subsystems, reliability, use of state-of-the-art hardware, and ease in integrating the subsystems into the spacecraft. Figures 1.4.1 through 1.4.4 show the four configurations. The positive or negative features of the spacecraft or subsystems of each configuration are shown in figure 1.4.5.

In configuration II, the same spacecraft face is directed toward earth in both the orbit raising and synchronous orbit phases. Therefore, the high gain antenna is always directed at earth and the station walking operations are simplified. The disadvantage is that because the 30 cm thrusters are located on the west face, the only available location for an east-west station keeping thruster is on the east face. The accomplishment of station keeping with this arrangement becomes more complex and is not representative of the station keeping operations which will be required by future satellites. Additionally, the location of the high gain antenna within the existing shroud envelope prohibits any growth in the antenna size.

In configuration III, all of the 8 cm thrusters are located on the centerbody to decrease the possibility of solar array interaction with the attitude control system. However, by doing this the thrusters are located in positions where the thruster plumes might contaminate the solar array surfaces. The 8 cm thrust directions require that two thrusters be operated simultaneously, thus increasing the lifetime required of the thrusters.

For the first three configurations, the solar arrays are not always normal to the sunline during the orbit raising phase. Configuration IV maintains the solar arrays normal to the sunline at all times by adding a second articulation. The centerbody is divided into a rotating hub and a nonrotating section. A disadvantage is that the extra rotating joint, orientation device, and slip rings impact the spacecraft reliability. Additionally, east-west station keeping operations require rotations about yaw to provide the proper thrust directions. There are potential problems in the design of the thermal and attitude control systems.

Each of the configurations investigated had advantages and disadvantages considering both the accomplishment of the mission objectives and the development of the spacecraft systems. The baseline, configuration I, provides for the best demonstration of synchronous orbit attitude control and station keeping with ion thrusters. The spacecraft development appears to be simple compared to that for the other three configurations, each of which has potential development problems. Balancing the advantages and disadvantages of the configurations considered led to the selection of configuration I as the baseline for SERT C mission.

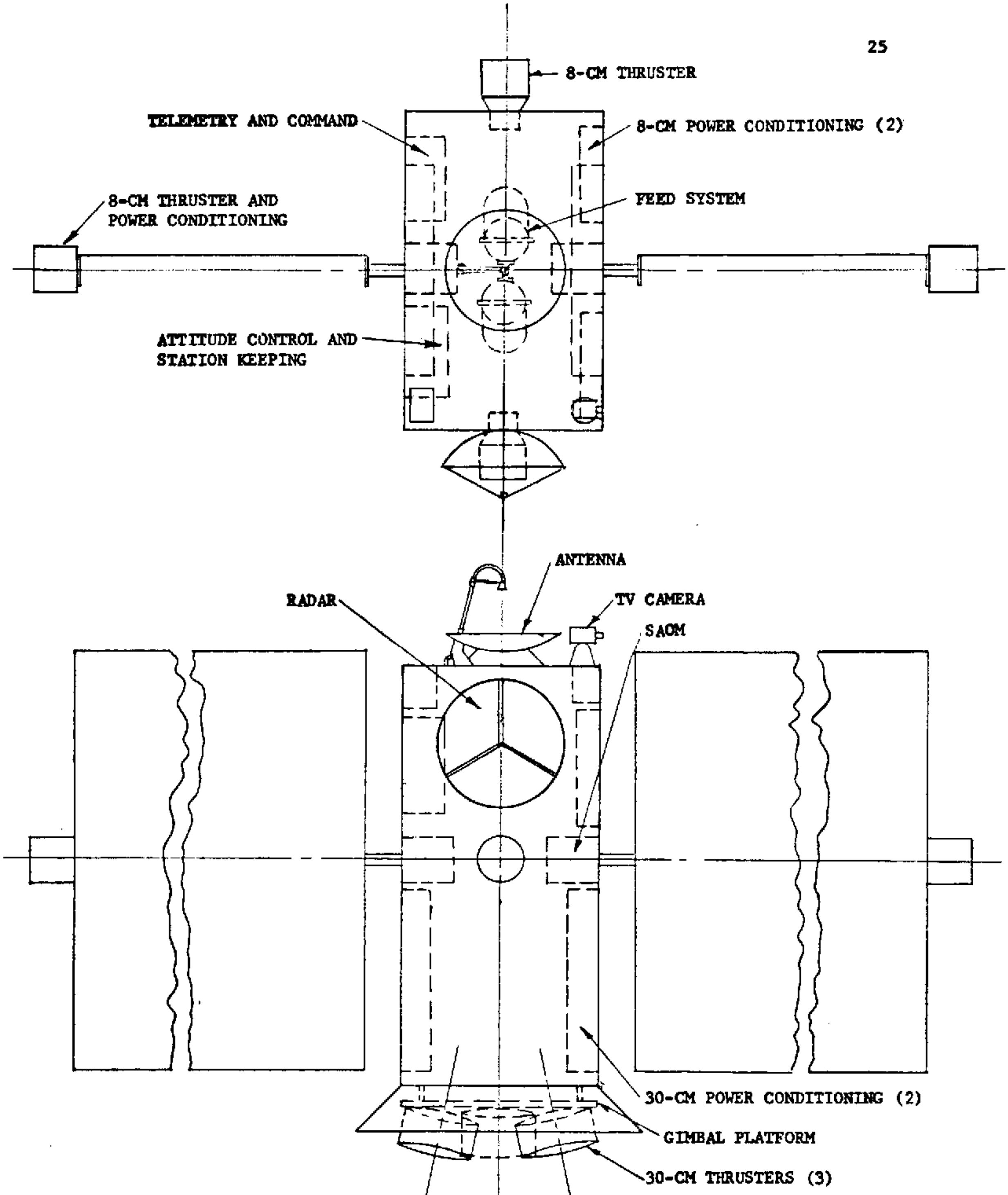


Figure 1.4.1 - SERT C Configuration I

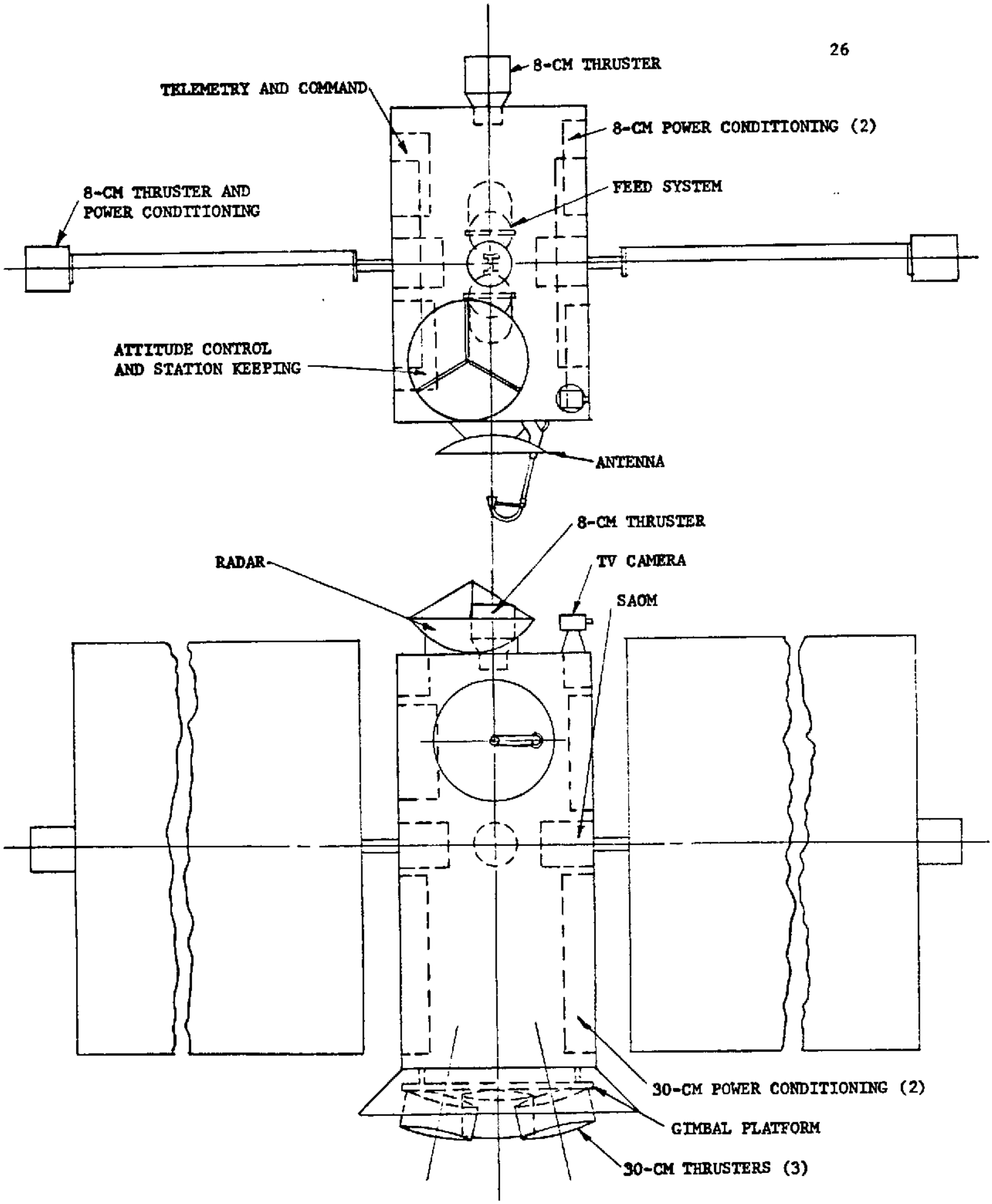


Figure 1.4.2 - Configuration II

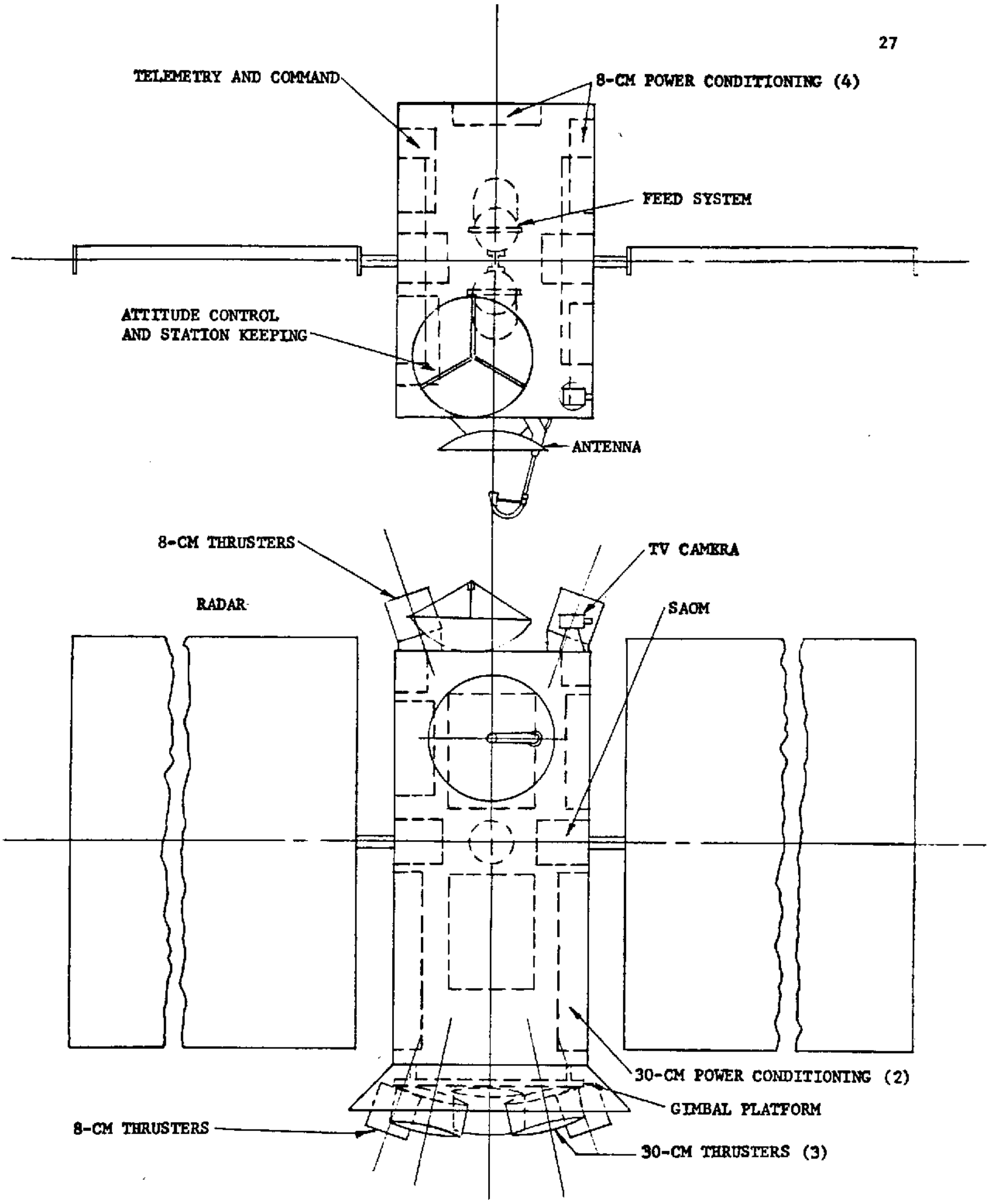


Figure 1.4.3 - Configuration III

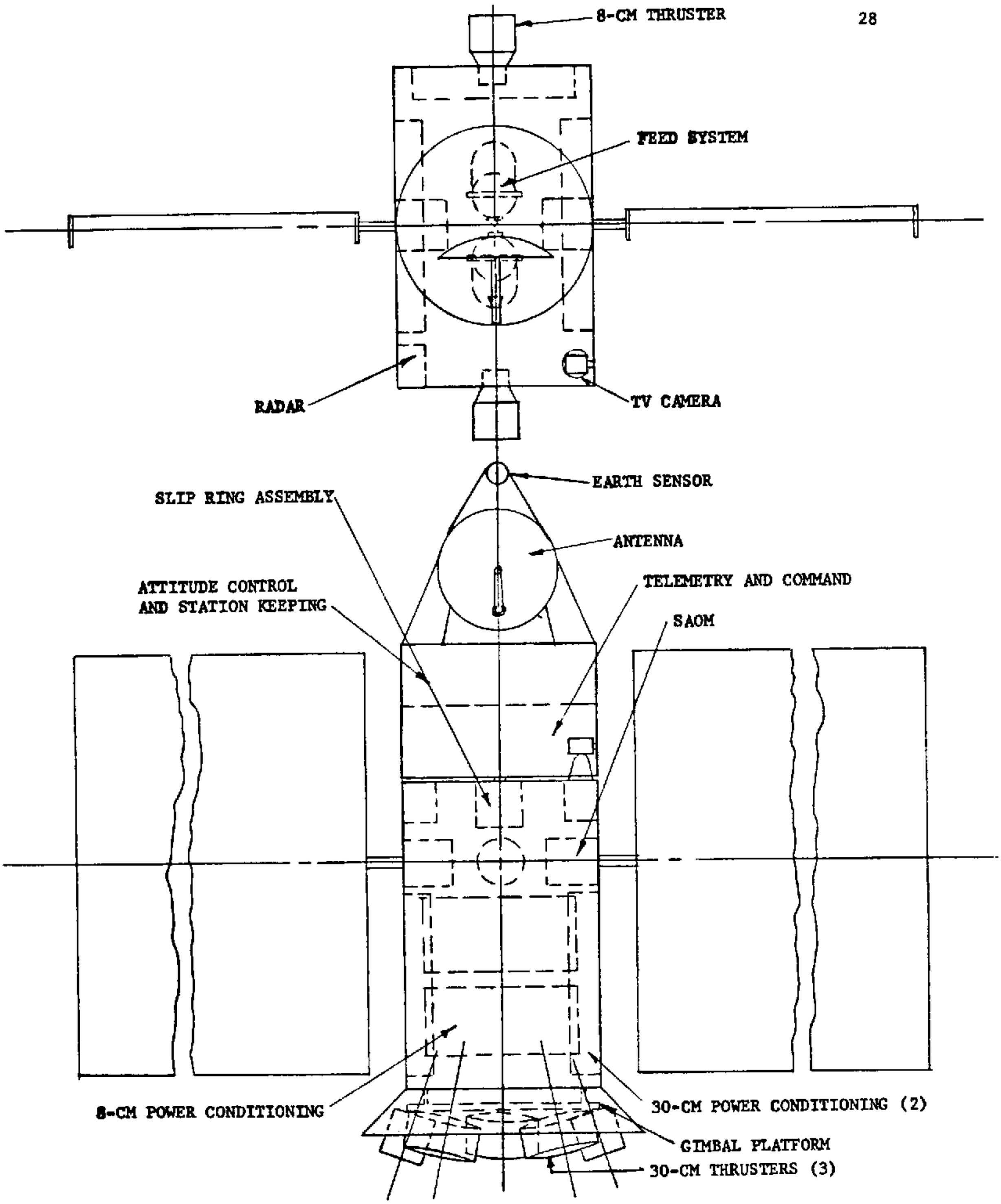


Figure 1.4.4 - Configuration IV

S/C Configuration		30-cm Thrusters	8-cm Thrusters	Solar Array	Attitude Control System
I. North-south array with array-mounted 8-cm thrusters and rotated S/C body	+		Ideal thruster geometry for N-S, E-W station keeping. Best station walking configuration for back up.	Minimum thruster contamination of solar array	Larger control torque (i.e., moment arm). Maximum 8-cm thruster function redundancy.
	-	Center body rotation required for station walking	Two thrusters required if life test required.	Increased structure weight	Array flexibility considerations more critical. Configuration makes two horizon sensors mandatory.
II. North-south array with array-mounted 8-cm thrusters	+	Best station-walking configuration	Ideal thruster geometry for N-S station keeping and life test. Life test requires only one thruster.	Minimum thruster contamination of solar array	Larger control torque (i.e., moment arm). Maximum 8-cm thruster function redundancy.
			Poor geometry for east-west station keeping	Increased structure weight	Array flexibility considerations more critical
III. North-south array with body-mounted 8-cm thrusters	+	Best station-walking configuration	All thrusters can use common tankage. All thrusters are body mounted		
			Poor thrusting angle, minimum redundancy. Does not take full advantage of electric propulsion capability. Two thrusters must operate simultaneously for N-S, E-W station keeping and momentum dumping	Thruster contamination. Higher aspect ratio needed.	Reaction wheel dump requires two thrusters; small moment arms for dumping. Decreases array resonance frequency
IV. East-West array	+	Shortest orbit raising time	All thrusters can use common tankage. All thrusters body-mounted.	Maximizes seasonal array output. No thruster interactions with array dynamics.	Provides 100% power in transfer.
8-cm thrusters 30-cm thrusters			Rotating body (with respect to fixed space) causes complicated N-S and E-W station keeping.	Higher aspect ratio needed	Earth sensor and high gain antenna design complicated. Yaw sensing complicated. Decreases array resonance frequency.

Figure 1.4.5 - Configuration Tradeoffs

Mission	T T & C	Thermal	Structure	Power	Experiments
Best antenna and TV camera configuration for rendezvous		Good thermal geometry			
Requires pitch maneuver to achieve synchronous orbit orientation.	High gain antenna not operational during transfer orbit	Array thruster thermal design required	Array thruster mounting design required	Power loss due to solar incidence angle	
	High gain antenna used for both orbit raising and synchronous orbit	Good thermal geometry			
	P/L envelope limits antenna growth	Array thruster thermal design required	Array thruster mounting design required	Power loss due to solar incidence angle	
	Minimum slip rings requirement	Least complex thermal design			Minimum slip ring requirement with all thrusters on the centerbody
Thruster life testing more complex			Requires thruster beam divergence shroud	Power less due to solar incidence angle	Least favorable configuration for directly regulated solar array experiment
Maximum available power			Array joints do not have to work after reaching geosync.	Less power loss due to solar incidence angle	
Requires roll control program for maximum power. More complex attitude control requirements. Stringent sensor platform rigidity requirements may require TV and radar mounted on stable platform. Requires yaw maneuver at synchronous orbit.	Maximum slip rings requirement.	Unusual thermal configuration	Requires a third rotating joint. Long, rigid structure required to mount Earth sensor and high gain antenna. May require beam divergence sputter shroud.	Complex control required to achieve maximum power	Requires an additional orientation and slip ring system. Worst configuration for directly regulated solar array experiment.

1.5 Launch Vehicle

The basic vehicle for use in this mission is the two-stage Thor Delta vehicle model 2910. An outline drawing of the vehicle is shown in figure 1.5.1.

The extended tank Thor first stage is 73 feet 4 inches long and uses the Rocketdyne H-1 main engine developing 205,000 pounds thrust. The fuel is RJ1 and the oxidizer is LOX. The main engine is gimbal-mounted to provide pitch and yaw control from lift-off to main engine cut off (MECO). Two liquid-propellant vernier engines provide roll control throughout first-stage operation, and pitch and yaw control from MECO to first-stage separation. Nine Castor II solid propellant motors of 52,000 pounds thrust each (burn time 38.7 sec) provide additional propulsive force. Guidance is provided by the second stage.

The second stage is the 8 foot diameter Delta employing the TRW TR-201 pressure fed gimballed engine and a new Delta Inertial Guidance System (DIGS) with a digital computer. Fuel for this stage is Aerozene 50 and the oxidizer is nitrogen-tetroxide, developing a thrust of 9600 pounds. Pitch and yaw control is provided through second stage burn.

A nitrogen gas system using 8 fixed nozzles provides roll control during powered and coast flight as well as pitch and yaw control after second-stage cut off (SECO).

The payload capability of the Delta 2910 into a circular parking orbit, inclined at 28.3 degrees is given in figure 1.5.2.

The payload envelope for the Delta 2910 with the 5724 attach fitting is shown in figure 1.5.3. The attach fitting is 24 inches high, approximately 57 inches in diameter, and weighs 83 pounds. The launch vehicle control system requires that the spacecraft natural frequency is above 15 Hertz.

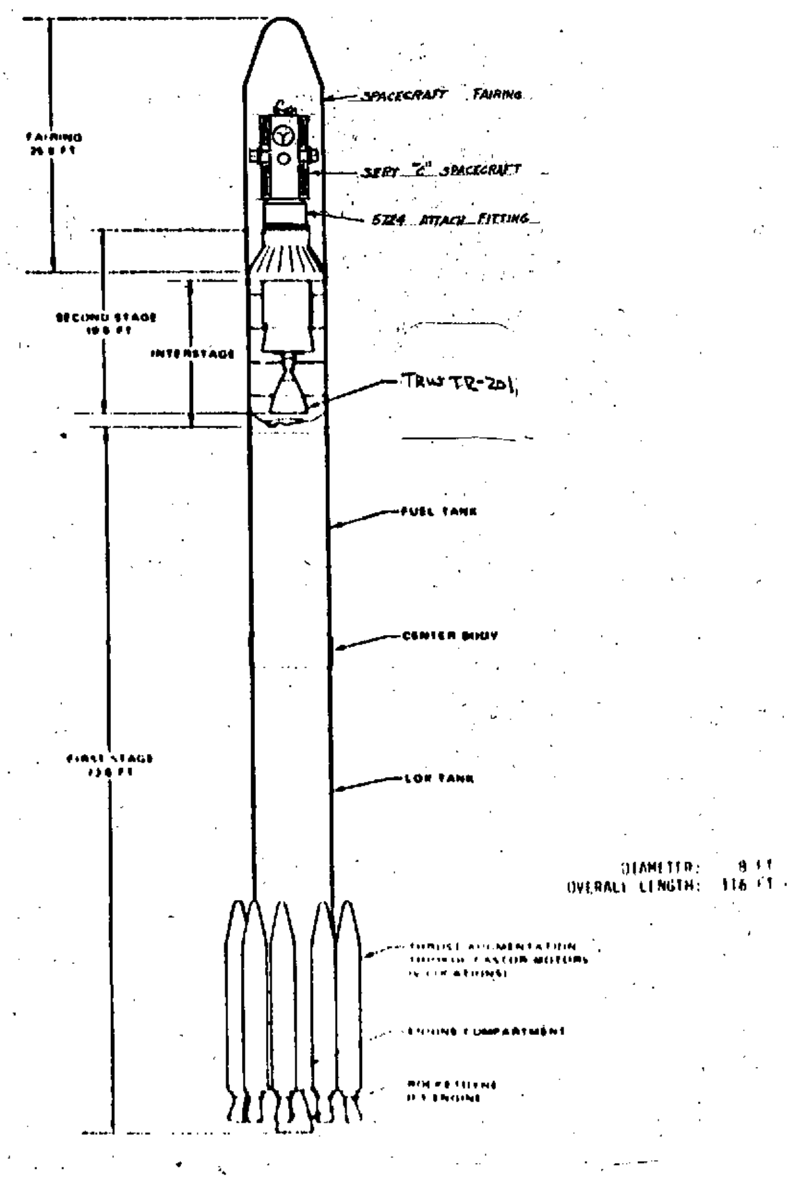


Figure 1.5.1 - Delta 2910 Launch Vehicle

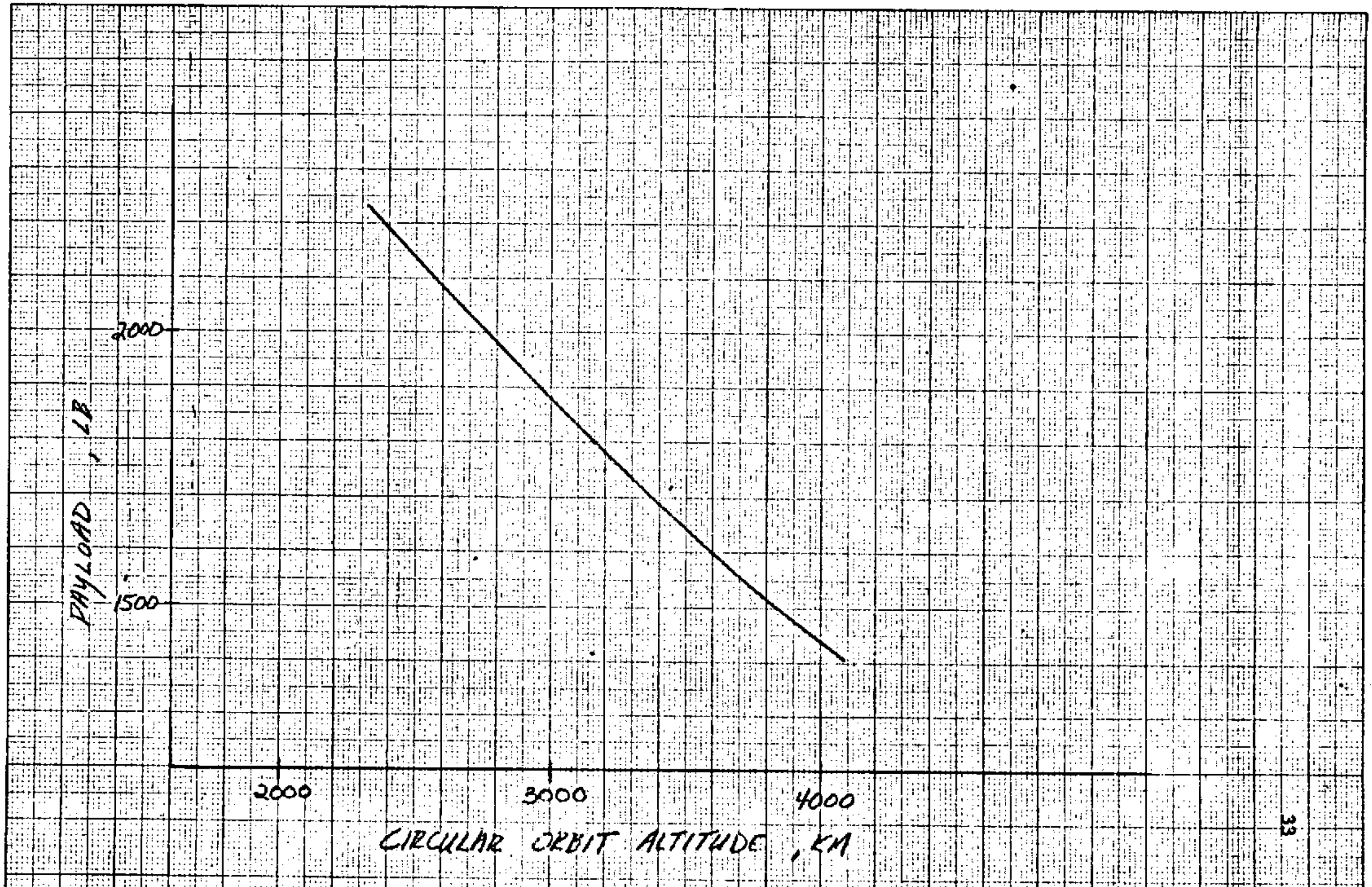


Figure 1.5.2 - Circular Orbit Capability for Delta 2910

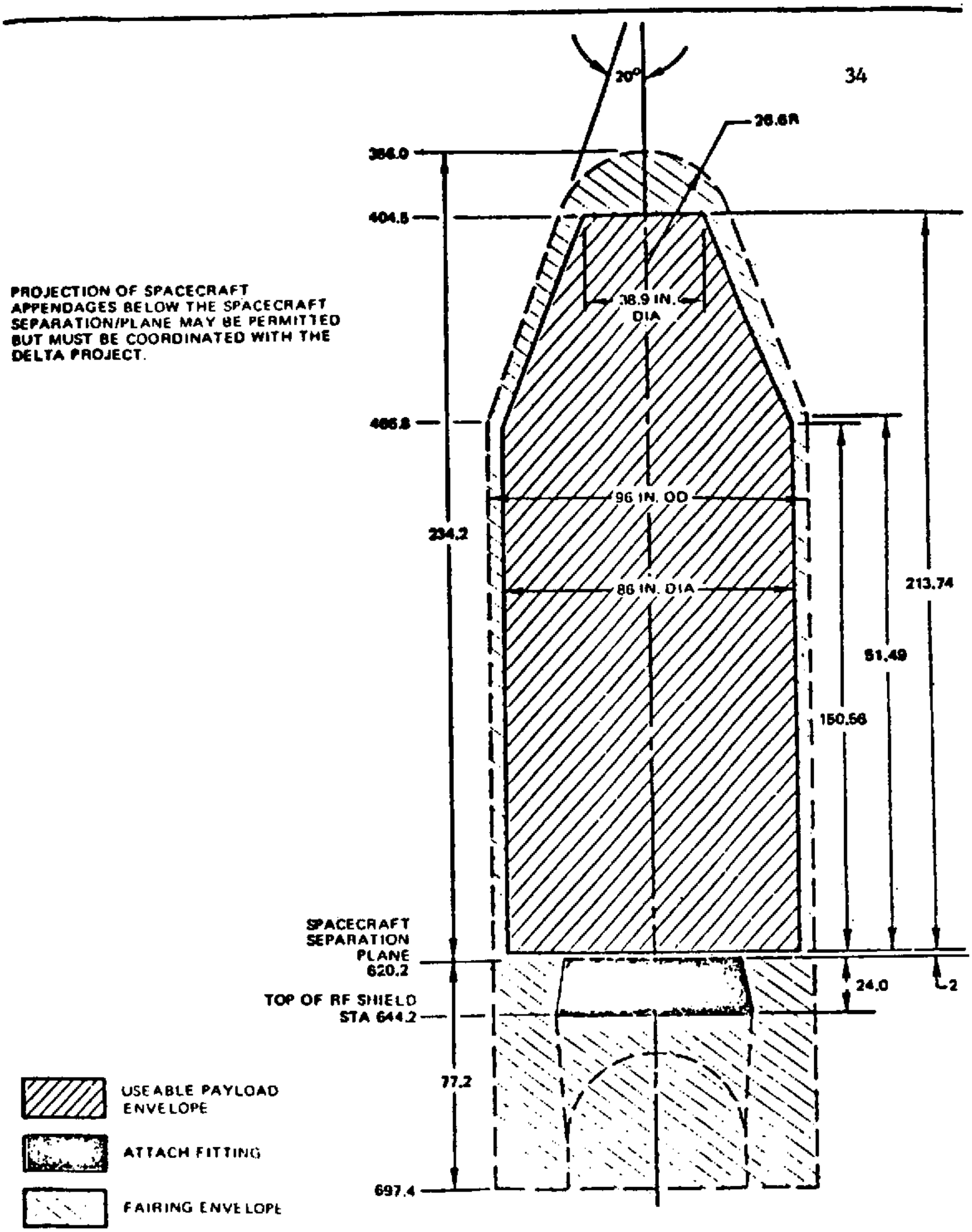


Figure 1.5.3 - Delta 2910 Payload Envelope with 5724 Attach Fitting

1.6 Summary

The program schedule as laid out in figure 5.3.1, is based on two considerations: 1) The project will have its official beginning in FY 75, and 2) During FY 74 enough S/C definition, supporting SRT and generation of SOW's will be performed to provide a full definition of project scope when official approval is granted. The effort prior to project approval will aid significantly in subsequent control of project cost and schedule commitments. The technologies addressed in the pre-project phase, such as solar array, thruster systems and mission design will support other on-going NASA flight programs, even if the project should not receive final approval.

With a FY 75 start, launch could occur as early as late CY 77. Pacing item will be the development of the solar array; however, supporting efforts from the MSFC SEP solar array development program may reduce the in-house LeRC effort and time required. If the start date is delayed until FY 76 or later, a corresponding delay in launch will occur. However, the start/launch delay need not be one-for-one if a strong supporting SRT program for SERT C is initiated. The extent to which a start delay can be mitigated is a direct function of the strength of the SRT program.

The S/C development costs (excluding contingency and inflation factors) of 19.3 million R&D and 40.6 million total program costs represent the accretion of individual subsystem costs together with the integration and supportive costs required to bring the project to a successful conclusion. This low cost figure, for a program of this size, represents the cost effectiveness of the LeRC approach of employing existing LeRC in-house capabilities coupled with off the shelf hardware.

2.0 MISSION REQUIREMENTS AND OPERATIONS

2.1 Introduction

The SERT C mission consists of two distinct phases which require mission analysis and trajectory design. The first phase is the orbit-raising or spiral-out trajectory from a low altitude, inclined parking orbit to geosynchronous orbit. The second phase consists of 5 years of operation in geosynchronous orbit where station keeping, station walking, and rendezvous will be demonstrated with electric propulsion. The following discussion is a preliminary mission profile for SERT C which has been employed in defining the performance of the low-thrust systems and the basic spacecraft configuration. A tentative or preliminary decision has been made to place SERT C into a circular rather than an elliptic parking orbit. This decision is based upon preliminary trade offs between the complexity of the attitude control system to accomplish the required thrust-steering, and the orbit-raising time. A detailed study must be performed before a final decision can be made.

<u>Orbit raising phase</u> - The spacecraft is launched due east from ETR by a Delta 2910 launch vehicle. A partial burn of the second stage will put the spacecraft on a transfer orbit. At apogee, a second burn of the second stage will place the spacecraft into a 3150 km altitude parking orbit inclined 28.3 degrees from the equator. The second stage of the booster will be used to acquire the proper space-

craft attitude before separation.

After an initial check out of the spacecraft systems, the 30 cm ion thrusters will be turned on. Thrust from these engines will cause the spacecraft to follow a spiral trajectory out to geosynchronous orbit; the semimajor axis of the instantaneous orbit will increase, and the inclination will simultaneously decrease. Since the low thrust transfer will begin well below the Van Allen radiation belts, the original 8 kW of solar array power available to the thrusters will degrade (it is estimated) to about half the BOL value during orbit-raising. Approximately 290 days will be needed to transfer the spacecraft into a geosynchronous orbit (semimajor axis 42164 km, zero inclination, zero eccentricity). Conservative assumptions were made regarding the efficiency of converting solar array power into thruster energy.

Operation of the electric thrusters will be continuous except for brief periods during which the spacecraft passes through the earth's shadow. The maximum possible shadow duration for a circular orbit will be about 36 minutes at the start of the mission, and the maximum will increase to about 69 minutes near the end of the transfer. For a mission this long, there are typically three distinct time intervals during which shadowing occurs. These intervals are of such a duration that approximately half the total number of orbits experience some shadowing. The total shadow time incurred during the spiral out is a function of the launch date (which sets the initial position of the sun) and the launch time (which determines the initial orientation of the orbit plane).

The subject of the required thrust vector orientation, or steering law, for a circle-to-circle transfer (circular parking orbit) is treated in the section titled Steering Law Selection. This section describes in some detail a simplified system for guiding the spacecraft during the orbit-raising portion of the mission. With the thrusters attached to the spacecraft as in the proposed configuration, the spacecraft yaw attitude must be varied with orbit frequency to produce a component of thrust normal to the orbit plane. Because the yaw axis is the only spacecraft axis about which rotation is allowed, the solar panel axes will not always be perpendicular to the solar radiation, so that the angle between the array normal and the sunline will vary throughout the orbit. The peak value of this angle is determined not only by the yaw oscillation magnitude but by the orientation of the orbit plane relative to the sunline. Therefore, the average power delivered to the thrusters over the orbit-raising phase is a function of launch time and launch date. The trade offs between the shadow time and maximum average power is discussed in the section titled, Launch Opportunity. A launch window can be determined from this information together with any other constraints which may be identified.

<u>Synchronous orbit phase</u> - Upon arrival in geosynchronous orbit at 60°

west longitude, attitude control and station keeping will be demonstrated with the use of 8 cm ion thrusters. The station walking and rendezvous functions will be performed using the 30 cm thruster system, with the 8 cm thrusters providing backup capability. A description of the synchronous orbit operations sequence is given in section 2.3.

The requirements for attitude control and station keeping for the SERT C mission have been derived from the potential mission applications at synchronous orbit. Section 2.3 discusses these requirements. High-power communication satellites present the most stringent attitude control and station keeping requirements of this satellite class. SERT C will demonstrate attitude control to an accuracy of .08 deg pitch, .08 deg roll, and .2 deg yaw. The station keeping accuracies will be $\pm$ .1 deg east-west and $\pm$ .05 deg north-south. Although a preliminary investigation of the station keeping operations has been performed, a detailed simulation of the station keeping operations is needed to confirm the feasibility of controlling to these accuracies.

Rendezvous experiments will be conducted with other satellites at geosynchronous orbit. Since most of the candidate rendezvous targets have no north-south station keeping, SERT C must change its inclination as well as station walk for rendezvous with another spacecraft.

2.2 Orbit Raising Phase

Parking Orbit Selection - The orbit achieved by the Delta booster, from which low-thrust electric propulsion begins, can be either circular or elliptical. A preliminary decision has been made to employ a circular parking orbit for the SERT C mission. This decision must be confirmed be detailed analyses of both the circular and elliptic orbit cases. These analyses will indicate the trade off between the complexity of the attitude control system and the orbit raising time.

The capability presently exists at Lewis to analyze low-thrust transfers from circular parking orbits, and the capability is being developed to analyze transfers from elliptical orbits. However, this latter capability is not at the point where a final decision can be made on the type of parking orbit to be used. From contacts with organizations that have made studies of elliptic orbit transfers, it appears that the optimum steering law does yield a savings in transfer time over the circular orbit case. These studies together with an analysis of the equations describing the optimum steering law also show that the elliptic transfer requires wide variations in the spacecraft pitch and yaw attitudes, particularly near the end of the orbit raising phase of the mission.

With the thrusters fixed to the spacecraft in the proposed configuration, this would require a more complex attitude control system than would be needed for transfer from a circular parking orbit. The increased complexity of the attitude control system could be justified only if the time saving resulting from an elliptical orbit is large. Alternatively, the parking orbit can be elliptical and an easy-to-implement steering law that is not optimum can be employed. This would, of course, reduce the saving resulting from the use of an elliptical orbit. These preliminary considerations have led to the circular parking orbit being the preferred approach in the orbit-raising outline which follows.

Steering Law Selection - Given an initial circular parking orbit, the SERT C mission will utilize an optimum steering law which simultaneously increases the orbit semi-major axis and decreases the orbit inclination. The steering provides the in-plane and out-of-plane components of thrust which assure that the geosynchronous orbit conditions of zero inclination and semi-major axis of 42164 km are achieved simultaneously. As the spacecraft approaches synchronous altitude the steering law will be revised to provide geosynchronous orbit arrival at 60° west longitude. This longitude is preferred for the first synchronous orbit experiments.

A steering law has been selected for use in the mission calculations of this project study. The selection criteria were ease of implementation in a guidance scheme and the simplicity of the attitude control system. A detailed simulation of the trajectory and spacecraft constraints will be required to investigate this law or other candidate laws before a final steering law is selected. There are additional areas requiring investigation. These are to determine the effect on the steering law of the long term or secular thrust level variation and the periodic thrust level variation. The secular variation is caused by the degradation of the solar array while the periodic thrust variation arises from the fact that the solar arrays are not normal to the sunline over a complete orbit revolution.

Previous steering laws for low-thrust transfers were derived (by T. N. Edelbaum of MIT/Draper Labs and others) on the basis of minimizing the characteristic velocity or ΔV required. The ΔV is directly related to the propellant required and is equivalent to minimizing the orbit raising time assuming constant thrust throughout the mission. The steering law used in the mission calculations of this project study was derived with that assumption. This steering law may not be the optimum one for the SERT C mission. This mission is not one of constant thrust but one which has secular and periodic thrust variations. The secular thrust variation is a function of the steering law employed. This is because the power degradation is a function of the altitude and inclination histories which in turn have been determined by the steering law. Minimizing the ΔV may not be equivalent to minimizing the transfer time.

The baseline steering law used in the project study is:

$$\sin\beta = \sqrt{\frac{R}{R_0}}\,\sin\beta_0 \cos u$$

where R is the orbital radius

R_0 is the initial value of orbital radius

β is the out-of-plane angle (spacecraft yaw angle) of the thrust vector (measured in a plane perpendicular to the radius vector, positive northward)

u is the argument of latitude (measured from the first ascending node)

β_0 is the value of out-of-plane angle at the ascending node at the beginning of the mission

This law utilizes current values of two orbit elements, R and u, and appears to be easy to implement in a guidance scheme. Figure 2.2.1 illustrates the variation of the angle β over a complete orbit in both graphical form and in a sketch. The angle u for a circular orbit is a linear function of time (and approximately so for a low-eccentricity orbit) given by

$$u = u_0 + \omega t$$
$$\omega = \frac{V}{R} = \sqrt{\frac{\mu}{R^3}}$$

where u_0 is the value of the argument of latitude at time zero

ω is the orbit angular velocity

μ is the product of the universal gravitational constant and the mass of the earth

Substituting this into the control law yields:

$$\sin\beta = \sqrt{\frac{R}{R_0}}\,\sin\beta_0\,\cos\left(u_0 + \sqrt{\frac{\mu}{R^3}}\,t\right)$$

This control law or algorithm would be programmed in the on-board computer.

<u>Attitude Control Requirements</u> - The attitude control system as described in section 4.1 provides the steering required by the above law. Any attitude error results in a misalignment of the thrust vector from the desired direction which will cause the transfer time to increase. A detailed analysis must be made before an accurate value for allowable attitude errors during orbit raising can be established. However, it is felt that errors in each axis on the order of 1 degree are probably allowable, because the error in thrust magnitude is proportional to the cosine of the attitude error. Attitude errors can reasonably be expected to be held to less than 0.5 degrees during this period. Demonstration of thrust vector orientation to these accuracies supports the objective of this mission as a SEPS/GEOSEPS precursor, because thrust vector orientation accuracies during powered flight are specified to be $\pm$ 1 degree for these missions.

<u>Baseline Mission</u> - The orbit raising for the baseline SERT C mission commences from a circular parking orbit having an altitude of 3150 km and an inclination of 28.5 degrees. At the beginning of the mission, 8 kilowatts of the solar array power is dedicated to the 30 cm thruster operation. Assuming 85% efficiency of the thruster power conditioners,

the initial power delivered to the thrusters is 6.8 kilowatts. Two of the 30 cm thrusters will operate throughout the orbit raising phase, while the third thruster provides backup capability. At the start of the orbit raising phase, there is more than adequate array power to operate the two thrusters at full power. As shown in figure 2.2.2, the power to the thrusters degrades rapidly during the first 50 days of orbit raising. As this occurs, the two thrusters will be throttled back equal amounts. Both thrusters will continue to operate in a throttled condition to the completion of the orbit raising phase. The power processor efficiency of 85 percent was employed in the baseline design to provide some conservatism.

Figure 2.2.3 shows the changes in the orbit altitude and inclination as the orbit raising operations proceed. The peak amplitude of the angle β in the steering law to accomplish these changes is shown in figure 2.2.4. A transfer time of 290 days is required to achieve geosynchronous orbit with an injected mass of 1540 pounds. The fuel expended during the mission is 270 pounds. The transfer time includes the effect of thruster shutdown during orbit shadowing and the power variation due to the offset between the array normal and the sunline. The transfer time could be reduced approximately 15 days if a power processor efficiency of 90 percent were used in the calculation.

Figure 2.2.5 shows the sensitivity of the orbit raising time to the synchronous payload weight. Noting the position of the baseline mission, this figure shows that a spacecraft growth of 300 pounds would maintain the orbit raising time to less than one year.

<u>Orbit Raising Operations</u> - A guidance system concept to accomplish the orbit raising operations is illustrated in figure 2.2.6. The various phases of operation are as follows:

Svstem Initiation

Following parking orbit injection, the STDN network will track the satellite and an orbit determination will be performed. From this information, the orbit period and the argument of latitude at the time of thrust steering initiation will be calculated. If the parking orbit has a small eccentricity, it will be assumed to be circular with a radius equal to the semimajor axis. The initial values of semimajor axis, inclination and other orbital elements will be inserted into a computer simulation of the orbit raising phase which will calculate a value of β_o, the initial steering law amplitude. This value will be determined such that zero inclination is achieved slightly before synchronous semimajor axis. (This will eliminate the necessity of steering angles of 90 degrees at the end of the mission). The value of β_o will be very nearly -27 degrees for a nominal trajectory starting from a circular parking orbit of 3150 km altitude.

Following a spacecraft health check out period, a value of $(\sin\beta_o)/\sqrt{R_o}$, a value of $\sqrt{R}$ equal to $\sqrt{R_c}$, and a value of U_o will be transmitted to the spacecraft. The value of U_o is such that the argument of the cosine function in the steering law will pass through zero (negative to positive) as the spacecraft passes through the ascending node. The guidance or steering algorithm in the on board computer will then be activated. This replaces the constant zero value command output to the yaw controller of the attitude control system. It will provide a time varying command angle β_c to the yaw controller which will in turn supply a torque about the yaw axis to rotate the spacecraft. The thrust vector may be offset from the spacecraft roll axis in order to orient it through the spacecraft cm. Therefore the commanded yaw steering angle may differ from the actual yaw angle of the center body by an amount up to the value of this offset. The offset which is not expected to be greater than about 5 degrees, will be compensated by introducing a bias of appropriate magnitude into the yaw control loop.

After determining that the spacecraft is executing the proper yaw motion and that it does not have motion about the pitch and roll axes (the closed-loop pitch and roll control system will null any motion about these axes), the thrusters will be turned on.

Normal Orbit Raising Operations

Operation of the thrusters will increase the semimajor axis of the orbit and decrease the inclination. The circumferential component of thrust will tend to reduce any eccentricity the orbit may have. At the same time, shutdown of the thrusters during the shadow periods will tend to change the eccentricity. (If shadowing occurs near the apogee the eccentricity will increase; if it occurs near the perigee, eccentricity will decrease.) If the eccentricity is small, the value of R can be assumed to be equal to the semimajor axis with little error resulting. During the orbit raising, the STDN network will track the spacecraft and perform periodic orbit determinations. As the spacecraft altitude increases and the orbit inclination decreases, it becomes necessary to increase the amplitude of the out-of-plane oscillation. At various times during the mission, a current value of $\sqrt{R}$ and a new calculated value of U_0 will be sent to the spacecraft to update these values in the algorithm programmed in the computer. The larger value of $\sqrt{R}$ will increase the maximum absolute value of the out-of-plane angle and also reduce the frequency of the oscillation required because the orbit period increases. The new value of U_0 will assure that the yaw angle oscillation is in proper phase with the variation of the argument of latitude.

Steering Law Update

If the orbit raising computer simulation were able to model the mission exactly (i.e., perfect knowledge of the thrust magnitude time history, thrust vector orientation accuracy, etc.) the orbit raising trajectory would be as predicted. There will, however, be some perturbation or disturbance to the trajectory which must be corrected during the spiral out. When the deviations between the actual trajectory and the predicted trajectory have reached specified values (to be determined) a new optimum trajectory will be calculated. This trajectory will minimize the transfer time from the current orbit to the geosynchronous orbit. The trajectory simulation code will again be run to calculate a new value of $(\sin\beta_0)/\sqrt{R_0}$ which will be inserted into the guidance law on board the spacecraft. To determine the new optimum trajectory, spacecraft parameters such as weight and power available must be known. This means that the spacecraft must be instrumented so that these parameters as well as thrust magnitude can be determined.

Final Altitude & Eccentricity Adjustment

The parameter $(\sin\beta_0)/\sqrt{R_0}$ is determined (initially and at subsequent later times) so that zero inclination is achieved slightly before synchronous semimajor axis is reached. When the inclination becomes zero a command will be sent to the on board computer to supply a constant command angle to the yaw controller. The constant yaw angle will orient the spacecraft so that no out-of-orbit plane component of thrust exists. Under this condition, the inclination will remain zero, the semimajor axis will increase, and any small remaining eccentricity will be reduced somewhat. (If there are any shadow periods at this time, the associated eccentricity change would be small because the maximum shadow period would be close to 1/24 of the orbit period.) As the spacecraft approaches synchronous altitude a short coast phase may be necessary. This coast will allow the spacecraft to drift to a longitude relative to 60^o west such that upon startup and continued thrusting, the spacecraft will arrive at 60^o west in the geosynchronous orbit. When synchronous semimajor axis is reached, action will be taken to reduce any final eccentricity error to an acceptable level. This will be done by thrusting circumferentially but with the thrust reversed at each crossing of the minor axis, thrust being nearly in line with the velocity during the apogee half of the orbit, and in the opposite direction during the perigee half of the orbit. This procedure will reduce the eccentricity with no effect on inclination and none on semimajor axis if there is no shadowing at this time. (If there is shadowing, the semimajor axis may have to be adjusted again slightly.) When the eccentricity reaches zero, a synchronous equaorial orbit will have been achieved and a command to turn off the thrusters will be sent.

Launch Opportunity - As the spacecraft executes the motion about its yaw axis, the power obtained from the solar panels varies over each orbit. The limits of variation are a function of the maximum absolute value of the yaw angle during the orbit, the orbit inclination, the orientation of the orbit plane (or the ascending node longitude), and the position of the sun relative to an equatorial coordinate system (or the day of the year). The limits themselves vary over the course of orbit-raising for the following reasons: the maximum absolute value of the yaw angle (which occurs at the nodes) increases as altitude increases, (from 26.8 to 71.7 degrees for the nominal mission) the orbit inclination decreases from 28.3 to zero degrees, the ascending node regresses because of the oblateness of the earth (approximately 170 degrees for the nominal mission), and the sun appears to move, in an equatorial coordinate system, in a circle inclined 23.5 degrees to the equatorial plane with a period of one year. The variation of power over an orbit due to solar panel misalignment for a particular set of conditions is illustrated in figure 2.2.7.

The orbit represented is that which presented the lowest value of instantaneous power over the entire orbit-raising phase for the indicated launch date and time. Note that, although the minimum instantaneous power drops to 70 percent of the maximum available from the array, the average over the orbit is 90 percent. Obviously, to take advantage of this higher average requires being able to throttle the thruster in a corresponding fashion. The maximum throttling rate required appears to be about 1 percent per minute.

For the present spacecraft configuration and the baseline mission profile and steering law, the only factors that can be controlled so as to reduce the overall power loss due to solar panel misalignment are the initial orientation of the orbit plane relative to the ecliptic plane (determined by the time of day the spacecraft is injected into orbit) and the initial position of the sun (determined by the day of the year the mission begins). After injection into parking orbit, the orbit plane regresses and the sun changes its position relative to the earth, both of these effects influencing the limits of variation of the solar power over an orbit. This, in turn, affects the total power delivered to the thrusters during the course of the mission.

The same two mission parameters, launch date and time, also affect the time that the spacecraft spends in shadow. During time in shadow, the power drops to zero and the orbit-raising time is increased over the time required with no shadowing, by an amount approximately equal to the total time spent in shadow. Without considering other launch constraints which may arise, it is obvious that the launch date and time should be selected so as to maximize the time integral of power delivered to the thrusters. This would result in the shortest orbit-raising time.

The initial position of the sun is set by the launch date. Four extreme sun positions were selected for examination: the vernal equinox (March 21) when the sun lies in the equatorial plane, the summer solstice (June 21) when the sun has risen to a maximum angle of 23.5 degrees above the plane, the autumnal equinox (Sept. 23) when it again lies in the plane, and the winter solstice (Dec. 22) when it has sunk to 23.5 degrees below the plane. The orbit-raising time for each of these dates and for a complete range of launch times (0000 to 2400 hrs GMT) is presented in figure 2.2.8. The parking orbit for all the cases is circular; other initial conditions and the final conditions are listed on the figure. It was assumed in this analysis that the antinode (point of maximum latitude) of the parking orbit is located at the celestial longitude of the launch site at the time of launch. This determines the position of the orbit plane in inertial space.

It can be seen from figure 2.2.8 that the minimum orbit-raising times occur for launch dates near the solstices ($\gtrsim$ 290 days); however, they are only slightly lower than the lowest times for launches near the equinoxes ($\gtrsim$ 295 days). So, it appears that the initial position of the sun (or the launch date) does not have a significant effect on a long duration mission of this kind. The initial orientation of the orbit plane (or the launch time) does have a significant effect upon orbit-raising time. For a vernal equinox launch (top curve), the shortest mission time is associated with a launch time of about 1200 hours; three months later, (second curve), the best launch time moves forward about 6 hours to shortly after 0600 hours. The same pattern follows for the other two dates presented. Interestingly, the orientation of the parking orbit in inertial space is nearly the same for all four minimum-time cases. From this figure it can be seen that a delay in launch date from some selected value should be accompanied by a moving-up of the launch time from its original nominal value.

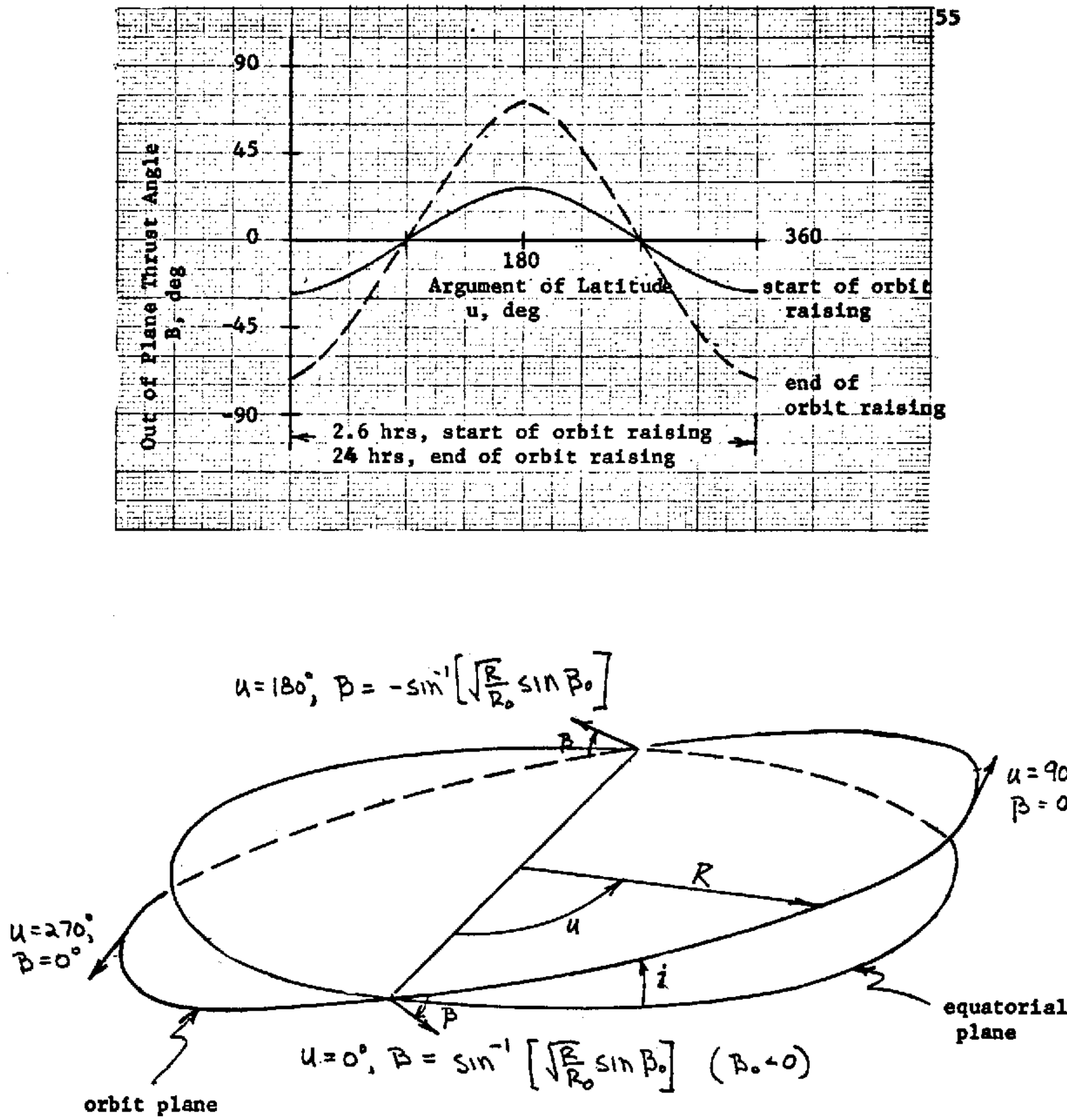


Figure 2.2.1 - Near-Optimal Thrust-Steering Angles at Several Points Along an Orbit for a Transfer from a Circular Parking Orbit to Geosynchronous Orbit

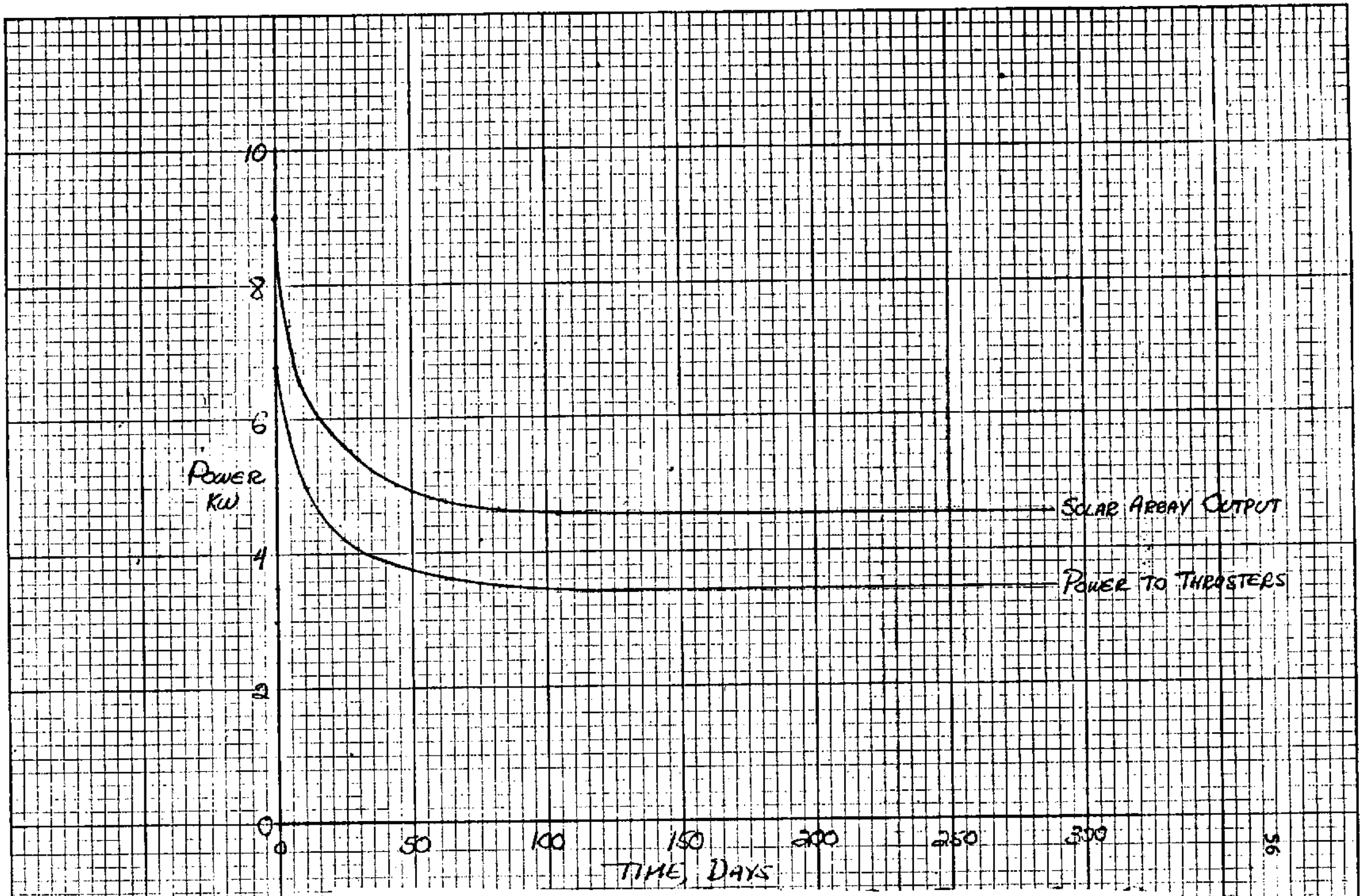


Figure 2.2.2 - Solar Array Power vs Time During Orbit Raising

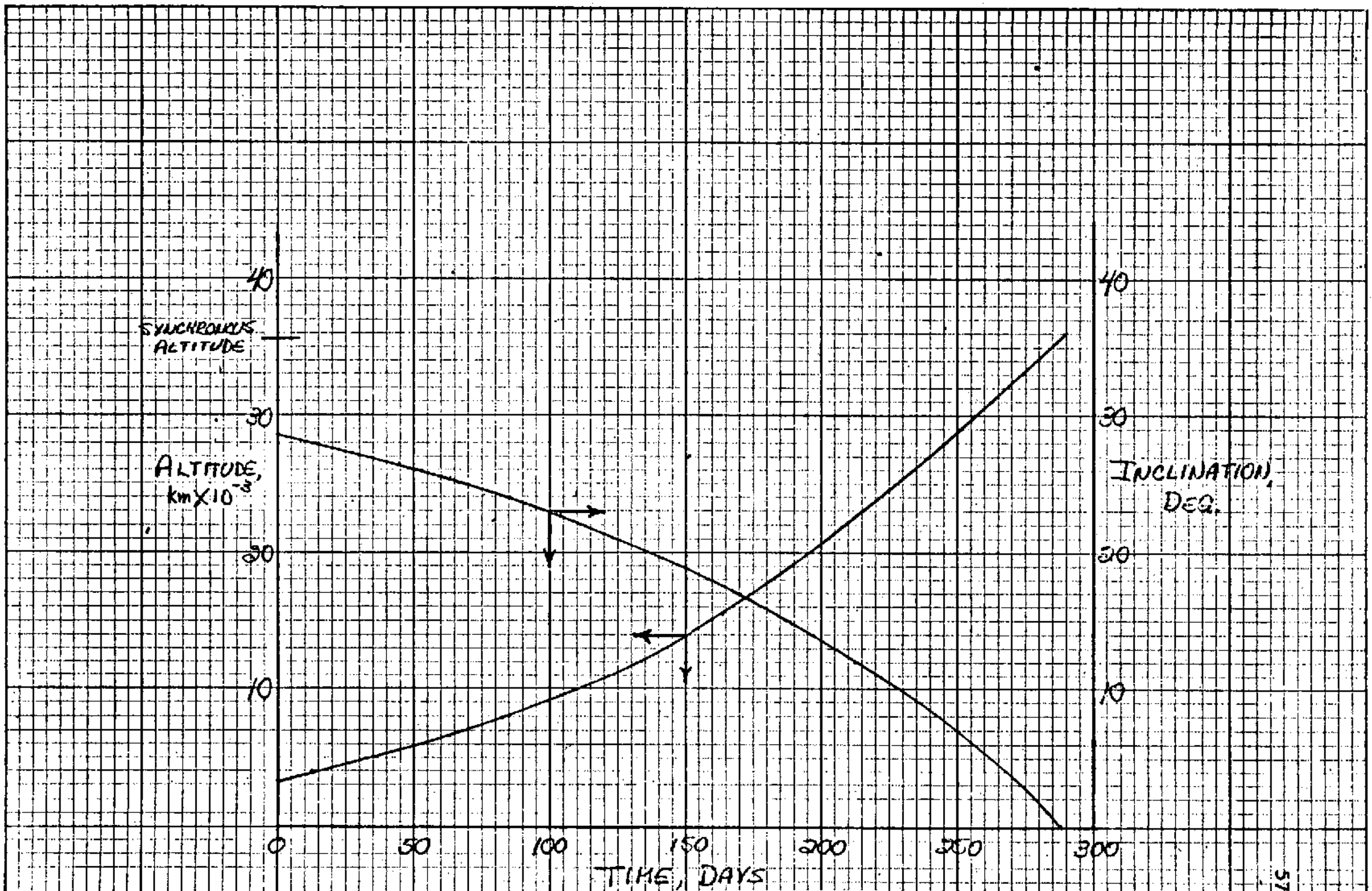


Figure 2.2.3 - Altitude and Inclination vs Time During Orbit Raising

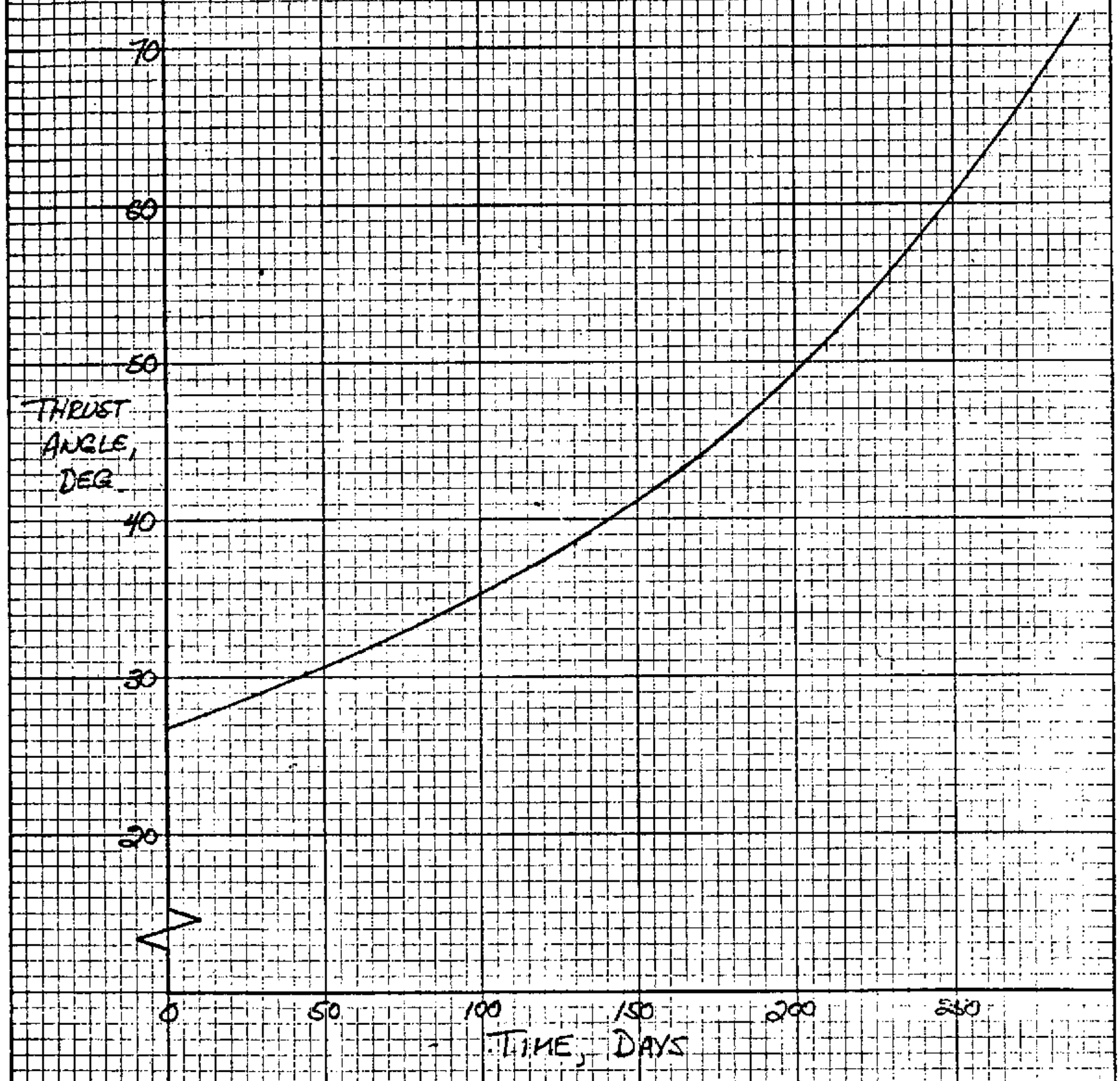


Figure 2.2.4 - Peak Out-of-Plane Thrust Vector Angle vs Time During Orbit Raising

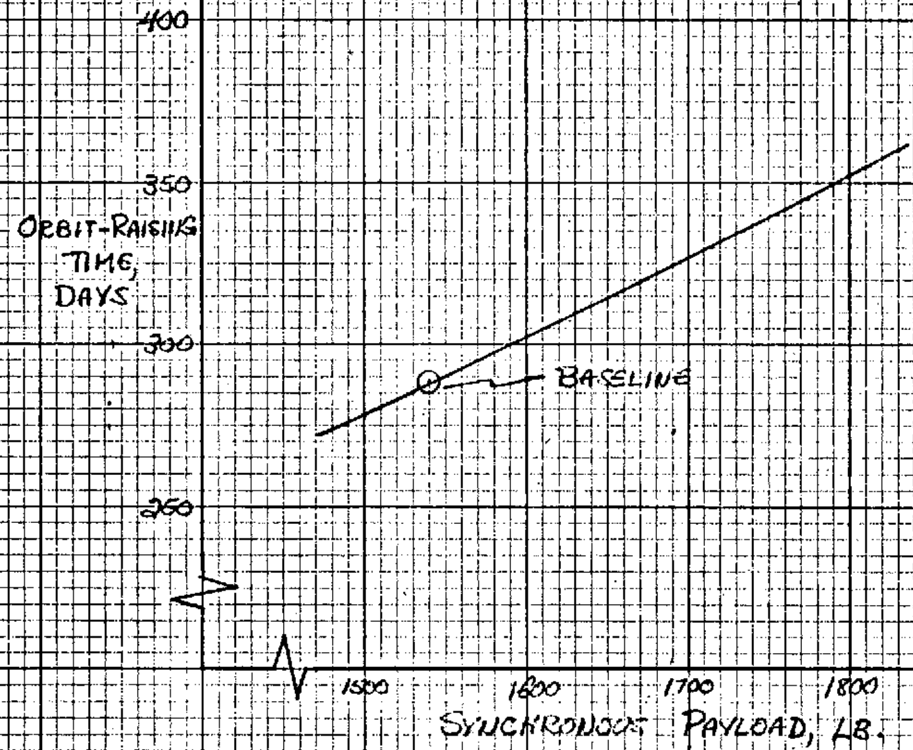


Figure 2.2.5 - Sensitivity of Orbit-Raising Time to Synchronous Payload Weight

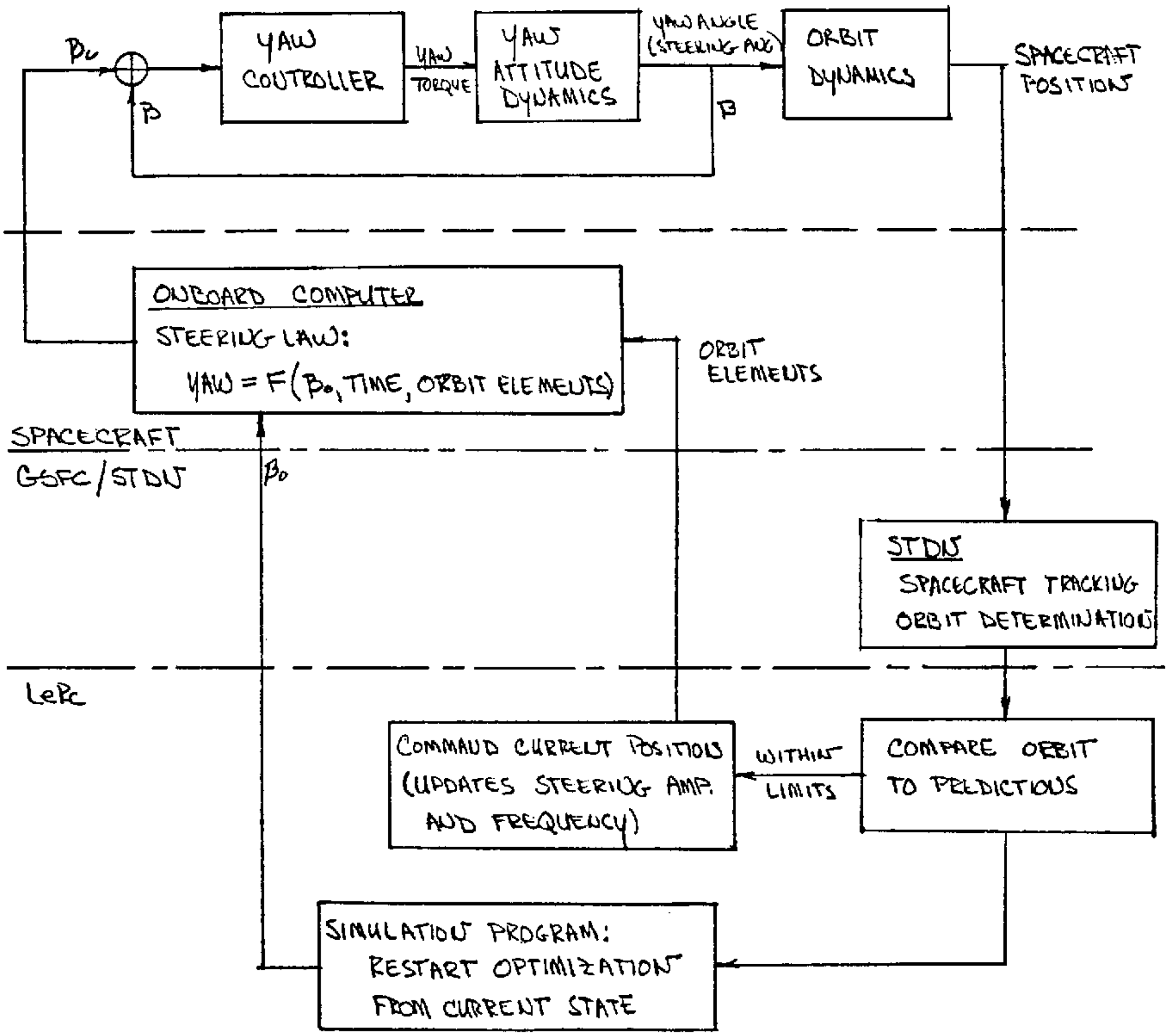


Figure 2.2.6 - Proposed Guidance Scheme

LAUNCH DATE AND TIME JUNE 21 1200 hrs GMT

Orbit	Parking	Orbit Presented
Time, days	T	T+290
Orbit Inclination, deg	28.3	~0
Eccentricity	0	0
Radius (Altitude), km	9528(3150)	41695(35317)
Period, hrs	2.57	~24
Ascending Node		
(Relative to the Vernal Equinox), deg	0	188
Spacecraft Mass, lbm	1800	1525
Solar Power (Maximum Available), kW	8	4.5
Maximum Absolute Value of Yaw Angle during		
the Orbit (at the Nodes), deg	36.8	70.7

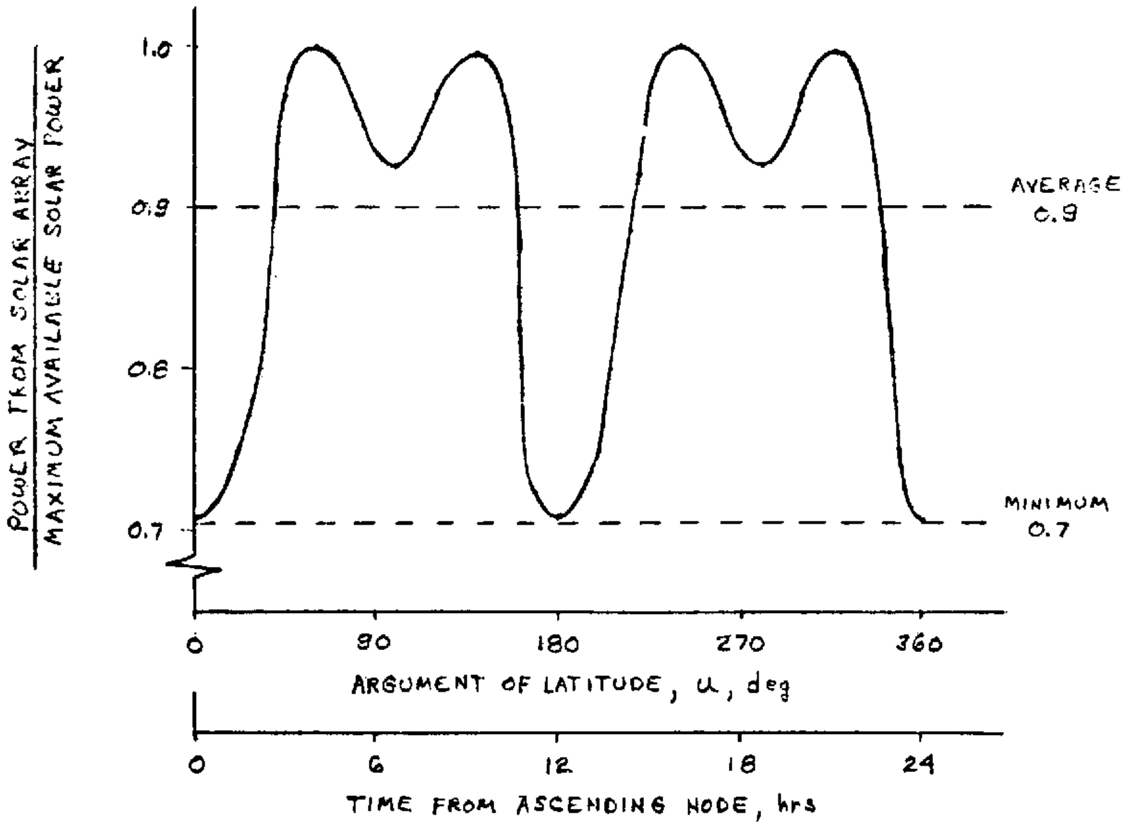


Figure 2.2.7 - Variation of Solar-Electric Power Over that Orbit which has the Lowest Value of Instantaneous Power (for the Specified Set of Conditions).

