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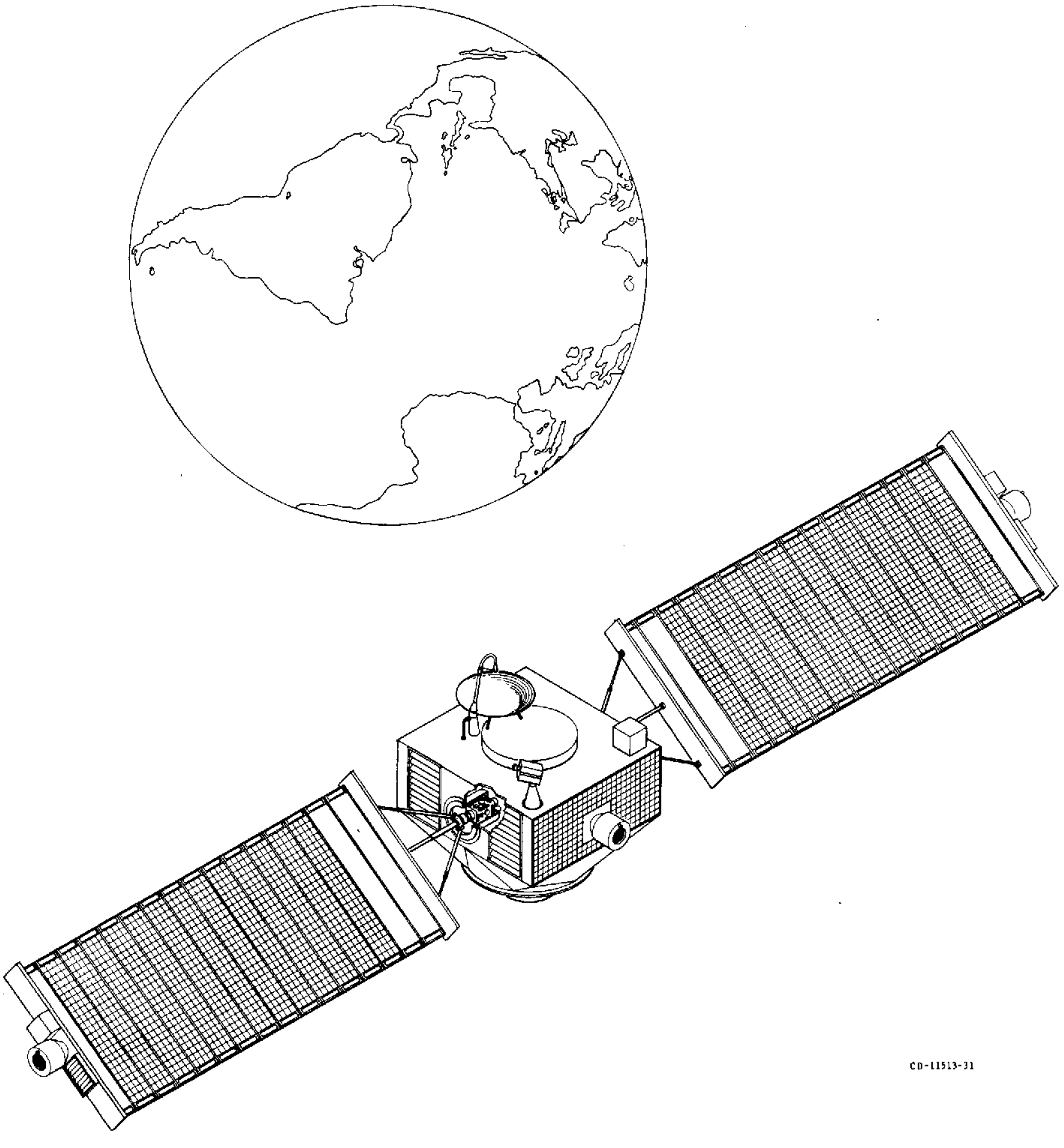
SERT D SPACECRAFT STUDY

Lewis Research Center
Cleveland, Ohio

SERT "D" SPACECRAFT



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ABSTRACT

The SERT D (Space Electric Rocket Test - D) study defines a possible spacecraft project that would demonstrate the use of electric ion thrusters for long-term (5 yr) station keeping and attitude control of a synchronous orbit satellite. Other mission objectives included in the study were: station walking to satellite rendezvous and inspection, use of a low-cost attitude sensing system, use of an advanced solar array orientation and slip ring system, and an ion thruster integrated directly with a solar array power source. The SERT D spacecraft, if launched, would become SERT III, the third space electric thruster test. (SERT I was launched in 1964 and SERT II in 1970).

FOREWORD

This SERT D Project Study has been prepared by members of the Spacecraft Technology Division who constituted a SERT design study team. Additional study team members included staff from the Electrochemical Technology Section and the Solar Cell Applications Section of the Energy Conversion and Materials Science Division. Elmer H. Davison and Robert C. Finke headed this SERT design study team. The team was assisted by all members of the Spacecraft Technology Division who contributed their advice and knowledge during the course of the project study.

This document has been reviewed by E. H. Davison of the Spacecraft Technology Division, Lewis Research Center, and J. Lazar of the Space Propulsion and Power Division, NASA Headquarters.

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  Mission Objectives - E. H. Davison

  Baseline Spacecraft Configuration - R. J. Zavesky

  Subsystem Trade-Off Studies - R. C. Finke

  Launch Vehicle - R. C. Finke

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6.0 Appendices

Directly Regulated Housekeeping Array - S. T. Gooder and A. C. Hoffman

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The layout of the detailed spacecraft configuration was done by R. J. Zavesky. Final editing, checking and report assembly was done by C. A. Low and W. R. Kerslake.

The contributors to this report would like to acknowledge the help of Linda Finke of the Spacecraft Technology Division for her skill and dedication in preparing the manuscript of this report.

CONTENTS

1.0 PROJECT PROPOSAL

1.1 Introduction

The development of space technology in the United States led to early recognition of the usefulness of synchronous equatorial orbit satellites for practical applications. These applications in general fall into the areas of communications, navigation and weather observation. At some future date, the transfer of solar energy from space to earth may be added to the list of applications.

Initially, the unique feature of the synchronous satellite, its ability to remain over one earth location, could not be fully exploited because of inherent technology limitations. In general, these limitations were launch vehicle payload capability, long term satellite attitude stability and station keeping requirements, power sources for satellites, and aerodynamic shroud volume limitations. Presently launch vehicle payload limitations are no longer a problem although the cost of alleviating payload restraints still determines whether certain applications are practical. Long term satellite attitude stability was initially obtained by employing spin stabilization. However, spin stability in conjunction with the use of solar cells as a power source and the restrictions of aerodynamic shroud volume limitations seriously restricted the power available to a satellite. Since aerodynamic shroud volume limitations are an inherent restriction and solar cells appear to be the power source for the foreseeable future, spacecraft designers are developing light weight 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. These 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 light weight 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. These techniques require thrust devices with high specific impulse to avoid the requirement of allocating an appreciable percentage of satellite weight to propellent 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 propellent 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 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 Kaufman thrusters, it appears appropriate to apply their unique capability to the next generation of synchronous satellites.

The SERT D study described herein addresses the applications of Kaufman thrusters and other technology to the next generation of synchronous orbit satellites. Because the proposed SERT D satellite represents the next generation of technology to be employed in synchronous orbit, it is appropriate at this stage of development to demonstrate the technology and concepts required to standardize and lower the cost of this next generation of satellites.

1.2 Mission Objectives

A discussion of SERT D mission objectives follows:

Objective I - 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. Their use at synchronous altitude for station keeping and attitude control represents one of the most attractive application of these propulsive devices as has been detailed in many studies. The ability to execute this experiment exists at the LeRC and should be used to translate this technology into a practical capability for synchronous orbit use.

ATS-F will fly ion thrusters that will be used for demonstration of North-South station keeping and attitude control. This initial experiment will be valuable in promoting the use of ion thrusters. However, the demonstration is short term and experimental in nature and, therefore, will not provide the practical demonstration of long term operation of ion thrusters required for future spacecraft projects. In contrast, SERT D will demonstrate the practicality of using ion thrusters as in-line functional elements of the attitude control and station keeping system for long duration missions. In addition, duration testing equivalent to five years of mission life will be accomplished. Thus, the successful performance of SERT D will greatly enhance the practical application of ion engines to future spacecraft.

There exists also a real requirement to demonstrate a low cost precision station keeping and attitude control capability for a 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, but its planned station keeping capabilities (± 1 degree north-south, ± 0.2 degrees east-west) and attitude control (± 0.1 degrees in pitch and roll, 1.1 degrees in yaw) can be improved upon significantly. The proposed SERT D would provide a demonstration of this improved capability with ± 0.05 degree north-south, 0.10 degree east-west, 0.08 degree pitch, and 0.2 degree yaw. In addition, SERT D would represent a NASA technology development readily available to U.S. industry and government users.

Objective II - As a corollary objective to objective I, SERT D will be used to demonstrate the ability of the electric propulsion thrust subsystem to station walk the spacecraft from its original on-station position to a different longitude. The longitude adjustment and inclination change maneuvering capability provided by the electric propulsion subsystem will allow a rendezvous mission to be performed.

Ground tracking stations will command the ion thruster system 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 the SERT D to be positioned close enough to perform a visual inspection by means of an on-board television system.

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

Objective III - 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 D 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. Demonstration of the performance and reliability of the SERT D system would permit substantial cost reductions in future synchronous orbit spacecraft.

Objective IV - Availability of a solar array orientation mechanism (SAOM) and power/signal transfer slip ring system is crucial to the development of low cost orbit platforms. Systems have been under development for a number of years within NASA that circumvent the problems of existing systems. For example, liquid metal slip rings (LMSR) have been developed at the LeRC 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 D.

Objective V - 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.

(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. For many electrical loads, this power conditioning equipment can be eliminated by using integrated solar array power conditioning techniques pioneered by the Lewis Research Center. 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 due to cell or interconnect failures is provided by diodes in parallel with cell groupings. These advanced power control techniques would be demonstrated on SERT D for appropriate thruster electrical loads. Such a demonstration would provide a significant step forward in spacecraft power system technology.

Objective VI - In addition to demonstrating technology, the spacecraft would provide an experimentation platform for a modest number of 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.

Objective VII - Perhaps the major benefit to be obtained from the development is that NASA would have available for future applications, a low cost synchronous orbit platform or bus. Such a bus could be placed in orbit in the future by its own launch vehicle or could be used as a shuttle/tug payload. It could be easily tailored to satisfy the power and size requirements of many users.

A low cost synchronous orbit platform has many applications. For example, users could add communication equipment to the bus to obtain very low cost communication systems. Such a communication spacecraft would be very attractive to developing countries because the precise station keeping capability of the bus would minimize ground system cost. Other applications of the bus are for navigation and weather satellites. As noted in the introduction, a new generation of synchronous orbit satellites is developing. The various technologies required by this new generation of satellites have been developed, but have not been combined into flight proven spacecraft. In executing a SERT D project, flight proof of the concepts and technologies required by the next generation of synchronous orbit spacecraft will be accomplished.

Objective Priority - The study showed that all mission objectives might not be achievable within present Thor/Delta launch capabilities. As a consequence, decisions are required on the priority of achieving the objectives and on the necessary weight contingency at project start. More detailed studies and further discussions are required before final decisions can be made. It appears, however, that it may be necessary to forgo a rendezvous demonstration in order to adequately demonstrate thruster life and operational procedures.

1.3 Baseline Spacecraft Configuration

The proposed SERT D is shown in figure 1.3.1 in its deployed in-orbit configuration. Structurally it consists of an earth tracking center body and fold-out arrays that track the sun. The solar array rotation axis is oriented in the north-south direction (i.e. perpendicular to the synchronous orbit plane). The earth facing side of the center body contains a high gain antenna for communication with the LeRC ground station. Two ion thrusters in conjunction with three-axis reaction wheels provide station keeping and attitude control for the spacecraft. The thruster system is fully redundant since only one body mounted and one array mounted thruster is needed for control purposes. The thrust vectors of the body mounted thrusters point east and west (tangent to the orbit velocity vector or roll axis). The thrust vector of the array mounted thrusters point north and south and are aligned with the array rotation axis which passes through the center of gravity of the spacecraft (pitch axis). Each solar array panel is provided a single degree of rotation by a solar array orientation mechanism (SAOM). Slip rings in the SAOM provide for the transfer of instrumentation and command signals, and electrical power across the rotating joint.

SERT III

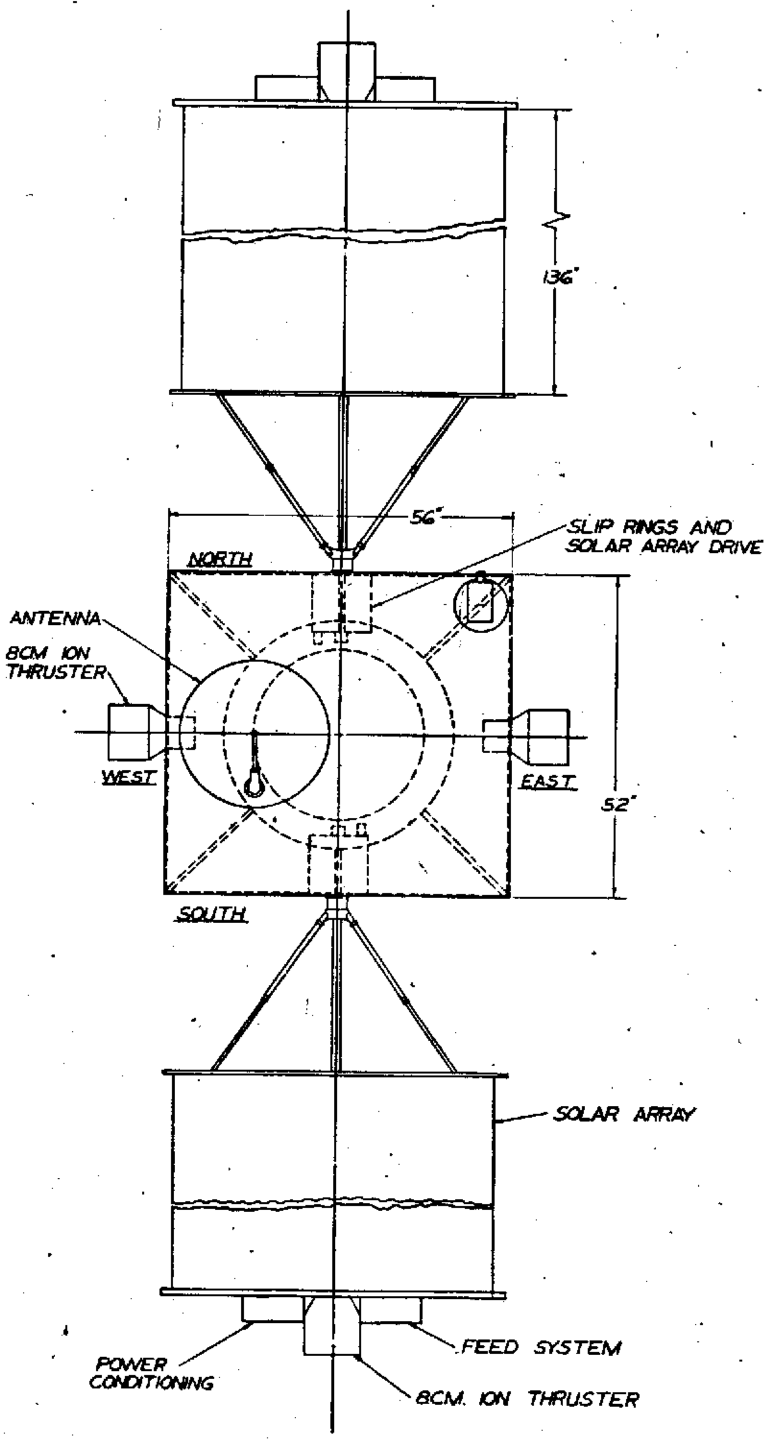


FIGURE 1.3.1

1.4 Subsystem Trade Off Summary

Launch vehicle payload limitations preclude the inclusion of all the desired experiments. Two alternative spacecraft configurations which satisfy payload restraints, but meet all mission objectives are discussed.

Configuration I carries an on-board randezvous radar which enhances the probabilities of locating a companion spacecraft. The radar is quite heavy (77 lbs) and the weight contingency of this configuration with the radar is 27.3 pounds. Preliminary studies indicate that radar may not be required for rendezvous. In this event, the radar could be removed and an ion thruster nickel-hydrogen battery experiment could be incorporated.

Configuration II (the proposed baseline spacecraft configuration) has the on-board radar removed, and depends instead on ground tracking stations to provide guidance until the spacecraft are in close enough proximity to allow the inspection television system to provide information for terminal guidance. Elimination of the on-board radar and addition of the ion thruster nickel-hydrogen battery experiment increases the weight contingency margin to 58.8 pounds. While Configuration II is the proposed configuration, it should be noted that radar rendezvous requirements might dictate the development of Configuration I.

1.5 Launch Vehicle

The basic vehicle for use in this mission is the three-stage Thor Delta vehicle model 2914; 116 feet long, 8 feet in diameter, and weighing 202,500 pounds gross. (See Fig. 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 cutoff (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 Aerojet AJ10-118F pressure-fed gimballed engine and a new "DIGS" guidance system with a digital computer. Fuel for this stage is Aerozene 50 and the oxidizer is nitrogen tetroxide, developing a thrust of 9400 pounds. This stage weighs 12,000 pounds gross. 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 cutoff (SECO). Two fixed nozzles fed by the propellant tank helium pressurization system provide retro-thrust after third-stage separation. (See Fig. 1.5.2)

The spin-stabilized third stage uses a solid propellant motor TE 364-4, developing a total impulse of 653,500 lbf-second at an average thrust of 14,870 pounds (16,900 lbs. maximum). This stage is 38 inches in diameter and 68 1/4 inches long. (See Fig. 1.5.3)

The committed capability of the launch vehicle is 1500 pounds into the synchronous transfer orbit. This capability does not include the adapter weight.

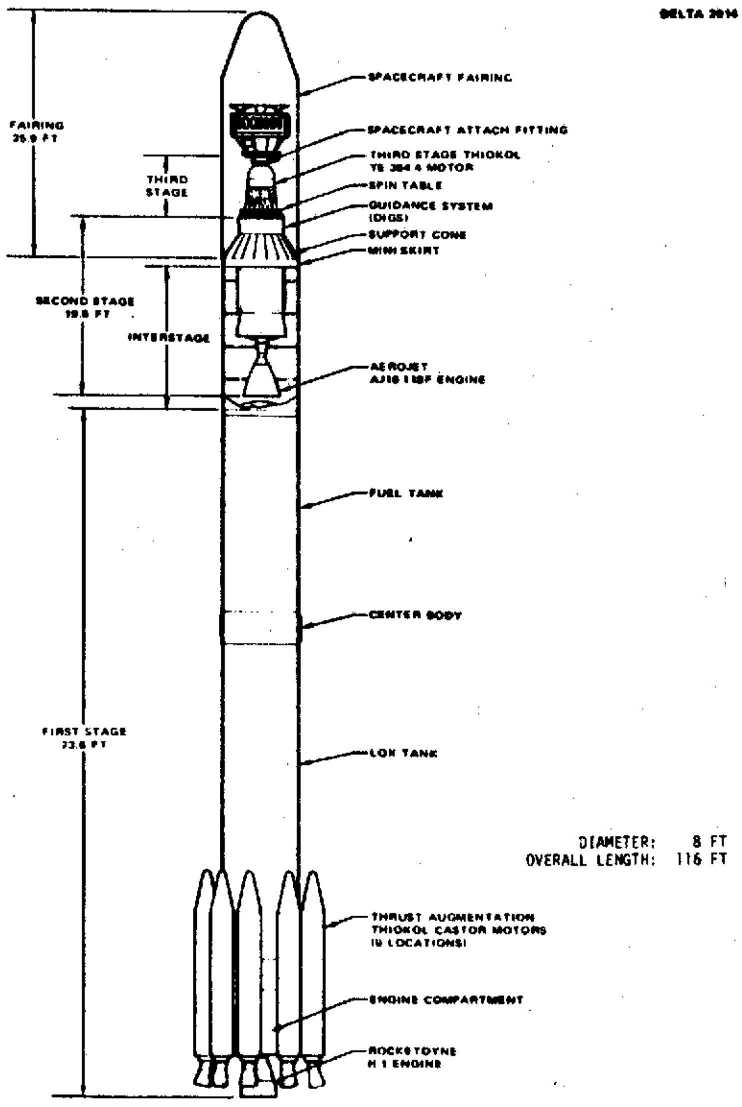


Figure 1.5.1 - DELTA 2914 Launch Vehicle Showing Interface with Spacecraft

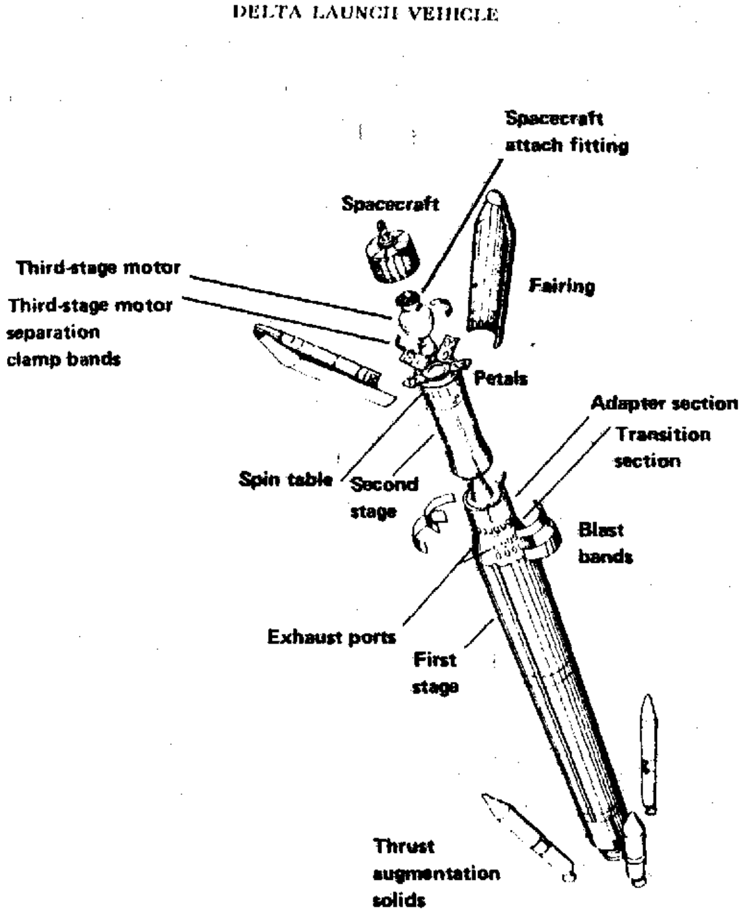


Figure 1.5.2 - DELTA Staging and Separation Events, Shown Here for an Applications Technology Satellite Launch

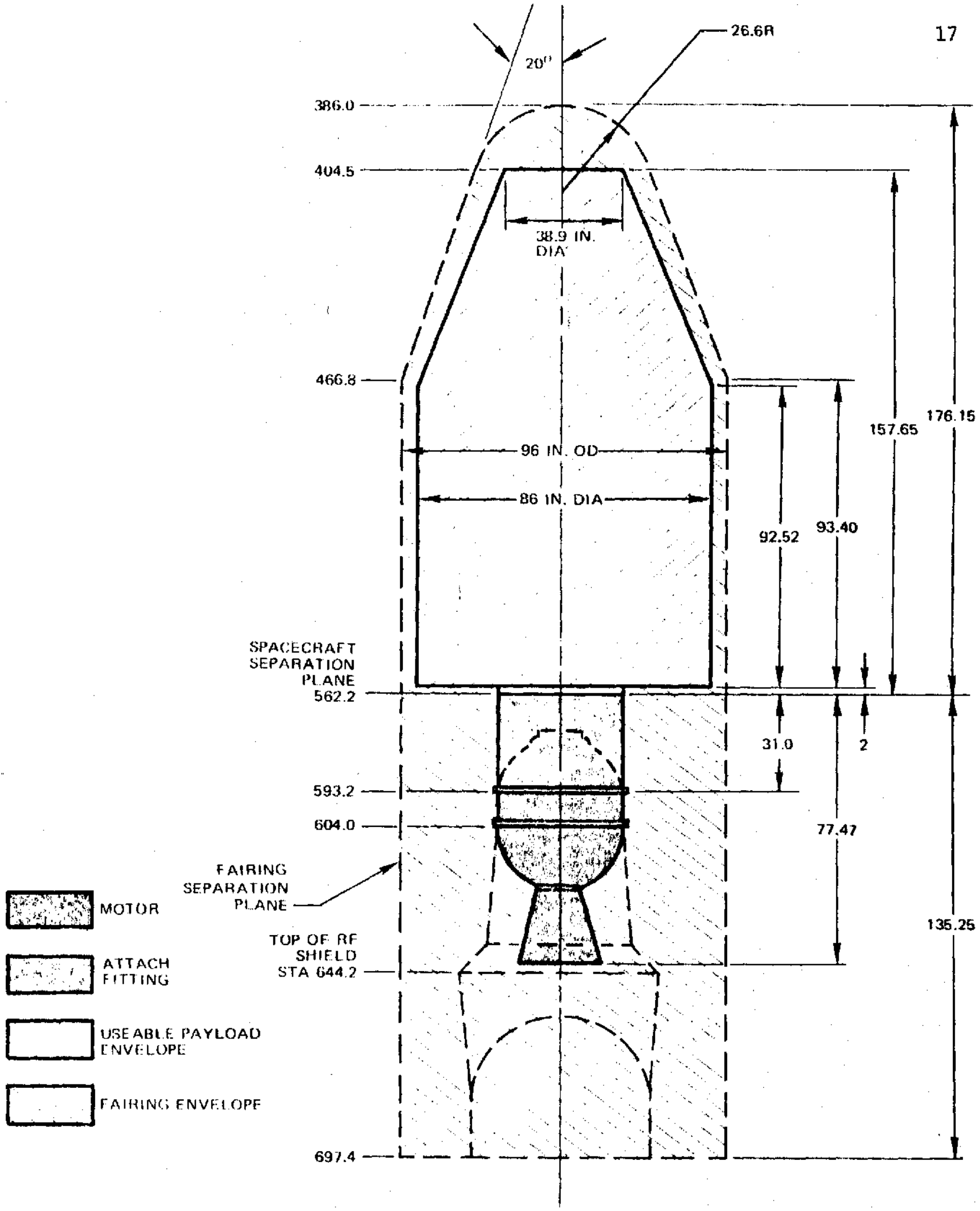


Figure 1.5.3 - Payload Envelope, TE-364-4, 3731 Attach Fitting

1.6 Summary

Configuration II allows the direct demonstration of thruster life by operating the two E-W thrusters in opposition. The duty cycles in this mode will be increased to provide the equivalent of 5 years station keeping thruster operation to be demonstrated in somewhat less than one year of spacecraft life. The equivalent experiment with Configuration I requires maneuvering the spacecraft in a complex pattern in order to retain its station.

Demonstration of thruster operation with a battery will be done with configuration II. This demonstration may be an important precursor for future communication satellite designs.

Configuration II places a greater emphasis on precise guidance and tracking by STDN stations. Mission profiles addressing this problem will be a part of the spacecraft design.

The greater benefits resultant to the mission, in conjunction with a larger weight margin clearly dictate that configuration II should be pursued as the SERT D baseline configuration.

A FY 75 start, would allow a launch in mid CY 77. However, a launch late in CY 77 would be more realistic and would provide schedule contingency.

The project cost (from project approval to launch) in Net R&D dollars for configurations I and II are 16.5 and 18.3 million, respectively. The manpower requirements for both configurations are approximately the same at 448 manyears. Both the dollar and manpower estimates include a 20% contingency and the dollar estimates include a 5% yearly inflation rate.

2.0 MISSION REQUIREMENTS AND OPERATIONS

2.1 Mission Profile

Introduction - The objective of the SERT D mission is to launch a spacecraft into a synchronous equatorial orbit, and to demonstrate the use of 8-cm ion thrusters for attitude control, east-west and north-south station keeping, station walking, and rendezvous functions. Accelerated life testing of the body mounted thrusters will also be performed. The station keeping experiments will be performed in the longitude quadrant centered about the Earth's minor axis at 105° west. The rendezvous experiments will be performed with other spacecraft in a synchronous, nearly equatorial orbit. The station walking demonstration will involve placement of the spacecraft at its target longitude following apogee injection, and maneuvering the spacecraft between station keeping experiments. The east-west thrusters will be subjected to an accelerated life test. Seven months on an accelerated cycle schedule will simulate five years of normal operation for station keeping and attitude control. The thrusters will also be operated over the five year life of the mission in the cyclic fashion required by station keeping and attitude control. Rendezvous operation may require thruster operation up to 48 days, depending upon the relative positions of SERT D and the rendezvous target. Rendezvous and station walking operation will be combined with the accelerated life testing demonstration.

Transfer Orbit - The SERT D will be launched with the Delta 2914 launch vehicle at a time compatible with the launch window constraints. Following lift off, the first stage burn and a partial burn of the second stage will place the spacecraft, assuming a 95° launch aximuth, into a 28.7° inclined 100 n.m. circular parking orbit. After a 17 minute coast in the parking orbit the second stage is restarted. The second and third stage burns perform a small plane change and inject the spacecraft, spinning at 60 rpm, into an inclined elliptical transfer orbit 26 minutes after lift off.

The apogee altitude of the transfer orbit is biased above the nominal synchronous altitude, such that following the apogee motor burn, the orbit is elliptical and has a westward drift on the order of 6 degrees per day. The station keeping experiments will be performed in the longitude range of 60° west to 150° west, where the boundaries represent the longitudes of maximum drift acceleration caused by the Earth's triaxiality. Assuming 5° elevation angles, the spacecraft is visible from Cleveland in the longitude range of 10° west to 153° west. Apogee injection to the west of 10° west is desirable if visibility from Cleveland during the drift phase is a consideration. Selecting a drift rate bias of 6°/day west and applying a typical 99% dispersion of ±12°/day results in drift rates which could range from 18 deg/day west to 6 deg/day east. Because the injection error distribution is

Gaussian, less than 10% of the apogee injections would result in an eastward drift. The drift rate bias is selected such that the spacecraft arrives at the target longitude within an acceptable period of time. The transfer orbit apogee is biased about the synchronous altitude to yield the desired drift rate after apogee motor firing. The magnitude of the apogee bias and the transfer orbit inclination are determined such that the spacecraft weight at station acquisition is maximized. The transfer orbit perigee is 100 n.m. After injection into the transfer orbit, the spacecraft telemetry and command functions, along with spacecraft ground tracking for orbit determination will be initiated as soon as ground station visibility permits. A sufficient number of ground stations and an adequate tracking schedule will be provided to determine precise transfer orbit parameters for spin axis attitude determination and apogee motor aiming. The spacecraft will remain in the perigee injection attitude until sufficient attitude information is available for the spin axis precession maneuver into the apogee motor firing attitude. Data from two horizon crossing indicators and a digital sun sensor are available for attitude determination. The optical axes of the horizon sensors are canted at a small angle (4° - 6°) with respect to the spin plane. The digital sun sensor measures the sun elevation angle from the spin plane and provides a command pulse for referencing the thruster firings for the precession maneuver.

In the perigee injection attitude, the earth sensor coverage is such that data is available for only a few minutes after transfer orbit injection. Shortly thereafter, the sensors lose the earth and do not reacquire until the spacecraft approaches apogee. A coarse precession maneuver will be executed in the neighborhood of the first apogee where the earth sensor data will be available to monitor the maneuver. During the maneuver, the spin axis is precessed in a manner which honors the thermal and power design limits of the spacecraft, maintains adequate antenna patterns, and provides earth sensor data to monitor the maneuver. The maneuver will consist of a series of attitude determinations, maneuver computations, and maneuver executions. The precession maneuvers will be performed with a hydrazine thruster aligned parallel to the spin axis. A final trim maneuver will be performed shortly before the second apogee, which is the first opportunity for firing the apogee motor.

As mentioned previously, the second and third stage of the launch vehicle placed the spacecraft into an inclined, elliptical transfer orbit having an apogee biased above the nominal synchronous altitude. The apogee motor is used to raise the perigee to near synchronous altitude and to remove the orbit inclination. Following injection, the spacecraft is despun, the solar arrays deployed, and the spacecraft controlled in a three axis mode. Following attitude acquisition, it is desired to place the spacecraft in the longitude quadrant centered about 105° west for purposes of conducting the east-west and north-south station keeping experiments. Any longitude within the quadrant is acceptable. However, an initial experiment at either 60° west or 150° west would demonstrate east-west station keeping at the points of maximum drift acceleration due to the earth's triaxiality.

Nominally, the orbit established following injection has a semi-major axis corresponding to a 6 deg/day westward drift rate. Table 2.1.1 shows the longitudes of the apogee injection as a function of apogee number for an apogee bias of 600 n.m. above synchronous altitude. The drift rate subsequent to injection is reduced by firing the 8-cm ion thruster mounted on the east face of the spacecraft and directed opposite to the velocity vector. For an injection on the second apogee and a nominal initial drift rate of 6°/day, the drift rate is reduced to zero, 23 days after injection with the spacecraft positioned at 150° west. Less westerly stations, of course, require less drift time, approximately one day for six degrees in the longitude region of interest.

Station acquisition times for off nominal injection, for example the 99% apogee injection dispersion, could increase the drift time to 150° west longitude to 80 days. During the station acquisition phase, the inclination dispersions are corrected using the 8 cm ion thrusters mounted on the north and south faces of the spacecraft. For an inclination error of .5°, the maximum expected dispersion, 40 days of continuous thrusting is required to remove the inclination.

The transfer orbit operations are shown in diagram form in figure 2.1.1, and as a sequence of events in table 2.1.2.

TABLE 2.1.1 - INJECTION LONGITUDES ASSUMING A 600 N.M. HIGH APOGEE BIAS

APOGEE NUMBERS	INJECTION LONGITUDE
1	101° east
2	62° west
3	134° east
4	29° west
5	167° east
6	4° east
7	157° west
8	37° east

TABLE 2.1.2 - MISSION OPERATIONS SEQUENCE

T_o = Launch + 26 Minutes - Injection into transfer orbit
Enable T & C
Initiate orbit determination

T_o + 3 1/2 hours - Initiate attitude determination

T_o + 4 1/2 hours - Reorientation maneuver

T_o + 5 1/2 hours - First apogee

T_o + 7 1/2 hours - Complete trim maneuver

T_o + 12 1/2 hours - Continue attitude determination

T_o + 14 hours - Trim to precise firing attitude

T_o + 16 1/2 hours - Second apogee
Fire apogee motor

T_o + 3 days - Complete attitude acquisition
Commence drift rate removal

T_1 = T_o + 80 days - Acquire station at 180° west (60° west)
Perform station keeping and attitude control experiments

T_1 + 60 days - Station Walk to 135° west

T_1 + 72 days - Perform S/K experiment at 135° west

T_1 + 132 days - Station walk to 120° west

T_1 + 360 days - Initiate rendezvous

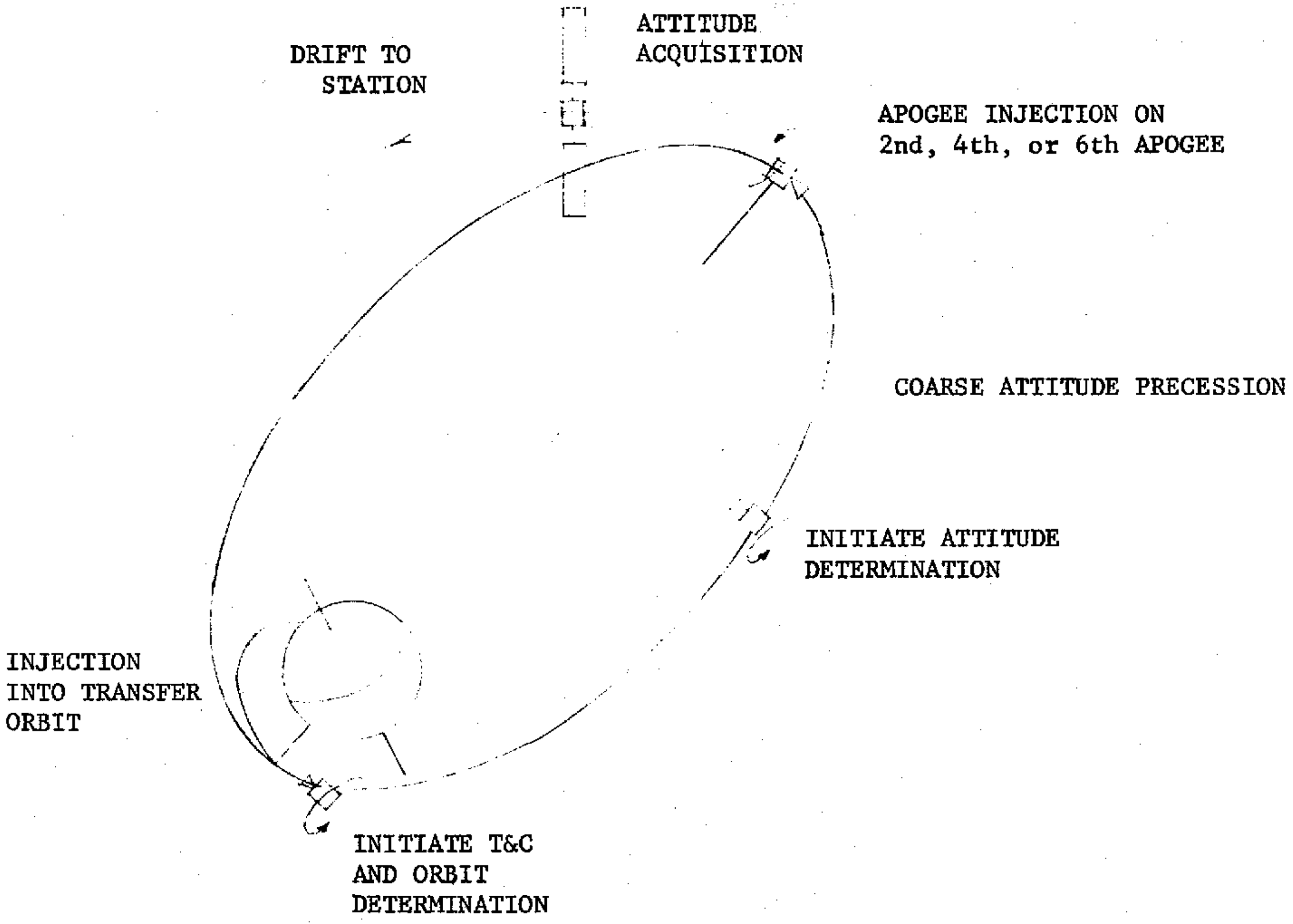


Figure 2.1.1 - Transfer Orbit Profile

2.2 Synchronous Orbit Operations

Introduction - Table 2.1.2 also shows the sequence of synchronous operations. Upon arrival at 150° west, precise station keeping and attitude control will be demonstrated for a period of two months. The spacecraft will then station walk to 135° west where the attitude control and station keeping experiments will be continued. The station walking capability of the 8 cm thruster system is shown in figure 2.2.1. Twelve days are required to accomplish the required longitude change of 15 degrees. The entire longitude quadrant centered at 105° west will be mapped with a series of station keeping and station walking experiments. These demonstrations will verify the achievable station keeping accuracies as a function of control period and drift acceleration.

