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DESIGN CONSIDERATIONS AND REQUIREMENTS FOR INTEGRATING AN ELECTRIC PROPULSION SYSTEM INTO THE SERT II AND FUTURE SPACECRAFT

by Raymond J. Rulis
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TECHNICAL PAPER proposed for presentation at
Eighth Electric Propulsion Conference sponsored by
American Institute of Aeronautics and Astronautics
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Abstract

Integration of a new propulsion system into a spacecraft requires establishing design criteria for interface control. This paper delineates problems encountered during the integration of an electric propulsion system in the SERT spacecraft. Design criteria are established for future applications for high voltage handling, thruster breakdown, thrust vector control, and mechanical, thermal and electrical interfaces. Design considerations based on results from experiments aboard the SERT II spacecraft are discussed. Testing philosophy is presented for achieving long system life times on-orbit. Ground support equipment for propulsion system and integrated spacecraft testing during launch base activities is delineated.

Introduction

The development history of the SERT II spacecraft, a forerunner of spacecraft utilizing electric propulsion for extended missions, is replete with the design philosophies and problems which will be encountered by mission planners and spacecraft designers contemplating the use of electric propulsion for spacecraft missions. It is the goal of this report to enlighten these designers and planners by delineating design criteria and philosophies which evolved from the development of the SERT II spacecraft, and which were obtained from specifically designed flight experiments.

Some of the major objectives of the SERT II flight program were to investigate and then establish definitive solutions to the problems of spacecraft systems integration, launch vehicle integration, and in-flight interactions between the propulsion system, the space environment and the spacecraft. Design requirements for integrating electric propulsion systems into spacecraft, and the manner in which they were implemented into the SERT II mission, are presented. Mechanical and thermal design considerations for specific areas of integration, such as, spacecraft and launch vehicle interfaces, thrust vector control, propellant storage, and environmental constraints, are discussed. Similarly, electrical requirements for wiring, connectors, high voltage handling and electrical transients associated with thruster arcing, are also presented. The very important area of power conditioning and control is reviewed, in particular, for the areas of high voltage containment, and thruster control and overload protection. Results from flight experiments, as they pertain to establishing design criteria, are presented for the areas of surface contamination, RF interference, control of spacecraft potential, and thrust measurement and control. Reliability, a key to long duration mission success, is discussed. Testing philosophies that form the foundation of the effort which yields the desired mission lifetimes are presented. The operational procedures and ground support equipment that were found necessary to support launch vehicle and launch base integration activities are also delineated.

Design Requirements and Implementation

Mechanical-Thermal Requirements

Unlike most space propulsion systems, which are short lived, the electrical propulsion system, by design, will more than likely be integrated into the spacecraft and operated for a large portion of the mission. Hence, the spacecraft designer must now concern himself not only with the spacecraft systems, but also with an active interacting long duration propulsion system. The mechanical-thermal design considerations for this integration effort, and the methods used for their implementation in the SERT II spacecraft, are presented below.

Launch imposed environment. - As with most spacecraft, the launch environment, shock, vibration and acoustical, greatly influenced the mechanical layout of the integrated SERT II spacecraft and thruster system. Table 1 delineates the qualification levels that the total system was subject to. Flight acceptance levels are basically the same but reduced by a factor of one-third. Inasmuch as the SERT II thruster system was designed to permit gimballing to correct for thrust vector misalinement, hence, to reduce resultant disturbance torques to acceptable values, the gimbal rings were restrained during launch with pyrotechnique pin pullers. To insure that the propulsion system would endure the launch environment, and to obtain realistic dynamic inputs, it was required that the environmental testing be accomplished with a fixture which nearly duplicated the spacecraft. The fixture used during the development testing was an early mechanical structure model of the spacecraft. A philosophy of closely simulating the final installation for qualification testing greatly enhanced success during the qualification and flight acceptance testing of the entire spacecraft. Problems uncovered during the environmental testing were representative of those which might be encountered with any light weight mechanical assembly. These failures included gimbal flexures, pin pullers, propellant feed tubes and rigid wiring.

Problems encountered during launch simulation testing pointed out the thoroughness with which the basic design effort must consider the effects of the environment on all design details. As a corollary, it is concluded that the spacecraft designer must assure himself that the propulsion system he is integrating has successfully demonstrated an ability to withstand the environment his structure will provide. He must define and control the specification for the environmental qualification levels at the structure interface.

Thrust vector control. - Probably the single most important design consideration that a spacecraft designer will be confronted with when integrating an electric propulsion system into a spacecraft is the thrust vector. Figure 1 is an overall view of the SERT II spacecraft. The location of the C. M. (center of mass) of the flight configuration is shown. To minimize the disturbance torques that would be generated by errors in alinement of the thrust vector and the spacecraft C. M., both SERT II thrusters were gimballed. Postulated uncertainty in the location of the thrust vector with respect to the thruster centerline due to grid alinement, thermal effects, etc., at the time of spacecraft design integration, was approximately 5 degrees. Accordingly, the gimbals were designed, with conservatism, to provide a ±10 degree correction. Table 2 depicts the allowable disturbance torques that could be tolerated by the SERT II spacecraft. It is apparent that these values are quite small. The gravity gradient restoring forces utilized for primary control permitted a maximum misalinement of the thrust vector (at 6.2 mlbs.) at thruster turn-on of 4.0 degrees before loss of control ensued.

The following design information and criteria evolved during the program in regard to thrust vector management:

1. The thrust vector must be considered as a primary source of disturbance torque.

2. The spacecraft attitude control system should be sized to handle, with margin, the disturbance torque generated by position errors of the thrust vector alinement. For multi-thruster installation, where the thrust vector may be modulated and not alined with the spacecraft C. M., the attitude control system must consider all possible variations in thrust level.

3. Results from flight data indicate that the thrust vector for the SERT II system does not change with endurance life. Beam probe measurements show the beam profile to remain essentially constant. (1) Spacecraft attitude position data reaffirms the stability of the thrust vector. Thermal transients, such as start-up and shutdown, do not result in distortion or positional shift of the accelerator and screen grids, and hence do not cause changes in the thrust vector.

4. Flight data also verified that if accurate control is maintained of the spacecraft C. M., and the mechanical alinement of the thruster accelerator plate to the spacecraft C. M. is controlled, the resulting thrust misalinement error will be small and that gimbal positioning is not required. For the SERT II spacecraft the C. M. was held to a theoretical position accuracy of ±0.5 inch. The accelerator plate was alined through the C. M. to within ±0.25 degrees. Resulting maximum thrust vector misalinement from flight data was calculated to be 0.53 degrees in roll for one thruster, and 0.24 degrees in pitch for the other thruster. Accurate yaw measurements are not obtainable.

Spacecraft interfaces. - Design considerations for the mechanical and thermal interfaces with the spacecraft that required especial consideration for the SERT II program, and which will require resolution for most electric propulsion missions are as follows:

1. The thermal design must recognize that the power conditioning assembly for the thruster will dissipate a very large (by spacecraft standards) amount of heat. For example, with a 1 KW input, the SERT II power conditioner rejected from 125 to 150 watts continuous, depending on the input voltage. Figures 2 and 3 show the location of the power conditioners on the spacecraft and the radiator area required to maintain safe operating temperatures of 120° F on the base plate, and 160° F on components. Conversely, the large radiator area also presents design problems when the power conditioning is not operative. To insure that the components do not become too cold, a semi-passive thermal design utilizing louvers or heaters should be considered. In order to effect a good thermal design, the joint and mounting interfaces between the power conditioner, thruster and spacecraft require particular emphasis. Proper surface finishes, mounting torques and thermal interface material must be specified and evaluated through thermal-vacuum test.