PARAMETER	PARKING ORBIT	SYNCH ORBIT
INCLINATION	28.3 Deg	0
ECCENTRICITY	0	0
ALTITUDE	3150 km	35,786 km
PERIOD	2.57 hrs.	24 hrs
SPACECRAFT MASS	1810 lb.	1540 lb
SOLAR ARRAY OUTPUT POWER	9.0 kw	4.55 kw

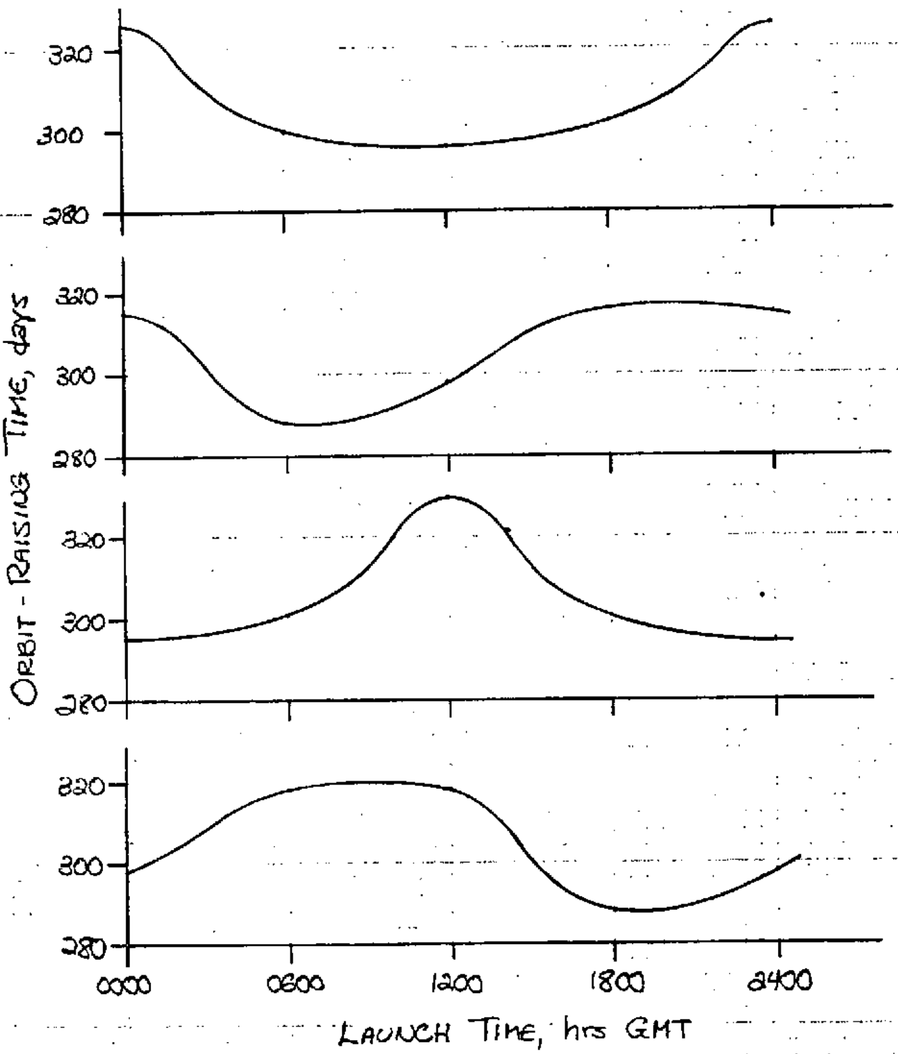


Figure 2.2.8 - Variation of Orbit-Raising Time with Launch Date and Time (baseline case)

2.3 Synchronous Orbit Phase

On orbit operations sequence - The station keeping experiments will be performed in the longitude quadrant centered about the earth's minor axis at 105° west. The first station keeping experiment will take place at 60° west which represents a longitude of maximum drift acceleration caused by the earth's triaxiality. During the final portion of the orbit-raising phase, a guidance strategy will be adopted which places the spacecraft at 60° west upon arrival at geosynchronous orbit. This strategy will demonstrate a gross rendezvous of a spiral out spacecraft with a spacecraft that has been in synchronous orbit.

Precise north-south and east-west station keeping will be demonstrated with the 8 cm ion thrusters at 60° west for a period of 60 days. The spacecraft will then be station walked using a 30 cm thruster to 75° west where another station keeping demonstration will be performed. The entire longitude quadrant centered about the earth's minor axis at 105° west will be mapped with a sequence of station keeping and station walking demonstrations. These demonstrations will verify the achievable station keeping tolerances as a function of the control period and drift acceleration. They will also identify an operational plan for station keeping with ion thrusters of future satellites.

As the station keeping experiments are performed, the position and status of other spacecraft in a synchronous, nearly equatorial orbit will be reviewed for candidates as rendezvous experiments. Depending upon the relative position of the SERT C spacecraft and the candidate rendezvous spacecraft, it may be advantageous to interrupt the station keeping experiments and proceed with a rendezvous experiment.

The entire sequence of mapping the longitude quadrant and performing a few rendezvous experiments should be completed within the first two years of the five year on-orbit lifetime. During the next three years, the spacecraft will be kept on station for extended periods of time at longitudes which maximize the length of the control periods. This will allow the amount of ground support to be reduced. Depending upon the fuel contingency available from the orbit-raising phase, the 30 cm thrusters will be periodically exercised to perform a station walk or rendezvous.

A profile of the anticipated on-orbit operations is given in table 2.3.1.

Station keeping operations - The forces which disturb the station of the satellite include lunar and solar gravity, earth's triaxiality, and solar pressure on the spacecraft. The station errors in the north-south (N-S) direction are caused by an increase in orbit inclination which results from gravitational attraction by the sun and moon. The effect on the spacecraft is to cause a daily latitudinal variation of the subsatellite point north and south of the equator, of magnitude equal to the orbit inclination. Station errors in the east-west (E-W) direction result from two sources. First, the earth's triaxiality causes a constant drift toward the earth's minor axis which must be nulled periodically. The second station error is caused by solar pressure on the spacecraft. The force resulting from this pressure causes an increase in orbit eccentricity. The effect of the eccentricity is to produce a daily longitudinal variation of the subsatellite point east and west of the desired station, of magnitude (in radians) equal to twice the eccentricity. As discussed in the section titled Station Keeping Requirements, the station keeping system is required to control the latitude to $\pm .05^{\circ}$ and longitude to $\pm .1^{\circ}$.

North-south station keeping is accomplished by controlling orbit inclination. This is done by firing an array-mounted 8 cm thruster for a period of time centered on an orbit node. If the control (or correction) is made daily, each thruster would run for 2.6 hours. The south thruster would fire on the descending node, and the north thruster on the ascending node.

East-west station keeping, when counteracting the effect of solar pressure, is accomplished by rotating the line of apsides of the eccentric orbit, using two tangential impulses, one-half an orbit period apart. The solar pressure changes both the eccentricity of the orbit and the orientation of the line of apsides. If the line of apsides is properly placed, the solar pressure will cause the eccentricity to decrease and then increase as shown in figure 2.3.1. Thus, the station keeping maneuver is accomplished by rotating the line of apsides each time the eccentricity reaches the allowable limit. The thrusting duty cycle is a function of the satellite area-to-mass ratio. Of the two impulses required to rotate the line of apsides, the one must be in the posigrade direction and the other in the retrograde direction. These impulses on SERT C are obtained directly by using the 8 cm thrusters mounted on the east and west faces of the spacecraft. The maximum daily east-west variation due to solar pressure on SERT C is about $\pm$ .16 degrees if left uncontrolled. To control the solar pressure variation to the required $\pm .05^{\circ}$, the thrusting time of each body mounted 8 cm thruster would be 4.0 hours once every 7 days. The array mounted 8 cm thrusters provide the backup capability. One thruster would be required to operate for 4.8 hours twice during one orbit every seven days. The required posigrade and retrograde impulses are achieved by alternatively yawing the spacecraft plus and minus 60 degrees.

East-west station keeping to counteract the effect of earth triaxiality is accomplished by periodically changing the orbit semi-major axis. The variation in semimajor axis as a function of variation in satellite longitude is shown in figure 2.3.2 for cases with station keeping and without station keeping. The coordinate system is referenced to the desired operating station. The deadbands for the acceptable longitude error ΔL_t caused by the triaxiality are also shown. The station keeping procedure is to first obtain a precise orbit definition and then set the semimajor axis at a value Δa_c less than the synchronous value as shown at point A, for stations east of 105° west. The perturbation will result in the orbit sweeping the arc ABC over the control period. At the point C, the semi-major axis is reset to point A. The control period over the arc decreases as the longitude of the operating station referenced to the earth's minor axis at 105° west increases. A change in semi-major axis requires two impulses. However, both of these are in the same direction, and are retrograde for stations located east of 105° west longitude. Therefore, the east body mounted thruster will be used for this operation. The west body mounted thruster will be used for stations west of 105° west. The solar-pressure and triaxiality corrections can be combined, resulting in some saving in propellant. If the correction were made every seven days with the east body mounted thruster, the total thrusting time per impulse would be 43 minutes. The array mounted thrusters are used as a backup for the triaxiality and solar pressure corrections.

Station walking and rendezvous operations - The station walking and rendezvous operations consist of changing the longitude of the synchronous satellite by a significant amount. The rendezvous operation may also involve changing the inclination of SERT C to rendezvous with spacecraft that are at synchronous altitude but do not have north-south station keeping.

Station walking is accomplished by changing the orbit radius to a value other than synchronous, producing a drift rate, and then changing the orbit radius back to synchronous when the desired longitude change has been achieved. To produce a westward station walk with low thrust, the maneuver consists of thrusting in the direction of the orbit velocity vector until half the desired longitude change has been made, and then thrusting opposite the velocity vector for the other half of the longitude change. An eastward station walking maneuver is accomplished by reversing the thrust directions. A 30 cm thruster is the prime system for performing the station walk maneuver. Figure 2.3.3 shows the minimum time required to accomplish a given change in spacecraft longitude. This assumes that there is continuous thrusting for the entire station walk. Because the 30 cm thrusters provide a larger acceleration than the 8 cm thruster, there is a larger change in the orbit radius for a given longitude change. Hence the 30 cm thruster provides a larger characteristic velocity associated with each longitude change. The characteristic velocity can be reduced by allowing a coasting time between thrusting periods, and accepting the longer station walk time. The figure also shows the minimum time and characteristic velocity requirements for a body mounted 8 cm ion thruster which would provide the backup capability.

Figure 2.3.4 shows the capability of the 30 cm thruster to change orbit inclination as required during the rendezvous operation. The velocity requirement and time to change the inclination by .5 degrees is given as a function of thrusting time per orbit. The thrusting periods are centered about the ascending and descending nodes. During this period, the thrusters are assumed to be normal to the orbit plane. As is evident from the figure, reducing the thrusting time to 10 hours (5 hours about each node) reduces the velocity requirement considerably without a significant increase in the total maneuver time. The 8 cm ion thrusters mounted on the solar array provide the required thrust vector directly because of their north-south orientation. However, when continuous normal thrust is applied over an entire orbit period the time to change the inclination by .5 degrees is 87 days. The 8 cm thrusters are, therefore, used as a backup and to provide vernier corrections to those achieved by the 30 cm thruster. Because the 30 cm thrusters are on the spacecraft face opposite earth, the spacecraft must be maneuvered to provide the components of thrust normal to the orbit plane. Maneuvering the spacecraft to provide the full thrust of the 30 cm thrusters normal to the orbit plane could result in insufficient solar array power at certain positions in the orbit. Therefore, a strategy must be devised to rendezvous by performing simultaneous station walking and inclination changing as during the orbit raising phase. A constant or cyclic yaw offset of the spacecraft will provide the required components of thrust collinear with the velocity vector and normal to the orbit plane while maintaining sufficient power from the array. The magnitude of the offset is a function of the required longitude change, inclination change, and position of the sun relative to the final line of nodes. The rendezvous times associated with this maneuver would be increased from the minimum time capability given in figures 2.3.3 and 2.3.4.

As the spacecraft approaches nearer to the rendezvous spacecraft, the 30 cm system will be shut down and the spacecraft will revert to its normal synchronous orbit attitude. At this point, the 8 cm thruster system will continue to accomplish the required orbit changes until the radar system acquires the target. The 8 cm thrusters will continue until terminal rendezvous.

<u>Attitude Control Requirements</u> - During those phases of the SERT C mission in which the 30 cm thrusters are not operating (which includes the entire synchronous orbit phase except station walking and rendezvous), the spacecraft orientation accuracy requirements are determined by the objectives which are to be achieved during this period. In its role as a SEPS/GEOSEPS precursor flight, SERT C should demonstrate an ability to control attitude to the accuracy required during the coasting or unpowered phases of these missions. Similarly, in its station walking and rendezvous phases, it should demonstrate the ability to control thrust vector and spacecraft orientations to the accuracies required during the SEPS and GEOSEPS

rendezvous operations. During the thrusting portions of these missions, thrust vector orientation is specified to be controlled to ± 1 degree, while during the coast phases, the most stringent attitude accuracy requirement is ± 0.5 degrees.

A further objective of SERT C is to demonstrate both a low cost attitude control and station keeping capability and also its use as a low cost synchronous orbit platform for future applications. Almost certainly one of these applications would be as a 3 axis stabilized high power communications satellite. A review of the attitude control accuracies specified for the first generation satellites of this type presently being developed (ATS-F and CTS) indicates that this type of application results in much smaller values for these tolerances than those for SEPS and GEOSEPS. (CTS attitude control accuracies are ± 0.1 degree in roll and pitch and 1.1 degrees in yaw). As the power level and transmission frequency of these satellites increases, the control requirements will become more stringent for reasons which will be explained in detail at this time.

The synchronous orbit attitude control accuracy requirements for high-power communications satellites are determined by the beamwidth of the spacecraft antenna, the power required at the ground antenna, and the power available on the spacecraft. For increasing transmission frequency, the antenna beamwidth becomes narrower, and the pointing accuracy becomes more stringent. For an application requiring a specified satellite effective isotropic radiated power (e.i.r.p.) for reception by earth terminals which have specific antenna and receiver characteristics, consideration must be given to the relative influence of the satellite transmitted power and antenna beamwidth in providing this e.i.r.p. In an application requiring only spot coverage, a trade exists between antenna beamwidth and the transmitter power requirements. Also the satellite antenna, by producing a narrow beamwidth, will minimize spillage of power into adjacent geographic regions and thus minimize interference.

In an application requiring coverage to a large geographic region of irregular shape, a single antenna beam may not meet requirements because of interference into adjacent regions. The solution is to produce a contoured antenna beam made up of a cluster of narrow beams. Thus, for an application of either spot coverage or large area coverage, narrow antenna beams may be required, resulting in stringent attitude control accuracy requirements. For a specified geographic coverage area, increased attitude control accuracy results in reduced solar array area.

The half-power beamwidths of satellite antennas operating in the 12-14 GHz range may be as small as 0.5^{o}. In determining the attitude control accuracy requirements, the tradeoff between satellite transmitted power and attitude control accuracy must be considered. As an example of analyzing this tradeoff, assume that the satellite is required to cover a geographic region with a circular antenna beam.

Let θ_c be the angle, subtended from the satellite, which just covers the geographic region (see figures 2.3.5a and 2.3.5b). If the attitude control system permits a maximum antenna pointing error of $\pm\,\theta_e$, then the spacecraft antenna must be designed to cover an enlarged area of angular width $\theta + 2\theta_e$ (see figure 2.3.6). If the minimum power flux density to be received by the coverage region is specified, then the required satellite transmitted power as a function of attitude control accuracy is given approximately by

$$\frac{P}{P^*} = \left[1 + \frac{2\theta_e}{\theta_c}\right]^2$$

where P is the required transmitted power and P* is the transmitted power for zero attitude control errors. This function is plotted in figure 2.3.7, where the value of θ_c is assumed to be 0.5 degrees. It is obvious from this figure that the transmitted power, and hence the solar array size can be reduced substantially by keeping the attitude errors as small as possible. There is, however, a practical limit to the accuracies which can be obtained at present, which is governed predominantly by the accuracies of the attitude error sensors. It is expected that the development of microwave interference techniques (interferometers) in the near future will permit attitude measurement accuracies on the order of .01 to .02 degrees, yielding overall positioning accuracies of about .03 degrees for the pitch and roll axes of a spacecraft. The accuracy about yaw will be somewhat less. However, yaw need not be held as accurately as pitch and roll. The effect of an actual yaw rotation upon the location of the ground spot depends upon the angle between the antenna pointing axis and the spacecraft roll-yaw plane. The allowable yaw error decreases as the angle increases. Thus, to obtain an effective yaw tolerance on the ground of $.03^{\circ}$ for a spot beam aimed at the northernmost latitude of interest, such as Alaska, only requires that the spacecraft yaw axis be held to about 0.2 degrees. Therefore, if the individual errors about roll, pitch and yaw are taken to be .03, .03 and 0.2 degrees respectively, figure 2.3.7 shows that the power penalty over the zero error case is about 65 percent for the worst case error and for a 0.5 degree antenna beamwidth. However, if errors of 0.1, 0.1, and 0.7 degrees are chosen, the power penalty goes to nearly 300 percent, a sizeable difference.

If the coverage region is to be covered by a contoured beam, then the attitude control accuracies should be chosen so that the enlargement of the coverage region due to pointing errors is not unduly large. As mentioned before, it is desirable to minimize the power spillage into adjacent areas because of interference problems. An example of coverage area enlargement for the state of Alaska is shown in figure 2.3.8. The satellite was here assumed to be located at 130°W longitude. The coverage area enlargement is shown for roll, pitch, and yaw errors of 0.03°, 0.03°, 0.20° and for roll, pitch, and yaw errors of 0.10°, 0.10°, and 0.70°. The enlargement due to the 0.03°, 0.03°, 0.20° set of errors is not unduly large and is representative of the desired closeness of fit. Thus, for both single-spot and multiple spot contoured beams, high power communication satellites will benefit by holding the attitude tolerances as small as possible.

A typical future high-power communications satellite which is to be held to $.03^{\circ}$, $.03^{\circ}$, and 0.2° will very likely use an interferometer system to sense attitude errors about all three axes, and reaction wheels as the prime torquers in order to obtain high accuracy pointing. Such a system can be flight demonstrated, at considerably less expense without using an interferometer, by using a high-quality earth sensor to provide earth-based pointing information about the roll and pitch axes, and a rate integrating gyro-sun sensor combination to provide yaw error. The main difference between this system and the interferometer system would be in the accuracies obtained in roll and pitch. It should be possible to maintain yaw attitude within 0.2 degrees with the sun sensor-gyro combination, but the best earth sensors have stated overall accuracies of .06 to .07 degrees, which would result in control accuracies of about .08 or .09 degrees, rather than the .03 degrees desired. The system thus demonstrated should be capable of achieving better pointing accuracies when required in future flights by the substitution of the more accurate interferometer system. In addition the demonstration would provide a flight qualified synchronous orbit platform capable of controlling a payload to "intermediate" attitude accuracies. To attempt to obtain pointing accuracies of .03 degrees without an interferometer would require using non-earth oriented sensors such as star trackers and sun sensors in the roll and pitch axes. Such a system would be more complex than the interferometer system because of the additional on board computation required (resolution of error signals into coordinates acceptable to the attitude control electronics and actuators, etc.) In addition, the system would not closely resemble that likely to be used on future satellites. Therefore, the SERT C spacecraft will use a two-axis earth sensor for pitch and roll error signals, and attitude accuracy goals will be $.08^{\circ}$, $.08^{\circ}$, and $.2^{\circ}$ roll, pitch, and yaw, respectively.

Station Keeping Requirements - As in the case of the attitude control requirements, the high-power communication satellite places the most stringent requirement on station keeping. The station keeping accuracy requirements for high-power communication satellites are determined on the assumption that the ground receiving antennas will have pointing but not tracking capability. In applications requiring a large number of receivers, the ground receiver costs may be a major portion of the total system cost. In such applications, the need for low-cost receiving antennas will rule out tracking capability.

The loss in the signal strength received by the ground antenna is due primarily to two factors. One factor is the pointing error of the ground antenna, i.e., the deviation of the electrical axis of the antenna from the nominal satellite position. The second factor is the station keeping error. Letting θ_p be the pointing error and θ_s the station keeping error, the worst case total error will be the sum $\theta_p + \theta_s$.

It will now be shown that the station keeping accuracy should be chosen so that $\theta_p + \theta_s$ does not exceed approximately one-half of the half-power beamwidth of the ground antenna. In the usual case, the e.i.r.p. of the satellite and the power to be received are both specified. Knowledge of these two specifications determines the ground antenna size if it is assumed that θ_p and θ_s are both zero. Since $\theta_p \neq 0$ and $\theta_s \neq 0$ in the real case, the ground antenna diameter must be made somewhat larger to allow for the nonzero errors and still meet the specifications on e.i.r.p. and received power. As the antenna diameter becomes larger the antenna half-power beamwidth, θ_o, becomes narrower. Figure 2.3.9 shows the allowable worst case error, $\theta_s + \theta_p$, as a function of ground antenna half-power beamwidth, θ_o. The ordinate and abscissa are here normalized by θ_o*, the value of antenna half-power beamwidth for the ideal case of $\theta_p = \theta_s = 0$. For point A, where $\theta_o/\theta_o^* = 0$, the antenna size would be infinite but the beamwidth would be infinitely small, and therefore no error in $\theta_p + \theta_s$ is allowable. At point C, $\theta_o/\theta_o^* = 1.0$, the antenna size is the minimum allowable to meet the specification on received power, and any value of $\theta_p + \theta_s$ other than zero will cause the received power to drop. Therefore, the optimum value of beamwidth (or antenna size) lies between these two extremes as shown by the figure. For specified value of worst case error, $\theta_s + \theta_p$, the curve produces two values of half-power beamwidth. Each of these corresponds to a different antenna size. The value which lies between points B and C on the curve would always be chosen because it represents a smaller antenna and therefore less cost. The actual value of θ_o chosen between points B and C would depend upon the funding available for the ground antenna, but it is obvious that from the station keeping point of view it is best to select the value of θ_o corresponding to point B. It can be seen that for any point on the curve of figure 2.3.9 between points B and C, the worst case allowable error, $\theta_s + \theta_p$, does not exceed approximately one-half θ_o.

Therefore, for future high-power communication satellites, θ_s should be chosen so that $\theta_s + \theta_p \approx \frac{1}{2}\theta_o$, where θ_p and θ_o are representative of ground receivers which may be used in future applications. The diameters of such antennas will probably be in the 6 ft to 10 ft range. For receiving frequencies in the 12-14 GHz range, the half-power beamwidth could be as small as 0.5^o. The pointing error θ_p is due to initial acquisition error, wind deflection, snow and ice loading, thermal distortion, refraction, and diffraction. The total pointing error from these effects would be approximately $\pm 0.2^o$. This error could be reduced by using a more expensive antenna, but in applications requiring many ground antennas, the added expense may be prohibitive. With $\theta_o = 0.5^o$ and $\theta_p = 0.2^o$, a reasonable value for θ_s is 0.07^o. A station keeping error of $\pm 0.05^o$ in the north-south direction and $\pm 0.05^o$ in the east-west direction will yield a value of 0.07^o for θ_s.

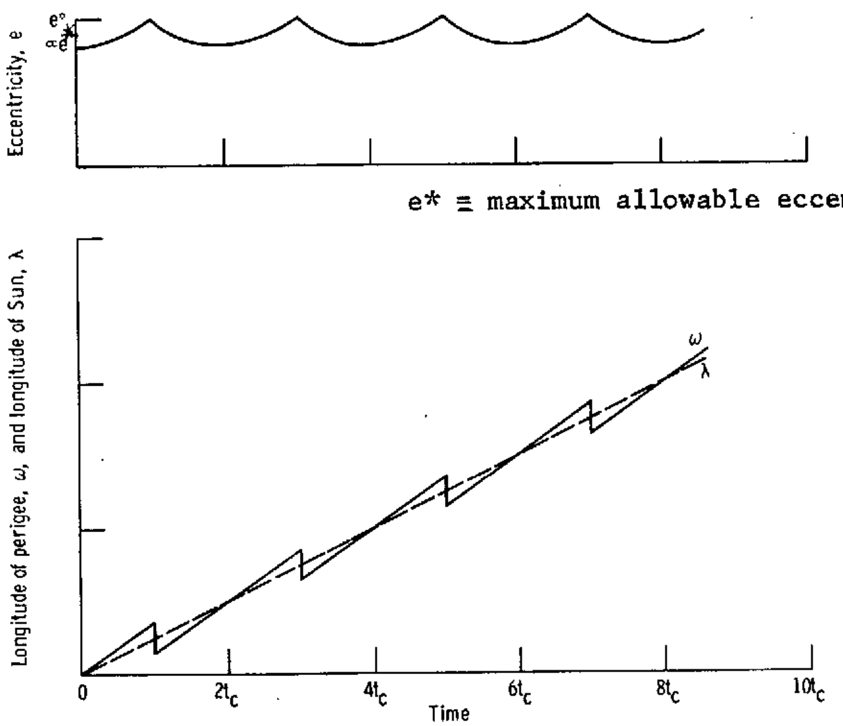
Thus, to demonstrate station keeping to the degree of accuracy which may be required of future high-power communications satellites, the station keeping tolerances should be in the range of $\pm$ 0.05 degrees in both the east-west and the north-south directions. Maintaining the latitudinal variations within the $\pm$ 0.05 degree band for SERT C should be possible, because daily corrections are planned in order to obtain maximum station keeping efficiency using the ion thrusters. Because the inclination change due to the lunar and solar attractions is about .002 degrees per day, it should not be difficult to maintain inclination within about the error tolerance associated with determining inclination (approximately $\pm$ .02 degrees).

Longitudinal station variations are caused primarily by two perturbation sources. The Earth's triaxiality causes a long period (about 2.2 yrs) oscillation around the Earth's minor axis points. Solar pressure causes a variation in orbit eccentricity which results in daily longitudinal oscillations, the magnitude of which are functions of the spacecraft area to mass ratio and reflectivity. For SERT C the maximum variation due to this effect would be about .16 degrees if uncontrolled. Therefore, because the variation due to triaxiality cannot be held to zero, it will be necessary to actively control the solar pressure variation to less than .05 degrees.

Previous analyses have indicated that controlling the longitudinal variation due to Earth triaxiality to .05 degrees may be difficult to achieve because of errors inherent in the orbit determination process. Therefore, in order to minimize the total east-west station variation it may be desirable to minimize the solar pressure variation as much as possible. A complete, detailed analysis and tradeoff study must be performed which will result in an east-west station keeping operational plan which minimizes the total east-west variation, in order to arrive at the actual longitude variation which can be maintained. Until such a study is complete, it will be assumed that the total east-west variation will be maintained within $\pm$ 0.1 degree.

TABLE 2.3.1 - ON-ORBIT OPERATIONS

T_o	Arrive at 60° West Longitude,
T_o	Start Station Keeping and Attitude Control Experiments
T_o + 60 Days	Station Walk to 75° West, Perform S/K Experiments at 75° West for 60 Days
T_o + 120 Days	Station Walk to 90° West, Perform S/K Experiments
T_o + 180 Days	Station Walk to 105° West, Perform S/K Experiments
T_o + 240 Days	Initiate Rendezvous Operations
T_o + 2 Years	Begin Long Term Station Keeping Operations



e* ≡ maximum allowable eccentricity

Figure 2.3.1 - Orbit Parameters vs Time when Correcting for Solar Pressure. Initial Conditions: Eccentricity≈e*; Longitude of Perigee, 0; Longitude of Sun, 0.

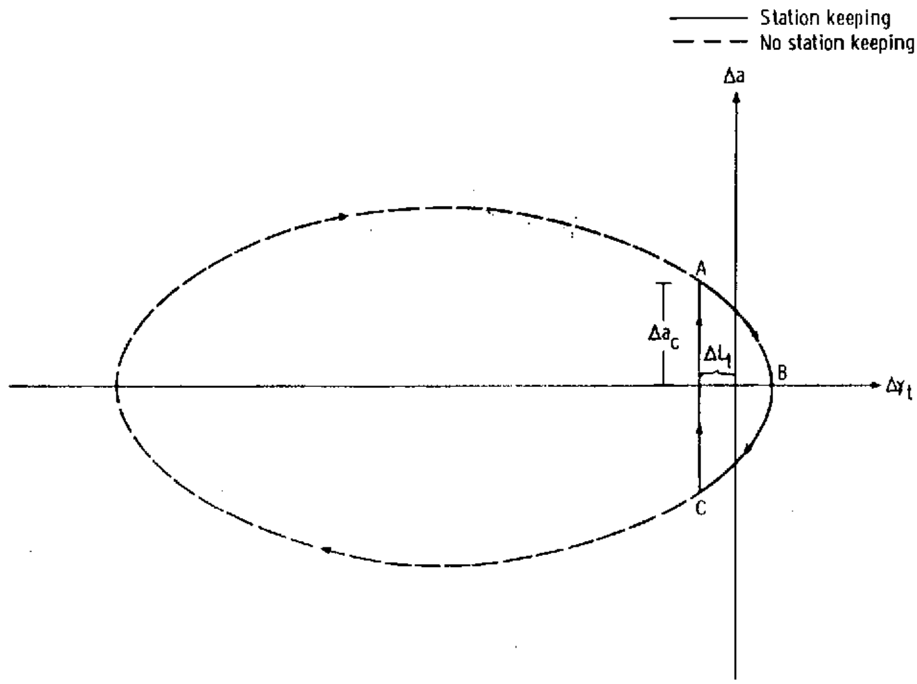


Figure 2.3.2 - Variation in Semimajor Axis as Function of Variation of Satellite Longitude. Perturbation, triaxiality.

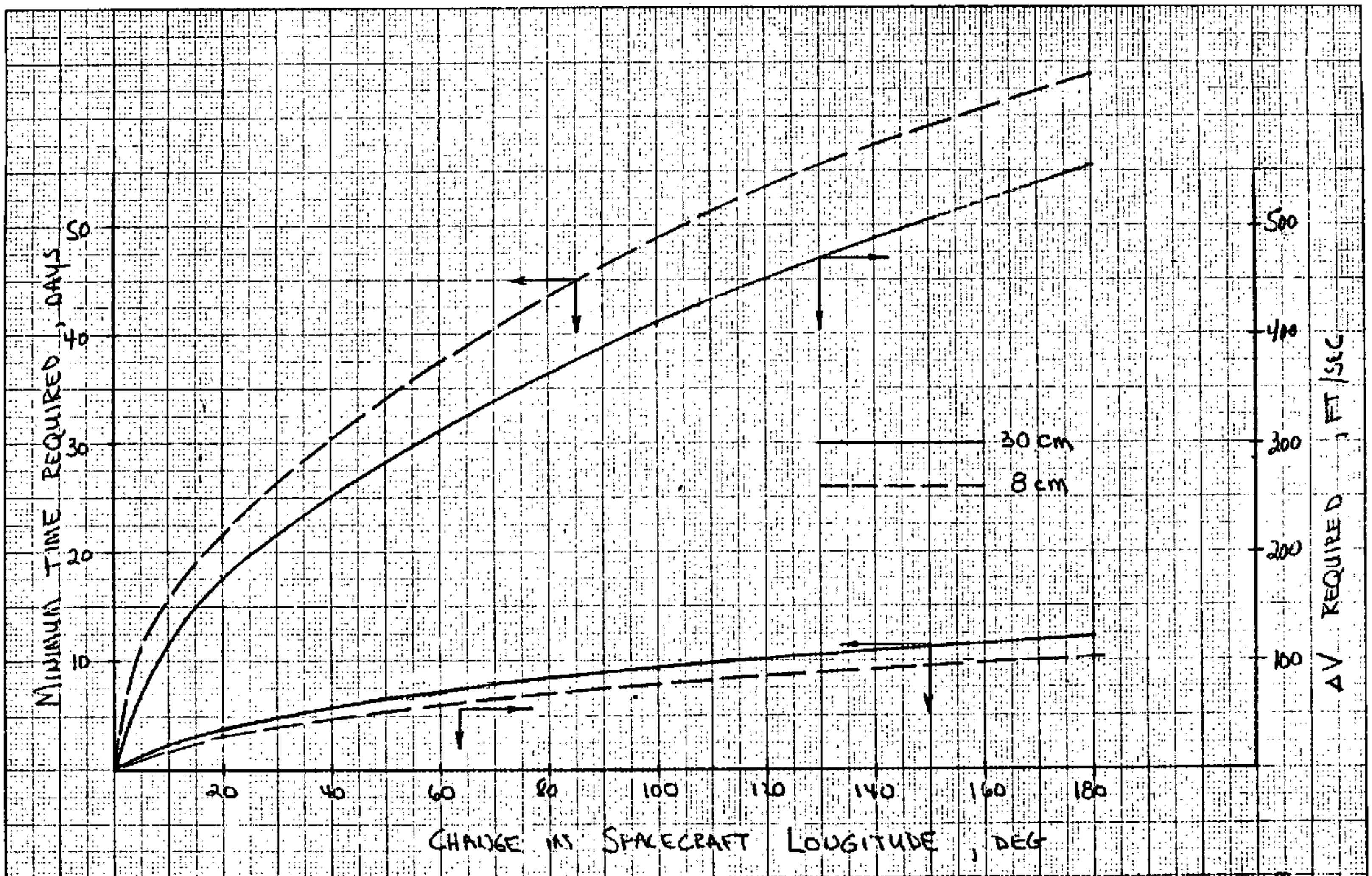


Figure 2.3.3 - Station Walking Capability

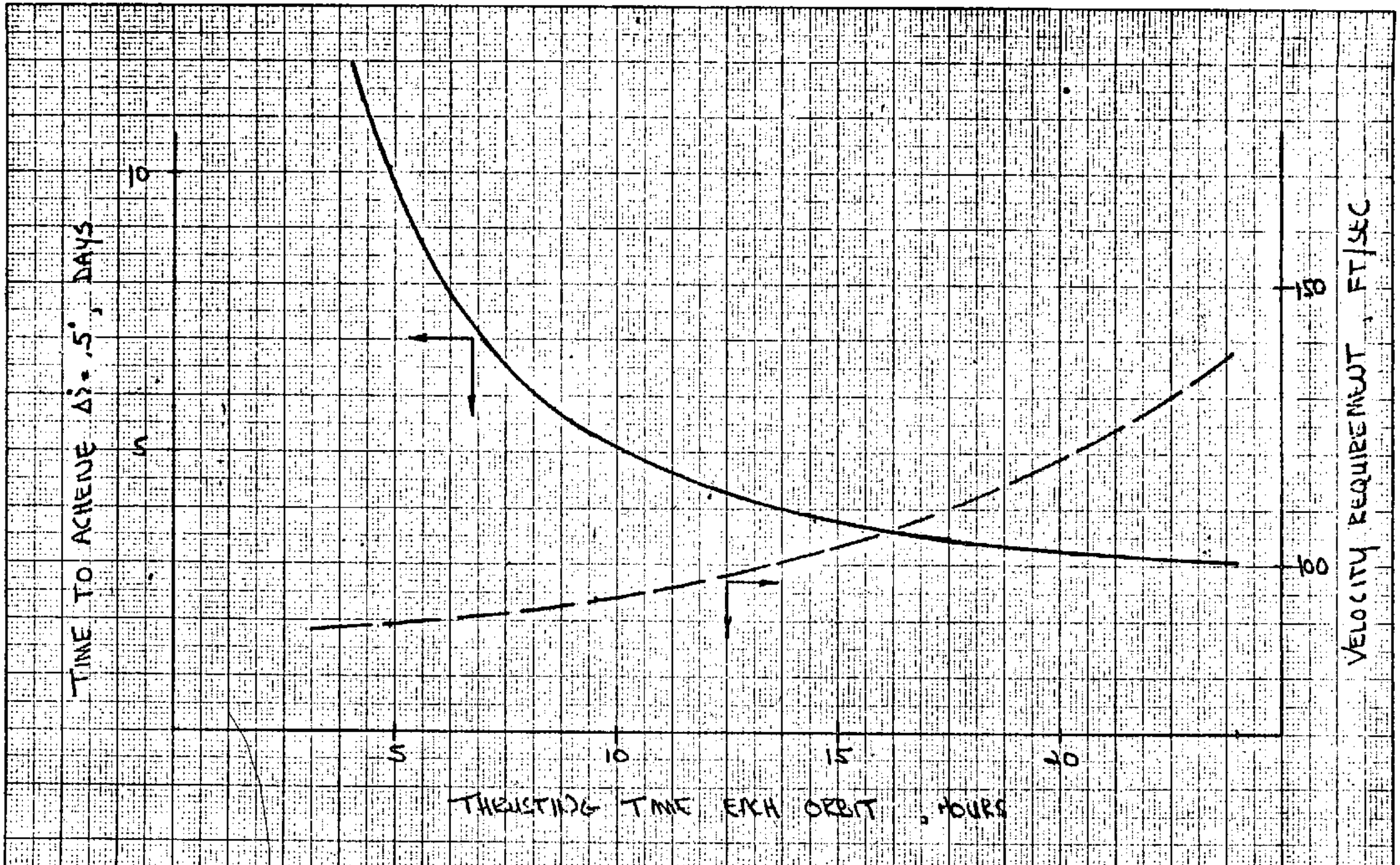


Figure 2.3.4 - Capability of a 30-cm Thruster to Change Inclination by .5 Degrees

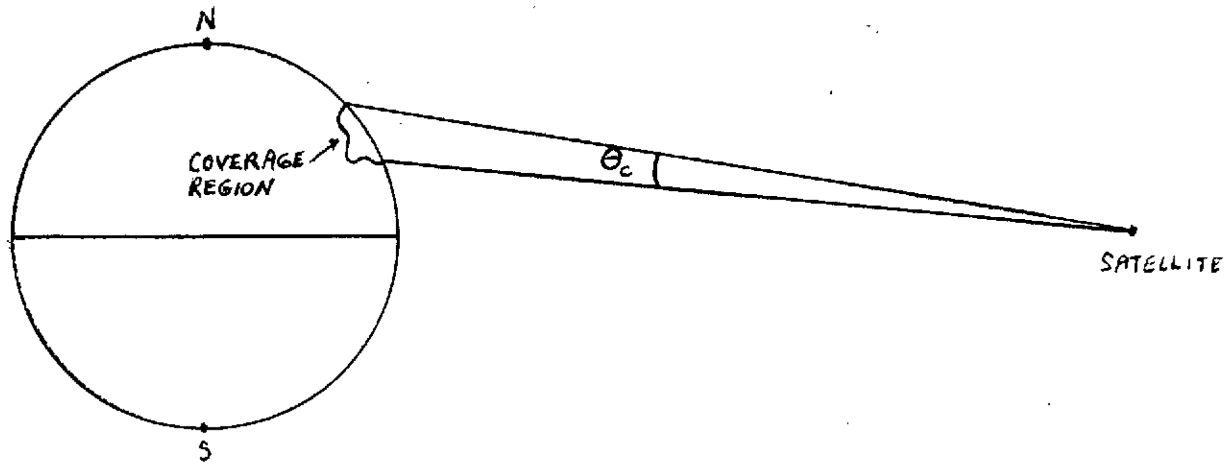


Figure 2.3.5a - Coverage Region as Viewed from Equatorial Plane

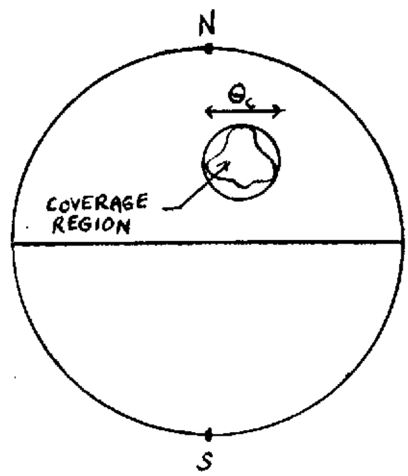


Figure 2.3.5b - Coverage Region as Viewed from Satellite

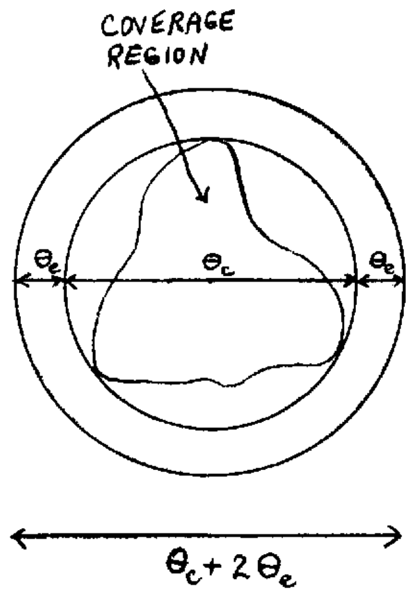


Figure 2.3.6 - Coverage area Enlargement for Single Beam

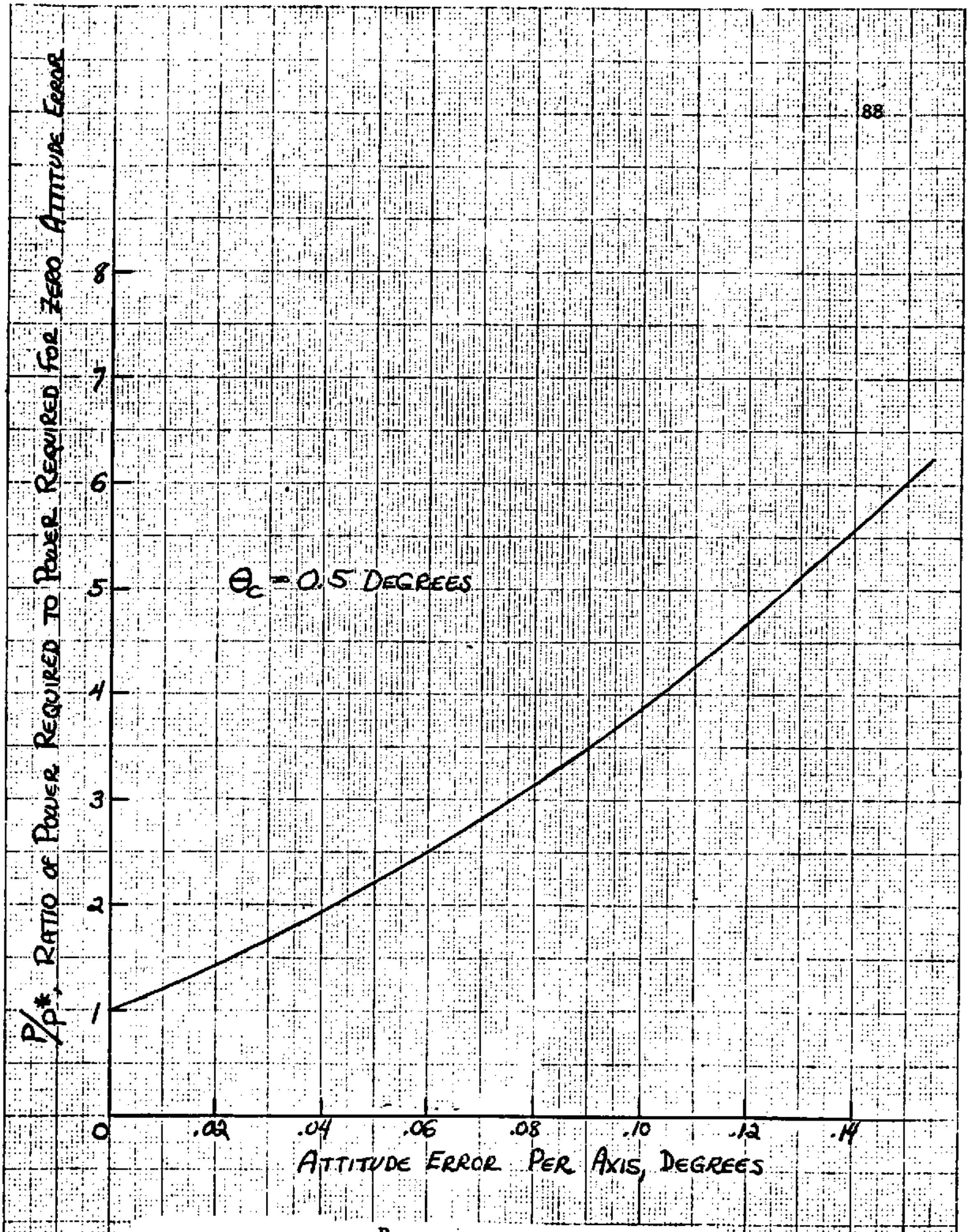


Figure 2.3.7 - $\frac{P}{P^*}$ vs Attitude Error Per Axis

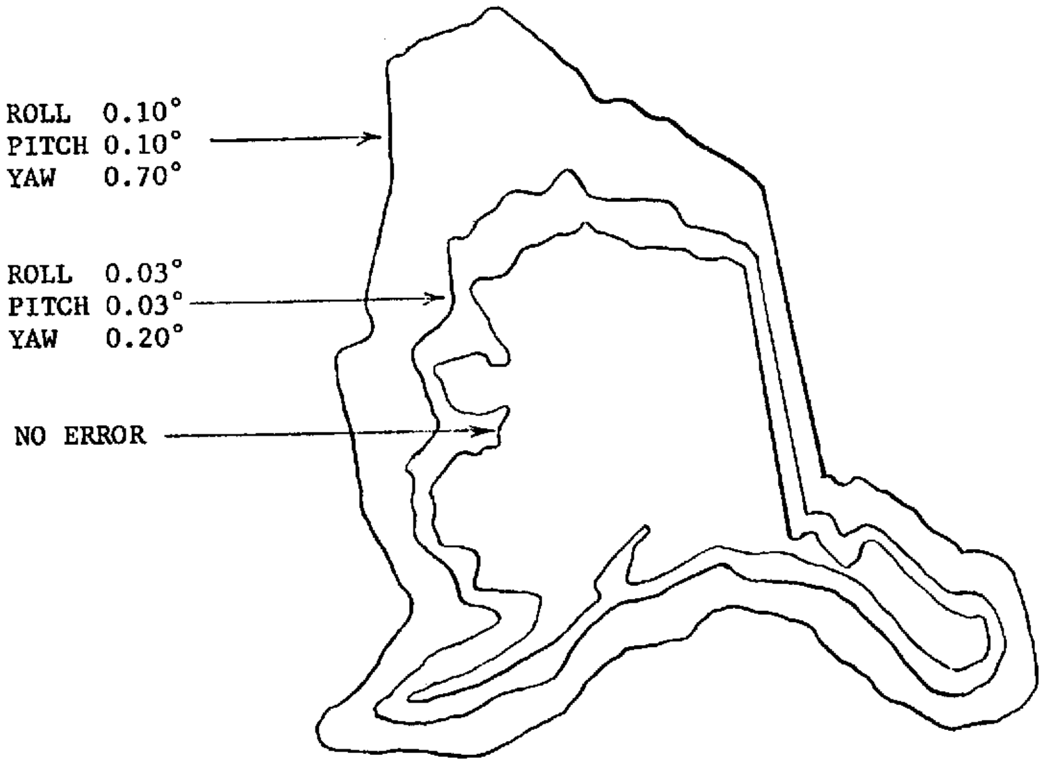


Figure 2.3.8 - Coverage Area Enlargement for Alaska
Satellite Longitude = 130°W

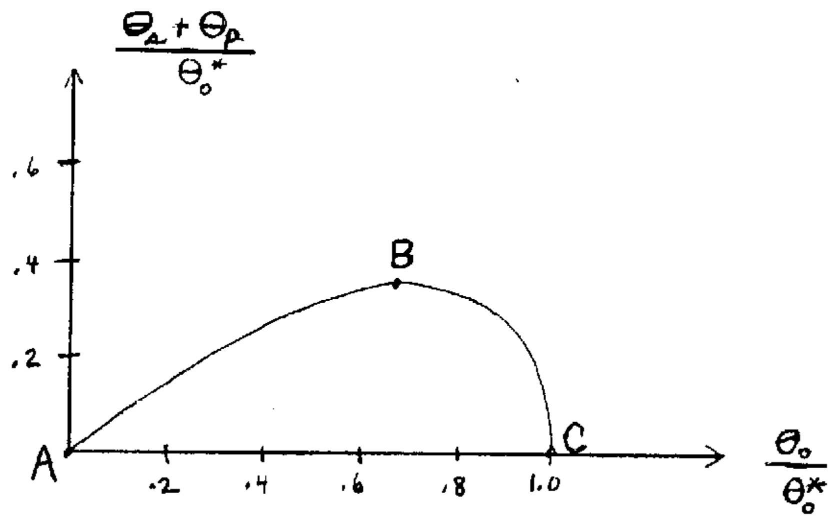


Figure 2.3.9 - Normalized Worst Case Error as a Function of Antenna Half Power Beam width

3.0 SPACECRAFT DESCRIPTION

3.1 Introduction

The baseline spacecraft configuration to be described in this section was arrived at through consideration of mission, system and subsystem requirements arising from several areas of pertinent technology, specifically: attitude control and thrust vector orientation requirements, thermal control requirements, orbit and trajectory requirements, and communications requirements. In most cases, more than one configuration approach was possible to achieve a specific requirement. In arriving at the baseline configuration, the various options were considered along with such factors as simplicity, reliability, and the use of state-of-the-art hardware.

The following sections describe in more detail the spacecraft requirements which exist during the spiral out and synchronous orbit phases, and the resulting baseline and alternate configurations.

3.2 Orbit-Raising Configuration Requirements

Spacecraft Orientation Requirements - The spacecraft orientation requirements during the orbit-raising period are governed by the 30-cm thrust vector pointing requirements. The thrust vector orientation in space during this period is derived from the requirement that orbit semimajor axis be raised to the synchronous value, and inclination and (if necessary) eccentricity be nulled, in the shortest possible time. This leads to the requirement for the thrust vector to oscillate above and below the orbit plane once per orbit. Unless the center of gimbal rotation of the net thrust vector were made to coincide with the center of mass (c.m.) of the spacecraft, obtaining these thrust vector offsets by thruster gimballing would produce large disturbance torques on the spacecraft. Also, the out-of-plane angles required would necessitate gimbal angles (27° to 72°) that are too large to be accommodated by a practical gimbal system. Therefore, thrust vector control must be obtained during orbit-raising by reorientation of the entire spacecraft.

Also, throughout the orbit-raising period, the solar arrays must be oriented as nearly perpendicular to the sunline as possible at all times in order to maximize the array output power and therefore the power to the 30 cm thrusters.

Whereas at the higher altitudes during orbit raising and at synchronous altitude the predominant disturbance torque on the spacecraft is due to solar pressure, at the lower altitudes gravity-gradient torques will dominate. The attitude control system will be required to handle these increased disturbance torques during the early part of the orbit-raising phase.

Orbit and Trajectory Requirements - As previously discussed in section 2.2, the use of the Delta 2910 launch vehicle constrains a circular parking orbit to lie below the Van Allen radiation belts, resulting in a significant decrease in array power due to radiation degradation during the orbit-raising phase. Further, to maintain the time to achieve synchronous, equatorial orbit at a value which provides an adequate but not excessive life test for the 30-cm thruster ($\sim$ one year) requires that the thrust level throughout the orbit raising phase be greater than that of one thruster operating at full power. These two requirements combine to establish the amount of solar array power required at the start of orbit raising.

Thermal Requirements - During the time that the 30-cm thrusters are operating, the primary sources of excess heat which must be rejected by the spacecraft thermal control system results from the inefficiency of the power processors for these thrusters. These power processors will generate initially a total of 900 watts of waste heat which must be rejected by radiation. To minimize the radiation area required, the spacecraft configuration must be such that the thermal dissipation surfaces do not directly face the sun for any significant period of time.

Communications Requirements - Obviously, throughout the orbit-raising phase, the capability must exist both for receiving telemetered data from the spacecraft and for issuing of commands to the spacecraft.

Eclipse Requirements - The spacecraft will periodically pass through the earth's shadow during the orbit-raising phase as described in section 2.2. Because power from the solar arrays will not be available during eclipse, neither the 30 cm or 8 cm ion thrusters will be operated. However, telemetry and command capability must be maintained, the attitude control system must continue to function (although some reduction in accuracy may be tolerable) and spacecraft temperatures must be held within prescribed limits by the thermal control system during these periods.

3.3 Synchronous Orbit Requirements

Spacecraft Orientation Requirements - In synchronous orbit, three major phases exist each of which has its individual spacecraft orientation requirements. These phases are: station keeping and attitude control, station walking, and rendezvous and inspection.

The 8-cm thrust vector must be oriented normal to the orbit plane to achieve north-south station keeping, and must be tangent to the orbit for east-west station keeping.

During station walking, the 30-cm thrusters are operating and the spacecraft orientation requirements are nearly the same as during orbit raising. The differences are that here no large oscillation about yaw is required of the station walking thrust vector, and its direction must be reversed halfway through the maneuver.

The orientation requirements during rendezvous and inspection are the least defined at the present time. However, both in-plane and out-of-plane accelerations will be required to achieve and maintain rendezvous, and it is likely that transfer of primary attitude "lock" from the earth to the target spacecraft will be required.

Common to all the above phases is the requirement that the solar arrays be continuously oriented as near to normal to the sunline as is necessary to provide sufficient power to all the required spacecraft systems and experiments.

Thermal Control Requirements - During the synchronous orbit phase, the 30-cm thrusters will be operating only periodically (during station walking and rendezvous). Also, eclipse periods will occur during which the spacecraft will be in the earth's shadow for as much as 70 minutes per day. These eclipse seasons will occur twice a year around the equinoxes. During eclipse, only necessary systems will be operated because no power will be available from the solar array. Thus, the thermal control system must be capable of maintaining the spacecraft temperatures within required ranges for the two extremes of maximum heat dissipation (during station walking and rendezvous) and minimum heat dissipation (during eclipse).

Communications Requirements - Again, during all synchronous orbit phases, both telemetry and command communications links must be maintained between the spacecraft and ground stations. During rendezvous and inspection, however, an additional communications link must be established between SERT C and the target spacecraft. Complicating this and bearing on the communication system configuration is the fact that, in general, both of these links will be required simultaneously while the two spacecraft are moving relative to each other.

Eclipse Requirements - During eclipse periods the requirements will be the same as during orbit raising. That is, communications, both telemetry and command, attitude control, and thermal control must all be maintained.

3.4 Baseline Configuration and Trade-Offs

Several basic spacecraft configurations were considered in detail prior to selection of the baseline configuration described above. These are shown in figure 3.4.1. The configuration labeled I is the baseline configuration chosen. However, each of the others has some advantages which will be discussed below, along with the disadvantages which resulted in it being dropped.

Baseline Configuration

The baseline spacecraft configuration comprises a center body which contains most of the spacecraft systems and two deployable solar arrays as shown in figure 1.3.1. Because of the large solar array areas required, two arrays were chosen mounted on opposite sides of the centerbody, to minimize solar radiation pressure disturbance torques on the spacecraft and for other reasons. During orbit raising, roll and pitch offsets are held to zero in accordance with the steering law. This results in the spacecraft yaw axis being pointed continuously at the center of the earth. Thus, one face of the centerbody must be continuously earth oriented throughout this period. Likewise in synchronous orbit, the high gain antenna and therefore the face on which it is mounted, must continuously face the earth. (A different face is earth oriented during orbit-raising than in synchronous orbit for reasons which will be discussed later). With this configuration each solar array is capable of continuous rotation about an axis perpendicular to its mounting face to allow it to be continuously oriented toward the sun. The arrays are stowed during launch as shown in figure 1.3.4 and are deployed following injection into the low altitude parking orbit.

The center body is a rectangular parallelepiped in shape of dimensions 3.3 foot by 4.5 foot by 8 foot. Let a coordinate system, with its origin located at the spacecraft center of mass (c.m.) be defined as follows: the x - axis (roll axis) lies in the orbit plane and has the same sense as the orbit velocity vector; the z - axis (yaw axis) is coincident with the earth radius vector and directed toward the earth; the y - axis (pitch axis) is perpendicular to the orbit plane and directed south. For zero attitude errors, the center body principal axes are aligned with this coordinate system. Of the six spacecraft sides, one faces radially inward and one radially outward. The other four face the north, south, east, and west directions.

Three 30-cm ion thrusters are attached to the face of the center body which is oriented in a westerly direction during orbit-raising. These thrusters will be used for orbit-raising and station walking. Two thrusters operate simultaneously during the early orbit-raising phase, the third providing redundancy. The three thrusters are arranged in a triangular array as shown in figure 1.3.1. Each thruster is canted slightly inward to align its nominal thrust axis through the spacecraft c.m. All three thrusters are mounted on a gimballed platform which has two degrees of rotational freedom relative to the spacecraft. This arrangement not only permits adjustment of the net thrust vector of the two operating thrusters to lie through the c.m. but also provides for obtaining control torque about the pitch and yaw axes during the early portion of the orbit-raising phase when the disturbance torques are high. By proper initial placement of the triangular array about the spacecraft roll axis, and by proper gimballing, depending upon which thruster or thrusters are operating, the thrust loss due to the inward cant of the thrusters can be held to about 1.5 percent maximum.

The solar arrays are mounted to the north and south faces of the centerbody so that their rotation axes lie north and south. Thus as the centerbody rotates at one revolution per orbit to maintain its earth-facing orientation, the solar arrays can be kept pointed in the direction of the sun by rotating them relative to the centerbody. The array rotation axes pass through the spacecraft c.m.

Four 8-cm ion thrusters are located on the spacecraft. Two of these are to accomplish north-south station keeping and therefore their thrust vectors must be oriented north and south. Because both the array rotation axis and the 8-cm thrust axis must pass through the spacecraft c.m. the 8-cm thrusters are located on the outer ends of the two array wings. They therefore provide both north and south accelerations for station keeping. Mounted with each thruster on the end of the array is its propellant tank and power processor. Because the thrust beams are deflectable ± 10 degrees in each of two mutually perpendicular axes the thrusters can be used to provide momentum dumping for the attitude control system as described below. The array mounted thrusters can provide torques about the roll and yaw axes by beam deflection. For this purpose, mounting the thrusters on the ends of the array provides a much larger moment arm and therefore a much larger dumping torque.

The other two 8-cm thrusters are located on the centerbody. A 90 degree rotation of the center body about pitch is made once synchronous, equatorial orbit is achieved. That is, in the synchronous orbit orientation, the 30-cm orbit raising thrusters will be located on the radially outward facing side of the center body. This will orient the two body mounted 8-cm thrusters east and west. These two thrusters can provide torques about the pitch and yaw axes as well as station keeping accelerations and will be used to provide pitch momentum dumping torques.

A common propellant supply system will be used for the orbit-raising thrusters and for the two body-mounted 8-cm thrusters. The propellant tank is designed in such a way that its c.m. is located at the c.m. of the spacecraft and so that its c.m. does not shift significantly as propellant is consumed.

The high-gain communications antenna is located on that side of the centerbody which is opposite to the 30-cm thrusters. Therefore, it faces earth after the 90 degree pitch rotation of the centerbody in synchronous orbit is performed. This location is the best one from the point of view of allowing room for growth of the antenna in future missions and also for avoiding the antenna feed structure launch load problems inherent in a side-mounted antenna.

The radar antenna must be located such that its axis is nominally 90 degrees from that of the high gain antenna to permit some maneuvering around the target spacecraft while maintaining earth lock of the high gain antenna. It can therefore be located on either the east or west face (synchronous orbit configuration). Its location, plus the amount of antenna gimballing available, will have a significant effect on the nature of the target spacecraft rendezvous maneuver. A more detailed study is therefore required before a final decision can be made as to radar antenna placement. The location of the TV camera allows observations of the SERT C solar arrays as well as of the target spacecraft to be made.

The north and south centerbody faces will be used as the thermal control surfaces because these are the only faces which never look directly at the sun. The 30-cm thruster power processors, which are designed to be self radiating, are located on these surfaces. The use of these faces as thermal control surfaces together with the dimensions of the launch vehicle aerodynamic shroud are the primary factors which govern the choice of the dimensions of the centerbody.

Trade Offs

Configuration I - All the spacecraft configurations examined were capable of satisfying mission objectives. Also, each had certain advantages and disadvantages. The baseline configuration, however, appeared to be the most straight forward approach in terms of design simplicity, satisfaction of design constraints, ease of accomplishing mission objectives, and growth or application potential.

Comparison of the configurations and the trade offs considered are shown in figure 1.4.5. The disadvantage listed in many instances arose from design decisions made. As a consequence, the disadvantages could be circumvented by modifying the basic design decision, but only at the expense of complicating the spacecraft. Such added spacecraft complexity would be unacceptable, however, because the guiding philosophy in all decisions was to minimize spacecraft complexity.

In general, configuration I offered maximum simplicity with the best antenna (radar and high gain) and TV camera configuration for rendezvous. It avoided the side mounted high gain antenna shroud and launch load constraints of configuration II and has the greatest potential for adaptation to future missions requiring larger antennas. The configuration was also best for station keeping purposes without compromising the functions performed by the primary thrusters.

The disadvantages of the baseline configuration are that (1) a pitch maneuver of the centerbody is required when using the 30-cm thrusters for station walking and rendezvous, (2) the fixed high-gain antenna is not available during orbit raising, (3) earth sensors fixed to the spacecraft with different orientation are used for the orbit raising and synchronous orbit phases thus reducing the usefulness of these sensors in a back up mode, and (4) some power loss is encountered because the solar incidence angle is not always 90 degrees to the plane of the solar array.

Most of the above disadvantages arose as a result of design decisions that were not necessarily dictated by the basic configuration. For example, it was not deemed desirable to provide a movable high-gain antenna so that it would be available during orbit raising since it is not required during that phase of the mission. It was also not deemed desirable to provide earth sensor rotation even if sensors could be found that satisfied both orbit raising and synchronous requirements. The power loss associated with the solar incidence angle is common to all configurations except configuration IV. This power loss had a minimal effect on the mission and was considered acceptable to avoid the complication of adding a second degree of rotation to the solar arrays. In general, the advantage of configuration I lay in the fact that it eliminated most of the disadvantages noted subsequently for the other configurations. Since selection of configuration I did result from a trade-off study, these trade offs will be evaluated further when more detailed design efforts are conducted.