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 D spacecraft and the candidate rendezvous spacecraft, it may be advantageous to interrupt the station keeping experiments and proceed with a rendezvous experiment. Otherwise, the rendezvous experiment will be deferred until the initial station keeping experiments are completed. The north-south and east-west thrusters will be operated simultaneously to produce the required inclination and longitude change for rendezvous. Figure 2.2.1 shows that the maximum longitude change of 180 degrees can be accomplished within 48 days. Any inclination change required can be accomplished at a rate of .1 degree per 8 days of thrusting.

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 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. 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 perturbation 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 section 2.3 the station keeping system is required to control both longitude and latitude to ± .05°.

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 1.3 hours. The south thruster would fire on the decending 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 2.2.2. 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 D can be obtained directly by using the 8 cm thrusters mounted on the east and west faces of the spacecraft. The maximum east-west variation due to solar pressure on SERT D is about ± .05 degrees if left uncontrolled. In order to minimize the overall east-west variation, however, it may prove to be desirable to control the solar pressure variation to a lesser value. If a maximum variation of ± .01 degrees were chosen, the thrusting time required of the body mounted thruster would be 45 minutes twice during one orbit every seven days. The array mounted 8-cm thrusters provide the backup capability. One thruster would be required to operate for 4.3 hours twice during one orbit every seven days. The required posigrade and retrograde impulses are achieved by alternatively yawing the spacecraft plus and minus 10 degress.

East-west station keeping to counteract the effect of earth triaxiality is accomplished by periodically changing the orbit semi-major axis. The variation in semi-major axis as a function of variation in satellite longitude is shown in figure 2.2.3 for cases with station keeping and without station keeping, for the triaxiality effect. The coordinate system is referenced to the desired operating station. The deadbands for the acceptable longitude error caused by the triaxiality are also shown. The station keeping procedure is to first obtain a precise orbit definition and then set the semi-major axis at the value Δa_c less than the synchronous value as shown at point A. 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, use of the east body mounted thruster appears to be attractive for this operation. Alternatively, or as a back up the array mounted thrusters can be used as previously described, in which case 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 22 minutes.

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 D to 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 is accomplished by reversing the thrust directions. Figure 2.2.1 shows the minimum time required to accomplish a given change in spacecraft longitude. The characteristic velocity associated with each longitude change can be reduced by allowing a coasting time between thrusting periods, and accepting the longer station walk time.

Because it may be desirable to perform a rendezvous maneuver with spacecraft which either do not have N-S station keeping or for which the N-S station keeping system has failed, it may be necessary to change inclination during rendezvous as well as station. The 8-cm ion thrusters mounted on the solar arrays would be used for this maneuver as they provide directly the north and south impulses required. However, because of the large ΔV requirements, considerable time is involved in changing inclination with these thrusters. For instance, the time required to change station by 180 degrees is about 50 days. During this time, the array mounted thrusters, firing continuously, can change inclination by about 0.6 degrees. Thus, unless long rendezvous times are allowed, rendezvous will be limited to those spacecraft having inclinations of less than 1 degree. It is possible that the hydrazine system could be used to change inclination for an extreme case, but it could not be used regularly without significantly increasing the hydrazine required.

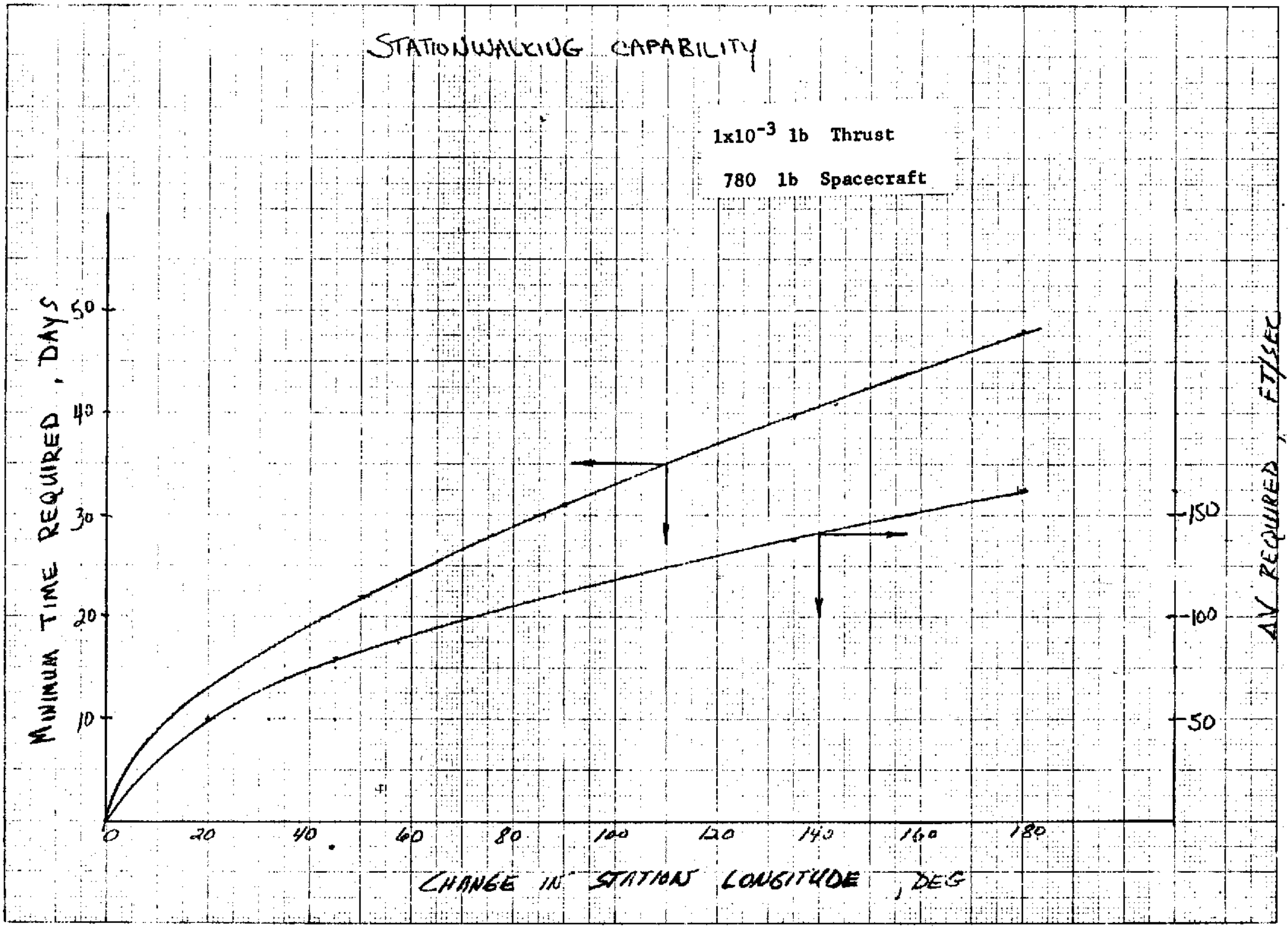
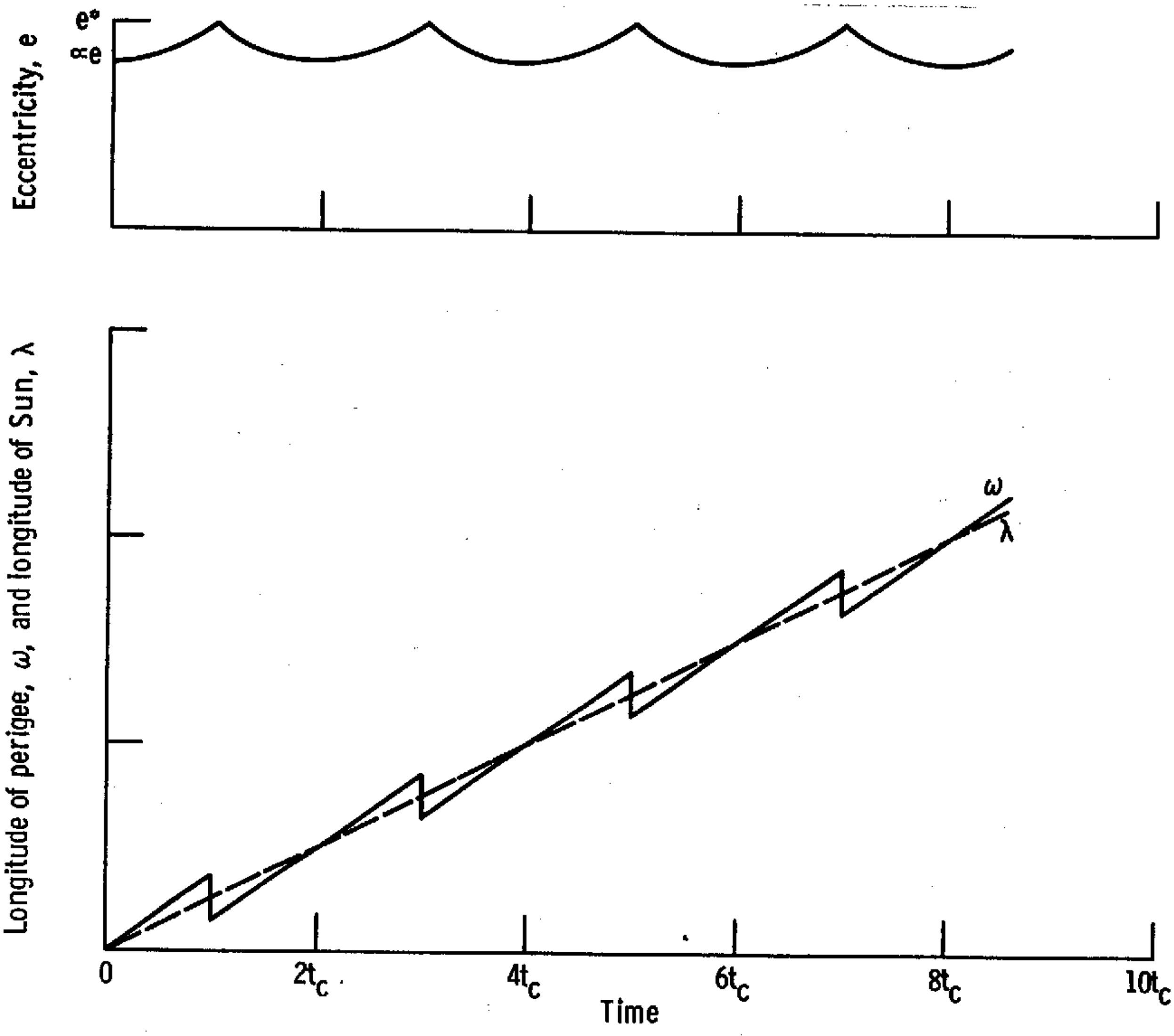


Figure 2.2.1 - Station Walking Capability



e* = Maximum Allowable Eccentricity

Figure 2.2.2 - Orbit Parameters vs Time when Correcting for Solar Pressure
Initial Conditions: Eccentricity e*;
Longitude of Perigee, 0; Sun Longitude, 0

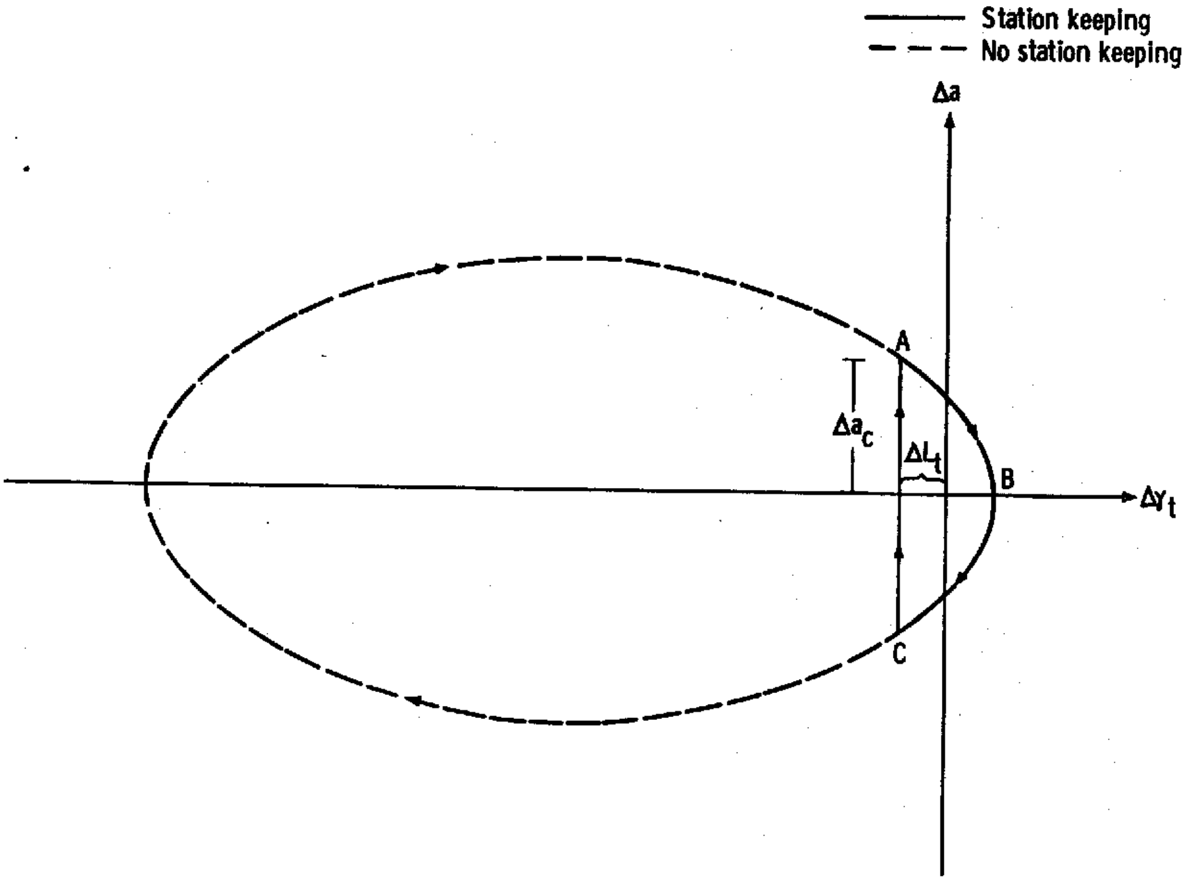


Figure 2.2.3 - Variation in Semimajor Axis as Function of Variation of Satellite Longitude. Perturbation, triaxiality.

2.3 Mission Requirements

The spacecraft orientation accuracy requirements are determined by the objectives which are to be achieved. In its role as a SEPS/GEOSEPS precursor flight, SERT D 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 D 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 executed (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 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. The pointing accuracy becomes more stringent as the antenna beamwidth becomes narrower, which occurs as transmission frequency is increased. 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 narrow antenna beamwidth trades-off in relaxing 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 power.

The half-power beamwidths of satellite antennas receiving frequencies in the 12-14 GHZ range may be as small as 0.5°. In determining the attitude control accuracy requirements, the trade off between satellite transmitted power and attitude control accuracy must be considered. As an example of analyzing this trade off, assume that the satellite is required to cover a geographic region with a circular antenna beam. Let θc be the angle, subtended from the satellite, which just covers the geographic region (see figures 2.3.1a and 2.3.1b). 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 2.3.2). 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 transmitted power and P* is the transmitted power for zero attitude control errors. This function is plotted in figure 2.3.3, where the value of θc 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-pitch plane (or the latitude of the ground spot). The allowable error decreases as this angle increases. Thus, to obtain an effective yaw tolerance on the ground of .03° 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 .02 degrees respectively, figure 2.3.3 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 sizable 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 as far as possible 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 2.3.4. 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°, .03°, and 0.2° may need to 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 or 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 D spacecraft will use a two-axis earth sensor for pitch and roll error signals, and attitude accuracy goals will be .08°, .08°, and .2° roll, pitch, and yaw, respectively.

Station Keeping Requirements - 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. Sinçe $\theta_p \neq 0$ 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 2.3.5 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 $\theta_p = 0_s = 0$. 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 specifications 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 a 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 2.3.5 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 $\theta_s + \theta_p = 1/2\ \theta_o$, 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 foot to 10 foot range. For receiving frequencies in the 12-14 GHz range, the half-power beamwidth could be as small as 0.5°. 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 ± 0.2°. 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 $\theta_o = 0.5°$ and $\theta_p = 0.2°$, a reasonable value for θ_s is 0.07°. A station keeping error of ± 0.05° in the north-south direction and ± 0.05° in the east-west direction will yield a value of 0.07° 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 ± 0.05 degrees in both the east-west and north-south directions. Maintaining the latitudinal variations within the ± 0.05 degree band for SERT D 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 ± .02 degrees).

Longitudinal station variations are caused primarily by two perturbation sources. The earth's triaxiality causes a long period (about 2.2 years) 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 D the maximum variation due to this effect would be about .05 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 ± 0.1 degree.

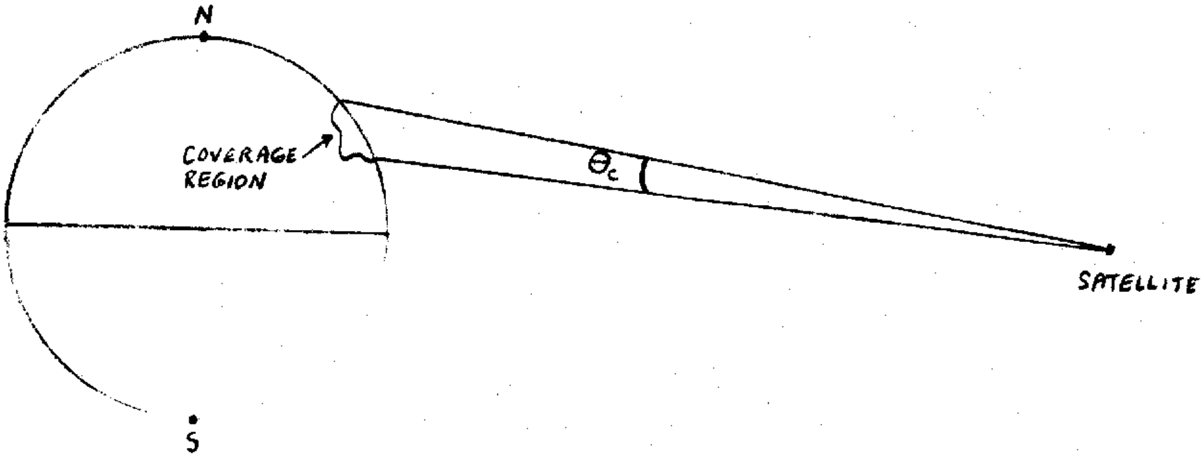


Figure 2.3.1a - Coverage Region as Viewed from Equatorial Plane

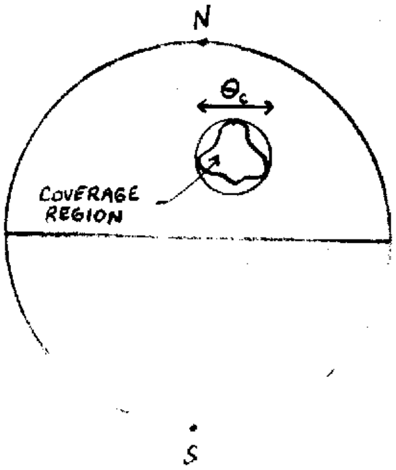


Figure 2.3.1b - Coverage Region as Viewed from Satellite

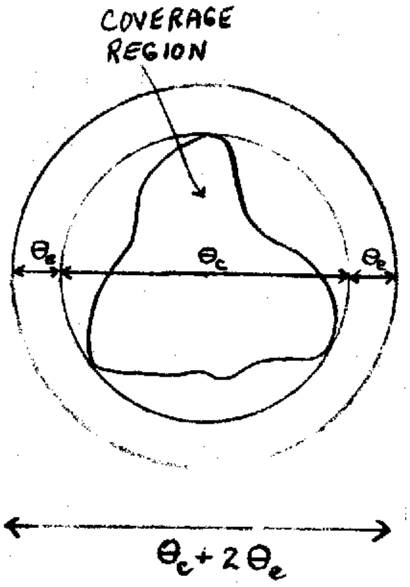


Figure 2.3.2 - Coverage Area Enlargement for Single Beam

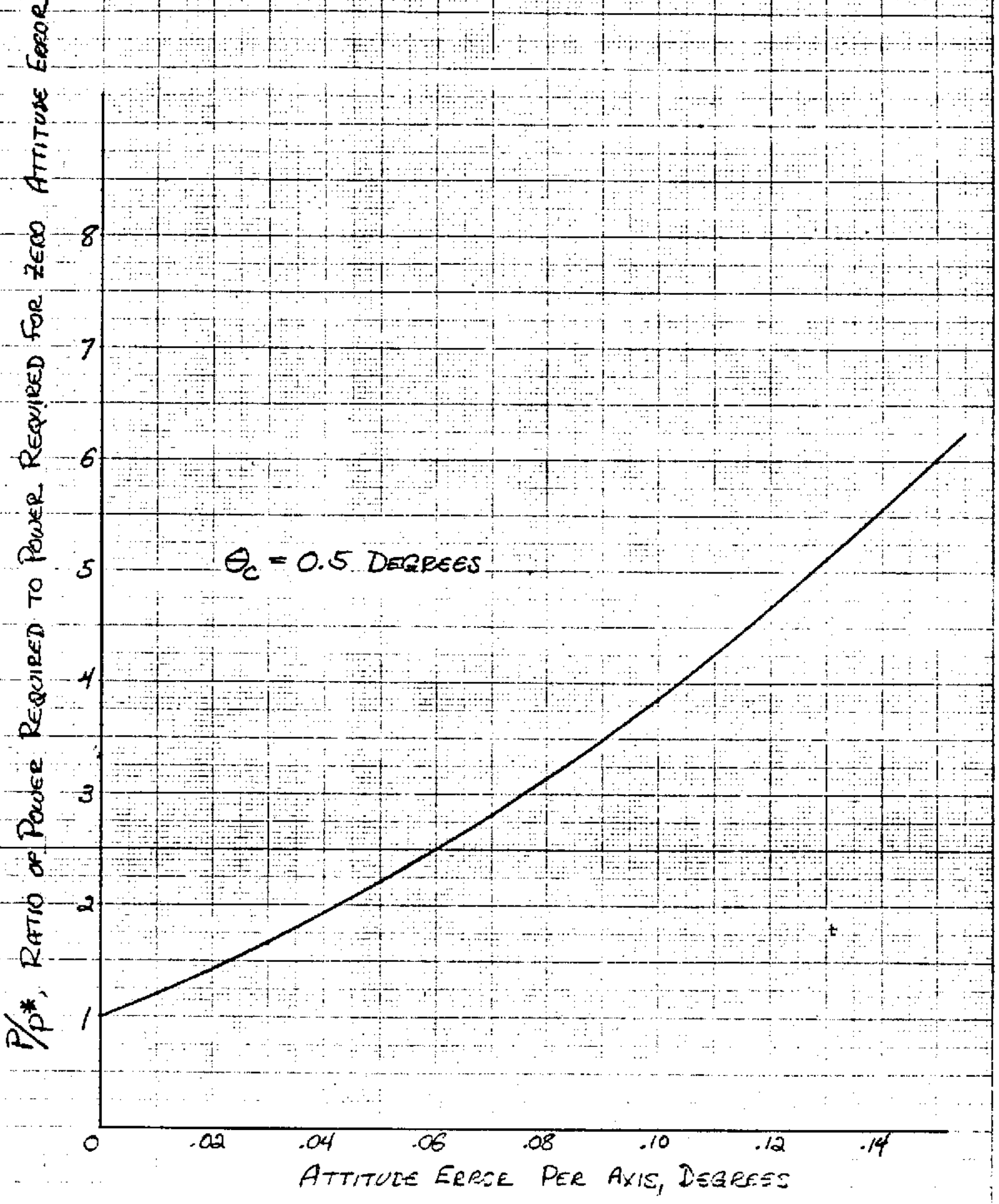


Figure 2.3.3 - $\frac{P}{P*}$ vs Attitude Error Per Axis

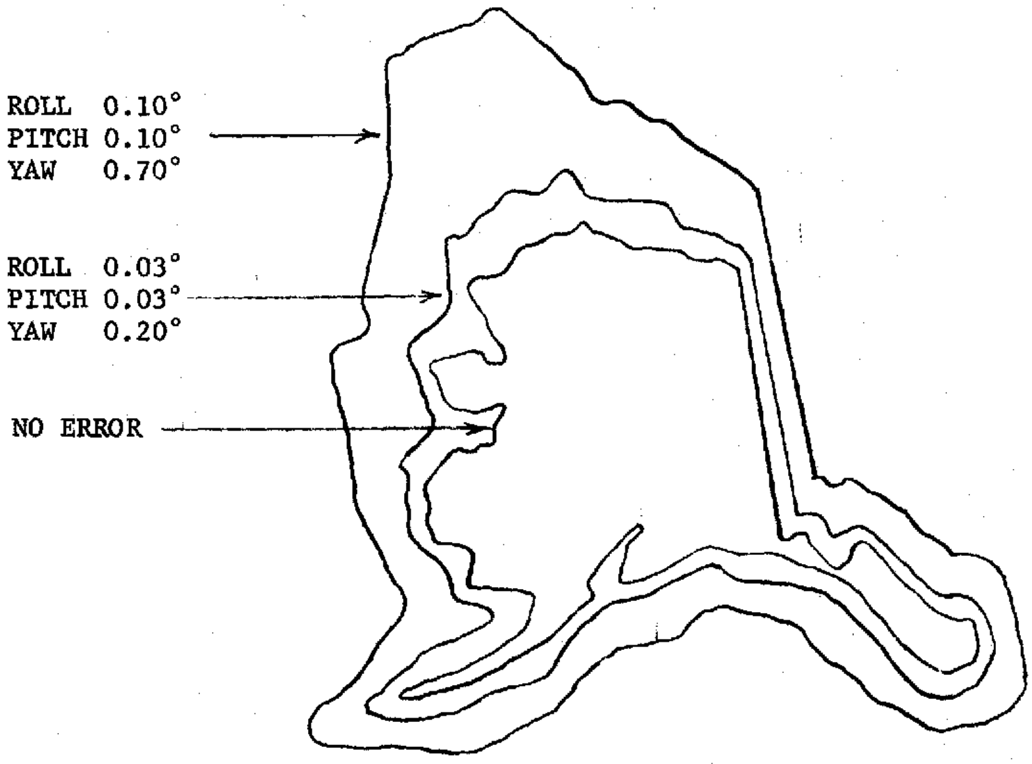


Figure 2.3.4 - Coverage Area Enlargement for Alaska
Satellite Longitude = 130°W

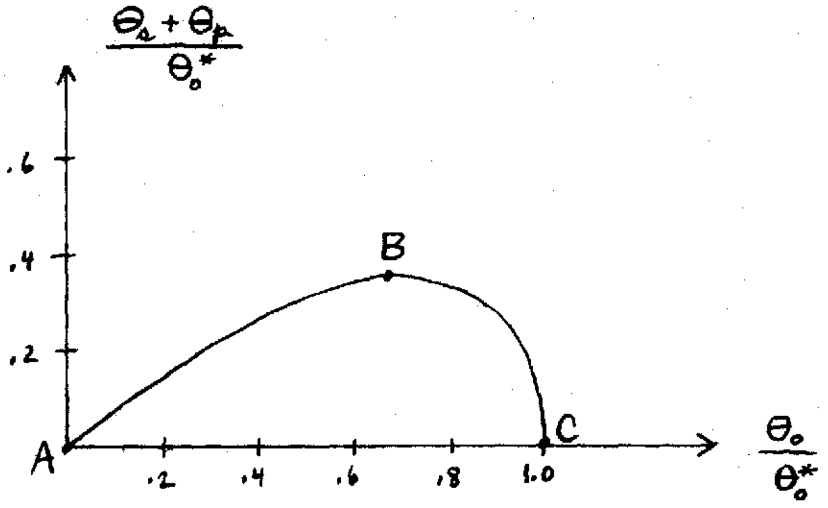


Figure 2.3.5 - Normalized Worst Case Error as a Function of Antenna Half Power Beam Width

3.0 SPACECRAFT DESCRIPTIONS

3.1 Spacecraft Configuration

Figure 3.1.1 shows the proposed SERT D spacecraft in its launch configuration. The in orbit configuration was shown previously as figure 1.3.1. A description of the major features of the spacecraft follows.

The spacecraft will be designed around the apogee motor and will conform to the Delta payload envelope shown earlier in figure 1.5.3. A thrust tube is formed around the apogee motor and is the inside wall of the spacecraft. A frustum of a cone structure will mount the spacecraft apogee motor thrust tube to the Delta 3731A attach fitting which mounts the spacecraft to the third stage Delta engine. A reinforced ring is mounted to the cone at the spacecraft separation plane and is also supported by four struts that extend from the spacecraft apogee motor mount to the inside of the ring.

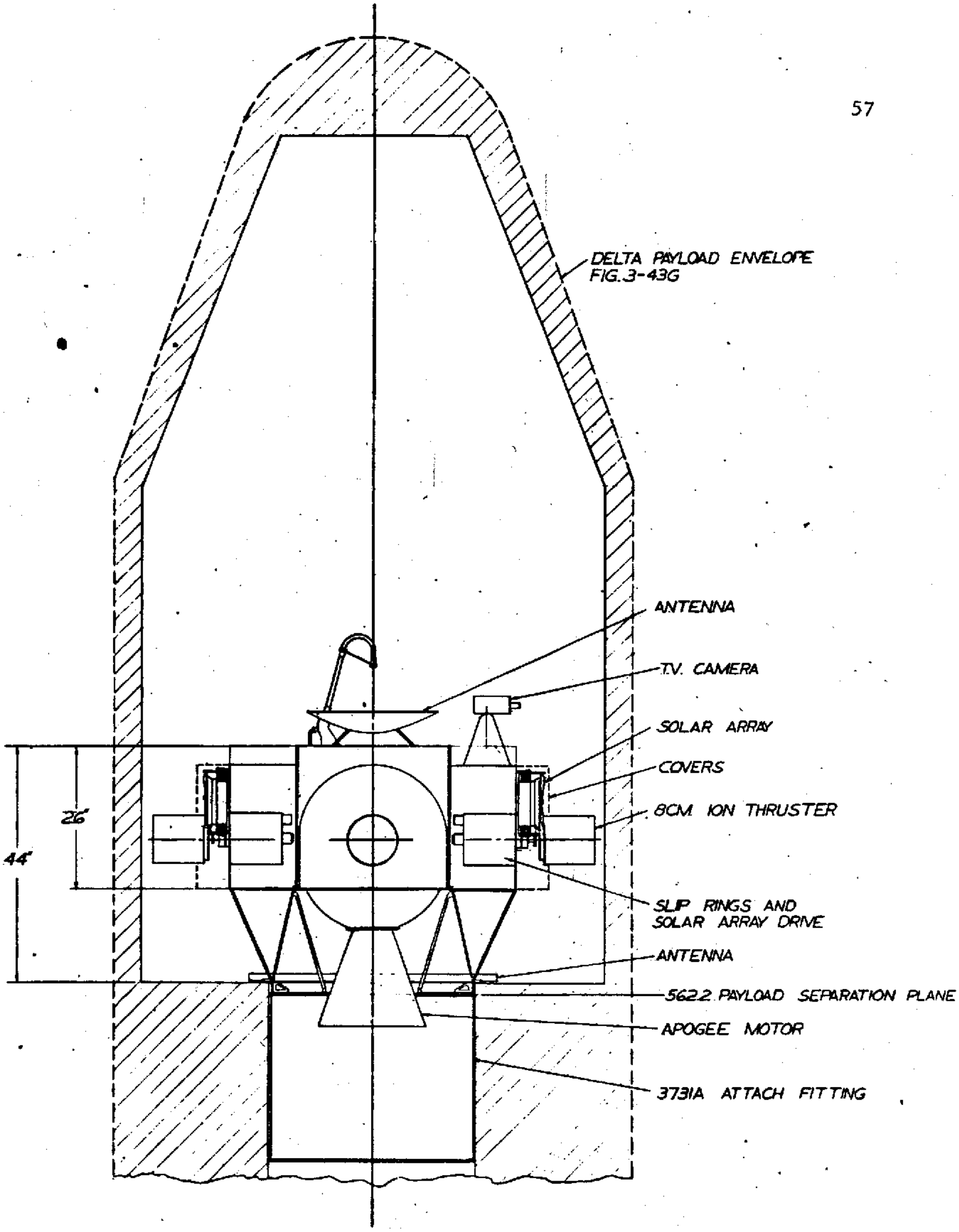
In the deployed in orbit configuration, the spacecraft is oriented so that the north and south panels of the center body always face deep space and are the best surfaces for radiating significant heat. Most of the spacecraft systems components requiring significant heat rejection will be mounted to these panels in addition to the solar arrays and their orientation mechanisms. Therefore, the panels will have to be structural members. Struts will be used to carry the loads from the panels directly to the reinforced ring and the attach fitting. A structural panel or shelf will act as a shear member between the base of the four outside panels and the thrust tube. The tops of the four panels are attached to the thrust tube by either struts or a plate which would act as a cover. More structural ties between the panels and the shear tube may be required to stiffen the center of the panels and act as shear ties across the spacecraft. The housings of the solar array orientation mechanism may be used for this purpose to support the north and south panels. The edges of the north and south panels can be joined by cross shear members. These would be used to mount the east and west 8-cm thrusters and the east and west face sheets.

The structure will be fabricated from aluminum sheet metal parts whenever 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 to the inner face.

The spacecraft must have enough area to radiate the heat that is generated by the thruster, power conditioning, telemetry, attitude control system, and experiments such as the TV camera and radar system. The spacecraft is oriented so that the antenna on top of the spacecraft center body faces the earth, the solar arrays extend from the north and south faces, and the body mounted thrusters are on the east and west faces. Because of this orientation, the north and south faces of the spacecraft always face deep space and are the best surfaces for radiating the excess heat. The areas of the north and south faces were selected on the basis of the heat that the center body of the spacecraft has to radiate. The width was determined by the width of the solar array storage configuration. The east and west panel width was determined by the constraint of putting the apogee motor, two solar array orientation mechanisms two folded solar arrays and two thrusters in line within the shroud envelope diameter.