2. Storage of propellant for the thruster(s) presents integration problems that are unique to electric propulsion. For a multi-engine installation, a choice must be made between totally self-contained thruster feed systems, as flown on SERT II, or a multi-feed single tank installation. Because the thruster operates at high voltage, the propellant storage and feed sytem must be either isolated electrically from the spacecraft or from the thruster. The spacecraft designer must consider the advantages and disadvantages offered by both types of installations. The SERT installation had provisions for both types of isolation; however, the development of a qualified feed system isolator was not compatible with program schedules.

The high density propellant used by electric propulsion systems presents other design problems. Dynamic loads presented by this concentrated mass as well as its effect on the location of the center of mass of the entire spacecraft must be considered. Propellant usage with mission time will affect the center of mass and, in turn, may cause attitude control problems due to thrust vector position errors with the changing center of mass of the spacecraft. The SERT II thruster installation considered both the affects of propellant mass location for dynamic inputs and for propellant usage. Figure 4 depicts the manner in which these problems were resolved - the thrust vector passes through the center of the propellant tank and the tank is located so as to minimize dynamic loads with relationship to the gimbal mount.

Thermal design layout of the propellant feed system must recognize that it is possible to both freeze or overheat the propellant. The location of the neutralizer feed system is particularly critical inasmuch as it is most likely to be exposed directly to the space environment. Non-operating systems can, if not properly thermally integrated, freeze. Multi-thruster installations also pose a major thermal problem. The thermal design of clustered thrusters must consider the affects of increased temperature on the vaporizer control for both the main and neutralizer feed systems. The necessity for a very thorough thermal design analysis for the installation of the propellant system is an obvious requirement.

Electrical Requirements

Electrical design requirements for electric propulsion system integration encompass all those that a spacecraft designer normally is faced with plus those that are truly peculiar to the electric propulsion system. The excellence of the electrical design in no small measure presages the degree of mission success. The single most important electrical design consideration faced by the spacecraft designer is the containment of high voltage electrical breakdown. With electric propulsion systems, high voltage breakdowns can be classified as those that are expected to occur as a characteristic of thruster operation, and those which must be contained by design. Described below are some of the more pertinent areas which must be considered to insure successful electrical integration of the propulsion system.

Outgassing. - High voltage breakdowns require a conducting media in addition to a relatively high potential difference between two electrodes. The medium most often associated with breakdowns in space is normally gaseous. Good design practice must recognize that outgassing must be controlled such that the combination of critical electrode spacing, potential difference and gas pressure is never realized during the operation of high voltage systems. Inasmuch as outgassing must have a source, only materials that have a very low vapor pressure must be used. Materials chosen should be viewed from the maximum operating temperature that they will experience. Construction of component enclosures must allow for the ready egress of evolved gases. In addition, the spacecraft design must provide an outgassing flow path that will readily vent all enclosed volumes. Figures 5 and 6 depict a typical design of enclosures for components used on the SERT II spacecraft and also the venting provisions provided by the spacecraft.

As stated previously, outgassing should be considered at the operating temperatures expected of the components and materials. In the case of the SERT II power conditioning system, the thermal design provided the power conditioner with an environment such that, in the off-state, except just prior to turn-on, the power conditioner was maintained at a temperature above that expected while operating. Hence, on high voltage turn-on, the evolution of gas, as a function of temperature increase, was minimized because the components had previously been outgassed at a higher temperature. This design feature was not required for operation, but came about as an extra when it was found necessary to maintain the low temperature limit of the power conditioners in the off-state.

Wiring. - Design considerations for high voltage spacecraft wiring, particularly that used in the thruster and interconnections between thruster and power conditioning systems, resulted in the selection for the SERT II spacecraft of MIL-W-81381(AS) Navy, "Wire, Electric, Polyimide-Insulated, Copper and Copper Alloy" as the standard wire. Of particular concern in the selection of the wire was the dielectric strength at relatively high temperatures, radiation resistance when exposed to the space environment, outgassing characteristics, cut-through resistance, long-life capability in vacuum under stress, and basic construction. A thorough search and evaluation of available high voltage wire resulted in the selection mentioned with a 6.5 mil insulation. This configuration afforded a dielectric strength of 22 KV at 260° C. Other types of high voltage wire were considered and used in protected and radiation-shielded components. Silicone rubber insulated wire was used in the power conditioning systems. The greater than 4000 hours of spacecraft and 10 000 hours of thruster system thermal-vacuum testing resulted in no electrical breakdown failures of the polyimide insulated wire.

In addition to the insulated individual high voltage leads, grounded shields were used over the harness between the thruster and power conditioner. The shield contains the electric fields adjacent to the insulation and provides a controlled breakdown path of spacecraft ground in the event of a breakdown in one of the leads.

High voltage connections. - Terminations for high voltage interface wiring, such as between the power conditioning system and the thruster, warrants particular attention. It is recommended that the decision be made early in the spacecraft design to sacrifice the ease of making electrical harness terminations, with standard type aerospace connectors, for reliability. Presently, no qualified commercially available standard high voltage connector is known to be in existence. Modified connectors, which are vented, have reportedly been successfully flight tested.(2) The high voltage connector design that was successfully implemented in both the power conditioning and thruster system design is shown in Figs. 7 and 8.

Specifications for the ceramic insulated terminals indicate an average corona start rating of 5.6 kV-rms, and an average flashover rating of 10.1 kV-rms. Both the thruster and power conditioner high voltage terminations are protected from the thruster and space plasma by stainless steel screen which also permits ready egress for outgassed material. The designs delineated above have been qualified very extensively through thermal-vacuum integrated systems testing. It is very strongly recommended that this basic design configuration be utilized for interconnection terminations between the thruster-power conditioner system.

High voltage design considerations. - To insure that a reliable, safe design has been effected for the high voltage system of the spacecraft, the designer must begin with and generate solutions to the most elemental details of the system. It is not the intention of this report to discuss the applicable philosophies, in detail, for insulation, shielding, encapsulation, etc. However, it is strongly recommended that the chain of design approval authority responsible for the spacecraft design integration acquire, as a minimum, a working knowledge of the causes of electrical breakdowns and of the solutions that have been generated by other designers. Numerous reports and publications have been authored on the subject of electrical breakdown. Reference 2 is an excellent current summary on the causes of high voltage breakdowns in spacecraft, and of design principles which should guide designers away from some of the pitfalls experienced by previous spacecraft programs. A good presentation on the theory, with confirming experimental data, of breakdowns between electrodes, due to electrostatic stress produced by high electric fields, is presented in Refs. 3 and 4.

High voltage breakdown problems were encountered during the development of the power conditioning system for the SERT II thruster system. Details of the design execution of the power conditioner are presented in Ref. 5. A discussion on the use of lightweight dielectric barriers between high voltage and low voltage components to contain electrical breakdown is presented in this reference.

Electrical transients. - As discussed previously, the nature of the ion thruster is such that high voltage electric breakdowns are an expected characteristic of the thruster. From a total spacecraft system integration viewpoint, the containment of the energy released by the high voltage breakdowns is a major design consideration. Both the SERT I and SERT II spacecraft encountered problems in associated spacecraft systems that resulted from the thruster arcing phenomenon. The telemetry system which measured thruster parameters was found to be particularly susceptible to transient voltage damage which was precipitated by thruster arcing. A number of ground rules that evolved from the electrical system integration effort on the SERT spacecraft are presented for consideration and guidance for this problem area.

1. Current limiting of all high voltage outputs from the power conditioner should be mandatory to suppress the transient which results from a breakdown in the thruster. Both resistive and inductive limiting were incorporated in the SERT II power conditioner.

2. All telemetry outputs that emanate from thruster measurements should be investigated, during a total systems test, to determine whether excessive transient voltage peaks result when the high voltage breakdowns occur. For the SERT II installation, it was necessary to install inductive-capacitive filters in each telemetry line from the power conditioner system to limit transient voltages to a level which was compatible with telemetry system components.