Configuration II - Configuration II is similar to the baseline configuration except that the centerbody does not rotate 90 degrees in pitch after synchronous orbit is attained. That is, the same centerbody side faces earth both during orbit raising and in synchronous orbit. The high-gain antenna is located on the earth-facing side. One of the body mounted 8-cm thrusters is located on the east face of the centerbody and the other is located on the radially outward side. One advantage of this configuration is that the high-gain antenna faces earth during orbit raising and when using the 30-cm thrusters for station walking and rendezvous. A 180° yaw maneuver is required for all configurations when the 30-cm thrusters are used for station walking. The disadvantage is that east west station keeping cannot be done entirely with the 8-cm thrusters because only a westerly acceleration is available. The easterly acceleration would be obtained either by operating a 30-cm thruster (resulting in very short operating times) or by rotating the spacecraft through 180 degrees about the yaw axis and using the 8-cm thruster. Alternatively, an 8-cm thruster could be placed on the west face. However, it is felt that significant design problems could be encountered in attempting to place the 8-cm thruster in the center of the 30-cm thruster array. Because the axis of the high-gain antenna is perpendicular to the longitudinal axis of the shroud envelope, shroud limitations could be encountered if a larger or gimballed high-gain antenna were ultimately desired. This antenna orientation is also unfavorable from a launch loads standpoint.

Configuration III - This configuration is similar to configuration II, except that all four 8-cm thrusters are mounted on the centerbody as shown in the illustration. The main advantages of this configuration are that it minimizes the weight located on the ends of the arrays thereby decreasing the possibility of having undesirable interactions of the flexible arrays with the attitude control system, it permits common propellant tankage for all ion thrusters, and it results in minimum slip ring requirements between the arrays and centerbody. There are a sizable number of disadvantages, however. First, because the thruster axes are not perpendicular to the array longitudinal axis, beam divergence would result in array contamination concerns unless beam divergence shrouds were provided on each thruster. Use of such shrouds would represent additional development effort. Second, any 8-cm thruster operation, N-S or E-W station keeping or momentum dumping requires two thrusters operating simultaneously. Third, to minimize the array contamination problem would probably require reducing the width of the array which would mean increasing an already high array aspect ratio. Fourth, the configuration is the least desirable from the viewpoint of the directly regulated solar array experiment because this is the only configuration where the solar array and the thruster it powers are not located on the same structural unit. Therefore, high voltages from the array and switch leads for array reconfiguration must be brought through the slip rings. Fifth, the thruster moment arm for momentum dumping is much smaller compared to that for configurations I and II which means increased thrusting time to accomplish the dumping operation. Sixth, because the angle between the thrust axis and the north-south direction is large, to attempt to minimize array contamination, the thruster operating time is significantly longer to provide north-south station keeping for a period of 5 to 10 years. This configuration was felt to be the least desirable overall of the four examined.

Configuration IV - The first three configurations all have the disadvantage that the solar arrays cannot be maintained normal to the sunline at all times, and therefore a loss in average array output power occurs. This is because the arrays have only one degree of freedom relative to the centerbody which is constrained to be earth facing. Configuration IV adds a second articulation. All components which are required to face the earth continuously are mounted in a pod which can rotate with unlimited freedom about an axis perpendicular to the array rotation axes as shown in figure 3.4.1. The 30-cm thrusters, propellant tanks and four 8-cm thrusters are mounted in the nonrotating part of the centerbody. During orbit raising, this configuration would be oriented with the array rotation axes north and south as in the other configurations. To keep the arrays normal to the sunline as the spacecraft traverses its orbit, the spacecraft would rotate about roll (thrust axis and pod rotation axis) until a pure solar array rotation orients the arrays normal to the sun. As the main centerbody rolls, the pod rotates so as to keep antennas, earth sensors, etc. fixed on the earth. Once synchronous orbit is reached and the 30-cm thrusters no longer are required to run continuously, the spacecraft would be rotated 90 degrees about yaw until the array rotation axes were oriented east and west. The array rotation axes would thenceforth be held perpendicular to the sunline and the arrays rotated to follow the sun as its declination varies. The pod would then rotate once per day to keep antennas and earth sensors fixed on the earth. For station walking and rendezvous, the spacecraft would be rerotated to the orbit raising orientation.

The main advantage of this configuration is that it permits the maximum power to be obtained from the array at all times. This means that approximately 10 percent less time is required to achieve synchronous orbit than for the other three configurations (assuming all configurations weigh the same). Among the disadvantages, both the on-orbit earth sensor and the high-gain antenna must be elevated 8-10 feet above the top edge of the solar array to avoid having their fields-of-view intercepted by the solar array twice daily. This configuration does not provide a simplified or realistic demonstration of the station keeping operations. The centerbody must be rotated to provide the proper thrust directions for east-west station keeping. This results in some loss in power. Because the other two 8-cm thrusters are directed south, north-south station keeping can be accomplished only over one half an orbit period. As with configuration III, moment arms for momentum dumping are small. Mounting the earth sensor on a tower to elevate it could result in relative motion between it and the centerbody with consequent reduced control accuracy. A third rotating joint, orientation device, and set of slip rings is required with resultant possible impact on overall spacecraft reliability. A roll control program which produces large roll offset angles is required to keep the arrays continuously oriented normal to the sunline which would complicate the attitude control system, possibly to the extent that the 10 percent additional power obtained would not be worth the increased complexity. Finally, because there are no sides of the centerbody which at one time or another do not look directly at the sun and because of the rotating joint across which heat may have to be transferred, it was felt that significant undefined thermal problems may exist with this configuration.

Summary - In summary, each of the configurations investigated had advantages and disadvantages considering both the accomplishment of the mission objectives and the development of the spacecraft systems. The baseline, configuration I, provides for the best demonstration of synchronous orbit attitude control and station keeping with ion thrusters. It also provides the optimum antenna/camera configuration. The spacecraft development appears to be simple compared to that for the other three configurations, each of which has potential development problems. The above and other considerations weighed against the disadvantages of configuration I led to the selection of configuration I as the baseline for the SERT C mission.

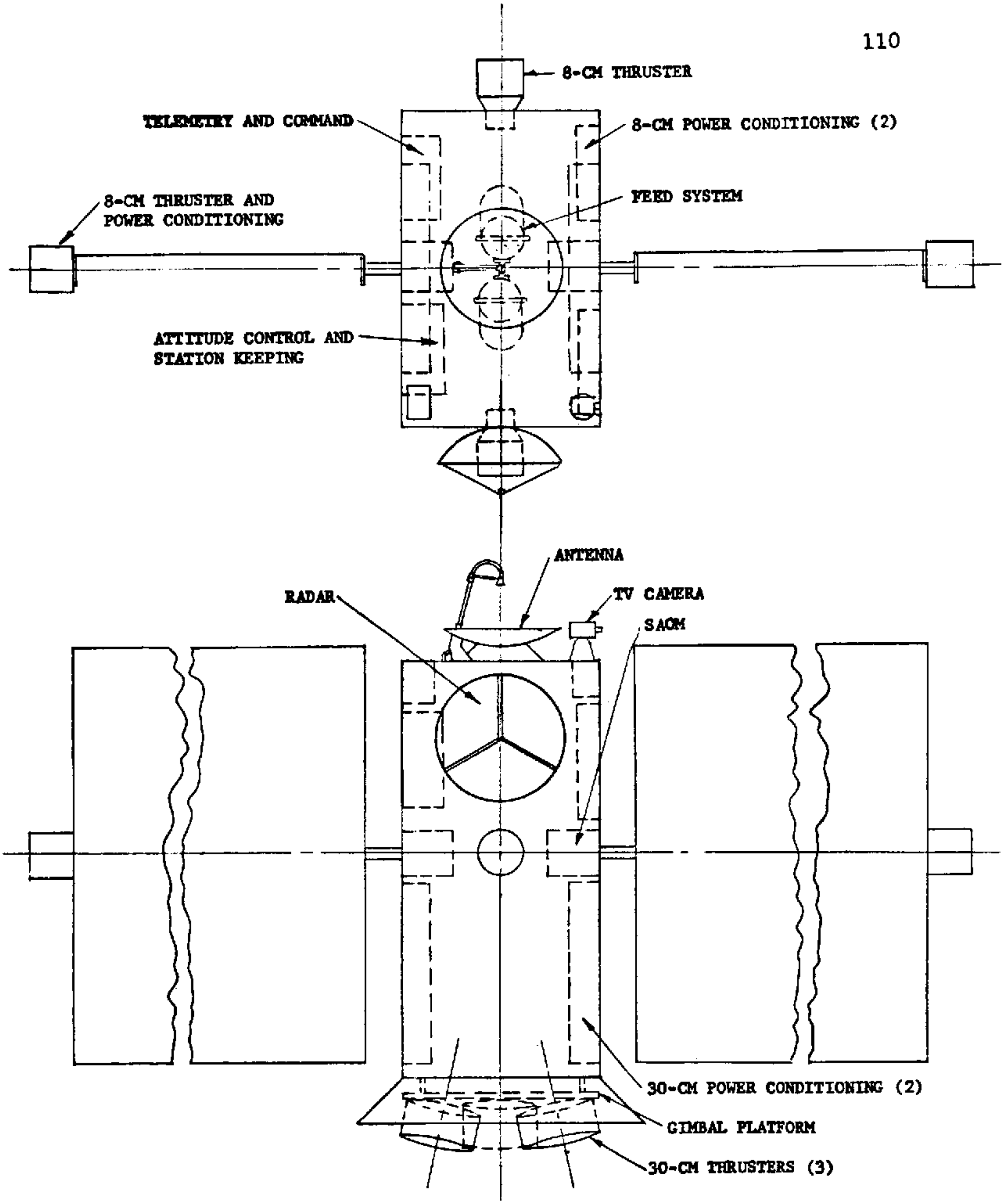


Figure 3.4.1a - SERT C Configuration I

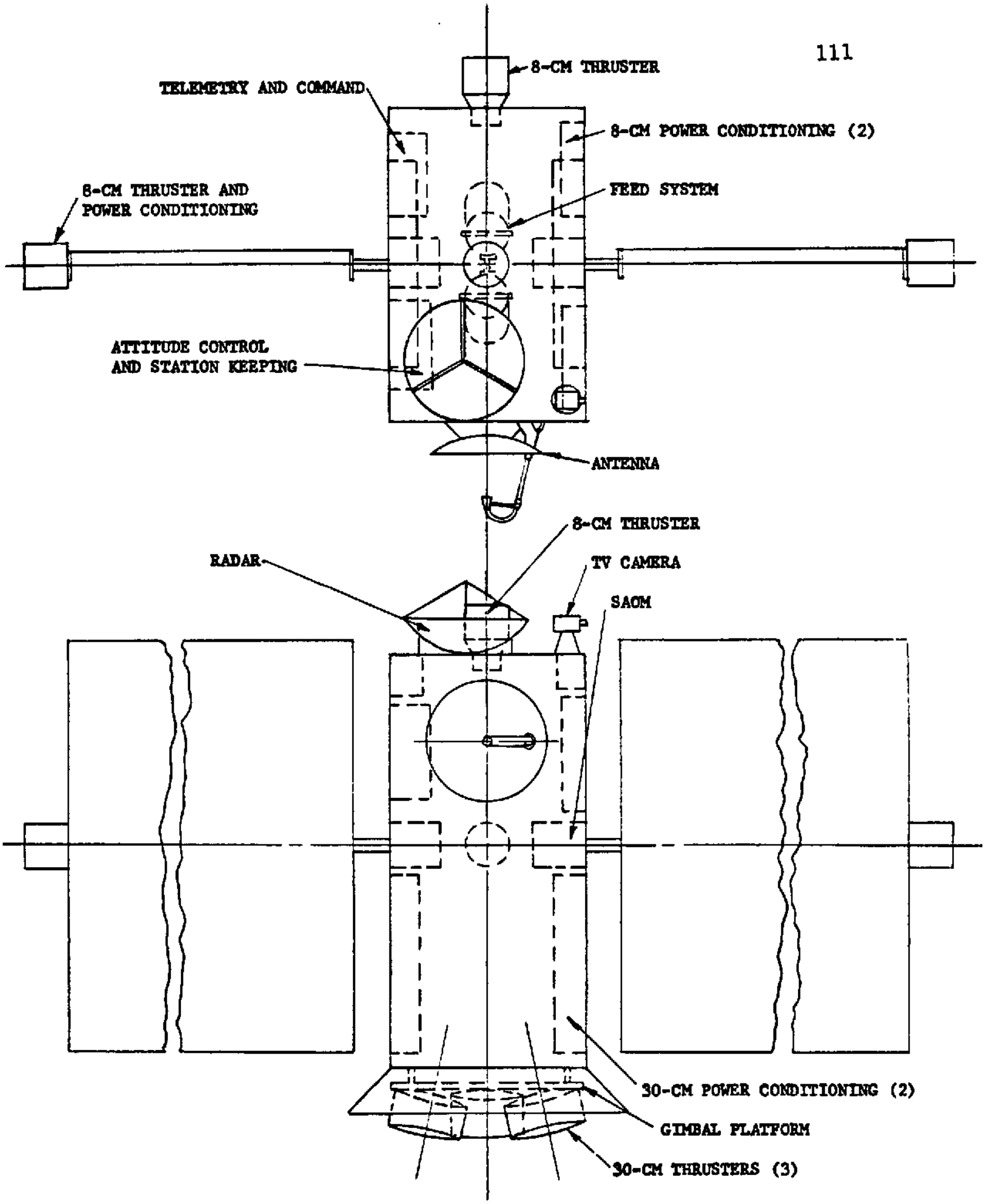


Figure 3.4.1b - Configuration II

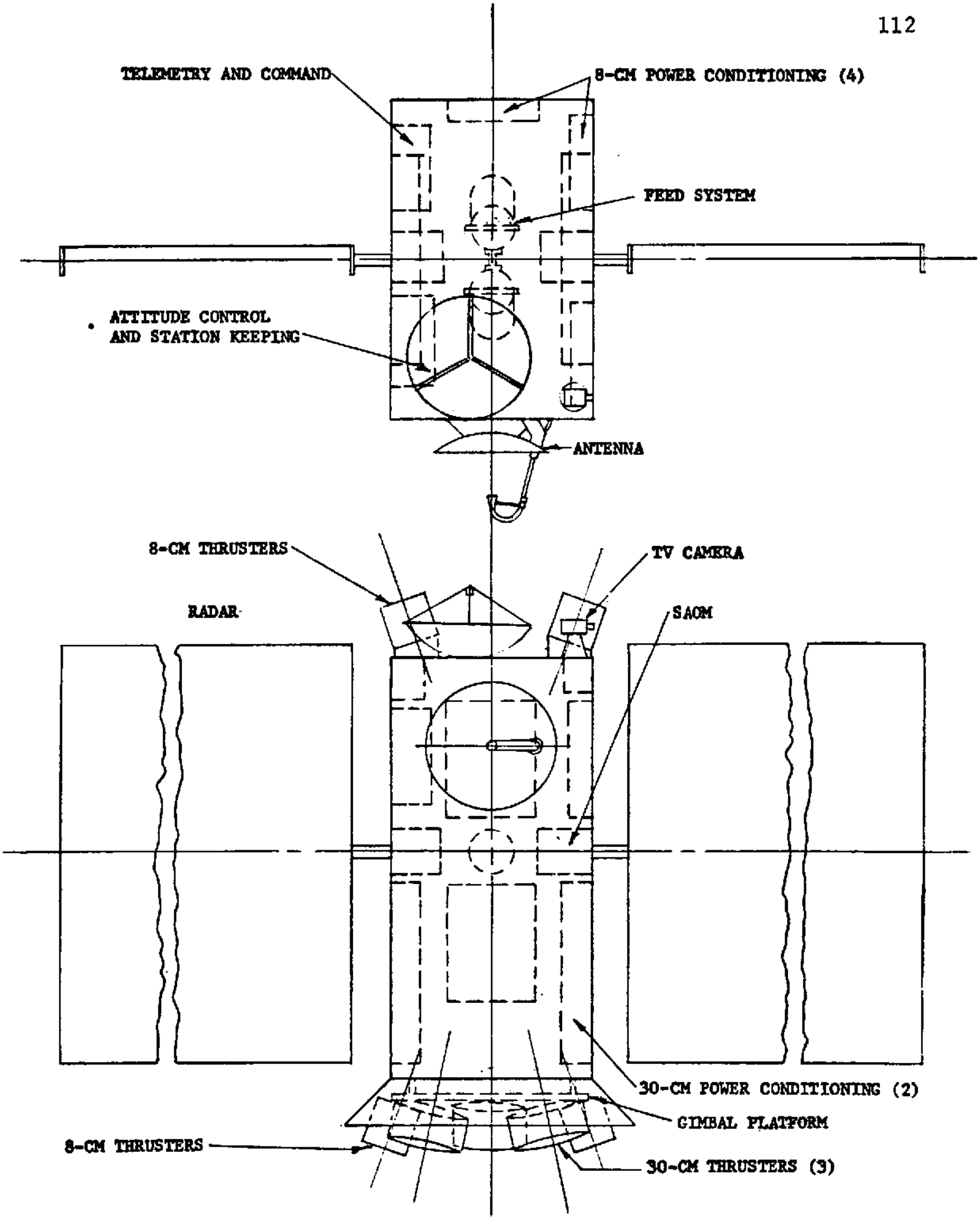


Figure 3.4.1c - Configuration III

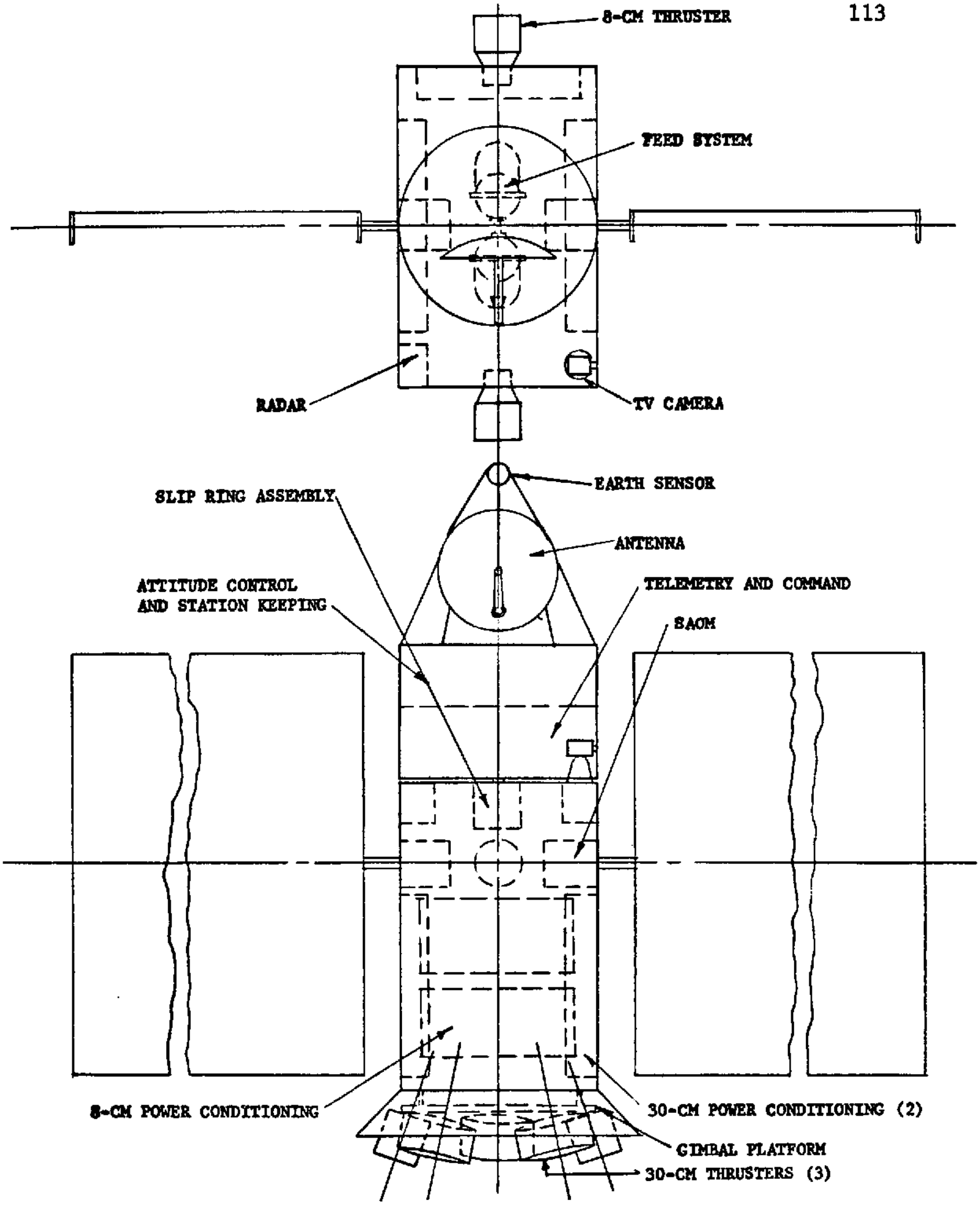


Figure 3.4.1d - Configuration IV

4.0 SUBSYSTEM DESIGN

4.1 Attitude Control

Purpose - The purpose of the attitude control system on the SERT C spacecraft is to provide for maintaining the spacecraft in its prescribed orientation during orbit raising and in its prescribed orientation and station in synchronous orbit within desired tolerances, to provide a test bed with which to demonstrate the capability of the ion thrusters to perform in an accurate attitude control and station keeping system, and to provide a flight proven attitude control system for future use of the SERT C as a spacecraft bus to which additional systems could be added.

Disturbance Environment - The disturbance torque environment with which the Attitude Control System (ACS) must contend is given in table 4.1.1 which lists the predominant disturbance torques and their approximate magnitudes. The ACS must provide 3-axis torques to counteract these disturbances and to unload the angular momentum imparted to the spacecraft by the secular components. It will be noted from table 4.1.1 that at the parking orbit altitude the gravity-gradient torques predominate over those from solar pressure. The solar pressure torque, however, is fixed in inertial space whereas the gravity-gradient torque is not. As a result the momentum imparted to the spacecraft by the gravity gradient torques is bounded as a function of time while the momentum due to solar pressure has a secular component which increases with time. The gravity-gradient torques decrease as the cube of the orbit radius, whereas the solar pressure torques are independent of altitude.

The size, or momentum storage capacity, of the reaction wheels is determined by the magnitude of the disturbance torques which they must compensate and the length of time between wheel dumping. In order to keep the wheel sizes at a minimum, it has been assumed that the wheels are required to accommodate only those disturbances encountered in synchronous orbit (i.e., essentially sized by solar pressure torques) and that an interval between wheel dumpings of one day is acceptable. Even with these assumptions, the momentum storage capacity must be about 12 lb-ft-sec, which is a rather large wheel. However, if a smaller wheel were chosen, it would require dumping more than once daily, which may be operationally undesirable. At the lower altitudes where gravity-gradient is significant, the gimballed 30-cm thrusters will be used to provide additional control torque to accommodate the additional disturbance torque. During the coast periods, of course, the 30-cm thrusters are not available, and an increase in dumping frequency may be required. The pitch wheel will be considerably smaller than the yaw and roll wheels because of the lower disturbance torque in that axis in synchronous orbit. The pitch wheel size will be about 3 lb-ft-sec. Note that, because of the 90 degree center body rotation in synchronous orbit, the functions of the yaw and roll reaction wheels interchange, the orbit raising yaw wheel becoming the synchronous orbit roll wheel, and vice versa. The switches shown in figure 4.1.1 are in their synchronous orbit positions. The 8-cm ion thrusters are used to unload the reaction wheels periodically, both during orbit-raising and in synchronous orbit, and to provide translational accelerations for station keeping. Two of these thrusters are located on the two spacecraft faces which are oriented east and west in synchronous orbit. The other two thrusters are located on the tips of the north and south panels. Each thruster has the capability of deflecting its thrust vector ± 10 degrees in two perpendicular axes. The thrust vector for zero deflection is nominally oriented through the spacecraft center of mass. The array-mounted thrusters are used to provide N-S station keeping in synchronous orbit and to provide torques to unload the roll and yaw reaction wheels during all mission phases. The body mounted thrusters provide a torque to unload the pitch reaction wheel. During orbit raising, yaw and pitch momentum dumping could be provided by the 30-cm thrusters and roll dumping by the body-mounted 8-cm thrusters. However, the latter operation would require that both of these 8-cm thrusters run about 25 percent of the time. It appears more desireable to use an array mounted thruster to perform roll and therefore also yaw dumping during this time. The body mounted thrusters also provide backup yaw torques in synchronous orbit, although of much smaller magnitude than that of the array-mounted thrusters. East-west station keeping is accomplished with the body mounted thrusters.

Attitude error sensing is provided by two-axis earth sensors located on the earth-facing sides of the center body, two-axis sun sensors located on the solar arrays, and single axis integrating gyros in the center body. The earth sensors provide error signals about the roll and pitch axes. The sun sensors provide the yaw error over a large portion of the orbit, and the integrating gyro, which has its input axis aligned with the yaw axis, "fills in" for the sun sensor for those periods when the sun sensor does not provide yaw error with sufficient accuracy. The rate gyro package is used in conjunction with the sun sensor and earth sensor during attitude reacquisition.

The attitude control system section of the on-board computer accepts the various attitude error signals, performs the required compensation and amplification, and provides control signals to the reaction wheels. It also performs the necessary calculations to transform the sun sensor outputs into a yaw error signal, and to determine the required yaw offsets during orbit raising as described in the mission outline section. Synchronous orbit station keeping and wheel unloading are of sufficiently low frequency (once per day) and duty cycle that it may be possible either to command these operations from the ground or program them into the on-board computer. Table 4.1.2 shows a weight breakdown of the ACS. In accomplishing the design and fabrication of this system, one of the primary goals is the use wherever possible of state-of-the-art technology, off-the-shelf hardware, and minimum new development.

System Operation - The attitude control system will be required to operate in several modes during the various mission phases. These are:

1. Parking orbit and thruster startup
2. Orbit raising
3. Normal synchronous orbit
4. Stationkeeping
5. Momentum-dumping
6. Eclipse
7. Stationwalking
8. Target spacecraft rendezvous and inspection
9. Reacquisition
10. Backup

Discussion of the system operation during each of the above phases follows.

Parking Orbit and Thruster Startup - After injection into the parking orbit, the DELTA second stage control system will be used to orient the spacecraft to the desired attitude, which has the spacecraft roll axis (thrust axis) parallel to the velocity vector, the pitch axis (solar array axis of rotation) aligned perpendicular to the orbit plane, and the yaw axis parallel to the orbit radius vector. The Delta second stage control system will hold the spacecraft in the desired attitude for about 2 hours after injection (longer times can be obtained if more batteries are provided on the second stage). During this time, the SERT C attitude sensors will be powered up, and at a preselected time, the SERT C attitude control system will be activated followed immediately by separation of the second stage. This sequence should minimize the effects of any separation disturbances imparted by the second stage, while not allowing a time period during which the spacecraft is uncontrolled. With the solar arrays folded, the reaction wheels can damp tipoff rates as high as 1 degree per second. When attitude transients have damped out, the solar arrays will be deployed and oriented toward the sun. Following some period of time for spacecraft outgassing and battery recharging, startup of the 30-cm ion thrusters will be performed. Because of the possibility of the net thrust vector of the ion thrusters not passing through the spacecraft center of mass, some provision, such as gimbals, will be required to enable nulling of this offset.

Procedures will be devised to determine the magnitude of the thrust vector offset in both the pitch and yaw axes by observing in real time the pitch and yaw attitude histories following thruster startup, and gimbal commands will be issued to null the offsets. This will be completed as soon as possible and prior to initiating the yaw program described in the Mission Requirements and Operations section.

<u>Orbit Raising Phase</u> - During the normal orbit-raising phase the spacecraft is controlled by reaction wheels and thruster deflections which respond to attitude error signals generated by the roll, pitch and yaw sensors. Because the wheels and thruster gimbals are proportional devices, the control loop is a proportional one, as opposed to a deadband control loop associated with attitude control thrusters. Proportional control produces better overall control accuracy. The overall control accuracy should be only slightly worse than the sensor accuracy, because the sensor now represents the predominant error source in the loop.

Two earth sensors, one for orbit-raising and one for synchronous operation, are used because: (1) a different center body face is oriented toward the Earth in synchronous orbit than during the orbit raising phase, (2) no earth sensor presently exists which can operate over the altitude range encountered during orbit-raising and also provide the more stringent accuracies desired in synchronous orbit. Earth sensor rotation to provide backup capability was not deemed a good trade-off against the added spacecraft complexity and alignment problems. However, this trade-off consideration requires further detailed study.

Two two-axis sun sensors, one mounted on the root of each of the solar array panels, provide nearly direct yaw information when the satellite is located at the "dawn" and "dusk" points in the orbit. At other points, the yaw information can be extracted from the sun sensor data by calculation, knowing the orbit parameters and sun right ascension and declination. Sensors are provided on each panel in order to obtain sufficient field of view to encompass the relatively large angles (40°-45°) between the solar array normal and sunline encountered during orbit-raising. The magnitude and accuracy of the yaw error signals obtained in this way is dependent upon the spacecraft location in its orbit. For some period of time twice each orbit, these error signals will decrease to the point where the information provided by them may no longer be usable for attitude control. At some point prior to this time, therefore, the sun sensor information will be used to update the gyro, which has its input axis oriented parallel to the yaw axis. The gyro will then provide yaw information for attitude control purposes until the accuracy of the information from the sun has returned to a usable value. Depending upon the allowable yaw error tolerance and the sun declination, it may be possible to avoid using the gyro entirely for significant portions of time, which means that under these circumstances it could be turned off, thereby increasing confidence in its ability to operate for the entire mission.

The motion about yaw to provide the required out-of-plane thrust component for inclination changing will be obtained by having the on-board computer provide the desired yaw offset history in real time to the yaw control loop as a reference. The momentum storage requirement for the yaw wheel to perform this function is about

5 ft-lb-sec at the lower altitudes and drops quickly as altitude and orbit period increase. To keep the reaction wheel sizes to a minimum, the gimballed 30-cm thrusters will be used as additional torque sources during orbit raising. Obviously, during eclipse periods, the wheels alone must provide attitude control. Also, the 30-cm ion thrusters do not produce a direct torque about the roll axis. However, because the center body rotates at orbit rate and because the yaw offsets are generally large, all three axes are momentum cross coupled. It should therefore be possible to use the ion thrusters to obtain some indirect roll control. In addition, the array and body mounted 8-cm thrusters provide direct roll torques during orbit raising. A detailed analysis of the attitude control system will be required to determine the best way to apportion the control torque requirements between the available torque sources during orbit raising.

Normal Synchronous Orbit Phase - As discussed in section 2.3, the attitude accuracy goals are $.08^{\circ}$, $.08^{\circ}$ and $.2^{\circ}$ in roll, pitch and yaw, respectively. The attitude control system will demonstrate these accuracies during this phase of the mission. Attitude control system operation during this phase is essentially the same as that during normal orbit-raising. Proportional attitude control will be established using the reaction wheels as actuators, the synchronous orbit high accuracy earth sensor, the computer, and the yaw sensing system. Because of the 90° pitch rotation which is performed after synchronous orbit is achieved, a different yaw rate integrating gyro will be used for on-orbit yaw sensing than was used for orbit raising. A detailed analysis of this phase will be performed to assess the capability of the proposed system to meet the accuracy goals. The analysis will consider spacecraft environment, hardware characteristics including computer quantization, and spacecraft flexibility in determining the best method of system operation to meet the accuracy goals.

Station-Keeping Operation - The stationkeeping operations will be accomplished by operating the 8-cm ion thrusters on a periodic basis to provide translational impulses to the spacecraft as described in section 2.3. Attitude control system operation during this mode can be accomplished by using either the reaction wheels or the 8-cm thrusters as the primary attitude actuators. Because the north-south ion thrusters are located on the ends of deployed solar arrays, there exists the possibility of a misalignment of the thrust vector from the spacecraft center of mass of more or less arbitrary magnitude and direction. This misalignment will produce a disturbance torque which will affect the spacecraft attitude and which must be compensated by the wheels. Three alternatives are available. The first is to do nothing about the offset which could, in the worst case, result in the roll and yaw wheels saturating quickly and result in frequent momentum dumping. Alternatively, the magnitude of the offset could be determined by some method such as observing the rate of change of attitude, and a beam deflection made to correct the offset. If, on the other hand, the method of operation is switched so that the ion thrusters are used as the primary actuators in the control loops during station keeping rather than the wheels, the misalignments would be automatically nulled whenever the station keeping operation were initiated. Because the time required to deflect the thruster beam and the acceleration due to the thruster are both small, any attitude transient resulting from the switch over should be of small magnitude. A detailed analysis must be made of the various alternatives before a final decision as to how best to handle thruster misalignments can be made.

Momentum Dumping Operation - With the attitude disturbance torques shown in table 4.1.1 wheel unloading about the roll and yaw axes would be required once each day for about 16 minutes. There is no requirement to perform the unloading at any specific point in the orbit, and therefore, it could in general be coordinated with the station keeping operation. Pitch unloading will be performed daily. The time required will be about 48 minutes because of the smaller moment arm.

Wheel unloading is accomplished by decelerating the reaction wheel until its speed is zero. The deceleration produces a torque on the spacecraft which must be compensated by the appropriate 8-cm thruster. In addition, torque must be provided either by the reaction wheel or by the ion thruster to control against the ambient disturbance torque during dumping. In order to minimize the dumping time, the ion thruster will be operating at or near its maximum angular deflection throughout the dumping operation which would leave insufficient torque margin for direct attitude control. It, therefore, appears that use of the reaction wheel as the primary control actuator during wheel unloading is the best procedure. However, a detailed analysis of the operation again is required prior to making a final decision.

Eclipse Operation - During the periods of eclipse the attitude control system will obtain pitch and roll reference from earth sensors as in the normal mode. Yaw sensing will be provided by the gyro because the sun sensor will be inoperative. Stationkeeping or momentum dumping cannot be performed during eclipses because thruster power is not available. For those cases in which station keeping impulses would be desirable during an eclipse, the procedure can be modified so that the operation is accomplished in sunlight.

Station Walking - During the station walking operation, the attitude control system operates as in the orbit raising mode, the main difference being that the yaw offset is essentially held to zero. Therefore, prior to initiation of a station walking maneuver, the center body must be rotated through 90 degrees in pitch. Because the 30-cm thrusters are not required to accommodate the larger disturbance torques as at low altitudes, they can be used to dump the pitch and yaw reaction wheels during the station walking maneuver. The roll reaction wheel is dumped by one of the array mounted thrusters. Alternatively, the 30-cm thrusters could be used to provide all or part of the control torque requirements about pitch and yaw during this period. This would have the effect of either increasing the time between wheel dumping periods or eliminating the necessity of operating the yaw and pitch wheels during station walking. Again a more detailed analysis is required to determine the best procedure. The torque to rotate the spacecraft through 180 degrees about yaw at the middle of the maneuver can be provided either by the yaw reaction wheel or by deflecting the 30-cm thrust vector.

<u>Target Spacecraft Rendezvous and Inspection</u> - The phase is initiated when the SERT C and a target spacecraft are sufficiently close that the on-board target sensing system (or systems) can locate the target. Depending on the overall field of view of the system, it may be necessary to switch to an attitude control mode which utilizes error signals from the target sensor as inputs to the control system in order to stay "locked on" the target. The target sensor would provide errors in two axes, while the third would be derived from either the sun sensors or earth sensor, or a combination of these. Furthermore, for the terminal "close in" maneuvers, it may be desirable or even necessary to use a high-thrust chemical system. This phase represents an area in which a detailed study must be performed before any conclusions can be made as to the best

and most feasible operational procedure. It is obvious, however, that a control system configuration and operational mode is required here which is more complex than any of the others.

Reacquisition - Reacquisition of spacecraft attitude will be required in the event of loss of spacecraft orientation due to a malfunction or other unpredictable occurrence. In general, following a disturbance, the spacecraft will be left with an attitude substantially off normal, spinning about one axis with the spin axis coning. The worst case situation is one in which the attitude is such that no sunlight falls on the solar arrays and no attitude information is provided by the sensors. In this case the ion thruster system is inoperative. In the absence of a chemical system, therefore, the reaction wheels are the sole source of torque for rate damping. Presently the worst conceivable disturbance would be caused by an ion thruster vectoring failure such that maximum possible torque were produced. In this event rates could be obtained in a few minutes which exceeded the wheel capacity. Such a failure mode must therefore be "designed out" of the ion thruster systems. The (less desirable) alternative is to provide a chemical mass expulsion system for reacquisition.

With the rates nulled, a search for the sun can be initiated. Once the sun is found, the array mounted sun sensor can be used for two axis attitude control. A period of recharging batteries and removing stored angular momentum would be initiated, since the ion thrusters would then be available. At the proper time in the orbit, a search for the earth will be initiated by introducing a programmed rotation of the center body relative to the solar array, and if required, a rotation about the sunline. When the earth is acquired by the earth sensor, two axis earth control will be initiated. The yaw offset (up to 23.5 degrees, depending upon time of year) will then be nulled using the sun sensor, and solar array stepping will begin.

As the risk of attitude loss is probably greatest after eclipse, when battery power is low, a detailed analysis of this mode is required. An assessment of the necessity for a chemical propulsion system backup for this mode must also be made.

Backup Mode - The backup mode represents any control system reconfiguration which is made to compensate for failure of a component. Without giving any consideration to redundancy of components, which must await a detailed tradeoff of required reliability vs allowable cost and weight penalties, it should be noted that the spacecraft configuration provides some inherent backups. If sufficient propellant were included, either array thruster is capable of performing all the stationkeeping maneuvers and roll and yaw dumping. The body mounted 8-cm thrusters provide redundant primary pitch wheel dumping torques and also provide backup torques for yaw wheel dumping in synchronous orbit. In the event of wheel failure, the 8-cm ion thrusters can provide backup control torques about all three axes via beam deflection, although at the expense of mission time and attitude accuracy. The sun sensor can provide backup pitch error information, as well as roll information for part of the orbit. Careful consideration will be given in the design of the Attitude Control and Station Keeping System logic and electronics and also in its interfaces with other spacecraft systems to making sure that these backup capabilities can be used in the best possible fashion.

TABLE 4.1.1 - DISTURBANCE TORQUE PEAK VALUES

SOURCE	ROLL (FT-LB)	PITCH (FT-LB)	YAW (FT-LB)
Solar Pressure	1.1×10^{-4}	2.0×10^{-5}	1.1×10^{-4}
Magnetic *	7.3×10^{-5}	- -	7.3×10^{-5}
Gravity-Gradient *	2.3×10^{-3}	1.3×10^{-5}	2.3×10^{-4}

* at 3250 KM altitude

TABLE 4.1.2 - ATTITUDE CONTROL SYSTEM WEIGHT AND POWER BREAKDOWN

COMPONENT	WT (LB)	POWER (WATTS)
Coarse Sun Sensor	0.1	Reacquisition only
2 - axis Earth Sensor (Orbit Raising)	16.8	13 max 10 avg
2 - axis Earth Sensor (On-Orbit)	8.0	1.0
2 - axis Sun Sensor	4.4	1.0
Rate Integrating Gyro (orbit-raising) Rate Integrating Gyro (on-orbit)	3.6 3.6	45 Start 12 Run
Rate Gyro Package	2.5	13 (reacquisition only)
Reaction Wheels	50	75 Peak 15 Continuous
Attitude Control Electronics	5	5
Total System Wt*	94.0	

* Assumes chemical system not required for rendezvous and reacquisition

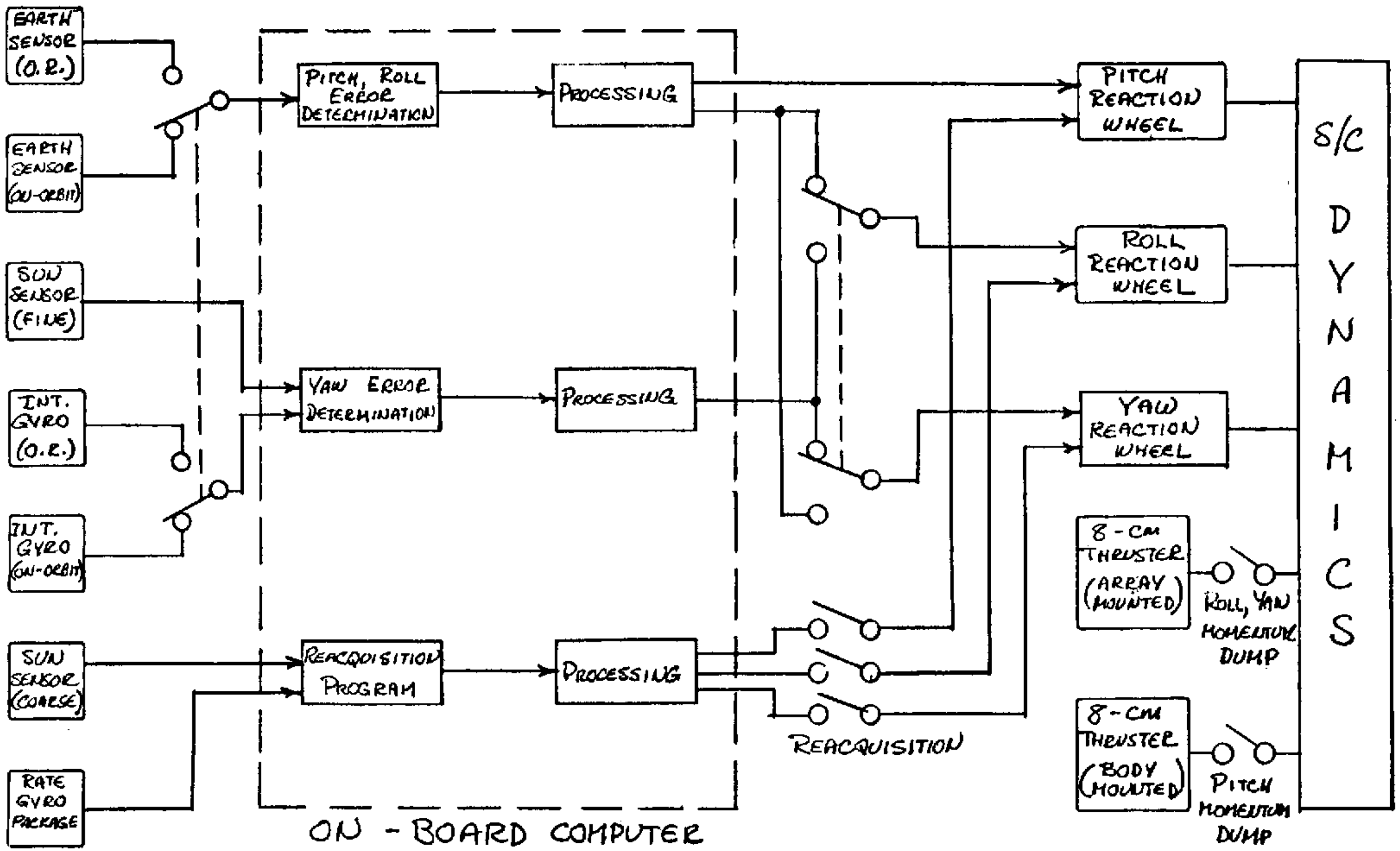


Figure 4.1.1 - Block Diagram of Attitude Control System

4.2 Electric Thruster and Power Processing

Ion thruster (and power processing) subsystems for both prime and auxiliary propulsion are included in the spacecraft design. These subsystems will be described separately.

Prime Propulsion Thrust Subsystem - The prime propulsion subsystem consists of three 30-cm thrusters; three power processors; the thruster array structure, including a gimballing system; and associated electrical cabling, propellant feed lines, and main propellant storage tanks.

30-cm Thrusters

The 30-cm thruster is being developed to flight hardware status as part of the SEP program. Design characteristics of the thruster and its power conditioner are presented in Table 4.2.1. The mass breakdown of the system is presented in Table 4.2.2. Figure 4.2.1 is a cutaway showing the main features of a 30-cm thruster and figure 4.2.2 is a drawing of the flight-configured engineering thruster model (ETM) to be delivered by Hughes Research Laboratories (HRL) to LeRC in October of 1973. Three identical 30-cm thrusters are arranged in an array such that the axis of each thruster is canted approximately 12° from the roll axis of the spacecraft.

The operation of these thrusters, which is conceptually the same as for the auxiliary propulsion thrusters, has been described extensively but will be briefly reviewed here for completeness. Liquid mercury propellant is vaporized in three porous tungsten phase separators and then passed through three isolators whose function is to isolate the thruster voltages from the main propellant storage tanks. Two of the isolators direct flow into the discharge chamber, both through a propellant manifold and the main hollow cathode. A small amount of propellant is provided to the hollow cathode neutralizer. Ions are produced in the discharge chamber by the bombardment of neutral propellant by electrons emitted by the main hollow cathode. Efficient ion production is achieved primarily by proper geometric discharge chamber design and optimization of the shape and magnitude of the magnetic field produced by the permanent magnets surrounding the discharge chamber. The ions thus formed are accelerated and extracted by the multiaperture , dished, two grid accelerator system. The resultant beam of positive ions is neutralized by electrons emitted by the hollow cathode neutralizer. A ground screen surrounds the thruster to prevent the beam plasma from interacting with the thruster surfaces.

Each 30-cm thruster can be operated over a 5-1 range of input power and is designed to provide high efficiency and stable operation over a 2-1 input power throttling range. Table 4.2.1 gives a typical performance values at full (nominal) and half thrust levels. The extremely high current carrying capacity of the accelerator system and the proposed power processor designed allow considerable flexibility in thruster throttling techniques. Proposed designs allow thruster operation either at near constant or variable specific impulse over the throttling range dependent upon refined mission analysis trade-off studies.

The weight of an individual thruster of the ETM design is given in Table 4.2.2. This weight includes the thruster body, approximately 30-cm of electrical cabling along with a connector, and several centimeters of propellant feed line upstream of the phase separators.

30-cm Power Processors

The power processors for the 30-cm ion thrusters will convert power received from the solar array bus into usable power for the thruster. Table 4.2.3 gives the maximum and nominal output characteristics of the twelve different power sources required for thruster operation.

Thermal Vacuum Breadboard (TVBB) power processors are presently being developed in two on-going contracts with delivery of an SCR model scheduled for November, 1973 and a model using transistor switching scheduled for April, 1974. The design goals of these contracts are an overall efficiency of 92 percent or greater and a weight of 31 pounds or less. They must be capable of operating over an input voltage range from 200 to 400 volts. These power processors must be capable of operation in vacuum with waste heat rejected by radiation. The frontal area of the processors is between 1100 and 1300 in^2, with aspect ratios of 2.2:1. The area, aspect ratio and mass of the ETM processor will depend upon information gained during operation of the TVBB processors and final spacecraft trade-off studies.

Thruster Array Structure

All three thrusters are hard mounted to a single support structure. This structure serves to (1) mechanically support the thrusters through the launch and flight phases, (2) provide a degree of thermal isolation between the thrusters and spacecraft and (3) serve as a single bed for two axis gimballing of the thruster array.

The single support structure mounted behind the thrusters is near optimum for simplicity, weight, and thermal isolation properties. It is proposed to gimbal the entire array about two axes by gimbals attached to the support structure. Specification of the location of the gimbals both with respect to the transverse axes of the spacecraft and with respect to each other, awaits detailed mission analysis trade-off studies. The entire array will be supported through the launch phase by mechanical supports which will then be removed by a pyrotechnic device after launch. This approach will allow minimum gimbal and support structure weight. The estimated weight of the array support structure, including gimbals, is 10 pounds.

Propellant Storage and Electrical Cabling

A common propellant reservoir for all 30-cm thrusters is located near the center of the spacecraft. Because of other spacecraft components, such as the solar array rotation mechanism, it is difficult to locate a single propellant tank at the center of mass of the spacecraft. Therefore, two half-size tanks are located on either side of the center of mass. These tanks are connected by a common line and the positive gas expulsion system (similar to the successful SERT II feed technique) depletes each tank equally to maintain a near constant center of mass with propellant use. Each tank will contain about 150 pounds of mercury, 10 pounds of which is allocated to the two body mounted auxiliary propulsion thrusters. The weight of the propellant tanks, and feed system is 30 pounds.

To specify the weight of the cabling system it is necessary to specify the diameter of the various power leads as well as their length. Other requirements, such as the possible necessity of twisted pair wiring for EMI suppression, also play a secondary role in determination of cabling weight. An estimate of cable weight was made using the assumption that the distance between the thrusters and the power supplies was to be 130 cm. The diameter of the individual leads was selected on the basis of a trade-off study of cabling power losses on cabling weight. The cabling weight including connectors and insulation was estimated to be fifteen pounds.

Auxiliary Propulsion Subsystem - The auxiliary propulsion subsystem comprises four 8-cm thrusters of identical design. Figure 4.2.3 is a cutaway of an 8-cm thruster. The two-axis, $\pm 10^\circ$ vectorable, dished accelerator grids have a design life of 20,000 hours. The actual diameter of the discharge chamber anode is 8.58 cm. Main and neutralizer cathodes are hollow cathodes with enclosed keepers and have a design identical with previously developed 5-cm thruster (SIT-5) cathodes. The main flow, all of which passes through the main cathode, is controlled by a porous tungsten vaporizer and electrically insulated by a flow isolator identical to SIT-5 design. The neutralizer cathode position and flow system are similar to SIT-5 design and contain a flow isolator capable of withstanding 100 V.

A single propellant tank for each of the two solar array mounted thrusters is located directly behind the thruster. Both the north and south thrusters are redundant in that either can completely perform the necessary north-south station keeping and roll and yaw momentum wheel dumping for the five-year mission. Thus, each thruster carries approximately twice the expected amount of propellant required. The propellant tank design uses a flexible expulsion diaphragm and gas pressure to feed propellant as in the SERT II thruster feed system. The spacecraft body-mounted thrusters draw their propellant from the common reservoir that also feeds the 30-cm thrusters. Table 4.2.4 gives thruster weights, cycles, and operating times for each thruster. Table 4.2.5 lists presently obtained thruster operating conditions, which were used to design the SERT C mission. If performance improvements are made, the propellant weight saving may be off loaded or retained as propellant reserves.

The 8-cm ion thruster power processor is designed to be a self-contained unit consisting of interface command logic, digital and analog control circuitry, low and high voltage power supplies, housekeeping bias supplies and telemetry output circuitry. The processor obtains 150 watts of power from an unregulated solar array bus, and less than three watts from a regulated bus.

The digital interface and control subsystem can receive 16 parallel bit digital "value words" by means of a series to parallel converter. The "value words" contain thruster operating commands and set points. This feature permits a high level of ground control over the thruster subsystem. The system provides command/control voltage isolation between the spacecraft common and power processor common. Isolation is also provided between telemetry output signals (referenced to spacecraft ground) and processor common.

Including appropriate thermal packaging, the processor weight will be 9.0 pounds and will operate at an efficiency of 80 percent. The allowed temperature range for nominal operation is -15° C to +50°C with a baseplate area of 144 square inches. During eclipse, the array mounted processor may, in addition to louvers, require five watts of standby power to maintain acceptable temperatures. The south thruster runs off the hybrid Directly Regulated Solar Array (DRSA) power processor.

Design of Solar-Array-Mounted Thruster System

The thruster system will be mounted on the end of the Astromast that deploys the solar array. As the solar array unfolds, the thruster system will be automatically deployed with the array to a position (as shown in Figure 1.3.1 SERT C S/C) at the end of the array. The center of the thruster and its thrust vector will be positioned directly over the end of the Astromast and pointed through the spacecraft center of mass.

The thruster propellant tank will be mounted directly behind the thruster. The propellant tank will be close enough to the thruster that a propellant line valve will not be necessary to protect the vaporizer porous tungsten against launch-induced propellant line pressure surges.

The power processor will be mounted on the array palette on the side of the thruster. Thermal control louvers will close during eclipse to protect the power processor against low temperatures. During normal operation, the power processor is designed to be self-radiation cooled. Likewise, the thruster will be designed to self-radiate its waste heat. During eclipse periods, the thruster and propellant have sufficient thermal mass to avoid mercury freezing. The neutralizer propellant line would be the first component to freeze. In-house tests have indicated a freezing time constant for an unprotected neutralizer line to be about equal to the longest eclipse period, 72 minutes. Therefore, a small amount of thermal shielding of the neutralizer line to the thruster system may be employed.

TABLE 4.2.1 - 30-CM THRUSTER SYSTEM PERFORMANCE

	Full Thrust (Nominal)	Half Thrust
Thruster Power, W	2620	1384
Thrust, mlb.	28.8	14.3
Specific Impulse, sec.	2955	2900
Total Mass Flow Rate, gm/hr.	15.9	8.04
Overall Thruster Efficiency	0.708	0.654
P/C Power	2880	1525
Beam Divergence Thrust Factor	0.985	0.985
Double Ion Thrust Factor	0.965	0.98
Power Processor Efficiency	0.92	.89

TABLE 4.2.2 - PRIME PROPULSION COMPONENT WEIGHTS

	Weight Lb.	Number Required	Total Weight Lb.
30-cm Thruster	17	3	51
Power Processor	35	3	105
Array Structure and Gimbals	10	1	10
Propellant Tankage and Feed	30	1	30
Electrical Cabling	15	1	15
Orbit Raising Propellant	300	1	300
Total System Dry Weight, lb.			211
Total System Loaded Weight			511

TABLE 4.2.3 30-CM THRUSTER POWER SUPPLY REQUIREMENT

Supply No.	Supply	Maximum	Nominal level	Regulation type and %	Ripple % p-p
1	Main vaporizer	10V at 2A	7V at 1A	I*, 5%	
2	Cathode vaporizer	10V at 2A	3.5V at 1A	I, 5%	
3	Cathode heater (3)	15V at 6A	9V at 4.5A	I, 5%	
4	Main isolator & Cathode isolator heater (3)	10V at 4A	8.2V at 3.6A	I, 5%	
5	Neutralizer heater (3)	10V at 5A	8.5V at 4.2A	I, 5%	
6	Neutralizer vaporizer	10V at 2A	3.5V at 1A	I, 5%	
7	Neutralizer keeper(1)	20V at 3A	12V at 2A	I, 5%	2%
8	Cathode keeper	60V at 1A	10V at 0.5A	I, 5%	2%
9	Discharge(2)	40V at 15A	37V at 14A	I, 1%	2%
10	Accelerator	1000V at 0.1A	500V at 0.008A	V, 2%	3%
11	Screen	1100V at 2.2A	1100V at 2A	V, 1%	1%
12	Magnetic baffle	< 2V at 5A	< 2V at 5A	I, 5%	5%

(1) H.V. Section 1 kV at 20 mA (or special designs to assure starting of HRL 30-cm thruster)

(2) Open circuit voltage, V_D; $60\ V \le V_D < 100V$

(3) On at start only

* I ≡ current
V ≡ voltage

TABLE 4.2.4 8-cm THRUSTER WEIGHTS, CYCLES AND OPERATING TIMES

		North Array Mounted Thruster lb.	South Array Mounted Thruster lb.	East Body Mounted Thruster lb.	West Body Mounted Thruster lb.
Thruster Body		5.1	5.1	5.1	5.1
Propellant Tank		3.0	3.0	Uses 30-cm thruster common tankage	
Hg Propellant		19.2	19.2	4.3	3.4
Power Processor		9.0	8.0*	9.0	9.0
Total Weight		36.3	35.3	18.4	17.5
Duty Cycle, Hrs./ Time Between Cycles	A	2.6/1 day	2.6/1 day	-	-
	B	-	-	4.0/7 days	4.0/7 days
	C	-	-	two 0.717 hr/7 days pulses	-
	D	0.267/1 day	or 0.267/1 day	-	-
	E	-	-	0.80/1 day	or 0.80/1 day
Total Cycles		2739	2739	1696	1775
Total Operating Hours		4992	4992	2145	1129

* Hybrid DRSA power processor

A North-south station keeping

B East-west station keeping (solar pressure)

C East-west station keeping (triaxiality)

D Momentum wheel dumping (roll-yaw)

E Momentum wheel dumping (pitch)

TABLE 4.2.5 MEASURED 1-MLB 8-CM ION THRUSTER OPERATING CONDITIONS

(Dished misalinement vector grid)

	OPERATING VALUE	SUBSYSTEM POWER
Thrust* (ideal), mlb	1.14	
Thrust† (true), mlb	1.00	
Specific impulse,* sec	2501	
Total input power, W	137.87	
Total efficiency, %	45.5	
Power efficiency, %	63.2	
Total utilization,* %	71.9	
Discharge utilization,* %	77.4	
Total neutral flow, mA	100.2	
Power/thrust,* W/mlb	112	
eV/ion excluding keeper, V	300	
eV/ion including keeper, V	329	
Beam current, J_B, mA	72	
Net accelerating voltage, V_I, V	1220	
Neutralizer floating potential, V_g, V	-10 (est.)	
Output beam power, W		87.1
Accelerator voltage, V_A, V	-500	
Accelerator drain current, J_A, mA	.23 (est.)	
Accelerator drain power, W		.4 (est.)
Discharge voltage, ΔV_I, V	40	
Emission current, J_E, A	.54	
Discharge power, W		21.6
Cathode:		
Keeper voltage, V_{CK}, V	10.5	
Keeper current, J_{CK}, A	.2	
Keeper power, W		2.1
Heater voltage, V_{CH}, V	0	
Heater current, J_{CH}, A	0	
Heater power, W		0
Vaporizer voltage, V_{CV}, V	5.6	
Vaporizer current, J_{CV}, A	2.2	
Vaporizer power, W		12.3
Flow rate, mA	(93.0) by diff.	
Neutralizer:		
Keeper voltage, V_{CK}, V	15.8	
Keeper current, J_{CK}, A	.4	
Keeper power, W		6.32
Heater voltage, V_{CH}, V	1.6	
Heater current, J_{CH}, A	4.0	
Heater power, W		6.4
Vaporizer voltage, V_{NV}, V	1.35	
Vaporizer current, J_{NV}, A	.6	
Vaporizer power, W		.81
Flow rate, mA	7.2	
Neutralizer coupling power, W		.72 (est.)
TOTAL THRUSTER POWER, W		137.8

*Accounting for neutralizer floating potential

†Ideal thrust corrected for double ions and beam divergence

est. = estimated

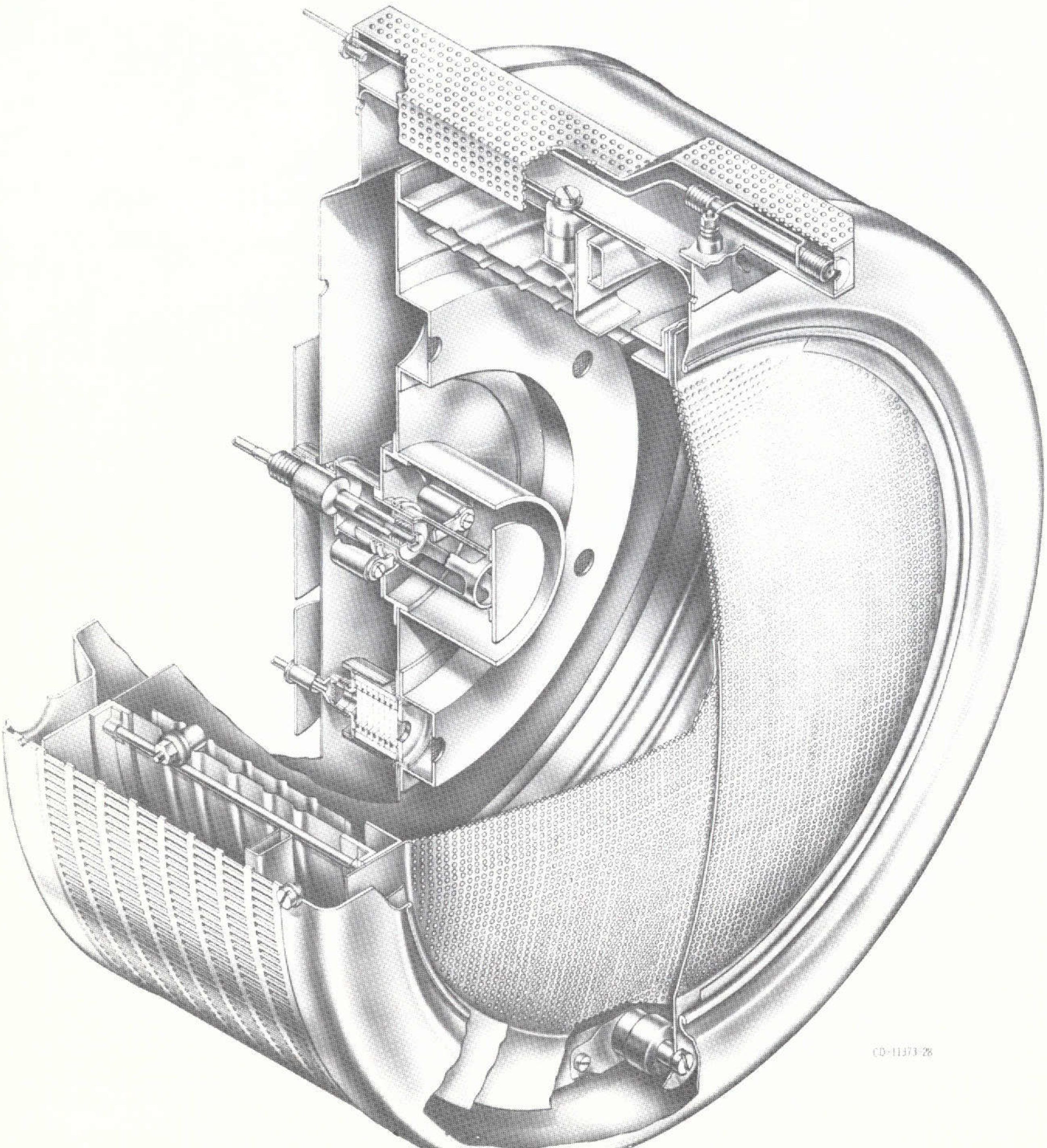


FIGURE 4.2.1 CUTAWAY DRAWING OF 30-CM THRUSTER

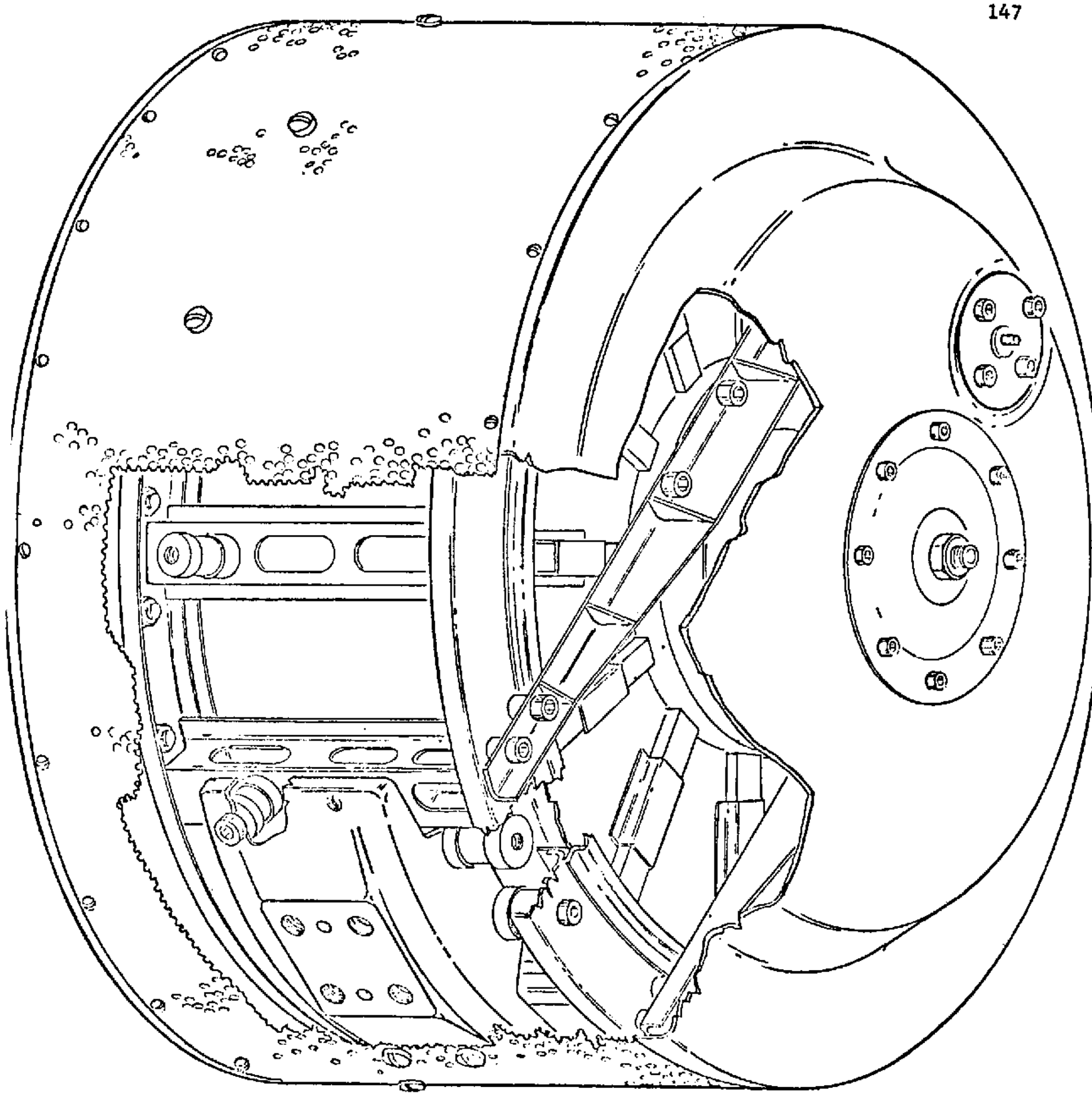


FIGURE 4.2.2 ETM THRUSTER DESIGN

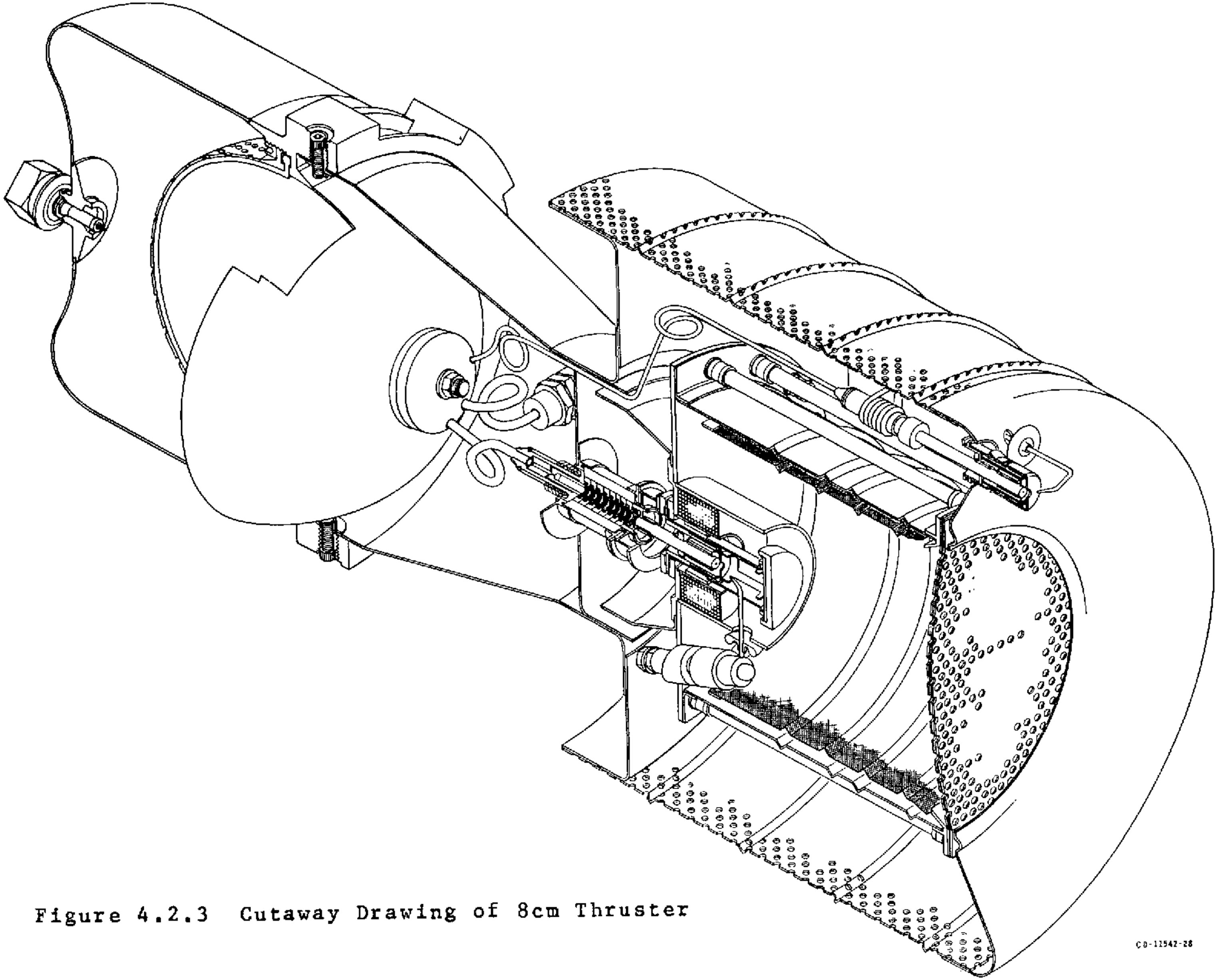


Figure 4.2.3 Cutaway Drawing of 8cm Thruster

4.3 Solar Array

Introduction - The SERT C power requirements vary during the mission life. During the orbit raising phase, power is required primarily for the 30-cm ion thrusters and for the spacecraft housekeeping system. In synchronous orbit, power will be required primarily for the 8-cm ion thrusters used for attitude control and station keeping experiments and housekeeping functions. The required power will be provided by extendible solar cell arrays which will be deployed after the 3150 km circular parking orbit is achieved. The solar arrays will rotate relative to the spacecraft center body once per orbit to minimize the angle between the solar array normal and the sunline.

Solar Array Description - The solar arrays are deployed from the north and south faces of the spacecraft center body. Each array (9 ft wide, 53 ft long) consists of two flexible blanket substrates (to which the solar cells are mounted) separated by an Astromast. See figure 4.3.1. The Astromast deploys the array and provides it structural stiffness. Each array blanket contains 22 active (containing solar cells) panels plus a blank panel at each end of the blanket. Each panel measures 26.5 by 48.5 inches. The total area of the 4 blankets is 850 square feet.