The spacecraft must be designed to provide mounting for the components. Component placement is determined by four considerations: 1) the dynamic balance of the spacecraft about the spin axis; 2) the moments of inertia about the principle axis; 3) the balance of heat that must be radiated from the north and south faces, and 4) a subsystem grouping. Most of the components will be mounted on the north and south panels of the spacecraft. The reaction wheels and the hydrazine tank will be mounted on the shear plate that connect the outer panels to the thrust tube and the panels at the apogee motor mount. The antenna and camera will be mounted on the top of the spacecraft. Figure 3.1.2 shows a top line schematic of the position of some of the major components. The attitude control components are mounted on the north panel and the telemetry and command system components are mounted on the south panel.

The solar arrays and solar array orientation mechanisms with slip rings are mounted on the north and south panels. The solar arrays will be folded against the panels and be held in place with straps and explosive bolts. A cover will be required over the north and south panels in order to mount the solar cells required for the transfer orbit power. A hole will be cut in the covers for the north and south thrusters. The covers will be ejected prior to solar array deployment. The solar array deployment mechanism will include a bar linkage which will move the array away from the spacecraft to achieve a better thermal view factor for the north/ south panels. Solar cells will also be mounted on the east and west panels of the spacecraft for transfer orbit power.



SERT D

FIGURE 3.1.1 - SERT D

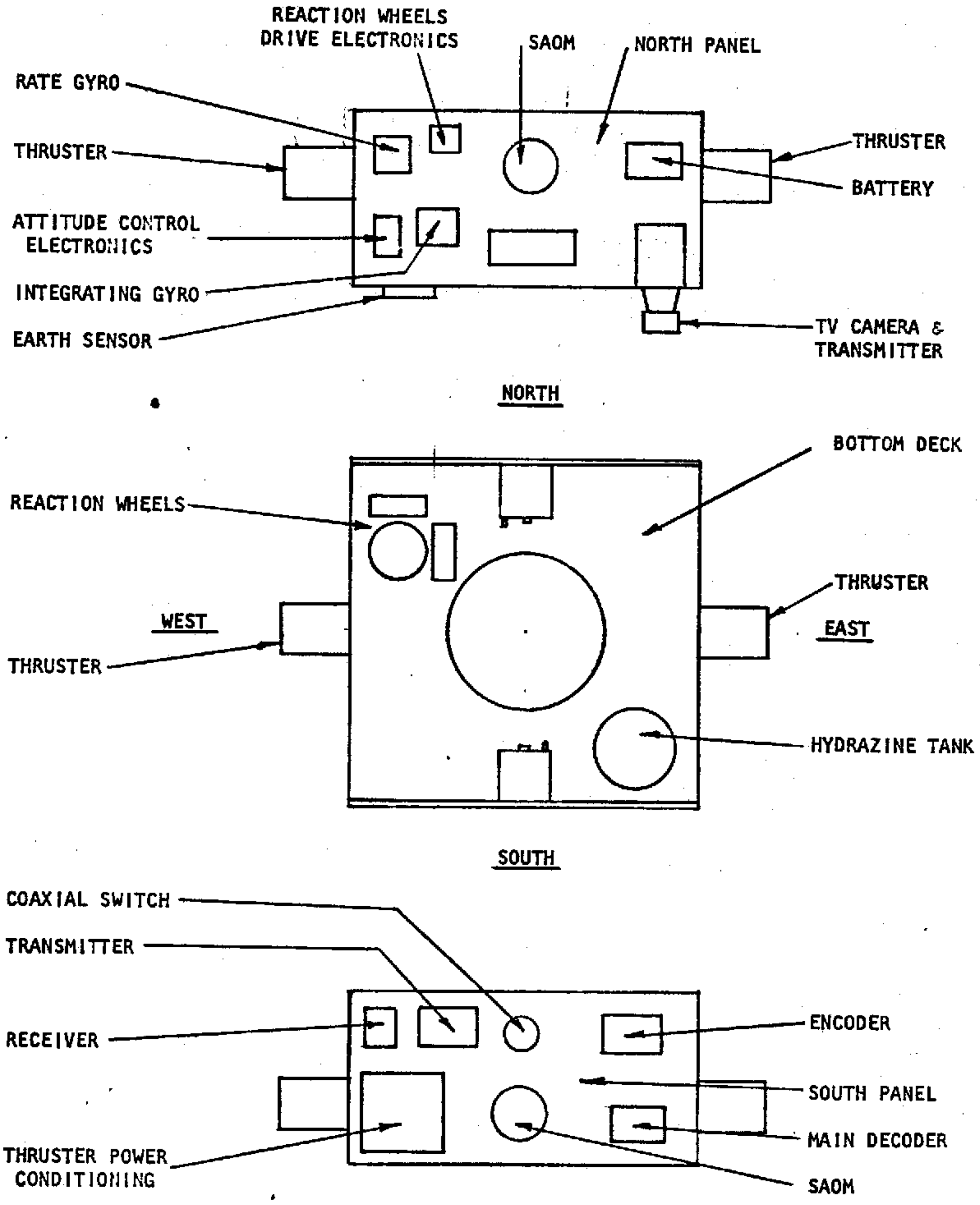


Figure 3.1.2 - Major Component Locations on SERT D Spacecraft

3.2 Configuration Trade Offs

To optimize the mission potential of SERT D and yet remain within the spacecraft launch weight budget, it was necessary to consider potential alternative experiments. The alternative experiments resulted in a number of spacecraft configurations. Two of the configurations are described herein.

Both configurations are basically the same as that shown in figure 1.3.1. The difference being that configuration I incorporates rendezvous radar at the expense of the ion thruster/nickel-hydrogen battery experiment of Configuration II.

Both configurations will accomplish all the stated mission objectives. However, Configuration I require more complicated maneuvers to demonstrate duration testing equivalent to five year mission life and to demonstrate station walking. Both demonstrations require 180 degree yaw maneuvers and involve other complications. Configuration II eliminates the need for yaw maneuvers and increases thruster redundancy. It also provides for a demonstration of powering a thruster from a nickel-hydrogen battery. Table 3.2.1 lists the spacecraft subsystems for Configuration I and associated weights and power requirements. As table 3.2.1 indicates, the TV and radar systems impose a severe weight penalty on the Configuration I spacecraft, reducing the contingency to marginal limits. Several desirable subsystem experiments were eliminated from Configuration I, in order to obtain the weight contingency. These were the NiH_2 battery and associated thruster station keeping demonstration plus the east panel thruster life test. Potential of using the STDN stations to provide the guidance data for the rendezvous maneuvers is under study. Development of this technique would allow the elimination of the radar system. Table 3.2.2 lists the spacecraft subsystems, Configuration II and associated weight and power requirements. As can be noted, eliminating the radar permits incorporation of the thruster/NiH_2 battery experiment and increases the weight contingency to 58.8 pounds.

TABLE 3.2.1 - CONFIGURATION I - WEIGHT AND POWER ESTIMATES (IN ORBIT-GEOSYNCHRONOUS)

SUBSYSTEM	WEIGHT, LB	AVERAGE POWER, WATTS
Structure	160	0
Apogee motor, empty	60	0
Attitude control	100	59.5 (max 100)
Solar blanket	18	0
North array thruster *	13.9	153 [1]
P.C.	9	
South array thruster *	13.9	153 [2]
(hybrid) P.C.	6	
East panel thruster *	17.5	153 [1]
P.C.	9	
HVSA	0	0
SAOM	14	1.5 (max 11.0)
Array structure	72	0
Body-mounted cells	7.6	
Ag-Zn battery	20	20 [3]
Radar *	77	123
TV + transmitter *	25.8	143
TT&C	36	27
Power system	50	27
Thermal control	40	0
Balance weight	10	0
TOTAL WEIGHT	759.7	
WEIGHT CONTINGENCY	27.3	

[1] 150 watts from 70 volt bus + 3 watts from 28 volt + bus

[2] 123 watts from HVSA + 30 watts from 28 volt bus

[3] Charging power

* These devices must share use of 70 volt bus

TABLE 3.2.2 - CONFIGURATION II - WEIGHT AND POWER ESTIMATES (IN ORBIT - GEOSYNCHRONOUS)

SUBSYSTEM	WEIGHT, LB	AVERAGE POWER, WATTS
Structure	160	0
Apogee motor, empty	60	0
Attitude control	100	59.5 (max 100)
Solar blanket	18	0
North array thruster *	13.9	153 [1]
P.C.	9	
South array thruster *	13.9	153 [2]
(hybrid) P.C.	6	
East panel thruster *	17.5	153 [1]
P.C.	9	
West panel thruster	17.5	153 [3]
P.C.	9	
HVSA	0	0
SAOM	14	1.5 (max 11.0)
Array structure	72	0
Body-mounted cells	7.6	0
Ni-N_2 battery	14	20 [4]
Ag-Zn battery	20	20 [4]
TV + transmitter	25.8	143
TT&C	36	27
Power system	50	27
Thermal control	45	0
Balance weight	10	0
TOTAL WEIGHT	728.2	
WEIGHT CONTINGENCY	58.8	

[1] 150 watts from 70 volt bus - 3 watts from 28 volt bus

[2] 123 watts from HVSA, 30 watts from 28 volt bus

[3] Powered by NiH_2 battery

[4] Charging power

* These devices must share use of 70 volt bus

4.0 SUBSYSTEM DESIGN

4.1 Body Cells and Solar Array

SERT D power requirements differ during the mission life. The initial requirement is for power during the transfer orbit. This power will be supplied by body-mounted cells in combination with spacecraft batteries. Once synchronous orbit is achieved, an extendible array will be deployed from the north and south faces of the spacecraft. This array will supply power for the experiments and housekeeping functions and will be continuously oriented facing the Sun. The extendible array is similar in many respects to the CTS array (Fig. 4.1.1). It was selected for the following reasons: (1) The CTS array will be thermally qualified for synchronous orbit in time for SERT D, which is important because lightweight array blankets can reach temperatures as low as -220°C in synchronous orbit during eclipse as compared to approximately -125°C for rigid panel arrays, (2) A pressure plate is extended ahead of the array to act as a spreader bar and will serve as a platform for the array mounted thrusters, propellant and power conditioning electronics.

<u>Body-Mounted Cells</u> - The body-mounted cells must supply approximately 80 watts of power during the transfer orbit phase of the mission. The spacecraft is spinning at 60 rpm and the spin axis may be tilted as much as 30 degrees from the normal to the

spacecraft/sun line. To minimize the area of the array required, it is proposed to use new high efficiency (13% versus 10% for standard cells) violet solar cells. With these assumptions, the total area required to supply 80 watts is approximately 23 square feet or 5.7 square feet per face of the spacecraft.

Extendible Array - The power requirements for the extendible array are 160 to 165 watts EOL (regulated to 29 volts) for housekeeping functions and approximately 150 watts EOL (unregulated 70 ± 20 volts) for experiments. It is proposed that the housekeeping array be the north blanket and the experiments array be the south blanket. The two blanket will be identical in size and consist of eleven active panels, or panels with solar cells, and three blank panels, or uncelled panels, for a total of fourteen panels. The blank panels are required to prevent solar cell shadowing under certain spacecraft attitude conditions. Each panel is approximately 10 inches long by 52 inches wide. The size of each blanket will then be 140 inches long by 52 inches wide. FEP teflon encapsulation solar cell blankets will be used for portions of the array. The following potential experiments will be considered contingent on the available weight margin.

(1) Experiments designed to separate the different types of degradation in synchronous orbit, e.g., UV, electrons, protons, solar flares. These experiments would require approximately 60 solar cells.

(2) Experiments designed to evaluate improvements in silicon solar cells, e.g., back surface field, soluble layer AR coatings matched with FEP covers, shallow junctions, wrap-around contacts. These experiments would require approximately 20 solar cells.

(3) Experimental modules identical to those used on the main array and instrumented to correlate their degradations to the predict degradation of the main array. These experiments would require 27 solar cells.

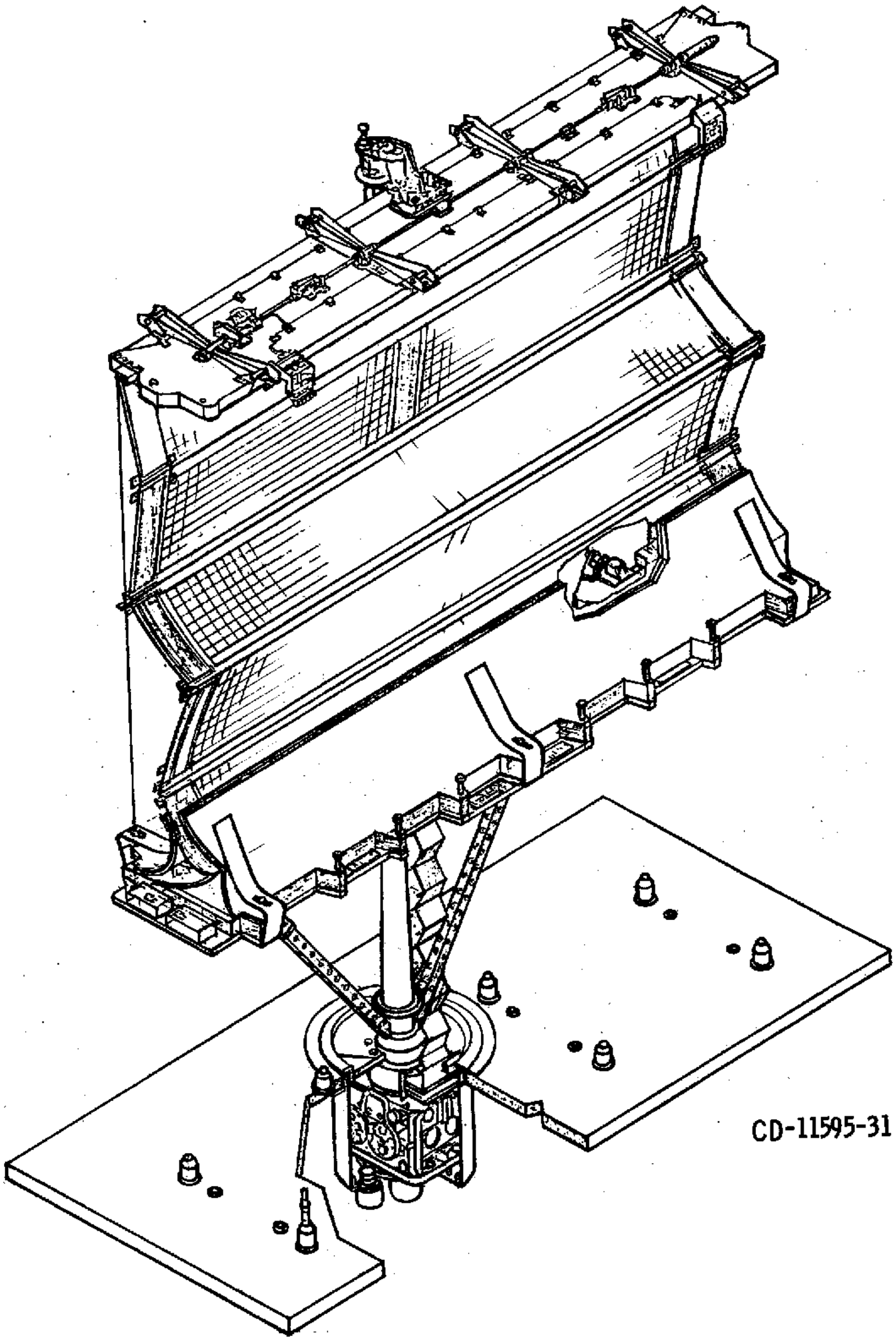


Figure 4.1.1 - Cts array.

4.2 Rendezvous Radar

Microwave Rendezvous Radar Subsystem (MRRS) - In order to provide the capability for rendezvous maneuvers with other synchronous satellites by the SERT D 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 D 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 4.2.1. The system characteristics are summarized in Table 4.2.1. Briefly, the modifications shown in the crosshatched area 1 of Figure 4.2.1 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 4.2.1 and the dish increased to 3-ft diameter. The Electric Assembly modifications for the noncooperative mode in crosshatched areas 3 and 4 of Figure 4.2.1 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 optimized 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 D 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 4.2.2 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 4.2.2 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 4.2.2.

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 4.2.2 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 4.2.2 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	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 D, 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.

Summary - 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 4.2.1 - 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 F.D.*
Minimum range	100 ft
Method of ranging	Phase comparison of transmitted and received tone modulation
Duty cycle	40/60
$P_{transmit}$	See Table 4.2.2
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	±90° each axis
Angle data	
Angular accuracy	Bias = 8 mr (per axis) Random (3σ) = 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 (3σ) = 0.25% at $R > 5$ n mi
Range rate	±4900 ft/sec
Weight	See Table 4.2.2
Power	See Table 4.2.2

TABLE 4.2.2 - 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

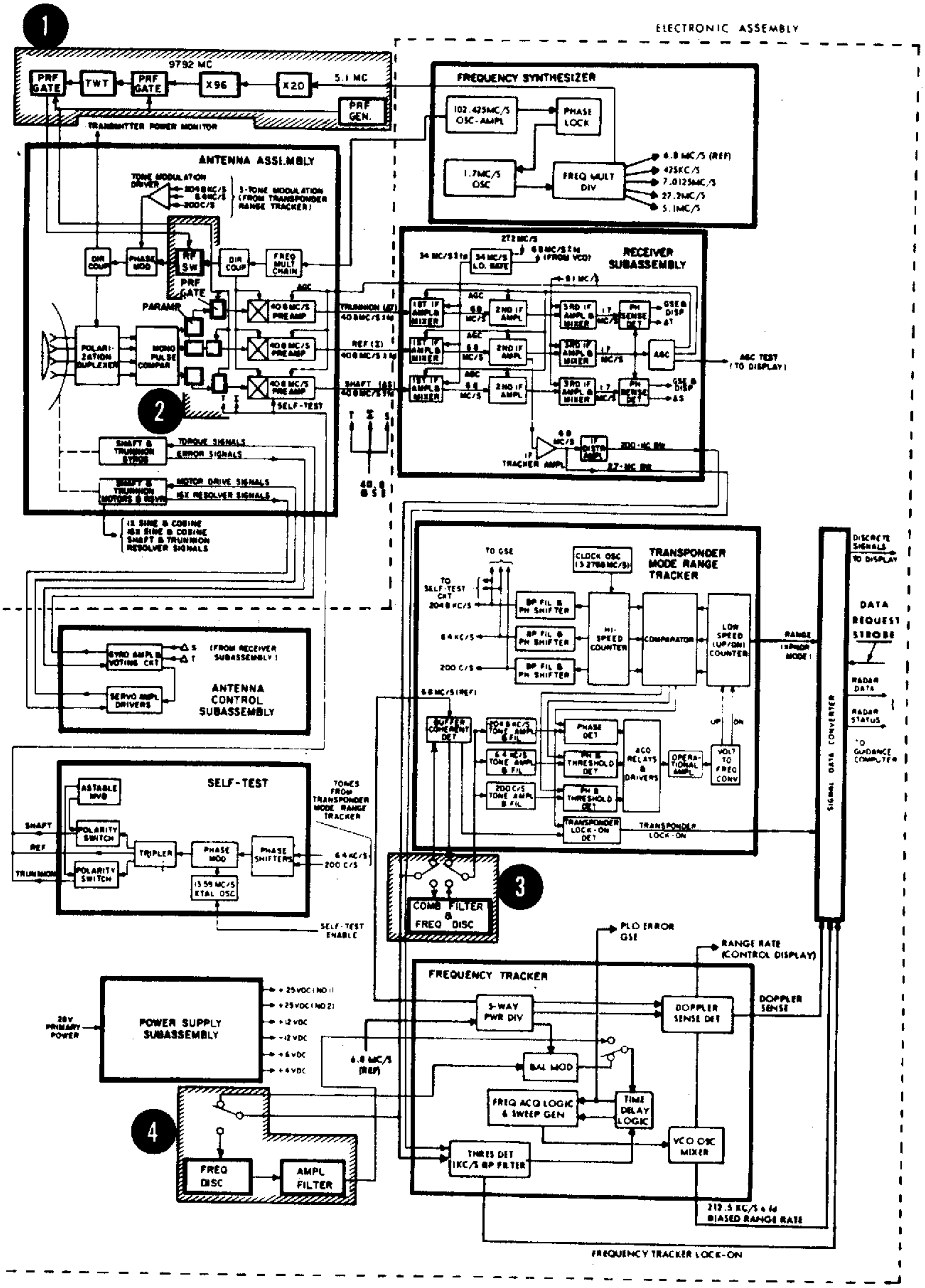


Figure 4.2.1 - Rendezvous Radar Block Diagram

4.3 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 4.3.1. A dimensional view of the GCTA configuration mounted on the lunar rover vehicle is shown in Figure 4.3.2.

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 4.3.1.

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 D), 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 4.3.2). 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 4.3.3.

Ground Command Format - 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 4.3.2 and 4.3.3.

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 4.3.4. 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 4.3.4

TABLE 4.3.1 - 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 -12 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 4.3.2 - 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 4.3.3 - GCTA COMMAND FUNCTION CODES

COMMAND FUNCTION	OCTAL CODE Vehicle Address	OCTAL CODE 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 4.3.4 - 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

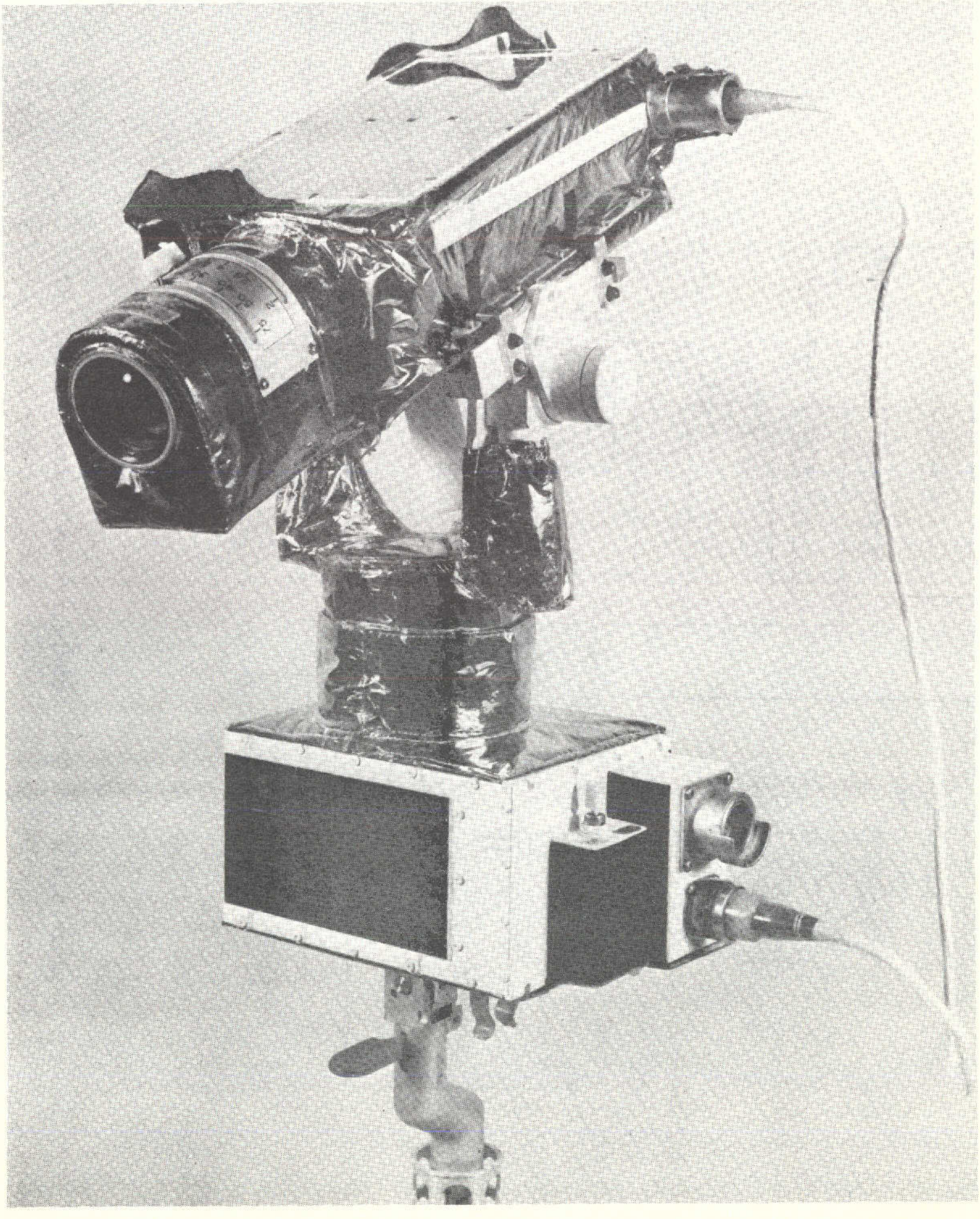


Figure 4.3.1 - RCA Ground - Commanded Television Assembly

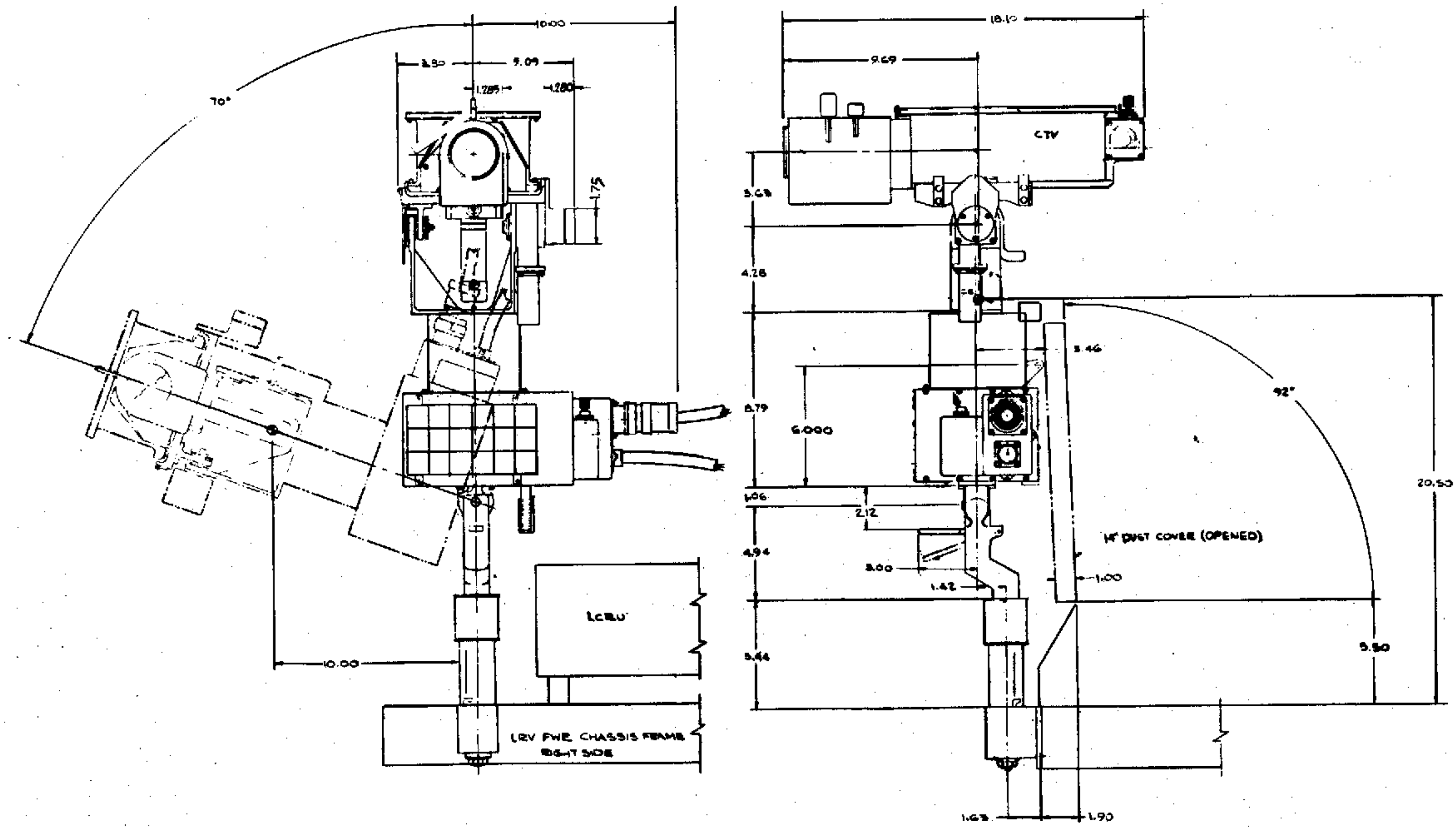


Figure 4.3.2 - GCTA Configuration Mounted on the LRV

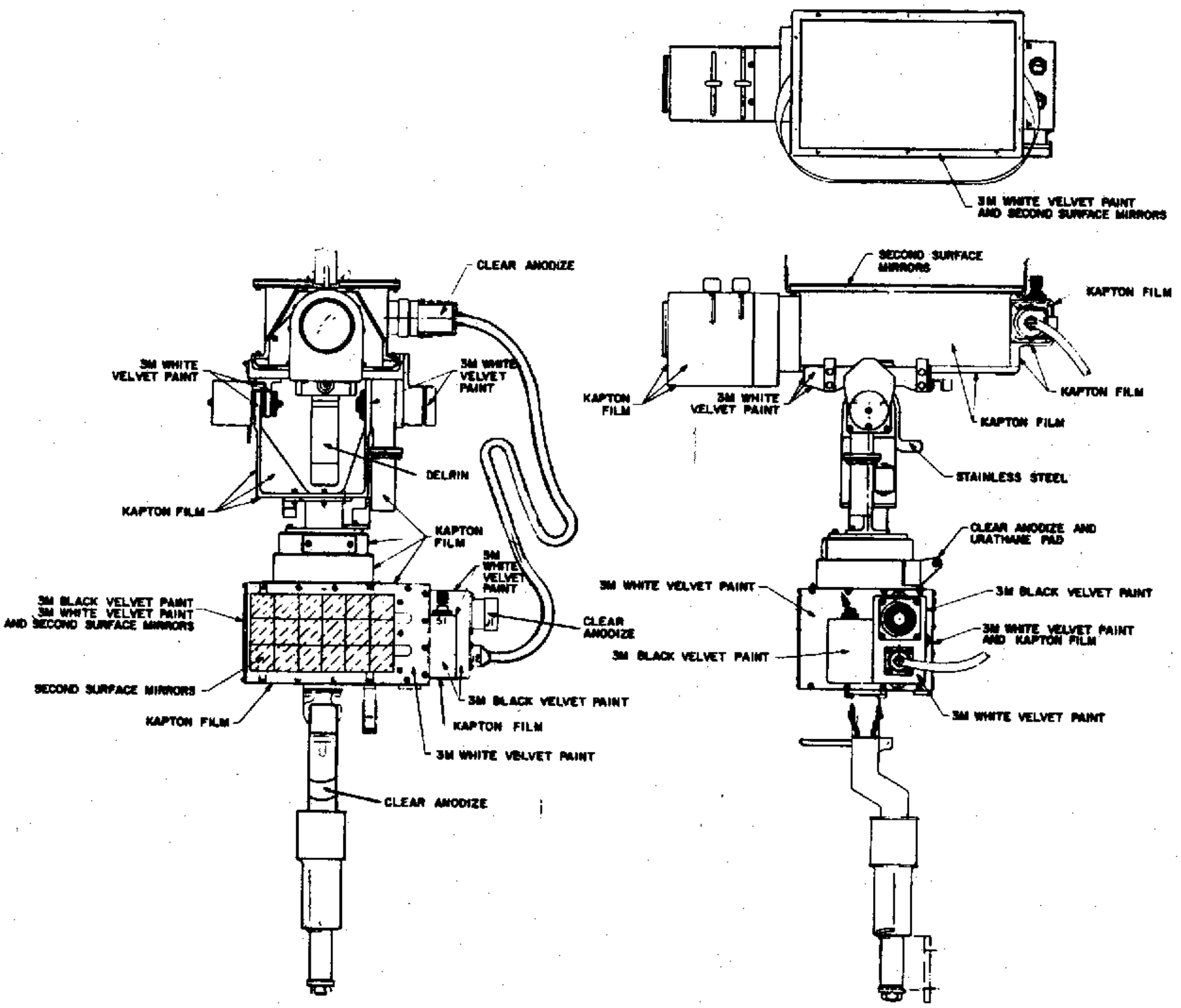


Figure 4.3.3 - GCTA Thermal Control

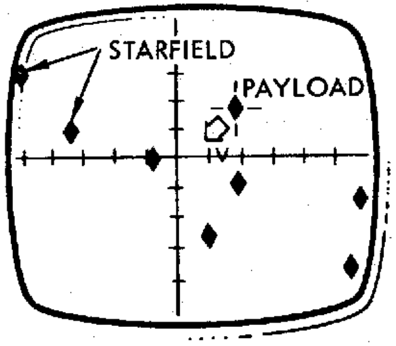
TELEVISION FIELD-OF-VIEW AND RESOLUTION

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

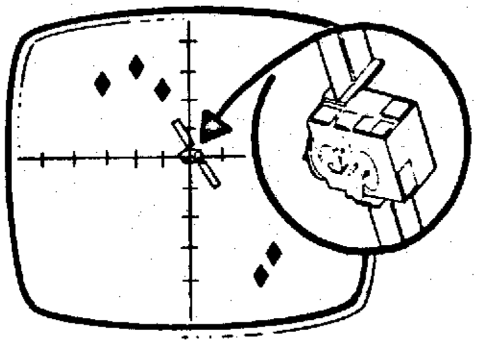
TERMINAL RENDEZVOUS

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

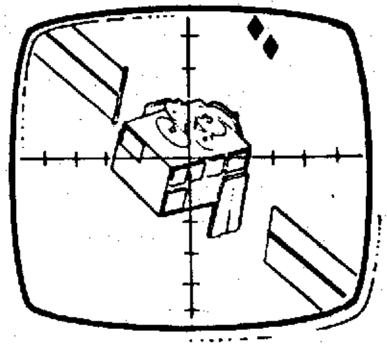


Figure 4.3.4 - Television Field-of-View and Resolution

4.4 Attitude Control

Purpose - The purpose of the attitude control system on the SERT D spacecraft is to provide for the initial acquisition of the on-orbit orientation, for maintaining the spacecraft in its specified on-orbit orientation 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 D 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 in synchronous equatorial orbit is given in table 4.4.1, 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.

System Description - A generalized block diagram of the attitude control and station keeping system is shown in figure 4.4.1. Primary attitude control torques are provided by three reaction wheels, one acting about each spacecraft principal axis. The 8-cm ion thrusters are used to unload the reaction wheels periodically. One thruster each is located on the east and west sides of the spacecraft center body. 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 oriented through the spacecraft center of mass. The array-mounted thrusters will be used to provide N-S station keeping, and to provide torques to unload the roll and yaw reaction wheels. The body mounted thrusters will be used to provide E-W station keeping; to provide a torque to unload the pitch reaction wheel; and will be used for ion thruster life testing. The E-W thrusters will also be used to station walk the spacecraft. The body mounted thrusters also provide a backup yaw torque, although of much smaller magnitude than that of the array-mounted thrusters.

East-west station keeping may also be accomplished with an array-mounted thruster by off setting the spacecraft about the yaw axis through several degrees for a portion of the orbit period. North-south station keeping may be accomplished with both body mounted thrusters deflected north or south. Attitude error sensing is provided by a two-axis earth sensor located on the earth-facing side of the center body, a two-axis sun sensor located on the solar array root, and a single axis integrating gyro in the center body. The earth sensor provides error signals about the roll and pitch axes. The sun sensor provides 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 during initial attitude acquisition.

The attitude control system electronics 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.

Station keeping and wheel unloading are of sufficiently low duty cycle that it seems desirable to command these operations from the ground. Table 4.4.2 shows a weight breakdown of the attitude control system.

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

1. Transfer Orbit
2. Attitude Acquisition
3. Normal On-Orbit
4. Station Keeping
5. Momentum-Dumping
6. Eclipse

7. Station Walking
8. Target Spacecraft Rendezvous and Inspection
9. Reacquisition
10. Backup

Transfer Orbit

During this phase the spacecraft spin axis orientation is determined and the spin axis is precessed into the proper Apogee Motor Firing Attitude (AMFA). This phase is described in more detail in the "Mission Profile" section.