3. Specifications should be imposed on all spacecraft systems and experiments for electromagnetic compatibility. All components and systems should be tested thoroughly to insure that they possess the ability to cope with electromagnetic radiation.

4. The total electrical design philosophy for handling ground paths, electrical harnesses, shields, etc., particularly across mechanical interfaces, should be reviewed early in the design integration effort to minimize affects of high voltage transients.

Thruster-Power Conditioner System Requirements

Most critical to the successful integration of an electric propulsion system into a spacecraft is the thoroughness in which the requirements for the power conditioning and control systems are defined and executed. The operational requirements and limitations of the thruster system must be defined in such a manner as to insure compatibility among the power supply, power conditioning, thruster, and in-flight operational control. Some of the more pertinent considerations, which the spacecraft integrator should have a basic awareness of and as they were applied to the SERT II spacecraft, are presented below. A detailed presentation of this area is found in Ref. 5.

Power supply. - Sources for the large amounts of electrical power required by electrical propulsion systems appear in the near future to be limited to large solar arrays. The power conditioner design must provide a capability to operate over a fairly large voltage excursion if the supply output is fed directly to the power conditioner. For the SERT II installation a voltage swing from 75 to 50 volts - no load to end of mission life - was specified as the operating range the conditioner design must cope with. In addition, undervoltage protection must be provided in flight for loss of solar array power due to attitude orientation, and solar eclipse encounters.

Short circuit and arc protection. - Recognizing that the thruster system has an inherent arcing characteristic, provisions must be incorporated in the power conditioning design to deal with this phenomenon. The SERT II system utilized so-called "instant-off" and an "overload shutdown" circuits to provide for arc suppression and to permit automatic shutdown in flight in the event repeated arcs were encountered.[5] Additionally, specifications were levied to insure total system reliability through initial design by requiring that all supplies be so rated that a continuous overload or short circuit between supplies would be tolerated.

High voltage considerations. - As previously mentioned in this report and now reiterated, the safe containment of high voltages is the single most important design consideration which must be imposed on the power conditioning designer. This consideration must be held paramount beginning with the circuit design, and carrying through the harness and grounding layout, mechanical layout, transformer design (vented or potted), connector design, dielectric insulation design, etc.

Reliability. - Long life propulsion missions inherently require high reliability. The spacecraft designer must consider various trade-offs, particularly for the basic design approach to the power conditioner. Presently, there appears to exist two basic philosophies of design for achieving high reliability. One approach stresses the use of minimum components, the other the use of redundancy by functional modular construction. There are decided advantages and disadvantages of both system approaches which may weigh more heavily one way or another, depending on the mission requirements and installation itself. Regardless of the design approach utilized, the mainstays associated with good design practice and reliability, namely, derating of component operating voltages, currents, and temperatures, stress analysis and testing, and the use of high reliability components, should be imposed through specification requirements.

Ground Command capability. - With long mission life, operating characteristics of discrete components, as the cathode and neutralizer, will change because of wear. Provisions should be incorporated in the control system to provide for readjustment of the initial operating points, as required, by ground command. Thruster start-up, shutdown, thrust modulation, and thrust vector control, though probably best managed by automated means on multi-thruster installations, should also be functions for which ground override command capability should exist.

Instrumentation. - Consideration should be given to providing the necessary instrumentation to determine the "well-being" of the thruster system, power system, and power conditioning system in-flight. The same instrumentation should permit the required flight readiness evaluation to be made of the integrated system at the launch site. Instrumentation for the above purposes should be thoroughly assessed against reliability requirements. Redundancy of design, e.g., for pressure transducer seals, should be stressed. Fail-safe provisions should be incorporated to protect the system being monitored, particularly in the area of high voltage measurements.

Design Considerations Resulting from Flight Experiment Data

Early in the formulation period of the SERT II program, it was realized that before electric propulsion could be considered ready for mission use, a number of outstanding questions associated with the operation of an integrated thruster system in space would have to be resolved. Accordingly, a number of flight experiments were generated to yield the desired data. Results from these experiments in relationship to providing design criteria for mission use are presented below.

Efflux contamination. - Flight results from an efflux contamination experiment yielded data from both of the two thrusters on the spacecraft. The experiment consisted of two small groupings of solar cells so designed that one effectively operated at the same thermal condition as the large spacecraft array, while the second group of cells was at a temperature which simulated a distance of approximately 2.0 AU from the sun. The location of the experiments was such that a small portion of the propellant and sputtered grid efflux was permitted to reach the exposed cells.

Results from flight confirm pre-flight ground test results(6) and indicate that the sputtered molybdenum is a major source of contaminant which causes serious coating of cells, optical surfaces and thermal control surfaces.(7) Mercury propellant efflux does not appear to be a problem. The spacecraft designer can readily obviate the thruster efflux as a serious problem by judiciously placing the thruster, or critical components, so no line-of-sight view of the accelerator plate exists. If placement presents a difficult problem, then shielding should yield the same results.

Electromagnetic radiation. - Concern has been frequently expressed over the possibility of radio frequency radiation generated by ion thrusters being a factor to consider in the design of electric propulsion spacecraft, in particular, the spacecraft communication systems. The RFI experiment carried aboard the SERT II spacecraft was primarily to investigate possible RF generation in the frequency bands of 300 to 700 MHz, 1680 to 1720 MHz, and 2090 to 2130 MHz. These are existant or planned frequencies for space communications.

Results as of this writing indicate that the background radiation from earth-based sources is of such intensity as to grossly reduce the resolution of the measuring instrument. The experiment layout is such that the receiving antenna looks at the ion beam, but the ion beam is directed nearly perpendicular to the Earth's surface. Thus, the background for the antenna is the Earth. The 300 to 700 MHz bands are nearly saturated at full scale almost continuously. The 1700 MHz and 2110 MHz bands are indicating noise levels that are about as expected from a purely thermal Earth. Radiation from the beam, when compared with that emitted from the Earth, does not appear to be a problem. However, the radiation levels that are of interest to deep space mission planners for communication systems are an order of magnitude lower than that received from the Earth. Issuance of design guidelines based on results from this experiment must await the completion of the data review and its analysis.

Thrust measurement. - With any spacecraft that incorporates an active propulsion system, an accurate knowledge of the thrust produced per unit of time is required if an end mission objective of being at a specific place at a specific time is to be achieved. In lieu of an accurate definition of thrust, the same knowledge must be obtained by measuring the change in flight parameters produced by the thrust.

One of the secondary objectives of the SERT II flight program was to accurately measure the thrust of the ion engine. Provisions were incorporated to measure thrust directly by means of a sensitive electrostatically-suspended accelerometer, and by orbit change. Instrumentation was provided to permit the thrust to be calculated from measurements of the ion beam current and accelerating potential.

In summary form, the results from flight yielded the following data: direct measurement by the accelerometer of thrust was accurate to within ±1.0 percent; thrust calculated from orbit change measurements was accurate to ±5.0 percent; calculated thrust from electrical measurements was accurate within ±2.6 percent. The above results are discussed in detail in Ref. 8.

It is recommended that missions which utilize electric propulsion, in particular, for multi-engine installations for deep space, provide provisions for direct thrust measurements.

Neutralization and control of spacecraft potential. - Another secondary objective of the SERT II mission was to investigate the interaction between the electrically integrated propulsion system and spacecraft, and the ambient space plasma. This was accomplished by measuring the potential of the spacecraft relative to the space plasma and the potential between the ion beam and the spacecraft. The existence of a significant spacecraft-space potential difference could affect the validity of certain types of experiment data, neutralizer lifetime and performance, and the net thrust from the ion thruster.