To meet mission requirements, the solar array must supply a total of 9 kW_e at beginning of life (BOL): approximately 8 kW_e at a nominal 300 Volts for the 30-cm thrusters and approximately

1 kW_e at low voltage for housekeeping. The use of new high efficiency solar cells (13.5% versus 10% for conventional cells) is proposed in order to minimize the array area and weight. Each active panel contains 1680 (2 cm x 2 cm) solar cells in 30 by 56 cell matrix. The panels have 168 cells in series and 10 cells in parallel to provide nominal 60 Volt power. The 300 Volt panels have 840 cells in series, 2 in parallel to provide power for the 30-cm thrusters. Power and instrumentation leads are located in the border areas at the sides of the blanket.

In synchronous orbit additional low voltage power for 8-cm thruster operation will be obtained by reconfiguring 300 Volt panels using relay switches. Some of the 300 Volt panels are also reconfigured to perform the Directly Regulated Solar Array (DRSA) experiment with an array mounted thruster. When the 30-cm thrusters are shut down a considerable excess of power is available from the array. This power is available to conduct a wide variety of experiments. Use of this excess power for experimental purposes is under study.

Array Deployment - Each array will consist of two blankets centered about a continuous longeron-type Astromast. The Astromast is a lattice boom consisting of coilable longerons, battons, and diagonal members which can be automatically deployed from a compact stowage cannister. The longerons and battons will be constructed from fiberglass rods while the diagonals will be made from flexible steel cables.

Because of its open section design this type of boom has distinct advantages over other types of extendible booms. First, the open section design results in the relatively high stiffness to weight ratio required to avoid coupling of the flexible array motion with the attitude control system. Secondly, the thermal bending and twist of the Astromast boom on orbit will be insignificant since all members receive equal solar illumination and each will have a small temperature difference across its diameter. Preliminary estimates indicate that a nine inch diameter boom will be required to support the split blanket array. This size boom can be housed in a cannister approximately 31 inches in length and 10 inches in diameter. Further studies are now being made in an effort to reduce the size and weight of the boom.

During the launch phase of the mission the Astromast will be in a retracted position and stowed in its cannister while the solar arrays will be folded concertina fashion and stowed under pressure between two plates as shown schematically in Figure 4.3.2. To prevent damage to the solar cells in this configuration the folds of the array will be interleaved with a cushioning material that will be attached to the lower plate or pallet. To deploy the arrays, the pallets are released from the spacecraft center body. The Astromast is then activated and the boom moves the entire pack away from the spacecraft until the telescoping tubes or elevation arms are locked in place (Figure 4.3.3). The distance the pack is moved from the

N-S panel will be dictated by thermal studies now being conducted. Next, the pressure plate is freed from the pallet by releasing the latching mechanism on the pressure plate. The Astromast is then reactivated causing the boom to move the pressure plate outward unfolding the array (Figure 4.3.4).

Solar Cell Experiment - The orbit raising phase of the mission begins below and passes through the Van Allen radiation belt maxima. The solar array will experience a space radiation dose of the order of 10^{16} equivalent 1 MeV electrons. This is one to two orders of magnitude greater than the radiation dose experienced by most spacecraft. The solar array power is predicted to degrade to approximately 50% of the beginning of life power. This degradation will occur during the first half of the orbit raising phase. To monitor the solar array, it is necessary to include an experiment made up of instrumented solar cell modules identical to those on the main solar array. These modules will be located on one of the array panels.

Status - Although operational feasibility for the various components of the proposed array will be verified on various spacecraft, development will be required to integrate and flight qualify the components into the proposed solar array configuration. The Astromast has been developed and tested. The high efficiency (violet) solar cells will be flown as part of the power system on IMP-J (Nov. 1973 launch) and as an experiment on SPHINX to be launch in January 1974. The design of blanket components such as substrate, interleaf materials, hinges and folds, wiring and interconnects can draw upon CTS array experience. CTS will be launched in 1975. The main development effort will be to integrate the various technologies into an array suitable for SERT C.

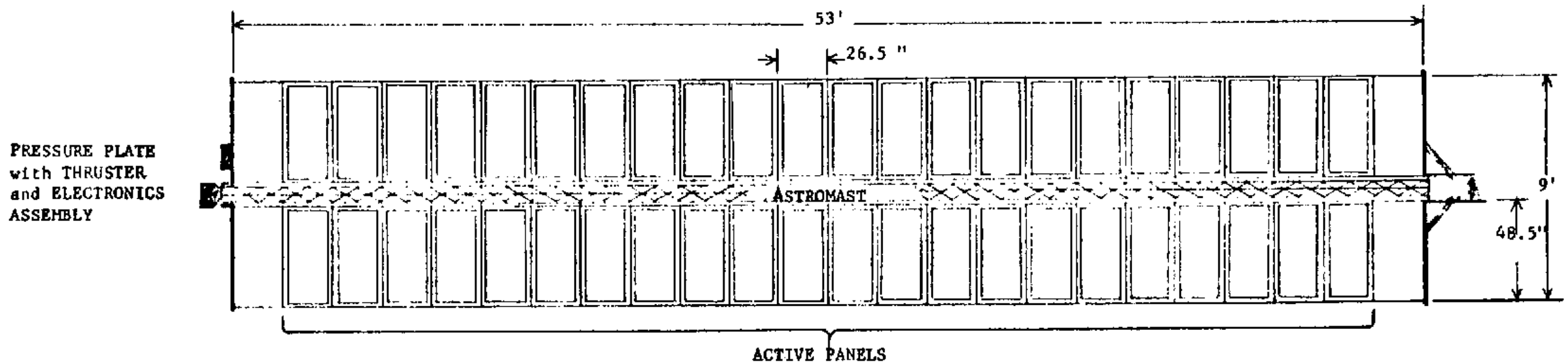


Figure 4.3.1 SERT C SOLAR ARRAY

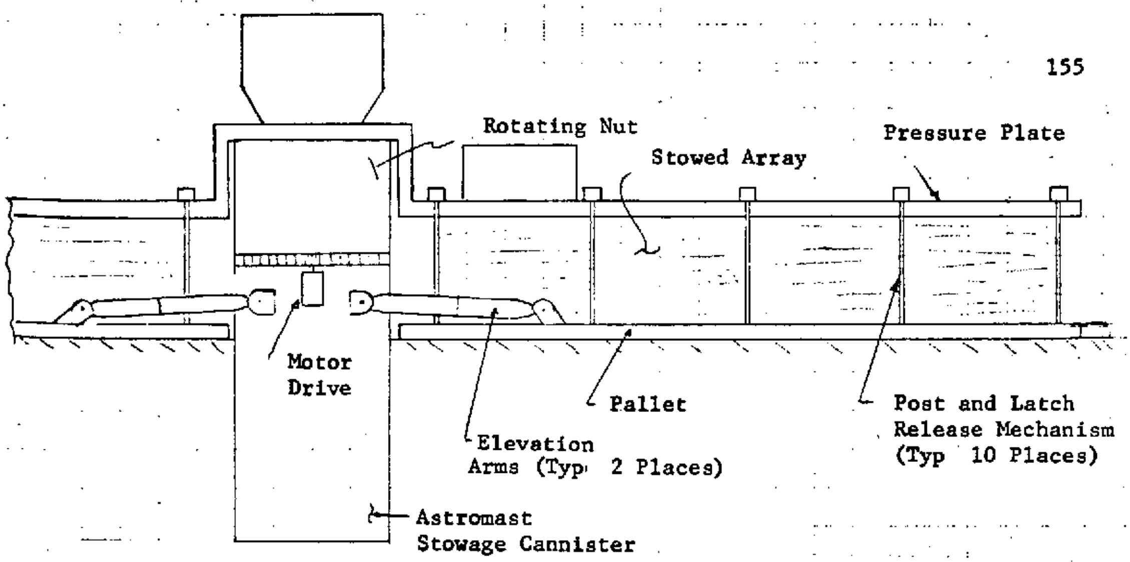


Figure 4.3.2 - Stowed Array

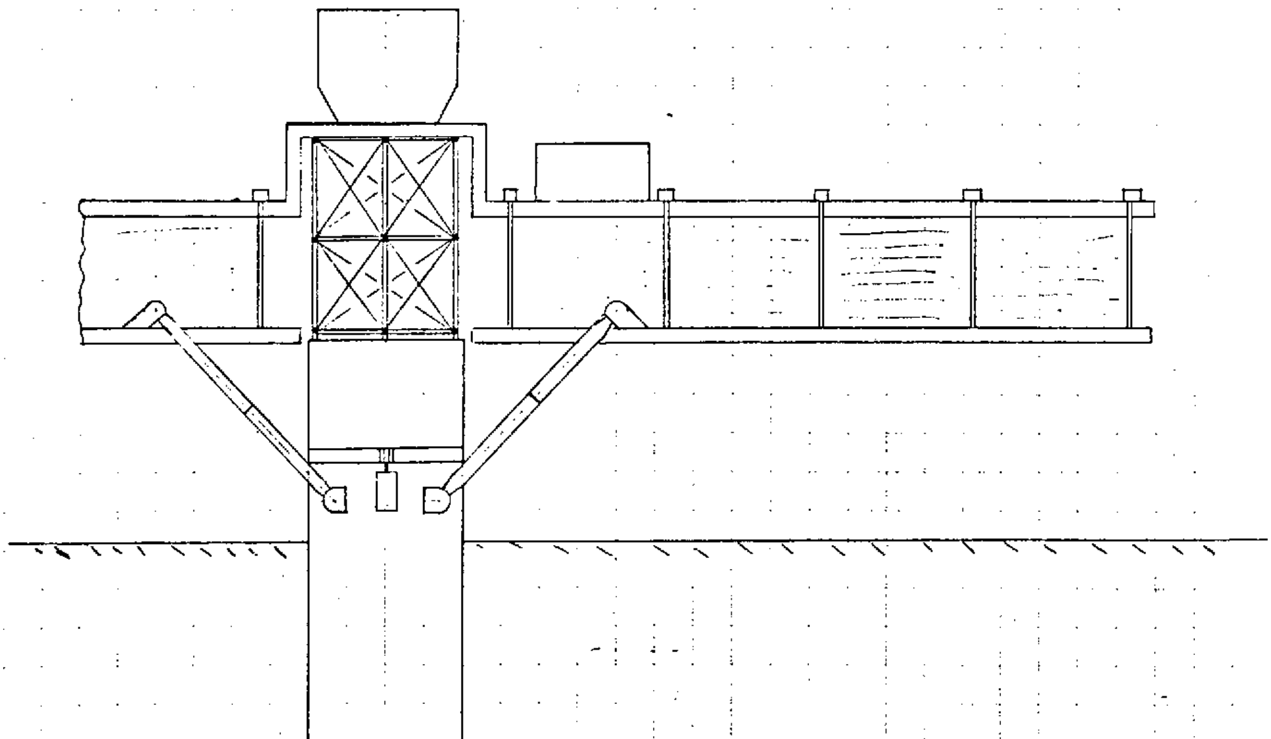


Figure 4.3.3 - Elevation Arms Locked in Place

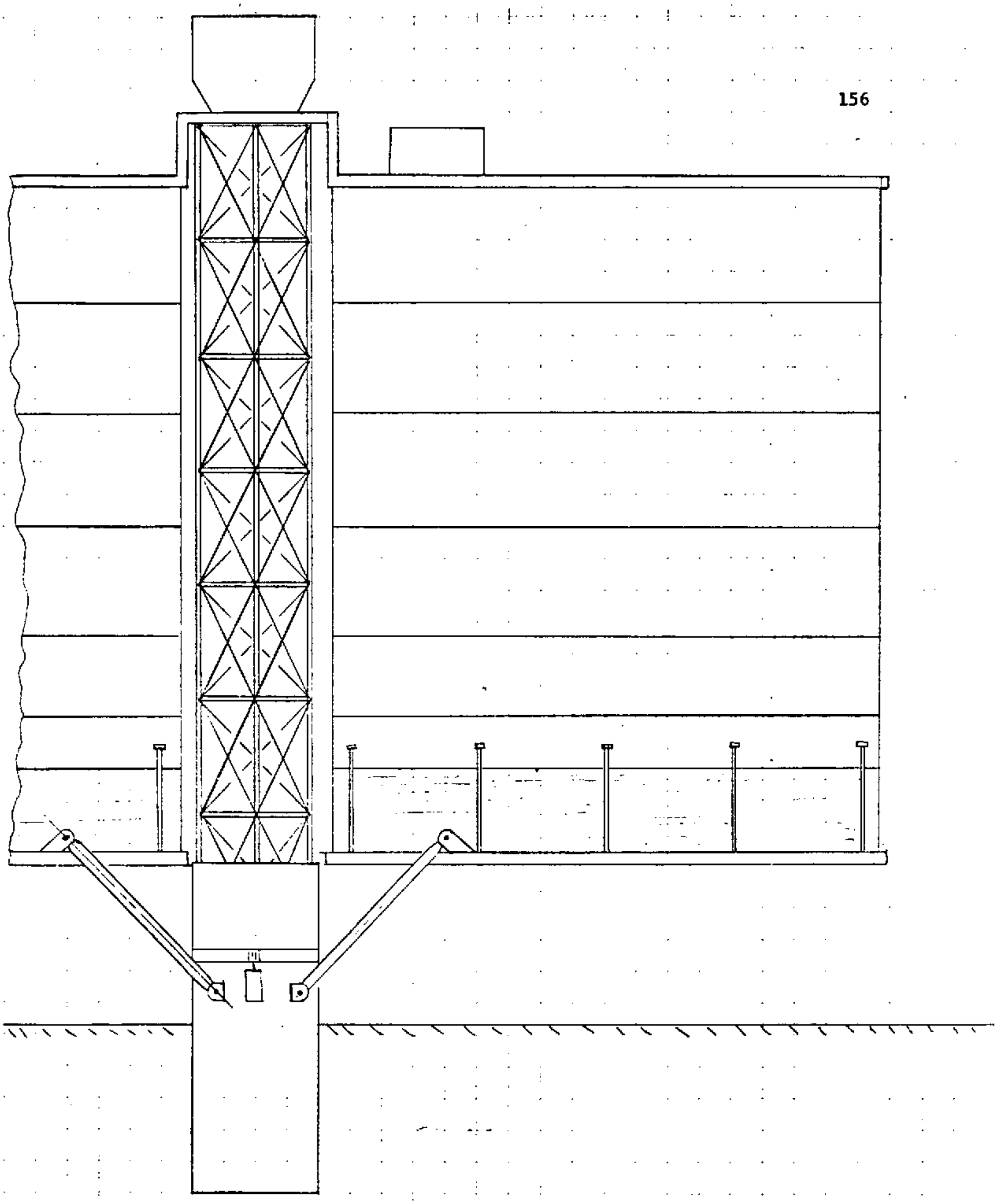


Figure 4.3.4 - Array Deployment

4.4 Solar Array Orientation Mechanism (SAOM)

System Description - The solar arrays are oriented with respect to the center body of the spacecraft by the solar array orientation mechanisms (SAOM). Power is transferred across this rotating interface by means of liquid metal slip rings (LMSR). The SAOM is a single ended device and two are required for the spacecraft. One controller electronics assembly will drive and control both SAOM's.

The SAOM basically consists of a small angle permanent magnet steppermotor, a single pass gear pair, and harmonic drive resulting in output steps on the order of 0.005°. Position is determined from an optical encoder, a multisegment disk with a light emitting diode-photo transistor excitation-sensor combination. The disk is mounted on the motor shaft to give the required resolution. The drive electronics consist of a crystal oscillator/frequency divider/down counter system stepping power to the appropriate motor field coils. The SAOM's will be driven synchronously.

During the initial phases of the mission from spacecraft initial activation to synchronous orbit operation the operation of the SAOM's will be controlled by the on-board computer. The drive electronics have provisions for varying the speed continuously from 1 to 20 revolutions per day. This variation will accommodate the range of orbit periods during the spiral out. The SAOM's will be operated open loop at one revolution per day in synchronous orbit. A slewing rate of 50 revolutions per day will provide for fast sun acquisition or other maneuvers requiring a fast slew. The SAOM's are capable of bidirectional operation.

The SAOM design does not incorporate any mechanical redundancy. The final design is to be subjected to a life test to verify the selection of materials, lubricants, and components. Redundancy is employed in the drive circuitry. The SAOM system is designed for two years ground storage and five years operational life.

Ten commands are required to operate the SAOM's and the driver and control electronics in the desired modes. The commands consist of the following:

Electronics power on

Electronics power off

Rate 1 revolution per day

Rate 50 revolutions per day (slew)

Rotation forward

Rotation reverse

Driver electronics primary

Driver electronics backup

Open loop operation

Operation by central processor unit

Data required from the SAOM system is:

Encoder output (1 each SAOM):	2 digital
Temperature (4 each SAOM) :	8 analog
Motor current (1 each SAOM) :	2 analog

The drive electronics will be contained in a separate assembly package.

A LMSR assembly will be attached to each SAOM and will be part of the spacecraft power transfer system. The LMSR concept has been under study and development for a number of years (approximately three years of development were funded by LeRC) and a near-optimum electrode-pair configuration has evolved; this configuration is being employed. Redundancy in both the ring and brush electrodes is being employed; there are multiple brushes per ring and power and signals are carried in parallel over redundant rings. The LMSR will be life tested simultaneously with the SAOM.

Weights for the SAOM/LMSR system are:

LMSR: 2 pounds each or 4 pounds total

SAOM: 8 pounds each or 16 pounds total

Drive electronics: 4 pounds

Total power transfer system weight: 24 pounds

The drive electronics package is approximately 50 cubic inches in volume, 4x4x3 inches, excluding connectors. The SAOM/LMSR assemblies will be integrated with the lattice structure deployable booms. A volume of 14 inches in diameter by 7 inches long allows optimization of the mechanical configuration.

Power consumption is:

Each LMSR: none

Each SAOM: 1 watt average at 1 rpd (10 watts peak/step)
6 watts average at 20 rpd
10 watts average at 50 rpd

Drive electronics: 4 watts continuously

Summary:	1 rpd	:	6 watts
	20 rpd	:	16 watts
	50 rpd	:	24 watts

4.5 Electrical Power

Introduction - The power generated by the two solar arrays on SERT C will be used to feed the three power buses in the electrical system. These power buses are: a regulated 28 volt housekeeping bus, an unregulated nominal 60 volt bus, and a nominal 300 volt unregulated bus. The total beginning of life (BOL) capability of the two arrays is 9000 watts. At the end of life (EOL) the capability is expected to be 4500 watts. Initially, approximately 8000 watts (at 300 V) will be used to supply power for the 30-cm thrusters,and the remaining 1000 watts will be used for housekeeping and experiments.

Once the spacecraft reaches synchronous orbit, part of the 300-volt source will be reconfigured into a low voltage source that will be used to power 8-cm ion thrusters. Also, at some time during the mission another part of the 300-volt source will be reconfigured into the Directly Regulated Solar Array (DRSA) which will be used to power one of the array mounted 8-cm ion thrusters. The solar array can be reconfigured as required to 300 volts for station walking with the 30 cm thrusters. One of the 8-cm ion thrusters can be powered by a nickel-hydrogen (Ni-H_2) battery. A more detailed description of the Ni-H_2 power source can be found in Appendix A.

Power Profile - The power profile for SERT C electrical loads is shown on figure 4.5.1. This plot shows the power requirements during the different phases of the mission. As can be seen from the power profile, the power required varies considerably with ion thruster operation. A breakdown of the different loads and their power requirements is given in table 4.5.1. This table also indicates the power source if the power does not come from the regulated housekeeping bus. The power profile plot and the power requirement table determine the configuration of the electrical power system of SERT C. Figure 4.5.2 shows a representative block diagram of the power system. All of the main units that are necessary to regulate and control the power for the spacecraft are shown. These include the housekeeping regulator, the battery, the battery charge-discharge sensor, and battery charger. Also shown in the block diagram are the major spacecraft loads. A detailed discussion of each of these loads can be found in separate sections of this proposal.

Housekeeping Bus - The housekeeping bus supplies power to spacecraft systems such as attitude control, telemetry and solar array orientation mechanism. The housekeeping beginning of life power will be approximately 1000 watts while its end of life power will be approximately 500 watts. An array voltage of 60 was chosen so that the battery charge and discharge circuitry could be simplified. The SERT C spacecraft will have enough array power available so that during sunlit periods all of the housekeeping power requirements can be met by the housekeeping array. By connecting the battery through an isolation diode to the solar array bus, a dedicated discharge regulator for the battery can be eliminated. When the battery is needed, it will discharge through the main regulator. Further simplification can be gained if the array voltage is made larger than the required battery charging voltage because a buck-type charger (one which has a lower output voltage than input voltage) can be used rather than a boost charger. The reason that this system is not commonly used in spacecraft power systems is that it is hard to control the charge-discharge cycle of the battery if the battery is called upon to support the housekeeping bus quite frequently. This problem is eliminated on SERT C by the use of an ampere-hour meter/controller. The ampere-hour meter is described in a later paragraph.

<u>Regulators</u> - Because of the relatively high source voltage, the main regulator is required to operate at 60 volts nominally, and up to 120 volts following an eclipse. Both shunt and switching mode regulators were considered for use on SERT C. Shunt regulators are less complex and lighter than switching mode regulators, but they are less efficient and they impose a widely varying thermal load when required to regulate a widely varying

input voltage. Therefore, it was decided that a switching mode regulator would be the better choice. Several of the units which are a part of the Skylab electrical system could be used on SERT C. The power requirements for Skylab are greater than for SERT C, with the consequence that most of the units are capable of conservatively handling the power levels for SERT C.

The housekeeping bus regulator that is proposed for SERT C is the Gulton model EMVR 144. Some of its design features are modular packaging, short circuit protection, EMI filtering, and low output impedance. Its performance specifications are:

Input voltage:	33 to 125 V dc
Output voltage:	28 V dc
Power rating:	1500 watts nominal
Voltage droop:	0.04 volts/amp, ± 0.002 volts/amp
Efficiency:	94% nominal
Size:	10" x 10" x 4-1/2"
Weight:	14 pounds

The bus regulator consists of five identical pulse-width modulated modules which are current limited to 12.5 amperes. The maximum full output of the unit is 50 amperes and one of the power modules serves as a redundant unit. The regulator power modules receive operating voltages and drive signals from redundant power supply and drive converter modules. The system is sized so that loss of any of the duplicate modules will not affect regulator operation. To meet the SERT C requirements it should be possible to reduce the number of modules to three and still be able to meet all anticipated housekeeping power requirements. Two modules would give 25 amperes at 28 volts, (700 watts) with the third module serving as a redundant unit. By reducing the number of modules, the weight of the regulator can be reduced to approximately 9 pounds. It is proposed that a complete second regulator be included in the power system in a standby state.

Battery - During launch, eclipses, and during the portion of parking orbit when the solar array has not been extended, the entire satellite electrical system will depend upon a battery for its source of electrical energy. Due to the spiral from a low earth orbit out to a synchronous orbit, there will be a very large number of eclipses during the lifetime of the mission. As a result, the battery will have to undergo the same number of charge-discharge cycles as eclipses. Therefore, the only type of battery that can be considered for this mission is a nickel-cadmium (Ni-Cd) battery due to its high cycling capability. Table 4.5.2 indicates the important points that have to be considered in designing and operating the SERT C battery system.

Due to the length of the mission (5 years) and the number of charge-discharge cycles, it is recommended that a back-up battery be included in the power system. There is some flexibility available in the manner in which the back up battery is used. One method is to have the main battery on line and the second battery in standby. A second method is to connect the main and standby batteries in parallel so they share the load. Another possibility would be to alternately use the batteries for every other eclipse. In this way, each battery would have a much longer time to charge. This would be especially important during the spiral-out phase of the mission where some of the eclipses are separated by as little as 2 hours.

Ampere-Hour Meter - Because of the large number of charge-discharge cycles that occur during the spiral-out phase of the mission, it is felt that a more accurate means of monitoring the charging and discharge of the battery should be employed in the SERT C electrical system. One such system which uses ampere-hour meters is presently in use in the Gulton Airlock system for Skylab. The Gulton ampere-hour meter which could be used in SERT C with slight modification is Gulton model EMAM 131-1. It weighs about 5 pounds and has dimensions of 8" x 4" x 3". The ampere-hour meter is operated directly from battery energy and monitors the input and output ampere-hours of the nickel-cadmium battery. The meter also acts as the prime control for battery charging.

The ampere-hour meter is of the analog to digital type. The analog current information is converted into a series of pulses within the meter. These pulses trigger a digital counter chain which through its arrangement of gates can count either in a forward or backward direction. The ampere-hour memory is volatile; a charge state will not be stored if the ampere hour meter loses input power. At the output of the ampere-hour meter the digital signal is converted back to an analog form which provides a control signal to the battery charger logic. Ampere-hour meter accuracy is better than ±2% over the normal range of charge-discharge currents. Because of its importance to the charging system, it is recommended that the electrical system contain a redundant ampere-hour meter.

Battery Charger - Since the array voltage is 60 volts and the battery voltage at charging is 43.5 volts, a buck-type charger is proposed. The charger considered for SERT C is Gulton model EMBC 181, which is a modified SERT II switching regulator. It has been modified to function as a charger which can be integrated with the ampere-hour meter so that precise charge control can be maintained during the mission. The battery charger is current limited and supplies a temperature-compensated charge voltage with provision for a trickle charge mode. Because of the importance of the battery charger, a second, redundant charger is recommended for SERT C. In addition, the nickel-hydrogen ($Ni-H_2$) battery which will be part of an experiment on this mission will require a battery charger. It is possible to use the back up charger in order to charge the Ni-H_2 battery. The proposed charger weighs approximately six (6) pounds.

Special Purpose Power Conditioners - The preceeding discussions were concerned with the main elements of the power system. Yet to be considered are the power conditioners needed for parts of the telemetry and attitude control systems. The integrating gyro requires a power conditioner to generate several well-regulated sine and square waves. It has not yet been built in a flight-type configuration. After some thought, it was decided to allow eight (8) pounds for this conditioner.

The telemetry transmitters and receivers require several dc voltages other than 28 volts. A transmitter requires +16.5 volts at 12 watts while a receiver needs +12 volts and +5 volts at a total of 1.5 watts. To handle these requirements, it was decided to use Power Cube Corporation "Circuitblock" modules. Powercube products have flown in Apollo and Skylab programs and are being used in the SPHINX satellite which Lewis is presently working on. The following Powercube modules would be used: 1 - 27G100 W 40 generator, 1 - 12TRC10 regulator, 1 - 5TRC10 regulator, and 1 - 15/16.5 TRC10 regulator. For redundancy, another complete system would be needed. Each module is 1" x 1" x 2" and weighs 2.8 ounces. Therefore, for two systems the weight would be 1.4 pounds. The efficiency would be around 90 percent.

Harness, Switches and Fuses - The last items to be considered are the power system harness, switches, and fuses. Through general calculations it was determined that about forty (40) pounds of harness would be needed for the power system. This would include connections; however, it does not include the ion thruster harnessing.

As can be seen from the block diagram of figure 4.5.2, fusing would be required for the ion thrusters, radar, television camera, and television transmitter. These experiments are not as vital to the well-being of the spacecraft as the rest of the items on board. If they start to draw excessive power, they then become expendable. These units also happen to be the largest users of low voltage regulated power on board the spacecraft.

There will be a number of switches used in the power system for SERT C. Wherever redundancy exists, switches have to be used in order to choose between the main component and the redundant one. Switches are also used to keep a unit isolated from the rest of the system until it is needed. For SERT C it is possible that up to five (5) pounds of switches and fuses will be needed.

Summary - Table 4.5.3 contains a summary of the power system components along with their expected weights. As can be seen, the batteries weigh between forty (40) and forty-five (45) pounds and the rest of the power system weighs 94.4 pounds.

Total weight is between 134.4 and 139.4 pounds.

Before leaving the spacecraft power system section, something should be said about power constraints during the mission. For part of the spiral-out phase of the mission, there will be approximately 8000 watts of solar array power available for the 30-cm ion thrusters. However, after 50 to 100 days, the 300 volt array power will degrade to approximately 4000 watts. As a result, two 30-cm ion thrusters will be able to operate at full thrust for the first 25 days; however, as the array degrades the thrusters will be throttled back equally. During spiral-out at least one and possibly two 8-cm ion thruster will be used for short periods to dump the momentum wheels. The optimum arrangement for powering these thrusters is under study. Once the spacecraft gets to synchronous orbit there is considerable excess power available. Use of this excess power for experiments is also under study. Although there is presently more housekeeping solar array power available than required, the battery system is at about its maximum loading if long lifetime is desired.

TABLE 4.5.1 - SERT C POWER REQUIREMENTS

	Launch	Parking orbit	Spiral out	Eclipse operation	Near synchronous orbit	Synchronous orbit	Terminal rendezvous
TT&C	48.5/48.5	48.5/48.5	48.5/48.5	3.1/3.1	48.5/48.5	48.5/48.5	48.5/48.5
Integrating gyro	24.5/57.5	24.5/57.5	24.5/57.5	24.5/57.5	24.5/57.5	24.5/57.5	24.5/57.5
Attitude control		31/31	31/31	31/31	47/47	26/26	42/42
Solar array orientation mech.		24/24	16/16	6/6	24/24	6/24	24/24
30-cm ion thrusters (4)			5760/5760 (1)		4000/4000		0/2880 (2)
8-cm ion thrusters (4)			0/306 (3)		0/306 (3)	0/306	0/306
Computer		20/40	20/40	20/20	20/40	20/40	20/40
Radar							123
Television camera and transmitter							138

Power: $\frac{\text{Average}}{\text{Peak}}$ watts

Note: (1) and (2) from 300-volt bus
(2) Power correct if 30-cm thruster used for station walking
(3) There is a possibility of two 8-cm ion thrusters running at the same time. If only one is running at any one time, then the power numbers become 0/153 for one 8-cm thruster running. The power for the 8-cm thrusters comes from a nominal 60-volt bus, high voltage solar array (reconfigured from 300-volt array) and/or Ni-H_2 battery.
(4) Power requirements assume power processor design efficiency goals are achieved.

TABLE 4.5.2 - SERT C NICKEL-CADMIUM BATTERY

Number of cells	28
Minimum voltage	30.7 volts, 1.1 volts/cell
Maximum voltage (charge)	43.5 volts, 1.55 volts/cell
Average discharge voltage	35 volts, 1.25 volts/cell
Average discharge current	3.7 amperes
Capacity per cell	6.85 A-H
Battery weight	~20 pounds
Storage temperature	-40° C to 50° C
Charge and discharge temperature range	-10° C to 40° C
Recommended operating temperature	5° C to 10° C
Impedance	~2-4 milliohms

TABLE 4.5.3 - SERT C POWER SYSTEM WEIGHTS

Component	Component weight	Total weight, lbs
Ni-Cd battery	~20 lbs each	~40→45
Ampere-hour meter	5 lbs each	10
Battery charger	6 lbs each	12
Main regulator	9 lbs each	18
Telemetry power conditioner	11.2 oz each	1.4
Integrating gyro power conditioner	8 lbs each	16
Wiring harness	40 lbs	40
Switches and fuses	5 lbs	5
TOTAL WEIGHT		134.5 lbs→ 139.4 lbs

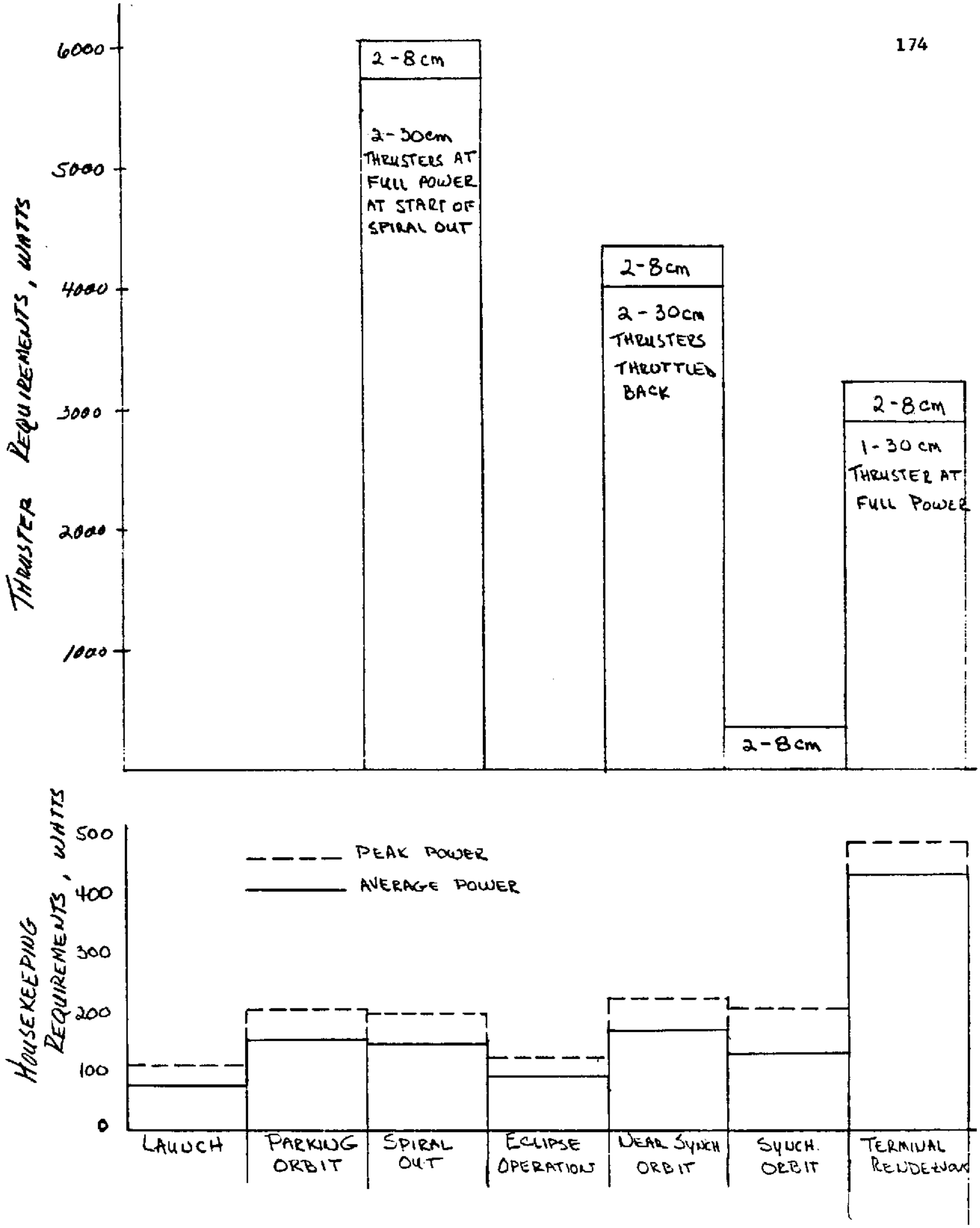


Figure 4.5.1 - SERT C Power Profile

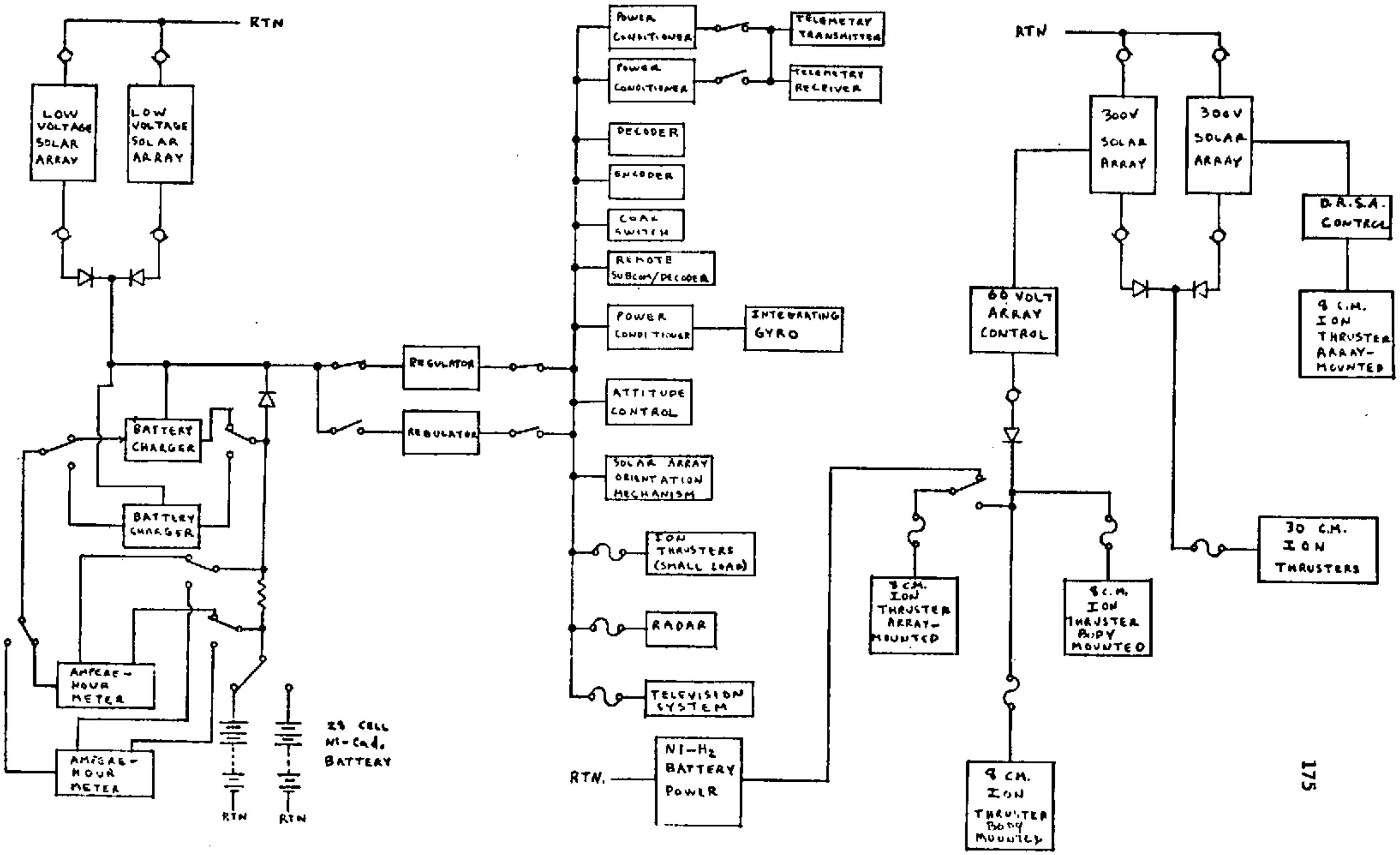


Figure 4.5.2 - SERT C Power System

4.6 DRSA Experiment

The Directly Regulated Solar Array (DRSA) experiment will be operated during the synchronous orbit phase of the mission. Three regulated solar array sections will provide power directly from the array to the 8-cm ion thruster located on tip of the south solar array. The beam, discharge, and accelerator ($\simeq$ 85% of total thruster power requirements) will be powered by DRSA sections. The small heater-type thruster loads will be supplied by conventional power processing circuitry. Figure 4.6.1 is a block diagram of the 8-cm thruster power system.

The beam and discharge solar array sections are reconfigured from 300 Volt panels of the solar array. The array management electronics for regulation and reconfiguration of these supplies are located in the electronics package mounted on the outboard end of the solar array, adjacent to the thruster. The electronics package also contains the power conditioning circuitry for the low power thruster loads. The accelerator supply consists of a small number of edge-illuminated solar cells mounted on the sun-facing surface of the electronics package. This supply is self-regulating and requires no active control.

The beam supply requires that solar array panels be reconfigured by switches from parallel operation at 300 Volts for orbit-raising to series operation at 1200 Volts. Figure 4.6.2 is a schematic of the beam supply. Regulation of the beam supply is accomplished by means of shorting switches across blocks of cells on one of the panels. These switches will short-out excess solar cell blocks, removing them in a non-dissipative manner from the array to maintain the required voltage. Switch operation will be controlled by on-board logic, based on sensed power conditions at the load. The shorting switches are solid state circuits, incorporating voltage isolation, where necessary, and self-protection from load arcs. Switch operation is controlled by an on-board logic system consisting of a load sensor, reference, comparator, an up-down counter, and digital clock. Four regulation switches and four additional leads are required on one of the beam supply panels. In addition, these four panels will require by-pass diodes and one blocking diode per panel for failure protection.

The discharge supply consists of 300 Volt panels tapped and paralleled to produce the 40 Volts required. Figure 4.6.3 is the discharge supply schematic. One solar cell string on each of the panels is tapped at four places for supply regulation. The discharge supply is regulated in the same manner as the beam supply.

The accelerator grid supply consists of edge-illuminated solar cells, which produce high voltage at low current. The array will consist of a series string of 22 edge-illuminated cells, every two of which are paralleled by a 50 V zener diode. Figure 4.6.4 is a schematic of the accelerator supply. The zeners not only regulate the array voltage, but also provide by-pass protection in the event of a cell-open failure. One segment of the array is normally shorted-out by a switch. The switch will be opened, adding the segment to the array, in the event of a cell failure (open or short) or a zener short. The failure of a zener diode open will not cause the array voltage to exceed the regulation tolerance. Since the required area for this supply is small, the cells and zener diodes will be mounted on the sun-facing surface of the electronics package. The shorting relay located within the package is controlled by on-board logic consisting of a voltage reference and comparator. The vaporizers, keepers, and heaters are powered thru conventional power conditioning circuitry located in the electronics package. Figure 4.6.5 is a block diagram of the thruster power system. These low power supplies are mag. amp. controlled and driven from a single inverter. A single heater supply is switched between the neutralizer and cathode heaters since they do not operate simultaneously. The start voltage for the cathode and neutralizer keepers is supplied from the accelerator supply in order to eliminate a second output on the keeper supplies. The DRSA beam supply, rather than the accel, can be used for the keeper start if a higher voltage is required. The normal engine start up commands will be used for switching the supplies to the various loads during the start up sequence.

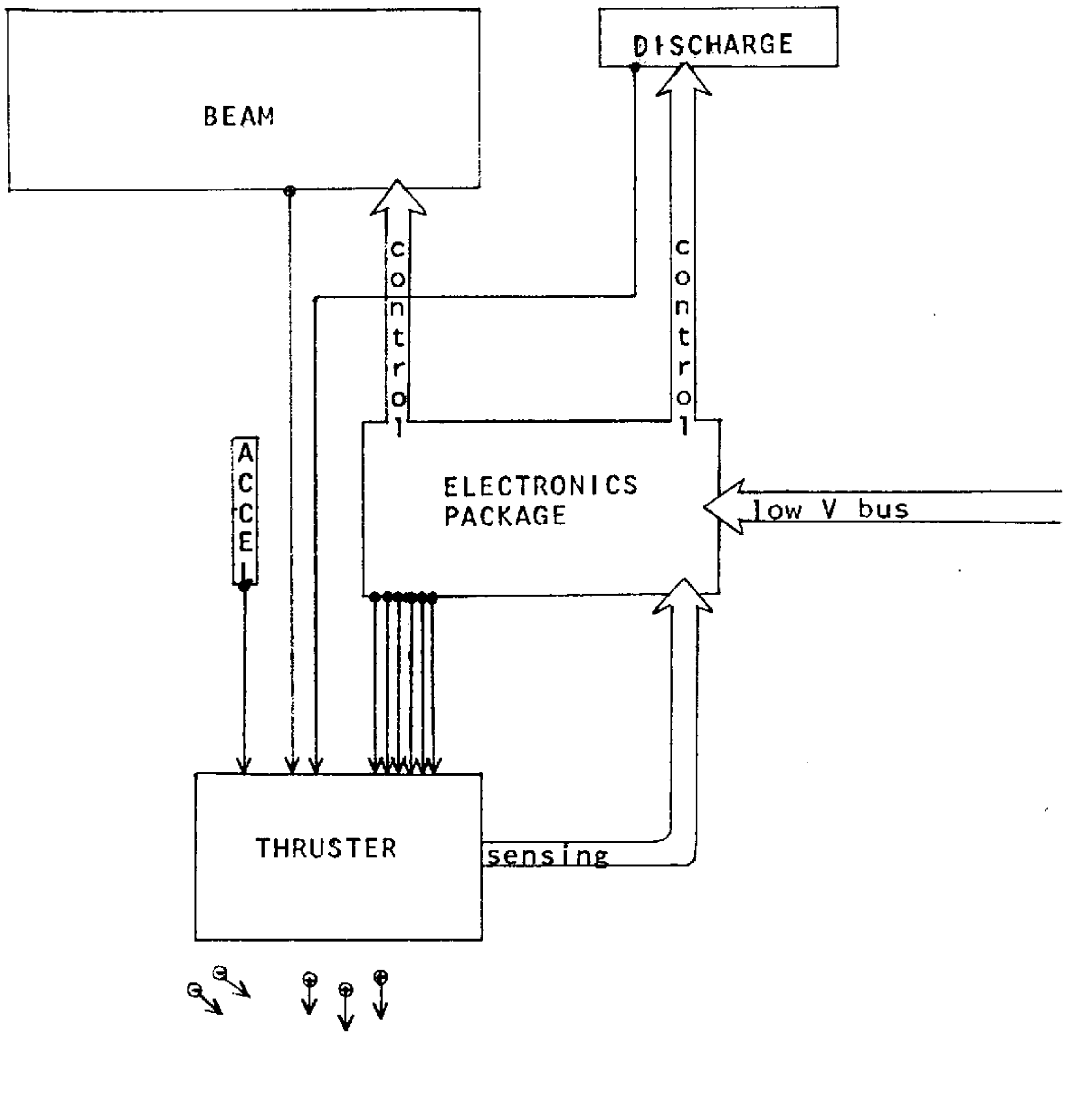


Figure 4.6.1 HYBRID POWER SYSTEM BLOCK DIAGRAM

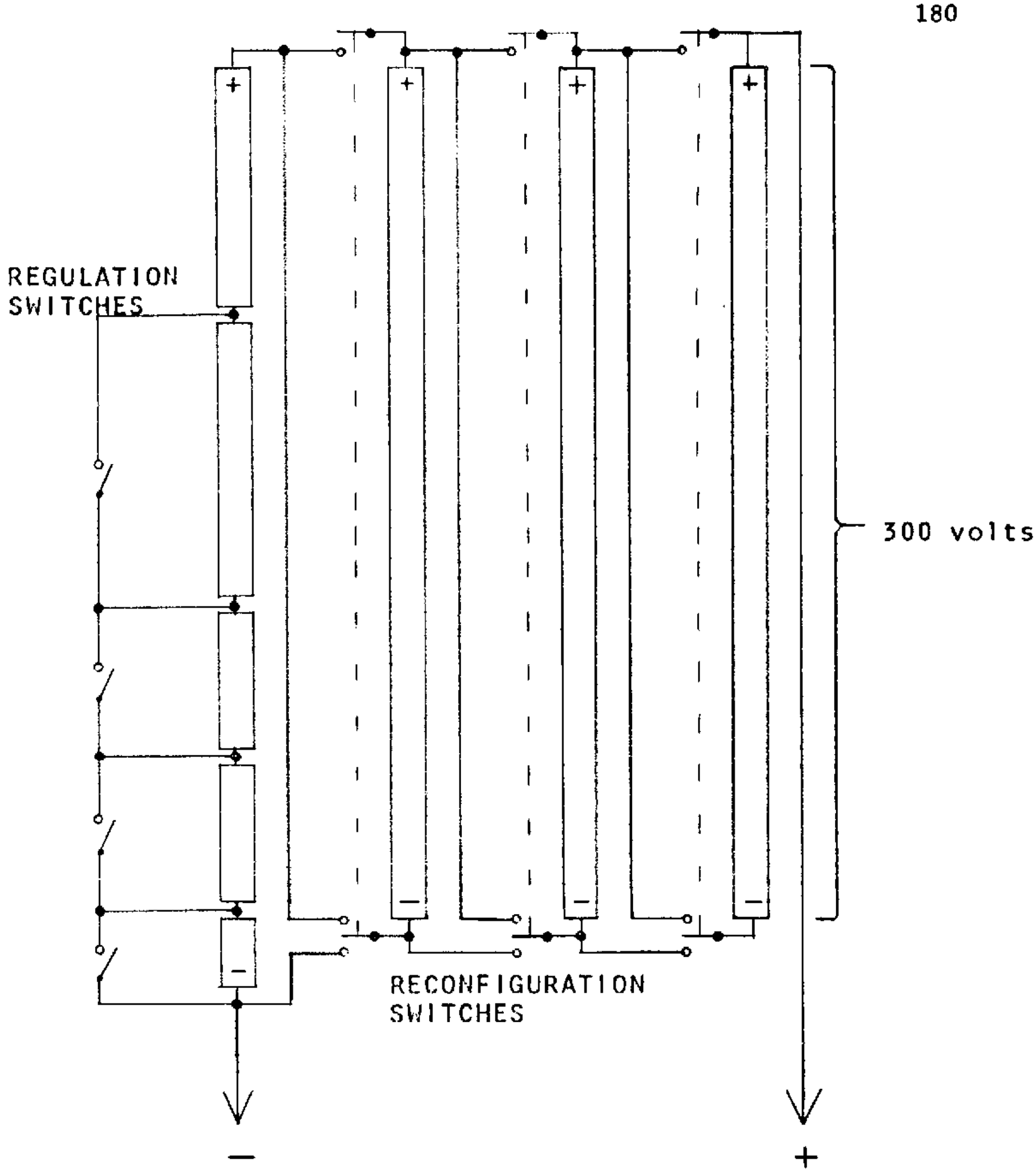


Figure 4.6.2 DRSA BEAM SUPPLY SCHEMATIC

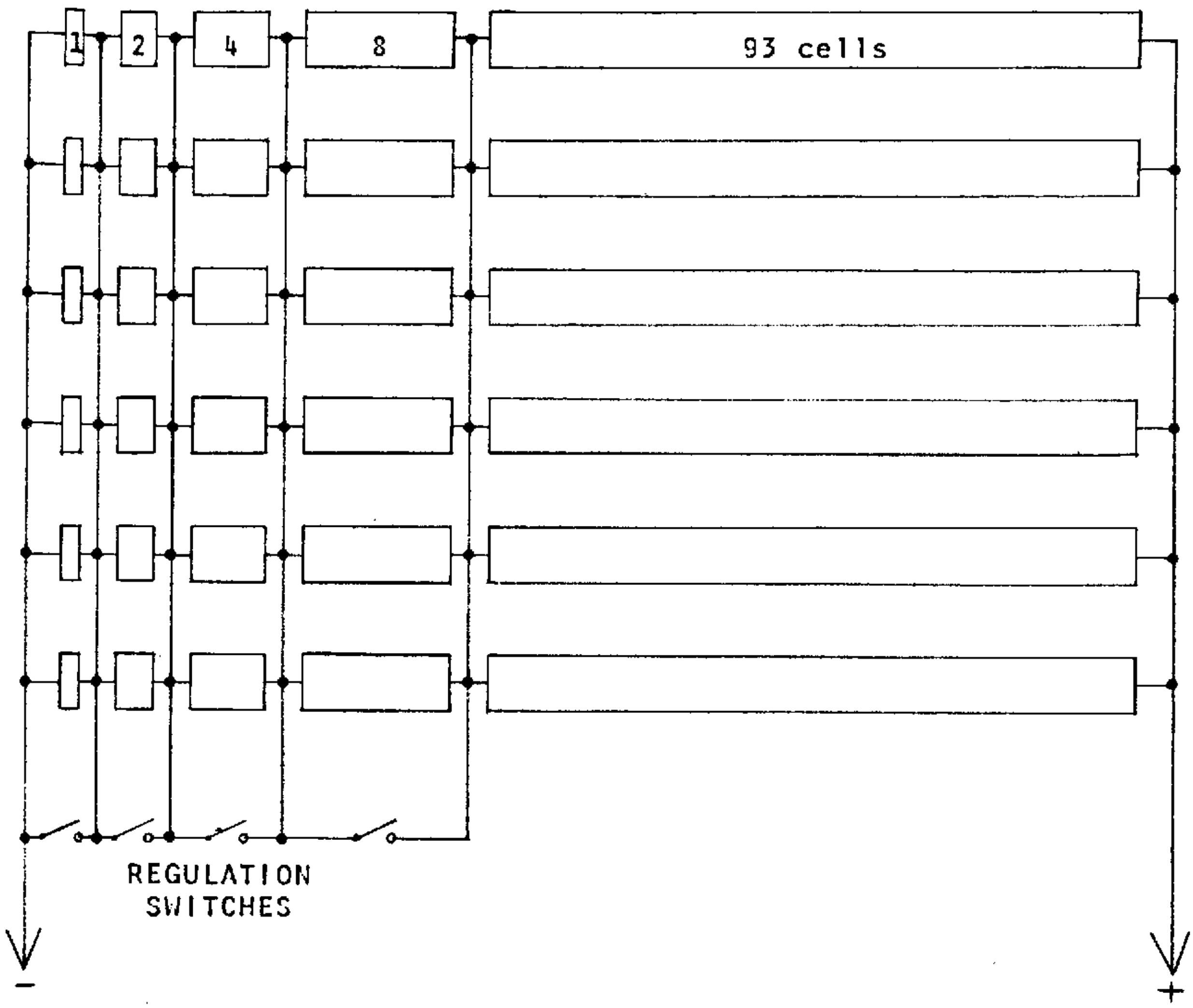


Figure 4.6.3 DRSA DISCHARGE SUPPLY SCHEMATIC

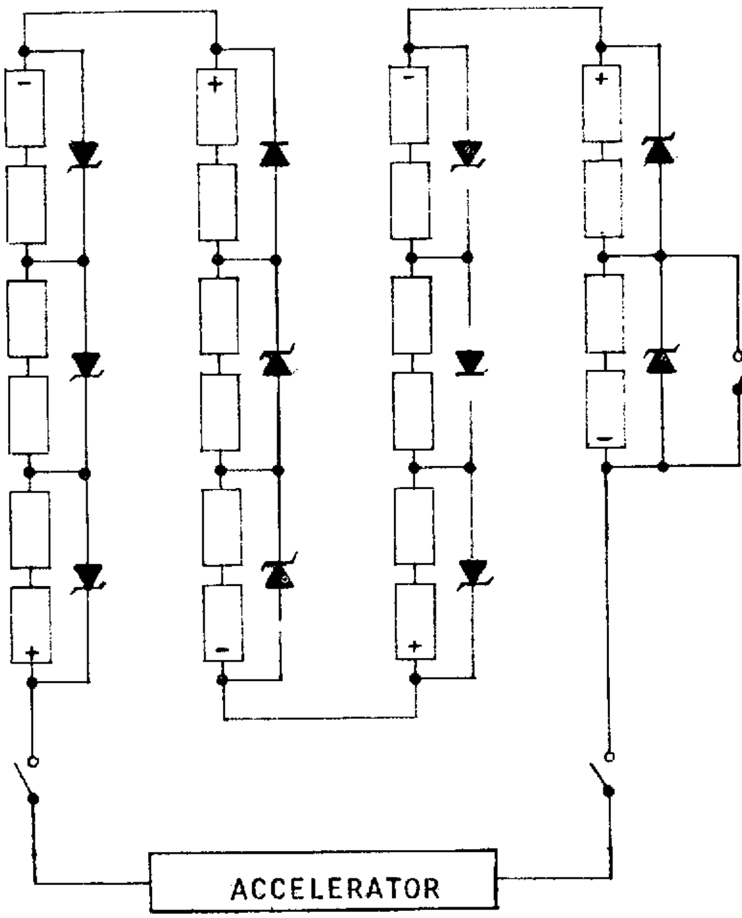


Figure 4.6.4 DRSA ACCELERATOR SUPPLY SCHEMATIC

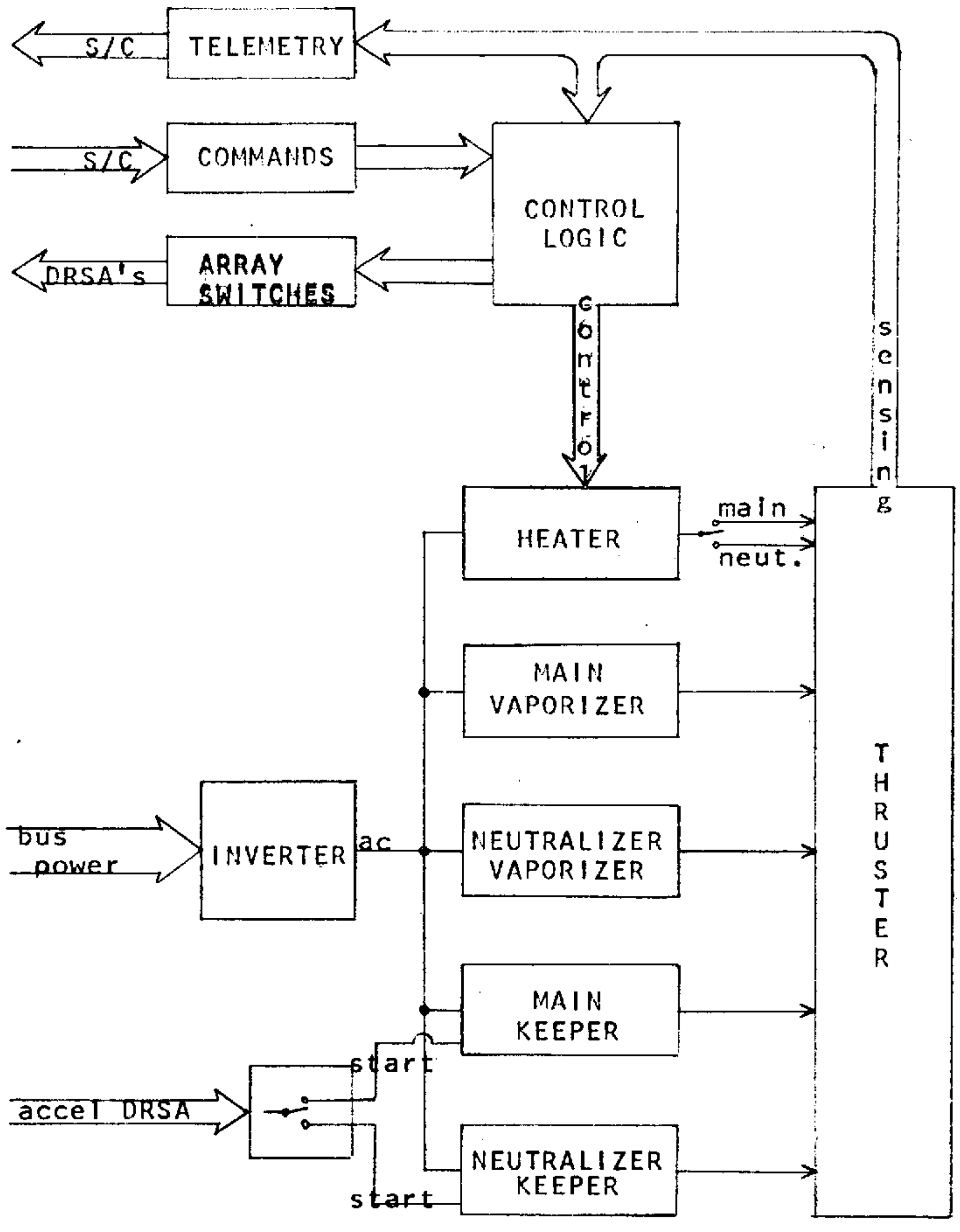


Figure 4.6.5 ELECTRONICS BLOCK DIAGRAM

4.7 Telemetry, Tracking and Command

Introduction - The tracking, telemetry and command functions of the SERT C spacecraft are performed by four major subsystems. These are the antenna, transponder, data handling and command subsystems. The subsystems are designed to meet the data and command requirements of the spacecraft and the experiments. In addition to these requirements, three constraints are imposed on the subsystems:

1. Maximum use shall be made of designs and hardware from existing programs in order to reduce program costs and development risks.
2. The subsystems shall be compatible with the existing Spaceflight Tracking and Data Network (STDN) facilities.
3. If the high gain antenna is not properly pointed toward the earth, the spacecraft shall be capable of receiving commands and transmitting telemetry data.

A block diagram showing a conceptual design of a communications system which meets the above requirements and constraints is shown in figure 4.7.1.

Antennas - The antenna system consists of two separate antennas. A 2 foot parabolic reflector with an S-band feed is mounted on the earth facing side (after acquisition) of the spacecraft body, and is aligned with the yaw axis of the spacecraft. This antenna provides 20 dB gain (at 55% eff.) and a half-power beam width of 17°. This angle subtends the geoid from synchronous altitude. The feed for this parabolic reflector is to be designed to accept a signal from the spacecraft television transmitter in addition to its functions in support of the telemetry and command system. The amount of "shadowing" of the parabolic reflector by the relatively large S-band feed has yet to be determined. A preliminary estimate of 5.6 dB is used to account for this shadowing in the link calculation. During transfer orbit or at times on-station, the high gain antenna will not be pointing toward the earth. In order to assure that the spacecraft is capable of receiving commands under these conditions, an omnidirectional antenna is to be mounted on the spacecraft. The form of this omnidirectional antenna has not been defined because the antenna pattern is strongly dependent on the geometry of the spacecraft. A unique design will be necessary to minimize the number and depth of the expected nulls in the omnidirectional pattern. The gain of the antenna is to be better than -10 dBi over 95% of a sphere which has its center at the spacecraft. This omnidirectional antenna will not be required to support the TV transmitter at any time.

Transponder - The transponder contains parallel redundant command receivers and standby redundant telemetry transmitters. Both receivers will be operating at the same time, but only one transmitter will be operating. The second transmitter will be turned on and switched to the load by ground command. The uplink and downlink frequencies will be assigned within the S-band ranges of 2.025 - 2.12 GHz and 2.2 - 2.3 GHz respectively.

In the commanding mode, the receivers demodulate the S-band signal and feed it to the command decoder. In the ranging mode, the ranging code or ranging tones are fed to the transmitter for retransmission to earth as phase modulation of the telemetry carrier. The transmitter will have the capability of generating the output frequency from an internal oscillator or coherently from the received ranging carrier.

Figure 4.7.2 shows a block diagram of the antenna system and the S-band transponder.

Command Decoding - Decoding of the commands from the transponder is performed by the command decoder on the spacecraft body and by remote decoders on each of the solar arrays. The command decoder contains circuitry which will reject any command that does not conform to an address code which is to be assigned to the spacecraft in accordance with the GSFC Aerospace Data Systems Standards. In order to provide an adequate number of commands, a nine bit word has been selected. The 9 bit word can yield a maximum of 316 commands, and provide bit pattern separation between the discrete commands and the data word commands. Since the present command requirements list (Table 4.7.1) has a total of 151 commands, there is the capability of providing redundancy for all of the discrete commands, and to provide four commands which are presently designated as spares.

Four data word commands are required for the spacecraft. Each thruster uses a portion of a data word command to set the operating conditions of the thruster/power conditioner. Eight bits of this data word command are used for that purpose. Three bits are required to steer the 8 bits to the proper registers in the power conditioner, and four bits are required to select the proper thruster. The remaining bit is not used. The TV camera uses the second data word command to perform the manipulating functions. This word is defined by the requirement to be compatible with the Lunar Rover Camera. The third data word command is required to set the status of the "dwell" capability of the telemetry system. This dwell capability is described in the telemetry section.

In order to provide a bit stream for use in re-programming the S/C computer, a 4075 - bit data word command is provided. This is the maximum usable word length allowed by the STDN standards. (21 bits are reserved for spacecraft and computer addressing). In order to provide a fast bit rate for the computer bit stream, a 1 kilobit rate for the command system has been selected.

In order to minimize the number of wires and slip rings from the decoder to the array-mounted thrusters, remote decoders will be mounted on the array, adjacent to the thruster/power conditioners. In order to minimize weight and volume, these remote decoders will be packaged in the same enclosure with the subcommutators for the array thruster/power conditioners.

<u>Verification</u> - In order to provide a complete check on a command that has been received by the spacecraft before the command is executed, automatic ground verification will be performed. This will be accomplished by transmitting the command word portion of the command message to the spacecraft control center by means of the telemetry

system. In the control center, each bit of the original command word will be compared with each bit of the telemetered command word and upon concurrence the intended command will be executed. An execute signal will not be transmitted if the verification process is interrupted or terminated, and if the decoder does not receive an execute signal within a pre-determined period of time the command word which has been stored in the decoder will be erased. A separate subcarrier oscillator will be provided in the spacecraft for the verification process so that commands may be verified and executed independently of the ground station decommutator synchronization status.

Data Handling - An 8-bit word is selected to provide an analog to digital conversion resolution of better than 0.5%. The input to the A/D converters is to be from 0 to 5 V. Status monitors in the form of bi-level flags will also be provided.

In order to minimize the complexity of the spacecraft encoder, two data rates have been selected. The fast data rate is 1 sample per second which meets the preliminary requirements of AC & SK, and exceeds the preliminary requirements of the other spacecraft subsystems. The slow data rate (32 sec sampling interval) has been selected to meet the preliminary requirements of all subsystems with sampling intervals of 15 sec. or longer. Data compression is to be performed at the Control Center rather than at the spacecraft, since this is more economical.

Table 4.7.2 lists the telemetry requirements. These requirements determine the number of minor frame words, which are listed on the same table. The requirements were developed from a survey of the measurement needs of the various subsystems. From the requirements and the number of minor frames, a telemetry major frame layout was generated and is shown in figure 4.7.3. In addition to the requirements, six words are allocated as spares, and two of these spare words incorporate the computer monitoring function. At the slow rate, this yields 176 spare measurements, exclusive of the computer monitoring function.

Words 12 through 18 are allocated for the thruster/power conditioner subcommutators. Each of these subcommutators provide the same interface with a thruster/P.C. However, the body thruster T&C functions are wired directly to the telemetry encoder and command decoder, and the array thrusters are wired to the remote subcommutator/decoder and thence to the encoder and decoder. These interfaces are shown in figure 4.7.4. The remote subcommutator/decoder concept reduces the slip ring requirements from 52 per thruster/P.C. to 8 per thruster/P.C. A similar arrangement is to be applied to the DRSA experiment.

In order to read a thruster/P.C. measurement at the fast rate (1 per sec.), the telemetry encoder may be commanded to "dwell" on any one of the 32 measurements which are contained in a thruster/P.C. subcommutator. This capability provides the option of observing trends in a measurement that would not be observed at the 32 second major frame rate.

The computer requires two adjacent words in a minor frame, but there is no requirement to monitor these words in every minor frame. Therefore, these words are transmitted every fourth minor frame.

Link Calculations - Table 4.7.3 shows link calculations for the command system for three different conditions. When the maximum STDN capability (a 10kW transmitter and an 85' dish) is used there is an adequate margin. When a 40' dish is used, the margin is still adequate, but 4.5 dB less than the 85' dish. If a 10 kW transmitter is used with the LeRC 15' dish, the margin is still adequate. These calculations take into consideration the possibility that the 2' spacecraft antenna is not properly pointed at the earth. The conclusion is that adequate command capability exists.

Table 4.7.4 shows the telemetry link calculations for the worse case (LeRC reception). With the 2' dish, there is an adequate data margin. When the omnidirectional antenna is used, the data margin is not adequate. The result is a significant increase in the probability of bit error. The probability of error is 4 errors per 100 bits received for the omnidirectional antenna compared with the 2' dish error probability which is less than 1 error per million bits received. The omnidirectional antenna problem can be overcome by using a STDN station with a 40' or larger dish to receive telemetry data.

The link calculations for the STDN tone range and range rate are shown in Table 4.7.5. The calculations are performed for the maximum slant range which occurs at zero degrees elevation. A STDN 40' dish and a 10 kW transmitter yield a 10 dB signal to noise ratio when the S/C omnidirectional antenna is limited to nulls which are 10 dB or less below an isotropic antenna. Use of an 85' dish will increase the S/N to 16 dB.

Table 4.7.6 summarizes the weight, volume and power requirements for the T&C system of SERT C.

TABLE 4.7.1 - COMMAND REQUIREMENTS

	DISCRETE COMMANDS	DATA WORD COMMANDS
THRUSTERS	14	1 - 16 bit
INTEGRATED ARRAY	7	
AC & SK	32	
TV CAMERA	16	1 - 16 bit
SAOM	12	
T & C	30	1 - 16 bit
RADAR	6	
SOLAR CELL EXP	2	
POWER	30	
COMPUTER	2	1 - 4075 bits
TOTAL	151	4
REDUNDANT	151	NOT APPLICABLE
SPARES	4	NOT APPLICABLE

TABLE 4.7.2 - TELEMETRY REQUIREMENTS

SUBSYSTEM	MINOR FRAME WORDS	MEASUREMENTS @1 Sec	MEASUREMENTS @32 Sec	DIGITAL FLAGS
THRUSTER/P.C. (7)	7		224	7
AC & SK	21	21		19
RADAR	10	10		1
SAOM	1		12	2
INT ARRAY			7	0
THERMAL			13	0
SOLAR CELL EXP	2		64	0
TV CAMERA	0			1
CMD VERIFY	0			5
POWER DIST. SYS	1		24	20
SYNC, AGC'S, FRAME COUNT, DWELL STATUS	6		192	10
COMPUTER	6		16*	7
SPARES			176	
	54	31	728	72

* 16 bit word sampled every 4th second

TABLE 4.7.3 - COMMAND LINK CALCULATIONS

	STDN MAX (10kW+85')	STDN NOM. (10kW+40')	LeRC MAX (10kW+15')	
Total ERP	120	115.5	107	dBm
Propagation Losses	-190	-190.0	-190.0	dB
S/C Ant. Gain	19.5	19.5	19.5	dB
Passive Element Losses	-2.0	-2.0	-2.0	dB
	------	------	------	
Total Recvd. Sig. Pwr.	-52.5	-57	-65.5	dBm
Expected Null Depth	-29.5	-29.5	-29.5	dB
Polarization Loss	-3	-3	-3	dB
	------	------	------	
Expected Null Level	-85.0	-89.5	98.0	dBm
Cmd. Rcvr. Threshold	-110	-110	-110	dBm
Signal Margin	+25	+20.5	+12	dB
Operating Margin	-3	-3.0	-3.0	dB
Adjusted Sig. Margin	+22	+17.5	+9.0	dB

TABLE 4.7.4 - TELEMETRY LINK CALCULATIONS

	2' Parabolic Antenna	Omni-directional Antenna	
Transmitter power (2 watts)	+33	+33	dBm
Modulation Loss	-4.0	-4.0	dB
S/C Cable Loss	-0.5	-0.5	dB
S/C Duplexer Loss	-0.5	-0.5	dB
TV Diplexer Loss	-1.0	0	dB
S/C Ant. Gain	20.7	-10.0	dB
Feed Shadow Loss	-5.6	0	dB
Path Loss (35,700 km)	-190.57	-190.57	dB
Ground Ant. Gain (15' Para, 55% Eff.)	38.2	+38.2	dB
Received Signal Power	-110.3	-134.4	dBm
Noise Power Density ($T_E = 134^\circ$ K)	-177.3	-177.3	dB/Hz
Bandwidth (1 kHz)	30.0	30.0	dB-Hz
S/N	37.0	12.9	dB
S/N Req'd. for 10^{-5} BER (FSK)	13.4	13.4	dB
Implementation margin	3.0	3.0	dB
S/N Req'd. for ØL	3.0	3.0	dB
Data Margin	+17.6	-6.5	dB

TABLE 4.7.5 - RANGE AND RANGE RATE LINK CALCULATIONS

Tone Range and Range Rate Tracking at 2.1 and 2.3 Gigahertz

A. Spacecraft RF Systems Specifications

Frequency:	Tracking Down Link	2.3 GHz
	Tracking Up Link	2.1 GHz
Antenna Characteristics		
Type:	Receive	2' Parabolic & -10 dB Omni
	Transmit	2' Parabolic & -10 dB Omni
Polarization:	Receive	Linear
	Transmit	Linear
Spacecraft Transmitter		
Power:	Total	33 dbm (2 watts)

B. Received Power Calculations, 2.3 GHz Down Link

1. Received power at maximum slant range (48,109 kilometers at 0° elevation)

	2' Parabolic	Omni
a. Transmitted power	33 dbm	33 dbm
b. Transmitting antenna gain	20 db	-10 db
c. Passive element and modulation losses, spacecraft transmitter to antenna	-6 db	-6 db
d. Free space attenuation	-193 db	-193 db
e. Receiving antenna gain (40' para)	46 db	46 db
f. Receiving polarization loss	-3 db	-3 db
g. System operating margin	-6 db	-6 db
h. Received signal power	-109 dbm	-139 dbm
i. R&RR receiver threshold	-156 dbm	-156 dbm
j. Signal to noise ratio	+47 db	+17 db

TABLE 4.7.5 - Concluded

C. Received Power Calculations, 2.1 GHz Up Link

1. Received power at maximum slant range (48,109 kilometers at 0° elevation)

	2' Parabolic	Omni
a. Transmitted power (10 kW)	70 dbm	70 dbm
b. Transmitting antenna gain (40-foot)	46 db	46 db
c. Passive element losses, spacecraft transmitter to antenna	-4 db	-4 db
d. Free space attenuation	-193 db	-193 db
e. Receiving antenna gain	20 db	-10 db
f. Receiving polarization loss	-3 db	-3 db
g. System operating margin	-6 db	-6 db
h. Received signal power	-70 dbm	-100 dbm
i. Receiver noise power (nominal value)	-110 dbm	-110 dbm
j. Signal to noise ratio	+40 db	+10 db

Note: When an 85-foot antenna is used, received signal power will be 6 db higher than the levels shown above.