Attitude Acquisition Phase

As soon as possible following apogee injection, the spinning spacecraft will be precessed so that the spin axis (yaw axis) is oriented toward the sun. The spacecraft will be despun using the hydrazine despin thrusters and rate gyro package to about 2 RPM (12 degrees per second), or a value in this range at which the arrays can be safely deployed, and that maintains sufficient angular momentum to insure spacecraft stability. The arrays will then be deployed, and the active side oriented toward the sun. A period of coast time will be provided for battery recharging, prior to completing the acquisition phase.

At the completion of the coast periods, the residual yaw rate will be nulled, and two-axis attitude control initiated using the sun sensor mounted on the array. At the proper time in the orbit, a search for the earth will be initiated by introducing a programed 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.

At this time, the spacecraft will be stabilized in its final orientation about all three axes, with control torques provided by the hydrazine thrusters in an on-off fashion. The array drive will then be initiated, allowing the array to track the sun. Finally, prime control will be switched to the reaction wheels.

Normal On-Orbit Phase

During the normal on-orbit phase, the spacecraft is controlled by the reaction wheels which respond to attitude error signals generated by the roll, pitch and yaw sensors. Because the wheels are proportional devices, the control loop is a proportional one, as opposed to a deadband control loop associated with thrusters used directly as actuators. The 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.

A two-axis sun sensor, mounted on the root of one solar array panel, provides good error signals and hence precise yaw information when the satellite is located at the "dawn" and "dusk" points in the orbit. The error signals degrade as the noon and midnight orbit positions are approached. The yaw information can be extracted from the sun sensor data by calculation, knowing the orbit parameters and sun right ascension and declination. As noted, the magnitude and accuracy of the yaw error signals obtained are dependent upon the spacecraft location in its orbit, and for some period of time, twice each orbit, will decrease to the point where the error signals 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 error signal from the sun sensor 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 the year, which means that under these circumstances it could be turned off, thereby increasing confidence in its ability to operate for the entire mission.

Station Keeping Operation

The station keeping operations will be accomplished by operating the 8-cm ion thrusters on a periodic basis to provide translational impulses to the spacecraft as previously described. 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 ion thrusters are located on the ends of relatively flexible 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 step disturbance torque which will affect the spacecraft attitude, producing a transient which must be compensated by the wheels to saturate quicker, in the worst case, and result in earlier initiation of the momentum-dumping mode.

If, on the other hand, the ion thrusters are used as the primary actuators in the control loops during station keeping; 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, the attitude transient resulting from the switch over should be of small magnitude. The latter method, therefore, appears to be the best one. However, a detailed analysis must be made taking into consideration such things as attitude transient characteristics, settling times, and overall stability before a final decision can be made.

Momentum Dumping Operation

With the attitude disturbance torques shown in table 4.4.1, wheel unloading about the roll and yaw axes would be required once each day for about 4 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 10 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 null 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 little torque margin for this latter purpose. 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 in the normal mode. Yaw sensing will be provided by gyro since the sun sensors will be inoperative. Station keeping 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 will be modified so that the operation is accomplished with a single impulse which occurs in sunlight.

Station Walking

During the station walking operation, the attitude control system operates as in the station keeping mode, the main difference being that the 8 cm thruster operates continuously. The operating body-mounted thruster will provide primary control torques about pitch and yaw, and the roll reaction wheel provides a control torque about roll. Since very little beam deflection (less than 1 degree) is generally required for attitude control, the effect on station walking of using the thruster for attitude control will be very small. The roll reaction wheel is dumped by one of the array mounted thrusters.

Target Spacecraft Rendezvous and Inspection

This phase is initiated when the SERT D 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 sensor or earth sensor, or a combination of these. Furthermore, for the terminal "close in" maneuvers, it may be desirable or necessary to reactivate the high-thrust hydrazine 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. The spacecraft condition from which reacquisition must occur will depend on how much time elapses after loss of orientation before reacquisition is begun. In general, however, the spacecraft will be left with an attitude substanially off normal and either tumbling or oscillating with relatively large rotational velocities. The worst case situation is one in which 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 and the reaction wheels must be used to null spacecraft rates based on information from the rate gyro package. 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 trade off 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 station keeping maneuvers, as well as roll and yaw dumping. The body mounted thrusters provide redundant primary pitch wheel dumping torques and also provide backup torques for yaw wheel dumping. North-south as well as east-west station keeping can be done by the body mounted thrusters and/or the array mounted thrusters. The hydrazine system can provide backup control torques about all three axes, in the event of reaction wheel failure, although at the expense of mission time and attitude accuracy, as can the ion thrusters via beam deflection. The sun sensor can provide backup pitch error information, as well as roll 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 4.4.1 - DISTURBANCE TORQUE PEAK VALUES

SOURCE	ROLL (FT-LB)	PITCH (FT-LB)	YAW (FT-LB)
SOLAR PRESSURE	$7.6\text{x}10^{-6}$	$3.48\text{x}10^{-6}$	$7.6\text{x}10^{-6}$
MAGNETIC	$4\text{x}10^{-7}$		$4\text{x}10^{-7}$
GRAVITY-GRADIENT	$5\text{x}10^{-7}$	$9\text{x}10^{-9}$	$5\text{x}10^{-8}$

TABLE 4.4.2 - ATTITUDE CONTROL SYSTEM WEIGHT AND POWER BREAKDOWN

COMPONENT	WT(LB)	POWER (WATTS)
Spinning Earth Sensors (2)	3.7	2.6 Spin Phase Only
Spinning Sun Sensor	1.2	.3 "
Coarse Sun Sensor	0.1	Acquisition Only
Earth Sensor (2 - Axis)	7.2	1.6
Sun Sensor (2 - Axis)	0.7	1.0
Rate Integrating Gyro	3.6	45 start 12 run
Rate Gyro Package	2.5	13 acquisition only
Reaction Wheels	20.0	23.6 peak
Attitude Control Electronics	12.5	10
Hydrazine Tanks (2)	5.8	-
Hydrazine Propellant	17.4	-
Hydrazine System Hardware (Thrusters, lines, filters, etc)	22.0	12
Wobble Dampers	1.5	-
TOTAL SYSTEM WT	98.2	

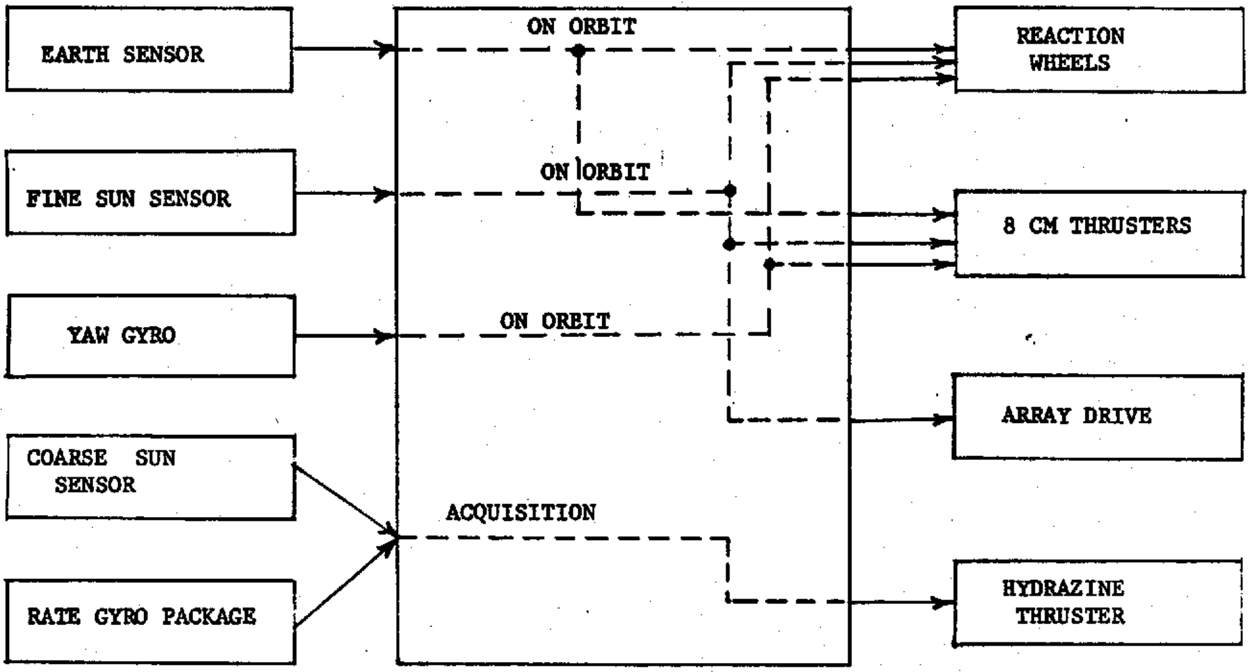


Figure 4.4.1 - Block Diagram of Attitude Control & Station Keeping System

4.5 Electric Propulsion

Thruster Description - Each of the 8-cm thrusters and propellant tanks are identical in design. Figure 4.5.1 is a cutaway of an 8-cm thruster. The two axis, ±10°, vectorable dished accelerator grids have an active area 8-cm in diameter and a design life of 20,000 hours. The actual diameter of the discharge chamber anode is 8.4 cm. Main and neutralizer cathodes are hollow cathodes with enclosed keeper 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 also equivalent to SIT-5 design and contain a flow isolator capable of withstanding 100 V.

A single propellant tank for each thruster is located at the side of the thruster to minimize overall thruster system length. A short thruster length facilitates mounting design of a thruster system on the end of a solar array and is necessary for the proposed SERT D spacecraft design to fit within the launch vehicle shroud. The propellant tank design uses a flexible diaphragm and gas pressure to feed propellant as in the SERT II thruster feed system. The propellant tank is sized for the thruster requiring the largest propellant mass. When less propellant is loaded for other thrusters, a propellant tank retainer sleeve will be used to control the propellant position during launch. Table 4.5.1 gives weights of thruster, tankage, and propellant for each thruster. Table 4.5.2 lists presently obtained thruster operating conditions. The values of table 4.5.2 were used to design the SERT D mission. If performance improvements are made, the propellant weight saving may be off loaded or retained as propellant reserves.

Thruster Power Processor Description - 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 from an unregulated 70 volt solar array bus, and less than 3 watts from the 28 volt regulated bus. The processor is capable of providing full power to the thruster for bus voltage swings of ±20 volts from nominal.

The digital interface and control subsystem can receive 16 parallel bit digital "value words" by means of a serial to parallel converter. 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 with 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 require five watts of standby power to prevent its temperature from dropping below -40° C.

Thruster Functions - The nominal 8-cm thruster operating condition produces a thrust of 1 mlb when corrected for double ions and beam divergence. The main function of the solar array mounted thrusters will be to perform N-S station keeping of the SERT D spacecraft for 5 years. One north thruster and one south thruster will fire alternately once per day for approximately 1.4 hours. The south thruster runs off the hybrid power high voltage solar array. The propellant loading of either north or south thruster is sufficient to do N-S station keeping for 5 years. Should either thruster fail, the remaining thruster could fire approximately 2.8 hours once per day for half of the days remaining in a 5 year mission. Full backup redundancy could be obtained by loading extra propellant into the north and south thruster tanks at the rate of 1 pound for every year of desired backup redundancy.

Roll and yaw reaction wheel unloading will be performed by vectoring the north or south thruster while it is thrusting for station keeping. This function for 5 years uses 0.4 pound of the 5.4 pound total propellant loaded for each north and south thruster. If 1.6 pounds additional propellant were loaded per thruster, the N-S thruster could perform a 5 year backup mission for E-W triaxiality station keeping. No separate E-W solar pressure station keeping is needed for the present spacecraft mass area ratio and station accuracy of 0.05 degree. A list of various functions to be performed by the thrusters and possible backup in the event of failure is shown in Table 4.5.3.

Body Mounted Thrusters - The spacecraft body mounted thrusters will fire east and west. The body mounted thrusters will be used to perform an accelerated life test of an ion thruster simulating north or south station keeping for a 1000 pound satellite in synchronous orbit for 5 years (1826 cycles). During this experiment the thruster will be on for 1.78 hours and off (to cool down) for 1 hour. The calendar time to perform this experiment is 0.6 years and the thruster will operate for a total of 3250 hours. This experiment can be combined with a station-walking experiment.

Additional propellant has been added to the body thrusters to perform the additional functions below:

1. Dump pitch momentum wheels for 5 years (0.5 pound of propellant)

2. Do E-W triaxiality station keeping for 5 years (0.4 pound of propellant)
3. Perform one later station-walking mission of 180° (1.0 pound of propellant)

Solar Array Mounted Thrusters - The thruster system will be mounted on a pressure plate (called palette) used to clamp the folded solar array in a stowed position during launch. As the solar array unfolds, the thruster system will be automatically deployed with the array to the position shown in figure 1.3.1

The center of the thruster and its thrust vector will be positioned directly over the end of the bi-stem boom and pointed through the spacecraft center of mass.

The thruster propellant tank will be mounted close to and on one side of the thruster. This position is necessary to reduce overall thruster length commensurate with a spacecraft design fitting within the launch vehicle shroud. 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 also will be mounted on the array palette on the other side of the thruster. The general shape of the power processor will be that of a square pancake. 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 tank 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 coupling of the neutralizer line to the thruster system may be employed.

TABLE 4.5.1 - THRUSTER WEIGHTS AND OPERATING HOURS

	North thruster (solar array mounted), lb	South thruster (solar array mounted), lb	East thruster (S/C body-mounted), lb	West thruster (S/C body-mounted), lb
Thruster body	5.1	5.1	5.1	5.1
Propellant tank	1.6	1.6	1.6	1.6
Propellant	5.4	5.4	8.1*	8.1 *
Power processor	9.0	8.0	9.0	9.0
Total weight	21.1	20.1	23.7	23.7
Total hours of thrust	2690†	2690†	4200‡	4200
Typical duty cycle, hrs	1.4	1.4	1.78	1.78

*6.2 lbs of propellant used in accelerated life test

†Includes 1826 cycles.

‡Includes 1826 cycles and 3250 hrs for accelerated life test.

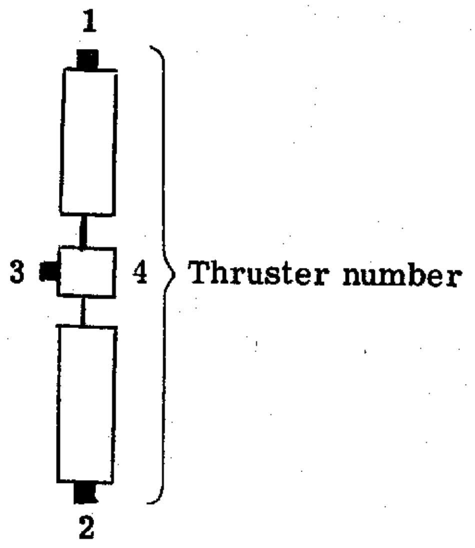
TABLE 4.5.2-MEASURED 1-MLB 8-CM ION THRUSTER OPERATING CONDITIONS

(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

TABLE 4.5.3 - THRUSTER FUNCTIONS
SERT D Dual Axis Vectorable Thrusters



Function	Prime	First backup (prime failure)	Second backup
Station keeping:			
N-S	1 & 2	1 or 2 for half of time remaining in 5 yr	3 or 4
E-W	3 & 4	(1 & 2)*	(1 or 2)*
Momentum wheel dumping:			
Roll	1 & 2	(1 or 2)†	Hydrazine
Pitch	3 & 4	3 or 4	None
Yaw	1 & 2	(1 or 2)†	3 4
Station walking:			
E-W thrusting	3 & 4	(3 or 4) and (1 and 2)	(1 and 2) or (3 or 4) (1 or2)
Cycling endurance test	3 & 4	None	None

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*If required, N-S station keeping will be reduced up to 1.6 years.
†If required, N-S station keeping will be reduced up to 0.4 years.

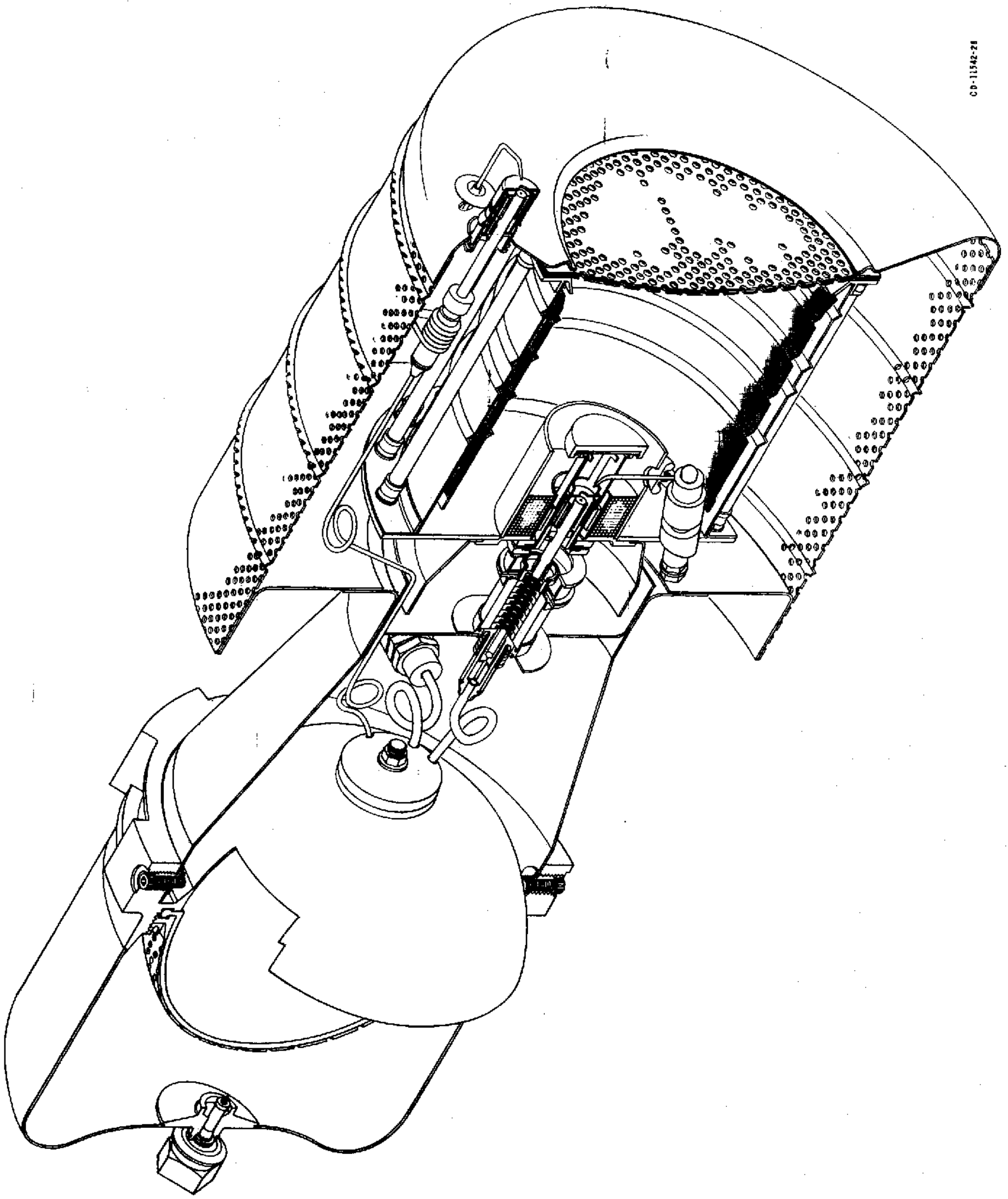


Figure 4.5.1 - 8-cm Ion Thruster Cutaway Drawing

4.6 Electrical Power

The SERT D electrical system is basically made up of 2 (two) power buses: a regulated 28 volt housekeeping bus and an unregulated 70±20 volt experiments bus. To power these two buses there are two solar arrays which each have approximately 160 watts capacity. When the south ion thruster is operating, part of the 70 volt array is reconfigured into a high voltage solar array (HVSA) that is directly tied to the thruster. The other three ion thrusters on SERT D use the 70 volt bus for almost all their power needs.

Before the solar arrays are extended, the housekeeping bus will be supported by a body-mounted solar cell which has a power output of 80 watts. This power is lost when the main solar arrays are extended.

A block diagram of the power system is shown in figure 4.6.1. From this diagram the different spacecraft power loads can be distinguished. Basically, they are the ion thrusters, telemetry and attitude control system, and satellite surveillance system (radar, if required, and television system). The telemetry and attitude control system comprise the largest continuous load requirement from the 28 volt bus during all phases of the mission. The telemetry system requires 27 watts at 28 volts at all times other than eclipse periods at which time the power requirements are reduced to 4.5 watts at 28 volts. The attitude control system power requirements can go from a low of 55.5 watts to a high of 94.5 watts during an eclipse and to a low of 73.5 watts to a high of 119.5 watts during solar array extension.

The surveillance system consists of radar (if required) and a television camera and transmitter. This system is used during the terminal rendezvous phase of the mission where the radar is used to locate the object of interest and the television camera and transmitter sends pictures back to a ground station. The radar is never on while the television camera and transmitter are on. The radar requires 123 watts at 28 volts while the television system requires 143 watts at 28 volts. They are powered from a separate regulator which uses the 70 volt solar array as the power source.

There are four (4) ion thrusters on SERT D. There are north and south thrusters mounted on the solar array and east and west body-mounted thrusters. The east body-mounted thruster and the north thruster require 150 watts from the 70 volt unregulated bus and 3 watts from the 28 volt regulated bus. The south thruster, as mentioned earlier, is operated off a reconfigured 70 volt array; however, it also requires 30 watts from the 28 volt bus. The west body-mounted thruster will operate from a nickel-hydrogen (Ni-H_2) battery which is part of an experiment aboard SERT D. A description of the Ni-H_2 battery system can be found in Appendix C.

The block diagram of the power system for SERT D shows the main units that are necessary to regulate and control the power for the spacecraft. These include housekeeping solar array regulator, battery, battery discharge regulator, and a battery charger. More will be said about each of these units later.

The SERT D mission will be made up of a number of different phases. The main phases relative to power requirements are launch, transfer orbit, apogee motor burn, three-axis acquisition and subsequent operation. Batteries supply spacecraft power during the launch phase. Batteries and body mounted solar cells supply power during the transfer orbit and apogee motor burn as well as during part of the three-axis acquisition phase. Batteries alone are used during part of the three-axis acquisition phase. Batteries and the solar arrays provide power during the latter portion of the three-axis acquisition phase. Subsequent synchronous operations utilize the solar arrays and batteries for power.

A natural evil of synchronous orbits is a seventy-two (72) minute eclipse. During this period only the essential systems are maintained off the batteries. These systems include the attitude control and telemetry systems, both of which comprise the majority of the power loading during eclipse.

Another phase of the mission is a rendezvous with another spacecraft. This phase has already been described earlier in this report.

From a power profile plot such as figure 4.6.2, the power constraints can be determined. The major constraint is the size of the arrays. The housekeeping array will not support either the radar or the television system. Because of the size of the 70 volt array, only one ion thruster can operate from it at any one time. If the radar and television system are powered from the 70 volt array, only one of the two can operate at any one time. An ion thruster and either the radar or television system cannot operate at the same time from the 70 volt array. Care has to be taken so that the battery does not become too deeply discharged. Time has to be allowed for the battery to be charged. Table 4.6.1 supplements figure 4.6.2 and shows the electrical loads during the various phases of the mission.

Up until now this proposal has dealt mainly in generalities concerning the hardware of the power system for the SERT D spacecraft. At this time a more detailed discussion of the components of the power system is offered. Some of the parameters that affect the design of the power system are total power required from the solar array, total power required from the battery, total weight allowed for the power system, efficiencies of the individual components (how much power will be dissipated), reliability, and off-the-shelf versus development items. In the case of SERT D the major guiding factor has been the weight allowed for the power system. Through calculations it was determined that about twenty-five

(25) pounds of harness would be needed. In addition, forty-five (45) pounds were allowed for the battery and the electronics of the power system. In order to maximize the weight available for the electronics, the use of lightweight batteries was explored. The one part of the power system which did not change no matter what was used for the other components was the power conditioner for the integrating gyro. The power conditioner is required 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. Therefore, the allowable weight for the battery and electronics was down to thirty-seven (37) pounds.

In researching power systems for this proposal a complete, flight-qualified system which was built by Philco-Ford for Goddard Space Flight Center was discovered. This system includes the solar array, main regulator, battery, battery charger, and battery boost regulator. Any part or all of this system can be purchased. The entire system minus the solar array weighs 27.76 pounds which is very light and, in essence, is the system proposed for SERT D. The power capability of this system can handle the requirements for SERT D and the designed lifetime of five years also meets a requirement of SERT D. Responsible for working on this power system at Philco-Ford Corporation are D. C. Briggs and H. N. McKinney. The same power system is part of the Synchronous Meteorological

Satellite (SMS). It relies on a partially shunted solar array for regulation. The solar array is tapped so that by the end of the mission the power required is equal to maximum power available from the array. In this way the shunt regulator will essentially not be needed at the end of the mission. Since it uses a shunt regulator, there will be some power dissipation during the mission. However, this dissipation is minimized by the partial shunt arrangement. Also, there are twelve (12) dissipative elements which can be placed anywhere in the spacecraft. The system has built-in redundancy. An interesting alternative to the Philco-Ford system is discussed in Appendix A.

By using the SMS power subsystem with its total weight of 27.76 pounds there is 9.24 pounds left for power conditioning for telemetry and a regulator for the radar and television system. The telemetry system requires a number of voltages at a total power of 27 watts. To handle these requirements, it was decided to use Powercube 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 - 27G100W40 generator, 1 - 12TRC10 regulator, 1-6TRC10 regulator, 1 - 3/5TRC10 regulator, 1 - 15/16.5 TRC10 regulator, and 1 - 8TR23 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 2.1 pounds. The efficiency would be around 90 percent.

There is now 7.15 pounds remaining from the original 45 pounds for the power system. The remaining parts of the system to be accounted for are a regulator for the radar and television system, switches, and fuses. The choice for the regulator are again Powercube "Circuitblock" modules because of their lightweight and flexibility. The maximum power required from the regulator is around 150 watts at 28 volts. Two (2) Powercube generators, each rated at 90 watts output and three (3) Powercube regulators, each rated at 65 watts output would handle the radar and television system. Total weight for this regulator would be 14 ounces or .875 lbs. This leaves around 6.27 lbs. for switches and fuses.

As can be seen from the block diagram of figure 4.6.1, 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 something goes wrong with them in that they start drawing excessive power, then they become expendable. These units also happen to be the largest users of power on board the spacecraft. If the Philco power system is used, then only one (1) additional relay will be needed in order to switch power conditioners for the telemetry system. A relay weighs around one (1) ounce. Therefore, there is a possibility of having around a six (6) lb. weight contingency in the power system. Table 4.6.2 gives a weight breakdown for the SERT D power system.

If the Philco-Ford power system is not used, there are several systems which could be used. However, none of them have the nice feature of being already integrated into a lightweight, simple package.

As noted previously, to help alleviate the weight problem with the power system, special lightweight batteries were investigated. The battery which holds the most promise at this time of being the lightest at a given power capability is the silver-zinc. However, past performance, especially in lengthy missions has not helped the silver-zinc battery's reputation. Lewis Research Center has done a great deal of work with this battery and is a strong advocate of its use in future missions. The Lewis group currently investigating the silver-zinc battery have provided a proposal for the SERT D mission which is presented in Appendix B.

Silver-zinc (Ag-Zn) batteries are sensitive to over charging. It is the main cause of battery failure. The recommended method for charging Ag-Zn batteries is by a two step charger. The following battery charger has been widely used in charging Ag Cd batteries. It can be adapted for charging Ag Zn batteries since they both require two (2) step charging. The voltage levels indicated are only used for reference and are not meant to be the levels that would have to be used for SERT D.

In essence it consists of initially applying maximum available rate of charge (up to about C/10) until a preselected voltage level is reached after which the charge voltage is clamped at this point until full charge is approached and the charge current has dropped to C/100. At this time the voltage point at which the charging is clamped is changed to 14.1 volts, and remains there until a discharge interval is required. Then the cycle is repeated. This technique has been found to greatly reduce gas formation during trickle conditions.

The battery charge control system of figure 4.6.3 is designed to provide charge currents up to the voltage limits peculiar to silver cadmium cells. The "full charge" current is equal to the solar array current less the load current up to the upper battery voltage limit at which point the battery is in a constant voltage charging mode. Continued operation in this mode reduces the battery current to a predetermined level at which point the constant voltage regulator switches down to the lower voltage limit. The battery then draws the current required to maintain it at this lower voltage level. Upon going into a discharge the system is reset and the charge cycle repeats itself at the onset of recharge. Transistor Qi is the charging path and is controlled by the upper and lower level voltage detectors. The OP Amp I.C. detectors utilize resistor divider sensors and compare the divider voltages to the Zener voltage of CR2. Their outputs are "OR" ed at the base of the driver transistor Q2. One detector is always disabled as determined by the state of the "Charge Voltage Level Selector" flip-flop. Both are disabled during discharge by the Discharging Detector output which sets the flip-flop to the lower voltage detector disable state and also drives the base of Q3 disabling the upper voltage detector. The state of the flip-flop is determined by the Discharging Detector and the Charging Detector, whichever was in the "ON" state last. Their input is the voltage drop across the battery current shunt located between cells number 5 and 6. For this battery the upper voltage limit has been selected to be 15.4 volts and the lower is 14.1 volts.

The sequence of operation starting in the discharging mode is as follows: During discharge the Discharging Detector is producing an output that has signaled the flip-flop to the 14.1 volt disable state and is keeping Q3 in the on state which in turn is disabling the 15.4 volt detector. At the end of discharge and at the start of the charge cycle, the Discharging Detector senses the current reversal in the shunt and its output goes to the RTN potential thus turning Q3 off and enabling the 15.4 volt Detector. As long as the battery voltage is below 15.4 volts, transistor Q1 will be saturated thus allowing the full array current less the load current to be put into the battery. When the battery voltage becomes 15.4 volts and 15.4 volt Detector will regulate the drive to Q1 such that the battery voltage is regulated at 15.4 volts.

In this mode the current required to keep the battery at the upper voltage level will reduce with time until at 100 ma, and for the delay time determined by C1, the "100 ma Charging Detector" will produce an output that toggles the flip-flop disabling the 15.4 volt Detector and enabling the 14.1 volt Detector. The 14.1 volt Detector then regulates the drive to Q1 such that the battery voltage is regulated at 14.1 volts. Regulation continues until a discharge, for the delay time as determined by C2, occurs. At this point the "Discharging Detector" produces an output that toggles the flip-flop back to the 14.1 volt disable mode and turn on Q3 also disabling the 15.4 volt Detector, thus Q1 drive current does not produce power losses in the drive circuitry. The battery discharges into the load through germanium diode CR1.

In referring back to the block diagram of the power system, figure 4.6.1, one can see that it may be necessary to be able to switch the battery cells from a series combination for discharging to a parallel combination for charging. This is a result of the battery having a higher open circuit voltage than the housekeeping bus. However, the battery voltage cannot be indiscriminately lowered since the efficiency of the boost regulator suffers. A compromise is required but it would be advantageous to be able to charge and discharge the battery cells while they are in a series connection.

Limited power budgets on small spacecraft present time sharing problems which force compromises in the exercise of certain subsystems. SERT D is no exception; the power budget precludes the operation of the north, south, and east thrusters at the same time. They all require power from the same solar array which can handle only one thruster at a time. It has been proposed that the west thruster operate from a battery system. Table 4.6.3 indicates the 8-cm thruster power requirements for SERT D. A conventional battery system for this duty cycle, with a five year mission life, would be prohibitively heavy for this spacecraft.

An experimental battery system employing nickel and hydrogen gas as the electrodes appears to have the capability to perform in this manner without excessive mass. Such an experimental system has been proposed by COMSAT and will be considered for SERT D. This experimental battery system is described in Appendix C.

TABLE 4.6.1 - SERT D POWER REQUIREMENTS

	Launch	Transfer orbit	Apogee motor burn	Orient. and Despin	Solar array ext.	Ion thruster oper.	Geo-sync. eclipse	Terminal rend.
TT&C	27/27	27/27	27/27	27/27	27/27	27/27	4.5/4.5	27/27
Integrating gyro					24.5/57.5	24.5/57.5	24.5/57.5	24.5/57.5
Attitude control		17.4/24.4	17.4/24.4	56.4/78.4	49.0/62.0	35/41	31/37	31/37
Solar array orientation mech.					1.5/11.0	1.5/11.0	1.5/11.0	1.5/11.0
Ion thrusters	3/3	3/3	3/3	3/3	3/3	153(1)/153	3/3	3/3
Radar								123(2)/123
Television camera and orientation	5/5	5/5	5/5	5/5	5/5	5/5	5/5	33(3)/33
Television transmitter								110(4)/110

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

Note: (1) 150 W from 70-volt bus, 3 W from 28-volt bus; (2), (3), and (4) all from 70-volt bus.