The results from flight data indicate that the spacecraft potential with an operating ion thruster is on the order of 12 to 28 volts.(1) Potentials of this magnitude have a negligible effect on the net ion beam thrust. It was demonstrated that the spacecraft potential could be varied by means of a bias power supply between the spacecraft ground and the neutralizer. Thus, the potential of the spacecraft can be adjusted so as to exert a minimum influence on experiments, overall spacecraft performance, and the ambient space plasma.

Reliability Through Testing

Propulsion System Endurance Tests

The SERT II program, when compared to other spacecraft programs, will probably best be known for its extensive testing program - in space and on the ground. From the onset, a basic philosophy was established to achieve in-flight reliability by first flying the spacecraft on the ground.

The mission requirement of a six-months endurance test of the propulsion system dictated that its long life characteristics be first established by extensive ground thermal-vacuum tests. The evaluation of the thruster system, particularly in the area of the main cathode and neutralizer, required what might be called "trend" tests of 500 to 1000 hours duration to verify or disprove design configurations.

When it was felt that the thruster design was firm, three life tests were initiated - two with the integrated power conditioner and one utilizing the total spacecraft systems. Results from these tests provided the confidence required for flight go-ahead. At that time, in excess of 10,000 endurance hours had been obtained on the basic thruster system, while the flight type power conditioner design had more than 5000 hours. The long duration tests provided information on the thruster system characteristics that changed with time, and proved the compatibility of the integrated system in such areas as materials and thruster-power conditioner control stability.

Total System Integration Testing

The extensive evaluation accorded the ion thruster system was also imposed on the total integrated spacecraft. Thermal-vacuum testing of the prototype spacecraft exceeded 3200 hours of which more than 2400 hours were with operational thrusters.

Figure 9 depicts the spacecraft system and the thermal-vacuum facility utilized for the endurance evaluation. Integrated systems testing provided the opportunity to assess the operation of the spacecraft under the various environmental conditions and operational modes which would be encountered throughout the planned mission. To insure the relevance and applicability of the data to a mission, configuration control must maintain the spacecraft being tested to the latest existant flight parts list. For the SERT II system, the systems integration testing sought answers to a number of specific design problems, confirmation of the design execution, and proficiency in flight control procedures.

Specific test objectives formulated for the SERT II mission, which are applicable for future missions, were as follows:

1. Establish electrical compatibility of the total integrated spacecraft system. Problem areas such as the effect of electrical transients, caused by thruster arcing, on other spacecraft systems should be particularly looked for.

2. Confirm the total thermal design. For the SERT II testing, the passive thermal control system used in flight demanded a very thorough evaluation program. The effectiveness of the radiator, or thermal system, which controls the power conditioner temperature must be established. Thruster components, such as the neutralizer system, which might see direct space exposure, warrant particular attention.

3. Evaluate the spacecraft under a simulated launch environment. Environmental testing of individual components, unless accomplished with large error or safety margin, will not provide the environment finally seen by these components in the integrated system. Early total system testing is recommended to establish confidence as early as possible in the spacecraft system.

4. Assess mission reliability. Long mission life times with active propulsion systems demand long duration evaluation tests on the ground.

5. Evaluate in-flight control procedures. Where mission requirements dictate a flight program with extensive ground control, then mission success depends on the effectiveness and response of the flight control personnel. Proficiency must exist to handle both the normal planned events and all possible emergency procedures. For the SERT II mission, the prototype spacecraft was "flown on the ground" from the flight control center, Fig. 10, for over six months. Experimenters, test conductors, and associated personnel were so trained that the actual in-flight control was near routine from the onset. All flight procedures, systems, and experiments had been previously exercised and analyzed on the ground.

Launch Base Activities and Launch Vehicle Integration

Thruster System Activities

The final phase, but one of the most important in the development of a propulsion system for space use, is the critical pad and launch operations phase. Solutions for the problems of thruster handling, preservation, launch vehicle integration, and final flight preparations were effected during the SERT II launch activities. Particular areas of concern are delineated below.

Thruster system handling. - Inasmuch as the SERT II type ion thruster incorporates a sensitive chemical catalyst in the cathode assemblies which can suffer degradation when exposed to moisture or contamination, the thruster system should be provided a relatively dry and clean atmosphere during storage, shipment, and when installed on the spacecraft. It is recommended that special containers be designed to provide this protection. Cleanliness should be assured during handling by utilizing a clean, white glove and uniform approach. Of particular concern is possible contamination of high voltage insulators from grease, oil, etc.

Environmental constraints during spacecraft assembly and launch vehicle integration. - The high standards of cleanliness normally associated with the thruster and spacecraft assembly are not found at the launch base, in particular, during spacecraft to launch vehicle mating. Humidity and temperature control can be obtained in the working areas provided on the gantry and should be required. Cleanliness is another matter. At least a minimum of protection should be afforded the thruster by bagging it with clean, static charge-free plastic until the spacecraft shroud is in position. The SERT II program bagged both the spacecraft as well as the thruster system. Figures 11 and 12 depict a typical environment at the launch facility for spacecraft assembly and launch vehicle mating.

Final pre-flight check. - Like most space propulsion systems, the ion thruster cannot be given a final operational check immediately before flight. The "health" of the thruster system is determined in the final thermal-vacuum testing of the thruster, preferably as a total spacecraft systems test. Very little validation can be done at the launch base, hence, it is very important that the configuration of the integrated system be left untouched after the final total systems test. Hi-pot tests of insulators and telemetry monitoring of the propellant storage system are basically all that is required to be accomplished at the launch base. It is apparent that the ion thruster poses few problems during prelaunch activities.

Power Conditioner Activities

Unlike the thruster system, the ion engine power conditioner and control system can receive a final operational verification test shortly before flight. Its handling at the launch base is not much different than that which any complex electronic system might be subjected to. As mentioned previously, one should recognize the importance of maintaining the total system integrity at the level reached during the final total systems test of the thruster and power conditioning system. If it is decided that a final operational check of the power conditioner is required to establish a last minute readiness status, because of a long time span between final systems test and the scheduled launch date, then special precautions are advised. The basic integration layout of the spacecraft must provide ready access to the interconnections between the power conditioner and thruster for test measurements and for inspection. The high voltage outputs of the power conditioner must be disconnected to permit electrical checkout with a thruster simulator. Inasmuch as lead lengths may attribute to calibration errors, in particular those with high frequency outputs from the power conditioner, the final checkout should be accomplished with the load simulator in close proximity to the spacecraft. Figure 13 depicts the ion thruster simulator used during the SERT II program.

From a reliability point of view, an axiom that is worth considering during the launch base checkout of the power conditioner is, "do not break electrical connections unless the original integrity can be re-established." Because the thruster system cannot be checked operationally at the launch base, the interconnections between the thruster and power conditioner must be such as to permit reconnections to be made with the highest degree of certainty. The open connectors, Figs. 7 and 8 (previously described), readily permit thorough inspection and the desired reliability assessment to be made, and for this reason a final launch base check of the power conditioner was made on the SERT II program.

Physical handling of the power conditioner at the launch base requires the same degree of care afforded the thruster system. White gloves, coats, etc. should be employed, and special care taken to insure that the high voltage components, connectors, etc., are not subject to contaminations during storage and handling.

Once the final checkout of the system is made with the simulated load, no further testing is possible. This makes for a very simple countdown with the launch vehicle. The launch base activities with both the SERT I and SERT II spacecraft have shown that the final checkout and launch readiness determination of an electrical propulsion system is relatively simple and straightforward, requiring a minimum of ground support equipment and personnel. As mentioned previously, the final system integrity of the total system is established during final thermal-vacuum systems testing and not at the launch site.