TABLE 4.7.6 - T&C SUMMARY

	No. Req'd.	Wt(lbs)	Volume	Power: Normal	Power: Decoder Stby	Power: Eclipse Operation
2' Parabola	1	10	2' Dia.	0	0	0
Omni Ant.	1	8	46" Annulus	0	0	0
TV Diplexer	1	0.25	4x2x1.5	0	0	0
Hybrid	1	0.29	3x1x3	0	0	0
Duplexer	2	2.25(both)	6x3x2(both)	0	0	0
Receiver	2	2.0 (ea.)	6x3x1$\frac{3}{4}$ (ea.)	3.0	3.0	3.0
Decoder	1	7.1	7.3x4x4.8	14.0	.1	.1
Encoder	1	7.0	6x6x3.2	18.0	18.0	0
Transmitter	2	1.1 (ea.)	5x1$\frac{3}{4}$x2 (ea.)	12.0	12.0	0
Coax. Sw.	1	0.75	2x1$\frac{3}{4}$x1.4	20W/10 sec		
Remote Subcom/Decoder	2	3.0 (both)	18 in^3 (ea.) --	1.5	1.5	0
		44.84 ≈45 lbs.		48.5 + 20 W for 10 sec	35.1	3.1

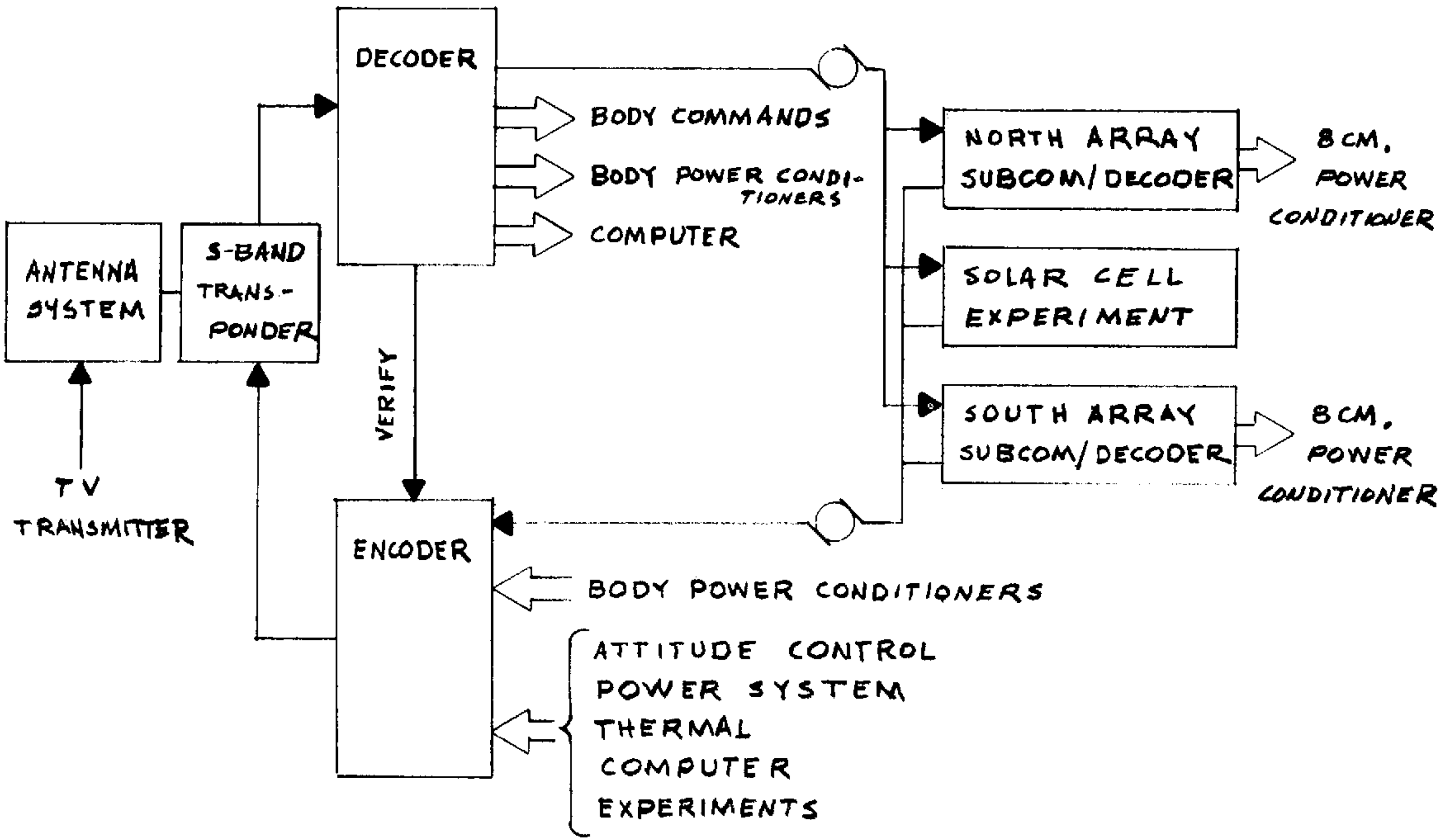


Figure 4.7.1 - SERT C T & C Concept

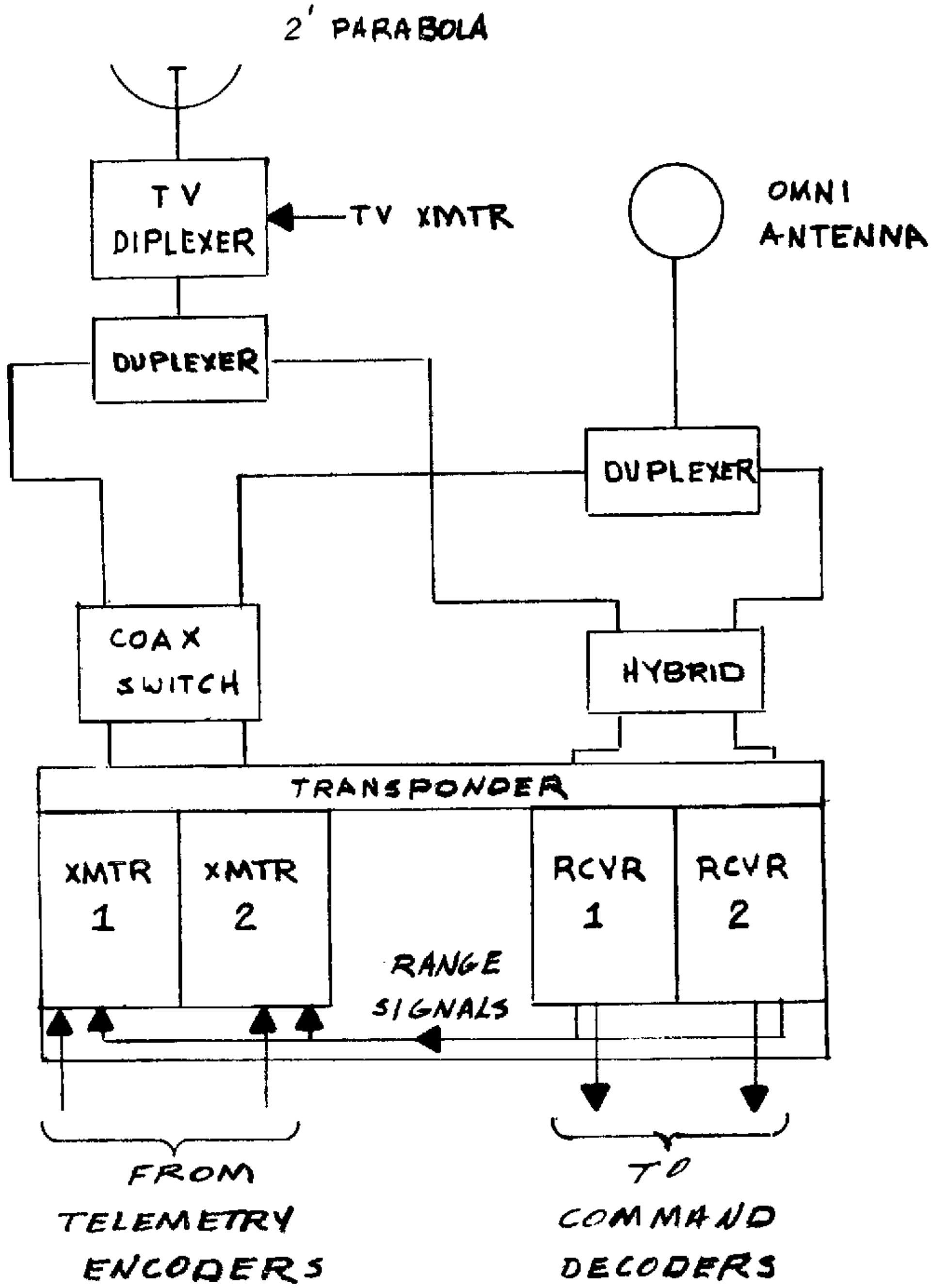
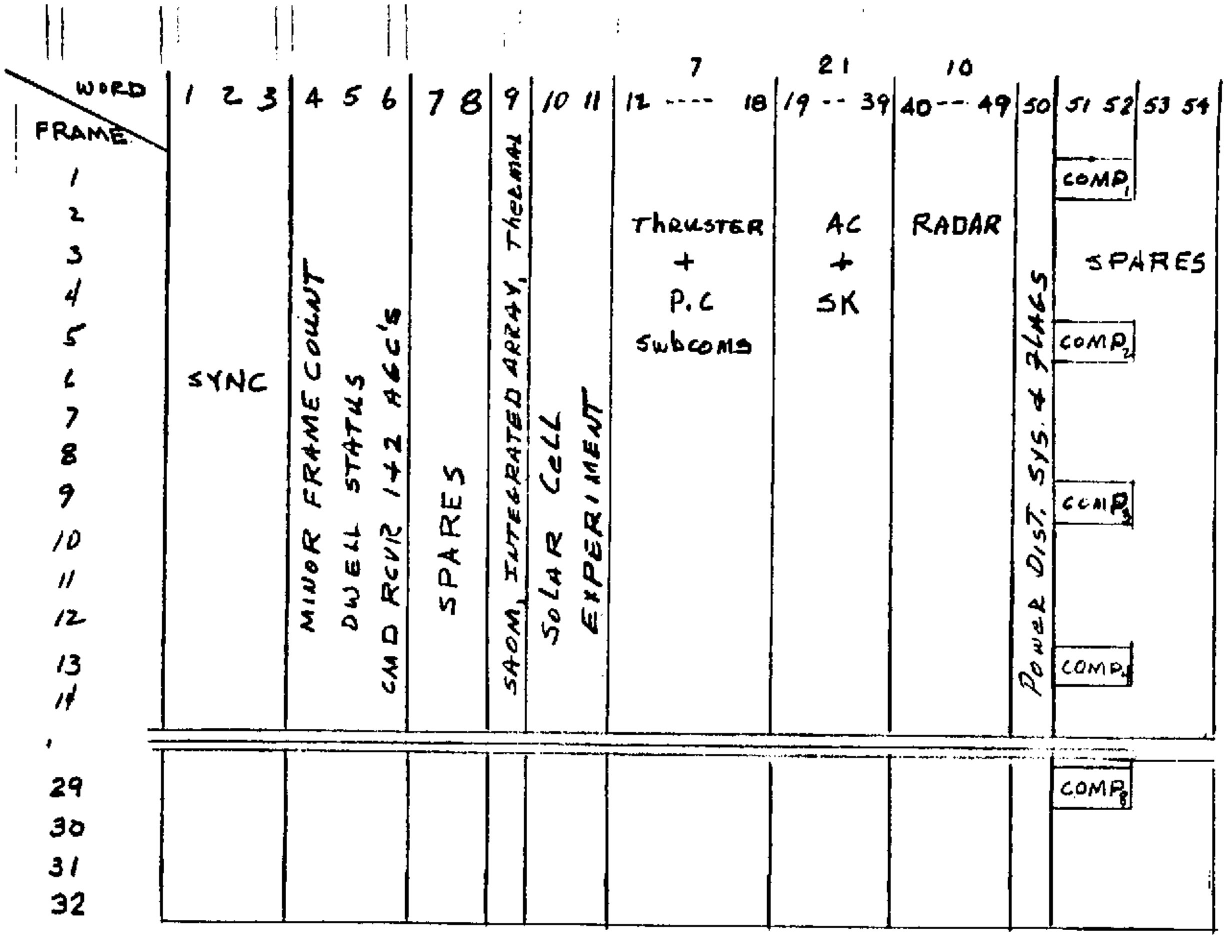


Figure 4.7.2 - SERT C Antennas and Transponder



MINOR FRAME PERIOD	1 SEC
MAJOR FRAME PERIOD	32 SEC
WORD LENGTH	9 (8 BITS + PARITY)
BIT RATE	486 BPS

Figure 4.7.3 - Telemetry Major Frame

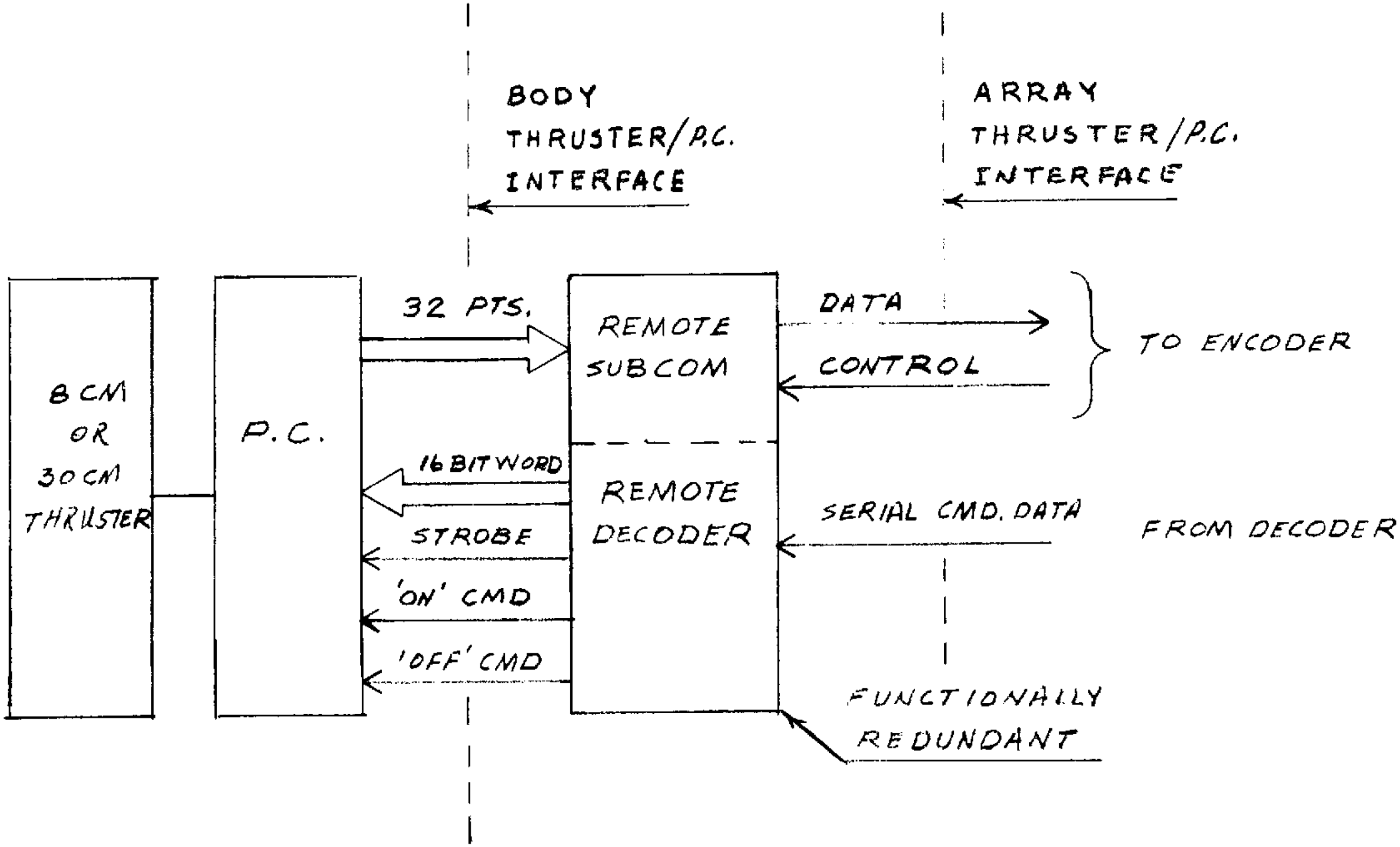


Figure 4.7.4 - Thruster/P.C. - T & C Interface

4.8 Computers

Introduction - The SERT C spacecraft is being equipped with an on-board computer to provide the general functions of performance monitoring and control of various spacecraft subsystems. As a result of the requirements imposed on the SERT C mission, the computer interfaces with most of the other subsystems. Although the principal functions will be attitude control, station keeping, and ion thruster control, the computer can and will assist in maintaining the overall well-being of the spacecraft. In the paragraphs to follow the computer system itself will be described in some detail along with a discussion of its physical and proposed functional relationships to the other subsystems.

Computing System - The computer being proposed for this program consists of a general purpose, binary, parallel, single address central processor. The arithmetic unit is fractional, fixed point, 2's complement with typical execution times of 2.4 to 5.6 microseconds add, 10.4 microseconds multiply, and 30.4 microseconds divide. It has 42 instructions including some double precision and its word length is 16 bits. The memory to be used is nondestructive-read-out plated wire and consists of 16,384 16 bit words. The input/output facilities include a 16 bit parallel input bus, a 16 bit parallel output bus, a serial input bus, a serial output bus, and three interrupt levels.

To provide for a reasonable probability of success for this mission, a fully redundant system is being proposed consisting of two central processing units (CPU) and a total of 32 K of memory. These will be arrayed in a cross-strapped configuration such that either CPU can have access to any portion of memory. Figure 4.8.1 shows a block diagram of this configuration. The total weight of this system will be approximately 14 lbs, with a volume of approximately 200 cubic inches. Assuming a mode of operation where the full redundant system is powered up, it will consume almost 40 watts. If available power is constrained, the system could be wired to function with half this power.

A method for interfacing the various subsystems and devices is shown in figure 4.8.2. The interrelationship of the telemetry and command system, the onboard computer, and the various subsystems is under investigation. Where the subsystem is mounted to the main body of the spacecraft, a parallel I/O interface will be used unless this would impose special conversion requirements at the subsystem. A case in point is the command subsystem that receives its information from the ground serially. By keeping as many communications as possible with the computer on the parallel ports, the CPU work load is lessened. Where data and control transmission difficulty become overriding in importance, serial communication will be selected.

For a better understanding of the computer's interaction with the various subsystems, a discussion pertinent to each follows. The discussion is necessarily general and brief in some cases due to the preliminary nature of the information available and because the interface between the various subsystems has not yet been fully negotiated between subsystem designers.

Attitude Control and Station Keeping Interface - One of the prime functions of the computer is to provide the link between the attitude sensors and the attitude and position determining devices. Herein will reside all the necessary algorithms for effecting attitude control and station keeping. Figure 4.8.3 shows all input and output subsystems and devices which play a part in this process. The earth sensors , sun sensors, and integrating gyros provide the computer with the information it requires to control the momentum wheels and, in addition, the 30-cm thruster gimbals during early mission periods.

Since the momentum wheels must be dumped daily, the 8-cm thruster beams must be deflected at the same time to compensate for the dumping torques. The computer could coordinate this activity based on a preprogrammed command schedule resident in memory. This will permit unattended daily dumping with manual update of the command schedule when position data becomes available from the tracking stations.

Another computer function as shown in figure 4.8.3 is that of the peak power monitor/thrust level control. For effective utilization of all available power during the orbit raising mission, it is necessary for the computer to periodically monitor the solar array and determine the maximum available power. Once determined, the computer can then determine the 30-cm thrusters thrust levels to derive maximum total thrust for minimum fuel consumption at the prevailing maximum power level. The ultimate objective here is to minimize the transit time to geosynchronous orbit position.

<u>30-cm Thruster Control Interface</u> - The 30-cm thrusters make use of PPU's designed for direct digital monitoring and control. As such they are not only ideally suited for computer control but their performance is enhanced by such an interface. As shown in figure 4.8.4 the parallel input & output buses could be used to interface to the PPU's. The output bus is used to transfer digital reference signals for each supply and/or control loop. These serve to establish a particular operating point for each active PPU function. Discrete digital commands are also sent via the output bus. These serve to establish particular modes of operation of the PPU's.

The parallel input bus will transfer performance data from the PPU's directly to the computer. The computer will acquire this data as required by sending appropriate data addresses over the output bus and reading the data value from the input bus. The computer can also pass thruster performance data to the telemetry subsystem via its interface with the computer.

Since the computer receives the data from the PPU's, the performance of the thrusters can be consistently monitored and their operation controlled from the computer. This capability is particularly important during orbit raising when two thrusters are operating simultaneously. If one of these should shut down for any reason, severe attitude disturbances will result within a small period of time if this fact is not quickly sensed and corrective measures taken. In this case it is possible for the computer to sense the shutdown and either shut the other thruster down or gimbal the thruster platform to compensate for the shutdown of one thruster.

8-cm Thruster Control Interface - The 8-cm thrusters use PPU's much like those used for the 30-cm thrusters particularly with respect to the data and control interface. The main difference in the interface characteristics is that rather than being parallel they communicate serially. This feature could be readily accommodated by using the computer's serial input and output buses as shown in figure

4.8.5. This technique is particularly important on SERT C since the two 8-cm thrusters which are used for north-south station keeping are mounted on the ends of the solar arrays. In this case the serial I/O communication makes possible full control and data monitoring by the computer over a couple of pairs of wires. And, as in the case of the 30-cm PPU's, the PPU's data can be sent from the computer to the telemetry system via its interface with the computer.

It should be noted from figure 4.8.5 that one 8-cm thruster acquires some of its power from the directly regulated solar array (DRSA) experiment. Since this type of array is intended to be directly controlled digitally, its control can be easily accommodated by the computer.

Telemetry Interface - As can be seen from the discussion in earlier paragraphs, the computer's interface to the telemetry subsystem is one of some importance. Due to the uniqueness of some of the other subsystem interfaces, the computer would have ready access to most of the data available from the subsystem and can readily pass it on to the telemetry subsystem. This could be accomplished via the parallel output bus to the telemetry encoder as shown in figure 4.8.6. For that data needed by the computer which is sampled only by the telemetry system, the computer would access the encoder subsystem interfaces proposed above, the computer already has ready access to most of the data available from the subsystem and could readily pass it on to the telemetry subsystem. This would be accomplished via the parallel output bus to the telemetry encoder as shown in figure 4.8.6. For that data needed by the computer which is sampled only by the telemetry system, the computer would access the encoder directly. By asking for data over the output bus, the encoder would feed the requested data to the computer via the input bus. Since the encoder cannot store data, the computer must receive information from the encoder identifying the data presently available in it.

It is reasonable to expect the computer to perform further services onboard the spacecraft as required. Some of these might take the form of data compression, limit checking, and alarm generation. The effect of these actions will be to reduce the amount of continuous data transmitted to earth. Each subsystem will be examined for applicability of these techniques.

Command Interface - There are certain functions associated with the computer operation which must be initiated from the ground. Some of these functions are shown on figure 4.8.7. It is planned that only one CPU will be actually executing instructions and controlling the spacecraft at a time. An optionally available feature can be one in which both CPU's are simultaneously processing with their results compared as a confidence check of functionality. Either computer can be started at one of the two preselected memory locations as a means of initializing their operation.

A further feature of the command system interface is the ability to reprogram the computers from the ground via the computer load data word mode of the command system. This applies not only to complete program reload but to the loading of command tables or queues to be executed in a preprogrammed manner by the computer. Their execution can be tied to a predetermined time or the occurrence of an event. Whole start up, shutdown, and recovery sequences can be preprogrammed in the computer.

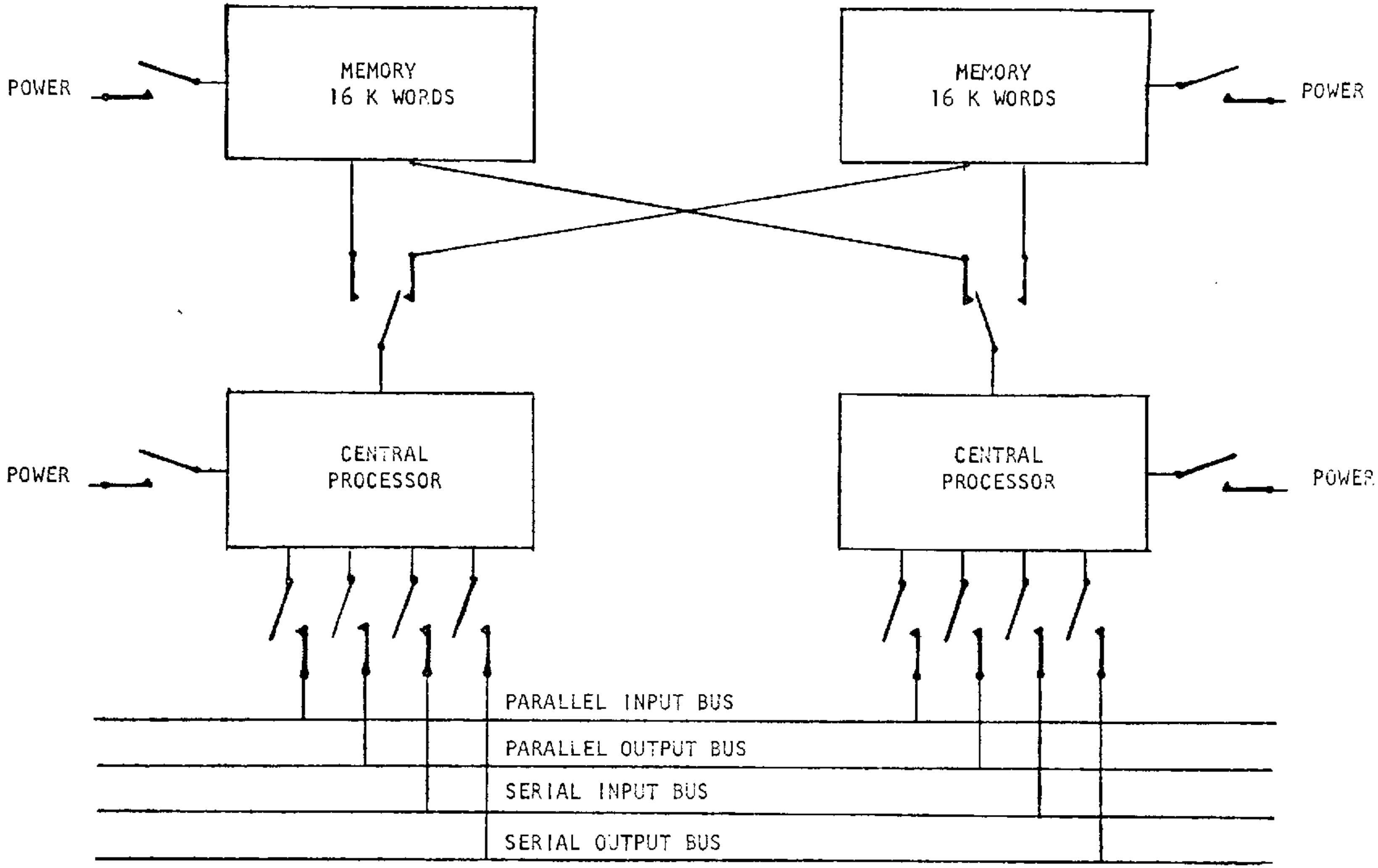
Housekeeping Subsystems Interfaces - There are many standard functions to be performed on any spacecraft which the computer may assist in. This is true on SERT C also. No attempt will be made here to exhaustively pursue this application but the following recognized functions are listed as examples:

Solar Array Drive

It is necessary to maintain the solar array positioned for maximum solar illumination at any time. The computer could receive inputs from the array mounted sun sensors, an optical shaft encoder, and a monitor of maximum available power. These inputs could be used in determining the required correction to be sent to the solar array drive mechanism.

Power System

There will be numerous power switching functions to be performed by the computer, some of which have already been mentioned in previous paragraphs. One in particular is the selection of thrusters for operation. Redundant and auxiliary power systems can be switched by computer control based on either sensed data or prestored ground command.



EACH SWITCH FUNCTION REPRESENTS A COMMANDABLE MODE OF OPERATION

Figure 4.8.1 - REDUNDANT COMPUTER CONFIGURATION

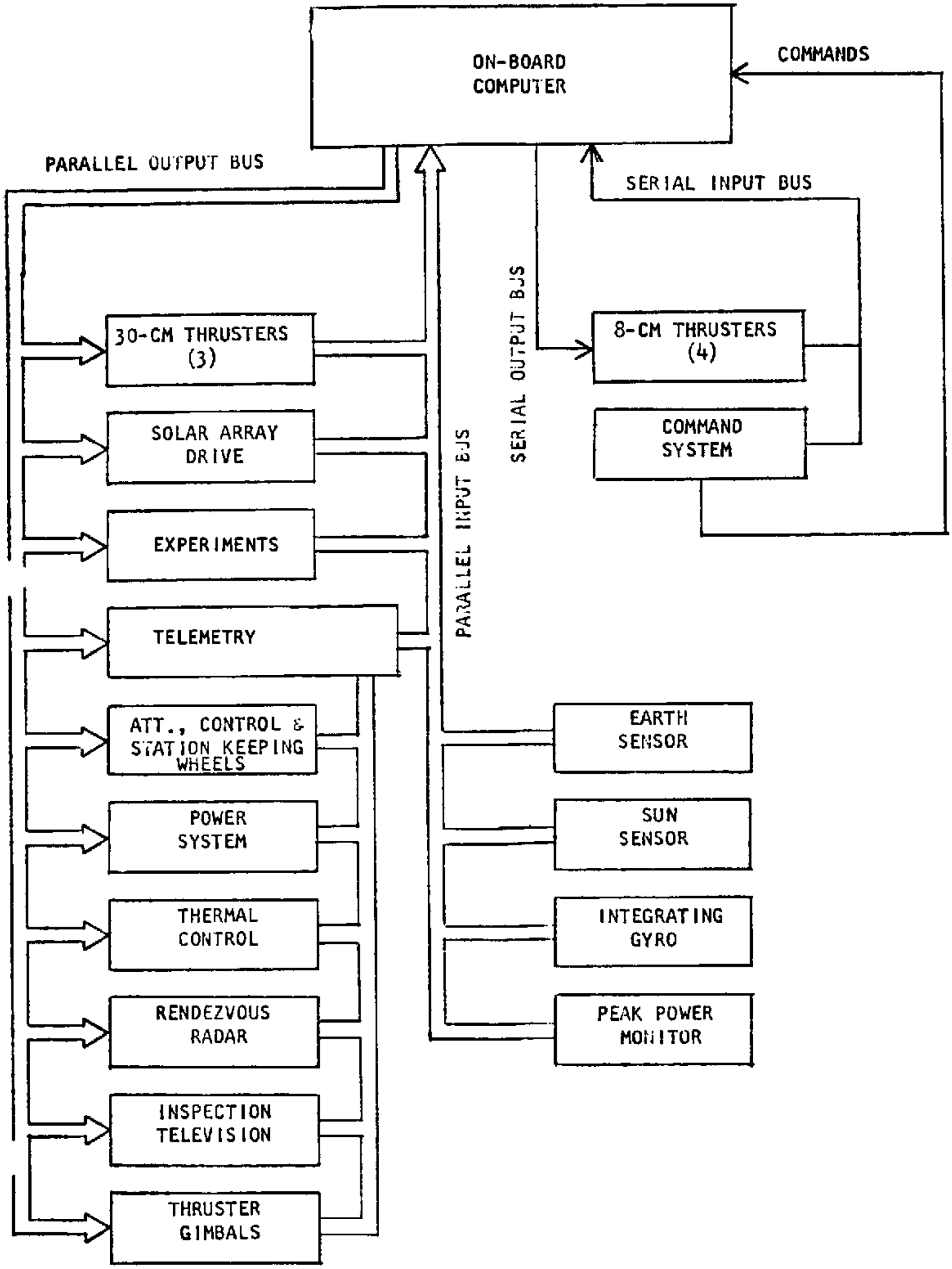


Figure 4.8.2 - GENERAL COMPUTER/SUBSYSTEM BLOCK DIAGRAM

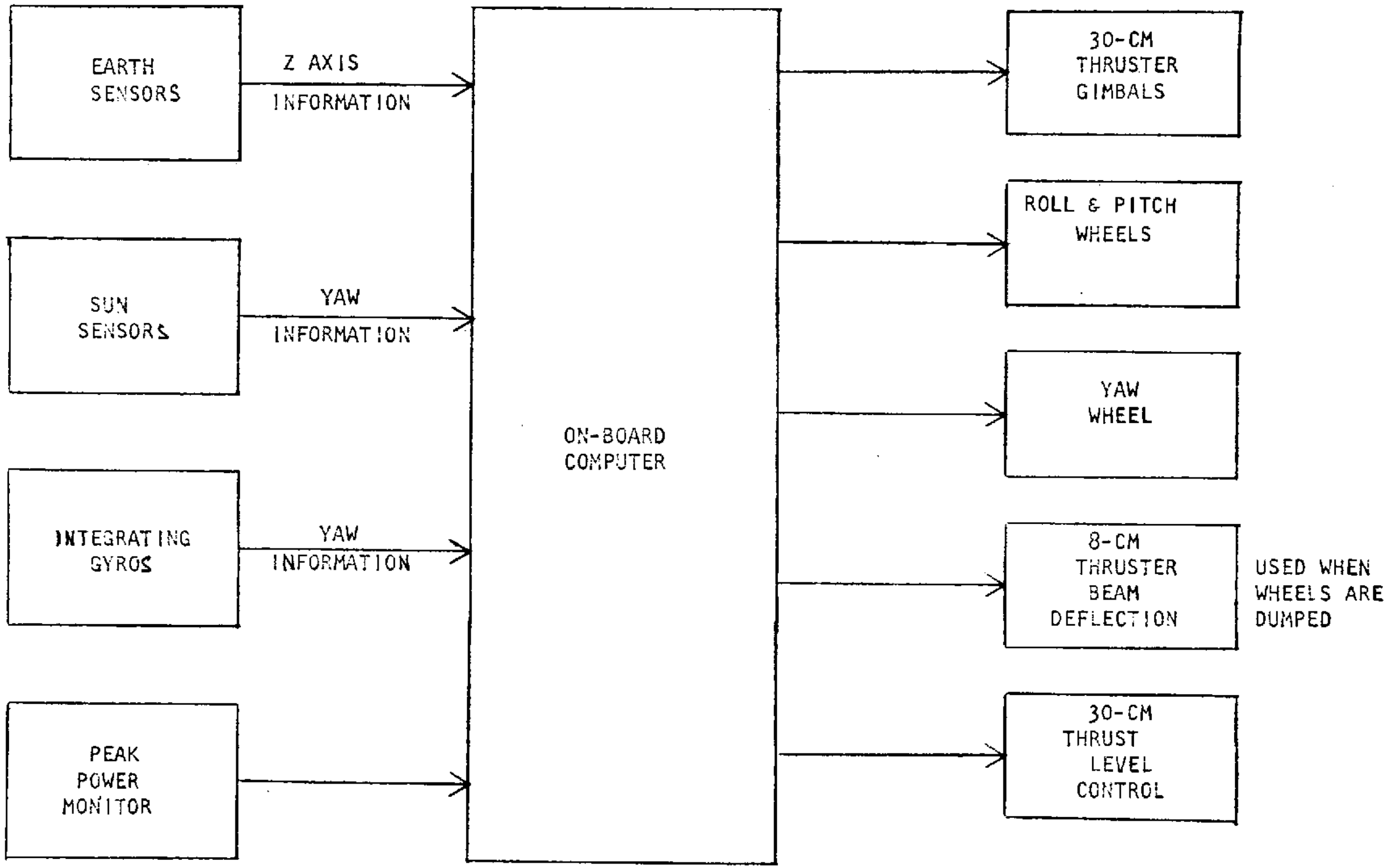


Figure 4.8.3 - COMPUTER CONTROLLED ATTITUDE CONTROL & STATION KEEPING SYSTEM

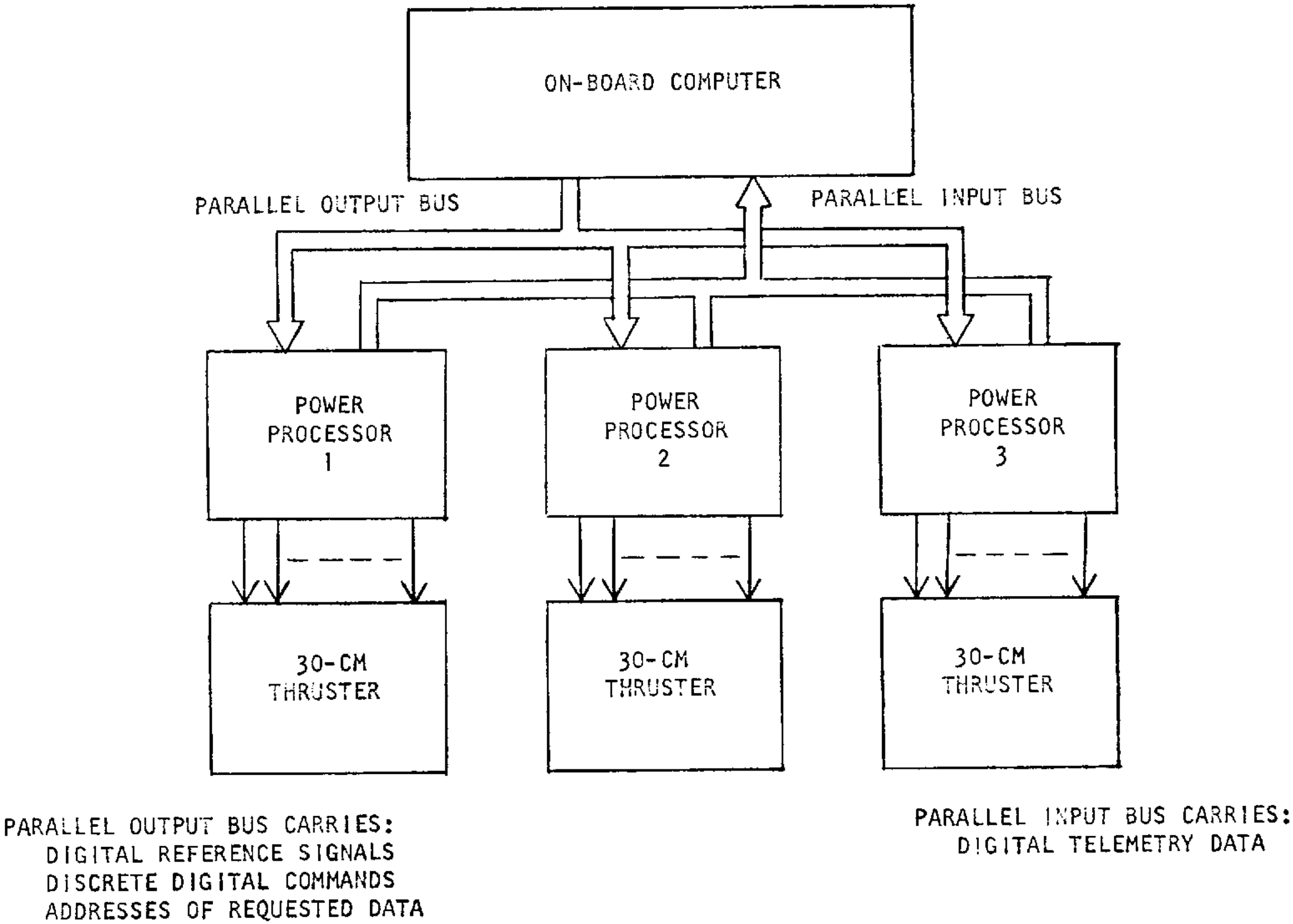
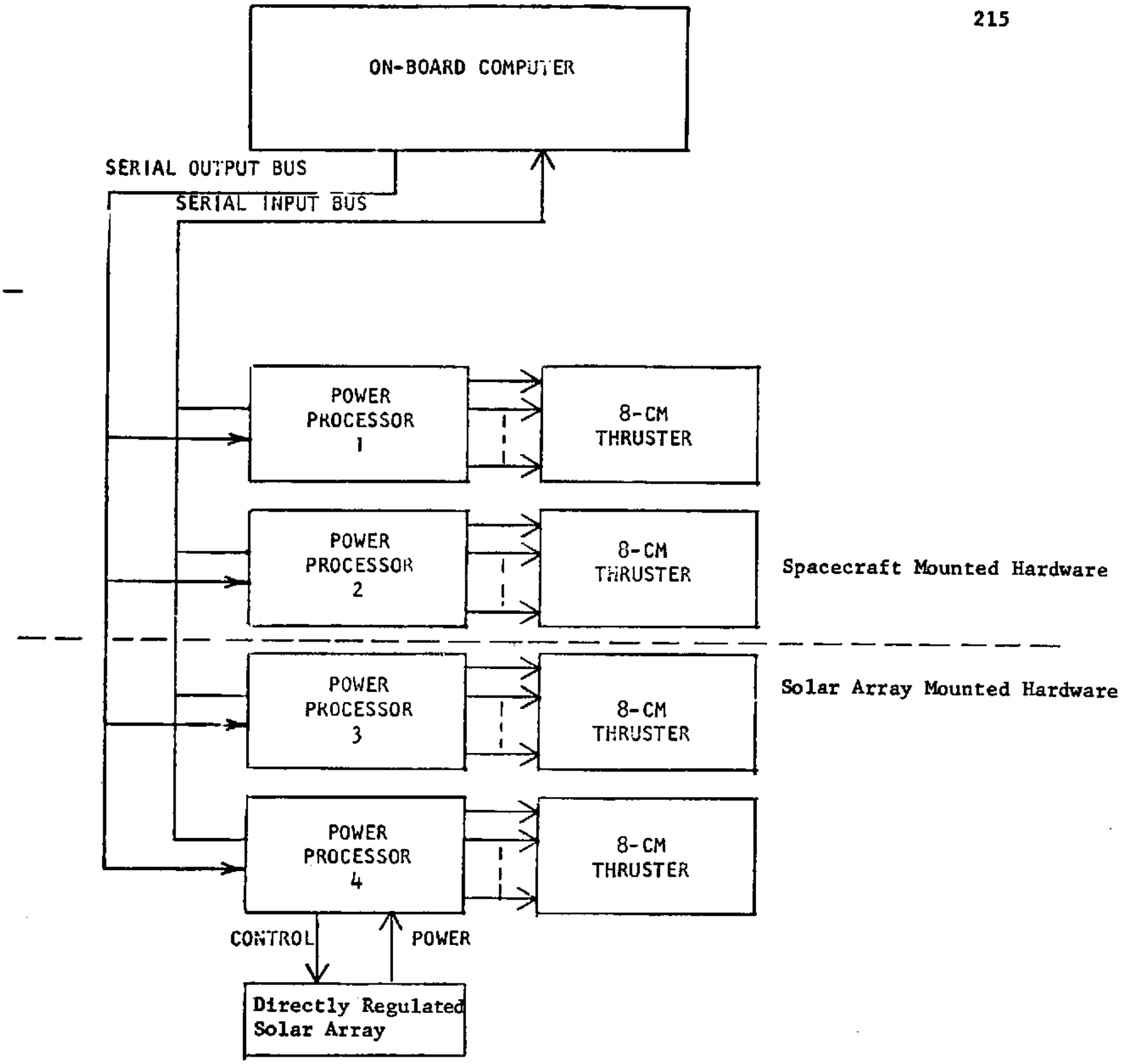


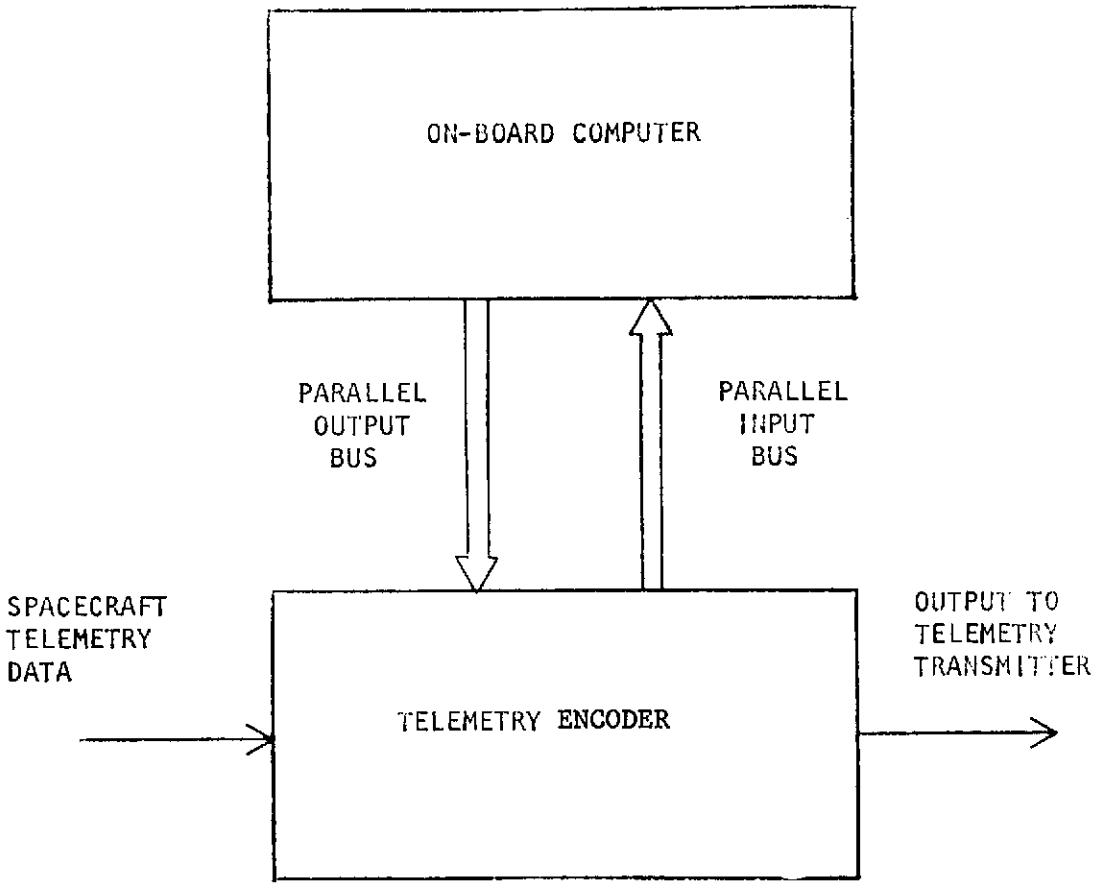
Figure 4.8.4 - COMPUTER/30-CM THRUSTER INTERFACE



SERIAL OUTPUT BUS CARRIES:
- DIGITAL REFERENCE SIGNALS
- DISCRETE DIGITAL COMMANDS
- ADDRESSES OF REQUESTED DATA

SERIAL INPUT BUS CARRIES:
- DIGITAL TELEMETRY DATA

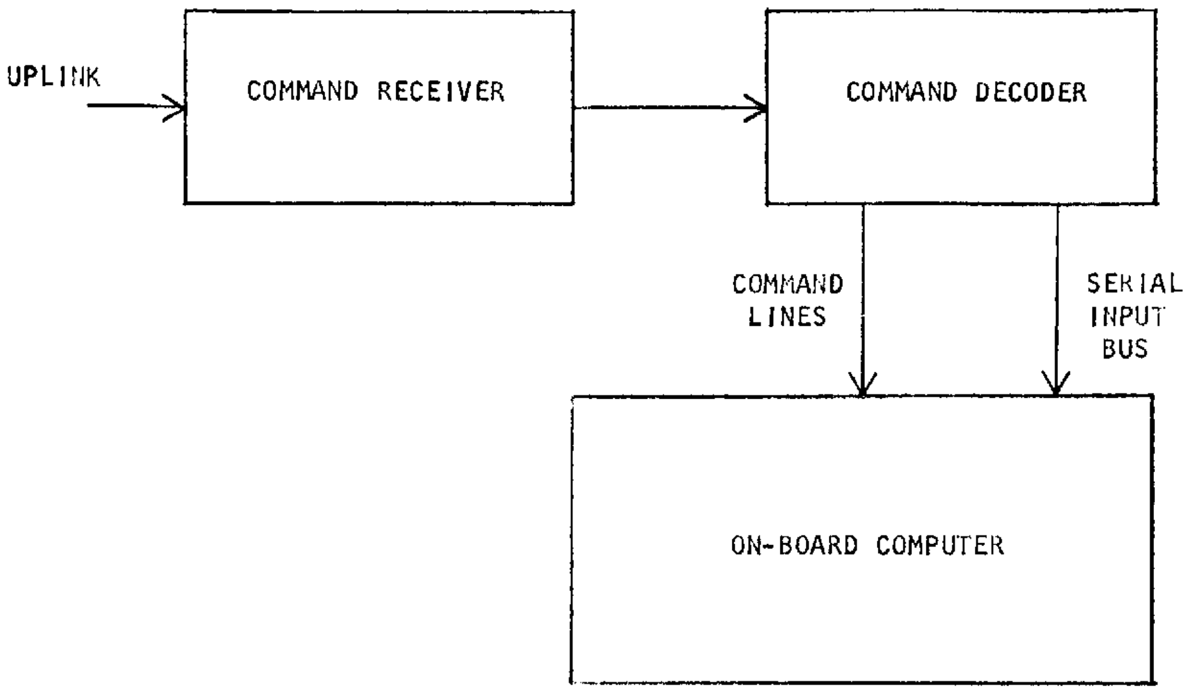
Figure 4.8.5 - COMPUTER/8-CM THRUSTER INTERFACE



PARALLEL OUTPUT BUS CARRIES:
DATA TO BE OUTPUT TO TELEMETRY
REQUESTS FOR SPACECRAFT DATA TO BE READ INTO COMPUTER

PARALLEL INPUT BUS CARRIES:
SPACECRAFT DATA TO BE READ INTO COMPUTER
IDENTIFICATION OF ENCODER DATA

Figure 4.8.6 - COMPUTER/TELEMETRY SUBSYSTEM INTERFACE



COMMANDS TO COMPUTER:
- LOAD MEMORY
- STOP CLOCK
- START AT PRESELECTED ADDRESS
- MASTER CLEAR
- STOP INPUT
- STOP EXECUTE
- CROSS-STRAP SWITCHES

SERIAL INPUT BUS CARRIES:
- INFORMATION TO BE LOADED INTO COMPUTER

Figure 4.8.7 - COMPUTER/COMMAND SUBSYSTEM INTERFACE

4.9 Thermal Control

The thermal design must provide a thermal control system that will keep all components within temperature ranges that will allow safe operation during the approximate one year of spiral-out orbit raising and 5 years of synchronous orbit operation. Due to the difference in the power requirements of the parking orbit, spiral-out phase, synchronous orbit and eclipse operations of the SERT C there will be a wide range in the amount of heat dissipation within the spacecraft and in the solar heating and/or deep space cooling.

Figure 1.3.1 shows the spacecraft in its spiral-out phase configuration. The north and south faces of this configuration (see Section 1.3) are utilized as the principal heat rejection surfaces. During the spiral-out phase these surfaces are subjected to solar incidence angles up to 45 degrees. This value corresponds to launch times which minimize the orbit raising time. Part of both the north and south faces are the radiator surfaces for the 30-cm thrusters' power processors. The north and south panels are dimensioned such that they can radiate the maximum internal heat generated (during spiral-out phase with two 30-cm thrusters operating) in addition to solar heat flux which they absorb. Since during an eclipse (which could be as long as 72 min), there is very little heat generated as compared with two 30-cm thrusters operating, the north and south sides must be partially covered with louvers to minimize the heat rejection to space. A preliminary thermal analysis has indicated that, at most, half of the north and south panels will need to be covered with louvers.

The 30-cm thrusters are thermally isolated from the rest of the spacecraft to minimize their effect on the spacecraft thermal design. These thrusters are protected from adverse space conditions by the use of high and low temperature insulation system, heaters, special coatings and paints.

All other external surfaces of the spacecraft will be covered with vented multilayer insulation blankets. This prevents both undesired solar heat absorption and radiative losses.

4.10 Rendezvous Radar

Microwave Rendezvous Radar Subsystem (MRRS) - In order to provide the capability for rendezvous maneuvers with other synchronous satellites by the SERT C spacecraft, an accurate terminal guidance radar system is desirable. Precise knowledge of the relative position between two spacecraft performing a rendezvous maneuver is difficult and often impractical to obtain from ground based radars, and therefore an accurate radar located onboard the spacecraft may be required. The MRRS that is being evaluated for SERT C is a follow on modification of the existing RCA Apollo I LM X-band Rendezvous Radar system.

In the Apollo application, the Rendezvous Radar was located in the Lunar Module (LM) and a companion transponder system was located in the "target vehicle" Command Service Module (CSM). The system operated as a cooperative mode-active system, where the radar "acquires" its associated Transponder at distances as great as 400 nautical miles and thereafter tracks it cooperatively in range, velocity, and angle.

The updated version presently being developed and proposed is a straightforward evolution of the space qualified LM rendezvous radar sensor, adding circuits to allow noncooperative operation and modernizing some of the technology, permitting both weight and serviceability improvements. MSI and hybrid circuit techniques have been incorporated toward these improvements.

Apollo Radar System Modifications - The modifications required to provide the LM radar the capability to track either a noncooperative or a cooperative target are shown in the system block diagram, Figure 4.10.1. The system characteristics are summarized in Table 4.10.1. Briefly, the modifications shown in the crosshatched area 1 of Figure 4.10.1 consist of adding a high power transmitter, which is selected for tracking a noncooperative target. To accommodate transmitting and receiving on a single antenna at the same frequency, the transmitter is gated at a 40/60 duty cycle and the receiver on a 60/40 duty cycle.

The RF switch, PRF gates, and Paramps are added in the antenna assembly as shown in area 2 of Figure 4.10.1 and the dish increased to 3-ft diameter. The Electric Assembly modifications for the noncooperative mode in crosshatched areas 3 and 4 of Figure 4.10.1 consists of providing: (1) A noncoherent AFC loop shown as a "Freq Disc/Ampl Filter" which parallels the "Bal Mod" that is used in the cooperative mode; and (2) The noncoherent range tone demodulation shown as a "Comb Filter and Freq Discriminator" which parallels the "Buffer Coherent Detector" that is used in the cooperative mode.

Parameter Approaches and Tradeoffs - Various approaches to a rendezvous sensor conducted over a period of several years by RCA included the consideration of pulsed, cw, and high prf pulse doppler types of radar. Frequency agility and pulse compression techniques were also included in the trade-off studies for an cptimized microwave rendezvous sensor. The system approach that has been selected by RCA for noncooperative target detection and tracking as a follow-on for the Space Shuttle/Orbiter Program and that would also be applicable to the SERT C is the high duty cycle pulsed doppler mode of operation.

The parameters for the noncooperative mode can be met with several distinct system variants. A matrix of the various implementation approaches is contained in Table 4.10.2 where the effect on physical parameters -- weight, size, and power consumption can be related. Variables are probability of detection (90 or 99 percent), noise figure, transmitter power, the use of frequency diversity, or smaller aperture antennas.

Radar Transmitter - The transmitter for pulsed doppler operation is a traveling wave amplifier. The power level would be as shown in Table 4.10.2 for the parameter trade-offs chosen. The transmitter would be housed on the gimbal-mounted portion of the antenna assembly to avoid using rotary joints. The amplifier is wide band, supporting the diversity range of the frequency agile transmissions.

Tube development would not be required since available designs exist covering the range of power levels considered in the trade-offs of Table 4.10.2.

Antenna Assembly - The Rendezvous Radar Assembly includes an antenna reflector, antenna feed, monopulse comparator, gimballing elements, and internally mounted components such as gyros, resolvers, LO/coop transmitter frequency source, phase modulator and mixer preamplifiers.

The antenna is a four-horn amplitude comparison monopulse type. A Cassegrainian configuration is utilized to minimize the total depth and to allow the radiating horns to protrude through the dish. Circular polarization is used to minimize the signal variations resulting from attitude changes of the linearly polarized transponder antenna. Components are distributed on the antenna to achieve balance about each axis.

Rate-integrating gyros are used for line-of-sight space stabilization and line-of-sight angle rate measurement. These are located in the lower section of the trunnion axis to act as a counterweight. A two-speed resolver is mounted on each axis for high accuracy angle-data pickoff for the computer and for displays.

Radar Receiver - The receiver is a highly stable three-channel, triple-conversion superhetrodyne. Two channels are used for amplifying the shaft and trunnion axis error signals and one channel is provided to amplify the sum or reference signal. The receiver also includes phase sensitive detectors for generating angle error signals, an Automatic Gain Control (AGC) circuit for controlling the gain of the three receiver channels, an IF distribution amplifier unit for supplying reference channel signal to range and frequency trackers, and a local oscillator mixer for generating the second local oscillator signal.

Physical Characteristics - As indicated previously, the rendezvous radar system will consist basically of an antenna assembly and an electronics assembly. There are a series of trade-offs involved, and the resulting impact of different detection probabilities, noise figures, transmitter power, the use of frequency diversity, and smaller aperture antennas that are tabulated in Table 4.10.2 to show the effect on system weight and power. Of the two system noise figures shown, the 3 dB figure can be obtained today by the use of paramps, and the 4 dB figure is predicted for the field effects transistor (FET) in the X-band (9800 MHz) frequency region within two years.

It should also be noted that the RF power requirements differ for 90 percent and 99 percent probability of detection by only 1 to 2 dB, and therefore a radar with a 90 percent probability of detection at 30 n.m. will achieve a 99 percent probability prior to reaching a range of 25 n.m. Table 4.10.2 also shows a comparison of 90 percent and 99 percent probability, with and without frequency diversity, and three (3) and two (2) foot diameter antennas, with the resultant overall system weight and system input power requirements.

Radar Interfaces

A. Input Power

1. 28 V dc
2. Gyro excitation 115 V 400 Hz
3. Resolver excitation 28 V 800 Hz
4. Antenna assembly designate signal - two axis sine/cosine resolver

B. Outputs

Outputs	Format
Range	Digital, 15 bit serial
Range Rate	Digital PRF
Angle	One and sixteen speed sine and cosine, resolver outputs at 800 Hz
Angle Rate	Analog, DC voltage

Environmental - There are no environmental limitations on the proposed equipments. The system design as presented by RCA for the Shuttle Orbiter Program, and as proposed for SERT C, is based on the Apollo Rendezvous Radar system that was designed for deep space environment; i.e., Saturn 5 Launch and boost vibration, lunar temperature extremes of +270°F and -300°F, and deep space pressures. The antenna is designed for radiation cooling with no active cooling; the electronics assembly is designed for cold rail mounting and cooling.

<u>Summary</u> - The proposed radar system does not require any development items. Available TWT designs exist. Twenty-eight (28) systems were built, tested and qualified for the Apollo requirements and environment. Thus many of the proposed modifications to the system for the Shuttle Orbiter are for updating and improving the system weight, power requirements, and performance.

TABLE 4.10.1 - SUMMARY OF MICROWAVE RENDEZVOUS RADAR CHARACTERISTICS

Characteristic	Noncooperative long range
Frequency	X-band
Radar type	High PRF pulse doppler
Maximum range	30-mi on 1 sq meter w/o frequency diversity
Minimum range	100 ft
Method of ranging	Phase comparison of transmitted and received tone modulation
Duty cycle	40/60
$P_{transmit}$	See Table 4.10.2
RF source	Solid state source driving traveling wave amplifier
Antenna data	
Antenna reflector	3-ft paraboloid
Antenna feed	4-horn amplitude monopulse
Antenna subreflector	Cassegrainian
Antenna gain	37 dB, min
Antenna beam width	2.3 deg
Coverage	$\pm 90^{\circ}$ each axis
Angle data	
Angular accuracy	Bias = 8 mr (per axis) Random (3 σ) = 4 mr $5 < R < 400$ n mi 10 mr at $R < 2$ n mi
Angular rate	Maintain track for base input rates of 10° per sec about each axis simultaneously
Range data	
Range accuracy	Bias = 0.1% at $R > 50.8$ n mi ±120 ft at $R < 50.8$ n mi. Random (3 σ) = 0.25% at $R > 5$ n mi
Range rate	±4900 ft/sec
Weight	See Table 4.10.2
Power	See Table 4.10.2

TABLE 4.10.2 - SYSTEM PARAMETER TRADE-OFFS

Radar config.	Target size, m^2	Range, n mi	AA diam, ft	NF, dB	Frequency diversity	Prob. of det., percent	RF power	System weight, lb	System power, W
1	15	30	3	3	No	99	5.4	78.5	125
2			3	3	No	90	4.1	78.5	120
3			3	3	Yes	99	1.0	80.0	110
4			3	3	Yes	90	0.6	80.0	110
5			3	4	No	99	6.1	77.0	120
6			3	4	No	90	4.6	77.0	123
7			3	4	Yes	99	1.1	78.5	111
8			3	4	Yes	90	0.67	78.5	108
9			2	3	No	99	27.0	79.5	168
10			2	3	No	90	20.5	79.5	150
11			2	3	Yes	99	5.0	77.0	130
12			2	3	Yes	90	3.0	77.0	120
13			2	4	No	99	34.0	78.0	185
14			2	4	No	90	26	78.0	165
15			2	4	Yes	99	6.3	75.5	126
16			2	4	Yes	90	3.8	75.5	123

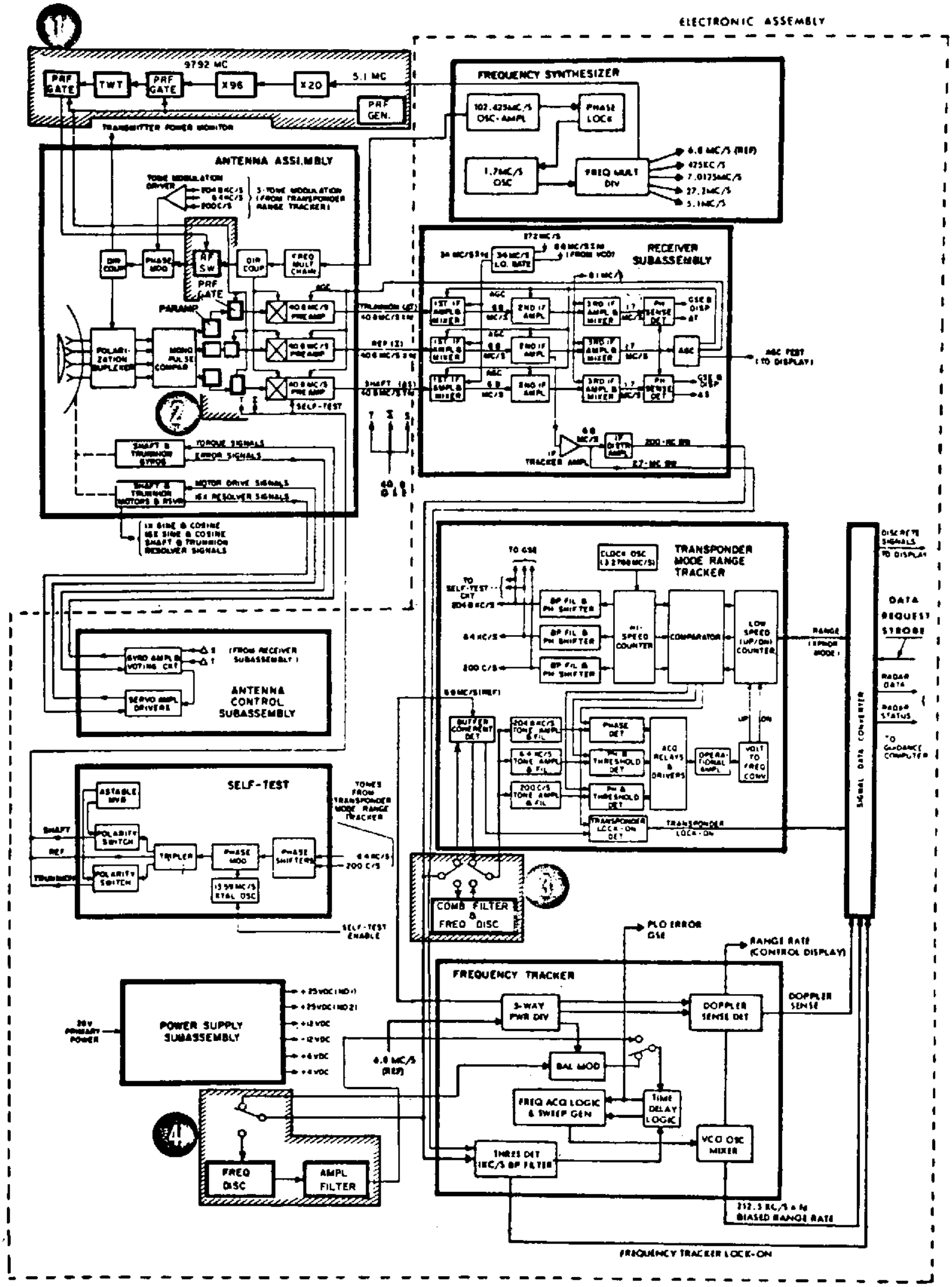


Figure 4.10.1 - Rendezvous Radar Block Diagram

4.11 Inspection Television

Ground Commanded Television Assembly (GCTA) - The GCTA is a color television system designed (by RCA, Astro Electronics Division) to operate on the lunar surface. The camera system can be operated manually (by astronaut) or by remote commands from earth. The assembly consists of a color television camera and a television control unit. The television control unit permits ground-commanded positioning and operation of the camera. The GCTA is illustrated in Figure 4.11.1. A dimensional view of the GCTA configuration mounted as on the lunar rover vehicle is shown in Figure 4.11.2.

Color Television Camera (CTV) - The CTV is a small, lightweight unit producing high quality, field sequential color television at standard line and frame rates. A simple conversion to black and white television operation can be accomplished by removing the rotating filter wheel which presently generates the field sequential color video. The camera uses a single silicon intensifier target (SIT) tube. A zoom lens is incorporated in the CTV with provisions for manual and remote control of zoom and iris settings. Automatic light control (ALC) operating on average or peak scene luminescence also is incorporated. ALC may be selected manually or by remote control. The television camera is approximately 18 inches long, 6.5 inches wide, 4 inches high and weighs 12.8 pounds.

The lens assembly houses a f/2.2 Angenieux lens with a zoom ratio of 6:1. Both zoom and iris may be manually or remotely controlled by electric drives in the lens assembly.

The camera body contains the SIT tube, color wheel assembly (to be removed for B&W operation) and all synchronization, deflection and video components required to provide a standard 525-line composite video format at the camera output. The SIT tube has a high sensitivity and a relative immunity to image burn. With the exception of the lens opening and a thermal radiator covering the top of the camera, the entire camera is covered with thermal insulation blanketing. The thermal radiator is constructed of second-surface mirrors.

The performance summary of the television camera is given in Table 4.11.1.

Television Control Unit (TCU) - The TCU provides an azimuth/ elevation mount for the television camera. The TCU receives a command subcarrier (70 KHz) from the spacecraft receiver (probably the command receiver on SERT C), and executes commands for: azimuth and elevation movement of the camera cradle; camera lens zoom, iris, automatic light control and power functions. The TCU accepts the video signal, adds a test signal (to check the system bandwidth and linearity), and routes the combined video to the FM transmitter for transmission to earth.

The TCU azimuth/elevation pedestal allows the television camera to pan 214 degrees to the right and 134 degrees to the left of forward and to tilt -45 to +85 degrees from horizontal. Geared stepping-motor drives are used. The power requirements of the TCU are as follows:

TCU	5 watts @ 29 ±4 VDC
Motors	2 watts/motor (intermittent)

Four motors are incorporated into the TV system and may be used simultaneously.