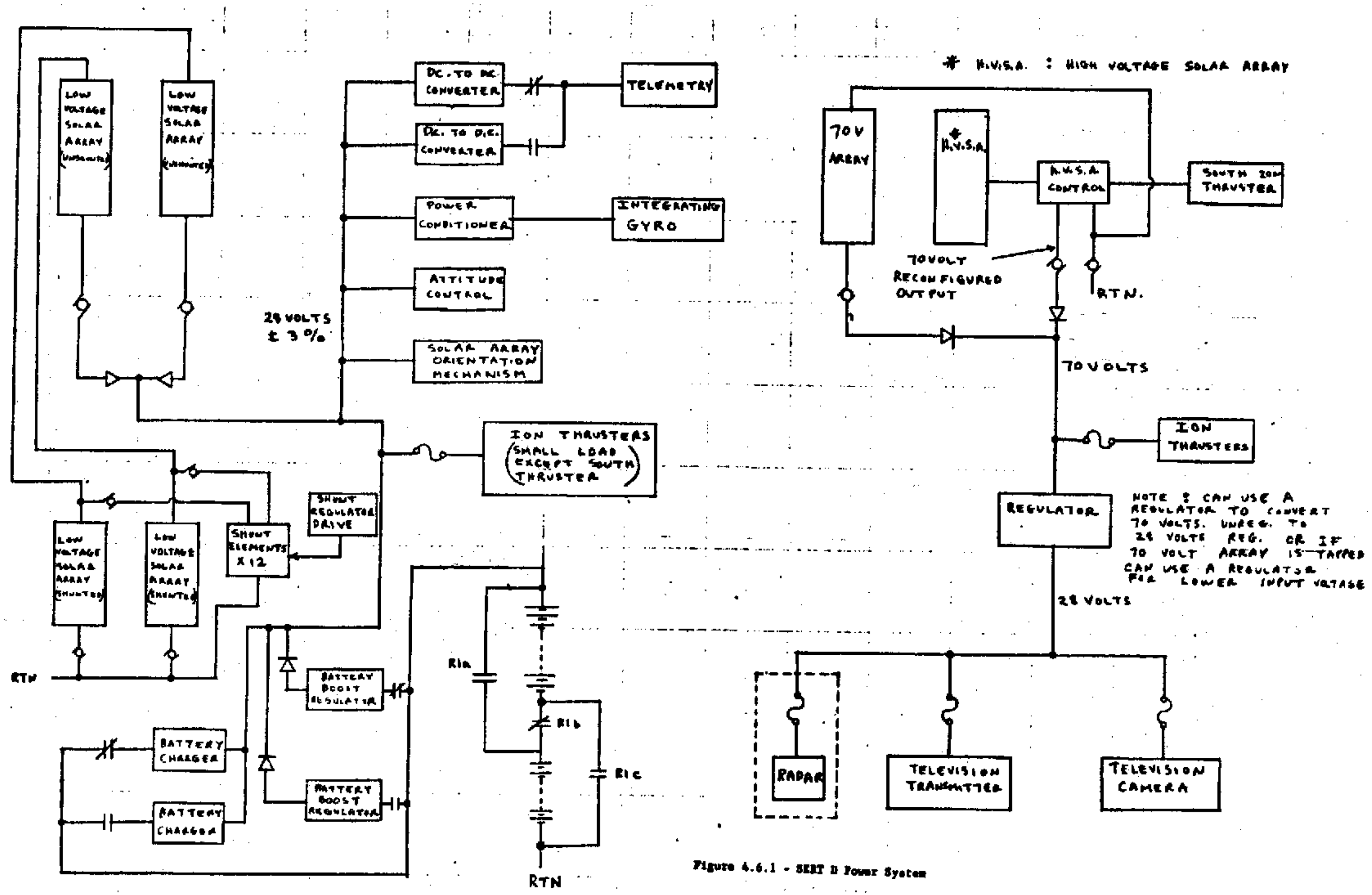
TABLE 4.6.2 - WEIGHT TABLE FOR THE SERT D POWER SYSTEM

Total Allowed Power System Weight	45.0 lbs
Main Regulator, Battery, Battery Charger, and Battery Boost Regulator (Philco-SMS System)	-27.76
Power Conditioner for the Integrating Gyro	- 8.0
Power Conditioner for the Telemetry System	- 2.1
Regulator for the Radar and Television Systems	- 0.875
Wiring Harness	-25.0
Fuses, Switches, etc.	6.26

Table 4.6.3 - 8-CM THRUSTER IMPULSE REQUIREMENTS FOR SERT D

OPERATION	THRUSTERS USED	THRUSTING TIME FOR THRUSTER PER CORRECTION	TIME BETWEEN CORRECTIONS
North-South Station Keeping	North & South	1.3 Hours	1 day
East-West Station Keeping (Solar Pressure)	East & West	45 Minutes	7 days
East-West Station Keeping (Triaxiality)	East (2 impulses)	22 Minutes per Impulse	7 days
Momentum Dumping (yaw - Roll)	North or South	4 Minutes	1 day
Momentum Dumping (pitch)	East or West	10 Minutes	1 day

NOTE: Above Assumes No Lifetest is Being Performed with Any Thruster



NOTE : CAN USE A REGULATOR TO CONVERT 70 VOLTS. UNREG. TO 28 VOLTS REG. OR IF 70 VOLT ARRAY IS TAPPED CAN USE A REGULATOR FOR LOWER INPUT VOLTAGE

Figure 4.6.1 - SERT D Power System

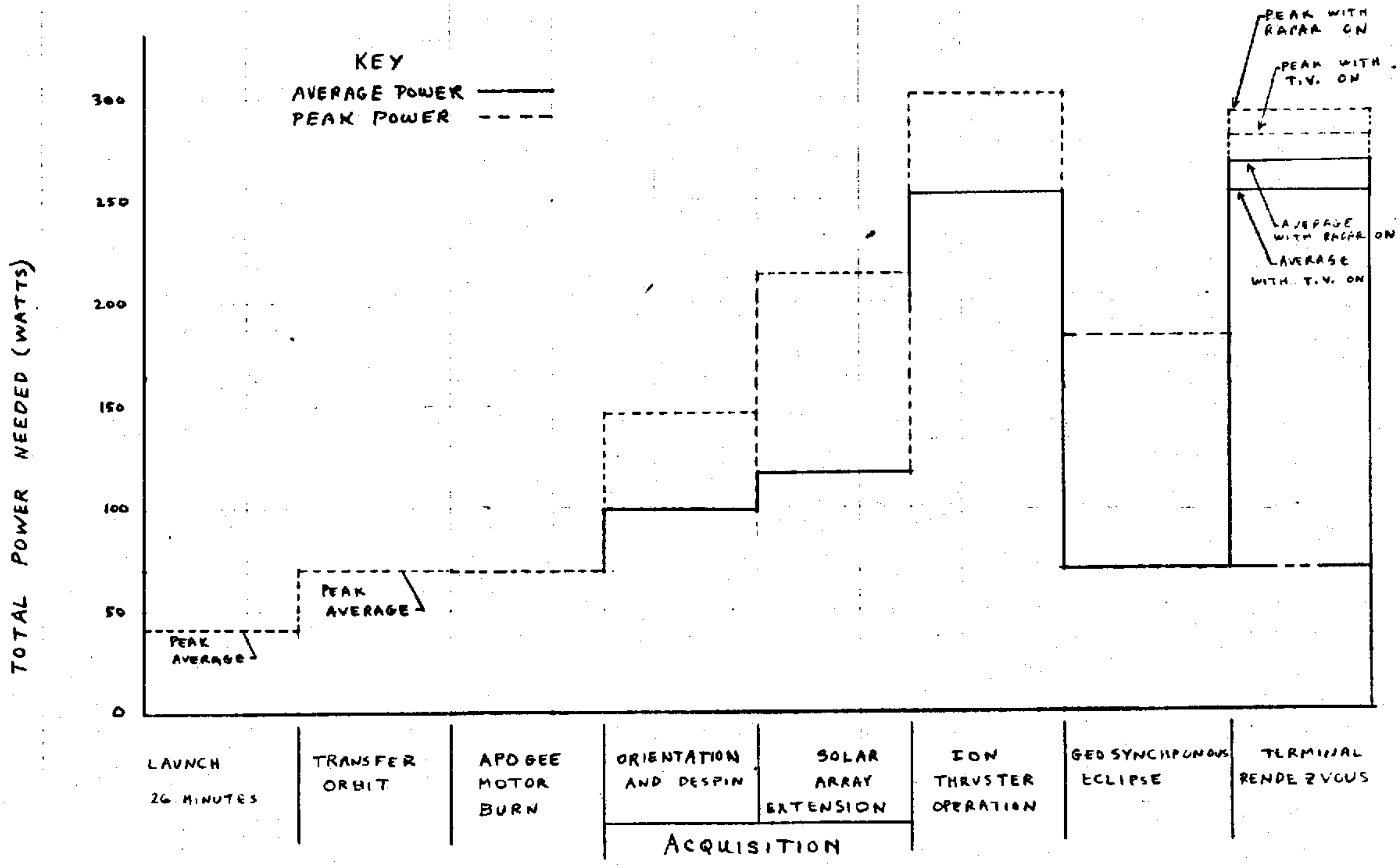


Figure 4.6.2 - SERT D Power Profile

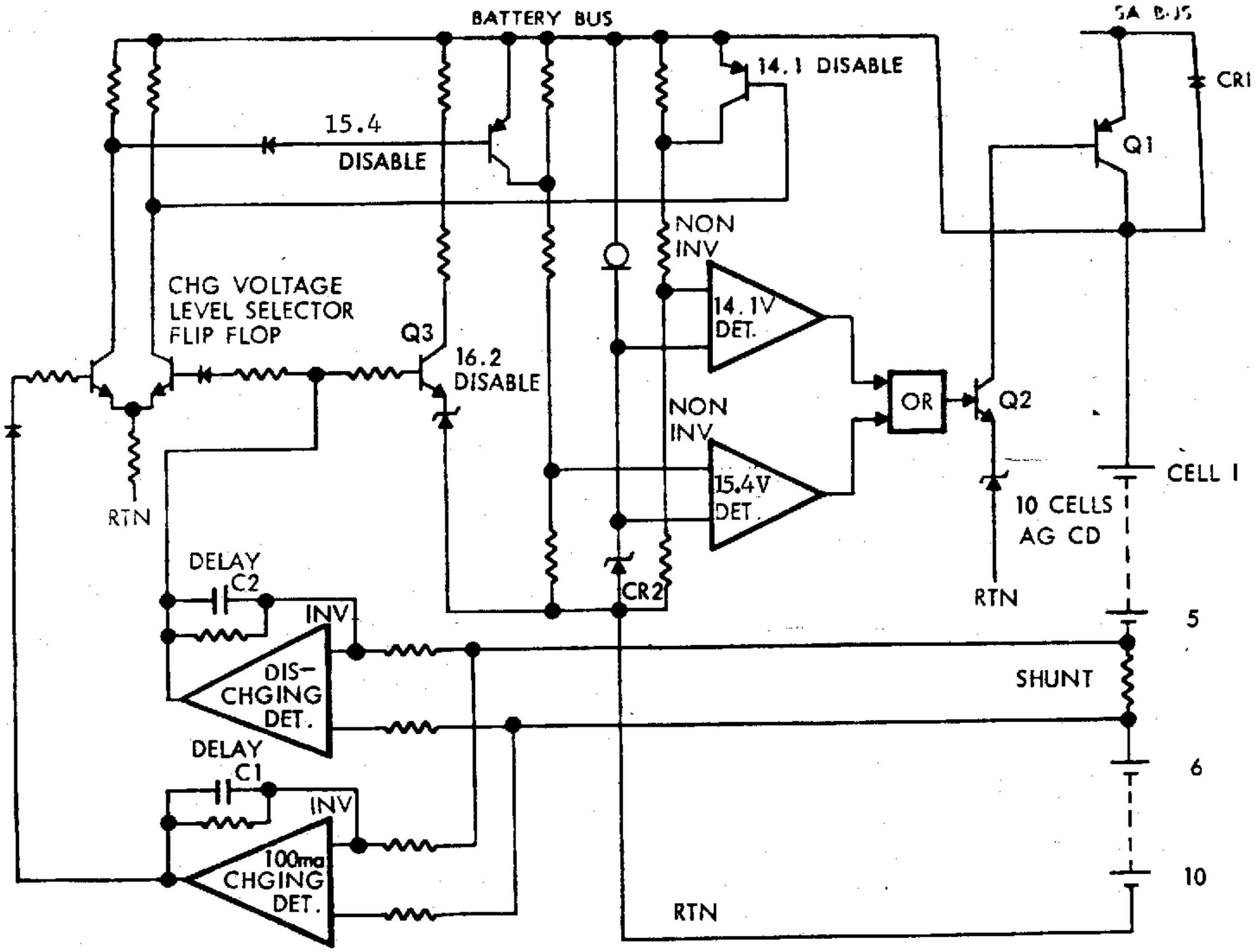


Figure 4.6.3 - Charger Block Diagram

4.7 Thermal Control

The thermal design will provide a thermal control system that will keep all components within temperature ranges that allow safe operation over a five year life. The structural size of the spacecraft must be such that enough radiative area is provided to satisfy worst case thermal operation with maximum solar heat input. The thermal control system must also have the ability to limit the radiative ability such that all components can survive the cold condition encountered during eclipse and final orbit acquisition. A unique thermal problem will be encountered with the thrusters mounted on the ends of the solar arrays. The thruster's supply lines must be maintained above the freezing point of mercury during eclipse and acquisition. The power conditioners for these thrusters must also be maintained above their electronic component's survivability temperature limit. These conditions can be satisfied by the use of louvers, high and low temperature insulation systems, heaters, heat isolators, special coatings and paints. All systems must be uniquely designed for the final spacecraft configuration.

Figures 3.1.1 and 1.3.1 show the spacecraft in its launch configuration and its deployed on-orbit configuration. The spacecraft is earth oriented with foldout solar arrays which parallel the earth's polar axis and continuously face the sun as the spacecraft rotates relative to the sun. The north and south faces of this configuration are utilized as the principal heat rejection surfaces. These surfaces are subject to seasonal solar plane angle variations of about ±23.5° and are therefore periodically exposed to low-angle solar illumination. During launch and transfer orbit these radiators and foldout solar arrays will have covers. These covers serve a dual purpose: (1) They prevent the spacecraft from radiating heat during the launch and transfer orbit phases when minimum heat is being generated internally, (2) They also provide a mounting surface for solar cells which provide extra electrical power for those phases when the solar arrays are still stowed. These covers will be ejected just prior to deploying the solar arrays. The north and south panels are dimensioned such that they can radiate the maximum internal heat generated in addition to solar heat flux. Their size is also such that during eclipse, the spacecraft components can survive without going below minimum allowable component temperatures.*

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. During the transfer orbit the nozzle of the apogee motor will be closed off by an insulated cover to minimize the nozzle radiation to space. This cover will blow out upon apogee motor firing.

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*All of the high heat dissipation electronics will be located on or near the north and south panels. Figure 3.1.2 shows the location of the major electrical components.

4.8 Telemetry Tracking and Command (TT&C)

Communications System - The tracking, telemetry and command functions of the SERT D 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 additional constraints are imposed on the subsystems:

1. Maximum use shall be made of designs and hardware from existing programs (such as the Communications Technology Satellite, CTS) 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. The spacecraft must be capable of receiving commands even if the high gain antenna is not properly pointed to the earth.

A block diagram showing a conceptual design which meets the above requirements and constraints is shown in figure 4.8.1.

Antennas - The antenna system (figure 4.8.2) consists of two separate antennas. A two-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.

During transfer orbit and possibly at times on-station, the high gain antenna may 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 body of the spacecraft. The form of this omniantenna has not been defined for this spacecraft. At this time the most likely candidate is an annulus which fits around the apogee motor nozzle. The gain of this antenna is to be better than 10 dB over 95% of a sphere which has its center at the spacecraft. This omniantenna will not be required to support the TV transmitter at any time.

Transponder - The transponder contains redundant command receivers and 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.

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, an eight bit word (at a 1 kilobit rate) has been selected. The 8 bit word can yield a maximum of 256 commands. Since the present command requirements list (Table 4.8.1) has a total of 143 required commands, there is an adequate capability (113 commands) to provide redundancy for some critical commands.

As shown in figure 4.8.1 the command decoder is required to provide all the commands for control of the equipment on the body of the spacecraft, to provide "set value" commands for each of the body mounted thrusters, to provide command interface for the Apollo Rover TV camera.

The remote decoders are required on each solar panel to provide command functions for the solar panel thrusters, for the integrated array experiment, and for the solar cell experiment.

Verification - 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 or the 16 bit value 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. No execute signal will be transmitted if the verification process is delayed, 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 dumped.

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 5V. Status monitors in the form of bi-level flags will also be provided.

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 requirements of the other spacecraft subsystems. The slow data rate has been selected to meet the requirements of all subsystems with sampling intervals of 32 sec or longer. Data compression is to be performed at the ground station rather than at the spacecraft.

The CTS telemetry encoder in its present form will not provide the functions required for SERT D because it has only approximately 40% of the required channel capacity. However, it has low power consumption and weight. An earlier version of the CTS encoder had greater channel capacity, but it had not reached the same state of development as the present CTS version before it was functionally changed. The encoder was changed because it was no longer required to support some CTS experiments that had been removed, partially as a result of the CTS weight reduction program. Since the package size has not changed from one version to the other, it is assumed that weight changes were minimal and resulted only from removal of a number of integrated circuits which weigh about 1 gram each. In order to minimize weight and power consumption, an encoder similar to the older CTS version would be purchased. The version to be purchased will have the same rates and number of words as the older CTS system. The remote subcoms and decoders will be packaged as single units to save weight. In order to consume approximately the same power as the CTS system, CMOS technology will have to be used in the encoder and remote encoder/decoder units at the expense of careful design to eliminate damage from ion thruster arcing and special handling of the devices in order to prevent damage during assembly operations. The characteristics of such a system are as shown in Table 4.8.2. Table 4.8.3 shows the channel capacity of the system. Table 4.8.4 shows the results of a survey to determine the telemetry data requirements. Even with this modified CTS system, concessions in data requirements will have to be made in order to meet the data handling system characteristics.

During the periods when the spacecraft must operate with the omniantenna, the data rate or the quantity of data will be reduced to a critical minimum in order to eliminate the necessity of using a transponder with an output rf power greater than two watts.

The telemetry encoder will also supply timing signals to the experimenters such as the solar cell experiment.

Table 4.8.5 summarizes the loads imposed on the structure, thermal, and electrical subsystems by the TT&C system. The major portions of the estimates are taken from the CTS. The TT&C system described is a minimum weight, minimum power system that essentially meets the requirements.

TABLE 4.8.1 - TOTAL COMMAND REQUIREMENTS

	DISCRETE COMMANDS	COMMAND WORDS	COMMENTS
THRUSTERS	8	1-16 bit 1-strobe	Based on CTS Thruster. Form of Strobe TBD
INTEGRATED ARRAY	7		
AC & SK	32		
TV CAMERA	16	REQ'D	Apollo Lunar Camera
SAOM	12		
T & C XPNDR	6		AE Transponder
T & C OTHER	24	REQ'D	Addressing of Decoders TBD
RADAR	6		Est. by T & C
SOLAR CELL EXP	2		
POWER	30		Est. by T & C
TOTAL	143	4	

TABLE 4.8.2 - CTS ENCODER CHARACTERISTICS (MODIFIED)

FRAME LENGTH	192
ANALOG WORDS/FRAME	152
ANALOG SUBCOM WORDS/FRAME	4
DIGITAL WORDS/FRAME	24
FRAME SYNC WORDS/FRAME	2
FRAME COUNTER, DIGITAL SUBCOM AND DIGITAL FLAG WORDS/FRAME	10
CHANNELS/SUBCOM	32
NUMBER OF ANALOG SUBCOMS	4
NUMBER OF DIGITAL SUBCOMS	5
BIT RATE	1536 BPS

TABLE 4.8.3 - MODIFIED CTS CHANNEL CAPACITY

	CHANNELS @ 1/sec	CHANNELS @ 1/32 sec	FLAGS
NORTH ARRAY SUBCOM		32	
SOUTH ARRAY SUBCOM		32	
2 S/C BODY SUBCOMS		64	
5 S/C BODY DIGITAL SUBCOMS		160	
S/C BODY ANALOG WORDS	152		
" " DIGITAL WORDS	24		
" " FLAG WORDS	4		32
TOTAL CAPABILITY	176	288	32

TABLE 4.8.4 - TOTAL TELEMETRY DATA REQUIREMENTS

	FAST RATE	SLOW RATE	FLAGS	COMMENTS
THRUSTERS	100	36		Total for 4 units. Two units operate simultaneously. PPU & NEUT. heater currents are required for the "off" pair.
INTEGRATED ARRAY		5		
AC & SK	21		19	
TV CAMERA			1	Apollo lunar camera
SAOM		12	2	Total for two units
T & C XPNDR	6	9	5	
T & C OTHER	30	5	15	
RADAR	10	2	1	Est by T & C
THERMAL		25		Includes Batt. Temp.
SOLAR CELL EXP.	1			Needs timing & control from T & C
POWER		10	20	Est. by T & C
TOTALS	168	104	63	

TABLE 4.8.5 - T & C SUMMARY

	VOLUME	WEIGHT	POWER
PARABOLIC ANT	2' Dia	10	0
OMNI ANT	46" Annulus	8	0
DUPLEXER (2 REQ)	6x3x2	22	0
HYBRID	3x1x3		0
RECEIVER (2 REQ)	6x1 3/4 x 3		3
MAIN DECODER	8x6x6		4 ¼ w standby
ENCODER	8x6x7		8
TRANSMITTER (2 REQ)	5x1 3/4 x 3		12
COAX SWITCH	2x1 3/4 x 1.4		20w for 30 sec
REMOTE SUBCOM/ DECODER	3x3x1 3/4	2 (both)	2 (both)
		42pounds	29 watts

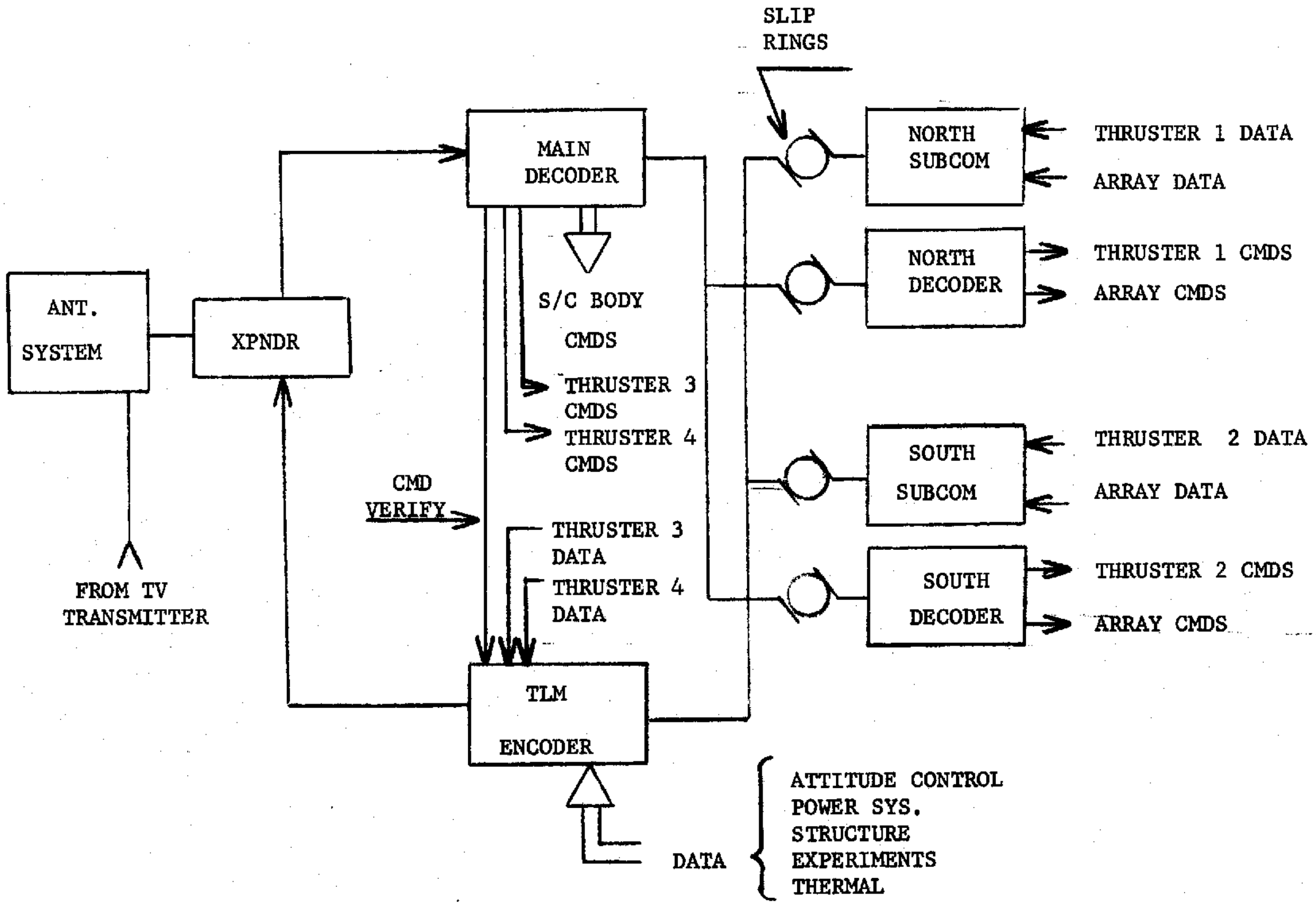


Figure 4.8.1 - SERT D T, T & C Concept

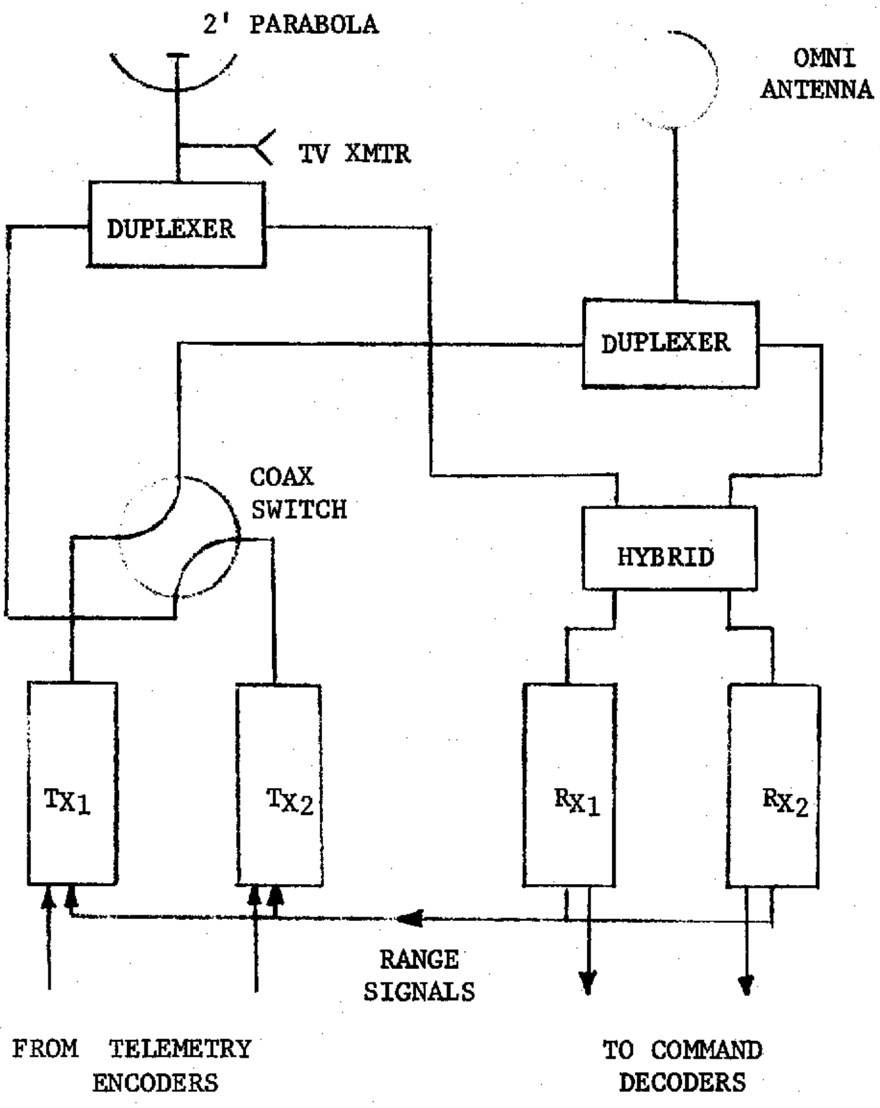


Figure 4.8.2 - RF Block Diagram

4.9 HVSA Experiment

Introduction - The high voltage solar array experiment for SERT D consists of three regulated supplies to provide power directly from the solar array to one of the array mounted thrusters. These supplies power the beam, discharge, and accelerator. The remainder of the thruster loads are supplied through conventional power conditioning circuitry. The portion of the solar array utilized for this experiment will be reconfigured and returned to the spacecraft power buses when the thruster is not being operated.

Figure 4.9.1 shows a block diagram of the entire thruster power system. The beam supply is powered by reconfiguring a portion of the experiment bus array. The discharge supply is powered by reconfiguring a portion of the housekeeping bus 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 HVSA experiment does not impose a penalty on the spacecraft array, since the portion of the solar array used for the experiment is not dedicated to its loads. The array construction is identical to that of the conventional portions of the array. Interconnection of the solar cell strings is modified and some by-pass diodes are mounted on the array. However, the number of loads required from the array is not significantly increased.

HVSA Description - Each HVSA power supply system consists of a solar cell array of sufficient size to meet the requirements of its load under worst conditions at the end-of-life. At other operating conditions, the array is capable of generating more voltage and/or current than is needed which requires that excess solar cells (series and/or parallel) must be in effect, removed from the array. There are a number of methods of controlling solar array output for direct load powering. The most suitable method of control is directly dependent upon the particular load current, voltage, and regulation requirements.

The beam and discharge power supplies are regulated by means of switches which will short-out excess solar cell blocks, removing them in a nondissipative manner from the array. Switch operation will be controlled by on-board logic, based on sensed power conditions at the load. The accelerator power supply will be regulated by zener diodes in parallel with the cells. A redundant segment is provided in this supply and is normally shorted out by a switch which will be opened when needed by on-board control. All switches and control logic will be housed in an electronics package mounted on the solar array tip, adjacent to the ion thruster.

The solar array construction and operating conditions for SERT D are similar to the CTS array: 8 mil solar cells with 4 mil micro-sheet covers mounted on a kapton-fiberglass laminated substrate. Several cell resistivities were considered and 1 ohme-cm cells were selected for the beam and discharge HVSA systems. The end-of-life (5 years in synchronous orbit) current capability of these cells is adequate for both supplies and their higher voltage per cell permits a substantial decrease in array size over 10 ohm-cm cells. Since the 1 ohm-cm cell output power is higher than the 10 ohm-cm after 5 years in synchronous orbit, this cell is also the better choice for the low voltage bus portion of the arrays. The table below lists the array temperature and illumination conditions and 1 ohm-cm solar cell characteristics at various orbit points for beginning and end-of-life.

ORBITAL POINT	T_p °C	Ilum %	BOL I_{sc} mA	BOL V_{oc} V	BOL P_{max} mW	EOL $(9x10^{14} e/cm^2)$ I_{sc} mA	EOL V_{oc} V	EOL P_{max} mW
Equinox	55	100	130.5	.530	52.3	107.0	.476	38.6
Winter solstice	50	98.3	128.0	.540	52.4	105.0	.486	38.7
Summer solstice	45	101.8	132.5	.550	55.4	103.5	.495	40.8

The solar array for the accelerator grid power supply supply system will be fabricated with edge-illuminated solar cells. The cells produce approximately 30 V and 1 mA in a 2x2 cm area.

Beam Supply - The beam supply for an 8-cm ion thruster requires 1220 V at 72 mA with ± 1% regulation. This supply requires a single cell series string, 2916 cells long. To protect the system integrity in the event of solar cells failing open, by-pass diodes will be placed across every 54 or 108 cells for an array total of 36. The table below lists the total array voltage at 72 mA at beginning and end-of-life. The end-of-life values assume that normally expected failure rates have prevailed.

	BOL	EOL	
Equinox	1416 V	1215 V	
Winter solstice	1453 V	1247 V	@ 72 mA
Summer solstice	1487 V	1277 V	

These diodes will be mounted on the array and will by-pass alternate sections of 6 and 12 series strings; each string is nine solar cells long. Figure 4.9.2 shows the array panel layout of the HVSA experiment portion of the solar array wing.

The array management electronics consists of a regulation section and a reconfiguration section, plus load switches and an array clamp. A total of 39 leads from the solar array to the electronics package are required for the management of the reconfigurable beam supply. This is an increase of three leads over the 36 leads required for the standard array configuration. Figure 4.9.3 is a schematic of the beam supply.

Regulation of the beam supply is accomplished by means of four (4) shorting switches across blocks of solar cells which are 27, 54, 108, and 216 cells long, respectively. 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. Reconfiguration of the beam array to the experiment bus voltage is accomplished by paralleling segments of the array. Each segment has a series length of 162 cells which is equal to the length of the experiment bus array. The eighteen segments of the beam array are paralleled by means of seventeen double pole, double throw relays. A blocking diode is provided for each segment for cell failure protection. The table below shows the power available from the reconfigured beam array at beginning and end-of-life. End-of-life data includes the effect of normal expected rates of cell failure.

	BOL			EOL		
	(w) Pmax	(A) Impp	(V) Vmpp	(w) Pmax	(A) Impp	(V) Vmpp
Equinox	152	2.16	70.5	99.9	1.58	63.3
Winter solstice	152	2.10	72.25	100.	1.54	65.0
Summer solstice	162	2.18	74.2	106.	1.59	66.7

Discharge Supply - The discharge supply for an 8-cm ion thruster requires 40 V at 530 mA with ± 3% regulation. This supply requires a 576 solar cell matrix with 96 cells in series, 6 cells in parallel. Protective diodes are not required for cell-open failures, provided that the array is matrixed. A cell open would limit the current capability of that parallel row to five solar cells. Under worst conditions at end-of-life, the row containing the open would operate at near short circuit current; that is, the row passes the full load current, but does not contribute to the array voltage. The rest of the array is capable of meeting the load requirements. The table below lists the total array voltage at 530 mA at beginning and end-of-life. The end-of-life values assume that the normal expected failures have occurred.

	BOL	EOL	
Equinox	47.0	41.0	
Winter solstice	48.0	40.0	@ 530 mA
Summer solstice	49.0	42.5	

The solar cell layout for the discharge supply as shown in figure 4.9.2 requires a series length of 8 cells. The total area required fills one and one-third panels. The remainder of the partial panel is available for a housekeeping array section and/or experiments. The discharge supply does not have additional protective diodes mounted on the array as in the case of the beam supply. The number of leads from this array to the electronics package is six. This is the same number of leads as is required for a housekeeping array of the same area. Figure 4.9.4 is a schematic of the reconfigurable discharge supply. Regulation of the discharge supply will be accomplished by means of three (3) shorting switches across array segments which contain 4x6, 8x6, and 16x6 cells respectively. The switches are controlled in the same manner as those in the beam power supply system. Tighter regulation of this supply can be achieved, if necessary, by the addition of regulation switches and one lead per switch from the array.

Reconfiguration of the discharge array to provide power to the housekeeping array is accomplished by shorting out the 16x6 cell block (switch R3) and connecting the array to the bus with the load switches. The discharge array will provide 22 watts BOL and 18 watts EOL to the housekeeping bus at the regulation voltage (29.3 V).

Accelerator Grid Supply - The accelerator grid supply for an 8-cm ion thruster requires 500 V at 0.23 mA with ±1% regulation. Edge-illuminated solar cells, which produce high voltage at low current are most suitable for this supply. 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 4.9.5 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.

Low Power Thruster Supplies - The vaporizers, keepers, and heaters are powered through conventional power conditioning circuitry located in the electronics package. Figure 4.9.6 is a block diagram of the thruster power system. These low power supplies are magnetic amplifier controlled and driven from a single inverter. In order to minimize the required electronics, the inverter is supplied from the regulated housekeeping bus rather than the unregulated experiment bus. 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 HVSA accelerator supply in order to eliminate a second output on the keeper supplies. The HVSA screen supply, rather than the accelerator, 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.