Conclusions

Some of the design pitfalls which were experienced during the integration of an electric propulsion system into the SERT II spacecraft have been presented. Solutions have been proposed for the known problem areas peculiar to electric propulsion integration. In particular, design considerations have been presented for the areas of mechanical, electrical, and thermal interfaces. It has been shown that the control of the thrust vector is as an important consideration with electric propulsion as it is with other propulsion systems. The spacecraft designer has also been offered solutions and guidance for the area of high voltage containment and system integration. It is concluded that effective control of high voltage systems is one of the most important design considerations facing the spacecraft designer.

Flight experiments have produced results which indicate that efflux from sputtered grid material could present serious problems if the basic design does not consider the view angle from the thruster exhaust to the spacecraft and protect against line-of-sight efflux. Preliminary data from other associated experiments show that the integrated spacecraft potential can be controlled by biasing the neutralizer, if desired, and that the area of RF generation from the beam requires further data analysis before design guides can be issued. Thrust measurement was reported to have been successfully achieved with a sensitive accelerometer to within an accuracy of ±1 percent.

It was also concluded that reliability, a key to mission success for long duration propulsion missions, can be established by conducting an extensive ground test program on the thruster system, power conditioner, and the total integrated spacecraft.

Criteria were established for the launch base activity phase of a flight program. It was concluded that an integrated electric propulsion system offers relatively few problems to consider during the final flight readiness verification of the system.

The design philosophies and guidelines enumerated, if considered early in the design integration phase, should materially assist the spacecraft designer in achieving a successful integration of an electric propulsion system in future spacecraft applications.

Test condition	Axis	Frequency range, Hz	Acceleration
Sinusoidal Acceleration	Thrust	10 to 13	2.3 g
		13 to 22	4.6 g
		22 to 400	2.3 g
		400 to 500	2.3 to 4.5 g
		500 to 2000	4.5 g
	Lateral	10 to 250	1.5 g
		250 to 400	3.0 g
		400 to 500	3.0 to 4.5 g
		500 to 2000	4.5 g
Random Acceleration	All	20 to 400	3.32 g's RMS
		400 to 2000	9.64 g's RMS
Shock	All	----------	14 g for 8 MS

TABLE I. - SERT II SPACECRAFT ENVIRONMENTAL QUALIFICATION SPECIFICATION

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6. Richley, E. A. and Reynolds, T. W., "Condensation on Spacecraft Surfaces Downstream of a Kaufman Thruster," TM X-52746, 1970, NASA, Cleveland, Ohio.

7. Staskus, J. V. and Burns, R. J., "Deposition of Ion Thruster Effluents on SERT II Spacecraft Surfaces," NASA TM X- , 1970.

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Orbital axis	Torque
Roll	3.6×10^{-3} ft-lb
Pitch	4.6×10^{-3} ft-lb
Yaw	2.6×10^{-3} ft-lb

TABLE II. - SERT II SPACECRAFT ALLOWABLE DISTURBANCE TORQUES

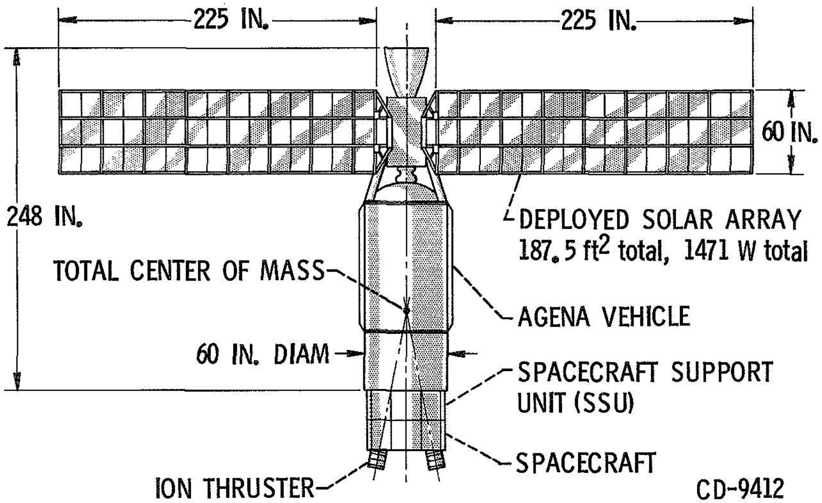

Figure 1. - SERT-II spacecraft.

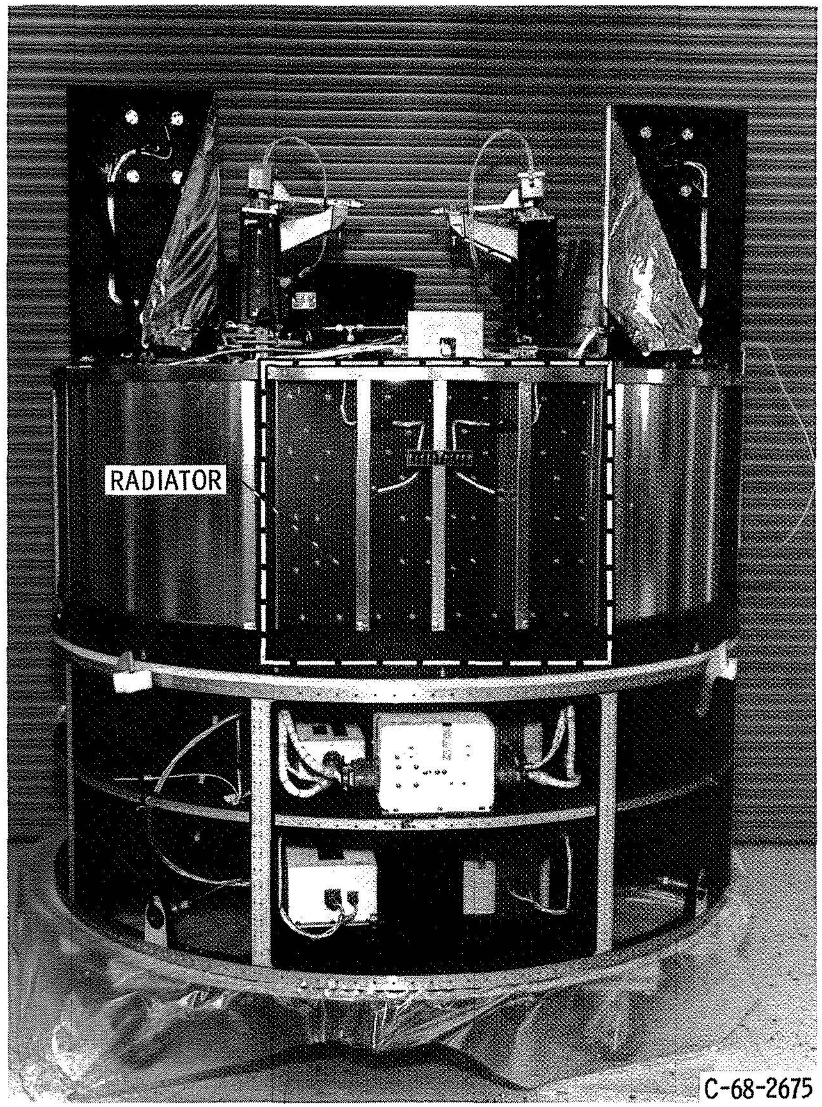

Figure 2. - SERT II spacecraft and power conditioning radiator.