The TCU electronics box receives a 70 kilohertz subcarrier from the command receiver and decodes 18 real-time commands for execution (Table 4.11.2). Motor drivers for control of camera azimuth, elevation, zoom and iris; and relay drivers for camera power and automatic light control are also in the TCU. Camera video is routed to the TCU where a vertical interval test signal is generated and added prior to transmission to earth by the television transmitter.

The TCU is partially covered with thermal insulation blanketing. The unblanketed surfaces are covered by second-surface mirrors or white and black paint, as illustrated in Figure 4.11.3.

<u>Ground Command Format</u> - The uplink command signal from the command receiver to the TCU is a 70-KHz subcarrier frequency modulated by

a composite command signal. The composite command signal consists of the linear sum of a 2-KHz message subcarrier and a 1-KHz coherent synchronization tone. The message subcarrier and the synchronization tone have equal carrier frequency deviation. Total peak deviation is ±5-KHz.

Digital command information is carried by the 2-KHz message subcarrier and is phase-shift keyed (PSK) at a rate exactly one-half the rate of the subcarrier frequency. The message subcarrier is modulated by a train of binary "ones" and "zeros" (referred to as sub-bits) at a rate of 1000 per second. The train is in two-level, non-return-to-zero (NRZ) binary code.

A sub-bit "1" is present when the 1-KHz synchronization tone and the 2-KHZ PSK tone are in phase at the start of a sub-bit interval. When the 2-KHz tone is 180 degrees out of phase with the 1-KHz tone at the start of the sub-bit interval, a sub-bit "0" is present.

Real-time command message structure consists of three vehicle address info-bits (using vehicle address sub-bit codes) followed by three system address info-bits and six data info-bits (using system/data sub-bit codes). The sequence is as follows:

	VEHICLE ADDRESS	SYSTEM ADDRESS	FUNCTION CODE
INFORMATION BIT NO.	1 2 3	4 5 6	7 8 9 10 11 12

The vehicle address (information bits 1, 2 and 3) is fixed as binary 0, 1, 1 (Octal 3). The system address (information bits 4, 5, and 6) is fixed as binary 0, 1, 0 (Octal 2).

The GCTA commands and their codes are given in Tables 4.11.2 and 4.11.3.

• FM Transmitter (TV) - Preliminary downlink calculations indicate that approximately 15 watts of RF (@ 2 GHz) will be required for satisfactory reception of the downlink television signal. This power requirement assumes the following:

(a) a 100°K receiver (ground)

(b) a 15 ft. dish (ground)

(c) a 2 ft. dish (spacecraft)

A spacecraft transmitter modulation index of 2 is planned for the FM transmitter. The transmitter video input signal bandwidth is assumed to be 2.5 MHz, which is the bandwidth utilized by the Apollo TV system.

A candidate transmitter is the unit used on the Apollo project. The following are some of its characteristics:

WEIGHT	3 lbs
SIZE	7" x 7" x 2"
FREQUENCY	2.2 GHz

All solid state components

The unit can be upgraded to 15 watts (125 watts input power) without any problems.

Antenna (Spacecraft) - It is anticipated that the antenna used to transmit the spacecraft telemetry can be utilized for transmission of the television signal. To accomplish this, a diplexer will be necessary to combine the TV and TLM transmitter powers into a single antenna feed. A 2 ft. diameter spacecraft antenna is presently anticipated.

Downlink Calculation (2 GHz) - A downlink calculation is presented on Table 4.11.4. A transmitter output power of 15 watts at 2 GHz produces a very good quality television picture provided that:

(a) a 2 foot spacecraft dish is utilized

(b) a cooled parametric amplifier is used at the ground station (100°K)

(c) a 15 ft. dish ground antenna is used (in conjunction with (b))

(d) a modulation index of 2 is used on the spacecraft transmitter

(e) video baseband bandwidth is 2.5 MHz

(Note: It is presently assumed that the 2 GHz telemetry frequency will be available for this purpose.)

GCTA Performance - The typical television field-of-view and resolution for terminal rendezvous, initial inspection, and final inspection of a target spacecraft from the proposed GCTA is shown in Figure 4.11.4

TABLE 4.11.1 - CTV PERFORMANCE SUMMARY

PARAMETER	CHARACTERISTIC
Sensor	SIT camera tube
Sensitivity	Greater than 32 dB signal-to-noise at 3 foot-lamberts
Resolution	80 percent response at 200 TV lines
ALC Dynamic Range	1000 to 1 (minimum)
Non-linearity	3 percent (maximum)
Gray Scale	10 -$\sqrt{2}$ steps
Video Output Level	1.0 volts p-p into 75 ohms Full EIA sync
ALC	Peak or average detection modes
Optics	
Zoom Ratio	6:1
Iris Control	f/2.2 to f/22
Pan Angle	$\left.\begin{matrix}+214\\-134\end{matrix}\right\}$ degrees from front
Tilt Angle	$\left.\begin{matrix}+85\\-45\end{matrix}\right\}$ degrees from horizontal
Power	14.8 watts @ 28 volts input
Weight	12.8 pounds

TABLE 4.11.2 - GCTA REMOTE CONTROL (GROUND) COMMANDS

COMMAND	FUNCTION
PAN RIGHT	Pan right at 3.03 degrees per second.
PAN LEFT	Pan left at 3.03 degrees per second.
PAN STOP	Stop pan.
TILT UP	Tilt up at 3.12 degrees per second.
TILT DOWN	Tilt down at 3.12 degrees per second.
TILT STOP	Stop tilt.
ZOOM IN	Increase focal length. Full range covered in 14.6 seconds maximum.
ZOOM OUT	Decrease focal length.
ZOOM STOP	Hold existing focal length.
IRIS OPEN	Decrease f/number. Full range covered in 13.1 seconds maximum.
IRIS CLOSE	Increase f/number.
IRIS STOP	Hold existing f/number.
ALC PEAK	Change CTV circuit connections to provide peak level light control.
ALC AVERAGE	Change CTV circuit connections to provide average level light control.
POWER ON	Power camera, enable LCRU FM Transmitter, commands S/C ON.
POWER OFF	Turn camera OFF, disable LCRU FM Transmitter when in TV mode.
S/C ON	Enable LCRU voice subcarrier (S/C).
S/C OFF	Disable LCRU voice subcarrier (S/C).

TABLE 4.11.3 - GCTA COMMAND FUNCTION CODES

COMMAND FUNCTION	OCTAL CODE Vehicle Address	System Address	Function	BINARY CODE (Bits 7-12)
PAN RIGHT	3	2	04	000100
PAN LEFT	3	2	01	000001
PAN STOP	3	2	02	000010
TILT UP	3	2	14	001100
TILT DOWN	3	2	11	001001
TILT STOP	3	2	12	001010
ZOOM OUT	3	2	24	010100
ZOOM IN	3	2	21	010001
ZOOM STOP	3	2	22	010010
IRIS OPEN	3	2	34	011100
IRIS CLOSE	3	2	31	011001
IRIS STOP	3	2	32	011010
ALC PEAK	3	2	44	100100
ALC AVERAGE	3	2	41	100001
POWER ON	3	2	54	101100
POWER OFF	3	2	52	101010
S/C ON	3	2	64	110100
S/C OFF	3	2	62	110010

TABLE 4.11.4 - 2 GHz DOWNLINK CALCULATIONS

Transmitter Power	+ 12	dBW (15W)
Transmitter RF Loss	- 0.5	dB
S/C Antenna Gain (2 ft., 0 = 17°)	+ 21	dB
Tr. Ant. Point Error	0	dB
Propagation Loss	- 191	dB
Pointing Loss to 37°N. Latitude	- 0.4	dB
Atmospheric Atten.	0	dB
Receiver Ant. Gain	+ 37	dB
Receiver RF Loss	- 0.5	dB
RECEIVED CARRIER POWER	- 122.4	dBW
K	- 228.6	dBW/°KHz
Receiver Temperature (100°K)	20	dB °K
Bandwidth (15 MHz)*	71.8	dB Hz
RECEIVED NOISE POWER	- 136.8	dBW
Carrier/Noise	14.4	dB
FM Improvement (m = 2)	21	dB
Pre-emphasis Improvement	3	dB
Noise Weighting (CCIR)	10.2	dB
S/N, PD-PK, WEIGHTED	48.6	dB

Very good quality TV picture

*BW_{BB} = 2.5 MHz (baseband of GCTA)

BW_{FM} = 2 fm (m + 1) = 2 x 2.5 (2 + 1) = 15 MHz

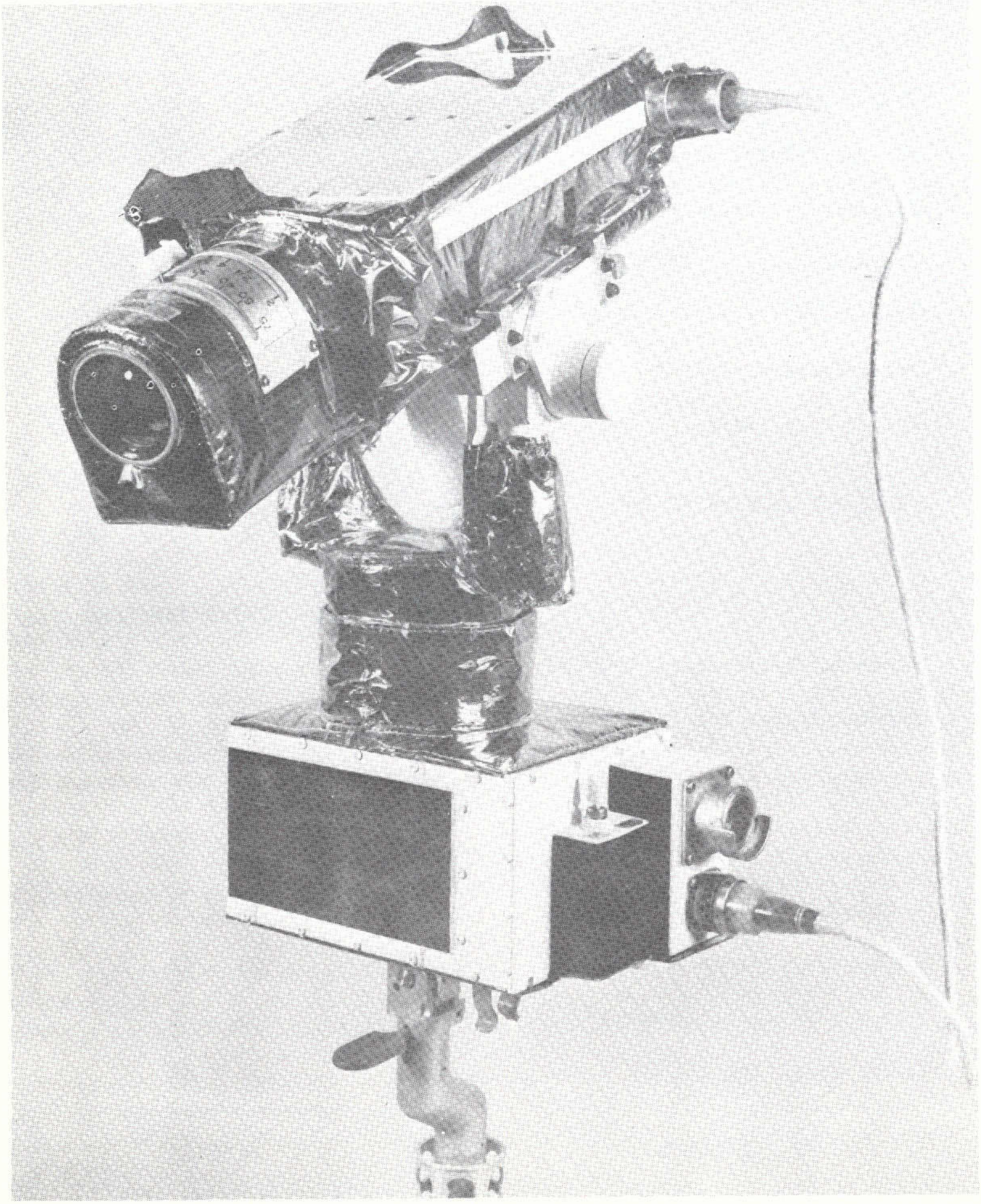


Figure 4.11.1 - RCA Ground - Commanded Television Assembly

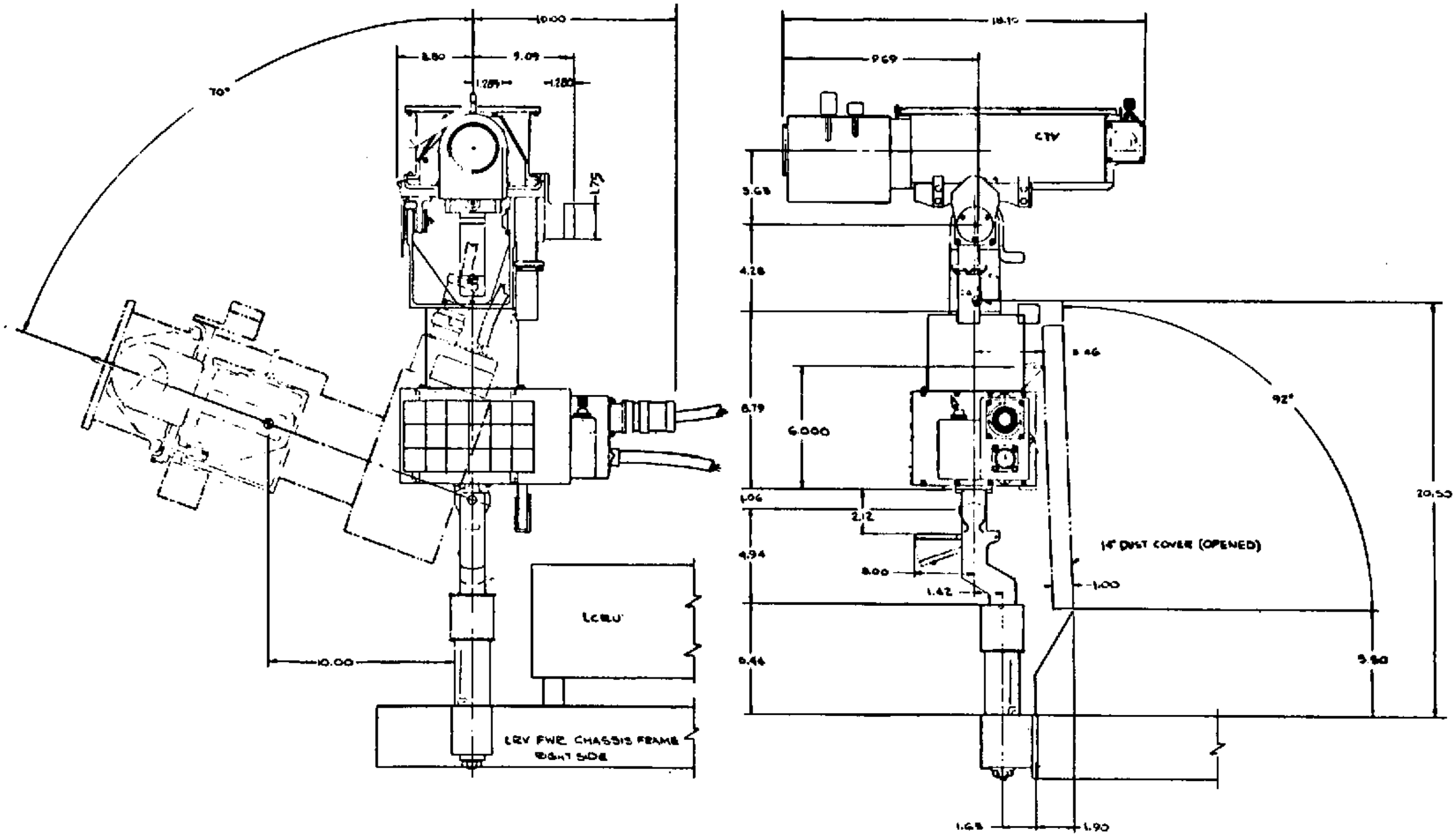


Figure 4.11.2 - GCTA Configuration Mounted on the LRV

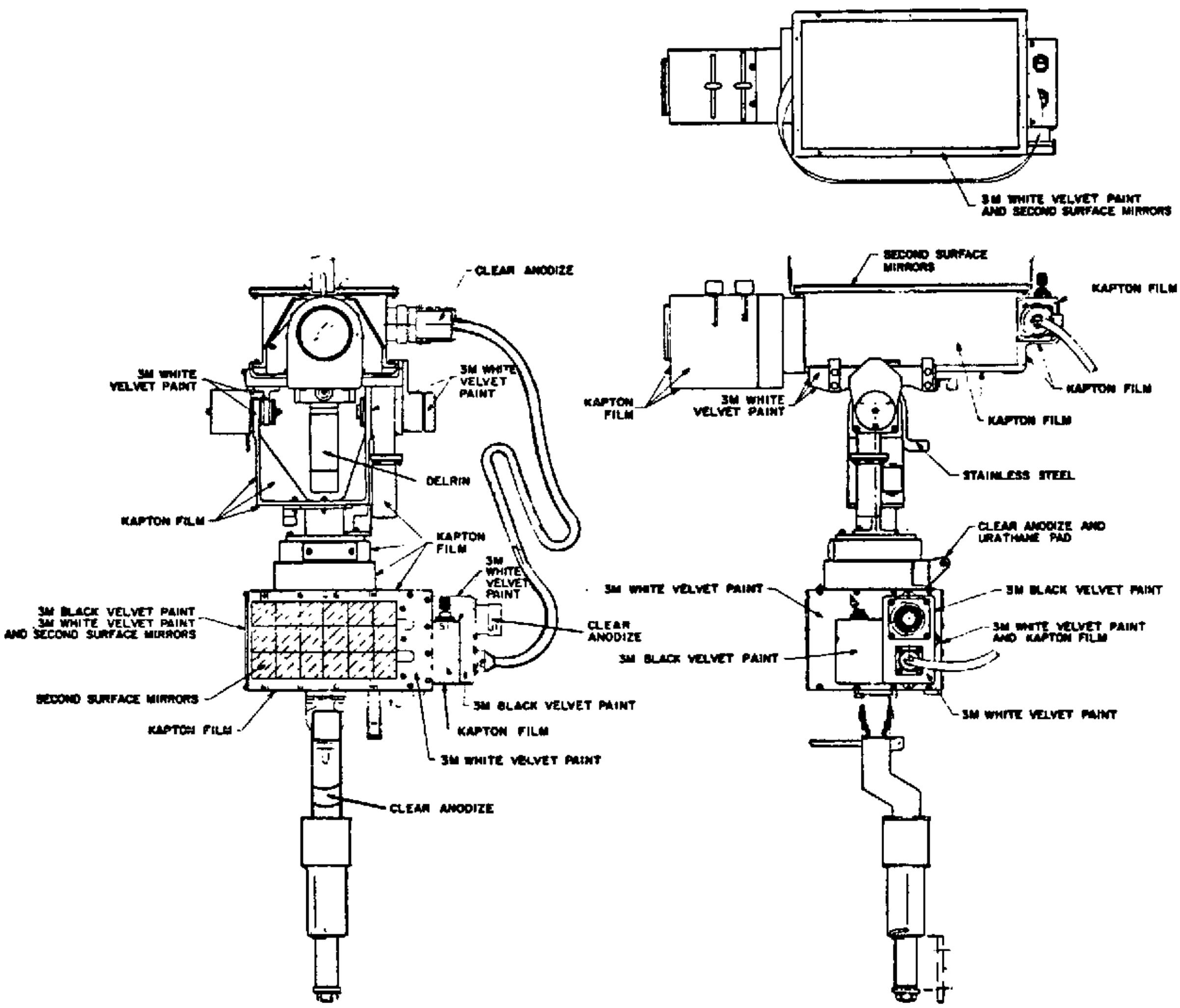


Figure 4.11.3 - GCTA Thermal Control

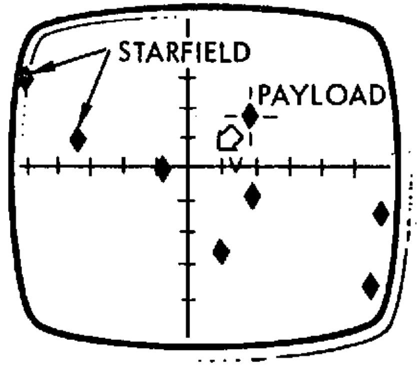
TELEVISION FIELD-OF-VIEW AND RESOLUTION

SUBSYSTEM CHARACTERISTICS

• TYPE:	RCA QTV-9 60 FRAMES/SEC BLACK & WHITE SPACE QUAL
• FOV ANGLE:	14.5 DEG
• FOCAL LENGTH:	50 MM
• MOUNT:	2-AXIS GIMBAL
• QTY:	1 CAMERA
• ILLUMINATION:	SUN

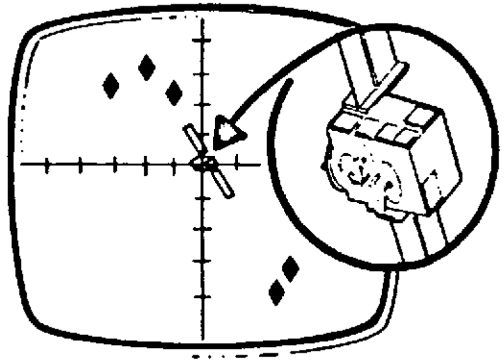
TERMINAL RENDEZVOUS

RANGE: 10,000 FT
RESOLUTION: POINT SOURCE



INITIAL INSPECTION

RANGE: 1,000 FT, FOV: 250 FT
RESOLUTION: 1 FT



FINAL INSPECTION

RANGE: 200 FT, FOV: 50 FT
RESOLUTION: 3 INCHES

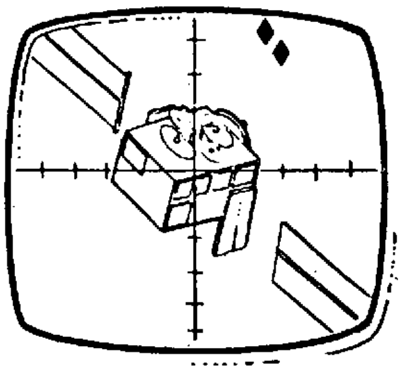


Figure 4.11.4 - Television Field-of-View and Resolution

4.12 Experiments

Two experiments in addition to the experiments described in section 4.3 and 4.6 have been proposed for the baseline configuration.

The first experiment is a colloid thruster which has been proposed by the Air Force Rocket Propulsion Laboratory, Edwards, California. This thruster has approximately the same thrust level of the 8-cm thruster proposed for SERT C. The colloid thruster is described in Appendix A.

A second experiment has been proposed by the Space Sciences Laboratory of the University of California, Berkeley. This experiment consists of a telescope which focuses radiation onto a detector. The purpose of the experiment is to investigate the possibility of galactic sources of extreme ultraviolet radiation. The letter of proposal is contained in Appendix B.

The third experiment is a nickel-hydrogen battery which will provide power to an 8-cm ion thruster. The nickel-hydrogen battery offers significant improvements in both energy density and cyclic life expectancy. This experiment proposal has been submitted by COMSAT Laboratories and is described in Appendix C.

5.0 PROGRAM PLANNING

5.1 Introduction

The LeRC has the unique experience and capability to design and engineer electric propulsion systems which employ the mercury bombardment ion thruster. Two prior flights, SERT I and II, have demonstrated the LeRC's ability to plan and implement low cost, value engineered spaceflight thruster projects. Additional expertise has been developed in the management of the SPHINX and CTS projects and support of the SEPS program.

Support facilities, uniquely designed for ion thruster system optimization and performance testing, coupled with a large, high-vacuum chamber dedicated to spacecraft thermal vacuum testing, facilities to dynamically test spacecraft structures and a ground station with a 15' diam dished antenna for spacecraft communications give the LeRC the in-house capability to fully support spaceflight efforts at whatever level is required to round out the program while remaining within planned budgetary limits.

Internal SRT programs in ion thruster, solar cell, battery, LMSR, and HVSA technology will support the subsystem component development and supply experimental devices to this project.

Support for the SERT C project will be drawn as needed from the Space Flight Programs Directorate with the cooperation of the Space Technology and Materials Directorate. Ancillary support will be provided by Engineering Design and Drafting groups, machine shops, R&QA, Procurement and Technical Services groups as required. The full resources of the LeRC can be utilized to bring this project to a successful conclusion.

5.2 Facilities

LeRC is equipped to perform environmental tests to simulate launch and space environments on spacecraft, subsystems and separate components.

The spacecraft test equipment includes a 28,000 pound electrodynamic system for sine and random vibration in three axes (5 to 2000 Hz) and an RF shielded enclosure (12' x 20' x 8' high) for electromagnetic tests.

For subsystems and components, a variety of additional equipment is available for simulating space environments. This equipment is listed below with some general parameters.

1. Vibrators - 1500, 3000, and 6000 force pounds
2. Shock machine - 150# at 400 g; 50# at 10,000 g
3. Accelerator (centrifuge) - 200# at 100 g; - 65°C to 85°C at 1 torr vacuum
4. Solar simulators - one and two xenon arc lamps
5. Corona tester ± 40 kV
6. Temperature and humidity chamber - 3' x 3' square, -100°F to +500°F, 20% to 95% R.H.

7. Gas analyzer - Quadrupole 1 - 500 AMU

8. Hydraulic/pneumatic pressure - to 3600 psi

9. Vacuum over 1 x 10^{-6} torr at 800°C; (65° - 800°C)

10. Thermo vacuum

    a. 18" diam x 28" high - 185°C to 175°C; 1 x 10^{-8} torr

    b. 28" diam x 48" long - 1 x 10^{-10} torr - 185°C to 175°C

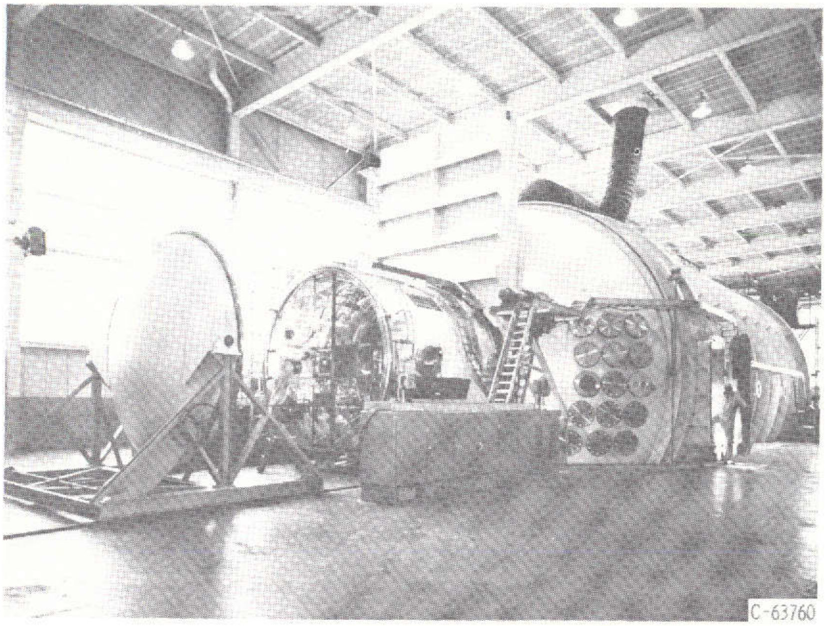
    c. 40" diam x 54" long - 760 torr to 9.6 x 10^{-4} torr in 100 sec; vibration (4000#); -175°C to 175°C (launch and ascent simulation simultaneously)

    d. 5' diam x 6' long - 300°F to 250°F; 1 x 10^{-8} torr.

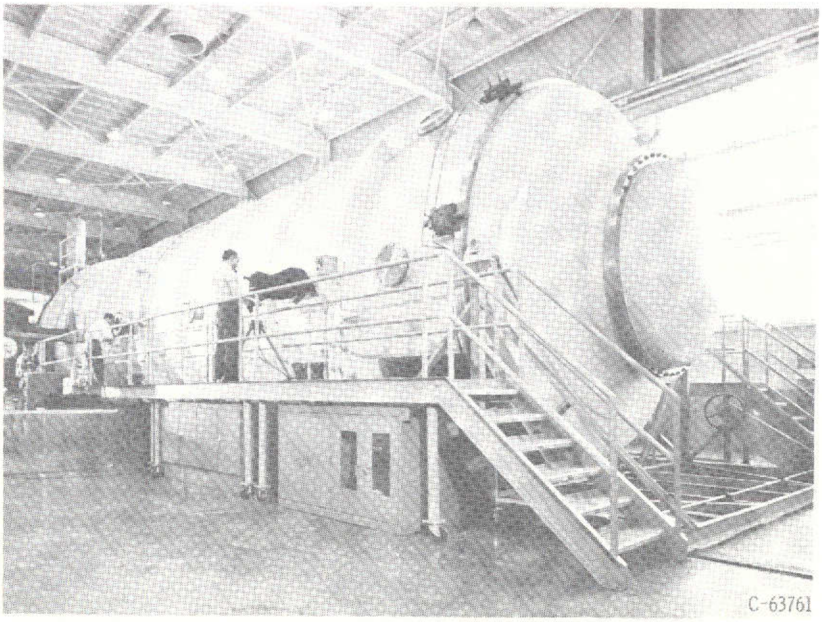
Table 5.2.1 lists representative LeRC vacuum test facilities. The two largest vacuum chambers, unique in their size and capacity, are described in Appendix D. The chambers are shown in figure 5.2.1.

TABLE 5.2.1 - LeRC ION THRUSTER TEST FACILITIES

TANK #1	- 5' DIA, 16' LONG, 3 ACTIVE PORTS, 4 - 32" DIA DIFFUSION PUMPS, LN_2 LINER
TANK #2	- $3\frac{1}{2} \times 7'$, 2 ACTIVE PORTS, 2 - 32" DIA DIFFUSION PUMPS
TANK #3	- 5' DIA, 16' LONG, 3 ACTIVE PORTS, 4 - 32" DIA DIFFUSION PUMPS, LN_2 LINER
TANK #4	- 5' DIA, 16' LONG, 3 ACTIVE PORTS, , 4 - 32" DIA DIFFUSION PUMPS, LN_2 LINER
TANK #5	- 15' DIA, 70' LONG, 3 ACTIVE PORTS, 20 - 32" DIA DIFFUSION PUMPS, LN_2 LINER
TANK #5N	- 5' DIA, 8' LONG, VERTICAL, ONE PORT, 1 - 14", 1 - 10" DIFFUSION PUMP, LN_2 LINER
TANK #6	- 25' DIA, 75' LONG, 6 ACTIVE PORTS, 22 - 32" DIFFUSION PUMPS, LN_2 LINER
TANK #7	- 10' DIA, 25' LONG, 1 ACTIVE PORT, 6 - 32" DIFFUSION PUMPS, LN_2 LINER
BELL JARS	- 24 BELL JAR TEST FACILITIES EXIST FOR COMPONENT DEVELOPMENT AND DURABILITY TESTING



(a) 25-Foot-diameter tank.



(b) 15-Foot-diameter tank.

Figure 5.2.1 - Vacuum facilities.

5.3 Master Phasing Schedule

The SERT C spacecraft test program will be carried out on three (3) models: (1) Struct./Thermal Model, (2) an Engineering Model, and (3) a Protoflight Model.

(1) The Struct. Model will be a full-scale structure in which mass dummy components will be mounted. Vibration, shock, and acceleration testing will insure the adequacy of the structure, and will supply dynamic response data at the subsystem locations providing an important subsystem design criteria.

After mechanical testing, the Struct. Model will be converted to a thermal spacecraft by replacing the mass dummy components with thermal models. Solar simulation testing will then verify the spacecraft thermal analysis and control system. The Thermal Model will also be used to demonstrate the capability of a facility to provide the proper thermal vacuum environment for later tests on the Engineering and Protoflight Models.

(2) The Engineering Model will then be tested. The spacecraft will be an electrically functioning spacecraft. The thermal control system and mechanical configuration will be identical to flight hardware. This model will be tested to demonstrate intersystem compatibility and complete operational capability. A thermal vacuum integration test will be performed to demonstrate proper

operation of all spacecraft systems over the entire qualification temperature range in vacuum. All spacecraft subsystems will be qualification tested during this phase of the program.

Among the system tests to be performed on the Engineering Model will be an Electromagnetic Interference Test, Antenna Range Test, Telemetry System Tests and Magnetic Moment Measurements as well as Spaceflight Tracking and Data Network Compatibility tests.

The Engineering Model spacecraft will not include an operational solar array. The solar array represents a major subsystem development effort which will be conducted in parallel with the center body development. The flight array will be protoflight tested as part of the protoflight spacecraft.

(3) The Protoflight Model will be a complete spacecraft including the flight solar array system. After all testing is completed, this spacecraft will be launched. Protoflight level testing will be performed on this model. All subsystems will be acceptance tested prior to installation in the Protoflight Model spacecraft. The Protoflight Model will be subjected to qualification loads for flight durations. In addition, mass properties, static balancing and alignment measurements of critical surfaces and components will also be made.

Launch site tests will demonstrate readiness for launch and flight. Launch vehicle compatibility and alignment measurements, power, data and tracking system tests will be conducted.

The milestone schedule for the project and critical subsystems is shown in figures 5.3.1 to 5.3.4. To support a launch in mid to late 1978 would require commitment of resources, primarily manpower and associated IMS funds, prior to project approval. Modest amounts of R&D funding should be made available also to initiate procurement of critical components for engineering evaluation and development of software.

5 YEAR

LEWIS RESEARCH CENTER

APPROVAL RESPONSIBILITY ______

ACCOMPLISHMENT RESPONSIBILITY ______

MASTER PROGRAM/PROJECT SCHEDULE
LEWIS RESEARCH CENTER

LEVEL

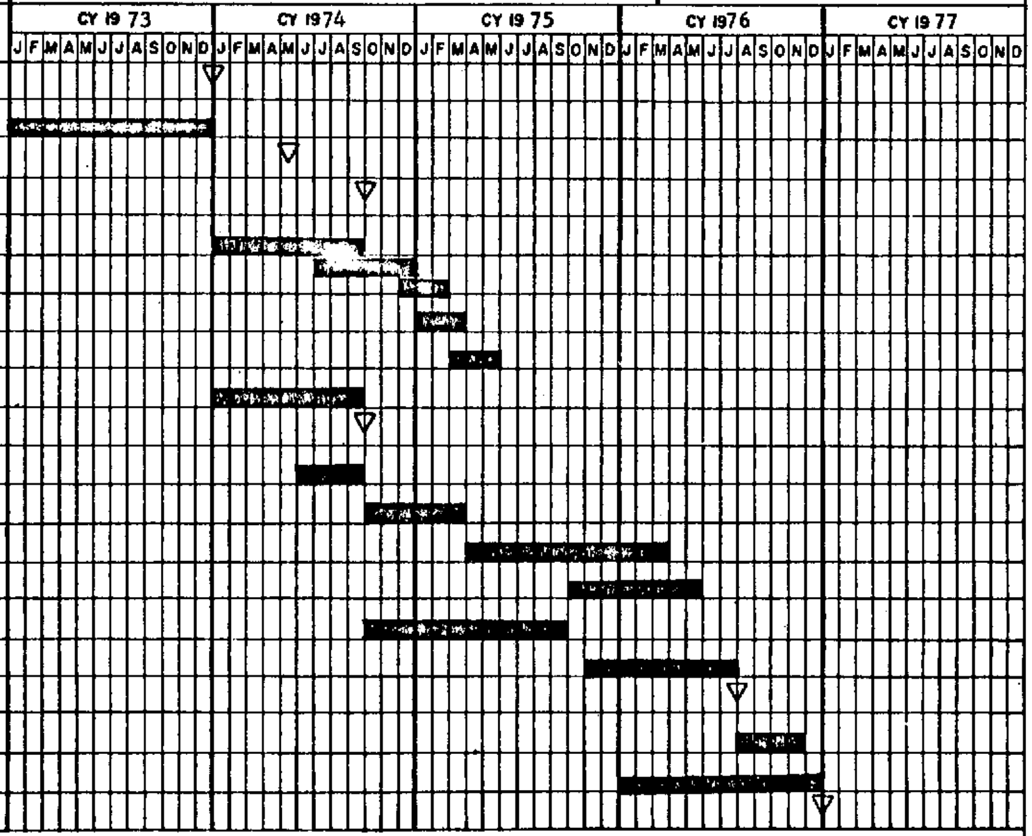
Figure 5.3.1 - SERT C SPACECRAFT

ORIGINAL SCHEDULE APPROVAL ______

LAST SCHEDULE CHANGE ______

STATUS AS OF ______

	MILESTONES
1	MISSION DEFINITION APPROVED
2	S/C & SUBSYSTEM DEFINED
3	PDP SUBMITTED
4	PROJECT APPROVED
5	STRU MOD DESIGN
6	STRU MOD FAB & ASS STRU MOD TESTING
7	STRU/THERMAL MOD CONVERSION
8	THERMAL MOD TESTING
9	SUBSYSTEM DESIGN
10	S/C & SUBSYSTEM D.R.
11	SOW/RFP PREPARATION (1)
12	RFP RELEASE & PROCUREMENT CYC
13	CONTRACT FAB & ASS
14	SUBSYSTEM TESTING
15	IN-HOUSE FAB & ASS
16	SYSTEM INTEGRATION TESTING
17	D.R.
18	ENG MOD QUAL TESTING
19	DESIGN UPDATE
20	D.R.



NOTES (1) SERT PROCUREMENT OFFICE ESTABLISHED BEGINNING FY 75

253

NASA-C-922 (Rev. 8-68)

5 YEAR

LEWIS RESEARCH CENTER

APPROVAL RESPONSIBILITY ____

ACCOMPLISHMENT RESPONSIBILITY ____

MASTER PROGRAM/PROJECT SCHEDULE
LEWIS RESEARCH CENTER

LEVEL

Figure 5.3.1 - SERT C SPACECRAFT
(continued)

ORIGINAL SCHEDULE APPROVAL ____

LAST SCHEDULE CHANGE ____

STATUS AS OF ____

MILESTONES

1	PROTOFLIGHT FAB & ASS
2	PROTOFLIGHT ACCEPTANCE TESTING
3	LAUNCH READINESS REVIEW
4	LAUNCH PREPARATION
5	SCHEDULE CONTINGENCY (2)
6	LAUNCH (3)
7	
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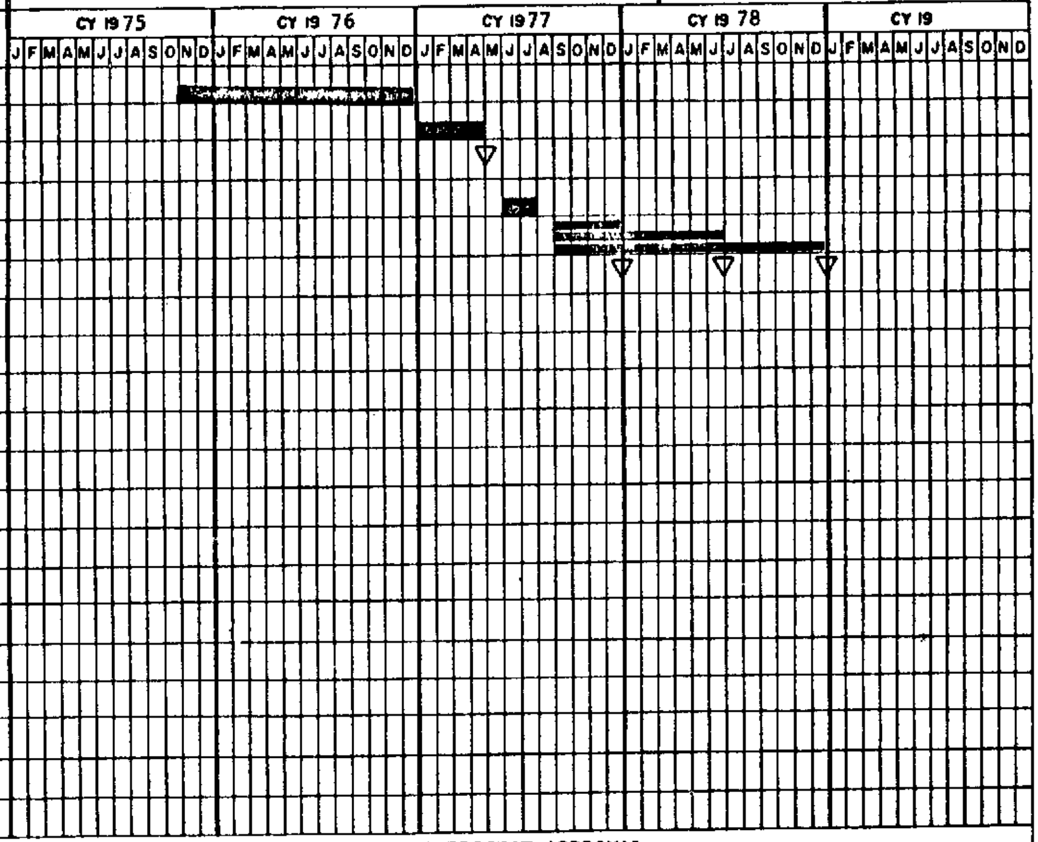
19	
20	



NOTES (2) SCHEDULE CONTINGENCY DEPENDS ON SUPPORT PROVIDED PRIOR TO PROJECT APPROVAL

(3) DEC 1977 UNREALISITC, JUNE 1978 POSSIBLE, DEC 1978 MOST PROBABLE

NASA-C-922 (Rev. 8-68)

5 YEAR

LEWIS RESEARCH CENTER

APPROVAL RESPONSIBILITY ____________

ACCOMPLISHMENT RESPONSIBILITY ____________

MASTER PROGRAM/PROJECT SCHEDULE
LEWIS RESEARCH CENTER

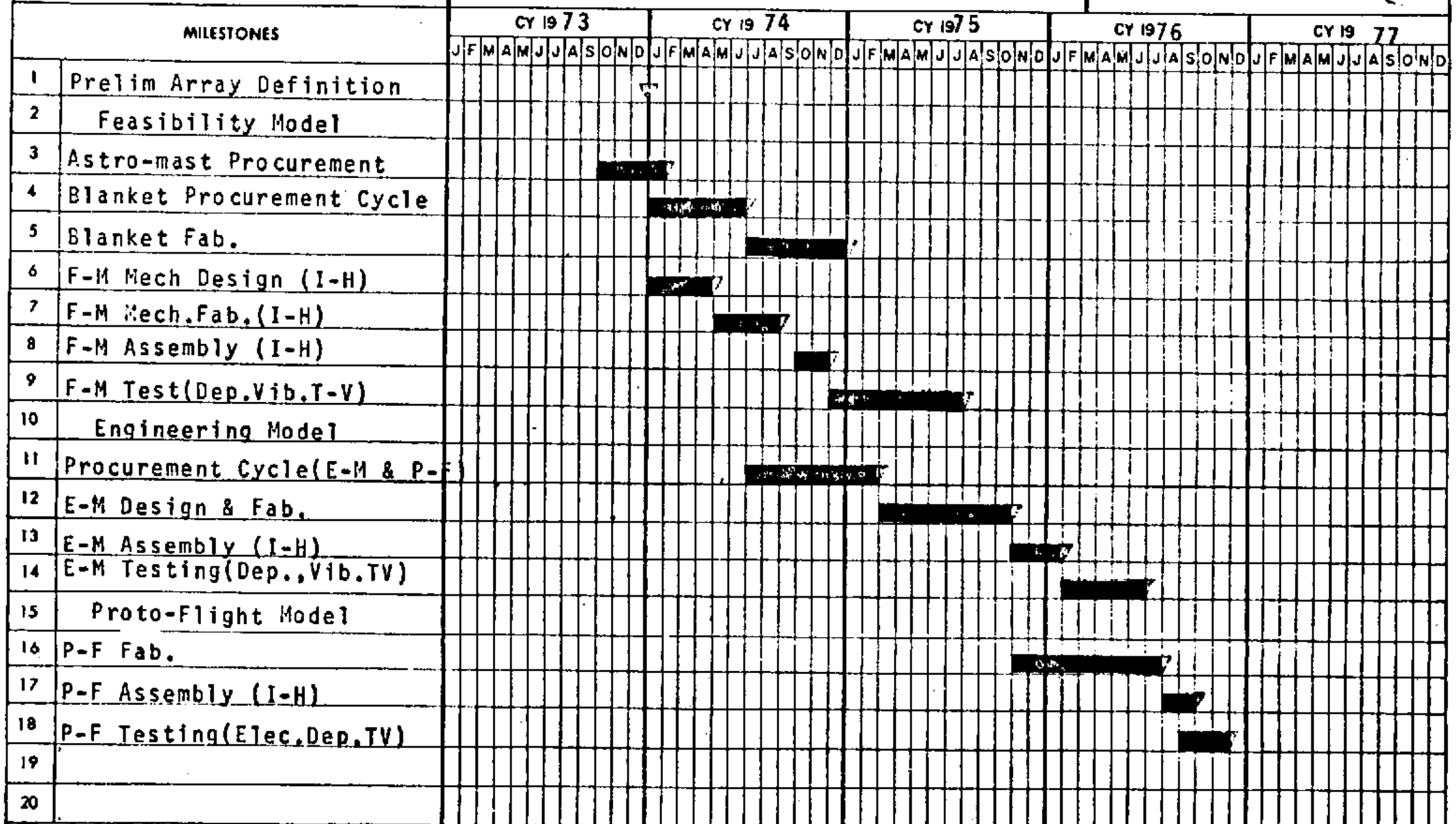
LEVEL

Figure 5.3.2 - Solar Array Schedule

ORIGINAL SCHEDULE APPROVAL ______

LAST SCHEDULE CHANGE ______

STATUS AS OF ______



NOTES

NASA-C-922 (Rev. 8-66)

5 YEAR

LEWIS RESEARCH CENTER

APPROVAL RESPONSIBILITY ______

ACCOMPLISHMENT RESPONSIBILITY ______

MASTER PROGRAM/PROJECT SCHEDULE
LEWIS RESEARCH CENTER

LEVEL

Figure 5.3.3 - ATTITUDE CONTROL AND STATION KEEPING ANALYSIS AND DESIGN

ORIGINAL SCHEDULE APPROVAL ______

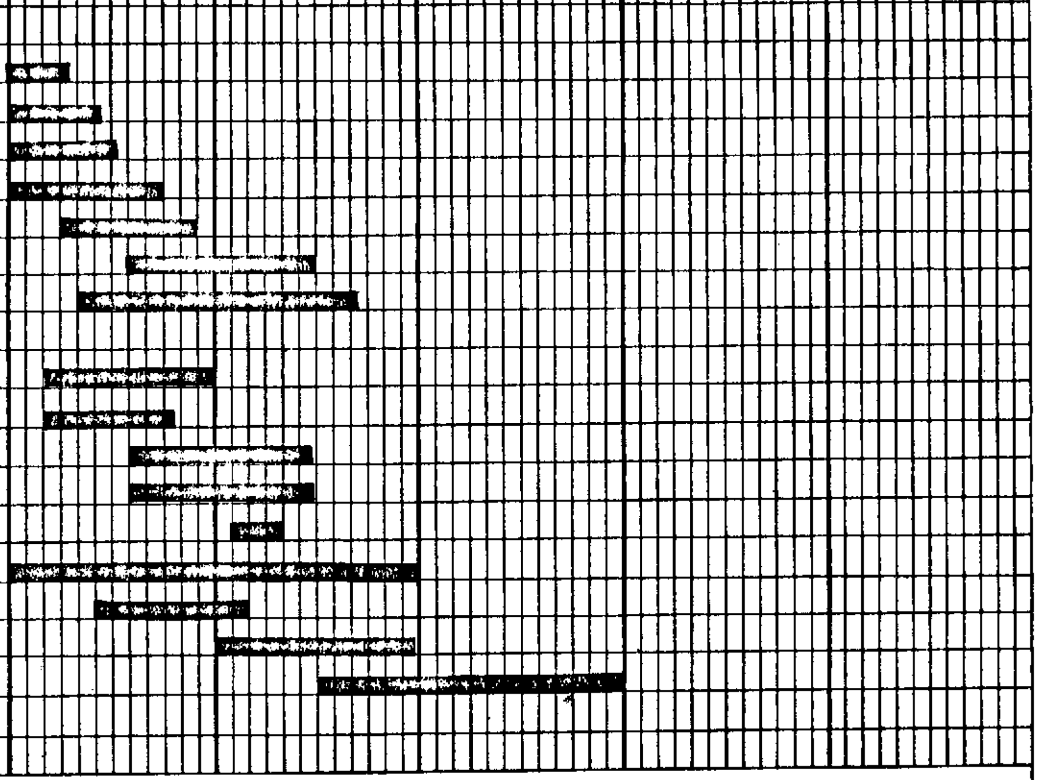
LAST SCHEDULE CHANGE ______

STATUS AS OF ______

CY 1974 | CY 1975 | CY 1976 | CY 1977 | CY 19

J F M A M J J A S O N D | J F M A M J J A S O N D | J F M A M J J A S O N D | J F M A M J J A S O N D | J F M A M J J A S O N D

	MILESTONES
1	COMPONENT SIMULATION:
2	WHEELS
3	THRUSTERS
4	SENSORS
5	SOLAR ARRAY
6	ORIENTATION MECHANISM
7	ON BOARD COMPUTER
8	INTERFACES
9	FUNCTIONAL SIMULATION:
10	RENDEZVOUS
11	REACQUISITION
12	ORBIT RAISING
13	ON ORBIT
14	STATION WALK
15	SUBSYSTEM DESIGN
16	COMPONENT SELECTION
17	FUNCTIONAL TEST SIMULATION
18	DESIGN UPDATE
19	
20	



NOTES

NASA-C-922 (Rev. 8-68)

5 YEAR

LEWIS RESEARCH CENTER

APPROVAL RESPONSIBILITY ________

ACCOMPLISHMENT RESPONSIBILITY ________

MASTER PROGRAM/PROJECT SCHEDULE
LEWIS RESEARCH CENTER

LEVEL

Figure 5.3.3 - ATTITUDE CONTROL AND STATION KEEPING SUBSYSTEM BUILDUP (continued)

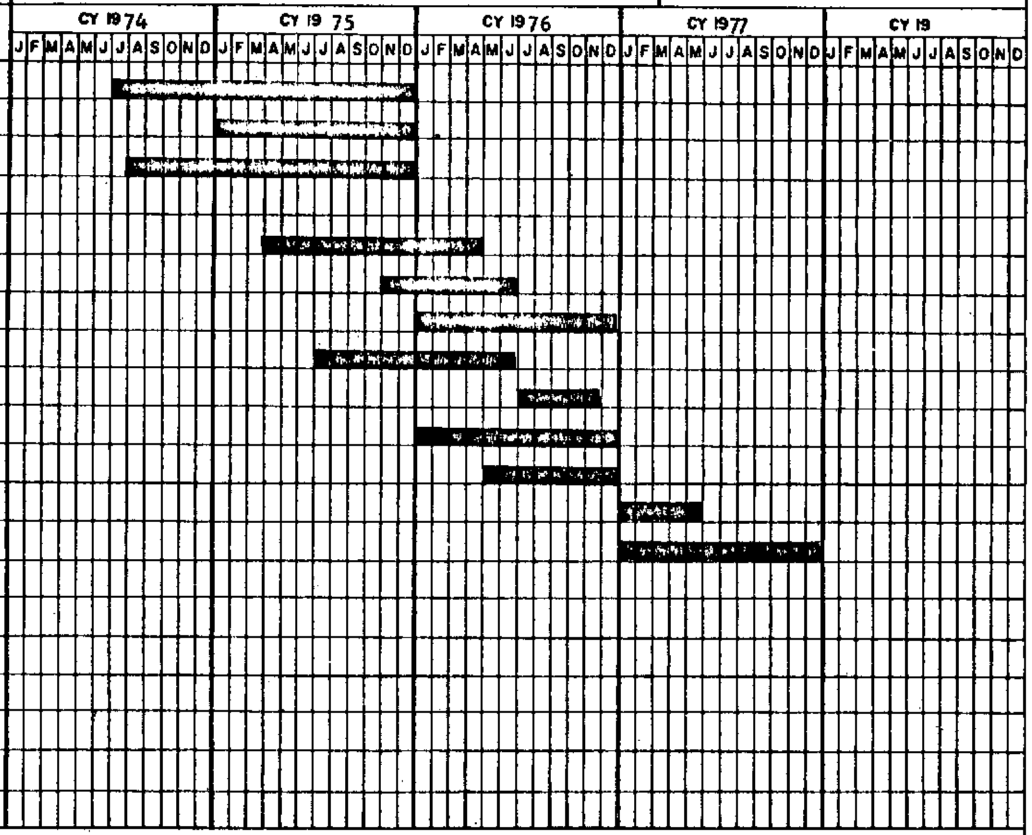
ORIGINAL SCHEDULE APPROVAL ________

LAST SCHEDULE CHANGE ________

STATUS AS OF ________

MILESTONES

1 COMPONENT PROCUREMENT
2 INTERFACE PROCURE OR FAB
3 TEST FACILITY MODS
4 FUNCTIONAL TESTS
5 COMPONENTS
6 INTERFACES
7 SUBSYSTEM
8 ENG MODEL FAB
9 ENG MODEL QUAL TESTS
10 DESIGN CHANGES ASSEMBLY & TEST
11 PROTOFLIGHT ASSEMBLY
12 ACCEPTANCE TESTS
13 PRELAUNCH ACTIVITIES
14
15
16
17
18
19
20



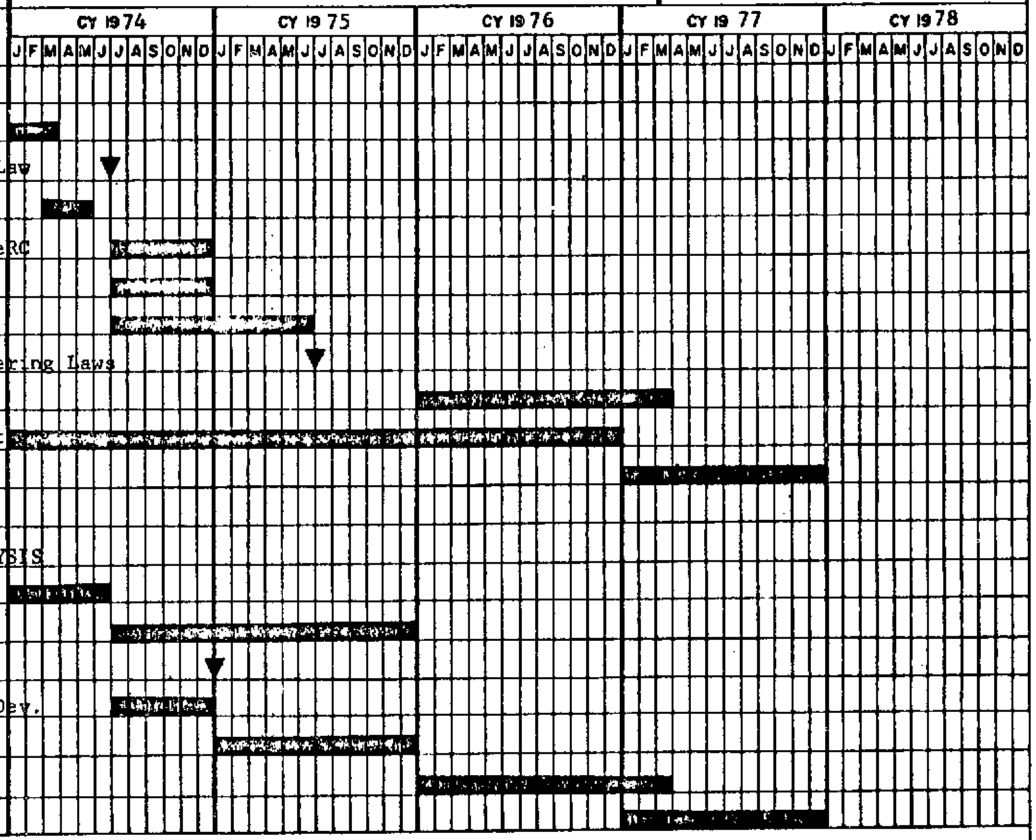
NOTES

NASA-C-922 (Rev. 8-68)

5 YEAR

LEWIS RESEARCH CENTER	MASTER PROGRAM/PROJECT SCHEDULE LEWIS RESEARCH CENTER (LEVEL)	ORIGINAL SCHEDULE APPROVAL
APPROVAL RESPONSIBILITY	Figure 5.3.4 - MISSION ANALYSIS	LAST SCHEDULE CHANGE
ACCOMPLISHMENT RESPONSIBILITY		STATUS AS OF

	MILESTONES
1	ORBIT RAISING MISSION ANALYSIS
2	Install EPSTOP
3	Select Circular vs Elliptic Law
4	Run GSFC/MIT Capability
5	Install GSFC Capability at LeRC
6	Install Simulation Program
7	Continue Performance Studies
8	Confirm Orbit Selection, Steering Laws
9	Develop Flight Software
10	Provide Continual M/A Support
11	Flight Preparations
12	
13	SYNCHRONOUS ORBIT MISSION ANALYSIS
14	Rendezvous Anal. Soft. Dev.
15	Rendezvous Analysis
16	Rendezvous Hardware Req.
17	Station Keeping Anal. Soft. Dev.
18	Station Keeping Analysis
19	Flight Software Dev.
20	Flight Preparations



NOTES

NASA-C-922 (Rev. 8-68)

5.4 Manpower and Cost Estimates

The cost elements used to determine the estimated program cost are given in Tables 5.4.1 and 5.4.2. Subsystem cost estimates were developed by considering manpower requirements, contracted and in-house R&D, IMS support and new facility requirements.

A description of the cost elements used in preparing the tables is as follows.

Man-years - Manpower requirements are presented for the three phases of the total program. These include an approximate one year period prior to project approval, a four year period from project approval to launch (project) and a post-launch period of six years duration. In determining R&D and R&PM resources, the project and post-launch manpower estimates are considered separately.

Hardware - Contracted R&D was subdivided into the areas of development, structure and thermal model fabrication, engineering and prototype hardware and ground support equipment (GSE) costs.

In-House Direct Research - This category includes test, instrumentation and hardware items which are project specific and which are required for subsystem in-house testing. This effort will permit more definitive definition of the hardware purchased for the engineering and protoflight spacecraft.

Unique IMS - This category includes IMS project support as defined by FY 74 RTOP guidelines supplied by NASA Headquarters. It includes such items as liquid nitrogen, computer services, other facility support, instrumentation, outside fabrication, materials, etc.

<u>Facility</u> - These costs cover on-site facility modifications or additions needed to meet subsystem specifications and test requirements.

Other costs such as IMS base support and R&PM personnel were calculated using the formulas prescribed by FY 74 RTOP guidelines. These formulas are shown in the headings of Table 5.4.2.

<u>Project Totals</u> - A contingency of 20 percent as well as an estimate of the effect of a yearly 5 percent inflation adjustment were added to the net R&D and total program costs. The latter estimate presumed an expenditure rate distribution of project dollar amount shown in Tables 5.4.1 and 5.4.2. The expenditure rate distribution assumed is shown in the following table.

Year	Percent of initial dollar estimate expended	Percent inflation of initial dollar estimate expended
1	10	5
2	40	10
3	40	15
4	10	20

The project cost resulting from application of the above factors and the manpower requirements are shown in Table 5.4.3. The values in Table 5.4.3 represent those traditionally reported as project cost and manpower requirements from project approval to launch.

TABLE 5.4.1 - SERT C MANPOWER/COST ESTIMATES (DOLLARS IN THOUSANDS)

(Manpower and R&D)

Subsystem	MANYEARS Prior PP	Project	Post Launch	HARDWARE Dev	Stru/Th	Eng	Proto flight	GSE	Contract Total	InHouse Direct R&D	NET R&D
THRUSTERS + gimbals 30 cm	1	8	1	150	0	405	525	0	1080	0	1080
8 cm	2	20	6	0	0	160	480	0	640	0	640
P.P.U. 30 cm	2	5	1	100	0	375	750	0	1225	0	1225
8 cm	1	10	5	0	0	300	1350	0	1650	0	1650
SOLAR ARRAY	5	43	2	100	0	1310	3674	200	5284	0	5284
SAOM	2	12	1	100	0	150	300	50	600	30	630
ATTITUDE CONTROL	12	51	2	0	0	1219	1550	50	2819	0	2819
COMPUTER	2	32	12	150	0	100	250	150	650	65	715
POWER SYSTEM	1	18	1	175	0	134	223	15	547	0	547
BATTERIES	0	5	0	0	0	20	60	0	80	0	80
T T & C	3	43	28	0	0	198	325	210	733	0	733
STRUCTURE	7	14	1	0	0	0	0	0	0	100	100
TV	1	2	1	200	0	200	400	120	920	50	970
RADAR		2		350	0	325	650	100	1425	50	1475
THERMAL	2	16	2	0	50	50	50	0	150	20	170
HYBRID P.C. (HVSA)	2	12	2	70	0	195	210	35	510	0	510
MISSION ANALYSIS (SK)	6	17	7	390	0	0	0	0	390	0	390
S/C DESIGN	2	32	0	0	0	0	0	0	0	0	0
S/C INTEGRATION	2	60	5	0	0	0	0	100	100	0	100
S/C TEST	0	30	5	0	0	0	0	150	150	0	150
TOTALS	48	432	82	1685	50	5141	10797	1130	18913	315	19268
w/20% Contingency											23122
w/5%/yr inflation											25767 *

* Depends on rate of expenditure; here assumed 10%:40%:40%:10%

TABLE 5.4.2 - SERT C MANPOWER/COST ESTIMATES (DOLLARS IN THOUSANDS)

(Total Project Cost)

Subsystem	IMS		IMS Base Support (2.4xMY)		R&D Resources		Personnel (31.7xMY)		R&PM IMS Base Supp (2.9xMY)		R&PM Resource		Facility	Total Project Summary Manyears		Total Project Summary Dollars	
	Unique	10K/MY	Proj	Post Launch	Proj	Post Launch	Proj	Post Launch	Proj	Post Launch	Proj	Post Launch		Proj	Post Launch	Proj	Post Launch
THRUSTER 30 cm + gimbal	220	80	19	2	1399	2	254	32	23	3	277	35	0	8	1	1676	37
8 cm	120	200	48	14	688	14	634	190	6	17	640	208	0	20	6	1328	222
P.P.U. 30 cm	90	45	11	2	1236	2	159	32	13	3	172	35	0	4	1	1408	37
8 cm	20	100	24	12	1674	12	317	159	29	14	346	174	0	10	5	2020	186
SOLAR ARRAY	100	430	103	5	5917	5	1363	63	125	6	1488	69	0	43	2	7405	74
SAOM	0	120	29	2	779	2	380	32	35	3	415	35	0	12	1	1194	37
ATTITUDE CONTROL	0	510	122	5	3451	5	1617	63	148	6	1765	69	100	51	2	5216	74
COMPUTER	50	320	77	29	1162	29	1014	380	93	35	1107	415	0	32	12	2269	444
POWER SYSTEM	0	180	43	2	770	2	571	32	52	3	623	35	0	18	1	1393	37
BATTERIES	0	50	12	0	142	0	159	0	15	0	174	0	0	5	0	316	0
T T & C	50	430	93	67	1306	67	1363	888	125	81	1488	969	0	43	28	2794	1036
STRUCTURE	0	140	34	2	274	2	444	32	41	3	485	35	0	14	1	759	37
TV	0	20	5	2	995	2	63	32	6	3	69	35	0	2	1	1064	37
RADAR	0	20	5		1500		63		6		69		0	2		1569	
THERMAL	400	160	38	5	768	5	507	63	46	6	553	69	0	16	2	1321	74
HYBRID P.C. (DRSA)	0	120	29	5	659	5	380	63	35	6	415	69	0	12	2	1074	74
MISSION ANALYSIS (SK)	400	170	41	17	1001	17	539	222	49	20	588	292	0	17	7	1589	309
S/C DESIGN	0	320	72	0	397	0	1014	0	93	0	1107	0	0	32	0	1504	0
S/C INTEGRATION	0	600	144	12	844	12	1902	159	174	15	2076	174	350	60	5	2920	186
S/C TEST	200	300	72	12	722	12	951	159	87	15	1038	174	100	30	5	1760	186
TOTALS	1650	4315	1026	195	25684	195	13694	2601	1201	239	14895	2754	550	432	82	40579	3087
w/20% Contingency																48695	
w/5%/yr Inflation																57076*	

* Depends on rate of expenditure; assumed 10%:40%:40%:10%

TABLE 5.4.3 - PROJECT TOTALS

COST (Net R&D) (includes contingency & inflation)	$25,767,000
MANPOWER	432 M.Y.

TABLE OF CONTENTS

1.0 MANAGEMENT INFORMATION

1.1 Payload Title

Colloid Thruster Flight Test

1.2 Security Classification

Unclassified	Propulsion system design Propulsion system performance characteristics Contract costs and schedules
Confidential (Gp-4)	Military planning schedules Overall program funding
Secret (Gp-3)	Vulnerability data (including physical access to affected hardware)
Secret-Restricted Data (Gp-1)	Nuclear radiation effects data

1.3 Associated Personnel

1.3.1 Experimenter

Captain Stanley G. Rosen
Air Force Rocket Propulsion Laboratory

Mailing Address: AFRPL/LKDS
Edwards, California 93523

TWX Address: 910-491-1351

Business Phone: (714) 553-2820

Home Phone: (805) 258-5771

1.3.2 Sponsor

Project Director: Captain Stanley G. Rosen
Air Force Rocket Propulsion Laboratory/LKDS
Edwards, California 93523
(714) 553-2820

Project Coordinator: Captain Craig A. Baer
Space and Missiles Systems Organization/DYAX
P.O. Box 92960
Worldways Postal Center
Los Angeles, California 90009
(213) 643-1128

1.3.3 Field Representatives

To be assigned. Information will be available about January 1974.

1.3.4 Real Time Data Analysis

	Sidney Zafran Thruster System Engineer TRW Systems Group
Mailing Address:	Bldg. 0-1, Room 1220 One Space Park Redondo Beach, California 90278
TWX Address:	910-325-6611
Business Phone:	(213) 536-1915

1.3.5 Early Data Analysis

Not applicable.

1.3.6 Data Reduction

	Sidney Zafran TRW Systems Group
Mailing Address:	Bldg. 0-1, Room 1220 One Space Park Redondo Beach, California 90278
TWX Address:	910-325-6611
Business Phone:	(213) 536-1915

1.4 Affiliated Organizations

Number: 1

Agency: TRW Systems Group

Item of Concern: Thruster System

Name: Frederick A. Jackson

Telephone No.: (213) 536-1915

2.0 PAYLOAD DESCRIPTION AND JUSTIFICATION

2.1 Purpose

This payload will study the operation of a colloid secondary propulsion subsystem in space. It will be the first space test of a colloid thruster.

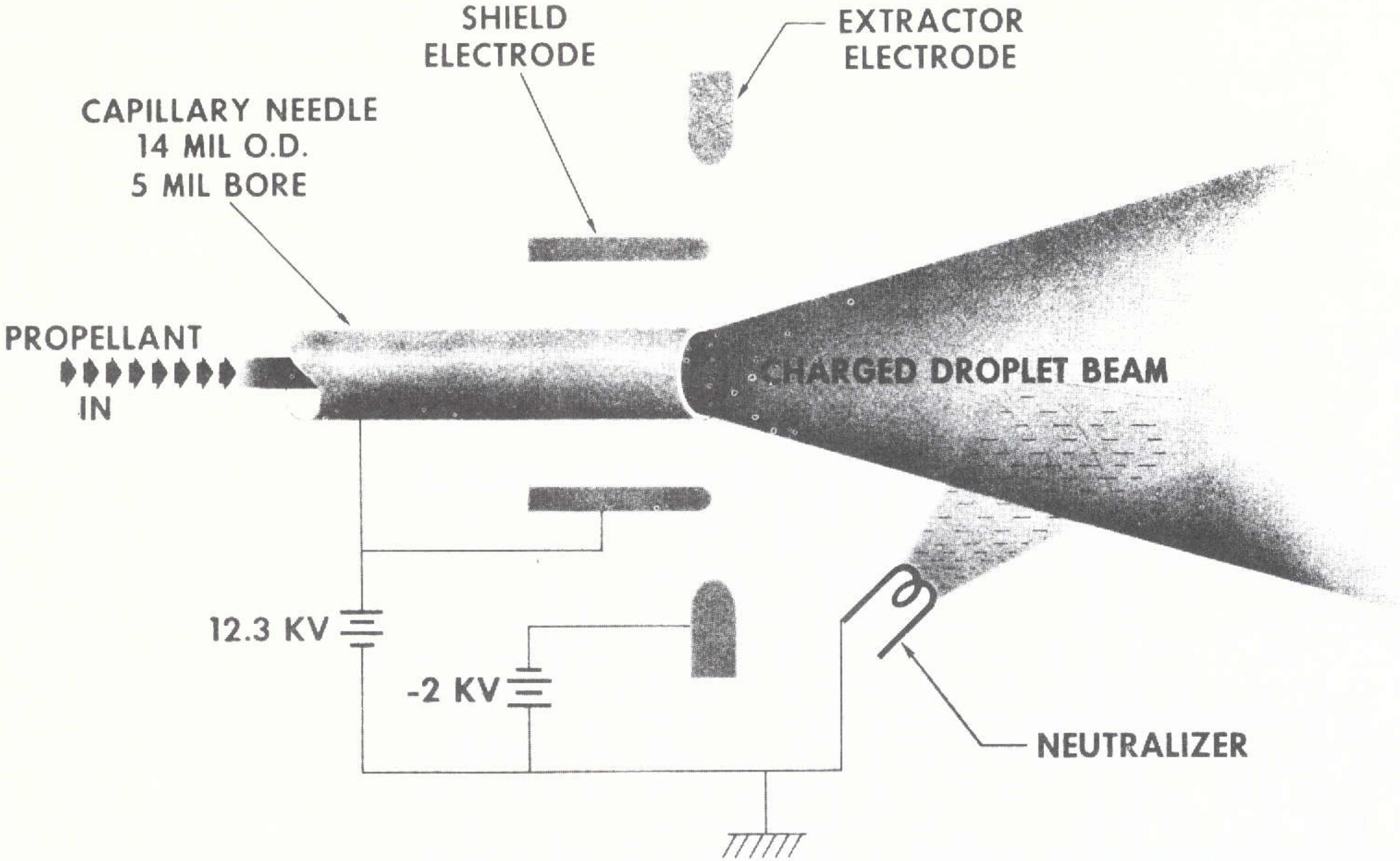
The principle of colloid thruster operation is illustrated in Figure 2-1 for a single capillary needle source. In practice, an array of colloid needles is operated in parallel to achieve the desired thrust level. The 14-mil diameter, 5-mil bore needle is centered within a 1/16-inch diameter shield electrode that in turn is centered within a 1/8-inch diameter extractor electrode aperture. The needle and shield are nominally operated at 12 kv, and the extractor at -2 kv with respect to the spacecraft structure. An electrically conducting fluid (19.3 percent by weight of sodium iodide in glycerol) is fed under low pressure to the needle tip. Thus, the fluid is subjected to an intense electric field at the needle tip. The forces exerted on the liquid surface are surface tension, liquid feed pressure, and electric pressure produced by the field. The combination of these three forces results in an unstable liquid surface from which charged multimolecular droplets are continuously ejected at high velocity to produce a resultant thrust. The positive droplets are neutralized by injecting electrons from a hot tungsten filament wire neutralizer into the exhaust plasma. (The extractor bias provides an electron barrier to the positive potential needles.) Typical thrust from a single needle is 2.3 micropounds at an exhaust velocity corresponding to 1500 seconds specific impulse.