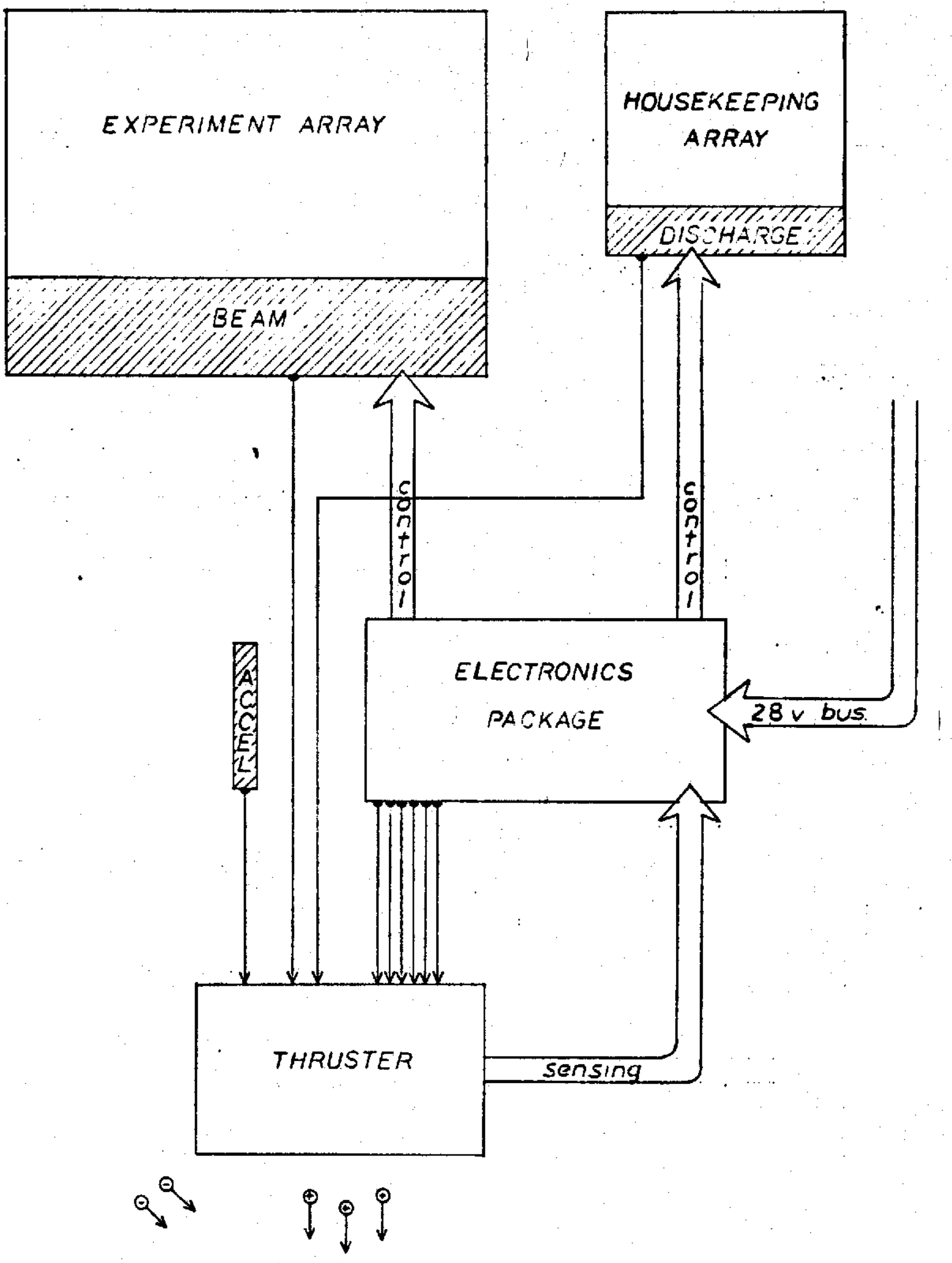


FIGURE 4.9.1 HYBRID SYSTEM BLOCK DIAGRAM

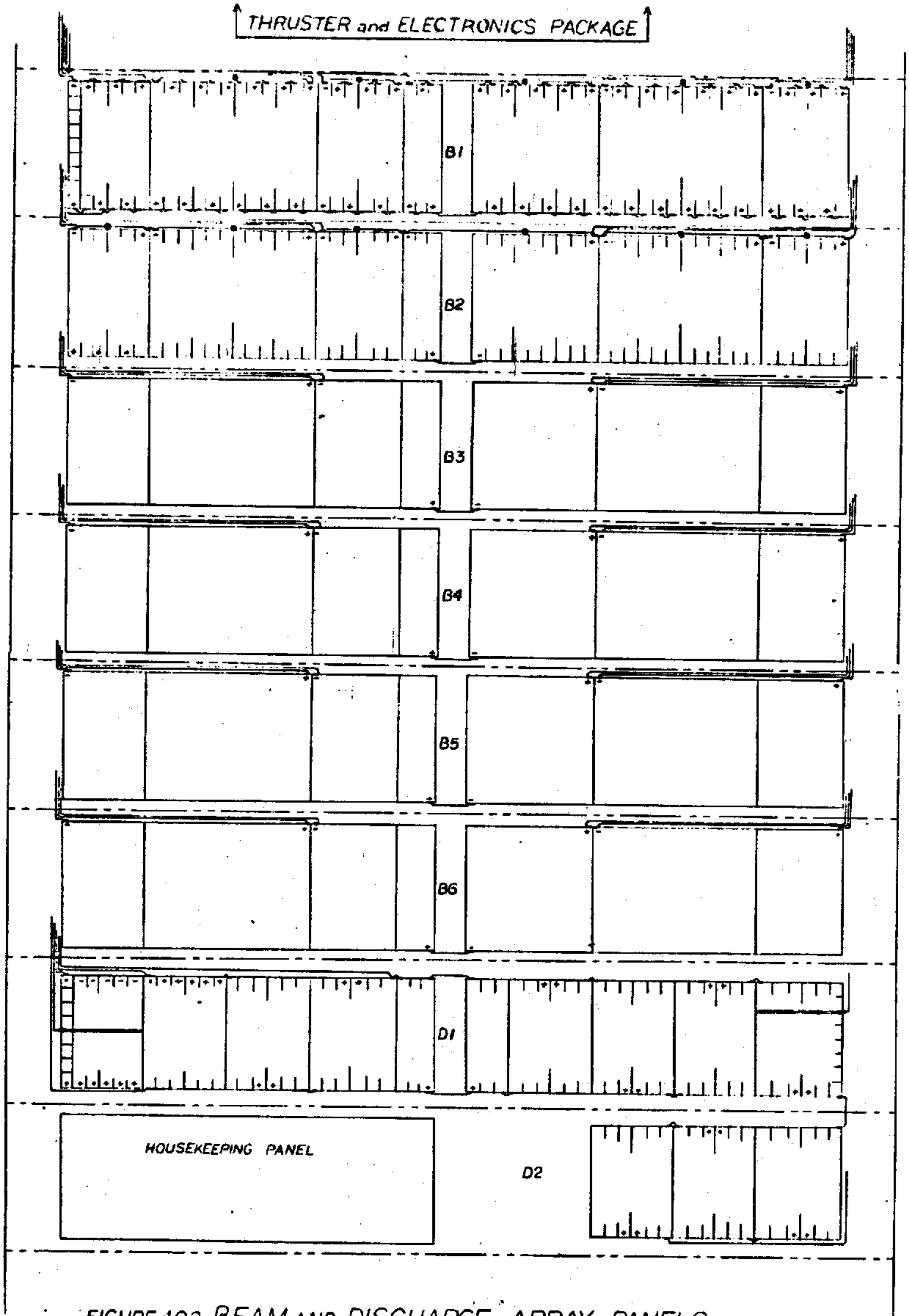


FIGURE 4.9.2 BEAM AND DISCHARGE ARRAY PANELS

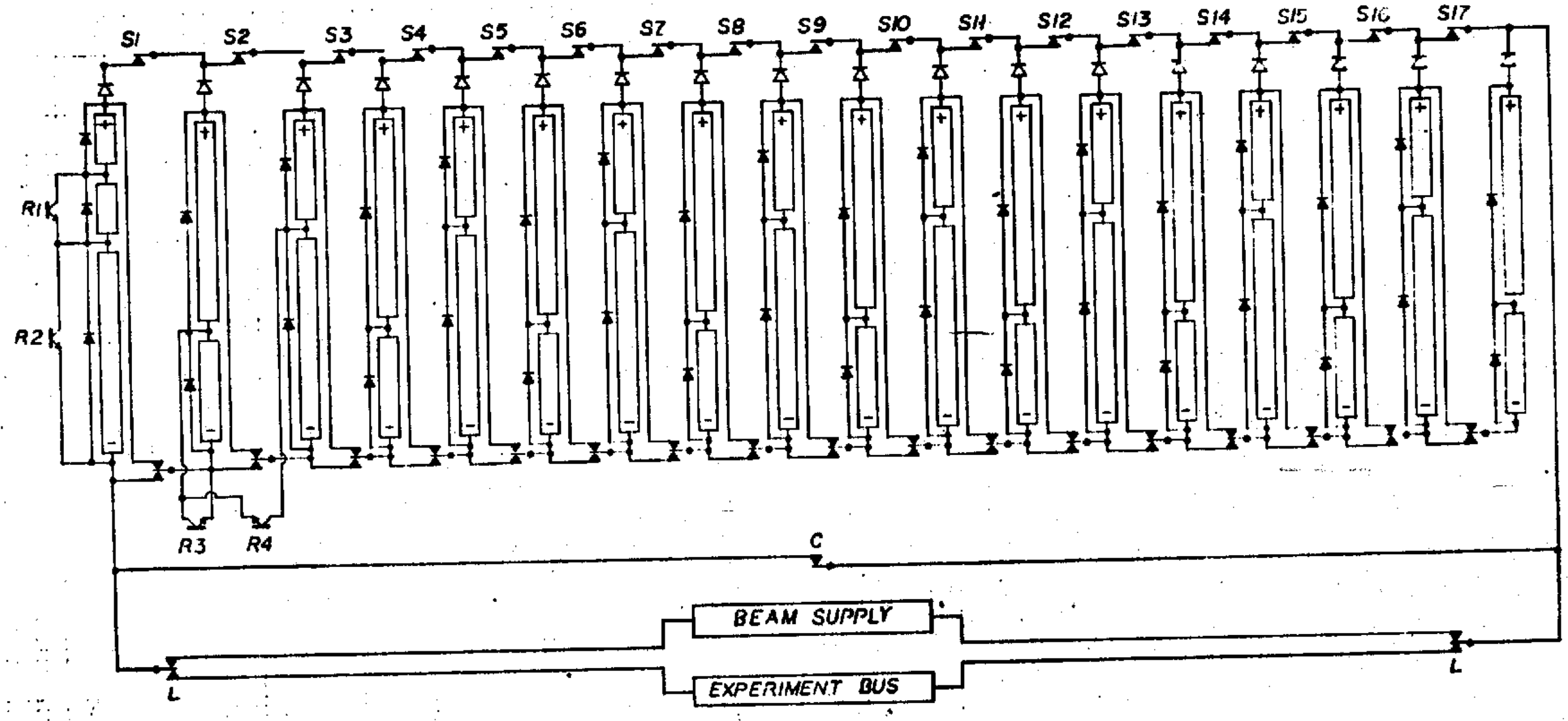


FIGURE 4.9.3 RECONFIGURABLE BEAM SUPPLY SCHEMATIC

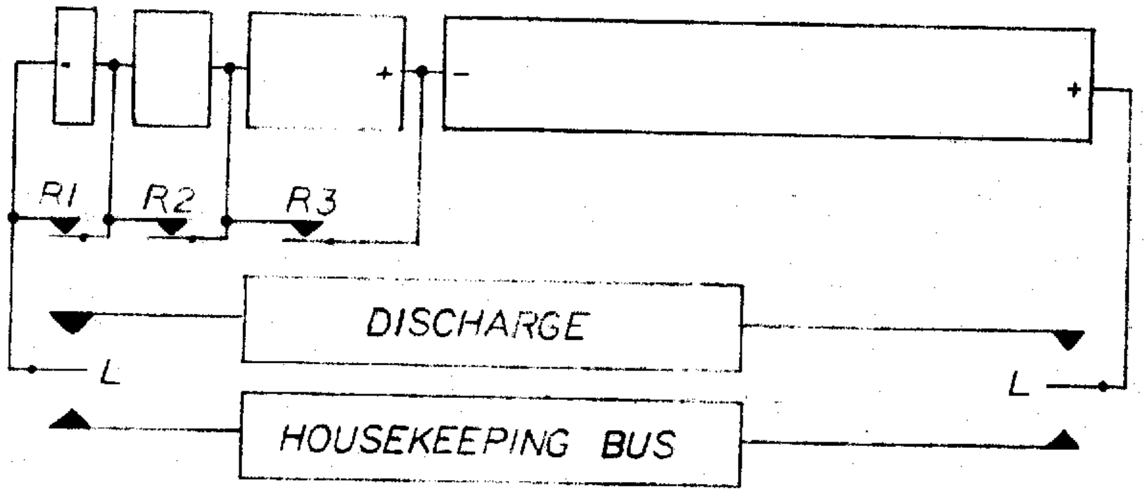


FIGURE 4.9.4 RECONFIGURABLE DISCHARGE SUPPLY SCHEMATIC

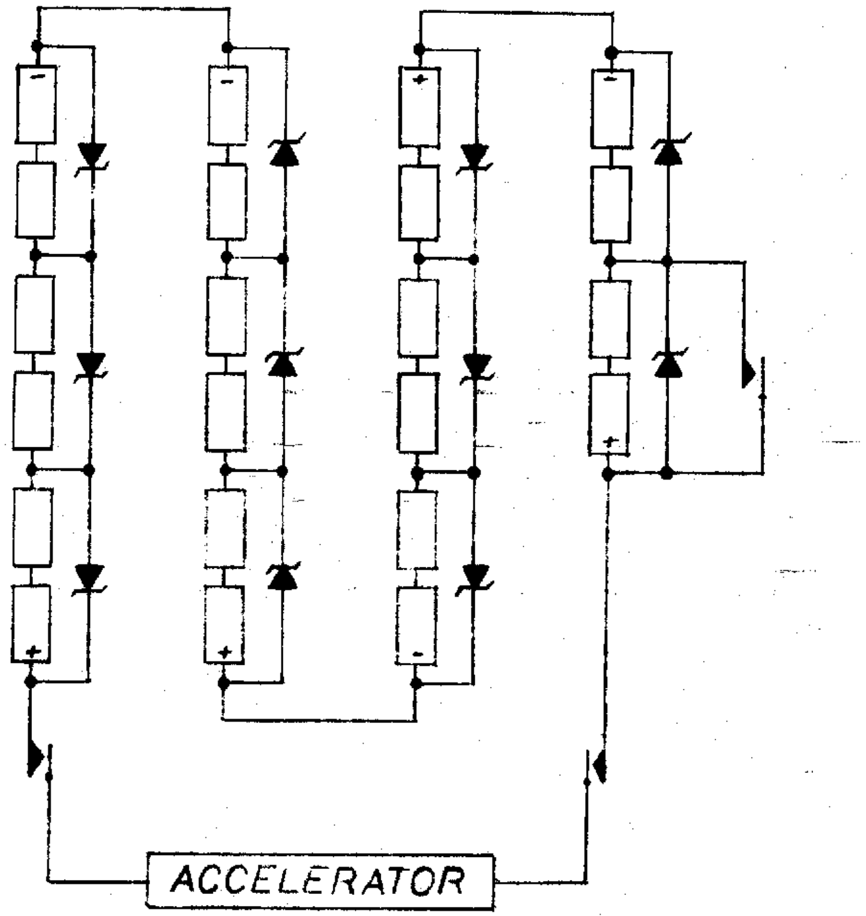


FIGURE 4.9.5 ACCEL SUPPLY SCHEMATIC

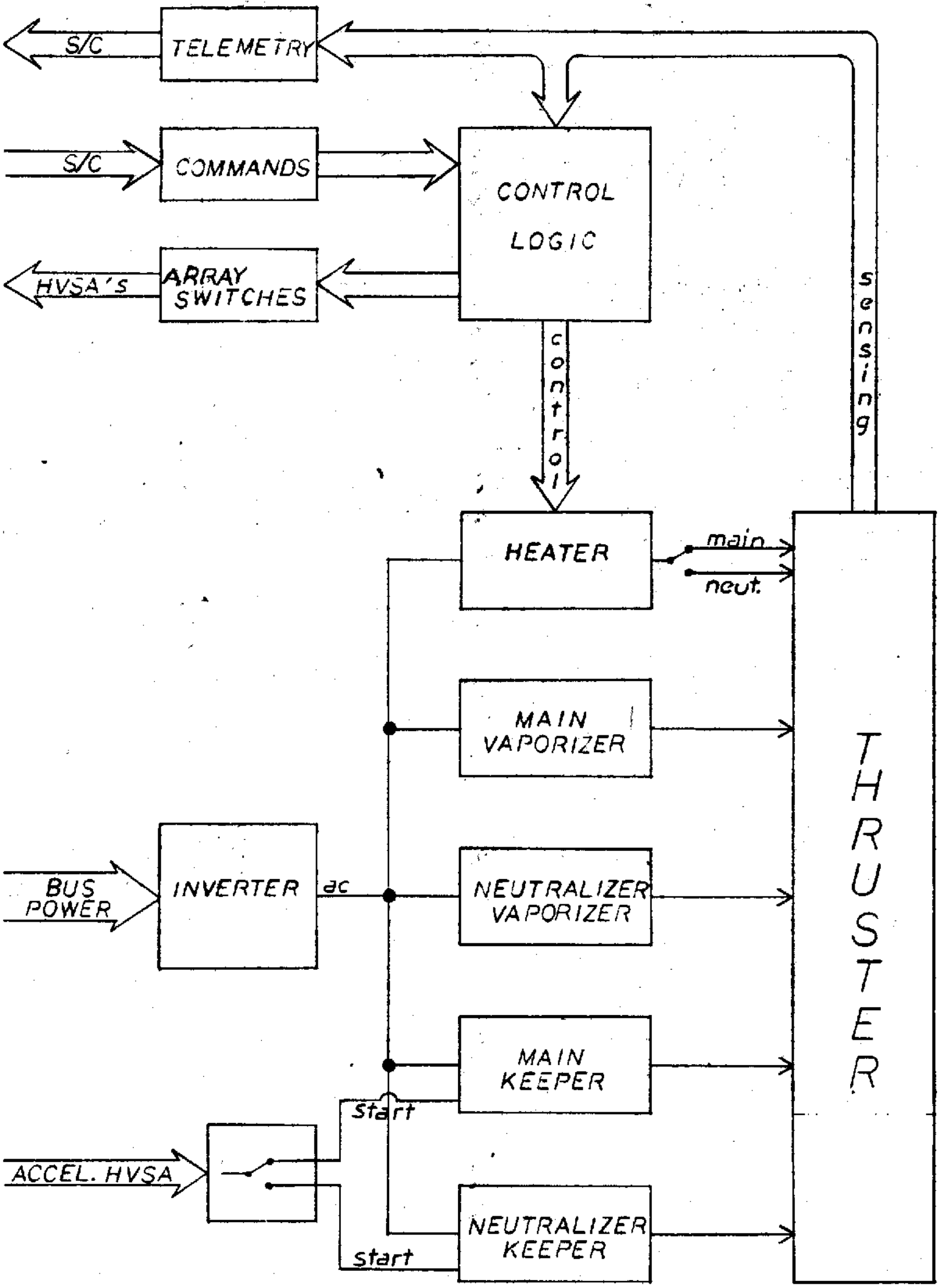


FIGURE 4.9.6 ELECTRONICS BLOCK DIAGRAM

4.10 Solar Array Orientation Mechanism (SAOM)

System Description - The solar arrays are oriented with respect to the spacecraft center body by the 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 electronics assembly will drive and control both SAOMS.

The SAOM basically consists of a stepper motor (1.8°/step), 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. Drive electronics consist of a crystal oscillator/frequency divider/down counter system stepping power to the appropriate motor field coils. The SAOMS will be driven synchronously.

In normal synchronous orbit operation, the SAOM will be operated open loop at one revolution per day. The SAOM can be operated at 1 and 20 revolutions per day, as required, by command. The control electronics have provisions for varying the speed continuously from 1 to 20 revolutions per day as required by mission operations; this will require a control input from an external source. The SAOM is 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 correct choice of materials, lubricants, and components. Redundancy is employed in the drive circuitry. The SAOM system is designed for 2 years ground storage and 5 years operational life.

Ten commands are required to operate the SAOMs and associated electronics in the desired modes. The commands consist of the following:

Electronics power on
Electronics power off
Rate 1 rpd
Rate 20 rpd
Rotation forward
Rotation reverse
Drivers primary
Drivers backup
Open loop operation
Closed loop operation

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 flag

The drive electronics will be contained in a separate box.

The LMSR assembly will be attached to the SAOM and will be part of the power transfer system for the spacecraft. The LMSR concept has been under study and development for a number of years (approximately 3 years of development were funded by LeRC) and the shaped flat blade-cup electrode configuration has been the most successful. This configuration is being employed. Fifty slip rings are being used in each of the two assemblies. There is redundancy in both the ring and brush electrodes; 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: 1.5 pounds each or 3.0 pounds total

SAOM: 5.0 pounds each or 10.0 pounds total

Drive electronics: 3.0 pounds

Total power transfer system weight: 16.0 pounds

The drive electronics package is approximately 50 cubic inches in volume (4x4x3 inches). The SAOM/LMSR assembly is 7.0 inches in diameter by 6.0 inches long (excluding connectors).

Power consumption is:

Each LMSR: none

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

6 watts average at 20 rpd

Drive electronics: 4 watts continuously

Summary: 1 rpd: 6 watts

20 rpd: 16 watts

Power Slip Rings - Most spacecraft configurations that can be envisioned for providing large amounts of dc power require the transfer of this power across a rotating interface. The transfer of electrical power across rotating interfaces in a spacecraft is typically accomplished with the well known brush-on-ring assembly in which a spring loaded, electrically conductive brush on one member rides against a precious metal ring in the other member. Such systems are well developed and are being used with remarkable success.

The concept of transferring electrical power between concentric rings bridged by the liquid metal mercury has been known for several years. Slip ring devices employing this concept are available commercially.

Advantages of Liquid Metal Slip Rings - Liquid metal slip rings provide several advantages over conventional brush-on-ring types.

1. Breakaway friction is eliminated, thus, greatly easing the attitude control problem for the spacecraft.
2. Extremely low interface (contact) resistance is obtained so power loss in the slip rings is negligible, even with currents over 100 amperes.
3. Electrical noise due to variation in contact resistance can be a small fraction of a microvolt with currents of 100 amperes.
4. There is no wearout mode.

A sketch of a liquid metal slip ring is shown in figure 4.10.1.

Hughes Aircraft Company and the General Electric Company have completed an experimental liquid metal slip ring development project for NASA Lewis Research Center. The projects include material selection experiments and the design and testing of assemblies in high vacuum. Gallium was used as the liquid metal because its very low vapor pressure makes evaporation negligible even at high vacuum, and because its melting point is 30°C (86°F), which permits freezing of the liquid metal for retention in the slip rings during transportation and launch. The gallium is readily retained in the slip rings by capillary forces against expected low in-orbit accelerations. The projects have demonstrated that the low interface resistance and electrical noise can be obtained with wetted electrodes, and have indicated that electrodes of nickel and stainless steel are compatible with gallium. Retention of the gallium by capillary forces at one gravity in a vacuum is readily accomplished by design.

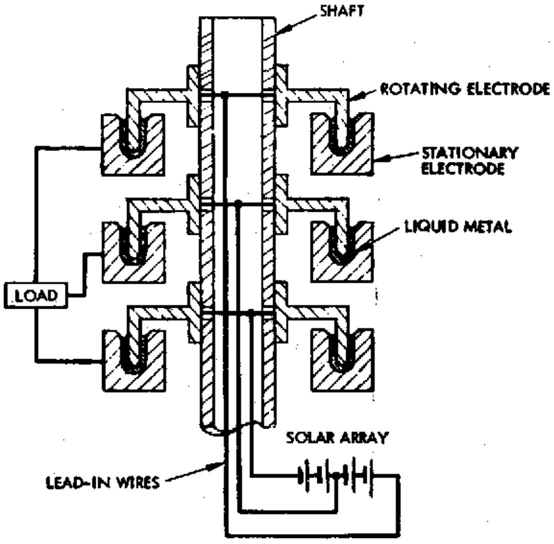


Figure 4.10.1 - Liquid Metal Slip Ring Assembly Schematic

5.0 PROGRAM PLANNING

5.1 Introduction

The Lewis Research Center 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.

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' diameter 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 D 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 equipment includes a 28,000 pound electrodynamic system for sine and random vibration in three axis (5 to 2000 Hz) and a 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.

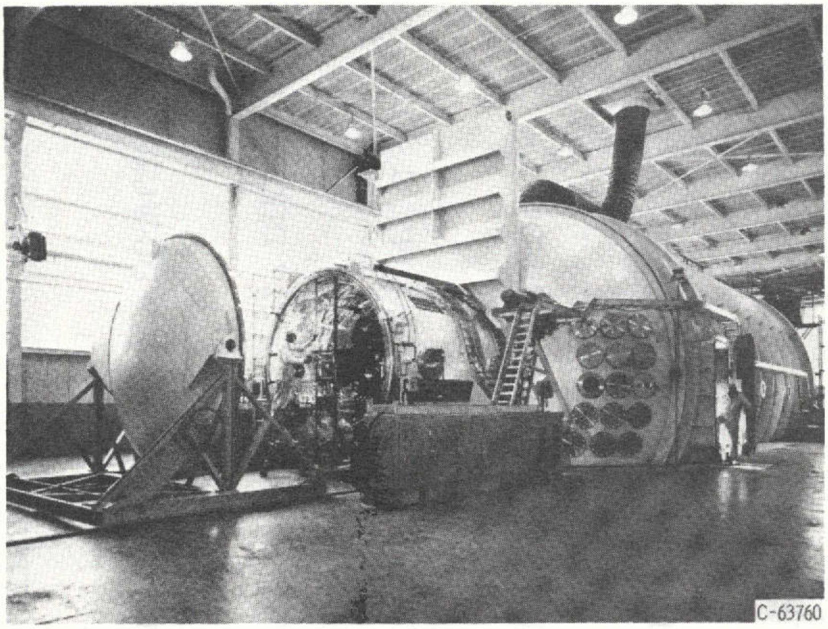
1. Vibrators - 1500, 3000, and 6000 force pounds
2. Shock machine - 150# at 400 g; 50# at 10,000 g
3. Accelerator (centrifuge) 200# at 100 g; - 65° C to 85° C at 1 torr vacuum
4. Solar simulators - one and two xenon arc lamps
5. Corona tester ± 40 kV
6. Temperature and humidity chamber 3' x 3' square -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" diameter x 28" high -185° C to 175° C; 1 x 10^{-8} torr
    b. 28" diameter x 48" long - 1 x 10^{-10} torr -185° C to 175° C
    c. 40" diameter x 54" long - 760 torr to 9.6 x 10^{-4} in 100 sec; vibration (4000#); -175° C to 175° C (launch and ascent simulation simultaneously)
    d. 5' diameter x 6' long -300° F to 250° F; 1 x 10^{-8} torr

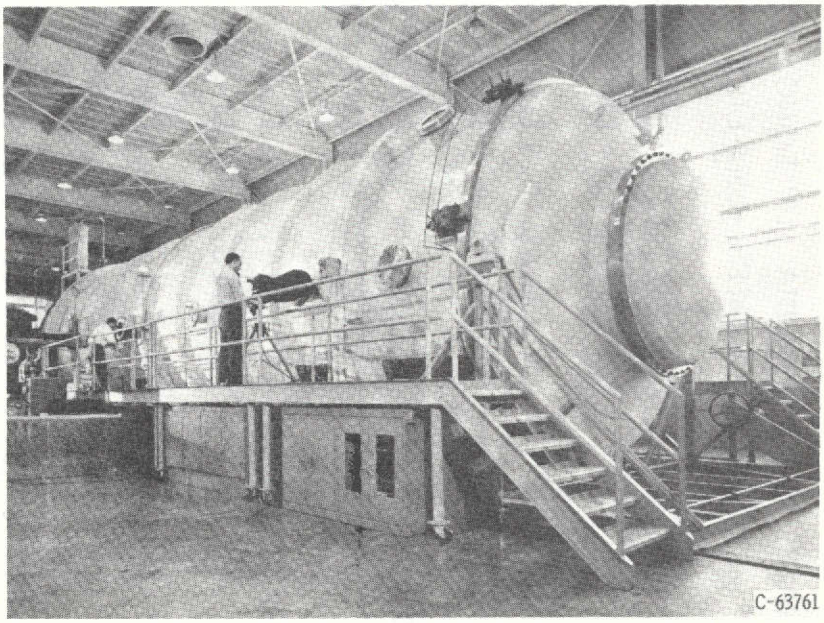
Table 5.2.1 lists representative LeRC vacuum test facilities. The two largest vacuum chambers, shown in figure 5.2.1, unique in their size and capacity, are described in detail in Appendix D.

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 D spacecraft test program will be carried out on three (3) models: (1) A Dynamic/Thermal Model, (2) An Engineering Model, and (3) A Flight Model.

(1) The Dynamic 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 important subsystem design criteria.

After mechanical testing, the Dynamic 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 system 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. A similar solar array will have been flight qualified on the Communication Technology Satellite (launch 1975). Testing on a subsystem level, however, will be performed during the SERT D program.

(3) The Protoflight Model will be a complete flight spacecraft including the flight solar array system. 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 and dynamic balancing and alinement measurements of critical surfaces and components will also be made.

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

The project schedule for SERT D is shown in figure 5.3.1.

To support a launch in late 1977 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.

5 YEAR

LEWIS RESEARCH CENTER

APPROVAL RESPONSIBILITY ____________

ACCOMPLISHMENT RESPONSIBILITY ____________

MASTER PROGRAM/PROJECT SCHEDULE
LEWIS RESEARCH CENTER

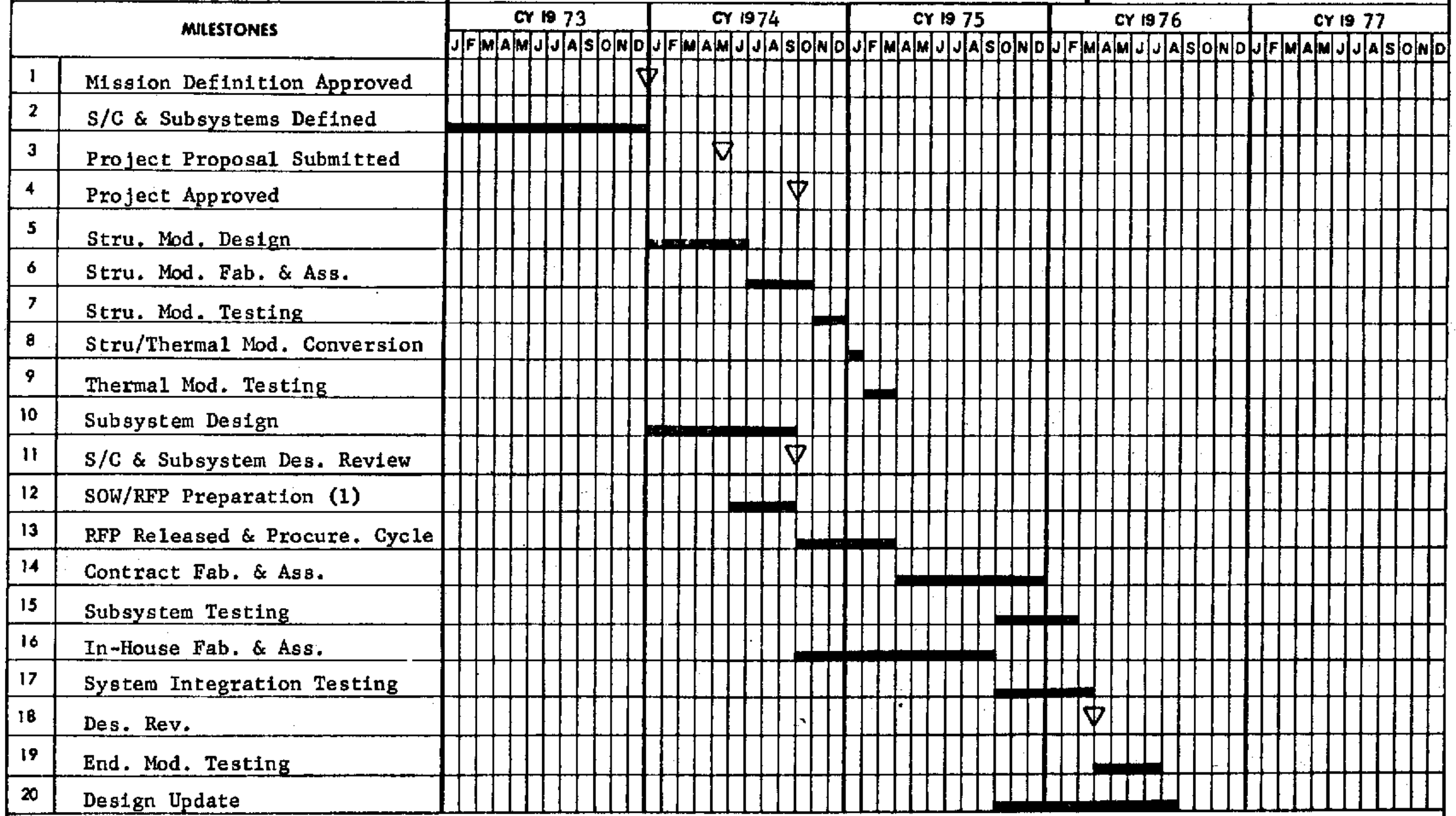
LEVEL

Figure 5.3.1 - SERT D SPACECRAFT

ORIGINAL SCHEDULE APPROVAL ______

LAST SCHEDULE CHANGE ______

STATUS AS OF ______



	MILESTONES
1	Mission Definition Approved
2	S/C & Subsystems Defined
3	Project Proposal Submitted
4	Project Approved
5	Stru. Mod. Design
6	Stru. Mod. Fab. & Ass.
7	Stru. Mod. Testing
8	Stru/Thermal Mod. Conversion
9	Thermal Mod. Testing
10	Subsystem Design
11	S/C & Subsystem Des. Review
12	SOW/RFP Preparation (1)
13	RFP Released & Procure. Cycle
14	Contract Fab. & Ass.
15	Subsystem Testing
16	In-House Fab. & Ass.
17	System Integration Testing
18	Des. Rev.
19	End. Mod. Testing
20	Design Update

NOTES

(1) SERT Procurement Office Established Beginning FY 75

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 (Cont.) - SERT D SPACECRAFT

ORIGINAL SCHEDULE APPROVAL ______

LAST SCHEDULE CHANGE ______

STATUS AS OF ______

MILESTONES

1 Protoflight Fab. & Ass.
2 Protoflight Testing
3 Launch Readiness Review
4 Launch Preparation
5 Launch (2)
6
7
8
9
10
11
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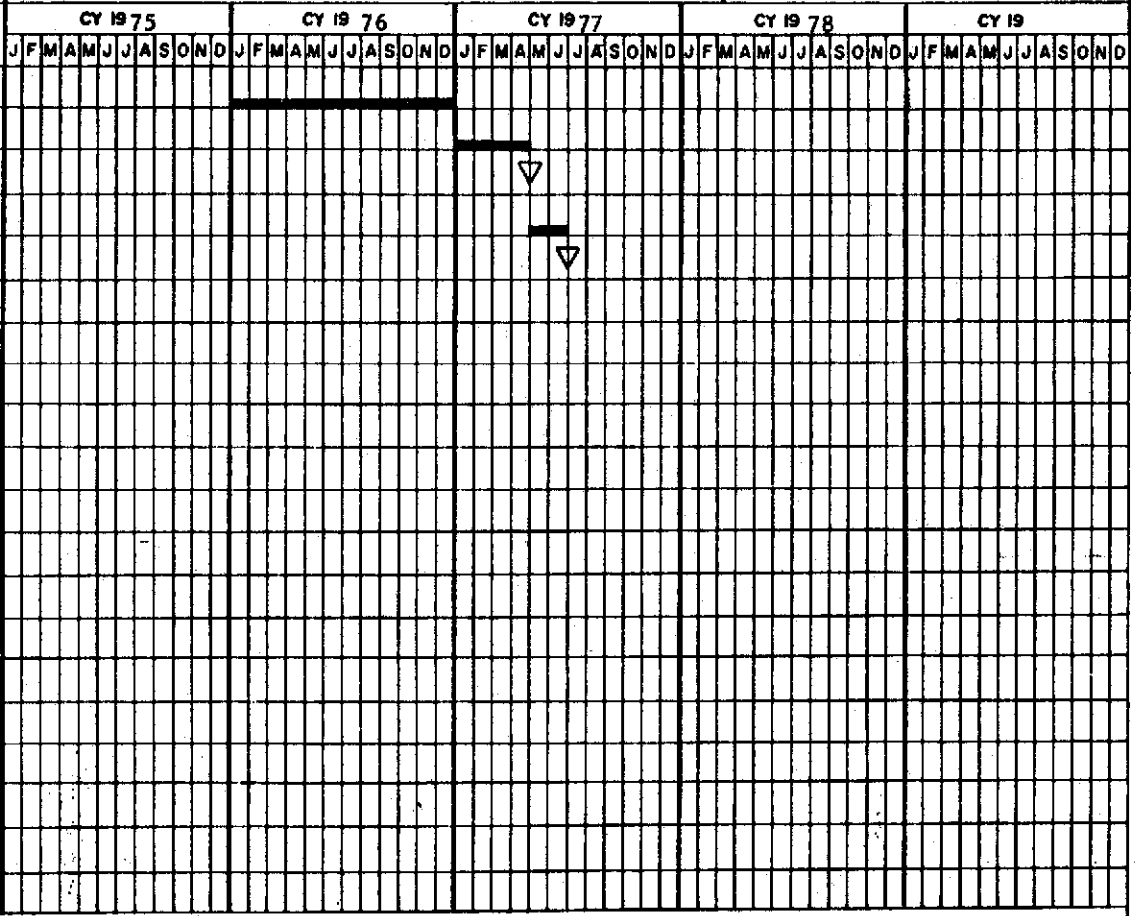


NOTES

(2) Represents earliest possible launch with no schedule contingency
More realistic launch providing schedule contingency would be late 1977 or early 1978
Schedule contingency and launch date dependent on support provided prior to project approval

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:

Manyears - Manpower requirements are presented for the three phases of the total program. These include an approximate one year period prior to project approval, a three to four year period from project approval to launch (project) and a post-launch period of five 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.

Facility - 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 and IMS base support were calculated using the prescribed FY 74 RTOP guidelines. These formulas are shown in the headings of table 5.4.2

Project Totals - A contingency of 20 percent as well as an estimate of the affect 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 costs resulting from application of the above factors are shown in table 5.4.3. These costs represent those traditionally reported as project cost from project approval to launch.