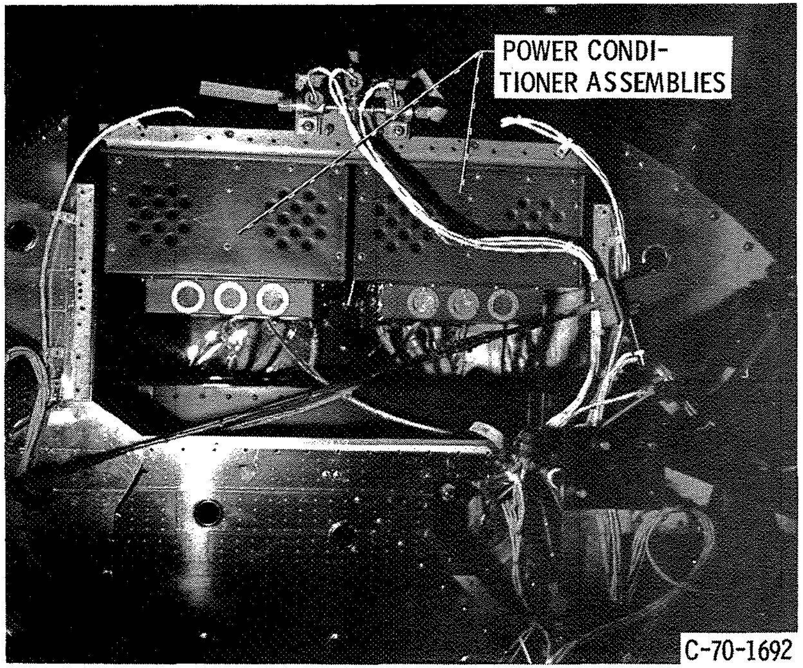

Figure 3. - SERT II power conditioning installation.

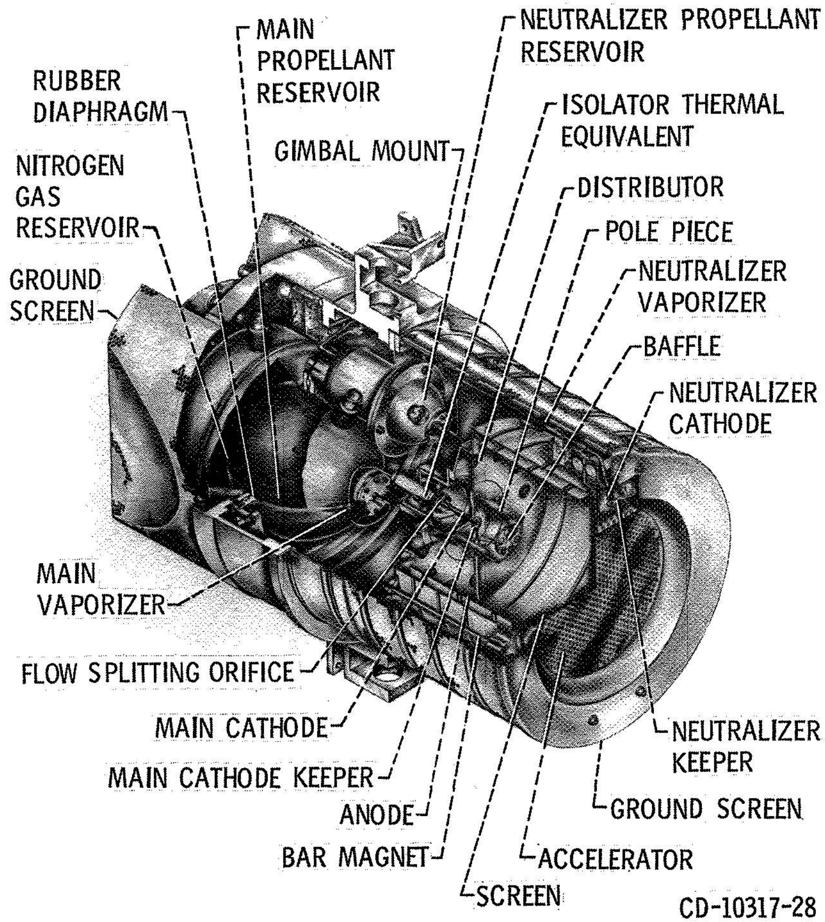

Figure 4. - SERT-II thruster system.

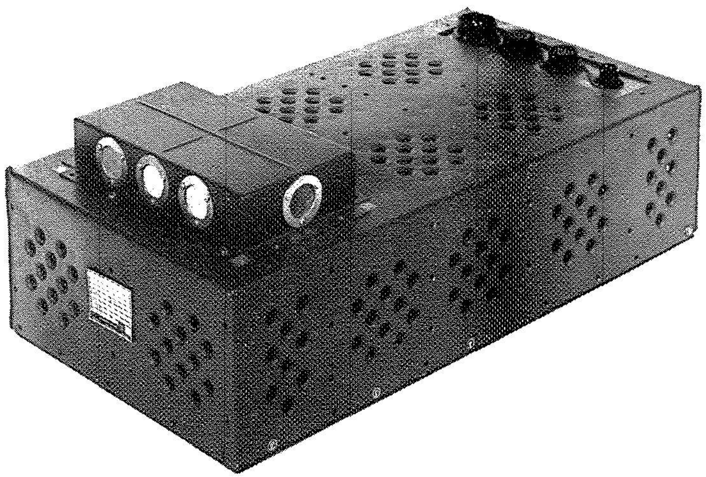

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Figure 5. - SERT II power conditioner showing outgassing holes.

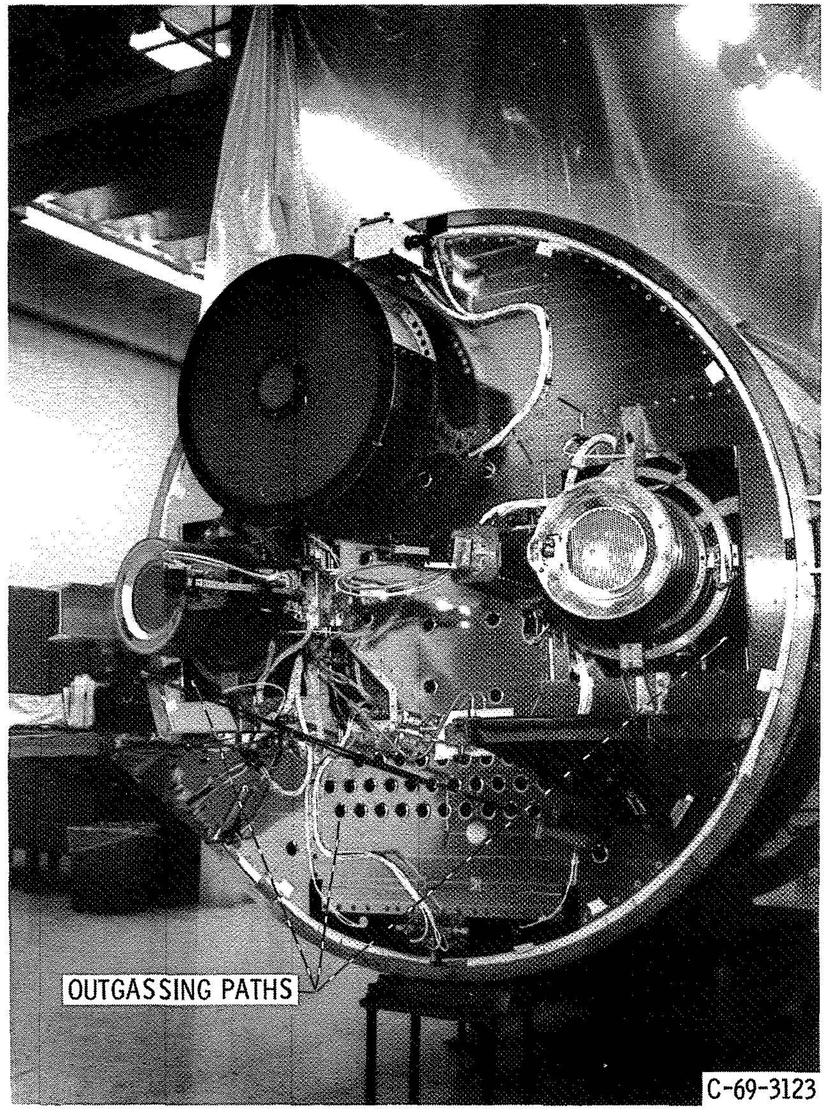

Figure 6. - SERT II spacecraft outgassing passages.