Colloid propulsion is presently in advanced development. Program progress through 1 June 1972 is given in Appendices I and II. Ground testing of the propulsion subsystem has included a 1000-hour demonstration test with a breadboard-model system that has a one millipound thruster, a neutralizer assembly, a positive expulsion bellows propellant feed system, and an inductor energy storage power conditioning and control system. A thruster module (rated at 83 micropounds thrust) has been life tested for 4350 hours.

2.2 Objectives

2.2.1 Primary

The primary objectives of the space test are to (1) measure thrust, (2) record telemetered parametric data on thruster operation, (3) measure the electric field between the spacecraft and ambient plasma while the thruster is operating, and (4) demonstrate that the propulsion subsystem is compatible with a typical host spacecraft.



THRUST	2.3μ LB/NEEDLE
I_{SP}	1500 SECS
PROPELLANT	GLYCEROL + NaI
PROPELLANT SUPPLY PRESSURE	0.25 PSI

Figure 2-1. Thruster Needle Source

Ground testing, at best, can only simulate the space environment. Thruster performance has been shown to be sensitive to the presence of pump oils in vacuum test facilities. Precautions are taken to minimize the influence of test chamber contaminants and to obtain data that are representative of the clean space environment where well wetting propellant at the thruster needle tips exhibits the desired characteristics. Nonetheless, a space test is necessary to assure that well-wetting performance is obtained in ground test stations. A correlation of space flight measured thrust with data taken on thrust stands and time-of-flight collectors (see Section 9 of Appendix I for a description of ground test instrumentation) is therefore a primary aim of this experiment.

For the colloid thruster to function properly, the neutralizer must give off a current of electrons that equals the current of charged droplets leaving the needle tips. The neutralizer injection potential (typically 50 volts) required to attract this electron current establishes the potential of the spacecraft with respect to its ambient plasma. In ground tests, the neutralizer injection potential is determined by electrically isolating the thruster subsystem from the test chamber and by measuring the voltage it reaches with the thruster and neutralizer in operation. Thus, the thruster subsystem is allowed to operate negatively with respect to the vacuum chamber in a manner that is analogous to the spacecraft operating negatively with respect to its ambient plasma. For this measurement of neutralizer injection potential to be valid, all the neutralizer electrons must couple to the thruster exhaust beam. If, for example, some neutralizer electrons were striking the test chamber, and secondary electrons from the test chamber target collector were entering the exhaust beam, a false indication of neutralizer injection potential would be obtained. Therefore, it is desirable to verify the neutralizer injection potential on the spacecraft. At synchronous altitude, however, the ambient plasma density is generally so low that a direct measurement of spacecraft potential is difficult to obtain. Instead, the electric field is measured at the spacecraft surface.

A successful space flight test will qualify the colloid thruster for use on operational spacecraft. Its high specific impulse when compared with chemical thrusters, and modest power requirement when compared with other electrostatic thrusters, makes it attractive for application in low thrust, high total impulse missions. North-south stationkeeping of geosynchronous satellites, for communications or navigational purposes, affords a good example of a low thrust, high total impulse requirement.

2.2.2 Secondary

Secondary objectives of the space test are (1) to obtain lifetime data on the colloid propulsion subsystem and (2) to measure the amount of exhaust material reaching the spacecraft structure.

Ground life testing is handicapped by the presence of backstreaming material from the test chamber target that returns to and deposits on the thruster face. This material leads to rapid neutralizer filament carbiding. It also accumulates on the shield electrode, resulting in thruster performance degradation with time. A long duration space test provides a test environment free from backstreaming material and consequently, lifetime data that are not degraded from spurious influences. Telemetered information, taken periodically during the long duration test, will monitor thruster subsystem status and performance as a function of operational lifetime.

It is desirable to monitor the rate of mass accumulation on the spacecraft structure in the vicinity of the thruster in order to verify the low arrival rates of propellant by-products outside the primary exhaust beam. These measurements can then be correlated with ground test data that have characterized the exhaust spray at wide divergence angles.

2.3 Technique Employed

The test will be conducted by operating the colloid thruster subsystem, monitoring its telemetered data, and observing its effects on the spacecraft. On-board measurements of thrust and spacecraft surface electric field will be made by means of an accelerometer and electric field meter, respectively. Mass accumulation on spacecraft surfaces in the vicinity of the thruster will be monitored with a quartz crystal. Operation of the thruster subsystem, accelerometer, electric field meter, and quartz crystal monitor are all discussed briefly below.

2.3.1 Colloid Thruster Subsystem

The subsystem components are physically separated into two packages: the thruster unit, and the power conditioning and control system (PCCS). Figure 2-2 shows the thruster unit containing the thruster, neutralizers, and propellant feed system. Two neutralizers are used, one for redundancy. The PCCS is contained in an electronics enclosure that converts 28 vdc spacecraft input power to the regulated power levels and forms required for thruster unit operation. The thruster unit and PCCS are tied together electrically.

To develop one millipound of thrust, 432 capillary needle sources are operated in parallel. The 432-needle thruster is assembled from a dozen modules, each with 36 needle sources and an individual extractor electrode. The modules are located four on a side of a square array. A heater, bonded to the back of the array frame, provides active thruster temperature control. The center of the array is left open for access to the propellant manifold that couples modules to the propellant flow controller assembly. A cover plate, that is connected to the extractor voltage supply, fits over the center of the array. From this point, voltage is supplied in parallel to each of the module extractors through individual load-limiting resistors.

Propellant is stored in a bellows assembly. Propellant loading through a fill valve causes the bellows to expand under tension. The bellows tension provides the expulsion pressure for the stored propellant. The flow controller assembly accepts propellant from the bellows and feeds it to the propellant manifold at a pressure determined by a beam current feedback loop in the power conditioning unit. Electrical power input to a zeolite canister heater (mass flow controller heater) responds to the feedback loop and releases or adsorbs carbon dioxide to vary the pressure within the flow controller as appropriate for regulation of mass flow.

The neutralizer assemblies are mounted along two thruster sides. Each tungsten neutralizer wire filament is parallel to the outermost row of needles. Neutralizer construction is such that the filament is shielded from a line-of-sight view of the thruster needles.

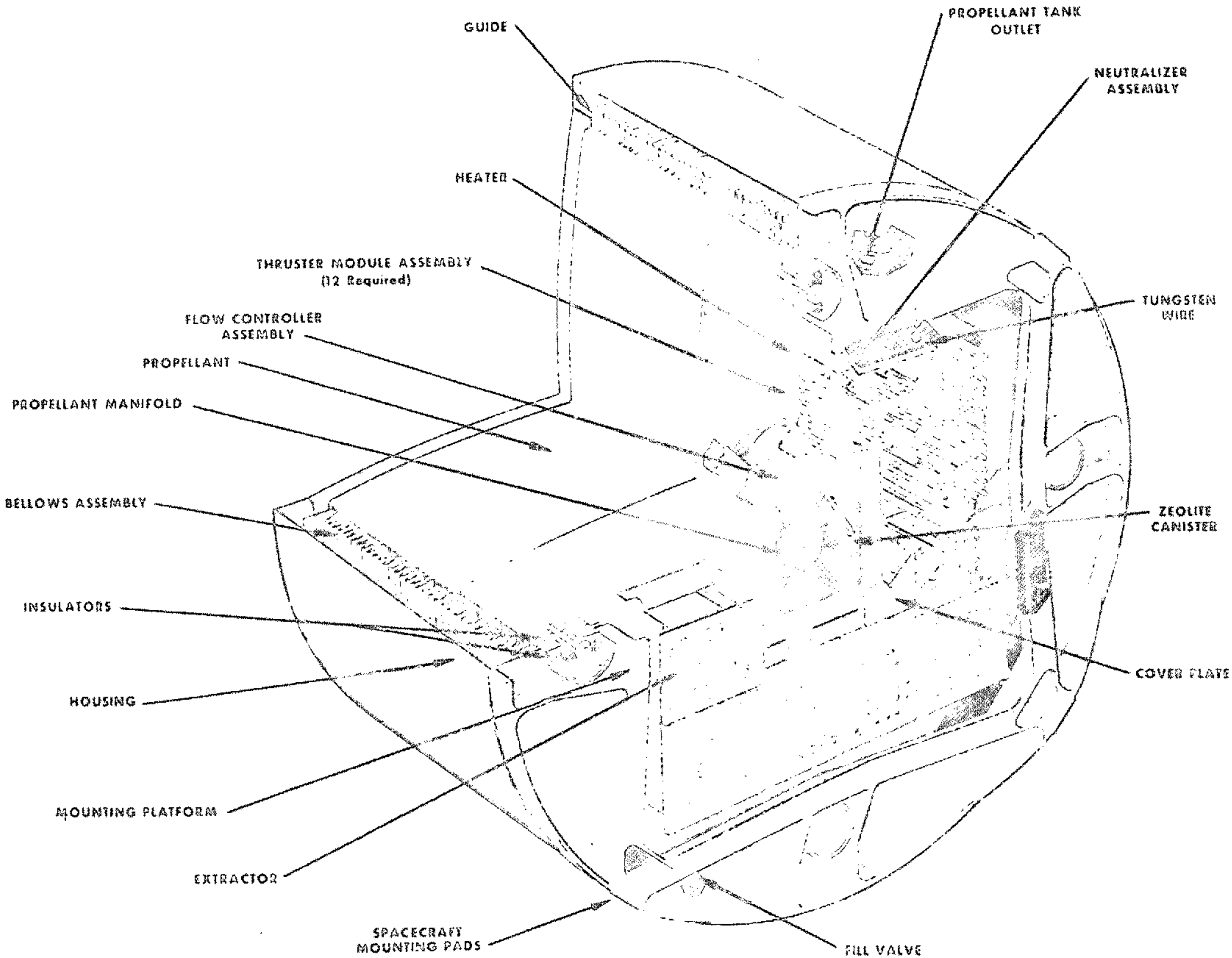


Figure 2-2. Thruster Unit

The mounting platform provides basic structural support for the thruster unit and attaches directly to the spacecraft at four mounting pads. Electrical insulators support the bellows assembly and provide high voltage isolation from ground. Four guide insulators also constrain the bellows during expulsion. The thruster is mounted directly from the bellows assembly, as are the other high voltage feed system components. Isolation transformers are used for the thruster heater and zeolite canister heater in order to minimize insulation requirements near the points of heat application. An outer housing is provided to enclose the thruster unit except for its exhaust beam area. A mask electrode (not shown in Figure 2-2) is placed in front of the extractor array to present a lower, more uniform voltage profile to the downstream plasma. The mask is biased at -300 volts.

The PCCS contains a high voltage converter for the needle, extractor, and mask supplies, a neutralizer control system, a mass flow controller, a thruster temperature controller, a command and protection system, and telemetry signal conditioning. A simplified PCCS block diagram, Figure 2-3, illustrates its major circuit elements and functions.

A two-stage input filter in the main power line attenuates ac switching current generated within the PCCS. This filter also suppresses input voltage transients, thereby protecting power conditioner components. An inductor energy storage high voltage converter provides the needle, extractor, and mask voltages. The three low voltage controls (neutralizer, mass flow controller heater, and thruster heater) and the telemetry system are energized from a multiple output inverter. Auxiliary dc voltages for internal needs are also derived from this inverter.

The PCCS command and protection system primarily controls the start-up and shutdown of the thruster system in a prescribed sequence. Seven command signals are used. Command logic is arranged so that false command signals are prevented from causing undesired action within the PCCS. The protection function acts to disable the mass flow controller under conditions of loss of needle voltage or sustained arcing.

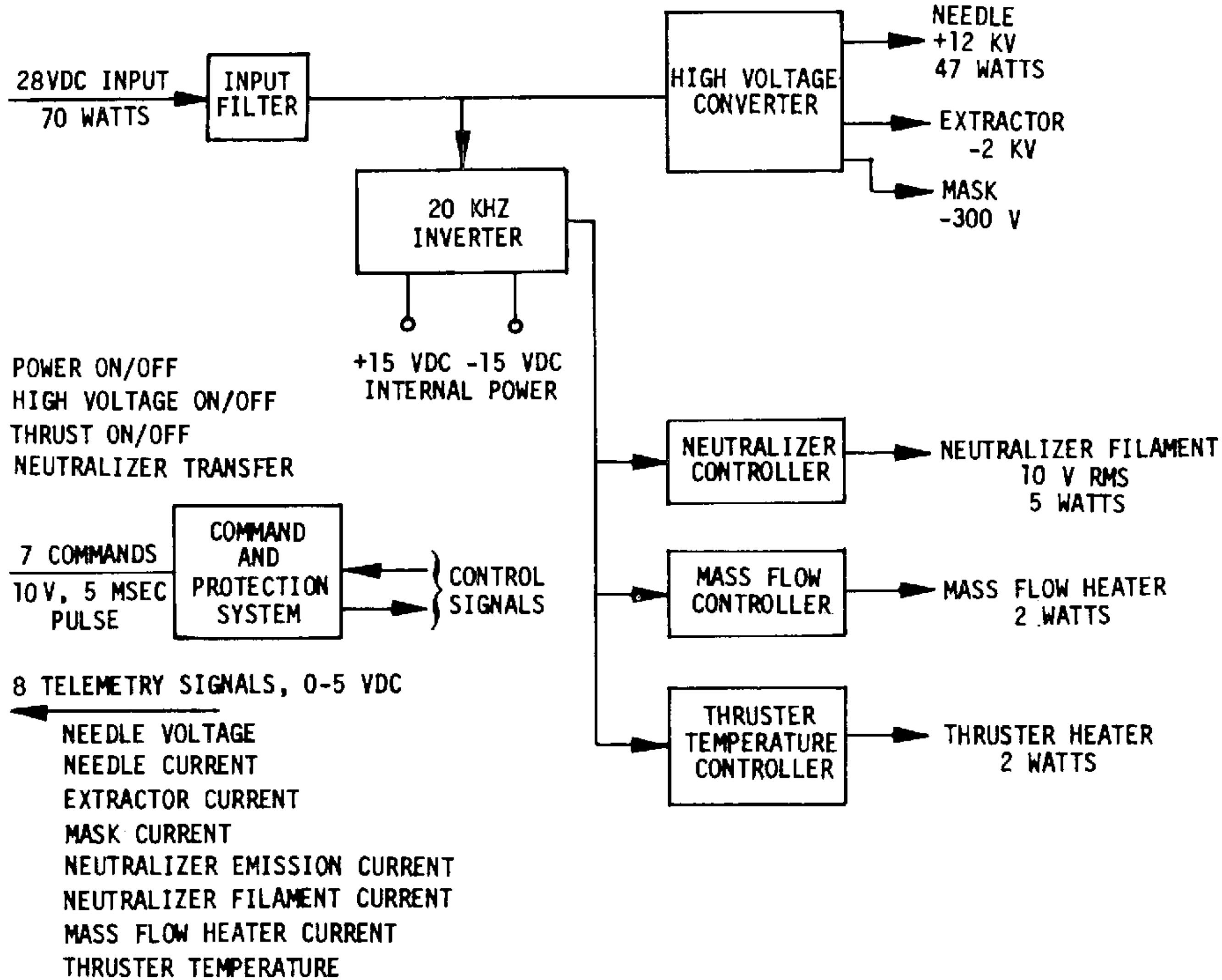


Figure 2-3. PCCS Block Diagram

Eight telemetry signals are provided for monitoring thruster system status. These are needle voltage, needle current, extractor current, mask current, neutralizer emission current, neutralizer filament current, mass flow controller heater current, and thruster temperature signals.

2.3.2 Accelerometer

The accelerometer selected to measure one millipound thrust on a one thousand pound satellite must be rated at 10^{-6} g. Bell Aerospace Company (division of Textron), Buffalo, New York, has developed a miniature electrostatic accelerometer (MESA) in this range that has been flight qualified. A proofmass is electrostatically suspended inside the instrument. Acceleration is measured by sensing the electrostatic force necessary to rebalance the proofmass. The MESA transducer is 4.0 in. diameter x 3.4 in. high. Its electronics are 7.0 in. x 6.0 in. x 3.7 in. The instrument weighs 6 pounds and requires 5 watts power at 28 vdc input. It has a 0-5 vdc analog telemetry output.

Bell Aerospace Company also makes a miniature digital accelerometer system. Here, precision acceleration measurement is made by accurately sensing the electromagnetic force required to rebalance a pendulous proofmass. The proofmass is constrained by a high gain, analog servo loop. An analog current signal through the proofmass-mounted torquer coil provides the rebalancing force. The analog signal is then converted to a digital pulse rate output proportional to input acceleration. The instrument weighs 1.8 pounds, occupies 20 cubic inches (its dimensions are approximately 2.5 in. diameter x 3.5 in. high), and requires 2.7 watts.

2.3.3 Electric-Field Meter

The present version of the electric-field meter (E-meter) is a third-generation design of electron-beam-deflection E-meters that have been developed at TRW. The first was an annular-ring design which successfully returned information on the ion-engine neutralization experiment during the Air Force Series 661A space-flight tests. The second version, which was based on the use of a tungsten-wire filament as the electron emitter, avoided some cathode problems of the original design. The present design overcomes two problems of the second version, viz., the need for stabilization of filament temperature and relative insensitivity to negative spacecraft potential. It is also less susceptible to the possibility of direct electron emission to space from the instrument.

Operation of the E-meter can be understood from the schematic diagram shown in Figure 2-4. A slab beam of electrons is generated by the electron gun, which consists of the filament and Einzel lens. The electron beam is accelerated across the drift space and focused upon the overlapping boundary between the two anode plates so that approximately one-half of the beam current falls on each plate. Any externally applied electric field that penetrates the window will tend to deflect the beam so as to unbalance the anode currents. This tendency is sensed by the amplifier which drives internal deflection plate voltage in the proper direction to cancel the effect of the external field. The deflection plate signal is a low impedance analog voltage that is directly proportional to the externally applied electrid field (of either polarity). It is the output signal of the instrument. To first order, it is independent of electron beam current or accelerating potential. Once the surface electric field is known, it is possible, in principle, to relate its value to the spacecraft potential through the plasma sheath equations and local spacecraft geometry. In practice, these calculations are supported by in-situ calibration of an active instrument on an appropriate mock-up in a large vacuum chamber.

The existing E-meter hardware is a design verification test model. The transducer is 1.5 in. x 1.5 in. x 0.5 in. The transducer and electronics may be mounted together in a single package that will be 3.0 in. x 3.0 in. x 1.25 in. The E-meter will weigh less than one pound, and will require 3 watts input power.

2.3.4 Quartz Crystal Monitor

Quartz crystal monitors (QCM) have been flight qualified by Celesco Industries, Costa Mesa, California, on a variety of satellites. They are capable of resolving mass accumulation on the sensor surface from 10^{-9} to 10^{-4} grams. The sensor assembly contains a sensing crystal and a matched reference crystal that is protected from ambient conditions. As mass accumulates on the sensing crystal, its oscillation frequency changes.

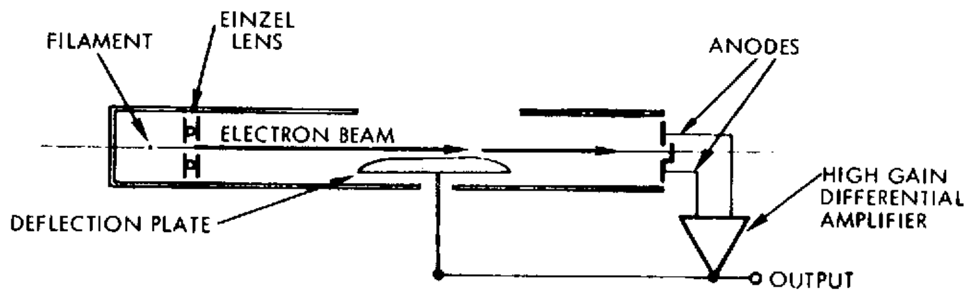


Figure 2-4. E-Meter Schematic

The difference in frequency between the sensing and reference crystal then becomes a measure of mass. The frequency difference signal between the two crystal oscillators is processed to provide a 0 to 5 vdc analog telemetry output. A thermistor in the sensor provides a second telemetry output for sensor temperature. An optional differentiating circuit can also be provided for an analog mass rate output. A typical QCM is 1.2 in. diameter x 3.25 in. high, weighs less than 0.5 pound, and requires less than 1 watt input power.

2.4 Applicable Documents

2.4.1 Drawings

Thruster system drawings will be available in June 1973.

2.4.2 Specifications

The thruster system specification will be available in June 1973.

2.4.3 Other Publications

2.4.3.1 AFRPL-TR-72-10, "Colloid Advanced Development Program Interim Final Report No. 1," February 1972.

2.4.3.2 AFRPL-TR-72-73, "Colloid Advanced Development Program Interim Final Report No. 2," August 1972.

2.4.3.3 Contract F33615-70-C-1694 CDRL Item No. B-019, "Colloid Thruster System Electromagnetic Compatibility Control Plan," 22 February 1972.

3.0 LAUNCH AND ORBIT CHARACTERISTICS

3.1 Launch Window

None.

3.2 Orbit Parameters

Apogee: Any
Perigee: Any
Inclination: Any
Eccentricity: Any

The one-millipound thrust level is sufficient for one or two thrusters to perform a number of space missions aboard spacecraft weighing several thousand pounds. Specifically, they could provide:

a) North-south stationkeeping of a 24-hour synchronous satellite in equatorial orbit

b) Drag compensation for a satellite in a low-altitude orbit

c) Elimination of line of apsides rotation for a satellite in a 24-hour elliptical orbit (typically with 30 degree inclination and 0.25 eccentricity).

If the thruster is to operate between 800 and 2400 n. mi., additional weight will have to be added to the subsystem to shield it from natural or trapped electrons. Under worst case conditions, i.e., a circular orbit at 1600 n. mi. in the trapped electron belt, about 2.5 pounds would have to be added to the weight summary in Section 5.3 for a one-year mission.

If the space test is to be conducted at low altitude, an electron emissive probe* is more suitable than the electric-field meter for determining neutralizer injection potential. An electron emissive probe has been successfully used by NASA/Lewis Research Center on its SERT II space test of an ion engine.

3.3 On-Orbit Life

On-Orbit Life: Min. 28 days
Max. 2555 days

The design life of the thruster is 7 years at 33 percent thrust duty cycle (typically on 8 hours out of every 24 hours). The longer the thrusting time, the closer the space test will come to achieving its secondary objective of obtaining lifetime data. It is felt that 500 hours thrusting time is sufficient to achieve primary test objectives. The minimum on-orbit useful life of the spacecraft is then determined by allowing 7 days to outgas the thruster subsystem and allowing 21 days to accumulate 500 hours operation.

* Ref: R. F. Kemp and J. M. Sellen, Jr., "Plasma Potential Measurements by Electron Emissive Probes," Review of Scientific Instruments, Vol. 37, No. 4, pp. 455-461, April 1966.

3.4 Special Requirements

None.

4.0 ON-ORBIT STABILIZATION/ORIENTATION

4.1 Nominal Look Direction

None.

4.2 Attitude Stabilization

4.2.1 Steady-State Offsets

Any.

4.2.2 Excursion Limits

360 degrees in yaw.

4.2.3 Excursion Rates

The thruster needles are presently designed for 0.25 psi feed pressure on the downstream side of the mass flow controller. A 10 percent allowable variation in flow rate from the innermost to outermost needles would permit 0.025 psi pressure increase from spinning about the yaw axis. For the one-millipound needle array, the outermost needles are located at a 4.6-inch radius from the thruster axis. Knowing that the propellant specific gravity is 1.4, the maximum spin rate is calculated to be 40 rpm.

Higher spin rates would require higher feed pressures for the thruster needles. This could be achieved by using smaller bore needles than are presently employed. The feed pressure to the needles varies inversely as the fourth power of the needle bore diameter. A small change in bore diameter has a large effect on feed pressure.

4.2.4 System Type

Three-axis stabilized is most desirable because it does not introduce any feed pressure gradients in the thruster. Any attitude stabilization system, however, can be used to achieve the thruster look direction.

4.2.5 System Supplier

The attitude control system will not be supplied with the payload.

4.3 Attitude Measurements

There are no specific requirements for on-orbit measurement of payload attitude. It may be desirable to detect the perturbations introduced by the payload thrust level on the attitude control system as a measure of payload thrust.

4.3.1 Source

The attitude sensing system will not be supplied with the payload. There are no specific requirements for the attitude sensing system.

4.4 Separation Requirements

None.

5.0 PHYSICAL DESCRIPTION

5.1 Engineering Layout Drawings

Thruster system drawings will be available in June 1973. The thruster unit, containing a one-millipound thruster and 25 pounds propellant, is about 13 in. diameter and 10 in. long. Its PCCS unit is about 6 in. x 7 in. x 10 in.

5.2 Electrical Connection

There are two electrical connections between the thruster unit and PCCS: (1) a high voltage cable containing the needle voltage and extractor voltage connections, and (2) the low voltage cable. Cable and connector drawings will be available in June 1973.

5.3 Weight Summary

A one-millipound, radiation-hardened colloid thruster subsystem, including the PCCS and 25 pounds of propellant, weighs about 50 pounds. At 1500 seconds specific impulse and 95 percent propellant expulsion efficiency, the thruster has 35,600 pound-seconds total impulse capability. It can, therefore, thrust continuously for 412 days at one-millipound thrust. Reduced thrusting times require less propellant and tankage weight, resulting in a lighter experimental package.

The PCCS weighs about 10 pounds. As mentioned earlier, it is packaged in a separate unit.

5.4 Moving Parts

None except for the propellant storage bellows, mass flow controller, and command circuit relays. It is anticipated that these parts will be non-critical for spacecraft integration purposes.

5.5 Ground Handling Procedures

Ground handling procedures will be available in June 1974. They will be primarily concerned with protecting the thruster needles from dust and contamination that would jeopardize their high voltage isolation in space. The two neutralizer filaments are the most delicate components. They are 0.002 in. diameter tungsten wires located as shown in Figure 2-2.

5.6 Environmental Design Criteria

The qualification hardware has not been fabricated. Qualification tests are scheduled for the fourth quarter of 1973.

5.7 Alignment

It is desirable to align the thrust axis within 0.5 degree of the nominal thrust direction. It should also be located within 0.2 inch of the nominal center of thrust.

The geometric thruster centerline is presently used as the thrust axis. It is difficult to make an accurate thrust vector measurement at the one-millipound thrust level. Thrust vectoring capability has been proposed for aligning the thrust axis in flight. This capability has not been developed for the experimental package.

6.0 ELECTRICAL POWER REQUIREMENTS

6.1 Voltage

Package Name: PCCS
Volts: 28 $\pm$ 4.0 vdc
Current (avg. amps): 2.5
Fuse Rating (amps): 6

The fuse must be capable of handling the inrush current shown in Section 6.6.

6.2 Voltage Regulation

The PCCS may be connected to an unregulated 28 $\pm$ 4 vdc power supply.

6.3 Standby and/or Warm-Up Power

Standby power will not be required.

Warm-up power for the thruster heater will be required to bring the thruster up to its 25°C operating temperature. The power level and duration depend on the thermal integration of the thruster subsystem into the spacecraft. The present thruster heater is rated at 2 watts.

6.4 Average Power

The PCCS operates at 70 watts input power for a one-millipound thrust level. Lower thrust levels use correspondingly lower power inputs.

6.5 Surge Current at Turn-On

The surge current profile at turn-on is sketched in Figure 6-1. Ten amperes peak current is required.

6.6 Surge Current After Turn-On

The surge current profiles when the PCCS power is turned on and when the high voltage is turned on are sketched in Figures 6-2 and 6-3. The peak current requirements are 20 amperes and 6 amperes, respectively.

6.7 Turn-On Constraint

The turn-on constraint is satisfactory.

7.0 ELECTRICAL CHARACTERISTICS

7.1 On-Off Control Devices

The thruster heater is switched on and off to maintain the thruster operating temperature constant at 25 $\pm$ 1°C. The thruster heater control is active during thrusting periods.

7.2 Internal Switching

None.

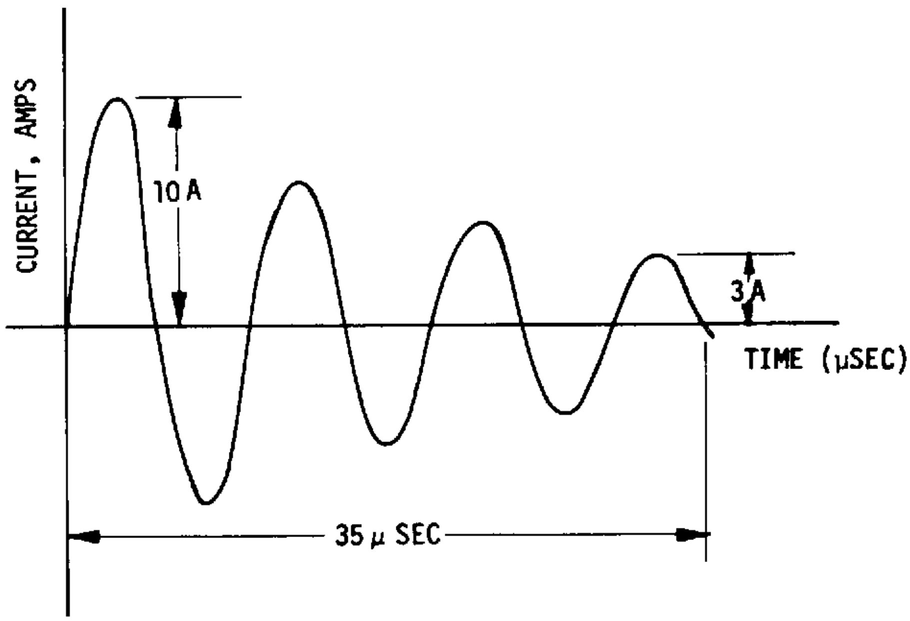


Figure 6-1. Surge Current at Turn-On (32 vdc Input)

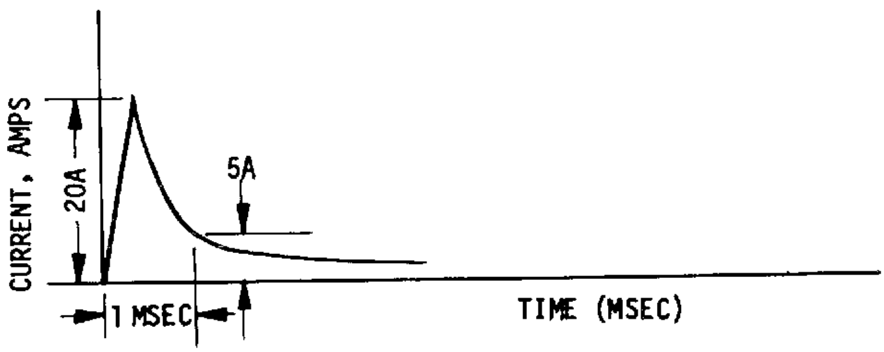


Figure 6-2. Surge Current at Power On (32 vdc Input)

C4

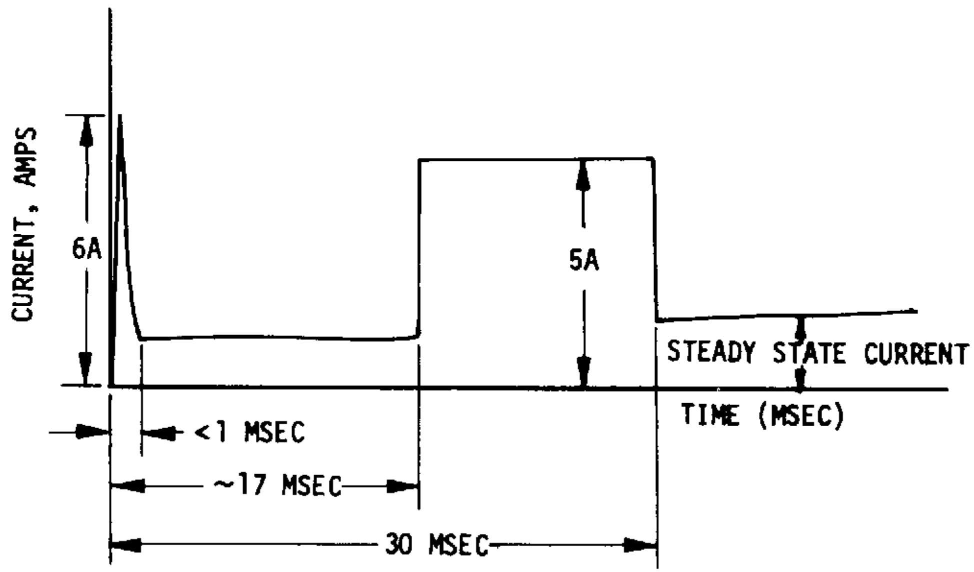


Figure 6-3. Surge Current at High Voltage On (32 vdc Input)

7.3 Input Line Filter

The input power line filter is schematically shown in Figure 6-4.

7.4 Power Isolation

All power is isolated from the spacecraft frame ground.

7.5 Grounding

7.5.1 Single Point Reference Convention

The single point reference convention is acceptable.

7.5.2 Package Internal Configuration

Case A can be used.

7.6 High Voltage Source Location

The three high voltage outputs are:

1. +12 kv needle voltage
2. -2 kv extractor voltage
3. -300 v mask voltage.

These outputs will be physically separated into a high voltage section of the PCCS. Their location in the thruster unit is discussed in Section 2.3.1.

8.0 EMI EMANATION AND SUSCEPTIBILITY

8.1 MIL-STD-826 (USAF)

MIL-STD-826 has been superseded by MIL-STD-461. The thruster subsystem is designed to meet applicable requirements of MIL-STD-461 as described in Appendix III.

8.2 Emanating Frequencies

The PCCS oscillator and converter characteristics are as follows:

(a) 20 kHz oscillator:	17 kHz to 24 kHz square waveform, 0.1 to 2 μsec rise/fall time
(b) 2 kHz oscillator:	2 kHz ± 10% square waveform, 0.1 to 2 μsec rise/fall time

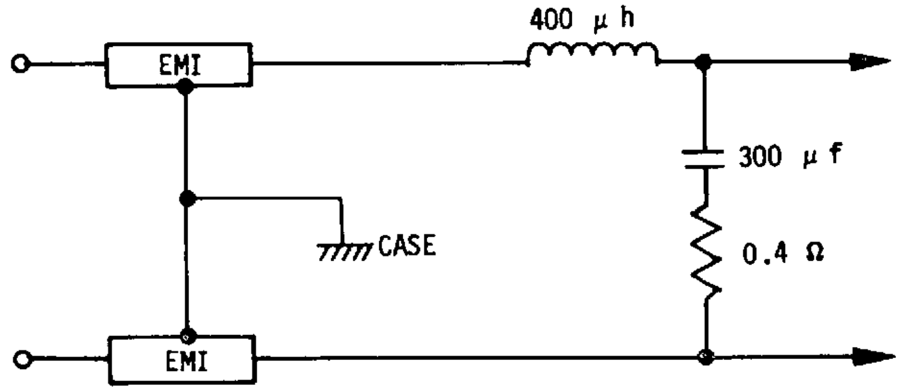


Figure 6-4. Input Power Schematic

(c) High voltage converter: 4 kHz to 12 kHz* asymmetrical square waveform; pulse amplitude and duration are variable, 0.1 to 2 μsec rise/fall time

8.3 Emanation Power

The input power to the PCCS oscillators and converters are as follows:

(a) 20 kHz inverter: 17 watts

(b) 2 kHz inverter: 0.8 watt

(c) High voltage converter: 54 watts

8.4 Shielding

It is possible to shield the PCCS by wrapping shielding around its enclosure.

8.5 Breadboard Circuit

The breadboard PCCS circuit is available.

9.0 DATA HANDLING REQUIREMENTS

9.1 Payload Outputs

9.1.1 FM Outputs

None.

9.1.2 AM Outputs

None.

9.1.3 PCM Outputs

None.

9.1.4 PAM Outputs

None.

9.1.5 Digital Outputs

None.

*At full thrust. At no load the frequency is lower: 30 to 150 Hz

9.1.6 Analog Outputs

Table of Analog Outputs

Function	Range (volts)	Rate (wps)	Distribution	Impedance (ohms)	Accuracy
Needle voltage	0-5	Not critical		≤ 5000	± 1%
Needle current					± 2%
Extractor current					± 5%
Mask current					± 5%
Neutralizer emission current					± 2%
Neutralizer heater current					± 3%
Mass flow heater current					± 3%
Thruster temperature	0-5	Not critical		≤ 5000	± 0.1°C

9.2 Telemetry System

No particular type of system is required.

9.3 General Requirements

9.3.1 Storage Unit

The need for an on-board storage unit and its total storage capacity depend on orbital parameters (not critical for this payload). The space test plan will be available in September 1974.

9.3.2 Time Reference Generator

An on-board time reference generator (clock) is not required.

9.3.3 Sync, Reset, Scan Pulse Characteristics

There are no requirements for synchronization, register reset, register gate, and register scan pulses.

9.3.4 Data Line Shielding

Shielded cables are satisfactory.

9.3.5 Other

None.

10.0 THERMAL CONTROL

10.1 Temperature Limits

Preferred Limits (°F)	Acceptable Limits (°F)	Survival Limits (°F)
0 to 100	-20 to 120	-40 to 180

The above limits apply to the PCCS. The thruster has its own integral thermal control to maintain its operating temperature at 25 ± 1°C. Thermal integration with the spacecraft will require that this temperature not be exceeded during non-operating periods. The thruster is exposed to space, and consequently, can radiate heat to space for thermal control. In general, thermal integration will isolate the thruster from the spacecraft for control purposes.

10.2 Power Dissipation

The thruster exhaust beam is rated at 47 w. The neutralizer takes 5 watts, the thruster heater about 2 watts, and the mass flow controller about 2 watts. The remaining power (14 watts) is dissipated in the PCCS.

10.3 Restrictions

Thruster system drawings will be available in June 1973.

10.4 Thermal Mass

Thruster system drawings will be available in June 1973.

11.0 MAGNETIC FIELDS

11.1 Magnitude

The maximum magnetic field at 18 in. is 75 gamma (C). At 36 in., it is 9 gamma (C).

11.2 Direction

Indeterminate at present.

11.3 Ferromagnetic Material

Thruster system drawings will be available in June 1973.

12.0 COMMAND AND CONTROL

12.1 Functions

Command No.	Type	Function	Effect
1	RT	Power On	Telemetry signals energized
2	RT	Power Off	Telemetry signals off
3	RT	High Voltage On	
4	RT	High Voltage Off	
5	RT	Thrust On	
6	RT	Thrust Off	
7	RT	Neutralizer Transfer	

12.2 Electrical Interface

The thruster subsystem accepts 10 $\pm$ 3 v peak pulses, 50 $\pm$ 10 msec in duration. The maximum pulse rise and fall times are 100 μsec. In the absence of a command, the signal level is 0 $\pm$ 0.5 vdc. The maximum load on each command input is such that the command source does not have to accept more than 1 mA nor supply more than 10 mA.

12.3 Experiment Control and Operation Schedule

The space test plan will be available in September 1974.

12.4 Duty Cycle

The space test plan will be available in September 1974.

12.5 Special Requirements

None.

13.0 POTENTIAL HAZARDOUS EQUIPMENT

13.1 Pressure Systems (Liquid or Gas)

13.1.1 Description

The propellant storage tank is loaded during thruster subsystem assembly. The propellant is stored at 2 psig.

13.1.2 Handling Procedures

Once loaded, there are no special propellant servicing and checkout procedures required.

13.2 Ordnance System

None.

13.3 Ionization Radiation Source

None.

14.0 METEOROLOGICAL SERVICES

None.

15.0 GROUND CHECKOUT AND CALIBRATION AT CONTRACTOR FACILITY

15.1 Pre-Spacecraft Integration

The test through prelaunch plan will be available in June 1974.

15.2 Post-Spacecraft Integration

The test through prelaunch plan will be available in June 1974.

15.3 Aerospace Ground Equipment (AGE)

(a) Contractor Support Equipment: None

(b) Experimenter Support Equipment:

1. Thruster unit simulator
2. Ground checkout console
3. Thruster system mockup model

(c) Special Facilities: To be specified in the test through prelaunch plan (available in June 1974).

16.0 GROUND CHECKOUT AND CALIBRATION AT LAUNCH SITE

16.1 Missile Assembly Building (MAB) Tests

The space test plan will be available in September 1974.

16.2 MAB AGE

(a) Support Organization Equipment: None

(b) Experimenter Support Equipment:

1. Thruster unit simulator
2. Ground checkout console

(c) Special Facilities: To be specified in the space test plan (available in September 1974).

16.3 Launch Pad Tests

The space test plan will be available in September 1974.

16.4 Launch Pad AGE

(a) Support Organization Equipment: None

(b) Experimenter Support Equipment:

1. Thruster unit simulator
2. Ground checkout console

(c) Special Facilities: To be specified in the space test plan (available in September 1974).

16.5 Launch Site Experiment Removal

None.

16.6 Prelaunch TT&C Support Requirements

None.

16.7 Launch Go/No-Go Criteria

To be specified in the space test plan (available in September 1974).

17.0 LAUNCH PHASE REQUIREMENTS

17.1 Trajectory Parameters

None required.

17.2 Photographic Coverage

None required.

17.3 Operation

The payload is not expected to operate during ascent.

18.0 ORBITAL DATA REQUIREMENTS

18.1 Ephemerides

If thrust data can be derived from orbital parameters, it would be desirable to correlate on-board thrust measurements and thruster operating time with the orbital information.

18.2 Real Time Experiment Data

It is desirable to obtain real time data during the initial turn-on of the thruster subsystem for a period of four hours. This will permit observation of the propellant filling process under zero-gravity conditions as propellant proceeds to fill the volume between the mass flow controller valve seat and the thruster needle tips. It would also be desirable to have this display available during the first five days of thruster operation, so that subsequent turn-on sequences may be monitored.

18.3 Magnetic Tapes

TRW has extensive data reduction facilities and capability. Almost any magnetic tape furnished will be satisfactory.

18.3.1 Recording Type

Pre-detection, post-detection, or digital-computer-compatible recording is not necessary.

18.3.2 Track Assignments

No track assignment preferences exist.

18.4 Data Distribution

Data distribution requirements will be specified in the space test plan (available in September 1974).

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SPACE SCIENCES LABORATORY BERKELEY, CALIFORNIA 94720

April 30, 1973

Dr. Harold R. Kaufman/Code RPE
National Aeronautics and Space Administration
Washington, D. C. 20546

Dear Harold:

During our telephone conversation a few days ago, I questioned you about the feasibility of placing a small scientific instrument aboard the SERT III spacecraft. You agreed kindly to forward my inquiry to the appropriate offices if I would supply some details of the instrument and describe our objectives in flying it on this vehicle.

Our group is devoted to astronomical observations at X-ray and ultraviolet wavelengths (1-2000Å), and the focus of this letter is upon wavelengths between 100 and 600Å, which all observers, excepting solar astronomers, have neglected completely. The reasons for this neglect lie firstly in a strong background flux (noise) produced by a geocorona of helium, extending to a radius of about 30000 km, and secondly in the strong photoabsorption by the interstellar medium which is expected at such wavelengths. However, we have reassessed the opacity of the interstellar medium and researched the latest measurements of its density locally in the galaxy, and on this basis we believe that observations between 100 and 600Å may lead to the discovery of galactic sources of extreme ultraviolet (EUV) radiation, and change radically our understanding of the interstellar medium, and of its interaction with the interplanetary medium.

As EUV astronomers, we find any possibility of flying an instrument aboard SERT III very exciting, for it would enable us, once in synchronous orbit, to make the first astronomical observations outside this geocoronal glow. Further, the slow spiral transfer would enable us to map the distribution of this glow, an object of great scientific interest in itself. We would propose a very simple instrument (see Figure I) consisting of a single, parabolic grazing-incidence mirror of circular symmetry, focusing radiation onto a detector. This telescope would have a diameter of about 6½" and a length of about 15", and strapped to the outside would be a small pack, 1" thick and about 2" x 3" in plain view, for detecting helium line radiation. The total mass would lie between 5 and 10 lbs, the power consumption would be about 0.8W, and the telemetry requirements would be three digital pulse channels (or 1 multiplexed channel) capable of handling three detectors counting at up to 3000 cts/sec. Please don't view these as rigid specifications, for the design may be adapted somewhat.

Regarding location, we would prefer a telescope to be situated where the possibility of viewing the sun would be zero, for example in the shadow of the solar panel. In this way special sensing and protective devices might be excluded. As for pointing, we would like to mount an instrument so that change in vehicle orientation would map as much of the sky as possible. If some opportunity to fly on SERT III were to arise, this matter would be examined carefully after we knew more about the vehicle and its mission.

It was good talking with you again, and thinking over it all, it has occurred to me that electric propulsion engineers and astronomers may have more in common than they realize. I feel sure that the earth is slowly becoming too noisy for astronomers, and that the forefront of this field will move one day to the Moon. How on earth are we going to get all that equipment up there - cheaply?

With best wishes. All the best.

Sincerely,

Ray Cruddace

Ray G. Cruddace
Space Astrophysics Group

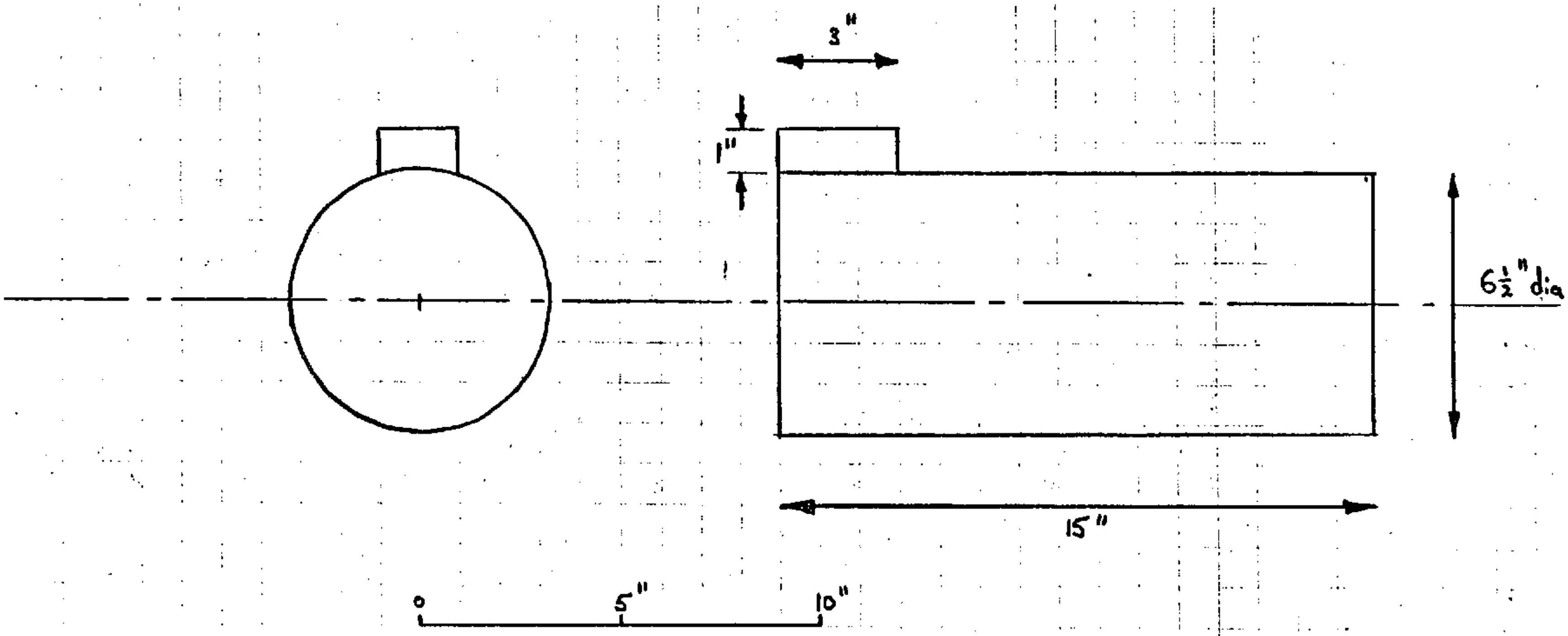

Fig. 1 Envelope of extreme ultra-violet telescope.

C APPENDIX - NICKEL-HYDROGEN BATTERY EXPERIMENT

(COMSAT proposal submitted by James Dunlop, Manager of Energy Storage, Physics Laboratory, Applied Sciences Division, COMSAT Laboratories)

OBJECTIVE

Ni/H_2 batteries offer several potential advantages compared with Ni/Cd batteries as a sealed secondary battery for satellite applications. The two major advantages are significant improvements in both energy density and cyclic life expectancy.

With the greater cyclic life expectancy of the Ni/H_2 battery, it is feasible to use battery power daily on a twelve or twenty-four hour cyclic basis for synchronous satellites with life expectancy of ten years. The adoption of this energy storage concept combined with an electric thruster for north-south station keeping offer a number of advantages which are described in a COMSAT Technical Review (CTR) note entitled "Battery Powered Electric Propulsion for North-South Station Keeping."

SYSTEM DATA

GENERAL

The Ni/H_2 battery is to be designed to operate with one SERT C thruster.

TABLE C-1. - System Definition

One SERT C Thruster, power required, including power conditioning to the thruster 153 watts	
Operating Time for Thruster	1.4 hours
Energy Requirement	214 watt-hours
Bus Voltage (regulated)	28 volts ± 1 volt
Thruster Duty Cycle	* optional once per day twice per day four per day
*The thruster duty cycle for the battery powered thruster is limited only by the solar array power available to recharge the battery.	

CELL DESIGN

Lightweight Ni/H_2 cells which are currently being built for extensive critical item and flight acceptance testing are of four types with different energy densities.

1. Conventional aerospace nickel hydroxide electrodes - 30 mils thick Inconel 625 pressure vessel
2. Conventional aerospace nickel hydroxide electrodes - 30 mils thick Lightweight pressure vessel
3. Thicker aerospace nickel hydroxide electrodes made specifically for the Ni/H_2 cell - 40 mils thick Inconel 625 pressure vessel
4. Thicker aerospace nickel hydroxide electrodes made specifically for the Ni/H_2 cell - 40 mils thick lightweight pressure vessel

Figure C-1 shows the energy density for each of these cells as a function of the energy stored for a complete discharge to 1.0 volt.

BATTERY DESIGN

The Ni/H_2 battery would comprise a number of cells connected in series To meet the energy requirements, one can vary the number of cells, the energy stored in each cell, the depth of discharge, and the efficiency of the dc/dc converter used with the battery Table C-2 shows how these variables influence the number of cells used for the battery and the effects on the energy density of the cells.

As the number of cells used in the battery is increased, the energy density of the cells and the battery decreases as previously shown in figure C-1. With the improvements in efficiency of power conditioning equipment available today, it is reasonable to use few cells in series and boost the voltage on discharge to the desired bus voltage to achieve higher energy density for the energy storage system. The effect of battery voltage on dc/dc converter efficiency is shown in figure C-2. With the Ni/H_2 system it is also reasonable to use 70 to 80 percent depth of discharge. Therefore, for these system requirements, the 66 watt-hour cells currently being built are selected for the battery design. The characteristics of a number of batteries using the 66 watt-hour cells are given in Tables C-3 through C-6.

As shown in Tables C-3 and C-4, the DOD (depth-of-discharge) has a significant impact on the usable energy density supplied by the energy storage system. Boiler plate cells (52 ampere-hour or 66 watt-hours) have already demonstrated on cyclic testing over 850 cycles at 75 percent DOD to date (see fig. C-3).

The advantage of increasing the number of cells in series to obtain higher battery voltage, and hence dc/dc converter efficiency, is not practical because of the loss in energy density as shown by comparing Tables C-3 and C-6 system energy density values. One other approach is to use a bipolar stack arrangement or multicell design within each pressure vessel. However, there is no experimental program presently under way to explore this approach and many potential problems do exist relating principally to electrolyte management.

66 WATT-HOUR CELL CONFIGURATION

A detailed drawing of the Ni/H_2 cell configuration is shown in figure C-4. The pressure vessel is designed with a safety factor of 1.5 operating over a pressure range of 100 to 600 psig. The case material is

Inconel 625. The standard 30-mil thick nickel hydroxide electrode is shown in figure C-5. Outside dimensions would be exactly the same for the 40-mil electrode.

A weight breakdown of the individual cell components is presented in Table C-7, which includes both the Inconel can and lightweight beryllium-nickel can using 30-mil thick nickel hydroxide electrodes. See Table C-7.

Inconel pressure vessels of the design have been fabricated and are currently under test. The lightweight beryllium-nickel pressure vessel is a more advanced design (proprietary with COMSAT Laboratories). Preliminary investigations of hydrogen embrittlement, corrosion in KOH, fabrication and weldability and yield strength show this material to be very promising.

BATTERY CONFIGURATION WITH 66 WATT-HOUR CELL

Since each individual cell is self-contained, the packaging of the cells into a battery is primarily a function of the spacecraft configuration, weight distribution and heat transfer considerations. For example, the cells could be uniformly positioned around the periphery of the spacecraft. They are designed with one feedthrough for the positive electrode with the negative electrode connected to the case; therefore, the case must be insulated from a common ground. This design can be changed to include two insulated feedthroughs if desirable.

To allow for a cell failure, either open circuit or short circuit, it may be desirable to provide a bypass diode circuit arrangement for each cell.

PROPOSED 5-CELL BATTERY FOR SERT C EFFICIENCY

The watt-hour efficiency for the nickel hydrogen cell itself is between 80 and 85 percent. The efficiency of a dc/dc converter for battery discharge is included in the design presented. For the 5-cell battery the efficiency of a dc/dc converter with a fixed gain is 82 percent.

CHARGE-DISCHARGE CHARACTERISTICS

For an individual cell, the charge-discharge voltage is shown in figure C-6. On discharge the voltage goes from 1.35 volts to 1.15 volts. Initially the EOD voltage is higher, 1.23 volts; 1.15 volts is estimated as the EOD voltage after several thousand cycles.

Using a fixed gain dc/dc converter the voltage on discharge would vary from approximately 30.5 volts to 26 volts, 1.35 to 1.15 volts per cell.

Either constant current or constant voltage charging methods are possible and depend on the overall system considerations. Constant current charging is desirable for a fixed duty cycle and charging time. The charging current would be selected to provide approximately 20 percent of ampere-hour overcharge. If constant voltage is selected to charge the cells, it is probably desirable to use a pressure switch sensing the hydrogen pressure to terminate the charge. Since pressure is an indication of the state-of-charge, this approach is feasible and does represent another potential advantage for the nickel-hydrogen cell.

LIFE EXPECTANCY

Nickel hydrogen boilerplate cells have already demonstrated over 3000 cycles in an accelerated test still in progress. At one cycle per day the life expectancy at 70 to 80 percent DOD is still not defined, but three-to five-year operation in terms of ampere-hour turnover has been demonstrated. Within the timeframe of this proposed experiment it is possible to completely demonstrate on a real-time basis the cyclic life and performance characteristics of prototype cells.

ENVIRONMENTAL LIMITS

The desirable temperature range for nickel-hydrogen is similar to nickel cadmium cells. We have demonstrated performance sufficient to meet these system requirements between the temperature limits of 25° C to -10° C.

The 66 watt-hour cells are designed to meet the shock and vibration specification for ATS-F.

If all five cells are positioned in series, the dimensions are 8.9 cm × 44.5 cm × 22.5 cm.

DEVELOPMENT SCHEDULE

The development schedule for the 5-cell battery is shown in figure C-7.

TABLE C-2. - THE EFFECTS ON THE NUMBER OF CELLS USED FOR THE BATTERY AND ON THE ENERGY DENSITY OF THE CELLS DUE TO CHANGES IN THE ENERGY STORED IN EACH CELL, THE DEPTH OF DISCHARGE, AND THE EFFICIENCY OF THE dc/dc CONVERTER

Number of Cells in Battery	Converter Efficiency*1 (%)	Depth of Discharge (%)	Total Energy (Wh)	Cell Energy (Wh)	Cell Energy Density (Wh/kg)			
					Standard Plates		40 Mil Plates	
					Inconel	Be-Nickel	Inconel	Be-Nickel
4	79.6	60	449	112	65.7	80.8	67.7	83.7
		70	385	96	64.7	80.0	66.7	82.8
		80	336	84	63.9	79.2	65.8	81.9
5	81.7	60	437	87.5	64.2	79.5	66.1	82.2
		70	375	75	63.0	78.4	64.8	81.0
		80	328	66	62.0	77.4	63.8	79.9
6	83.1	60	430	72	62.7	78.1	64.5	80.6
		70	368	61.5	61.4	76.8	63.1	79.2
		80	322	54	60.1	75.5	61.8	77.9
7	84.0	60	425	61	61.3	76.7	63.0	79.1
		70	364	52	59.7	75.1	61.4	77.5
		80	319	45.5	58.3	73.7	59.8	75.9
8	84.8	60	421	53	59.9	75.3	61.6	77.7
		70	361	45.5	58.3	73.7	59.8	75.9
		80	316	39.5	56.7	72.1	58.1	74.2

*1 The efficiency of a DC/DC converter as a function of input voltage is shown in Figure 2.

TABLE C-3 - CHARACTERISTICS OF A $Ni-H_2$ BATTERY WITH 5 CELLS, 66 Wh EACH

5 cells, 66 Wh each

DOD : 80%

	Standard Plates		40 Mil Plates	
	Inconel	Be-Nickel	Inconel	Be-Nickel
Cell Weight (g)	1066	854	1040	830
Cells				
Total Weight (g)	5330	4270	5200	4150
Energy Density (Wh/kg) (usable)	40.1	50.0	41.1	51.5
Battery				
Total Weight (g)	5860	4700	5720	4565
Energy Density (Wh/kg)	36.6	45.7	37.6	47.0
System				
Total Weight (g)	6560	5400	6420	5265
Energy Density (Wh/kg)	32.6	39.6	33.3	40.6

TABLE C-4 - CHARACTERISTICS OF A $Ni-H_2$

BATTERY WITH 6 CELLS, 66 Wh EACH

DOD : 65%

	Standard Plates		40 Mil Plates	
	Inconel	Be-Nickel	Inconel	Be-Nickel
Cell Weight (g)	1066	854	1040	830
Cells				
Total Weight (g)	6396	5134	6240	4980
Energy Density (Wh/kg)	33.4	41.7	34.3	43.0
Battery				
Total Weight (g)	7060	5650	6865	5480
Energy Density (Wh/kg)	30.3	37.9	31.2	39.1
System				
Total Weight (g)	7760	6350	7565	6180
Energy Density (Wh/kg)	27.6	33.7	28.3	34.6

TABLE C-5 - CHARACTERISTICS OF A $Ni-H_2$

BATTERY WITH 8 CELLS, 45.5 Wh EACH

DOD : 70%

	Standard Plates		40 Mil Plates	
	Inconel	Be-Nickel	Inconel	Be-Nickel
Cell Weight (g)	780	615	760	600
Cells				
Total Weight (g)	6240	4920	6080	4800
Energy Density (Wh/kg)	34.4	43.5	35.2	44.6
Battery				
Total Weight (g)	6865	5410	6690	5280
Energy Density (Wh/kg)	31.1	39.5	32.0	40.5
System				
Total Weight (g)	7565	6110	7390	5980
Energy Density (Wh/kg)	27.9	35.0	29.0	35.8

TABLE C-6 - CHARACTERISTICS OF A $Ni-H_2$ BATTERY

WITH 16 CELLS, 24 Wh EACH

DOD : 70%

	Standard Plates		40 Mil Plates	
	Inconel	Be-Nickel	Inconel	Be-Nickel
Cell Weight (g)	483	370	465	354
Cells				
Total Weight (g)	7728	5920	7440	5664
Energy Density (Wh/kg)	27.7	36.2	28.8	37.8
Battery				
Total Weight (g)	8500	6510	8185	6230
Energy Density (Wh/kg)	25.2	32.8	26.2	34.4
System				
Total Weight (g)	9200	7210	8885	6930
Energy Density (Wh/kg)	23.2	29.7	24.1	30.9

TABLE C-7 - WEIGHT BREAKDOWN OF TWO 66 Wh

CELL CONFIGURATIONS

Wt Breakdown, 66 Wh Cell, Inconel Can:

Diameter 8.9 cm	Height 20.6 cm
Nickel plates	461 g
Separators	9 g
Hydrogen electrodes	44 g
Screens	16 g
Electrolyte	115 g
Endplates	34 g
Tie rod	11 g
Tabs	38 g
	+
Stock total	728 g
Pressure Vessel	306 g
Feedthrough	32 g
	+
Cell total	1066 g
Energy Density	$\frac{66}{1.066} = 62$ Wh/kg

Wt Breakdown, 66 Wh Cell, Beryllium-nickel Can:

Stack	728 g
Pressure vessel	94 g
Feedthrough	32 g
	+
Cell total	854 g
Energy Density	$\frac{66}{.854} = 77.4$ Wh/kg

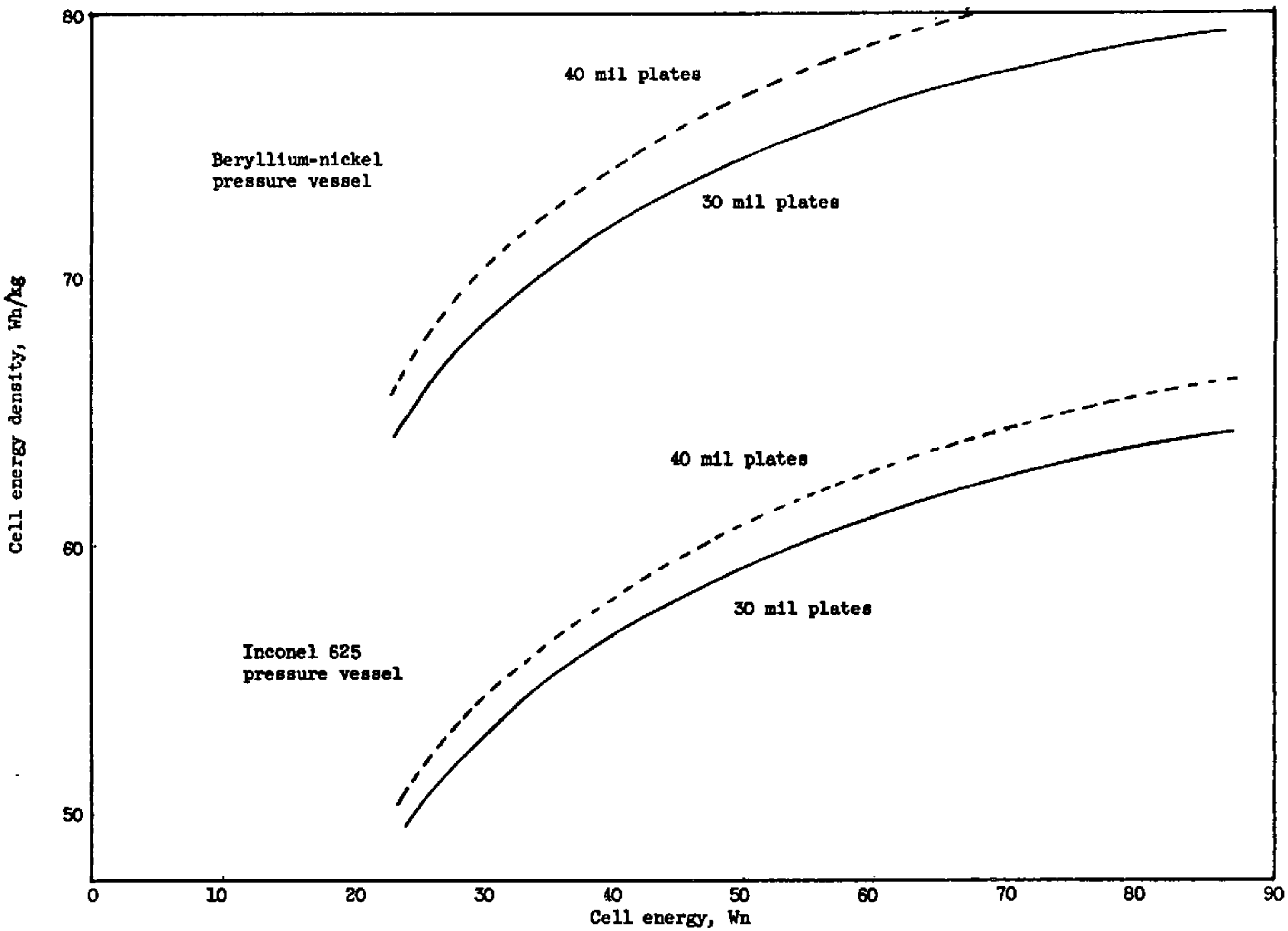

Figure C-1. - Cell energy density at 100 percent depth of discharge.

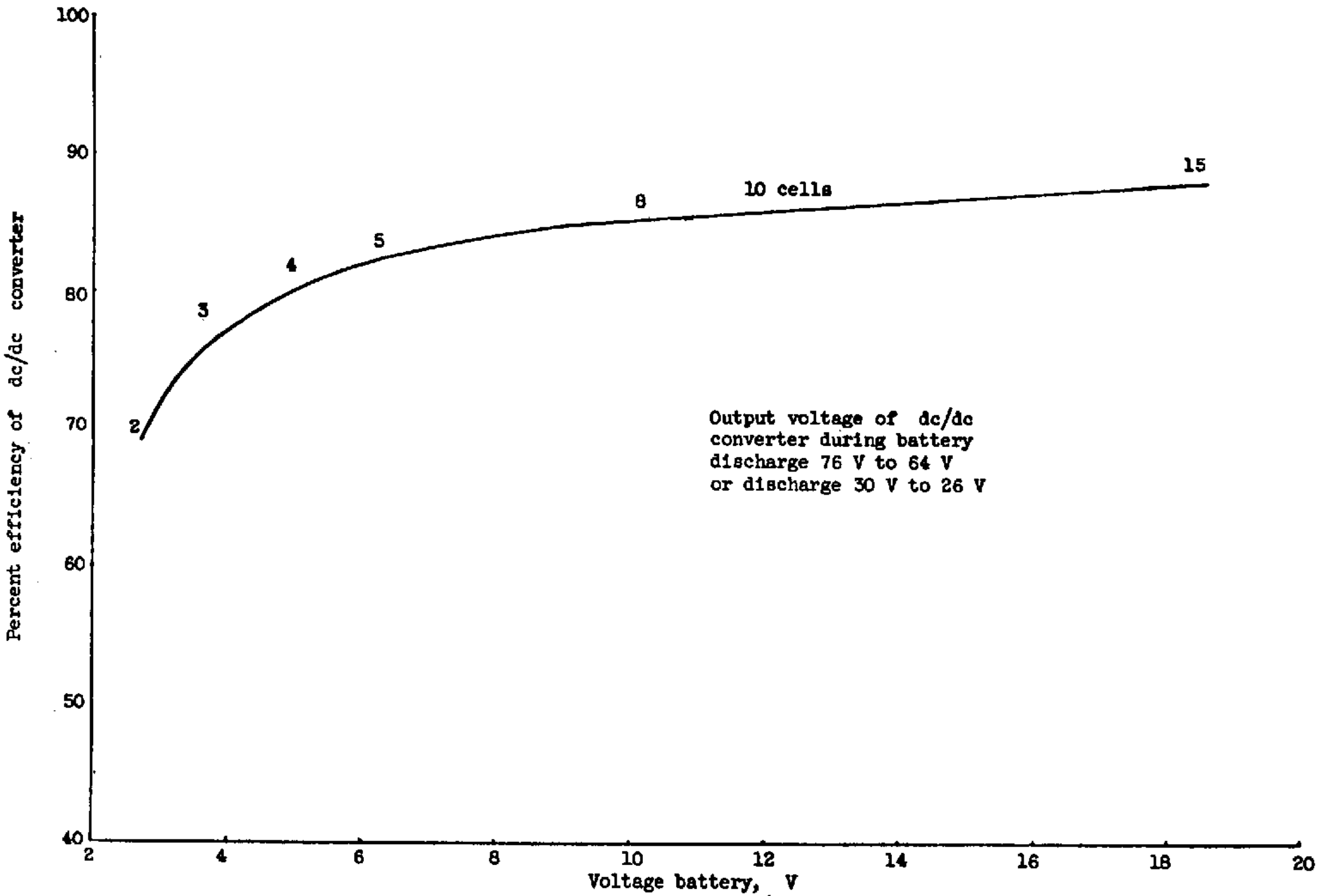

Figure C-2. - Efficiency of dc/dc converter versus battery voltage.