TABLE 5.4.1 - SERT D MANPOWER/COST ESTIMATES (DOLLARS IN THOUSANDS)
(MANYEARS AND R&D)

	MANYEARS			HARDWARE							
Subsystem	Prior PP	Project	Post Launch	Dev	Stru/Th	Eng	Proto flight	GSE	Contract Total	In House Direct R&D	NET R&D
THRUSTERS	2	20	6	0	0	160	480	0	640	0	640
PPU	1	10	5	0	0	300	1350	0	1650	0	1650
HYBRID P.C.	2	12	2	70	0	195	210	35	510	0	510
SOLAR ARRAY	3	25	2	140	0	546	1014	50	1750	0	1750
SAOM/LMSR	2	12	1	100	0	150	300	50	600	30	630
ATTITUDE CONTROL	12	48	2	0	0	1020	1212	60	2292	180	2472
POWER SYSTEM	1	18	1	65	0	254	505	15	839	0	839
BATTERIES (Ni-Cad)	0	5	0	0	0	20	60	0	80	0	80
TT&C	3	43	28	0	0	689	816	210	1715	0	1715
STRUCTURE	7	12	0	0	30	30	40	0	100	0	100
THERMAL	2	14	2	0	5	10	15	0	30	0	30
TV	1	8	5	200	0	200	400	120	920	50	970
RADAR		8		350	0	325	650	100	1425	50	1475
MISSION ANALYSIS	6	12	5	455	0	0	0	0	455	0	455
BATTERY EXP (Ag-Zn)	0	5	0	0	0	20	60	0	80	0	80
S/C DESIGN	2	32	0	0	0	0	0	0	0	0	0
S/C INTEGRATION	2	60	5	0	0	0	0	50	50	0	50
S/C TEST	0	30	5	0	0	0	0	75	75	0	75
APOGEE MOTOR	0	4	0	0	0	50	100	0	150	0	150
CONFIGURATION I (LESS BAT EXP)	46	373	69	1380	35	3949	7152	765	13281	310	13591
CONFIGURATION II (LESS RADAR)	46	370	67	1030	35	3644	6562	665	11936	260	12196

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

Total Project Cost

Subsystem	IMS		IMS Base Support (2.4xMY)		R&D Resources		Personnel (31.7xMY)		IMS Base Sup (2.9xMY)		R&PM Resour		Fa-cil-ity	Total Project Dollars	
	Unique	10K/ MY	Proj	Post Launch	Proj	Post Launch	Proj	Post Launch	Proj	Post Launch	Proj	Post Launch		Proj	Post Launch
THRUSTERS	120	200	48	14	1008	14	634	190	58	17	640	208	0	1648	222
PPU	20	100	24	12	1794	12	317	158	29	14	346	174	0	2140	186
HYBRID P.C.	0	120	29	5	659	5	380	63	35	6	415	69	0	1074	74
SOLAR ARRAY	0	250	60	15	2060	15	792	63	72	6	864	69	0	2924	84
SAOM/LMSR	0	120	29	2	779	2	380	32	35	3	415	35	0	1194	37
ATTITUDE CONTROL	0	480	115	5	3067	5	1522	63	139	6	1661	69	100	4828	74
POWER SYSTEM	0	180	43	2	1062	2	571	32	52	3	623	35	0	1685	37
BATTERIES	0	50	12	0	142	0	158	0	14	0	172	0	0	314	0
T T & C	50	430	103	67	2248	67	1363	888	125	81	1488	969	0	3736	1375
STRUCTURE	0	120	29	0	249	0	380	0	35	0	415	0	0	664	0
THERMAL	400	140	34	5	604	5	444	63	41	6	485	69	0	1089	74
TV	0	80	19	12	1069	12	254	158	23	14	277	172	0	1346	184
RADAR	0	80	19		1574		254		23		277		0	1851	
MISSION ANALYSIS	400	120	29	12	1004	12	380	158	35	14	415	172	0	1419	184
BATT. EXP (Ag-Zn)	0	50	12	0	142	0	158	0	14	0	172	0	0	314	0
S/C DESIGN	0	320	77	0	397	0	1014	0	93	0	1107	0	0	1504	0
S/C INTEGRATION	0	600	144	12	794	12	1902	158	174	14	2076	172	0	2870	184
S/C TEST	200	300	72	12	647	12	951	158	87	14	1038	172	0	1685	184
APOGEE MOTOR	0	40	10	0	200	0	127	0	12	0	139	0	0	339	0
CONFIG I (less batt)	1190	3390	896	175	19357	175	11823	2184	1082	198	12853	2385	100	32310	2560
CONFIG II (less radar)	1190	3360	889	170	17925	170	11727	2121	1073	192	12748	2313	100	30773	2483

TABLE 5.4.3 - PROJECT SUMMARY COST ESTIMATES

SPACECRAFT CONFIGURATION I	THOUSANDS
Net R&D	18,348
Total Project	43,618
SPACECRAFT CONFIGURATION II	
Net R&D	16,536
Total Project	41,545

A APPENDIX - DIRECTLY REGULATED HOUSEKEEPING ARRAY

Direct regulation of the SERT D housekeeping solar array by means of shorting switches offers several advantages over more conventional regulators. The regulation electronics, which consists of shorting transistors, control logic, and sensor, is light weight and has low power dissipation. High system reliability is easily obtainable with redundant logic and sensors and the use of quad transistors. Figure A-1 is the schematic of a four panel direct regulated housekeeping array.

Efficient mechanization of this system for SERT D requires modification of the solar panel layout from a basic panel length of 9 cells (CTS layout) to a panel length of 8 cells. Panel width remains unchanged. This change removes 54 unneeded cells from the panel and greatly simplifies array segment tapping for regulation. The proposed cell layout is shown for a single panel in figure A-2. As shown, the basic cell block width has been doubled to six cells in parallel. Again to reduce wiring complexity. In addition, the blocking diodes have been moved out onto the array. A suitable diode the size of a 1x2 solar cell is being tested for the Air Force. It is intended for use on flexible arrays under synchronous orbit conditions.

As shown in figure A-1, each panel requires eight leads from the cells to the diode board. To reduce harness weight, it is proposed that the array segments be paralleled on the array, rather than at the diode board. This can be done since each panel contains its own blocking diode. The diode board will contain the shorting transistors, the control logic and the sensor. Only the bus leads will be brought across the slip rings.

Solar array panel weight will not change significantly, since the added wiring and the blocking diode are off set by the removal of 54 solar cells. The weight of the control electronics, mounted on the array diode board, is estimated to be three pounds for a fully redundant system.

Because of weight restraints with SERT D, the light weight estimate for the direct solar array regulator makes it quite attractive. Also, its low power dissipation gives it an advantage over other types of regulators.

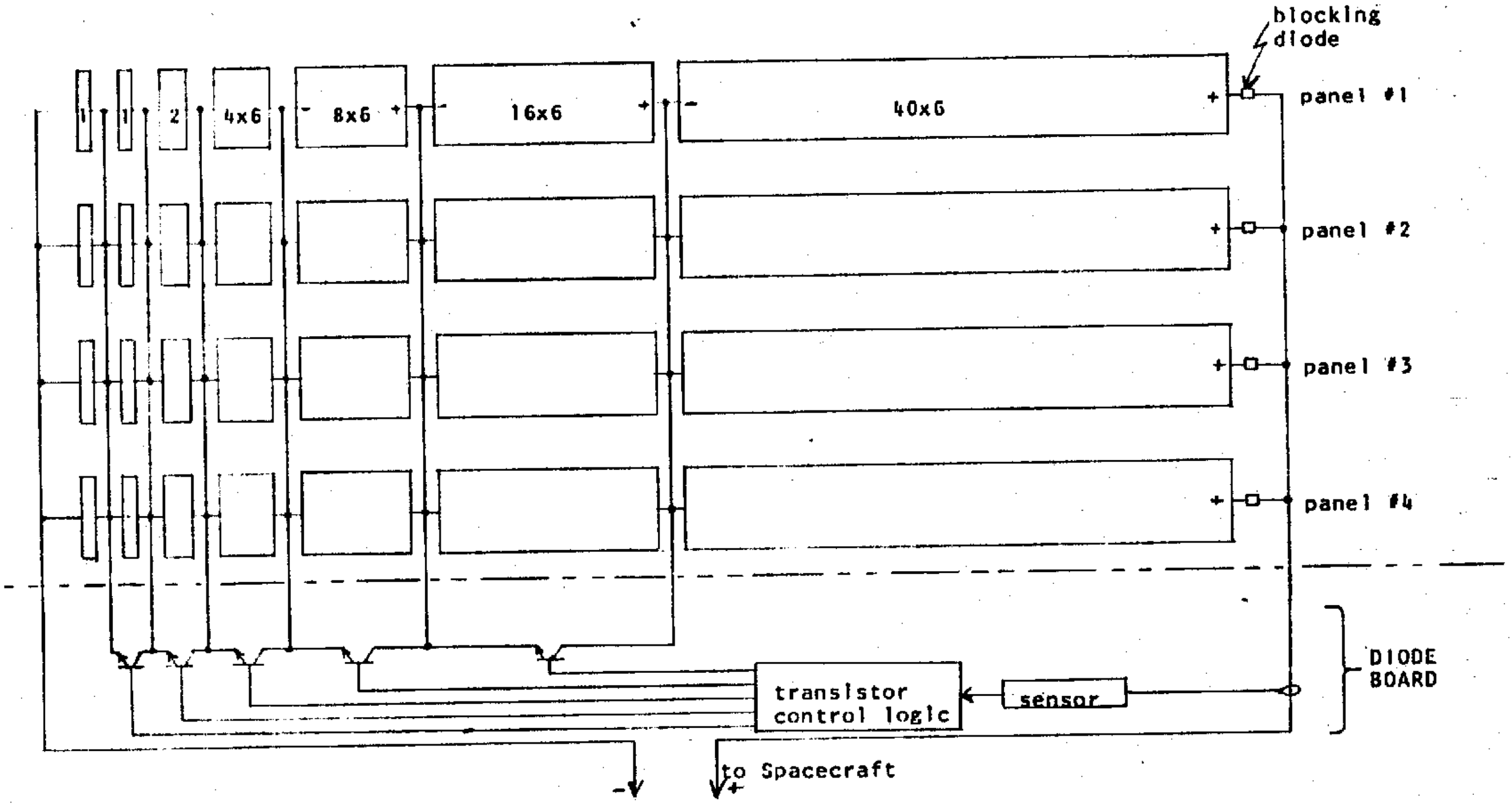


Figure A.1 - Regulated Housekeeping Bus Schematic

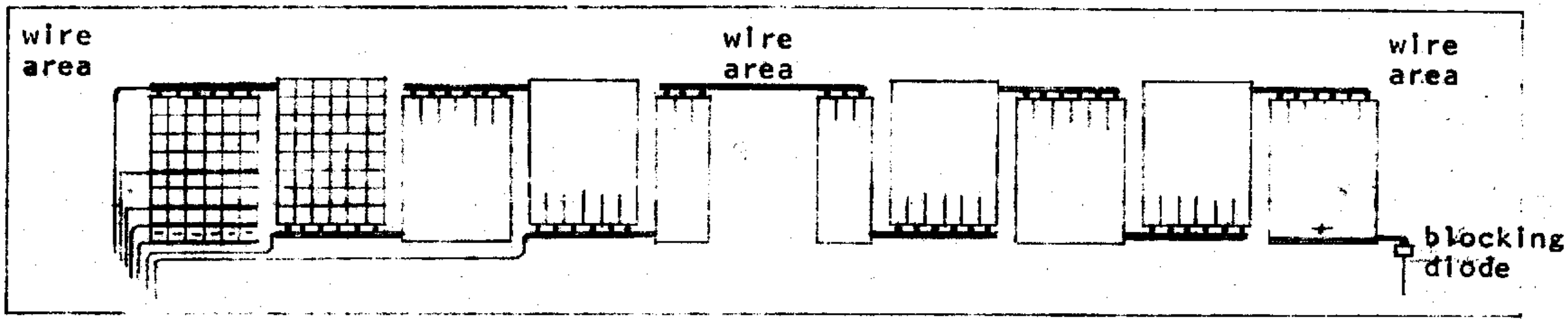


Figure A.2 - Modified Solar Array Panel for Regulated Housekeeping Bus

B APPENDIX - PROPOSED USE OF SILVER-ZINC BATTERY FOR SERT D

On the basis of cycling and net life data obtained over the past four years, the silver-zinc cell developed by NASA has the capability of performing the number of cycles required for the proposed five year SERT D mission in synchronous orbit. Cells cycling after four years of net life also attests to the operating lifetime.

The cell proposed for the SERT D spacecraft is a silver-zinc cell based on the design of the 40 AA cell developed for the Viking Lander. 70 watts for the housekeeping chores for SERT D requires a 6 amp hour cell. Design parameters for this cell are shown in table B-1. The average discharge current is 2.7 amperes or about the C/2 rate. 12 amp peak power pulses should be no problem. At the maximum eclipse period, 3.24 ampere hours are used from the cell, a 54 percent depth of discharge. To provide 26 volts, 18 cells are required operating at an average discharge voltage of 1.44 volts per cell. The cell weight is calculated to be at most .45 lbs/cell or about 9.5 lbs. per battery including case. A redundant battery is proposed which will be stored open circuit at low temperature. The battery will be composed of 2 strings of 9 cells each. For best results it should be operated between 10 and 20°C. It can operate between 0 and 40°C and withstand temperatures between -40°C and +65°C. It is anticipated that the battery will be on line at open circuit during the entire mission and will be used during sunlight periods for pulsing peak power and for housekeeping during eclipse periods. The battery will be charged after each use at constant current to a preset voltage limit at which time the current will taper to a preset limit. At this point the battery will be switched to open circuit. This charging technique is used on the Imp and OAO series of satellites and is called a 2 step voltage limiter. The rate at which the battery should be charged after each use should be at the 10 hour rate or about .325 amps based on maximum discharge depth. The low current level after taper at which the cell should be switched to open circuit voltage will have to be determined in tests to prevent overcharging. The voltage will also be protected by a cutout to prevent overdischarging and reversal of cells.

TABLE B.1 -Ag Zn CELL - 6 AMP HOUR CELL

OPEN CIRCUIT VOLTAGE	1.86
CLOSED CIRCUIT VOLTAGE	~1.44 @ 2.7 Amps
MAXIMUM CHARGE VOLTAGE	1.96 ± .01 Volts/Cell
MINIMUM VOLTAGE	1.25
AMP HOUR EFFICIENCY	~100%
WATT HOUR EFFICIENCY	~70%
CAPACITY RETENTION (1 YR)	85%
OPTIMUM OPERATION	10-20°C
USEFUL OPERATION	0 - 40°C
HEAT EVALUATION *	to be determined (charge & discharge)
WEIGHT/CELL	~.45 lbs

* Approximate numbers can be obtained from the equation

$$q = (E_{REV} - E)\ I + KIT$$

where q = watts of heat
E = voltage
I = amperes
K = 4.186 $\Delta S/F$
T = °K
ΔS = Cal/g-equivalent °K (or entropy units)
4.186 = conversion factor - calories to joules

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 D thruster.

TABLE C-1. - System Definition

One SERT D 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 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 *1 Efficiency (%)	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
<u>Cells</u>				
Total Weight (g)	6396	5134	6240	4980
Energy Density (Wh/kg)	33.4	41.7	34.3	43.0
<u>Battery</u>				
Total Weight (g)	7060	5650	6865	5480
Energy Density (Wh/kg)	30.3	37.9	31.2	39.1
<u>System</u>				
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
<u>Cells</u>				
Total Weight (g)	6240	4920	6080	4800
Energy Density (Wh/kg)	34.4	43.5	35.2	44.6
<u>Battery</u>				
Total Weight (g)	6865	5410	6690	5280
Energy Density (Wh/kg)	31.1	39.5	32.0	40.5
<u>System</u>				
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
<u>Cells</u>				
Total Weight (g)	7728	5920	7440	5664
Energy Density (Wh/kg)	27.7	36.2	28.8	37.8
<u>Battery</u>				
Total Weight (g)	8500	6510	8185	6230
Energy Density (Wh/kg)	25.2	32.8	26.2	34.4
<u>System</u>				
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

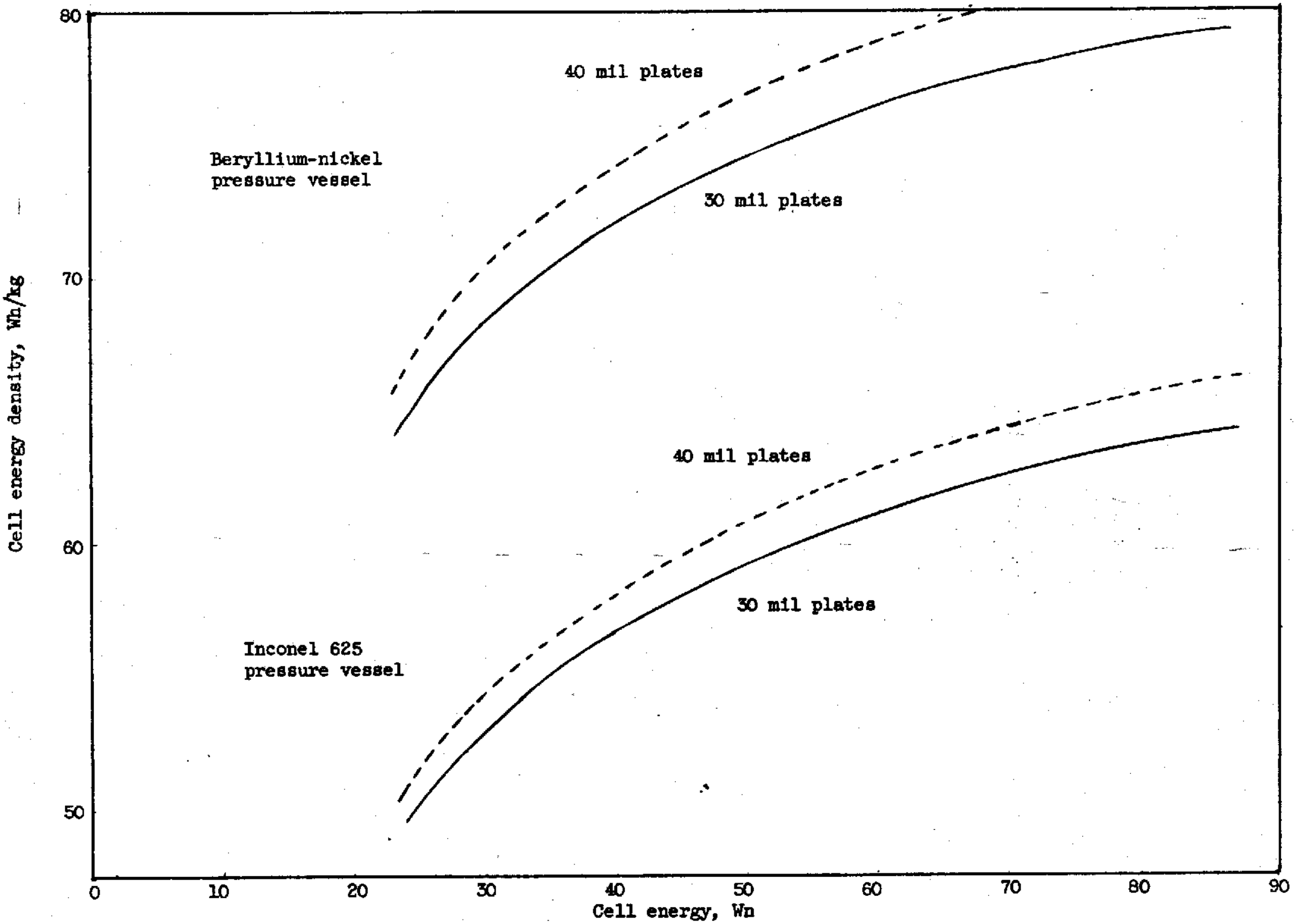
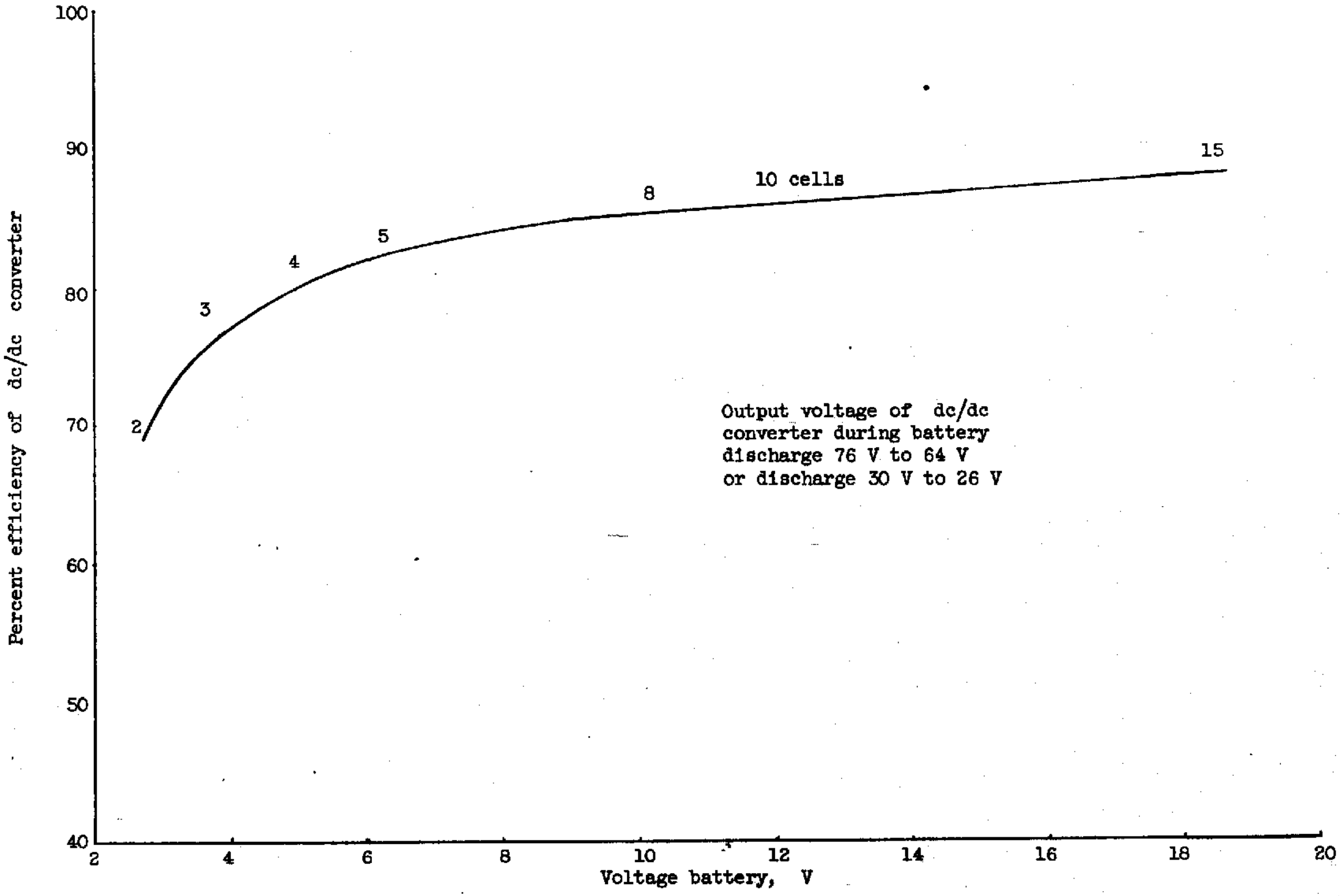


Figure C-1. - Cell energy density at 100 percent depth of discharge.



Output voltage of dc/dc converter during battery discharge 76 V to 64 V or discharge 30 V to 26 V

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

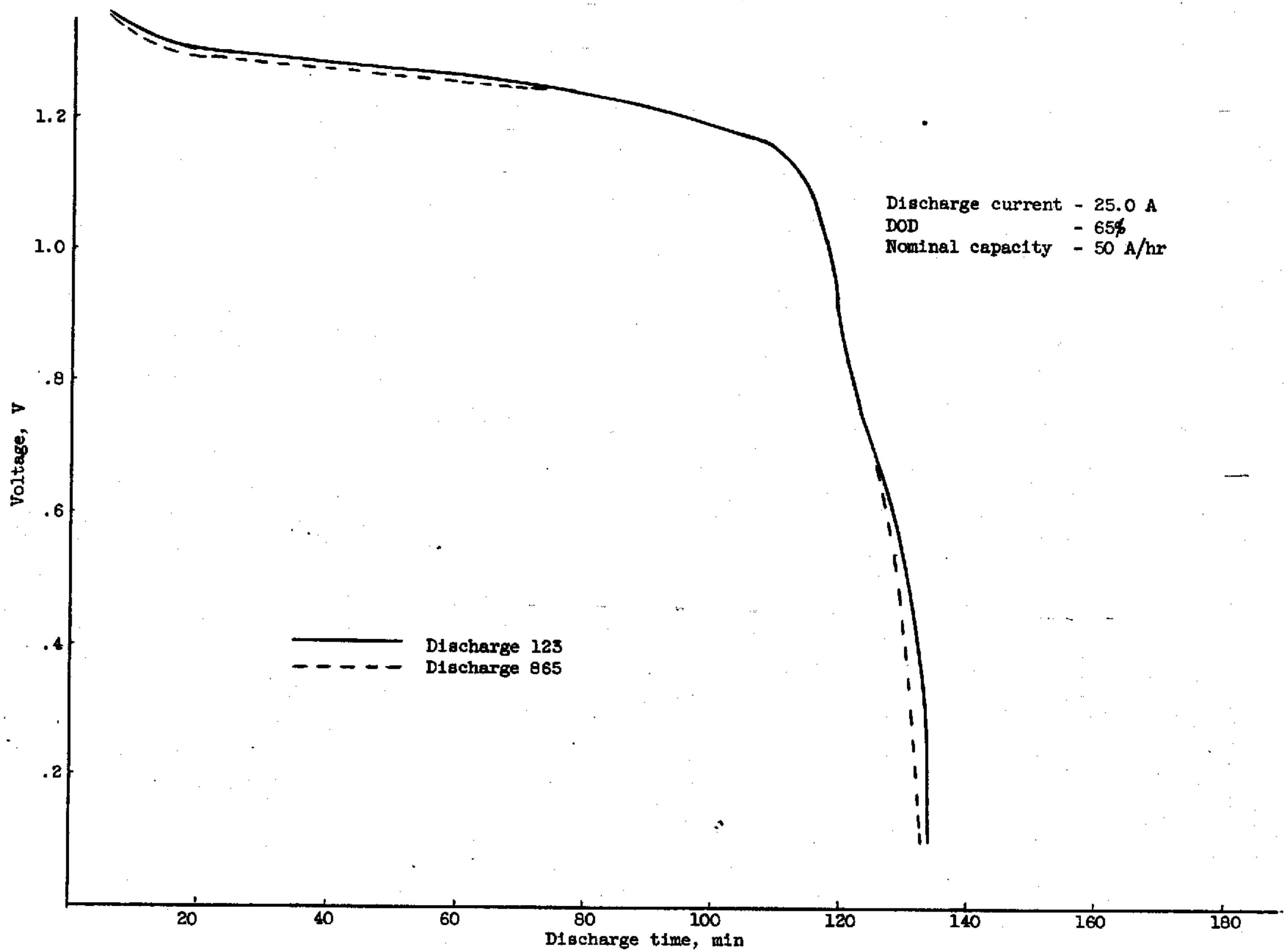


Figure C-3. - Discharge characteristic of Ni-H_2 cell.

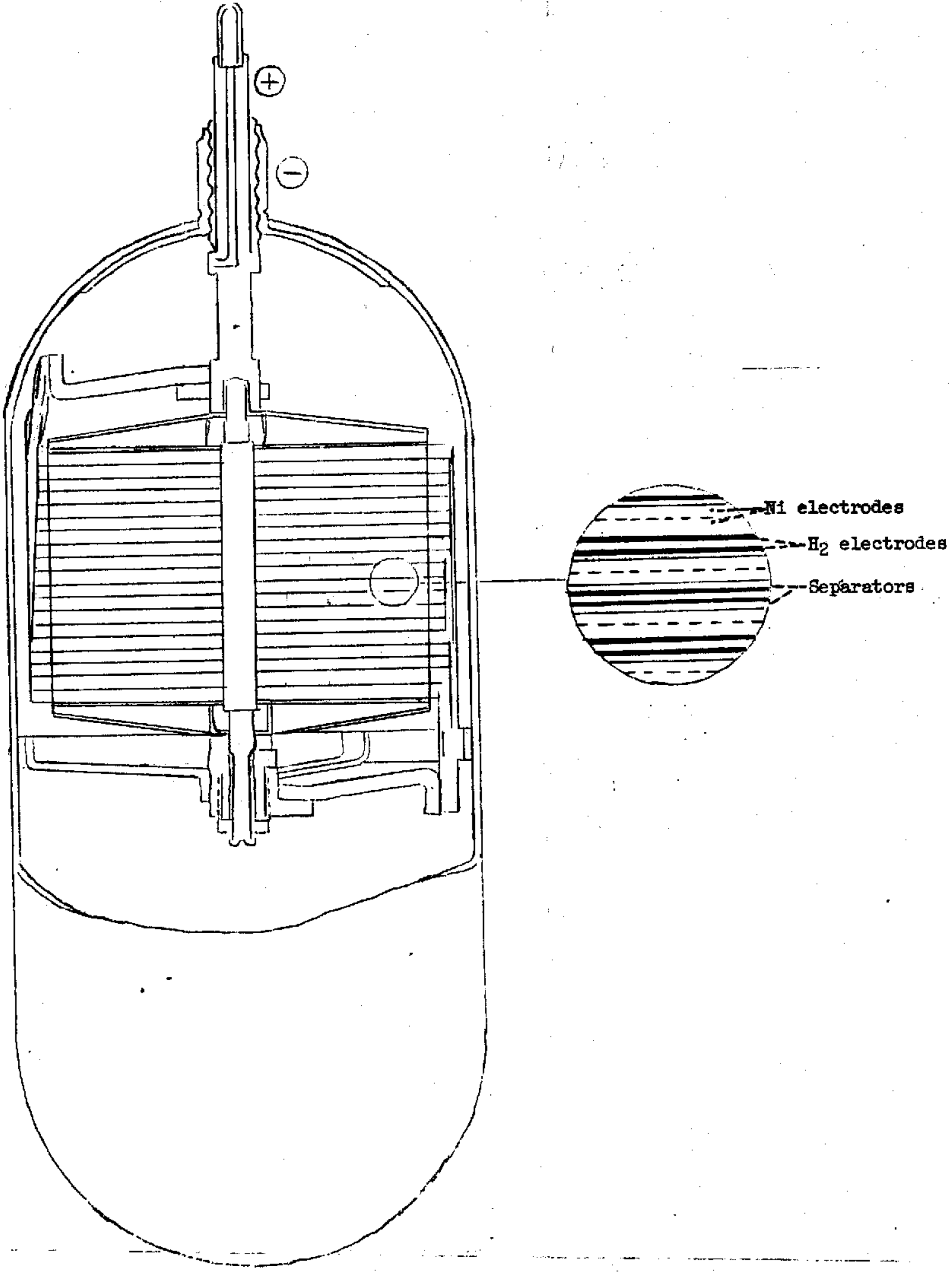


Figure C-4 - Ni/H_2 66 watt-hour cell (scale: 1:1).

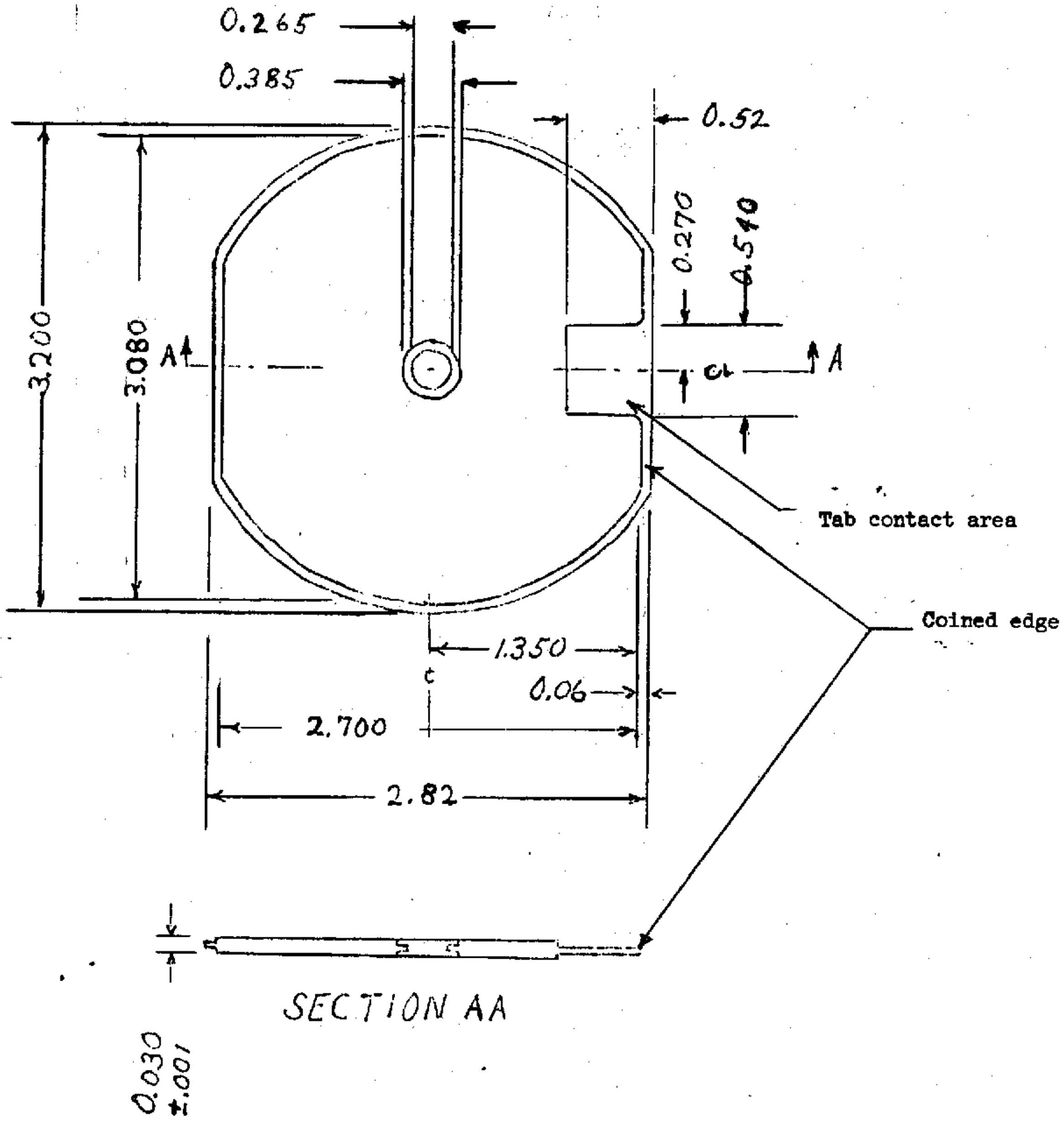


Figure C-5 - Nickel plate for Ni/H_2 cells (scale: 1:1).