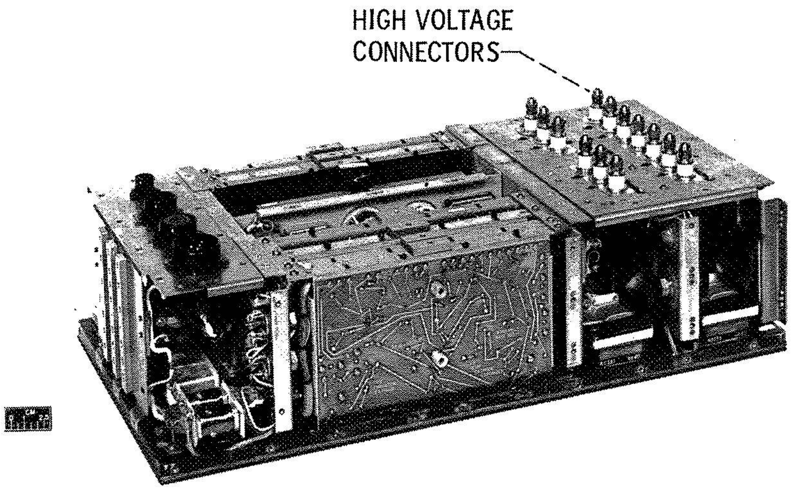

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Figure 7. - SERT II power conditioner showing high voltage connectors.

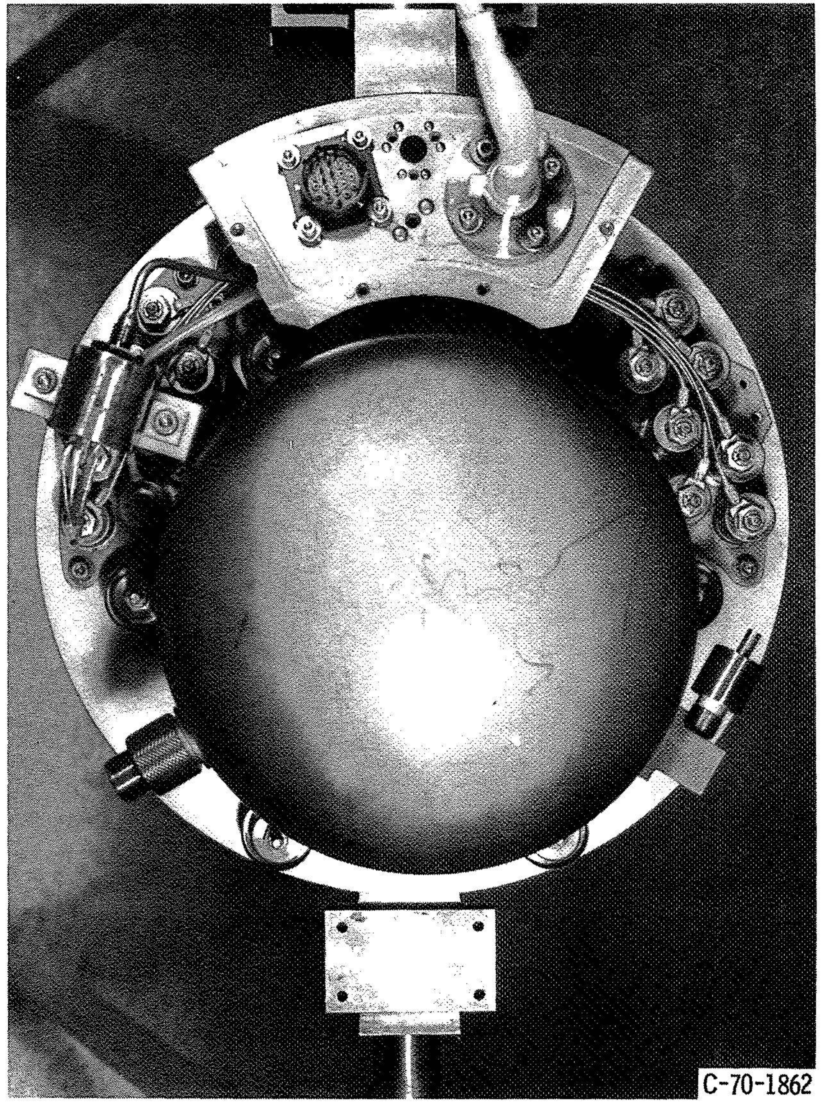

Figure 8. - Thruster system electrical connectors.

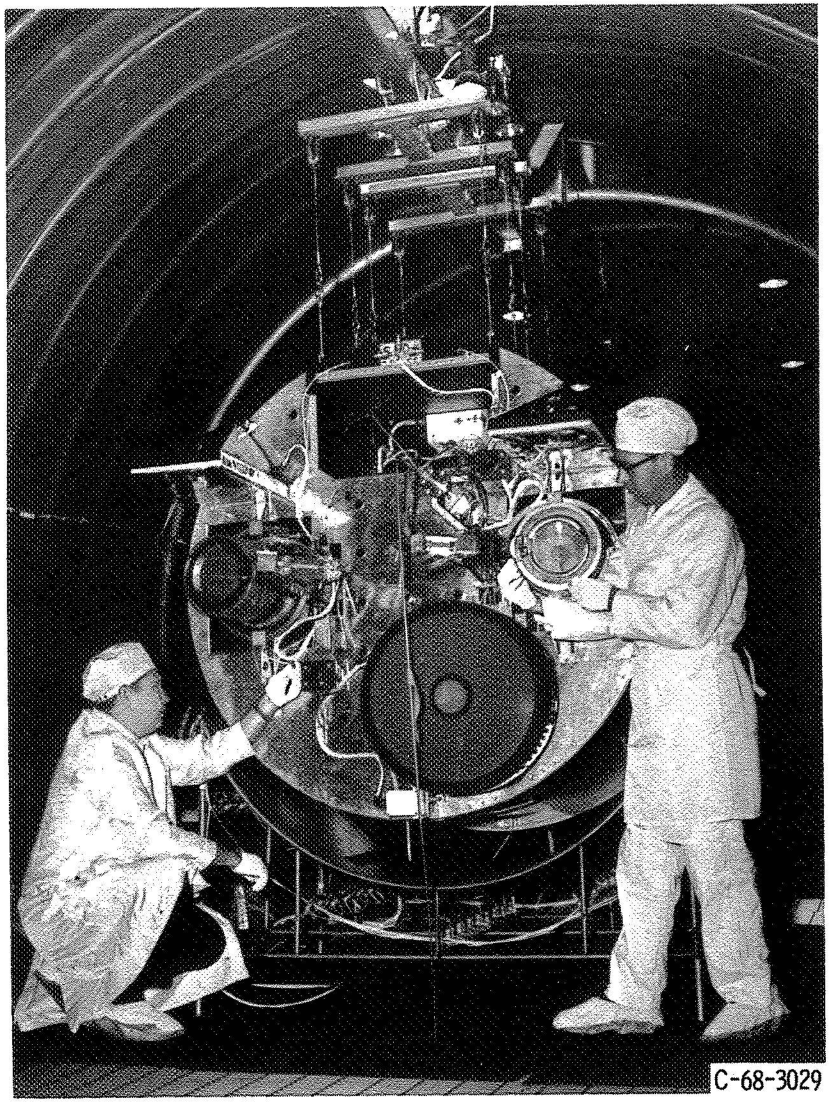

Figure 9. - SERT II prototype spacecraft in thermal-vacuum tank.