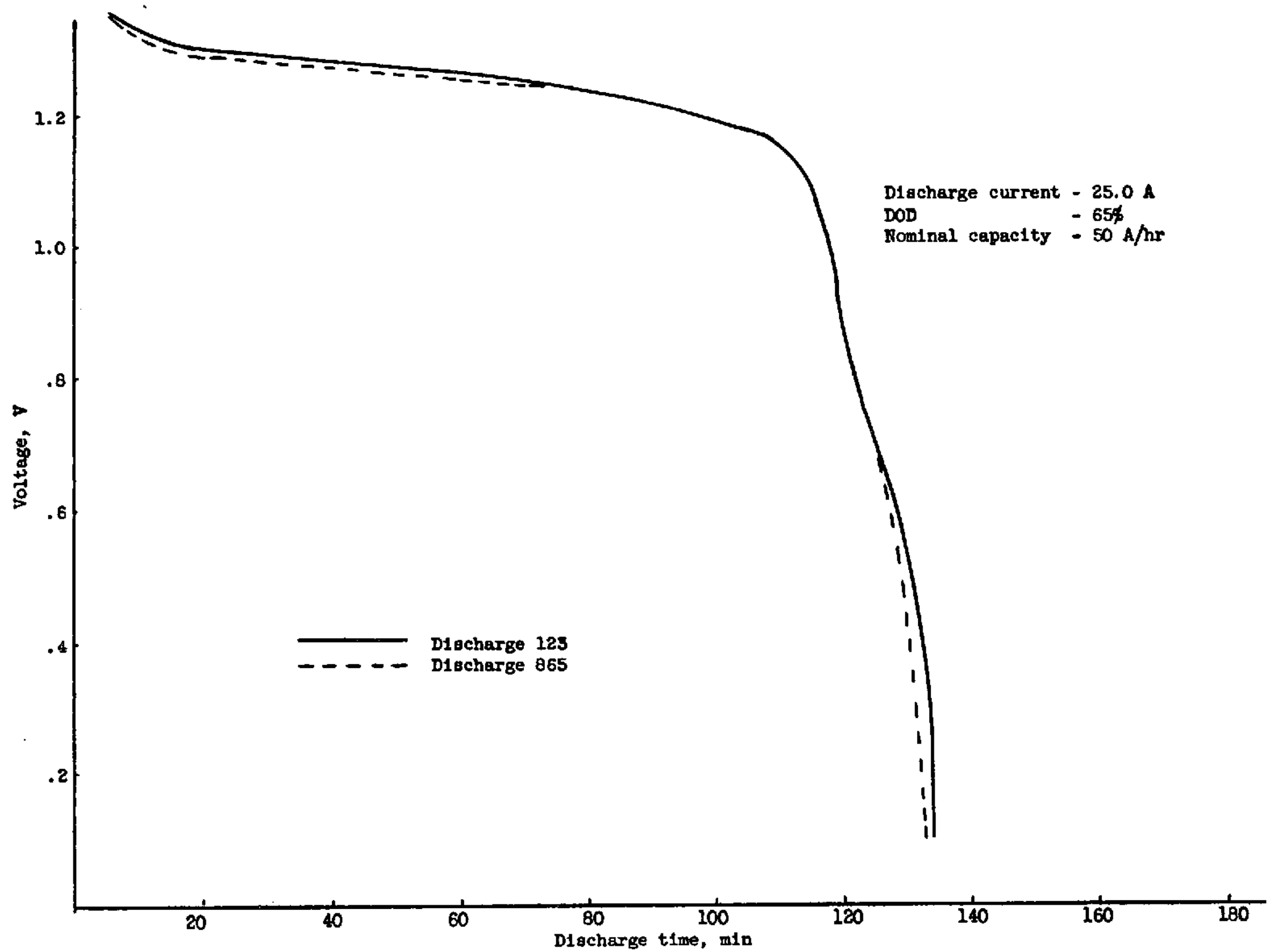

Figure C-3. - Discharge characteristic of Ni-H_2 cell.

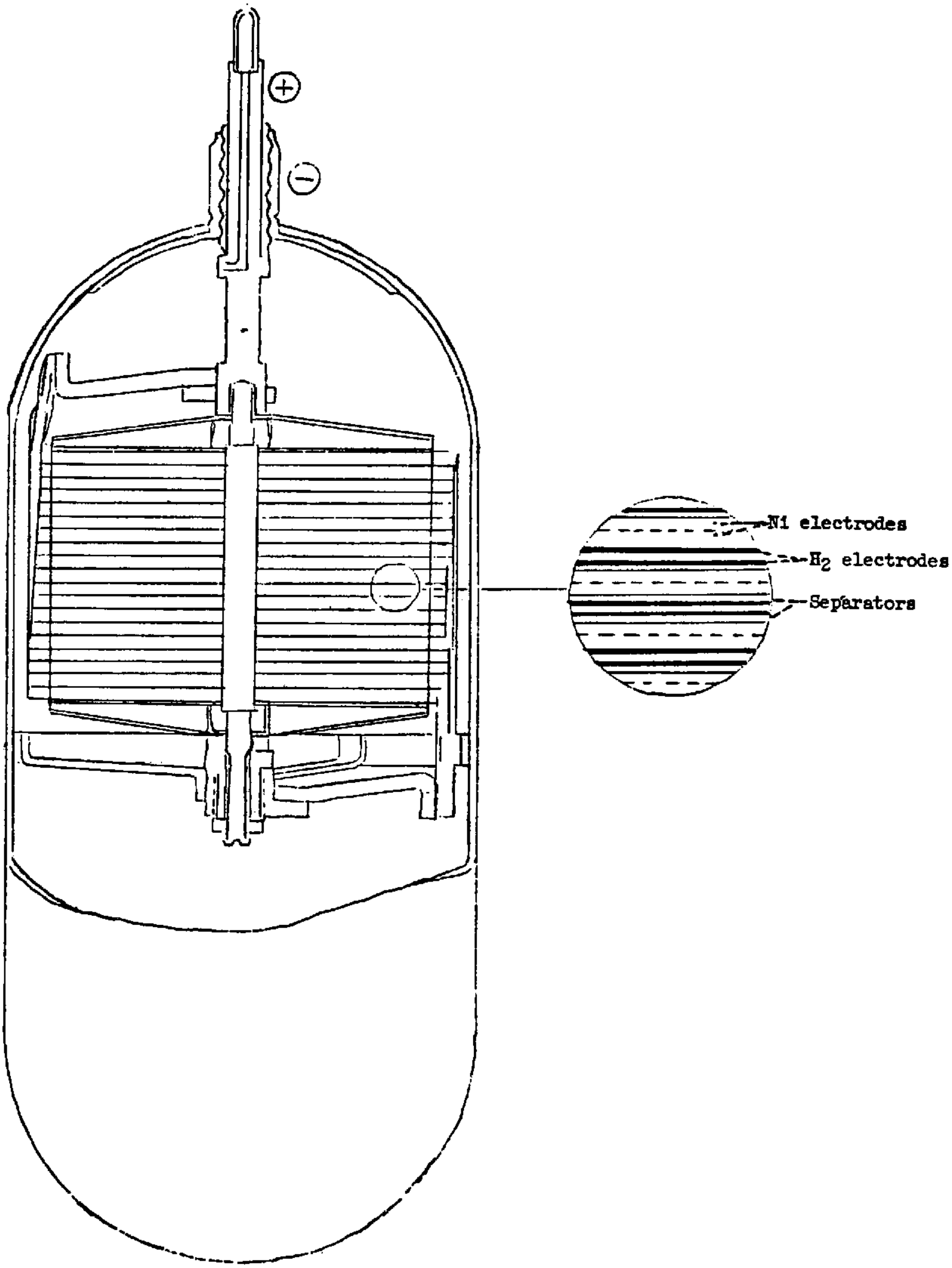

Figure C-4 - Ni/H_2 66 watt-hour cell (scale: 1:1).

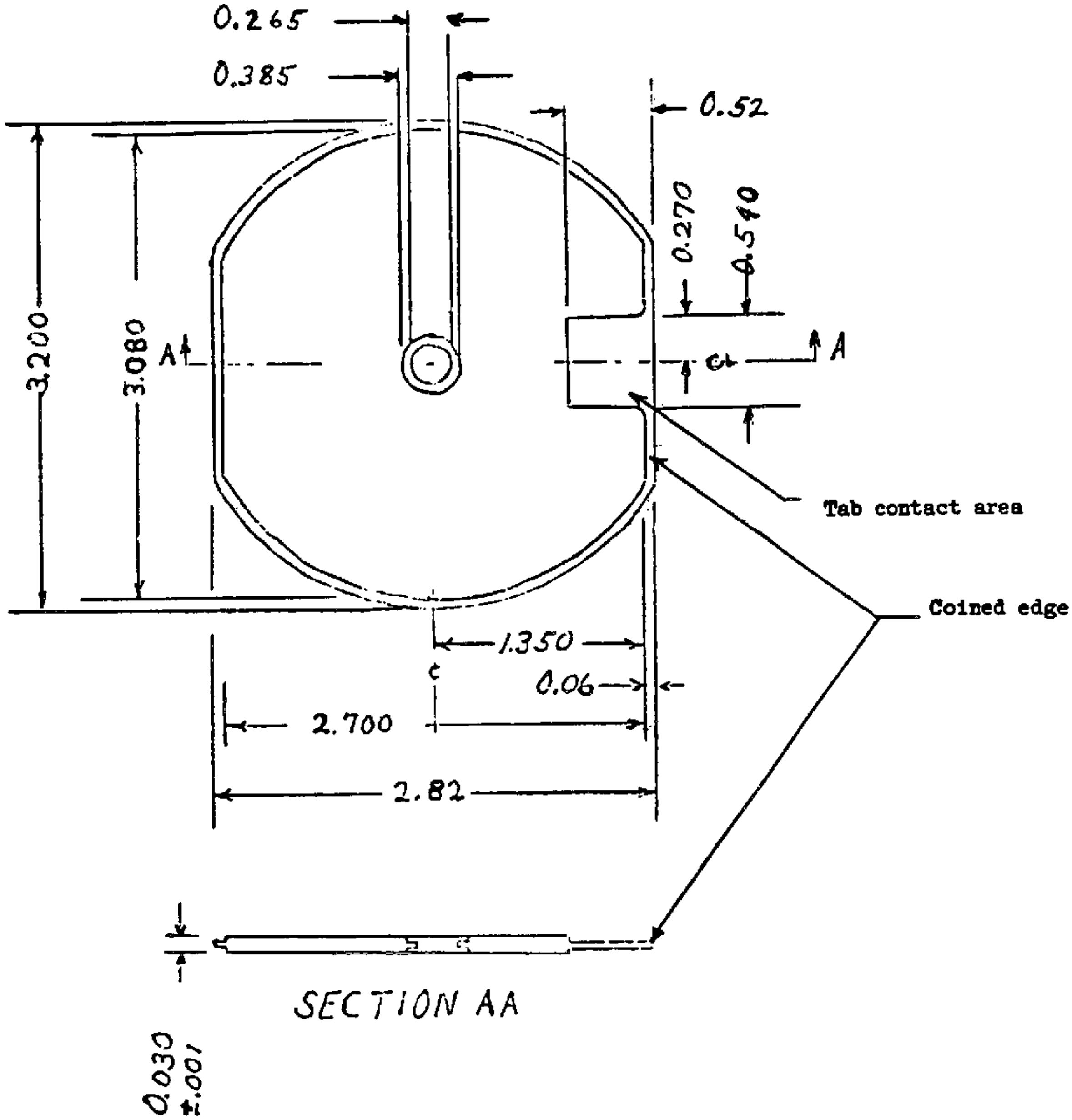

Figure C-5 - Nickel plate for Ni/H_2 cells (scale: 1:1).

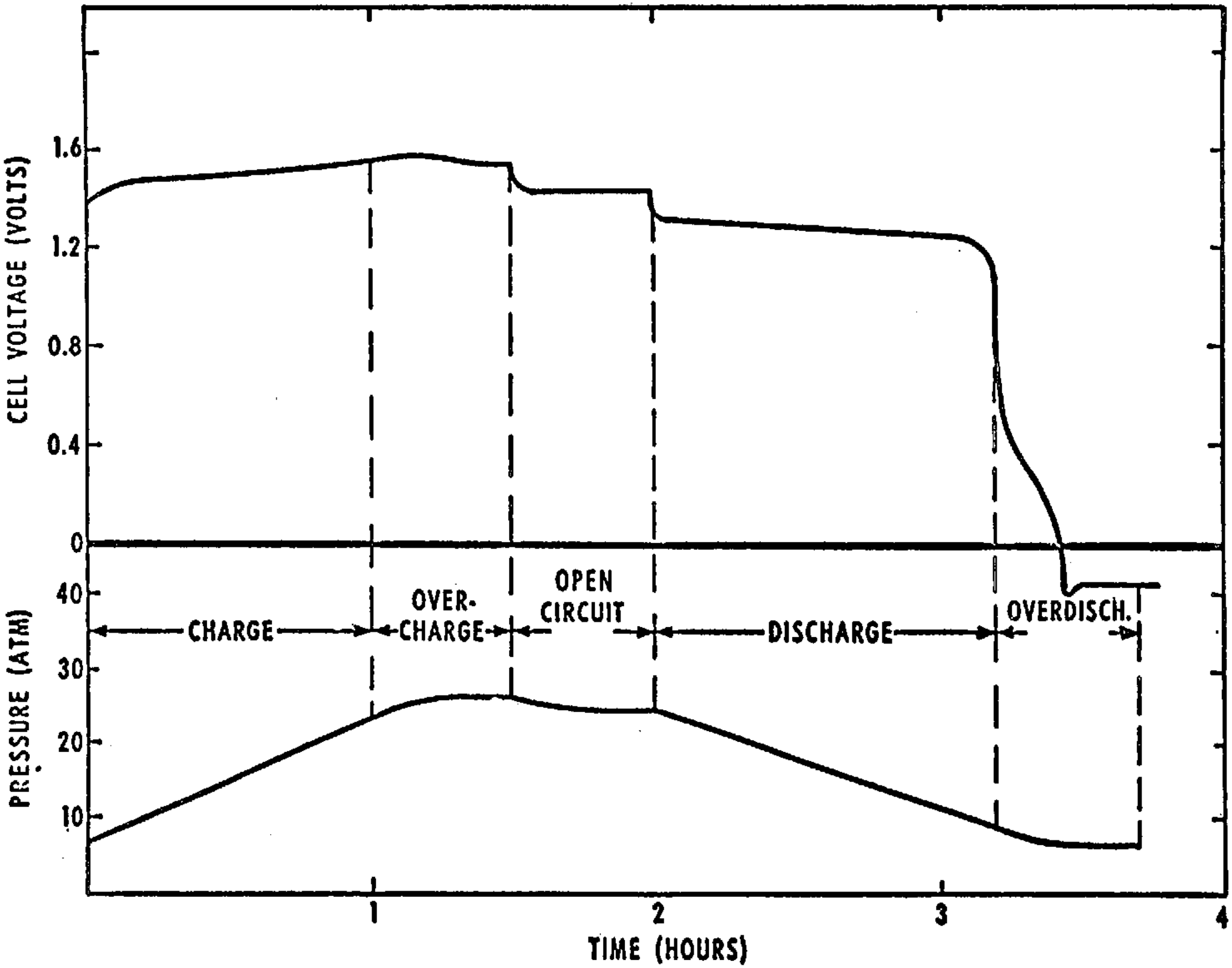

Figure C-6 - Charge-discharge characteristics of a single Ni-H_2 cell

Milestone Chart

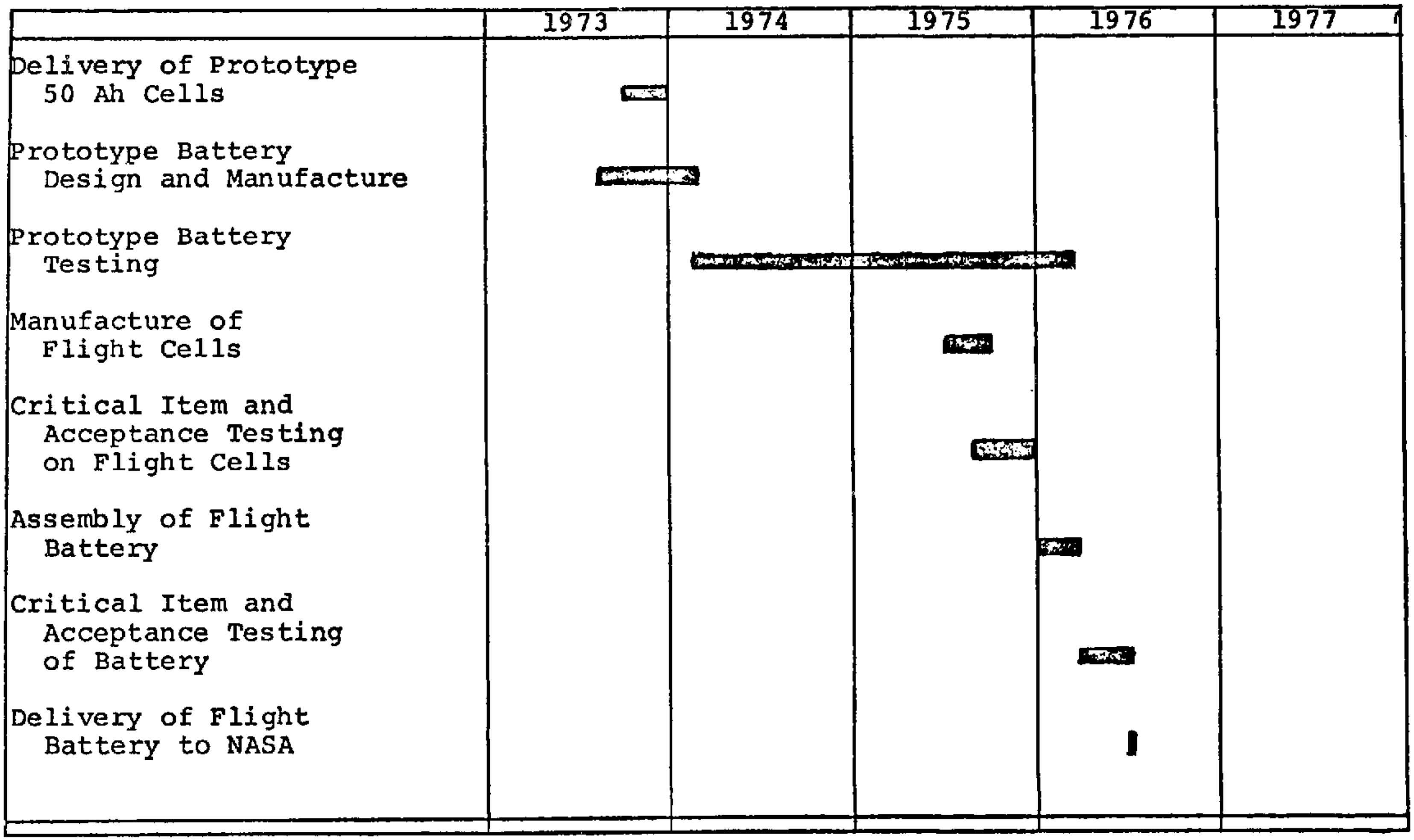

Figure C-7 - Development schedule for the 5-cell SERT C battery

25-FOOT-DIAMETER TANK

Description

The 25-foot tank is designed primarily for testing thrustors that employ condensable propellants. It is constructed from a 9/16-inch-thick steel clad material. The interior layer consists of 1/8-inch 304 stainless steel, while the outer layer is mild steel. The arrangement of components, the major dimensions, the details of the vacuum feed-through panels, and other related information of the 25-foot-diameter tank are shown in figure 3. Thrustors may be tested individually or in arrays; the primary limitations are mounting space and/or propellant condensing capacity.

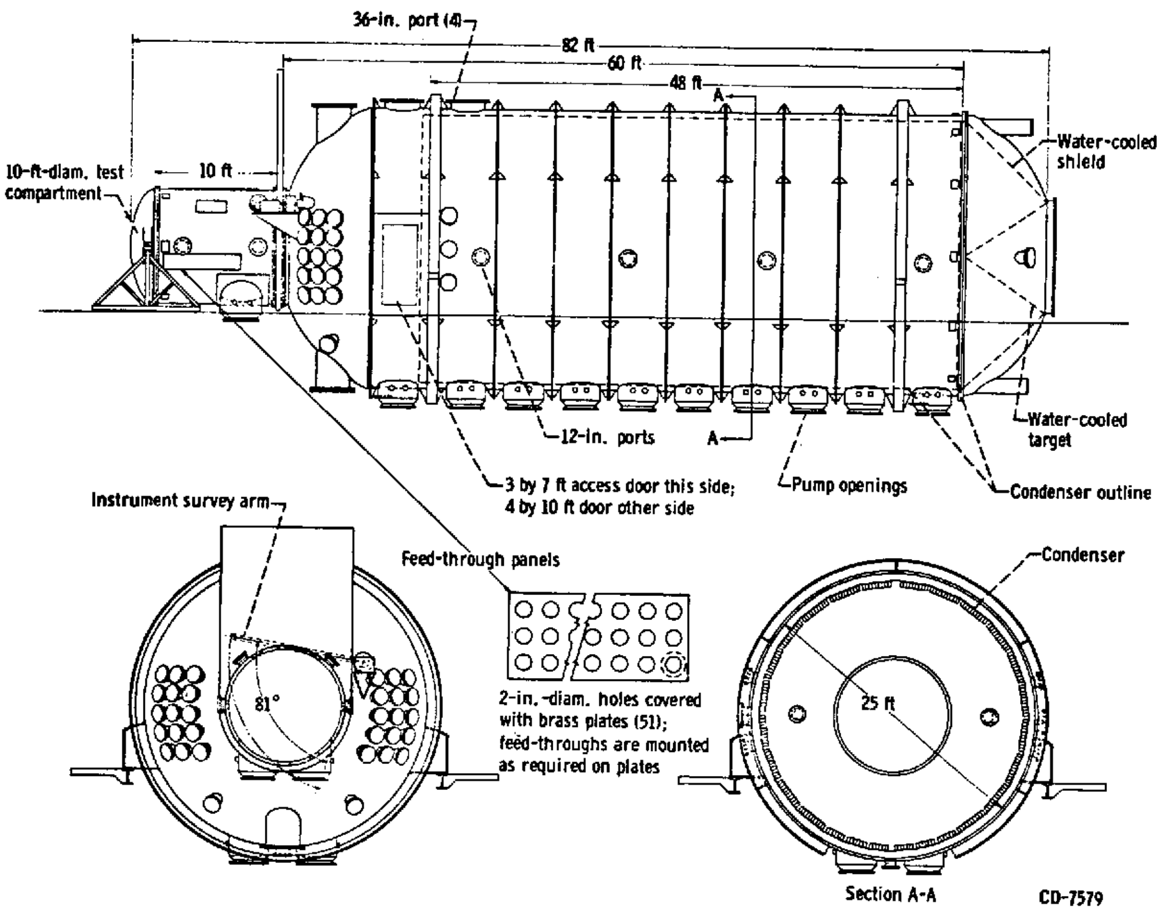

Figure 3. - 25-Foot tank layout.

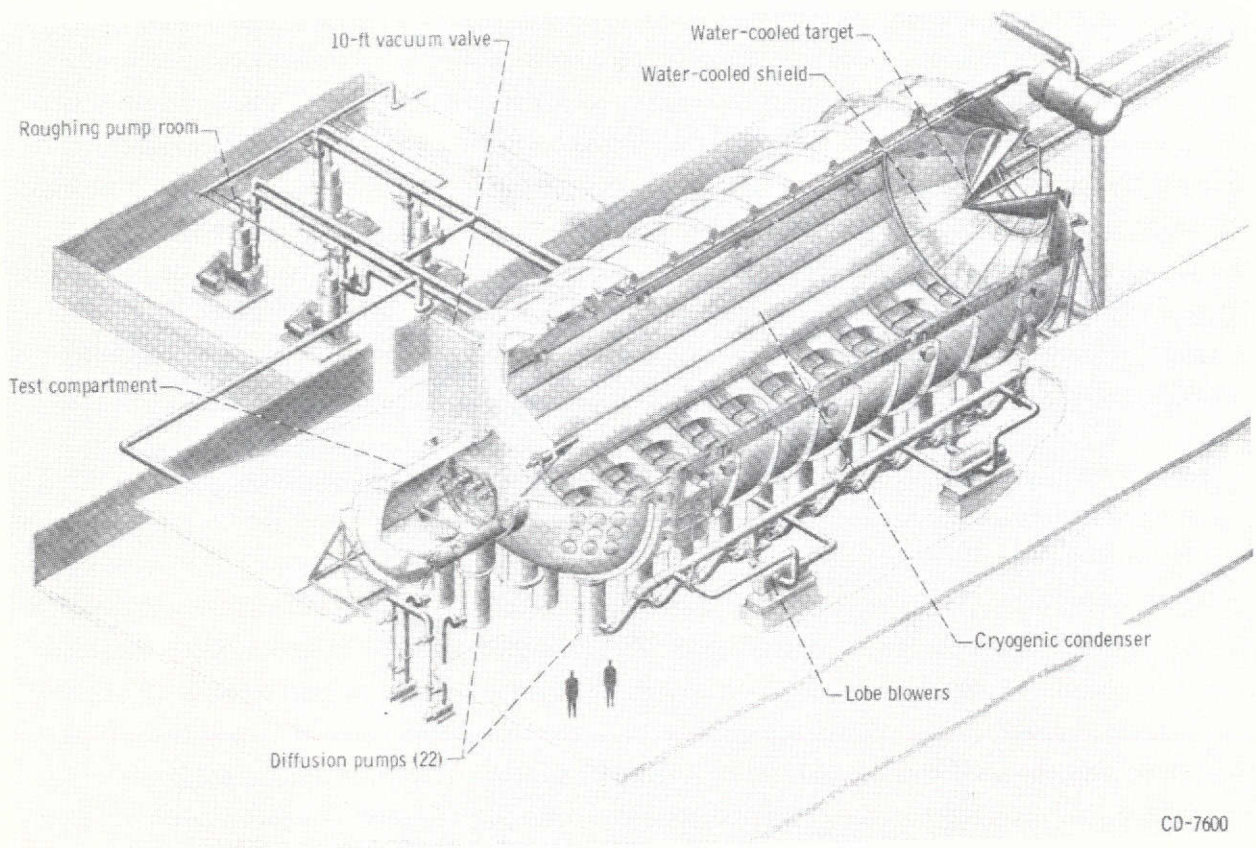

Figure 4. - Cutaway view of 25-foot-diameter tank.

The condenser system, which is designed to absorb up to 1 megawatt of power, is composed of three elements: the cryogenic condenser, the water-cooled target, and the water-cooled shield. These components are shown in the cutaway view of the tank (fig. 4). The cryogenic condenser is a copper honeycomb cooled by liquid nitrogen. This portion of the condenser can absorb up to 0.1 megawatt. The water-cooled target is designed to absorb 0.9 megawatt. The target is a conical-shaped stainless-steel double-walled structure, 8 feet from the bottom to the apex and 10 feet in diameter. Water is introduced at the apex of the target and is conducted through a helical passage to a manifold at the base of the target. In addition to the target, a water-cooled shield protects the interior surface of the 25-foot head. The shield is fabricated in pie-shaped segments from embossed stainless-steel plates.

The test compartment is separated from the main tank by a 10-foot-diameter vacuum gate valve of stainless-steel construction. The valve disk is raised and lowered by a counterbalanced chain, which is operated by an electric motor mounted exterior to the valve body with the drive shaft going through a rotating vacuum seal. The gate disk with two O-rings vacuum seals in one direction. To accomplish the sealing of the disk against the O-ring sealing surface, eight pneumatically operated pistons are used to hold the disk in place until the differential pressure across the valve is sufficient to seal it. Dry nitrogen gas is used to activate the pistons. The time required to raise or lower the disk is 5 minutes. The utility of the tank is increased by the capability of isolating the test compartment because thrustor changes can be made without warming the condenser or bringing the main tank to atmosphere.

Pumping System

The tank is equipped with a vacuum system to simulate space pressure conditions with normal testing in the 10^{-7} torr range. The pumping system is essentially identical for each tank and is shown for the 25-foot tank on figure 4. Air is pumped out of each tank through twenty 32-inch diffusion pumps into four lobe-type blowers installed in parallel. Two additional diffusion pumps are installed on the test compartment of the 25-foot tank. Following the blowers are four rotating piston-type roughing pumps. During the pumpdown cycle, the blowers are bypassed until the roughing pumps have lowered the system pressure to 15 torr. The blowers and diffusion pumps are then activated, and by the time the system reaches 2×10^{-3} torr the diffusion pumps have become effective. All eight roughing pumps can be used on one tank if desired. The tanks, containing no test apparatus, have been evacuated to 6×10^{-6} torr in less than 2 hours. Backstreaming of

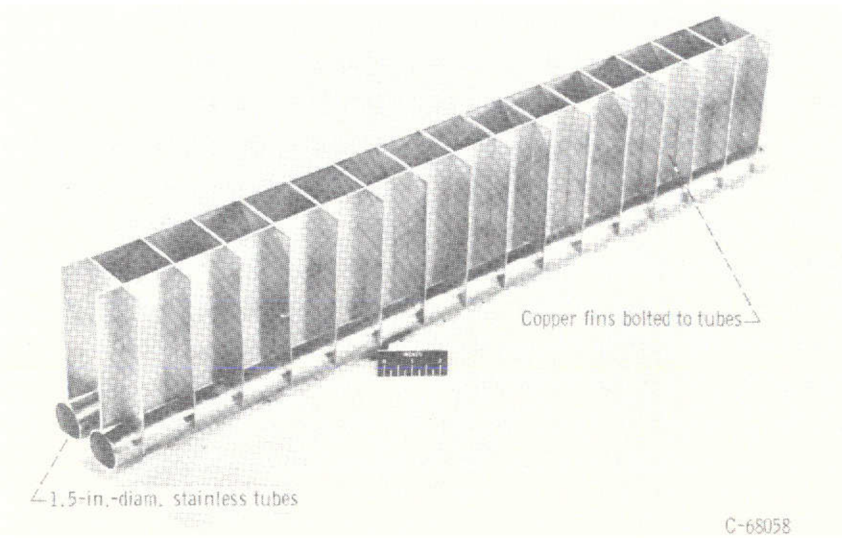

Figure 5. - Section of honeycomb condenser.

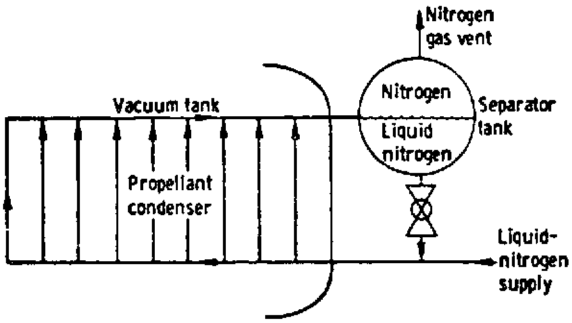

Figure 6. - Flow diagram of 25-foot-diameter tank condenser system.

oil from the diffusion pumps into the tanks is reduced to negligible amounts by water-cooled caps on top of the diffusion pump jet assembly and by liquid-nitrogen-cooled chevron-shaped baffles over the intakes of the pumps. Cold traps have been installed between the mechanical and diffusion pumps on the 25-foot tank. These traps protect the mechanical pumps from condensable propellants.

Cryogenic pumping is also used in maintaining tank pressures during thrustor operation. A finned liquid-nitrogen-cooled condenser lines the major portion of the 25-foot tank. The prime function of this cold surface is to condense and trap propellant from the thrustor exhaust. The condenser consists of a rectangular copper honeycomb structure built on 1.5-inch-diameter stainless-steel tubes as shown in figure 5. There are 28 000 square feet of copper surface designed to operate between -300° and -280° F, depending on the thrustor power load. The honeycomb material is bolted to the stainless-steel tubing and may be replaced if eroded by the exhaust beam.

The liquid-nitrogen system furnishes nitrogen to the propellant condenser, oil-diffusion-pump traps, and foreline traps. In addition, the liquid-nitrogen system furnishes gaseous nitrogen to raise the tank pressure to atmospheric conditions. The nitrogen system has a 56 000-gallon storage Dewar that is insulated by vacuum and perlite. There are two 100-gallon-per-minute pumps to supply liquid nitrogen to the oil diffusion traps, foreline traps, and propellant condenser at 40 pounds per square inch gage pressure. The liquid-nitrogen system for the diffusion pump traps and foreline traps is operated with a back pressure of 25 pounds per square inch gage, which gives a single-phase flow. Two-phase flow occurs in the propellant condenser. The nitrogen passes through a separator tank where the liquid nitrogen is returned to the inlet and the gaseous nitrogen is vented to atmosphere as shown in the flow diagram (fig. 6).

Hot nitrogen is used to warm the propellant condenser. The condenser warmup system has a separate blower and heat exchanger. The liquid nitrogen is first removed from the propellant condenser by a 50-gallon-per-minute transfer pump and returned to the storage Dewar. The gaseous nitrogen is then circulated through the 126-kilowatt heat exchanger and through the condenser. As the condenser surfaces warm up and the gas pressure increases, the gas is vented to the atmosphere to maintain a pressure of 15 pounds per square inch gage in the condenser. Another part of the nitrogen system contains a 500-kilowatt heater to vaporize liquid nitrogen. The vaporized nitrogen can be vented into the vacuum chamber to bring the chamber to atmospheric pressure.

Tank Pressure Instrumentation

Tank pressure is continuously monitored at a number of points by multiple Pirani and ionization gage systems. Each tank has an independent pressure monitoring system.

The Pirani gage system is a 24-channel (for each tank), pushbutton-operated unit, employing a single-channel Pirani gage box to provide power and readout for the system. Tube switching is accomplished by a 24-channel stepping switch arranged so that it automatically runs to the channel for which a pushbutton has been depressed. All channels have individual zero, and calibration controls are preset so that no adjustments are required when switching from tube to tube.

The multiple ionization gage systems, like the Pirani systems, are 24-channel devices. The selection of an ionization tube is determined by depressing the appropriate pushbutton. Bayard-Alpert tubes with iridium filaments are used with the system. The ionization system has eight rotary switches on the panel to power the degassing grids in the tubes (3 tubes/switch). The gage system will record continuously, on a strip chart recorder, the output of the tube being read. The scale on the recorder is in decades from 1×10^{-3} to 1×10^{-8}, and the recorder pen is shifted as the gage box switches range. The gage box will shift its range automatically as the pressure varies in the tank. Between readings, the gage filaments are kept at a reduced temperature to prevent condensation of vapor on the walls of the tube and absorption of gases by the tube elements.

Data from the thrustor ion beam can be obtained from sensing instruments mounted on an instrument survey arm. A schematic of the water-cooled arm is shown in figure 7. It sweeps an arc in a plane perpendicular to the beam. The motion of the arm may be stopped and reversed at any point in the 81° sweep angle. The speed can be varied from

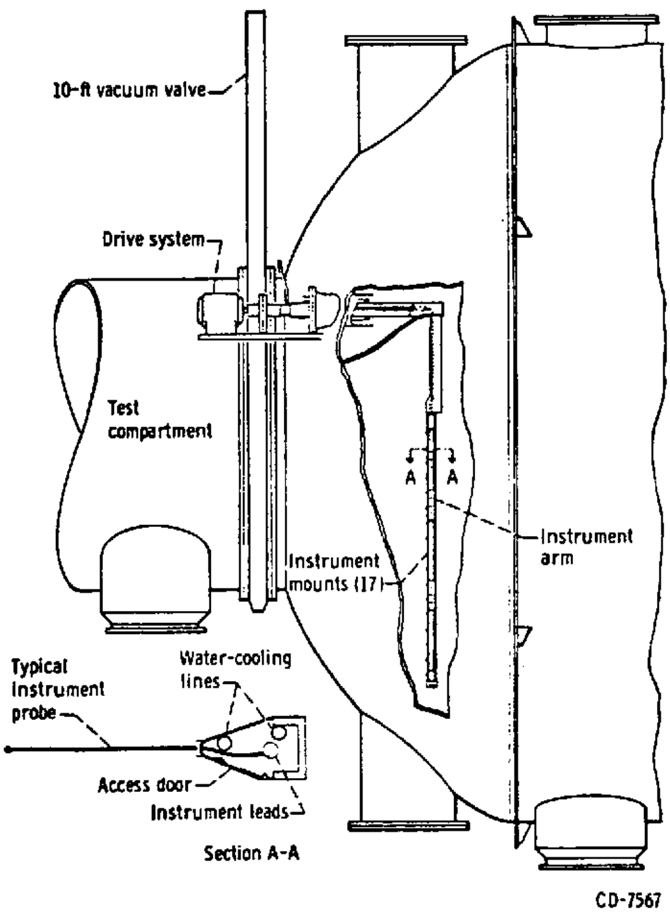

Figure 7. - Instrument survey arm.

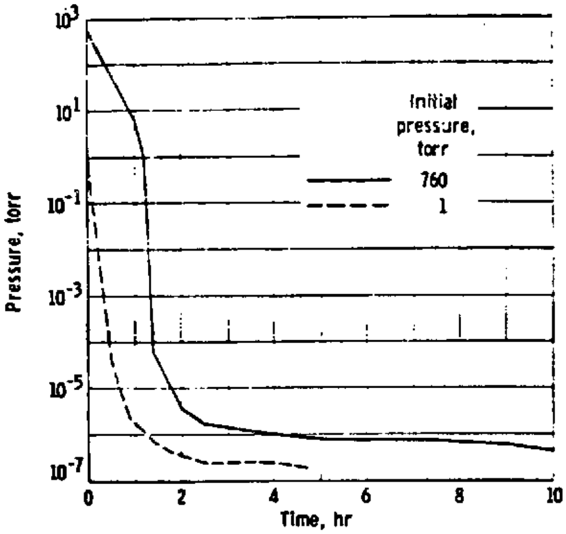

Figure 8. - Pumpdown rate with 20 oil diffusion pumps, liquid-nitrogen-cooled traps, and condenser. 25-Foot-diameter tank.

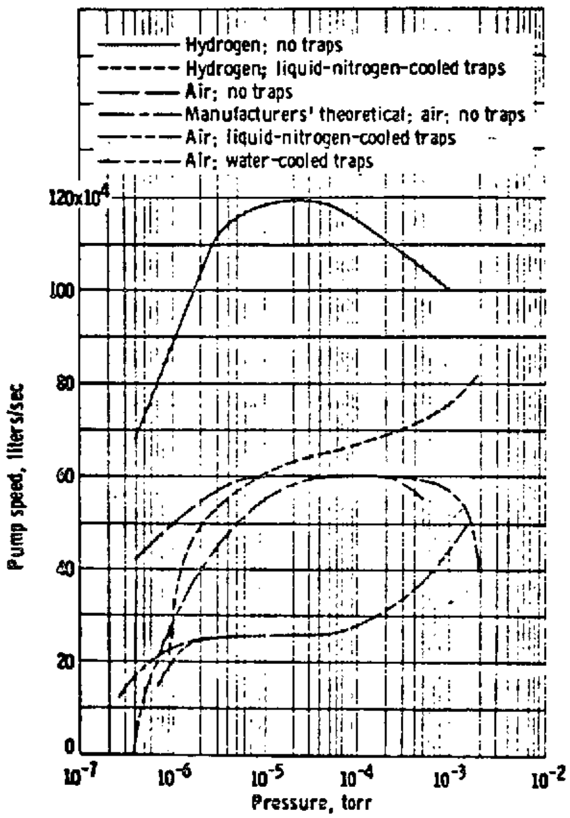

Figure 9. - Pumping speed data for 25-foot tank.

4 seconds to 4 minutes per sweep.

Seventeen instrument mounting pads are available along the leading edge of the arm at 6-inch intervals. The pads consist of 3/4-inch-diameter disks with a center tapped hole. Each disk is welded directly to the arm structure. Leads from the sensing device are threaded through the tapped hole prior to instrument mounting. The leads are attached to the inside of the hollow arm and conducted up to the end of the arm. The sensing lead harness is then brought out of the tank through a vacuum seal.

Tank Performance

The performance characteristics of the 25-foot-diameter tank have been determined with the pump traps at -320° F and the main tank condenser at both ambient temperature and -320° F. Calibrated flows of both air and hydrogen have been investigated.

The tank pumpdown rate with the condenser at -320° F is shown in figure 8. Two curves are presented: (1) starting at atmosphere after the tank had been open for several days, and (2) starting at 1 torr, the normal point for initiating pumping between phases of a test. These data were taken after the tank had been in use for approximately 2 years and the condenser was coated with mercury collected from ion-thrustor operation. The tank contained apparatus that included a 20-foot-diameter movable collector with drive system, a fixed 20-foot-diameter collector, an instrument survey arm, and several miscellaneous wiring harnesses.

Normally, model modifications are made

TABLE I. - PUMPING CHARACTERISTICS OF 25-FOOT-

DIAMETER TANK

Condition	Tank condenser, -320° F; pump traps, -320° F	Tank condenser, 60° F; pump traps, -320° F
Maximum system pump speed for air, liters/sec	----	[a]250 000
Maximum system pump speed for hydrogen, liters/sec	----	[b]660 000
Ultimate pressure in new clean tank, torr	6.5×10^{-9}	1.4×10^{-7}
Time required to obtain ultimate pressure, hr	35.0	20.0
Ultimate pressure in tank (after 2 yr use), torr	9.0×10^{-9}	----
Time required to obtain ultimate pressure, hr	26.0	----
Time required to open tank to atmosphere, hr	3.0	----
Time required to open test compartment to atmosphere, hr	1.8	1.8

[a]At 10^{-4} to 10^{-6} torr.
[b]At 10^{-4} torr.

by closing the 10-foot valve and bringing only the test compartment to atmospheric pressure. Then, the compartment is closed and pumping initiated. With three 20-centimeter-diameter ion thrustors installed the pressure reached 2×10^{-5} torr in 55 minutes. At this time the 10-foot-diameter valve was opened. Thirty minutes later the tank pressure reached 2.0×10^{-6} torr. Typical pumping speed data are shown in figure 9. Pertinent characteristics for the 25-foot tank, including the calibrated air and hydrogen flows, are given in table I.

After a test run, 1.8 hours are required to cool the diffusion pumps and bleed the test compartment to atmospheric pressure. Additional time may be required to remove toxic vapors prior to entering.

A 65-hour continuous run was made with a 50-centimeter-diameter electron-bombardment thrustor. A beam power of 22.5 kilowatts was maintained during this period. Mercury was the propellant. The test

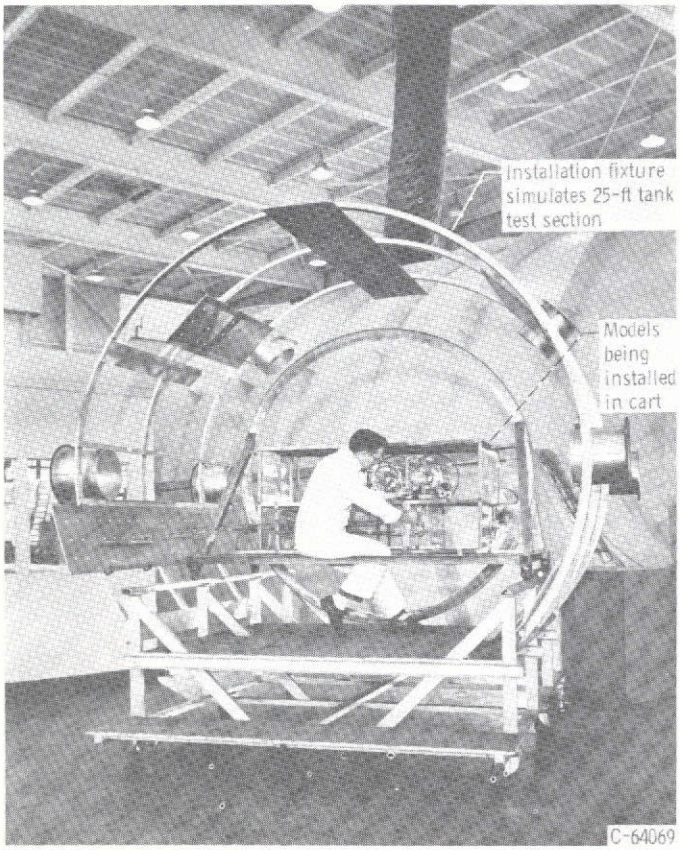

Figure 10. - Thrustor installation in cart and installation fixture.

compartment pressure was 2×10^{-6} torr at the start of the test and reached 7×10^{-7} torr after 30 hours. At 65 hours the pressure was 6×10^{-7} torr. The tank had accumulated several hundred hours of mercury thrustor time when these data were taken.

Thrustor Installation

Prior to testing, thrustors are mounted on a thrustor cart in the shop. Power, instrument, cooling, and heating leads are fitted from the cart to a fixture that duplicates the tank feed-through positions (fig. 10). When the tank is available for testing, the thrustor cart is transferred to the tank test compartment. The cart is rolled into the operating position on a set of rails. Thrustor leads are connected, the compartment door closed, and the system checked out. The test compartment is then evacuated by a separate pumping system. After the 10-foot valve is opened the apparatus is ready to be tested.

Figure 11 gives an indication of the mounting space required by an array of nine electron-bombardment thrustors. Thirty-seven thrustors of this size (16-in. o.d.) could be installed.

15-FOOT-DIAMETER TANK

Description

The 15-foot-diameter tank is 63 feet long and was designed primarily for environmental testing of space packages and thrustors. It is constructed from the same material as the 25-foot tank. One end of the tank is readily removable to allow large

Figure 11. - Nine ion thrustor array.

units to be installed conveniently for testing. The layout of the tank is shown in figure 12. Power, gas, and liquids may be supplied to the test apparatus through lines passing through any of the eight sets of feed-through panels.

Along the horizontal centerline of the tank are eleven 12-inch viewing ports. For ultraviolet applications quartz windows may be substituted for the standard glass units. The windows can be removed and exchanged for plates with instrumentation feed-throughs and other devices. One 24-inch-diameter and six 36-inch-diameter ports are available for special installation of equipment, services, and instrumentation as required. The size and location of the ports are shown in figure 12. All port openings are sealed with buna N O-rings. The pumping system for the 15-foot tank is identical to the 25-foot tank. The condenser for the 15-foot tank is fabricated from copper sheet with integral tubes and is painted black. Nitrogen is furnished to the 15-foot tank from the same source that supplies the 25-foot tank. To operate all systems in the 15- and 25-foot tanks at maximum capacity, 830 gallons per hour of liquid nitrogen are required.

Tank Performance

The performance characteristics of the 15-foot-diameter tank have been determined under several conditions and are tabulated in table II. The pumping speeds for both tanks are identical, as previously mentioned, and are plotted in figure 9.

After a test with the condenser at ambient temperature and the pump traps at 60° F, approximately 60 minutes are required to cool the diffusion pumps and vent the tank to atmosphere. If the condenser and traps are at -320° F, 3 hours are required to bring the tank to atmospheric pressure. Additional time is required for ventilating the tank

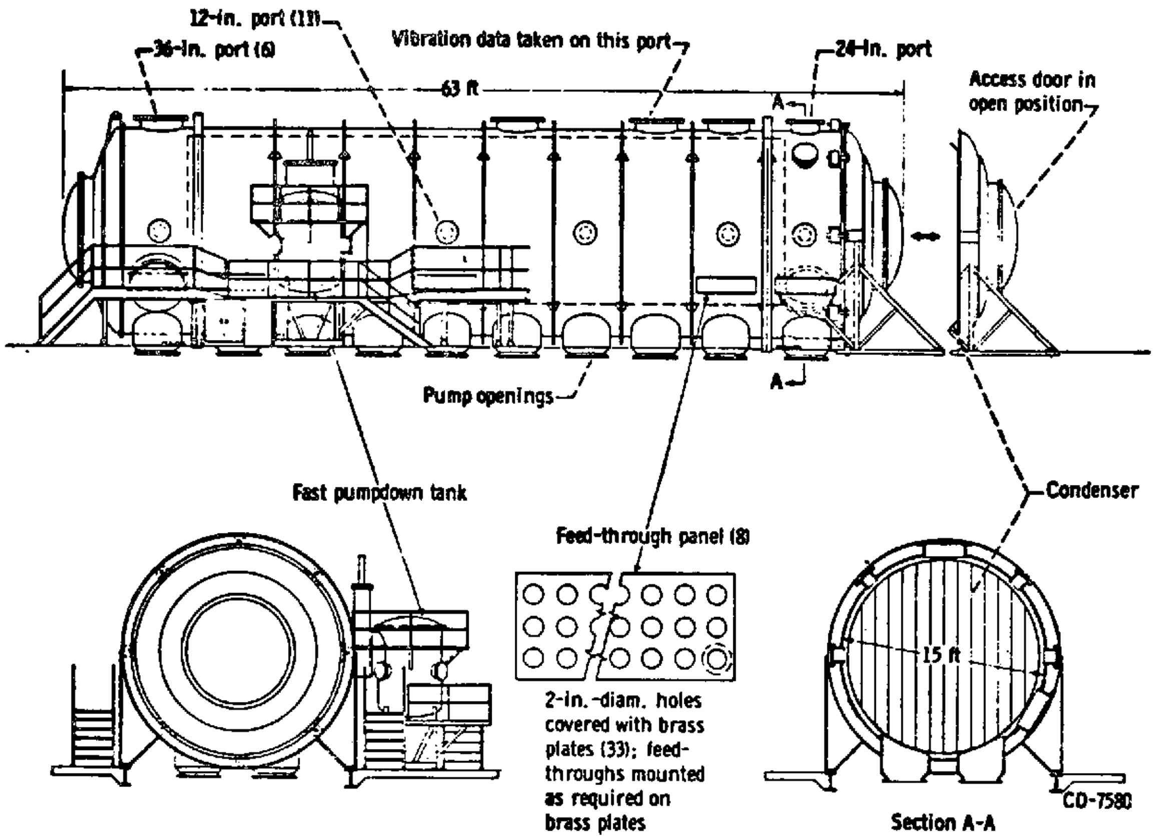

Figure 12. - 15-Foot tank layout.

TABLE II. - PERFORMANCE CHARACTERISTICS OF 15-FOOT-DIAMETER TANK

Condition	Tank condenser, 70° F; pump traps, -320° F	Tank condenser, 70° F; pump traps, 60[b] F	Tank condenser and pump traps, -320° F
Maximum system pump speed for air, liters/sec	[a]250 000	[a]250 000	------
Maximum system pump speed for hydrogen, liters sec	[b]660 000	------	------
Ultimate pressure, torr	1.3×10^{-8}	4.0×10^{-7}	1.0×10^{-8}
Time required to obtain the ultimate pressure, hr	17.0	11.0	17
Time required to enter tank after test, hr	2	1	~3

[a]At 10^{-4} to 10^{-6} torr.
[b]At 10^{-4} torr.

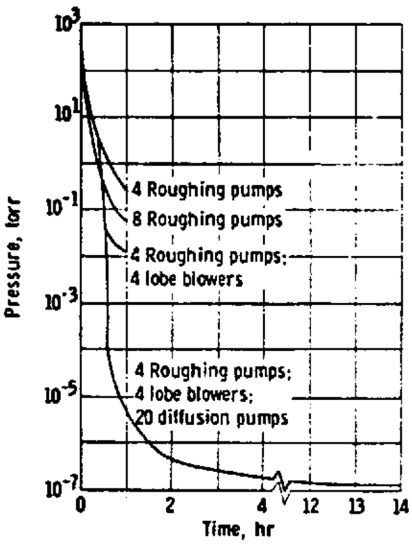

Figure 13. - Pumpdown rate for 15-foot tank.

prior to entering if toxic vapors are generated during the test.

The tank pumpdown rates for four pumping configurations are shown in figure 13. For the configurations utilizing both diffusion and mechanical pumps, the roughing pumps were activated at time zero. The lobe blowers were turned on in approximately 17 minutes at a pressure of 15 torr. The heaters on the diffusion pumps were energized approximately 10 minutes after closing the tank; the pumps became effective in 32 minutes. The main condenser and the pump traps were cooled to -320^{o} F at 10^{-6} torr. During this pumpdown the tank contained the following: the space electric rocket test (SERT) payload, a 30-kilowatt arc-jet thrustor, a 10-centimeter-diameter electron-bombardment thrustor plus the associated harnesses, an instrument probe, an 8- by 8-foot sheet-metal baffle, and a 3-foot conical water-cooled heat exchanger. After 11 hours, the pressure had dropped to 1.5×10^{-7} torr and remained essentially constant for 3 more hours when the run was concluded.

Another test series was run with the tank condenser and traps warm, the tank containing the equipment previously described. During one test, only four roughing pumps and four blowers were in operation. An ultimate pressure of 1.2×10^{-2} torr was obtained. Starting at atmospheric pressure the tank was pumped in 1 hour to 0.3 torr with four roughing pumps and to 8×10^{-2} torr with eight roughing pumps. These data are also shown in figure 13.

With a clean and empty tank, ultimate pressures of 4×10^{-7} and 1.3×10^{-8} torr were obtained with water-cooled and liquid-nitrogen-cooled traps, respectively.

Typical tests conducted in the 15-foot-diameter tank include the evaluation of a 1- and a 30-kilowatt arc jet and the SERT payload. The 1-kilowatt arc jet was tested with tank pressures near 8×10^{-5} torr. The results of this investigation are reported in references 4 and 5. During the 30-kilowatt arc-jet test a tank pressure of 0.85 torr was recorded with a hydrogen flow of 0.28 gram per second. The SERT test required activation of high-voltage systems, ignition of pyrotechnics, and ignition of ion thrustors during a typical tank run. Typical pressure fluctuations resulting from equipment and thrustor operation (maximum beam current was 250 ma) ranged from 6×10^{-6} to 8×10^{-6} torr. The operation of three pyrotechnic devices caused the pressure to decrease from 5×10^{-7} to 9×10^{-5} torr. The devices were ignited over a 5-second interval, and 17 seconds elapsed after the last firing before the tank pressure recovered.

An air bearing, which bleeds air into the tank for a known back pressure and flow

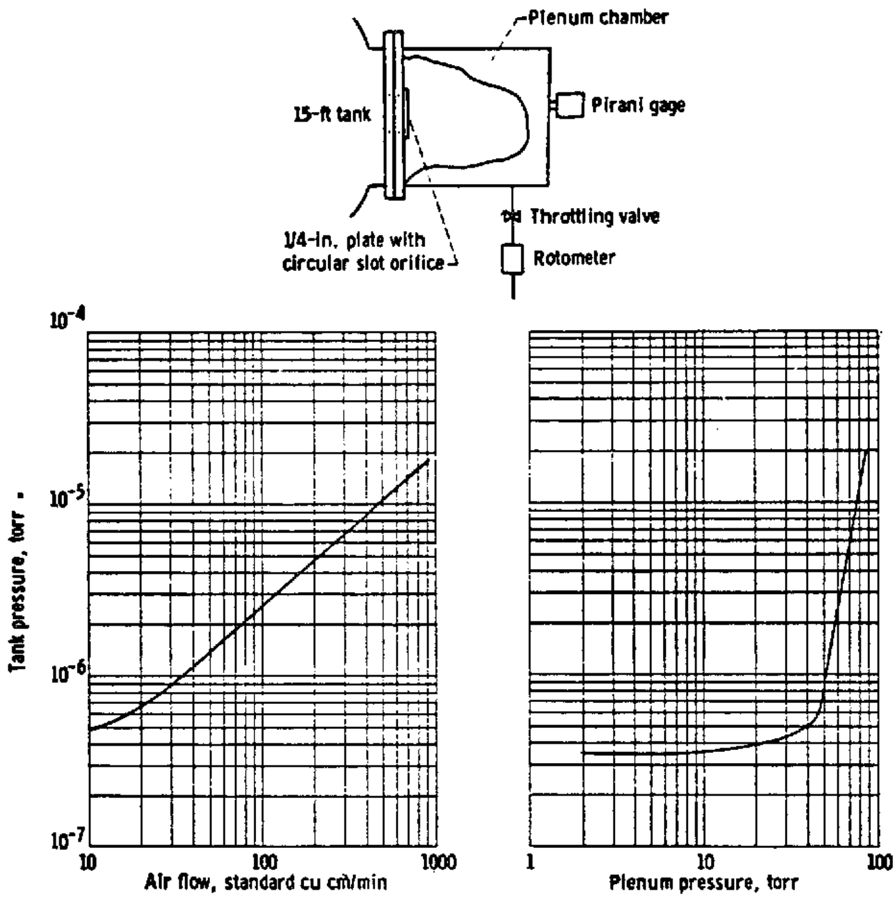

Figure 14. - Simulated air bearing leak.

rate, was required for a space payload test. To determine the effect of such a device, air was bled into the tank through a special calibration system shown in figure 14. Air flows into the tank through a 0.0005-inch circular slot, 2 inches in diameter, from a plenum chamber where the pressure can be controlled. The chamber was approximately 9 inches long, 6 inches in diameter, and mounted to a tank port. With the 15-foot tank at vacuum, the pressure in the plenum was controlled by throttling the quantity of supply air at atmospheric pressure. A rotometer was utilized for measuring the flow of supply air. Chamber pressure was measured with a Pirani gage. Data-indicated tank pressure, chamber pressure, and flow rates for this system are also shown in figure 14.

Tank Vibration Characteristics

Vibration data were taken on the 36-inch port located on the top centerline of the 15-foot tank shown in figure 12. Data were recorded with the tank at vacuum and all

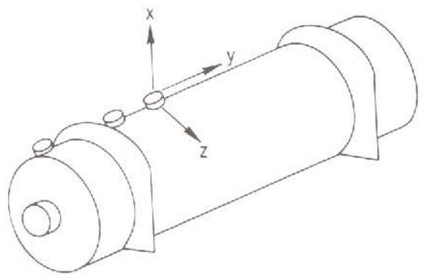

Figure 15. - Axis notation.

TABLE III. - VIBRATION DATA FOR

15-FOOT TANK

Axis	Frequency, cps	Displacement, in.	Maximum g load
x	3000	Below 10^{-7}	0.005
y	↓	↓	.12
z	↓	↓	.006

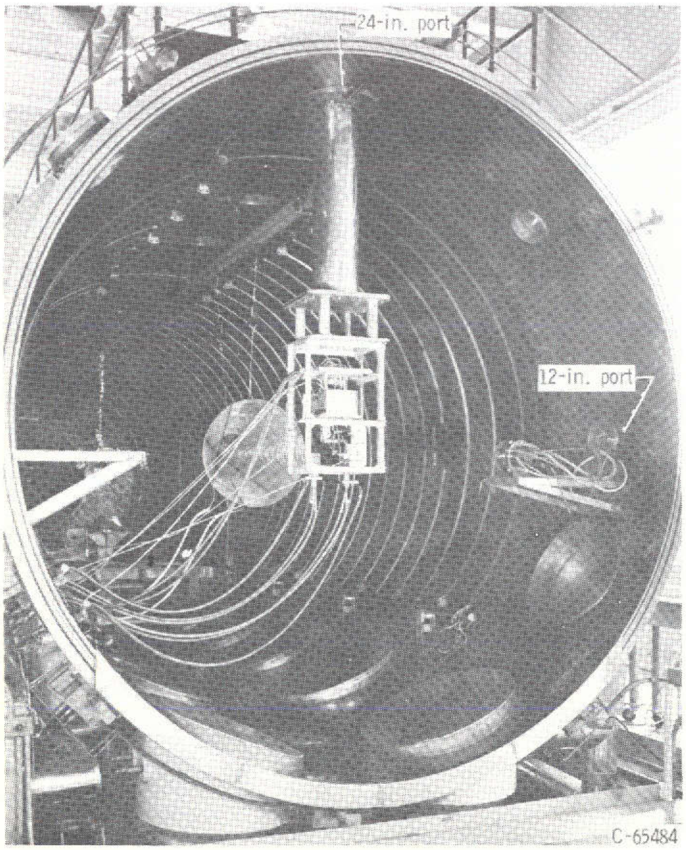

Figure 16. - Port-mounted thrustor.

pumps in operation to provide an order of magnitude indication of vibration conditions. For the design of ion-thrustor thrust stands or equipment where very precise information is required, vibration studies would be made in the exact location where the equipment would be installed. Measurements were recorded on the three axes shown in figure 15.

The vibration characteristics were detected with a piezoelectric accelerometer. The frequency range scanned was between 16 cps and 32 kcps. The maximum "g" condition occurred on each axis at approximately 3000 cps. At these maximum g conditions the displacement was less than 10^{-7} inch. Table III shows the maximum g loading for each axis and the corresponding frequency and displacement.

Model Installation

The 15-foot tank is designed so that models may be installed in any location to suit the requirement of the test. Large models may be handled in the tank with the assistance of an overhead monorail chain fall with a half-ton capacity. Models may be mounted from the walls or from one of the large port openings. A port-mounted model may be mounted on a port cover outside the tank. This has the advantage of providing a means of checking the mounting system prior to installation and simplicity of instal-

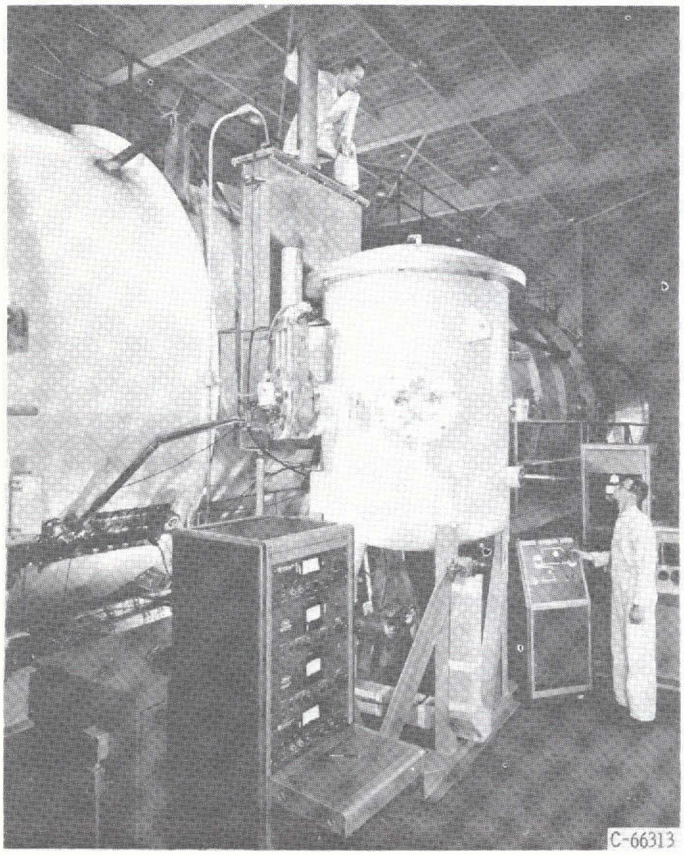

Figure 17. - Fast pumpdown tank.

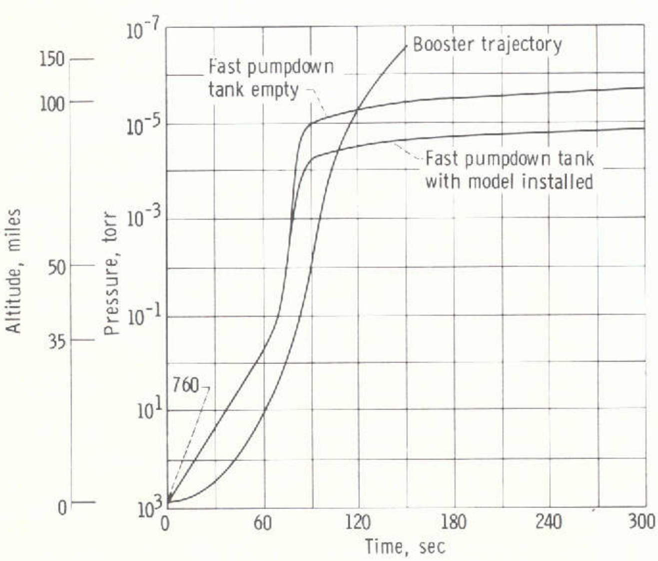

Figure 18. - Fast pumpdown tank performance data.

lation. A thrustor mounted from a 24-inch port and another mounted from the wall are shown in figure 16.

SPECIAL EQUIPMENT

Fast Pumpdown Tank

A fast pumpdown tank is available to simulate pressure conditions encountered in the early portion of a missile flight. This tank (fig. 17) is connected to the 15-foot-diameter tank by a 3-foot-diameter vacuum gate valve. It has an inside diameter of 46 inches and working height along the vertical axis of 70 inches. The top of the tank is removable for access to the model and for installation. Instrument, power, and service leads are fed into the tank through fittings mounted on three 12-inch ports on the side of the cylindrical tank. The tank pumpdown rate is compared with the pressure profile encountered during a booster flight in figure 18. Prior to the fast pumpdown tank operation, the 15-foot tank is evacuated and all pumps are kept in operation. The rotating piston pumping systems from either or both the 15- and 25-foot tanks are then used to reduce the pressure rapidly in the fast pumpdown tank to approximately 4 torr. The 3-foot valve is then opened to the 15-foot tank providing additional pumping. Condensable products are captured on the fast pumpdown tank

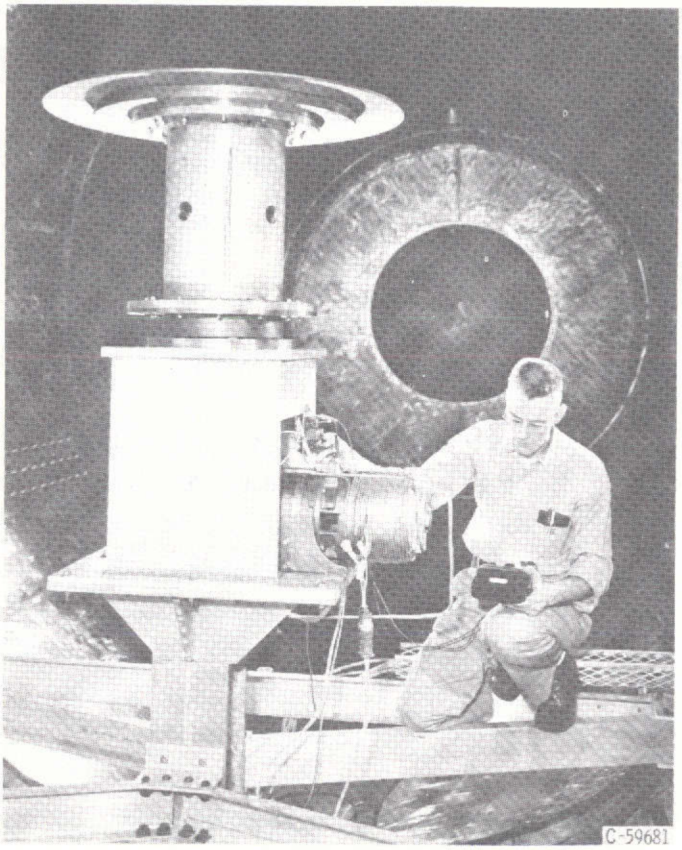

Figure 19. - Spin table with dummy load.

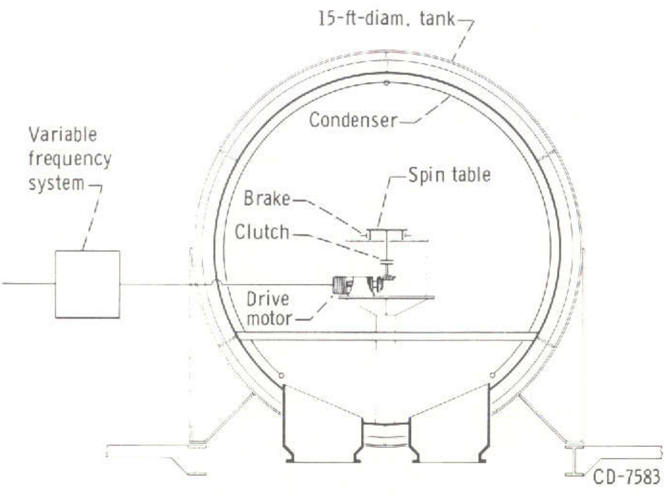

Figure 20. - Schematic of spin test apparatus.

condenser. The condenser is flooded with liquid nitrogen at the beginning of the pumpdown cycle.

Spin Test Apparatus

Figure 19 shows a spin test apparatus used in the 15-foot-diameter space tank for spin testing space payloads, components, or thrustors at controlled spin rates up to 282 rpm. Maximum clearance above the table is 8 feet with a side clearance of 5 feet. The spin table is approximately 20 inches in diameter and is driven through a shaft, electromagnetic clutch, and bevel gears by a variable-speed alternating-current induction gearmotor. Table rotation may be braked by an electromechanical braking unit. Figure 20 shows a schematic of the system. The variable-speed system consists of a direct-current motor with speed control driving an alternator at an infinite number of speeds. The speed of the gearmotor is a function of the frequency generated by the alternator and is adjustable up to full-field speed and can be maintained to 1 percent of a set value.

The spin table was operated in the 10^{-7} torr pressure range with a simulated SERT payload. This load had a weight of 278 pounds, a center of gravity 0.235 inch from the axis of rotation, and a moment of inertia of 7.45 slug-feet^2. The system accel-

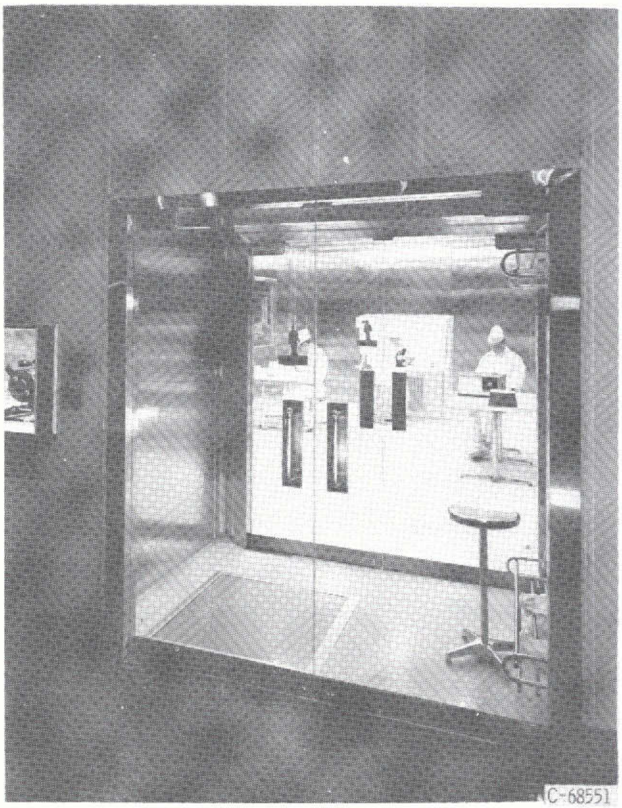

Figure 21. - Clean room.

erated to a speed of 282 rpm in 6 seconds and was braked to a full stop in $1\frac{3}{4}$ seconds with no overheating. With clutch disengaged, the unit coasted from 282 rpm to a stop in 7 minutes 35 seconds.

Clean Room

A clean room is located on the first floor adjacent to the control room. The room, shown in figure 21, is a nonlaminar design meeting the particle size distribution curves for a class 10 000 clean room as defined in Federal Standard 209 - Clean Room and Work Station Requirements, Controlled Environment (ref. 6). Class 100 is achieved on the laminar flow clean bench in the room. Supply air for the room is passed through an absolute filter, conditioned to a relative humidity of between 30 and 45 percent, and can be maintained with $\pm 2^{\circ}$ F of any desired temperature between 65° and 80° F. A positive pressure of 0.05 inch of water is maintained in the room. The stainless-steel walls are grounded to eliminate dust attraction due to a static charge.

The room is 27 by 13 feet with a 9-foot 4-inch ceiling height. Access is through a 4- by 8-foot air lock equipped with a vacuum cleaner for cleaning equipment prior to being taken into the clean room. The air lock is also utilized as a change room. Double doors provide ready access for moving large units of equipment into the room.

Special equipment includes the laminar flow clean bench with a work area of 24 by 47 inches, a work table with sinks completely covered with a hood, work tables, and a sealer for enclosing equipment in polyethylene bags.