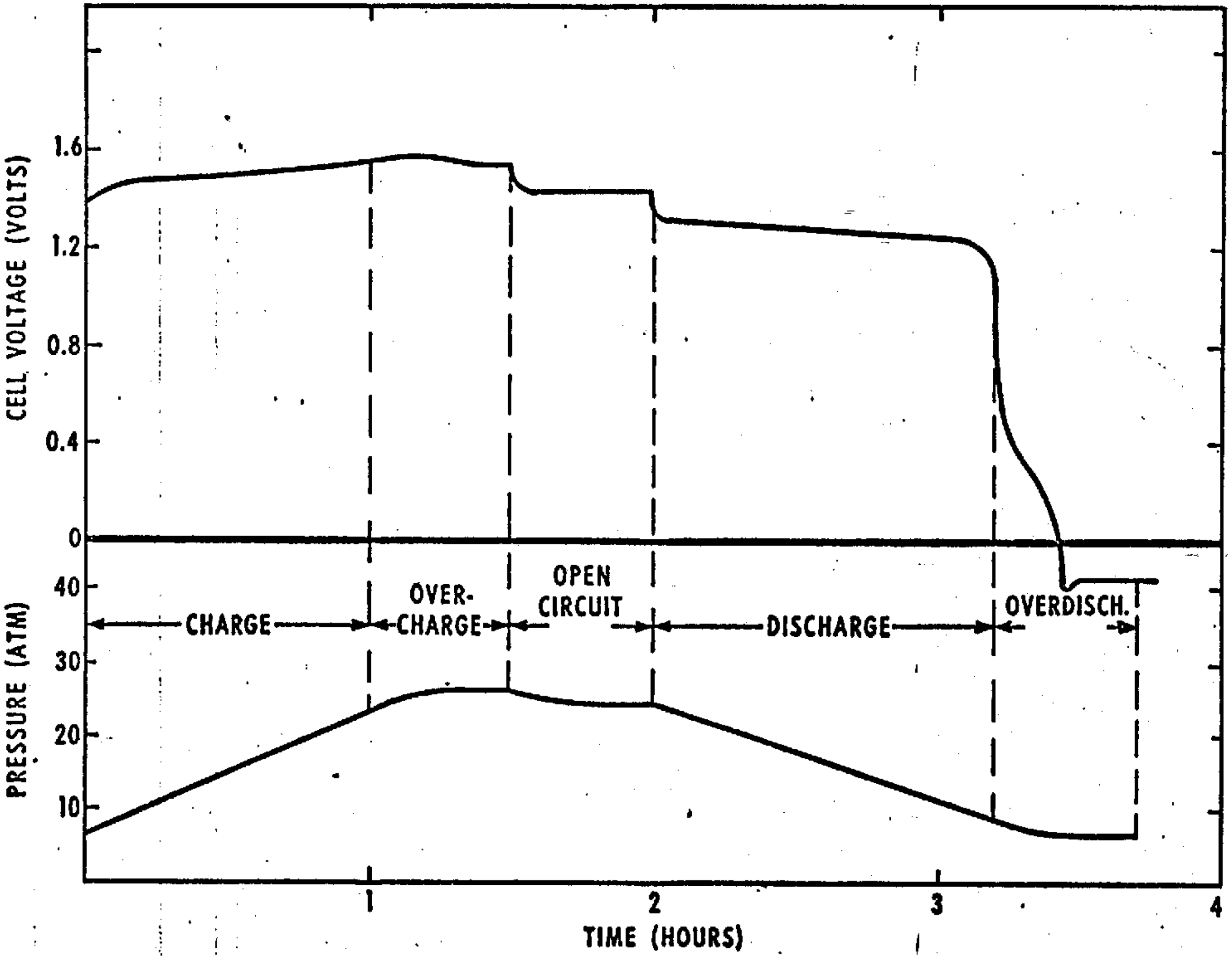


Figure C-6 - Charge-discharge characteristics of a single Ni-H_2 cell

Milestone Chart

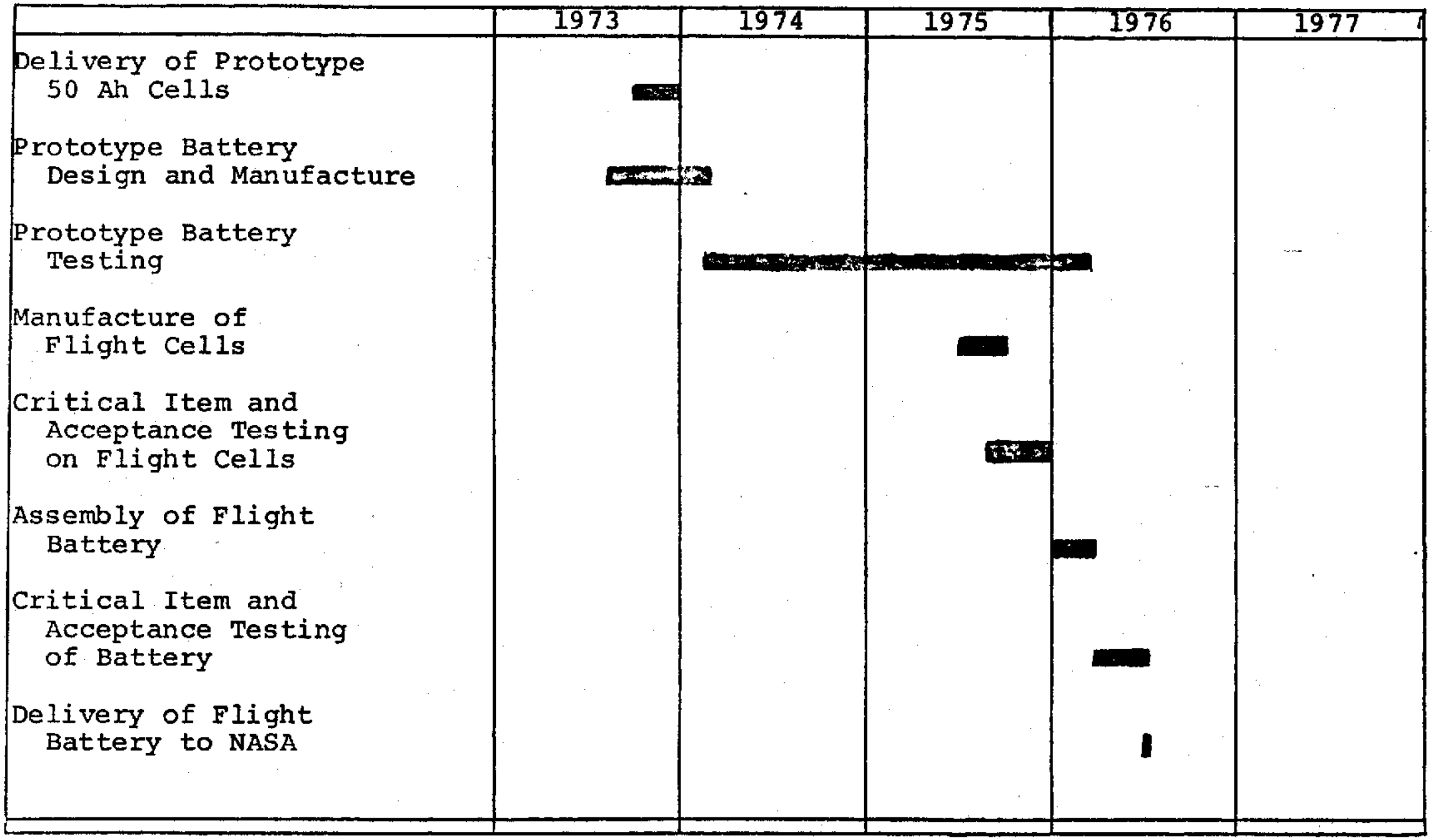


Figure C-7 - Development schedule for the 5-cell SERT C battery

25-FOOT-DIAMETER TANK

Description

The 25-foot tank is designed primarily for testing thrusters 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. Thrusters may be tested individually or in arrays; the primary limitations are mounting space and/or propellant condensing capacity.

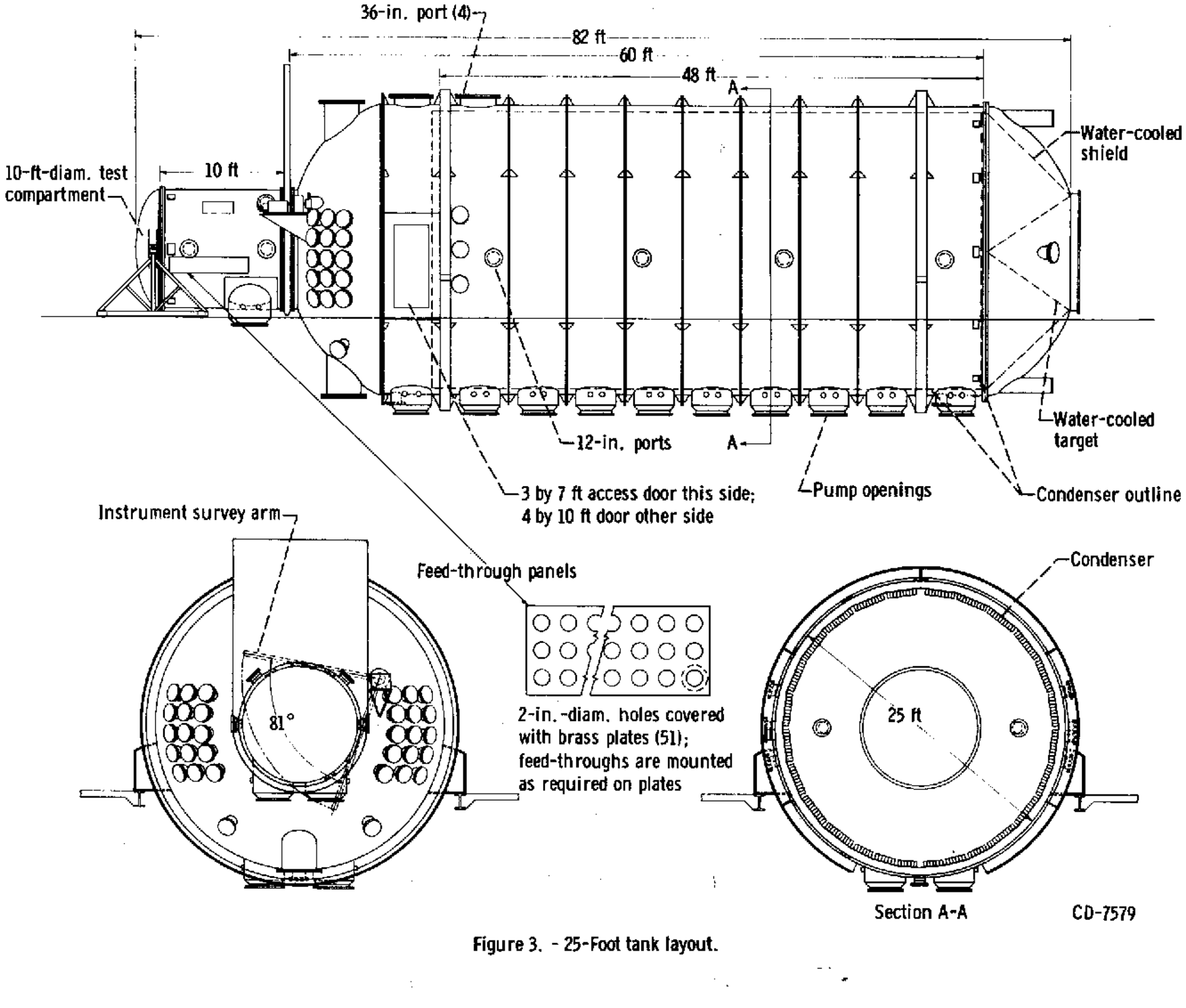


Figure 3. - 25-Foot tank layout.

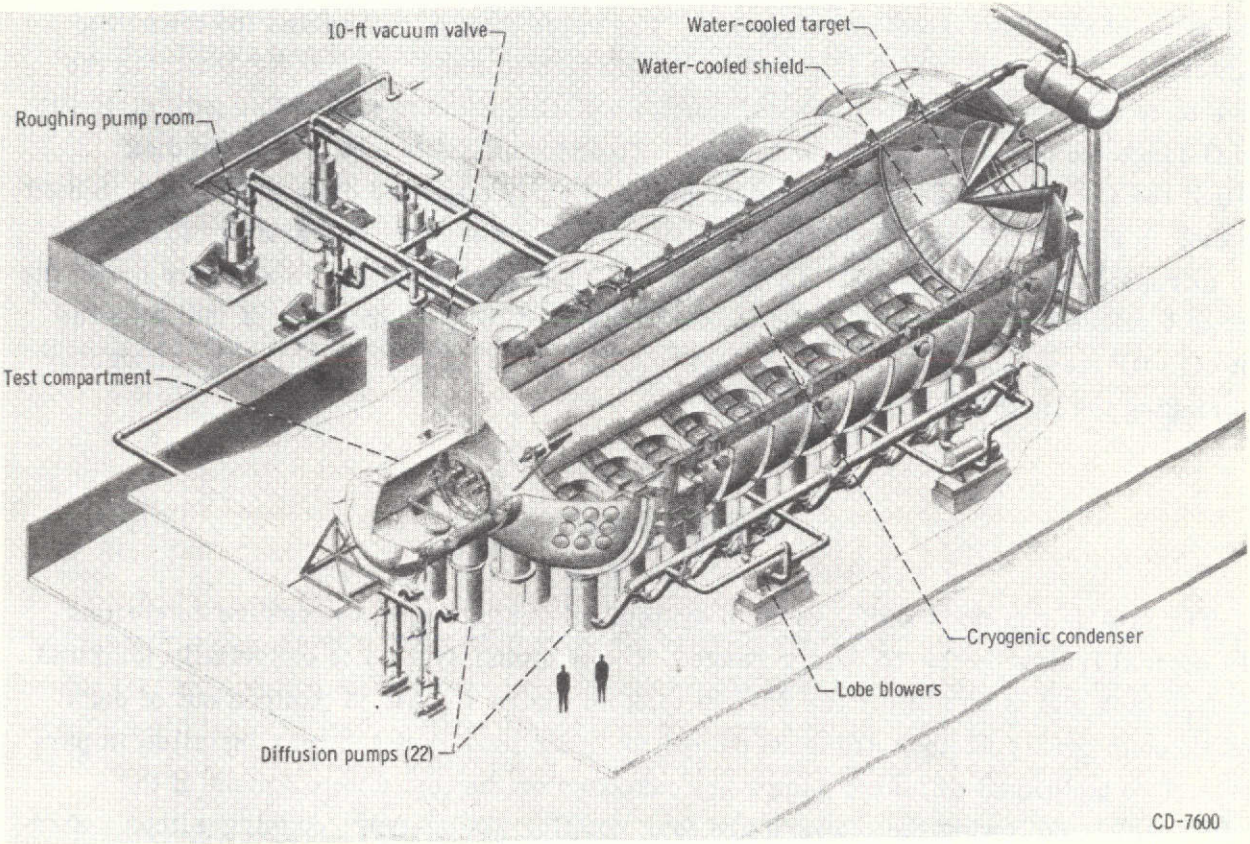


Figure 4. - Cutaway view of 25-foot-diameter tank.

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

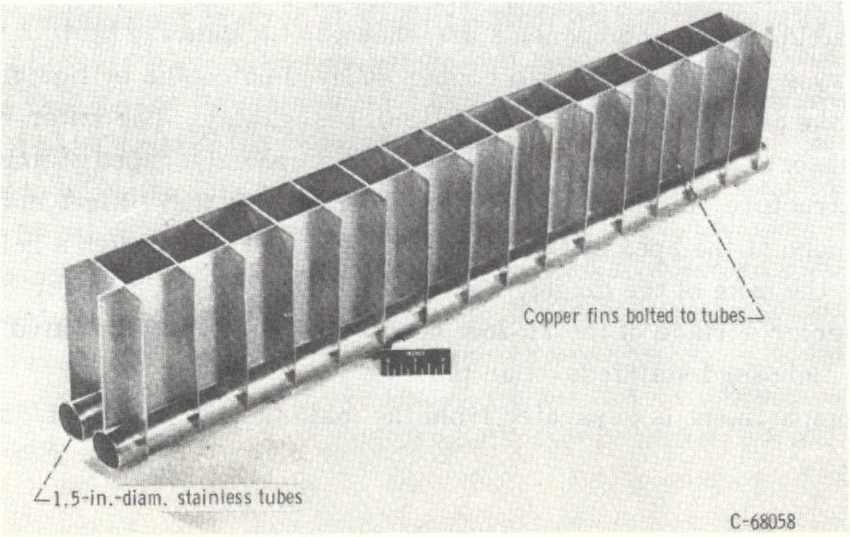


Figure 5. - Section of honeycomb condenser.

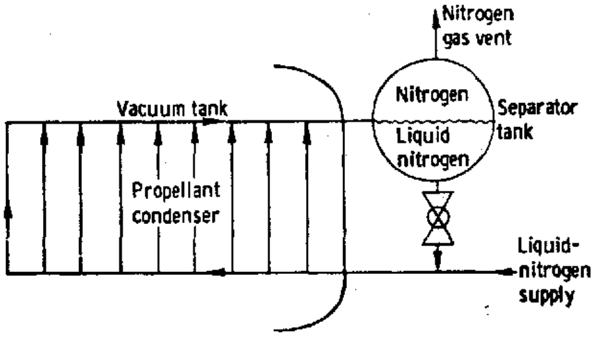


Figure 6. - Flow diagram of 25-foot-diameter tank condenser system.

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.

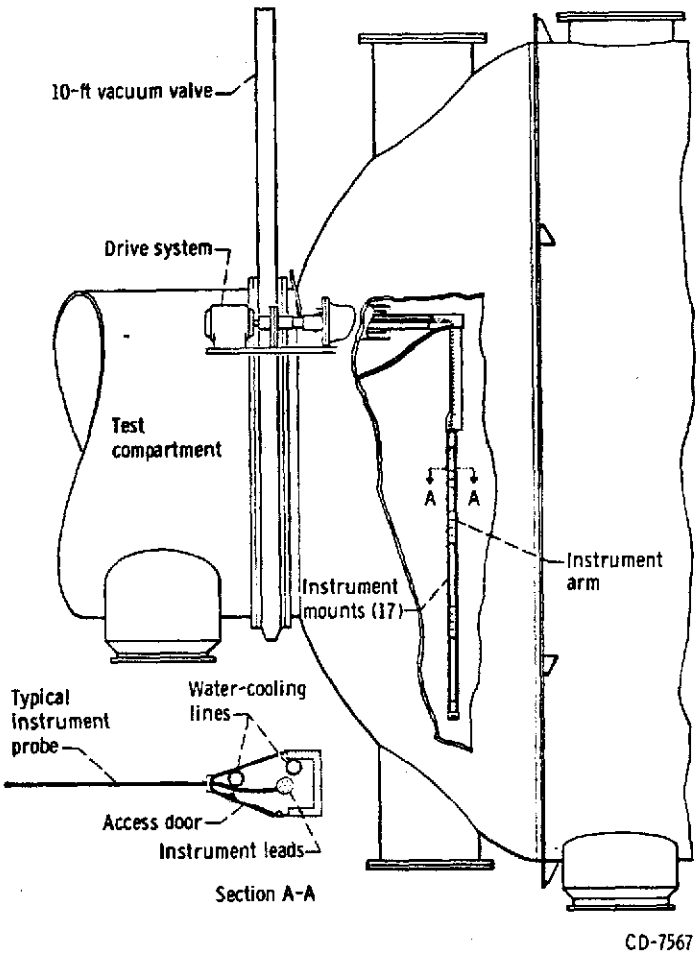


Figure 7. - Instrument survey arm.

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

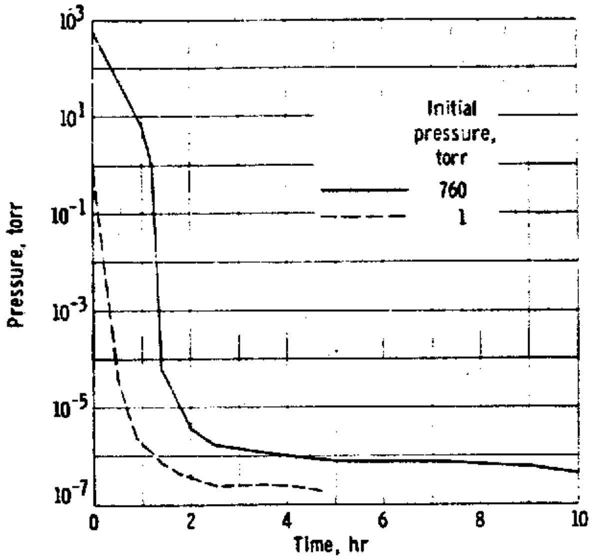


Figure 8. - Pumpdown rate with 20 oil diffusion pumps, liquid-nitrogen-cooled traps, and condenser. 25-foot-diameter tank.

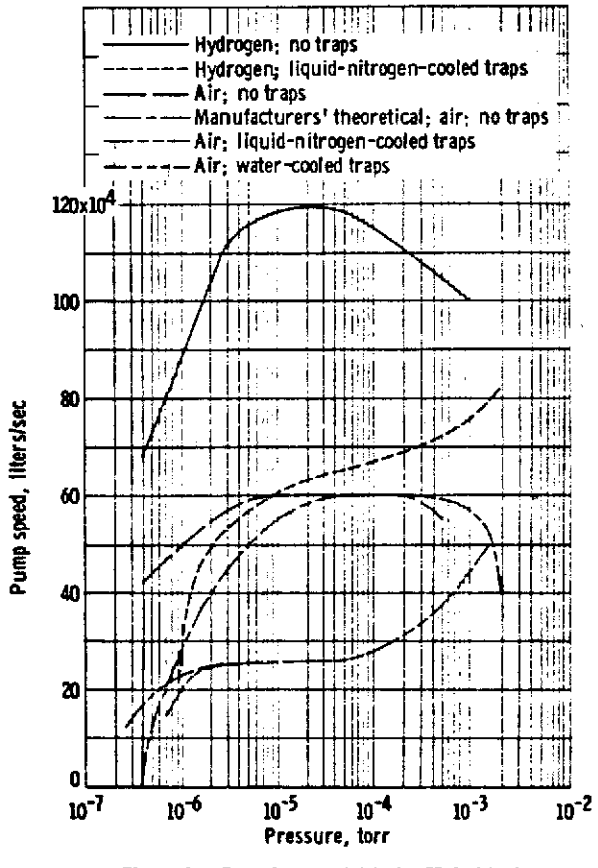


Figure 9. - Pumping speed data for 25-foot tank.

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 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.

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

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 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.

Figure 11. - Nine ion thrustor array.

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

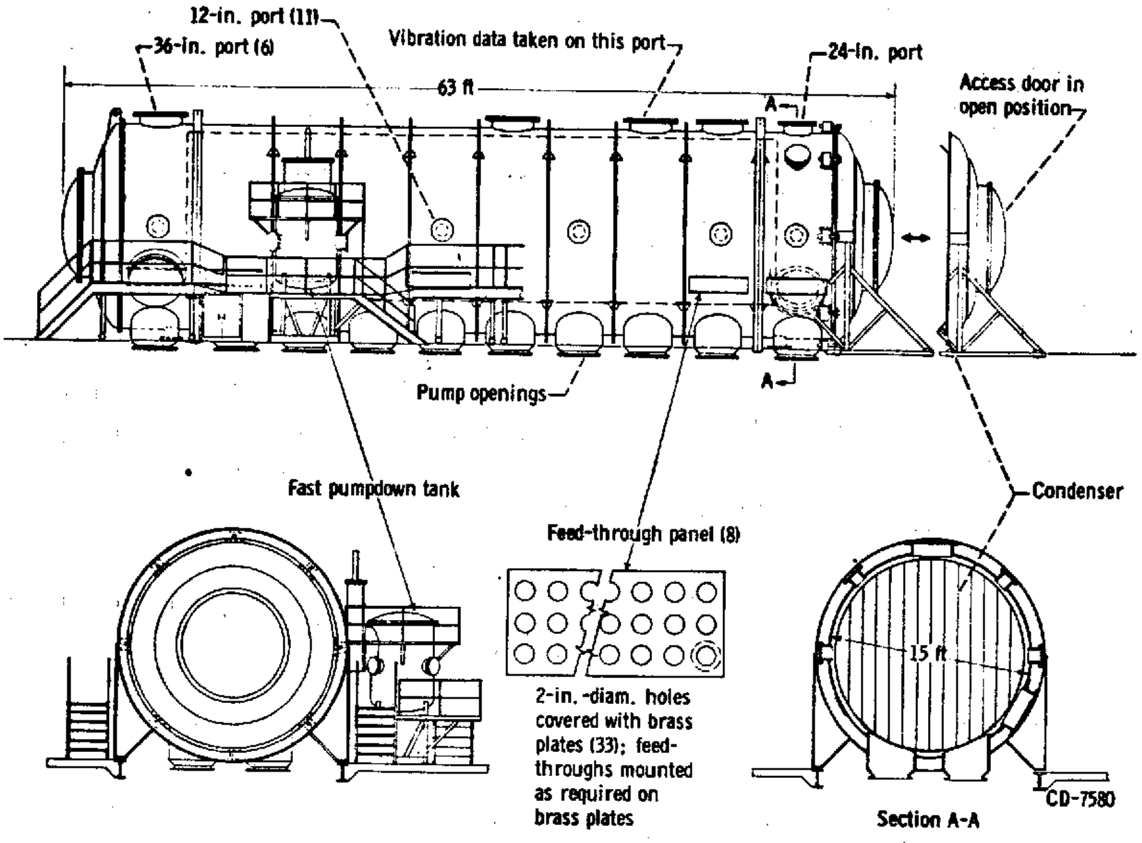


Figure 12. - 15-Foot tank layout.

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° 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.

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

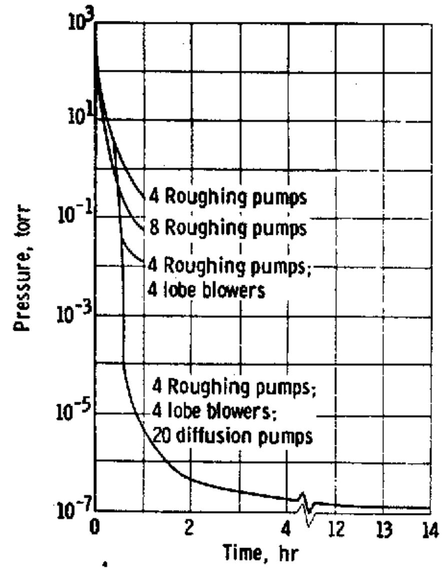


Figure 13. - Pumpdown rate for 15-foot tank.

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° 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

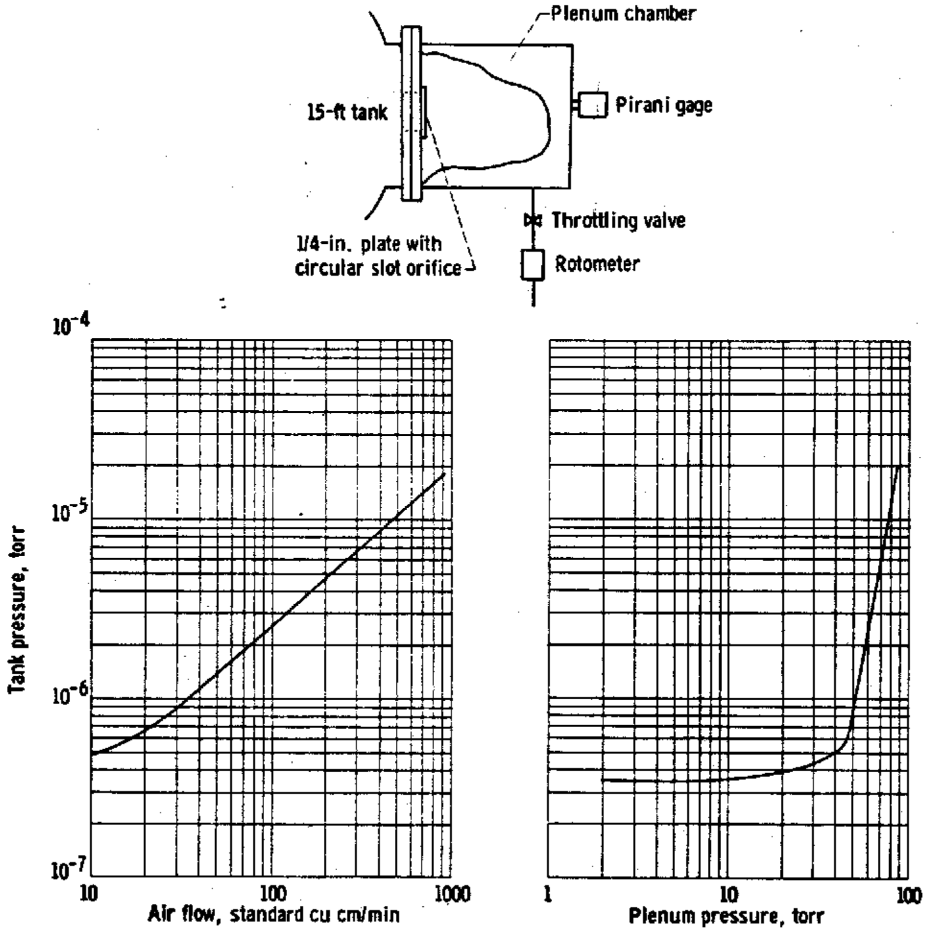


Figure 14. - Simulated air bearing leak.

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 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.

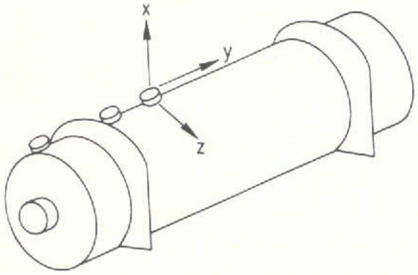


Figure 15. - Axis notation.

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

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.

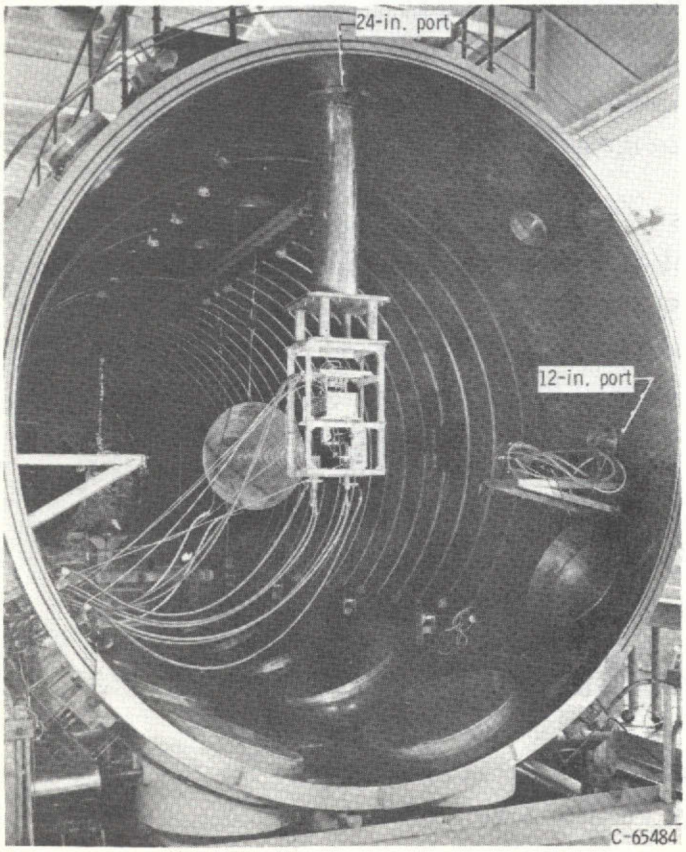


Figure 16. - Port-mounted thrustor.

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-

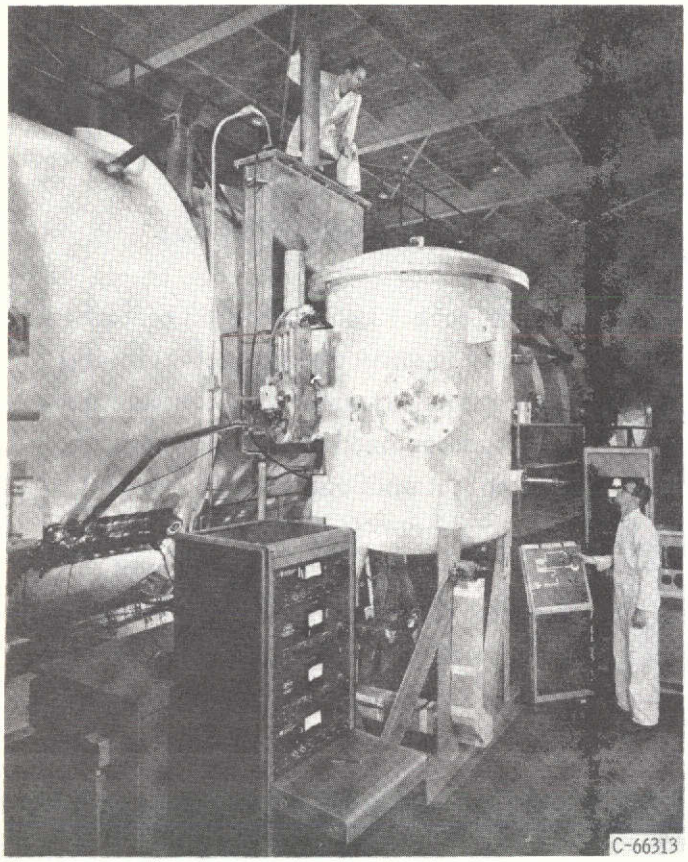


Figure 17. - Fast pumpdown tank.

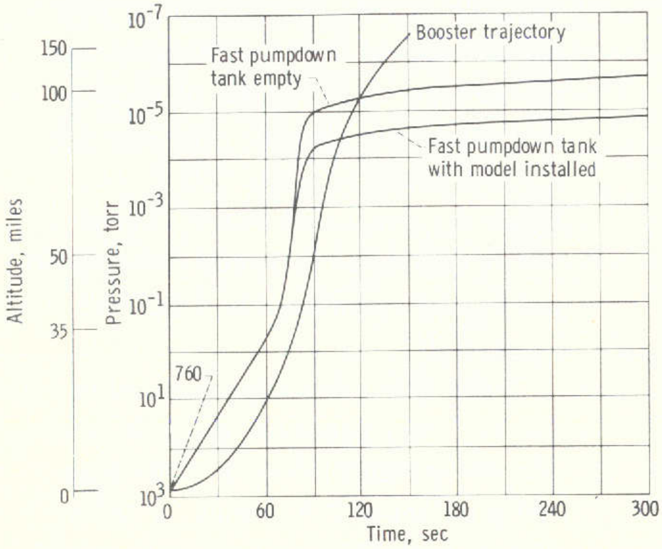


Figure 18. - Fast pumpdown tank performance data.

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 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-

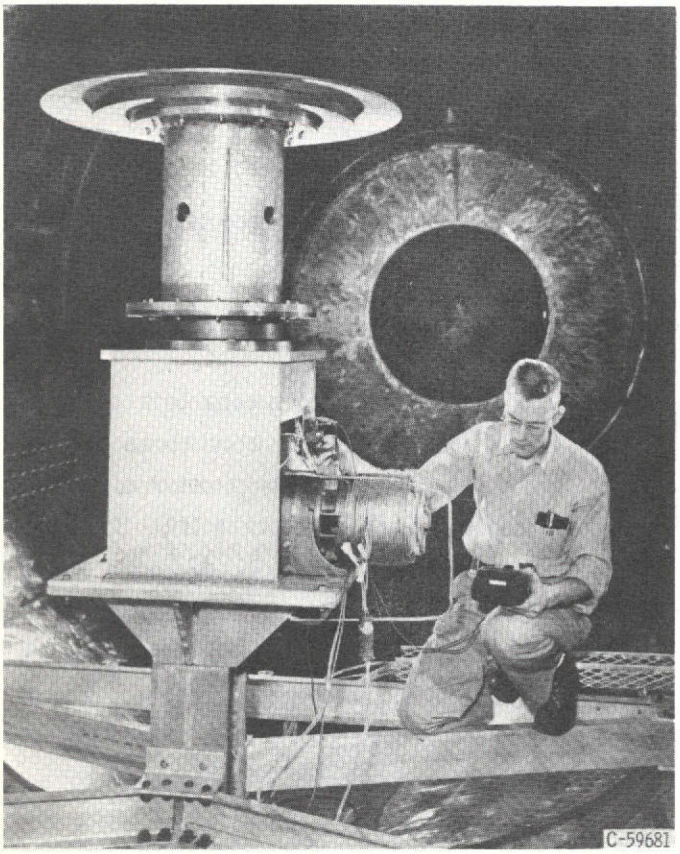


Figure 19. - Spin table with dummy load.

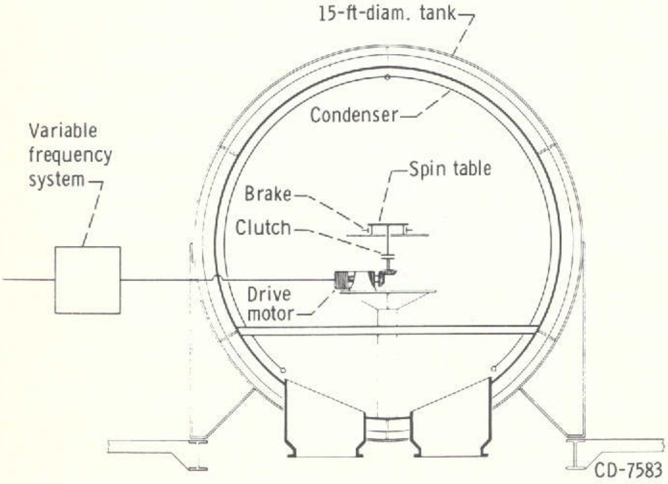


Figure 20. - Schematic of spin test apparatus.

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.

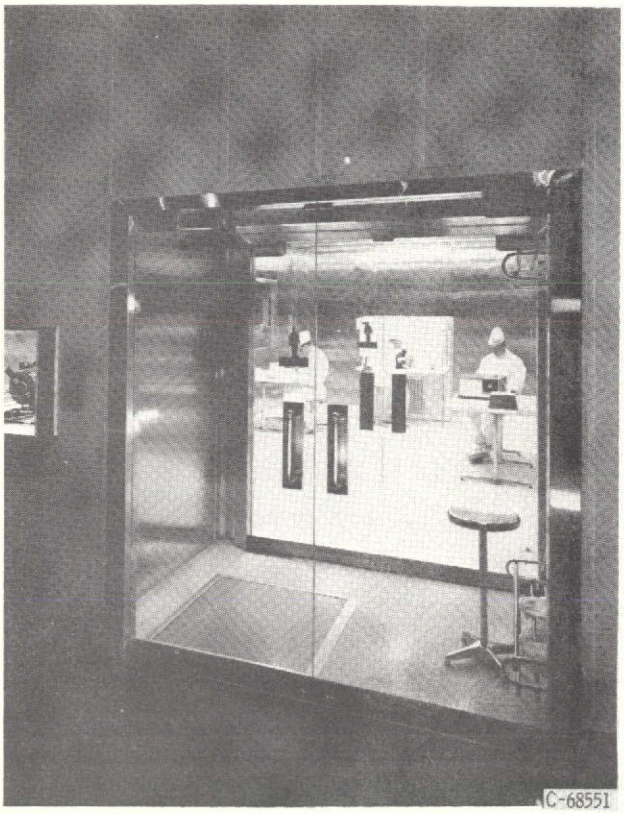


Figure 21. - Clean room.

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 ±2° 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.