Figure 10. - SERT II flight control center.

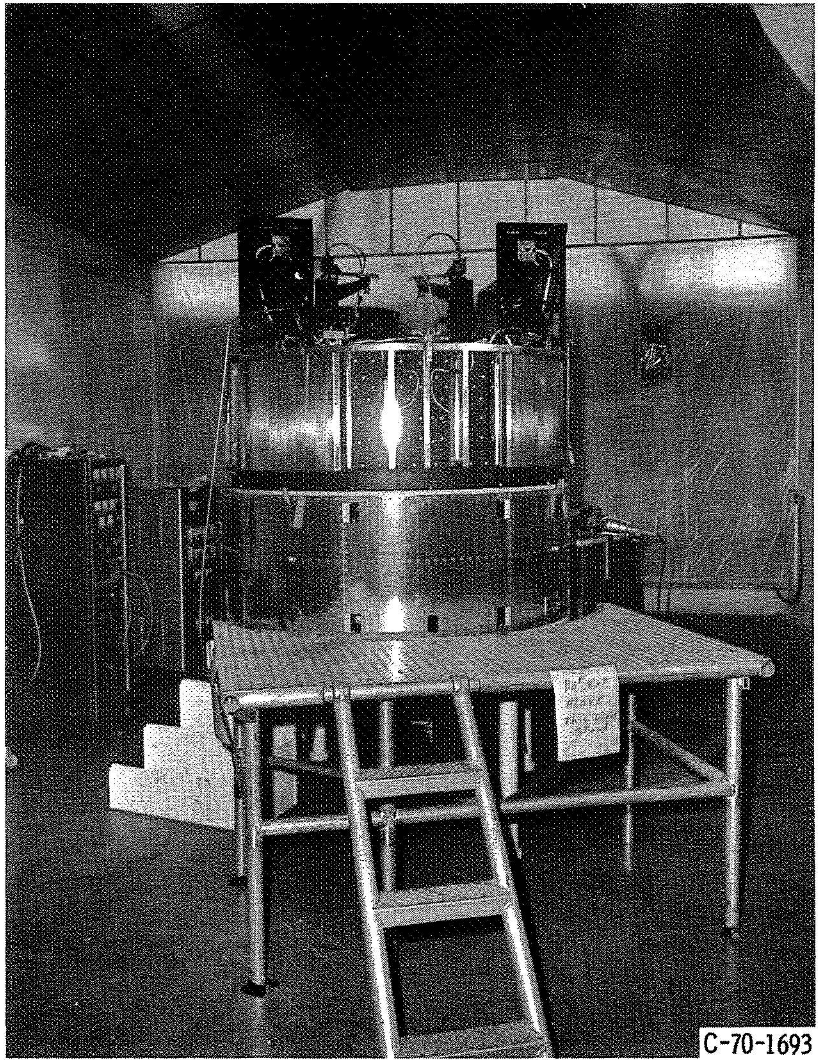

Figure 11. - SERT II spacecraft launch base assembly area.

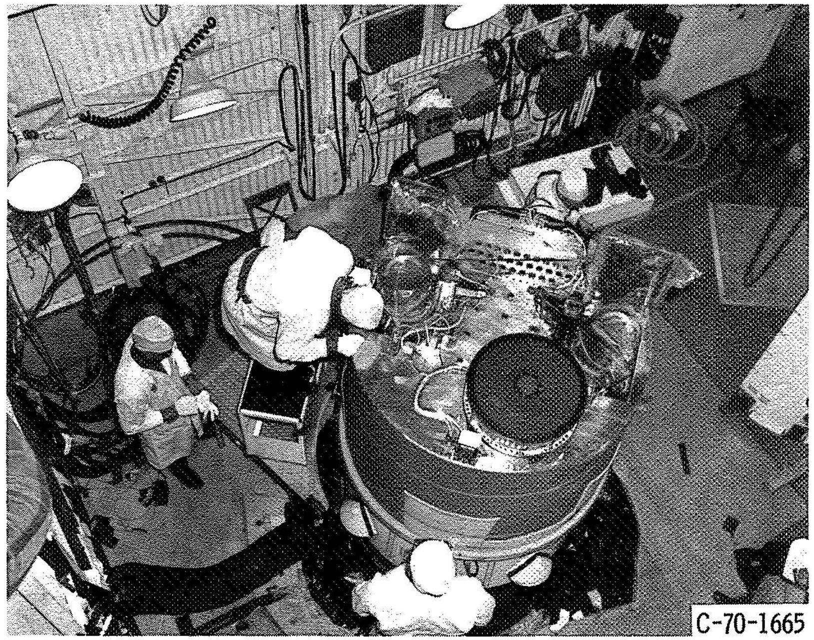

Figure 12. - SERT II spacecraft mated to launch vehicle.

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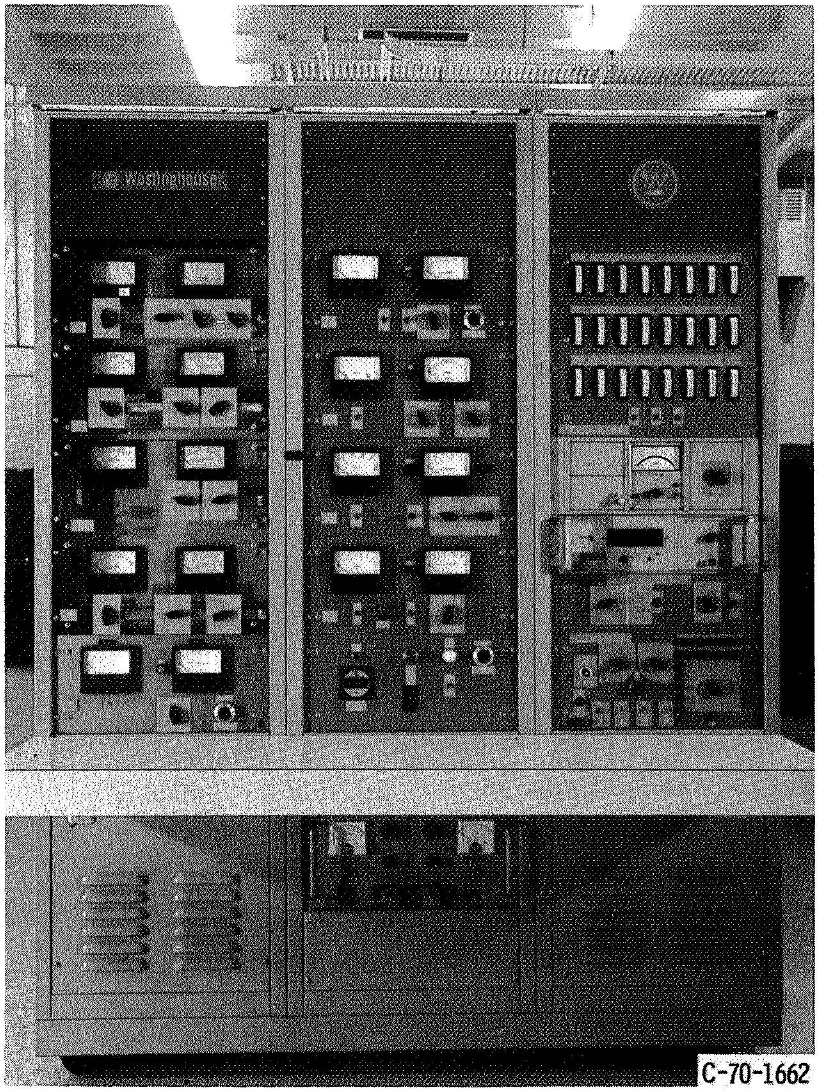

Figure 13. - SERT II ion thruster simulator.