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SERT II Papers

The following four papers on Space Electric Rocket Test (SERT) II have been condensed from papers presented at the AIAA 8th Electric Propulsion Conference, August 31–September 2, 1970. The first three represent a total of six preprinted papers from the NASA Lewis Research Center. W. R. Kerslake, who also assisted with the preflight set published in our January 1970 issue, has coordinated the effort on these three, and he wishes particularly to acknowledge the contributions of the following people to the SERT II effort and preparation of the original papers: F. D. Berkopec, R. J. Burns, D. C. Byers, G. S. Gurski, W. H. Hawersaat, A. C. Hoffman, R. J. Leser, V. K. Rawlin, K. F. Reader, J. B. Stover, and R. W. Vasicek. The fourth paper, from Lockheed, briefly covers solar array design aspects.

SERT II: Design Requirements for Integrating Electric Propulsion into a Spacecraft

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Design principles and experience gained from development of the SERT spacecraft and from results of experiments aboard SERT II are used to establish design criteria for high-voltage handling, thruster breakdown, thrust vector control, and mechanical, thermal and electrical interfaces for future applications of electric propulsion. Testing philosophy is presented for achieving long system lifetimes on-orbit. Criteria for propulsion system and integrated spacecraft testing during launch base activities are delineated.

Introduction

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. 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.

Presented as Paper 70-1123 at the AIAA 8th Electric Propulsion Conference, Stanford, Calif., August 31–September 2, 1970; submitted September 21, 1970; revision received December 15, 1970.

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Mechanical-Thermal Design Considerations

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 for the total system. It was required that the environmental testing be accomplished with a mechanical structure model which nearly duplicated the spacecraft. The failures experienced were representative of those which might be encountered with any light-weight mechanical assembly; they included gimbal flexures, pin pullers, propellant feed tubes, and rigid wiring.

An important design consideration is the thrust vector. Figure 1 is an over-all view of the SERT II spacecraft. The location of the c.m. (center of mass) is shown. 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 ~5°. Accordingly, the gimbals were designed, to provide a ±10° thrust correction. Table 2 lists the disturbance torques that could be tolerated by the SERT II spacecraft; they are quite small. The gravity-gradient restoring forces utilized for primary control permitted a maximum misalinement of the thrust vector (at 6.2 mlb) at thruster turn-on of 4.0° before loss of control ensured. The following design information and criteria evolved in regard to thrust vector management:

1) 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 multithruster 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.

2) Flight data indicate that the thrust vector for SERT II does not change with endurance life. Beam probe measurements show that the beam profile remains essentially constant.[1] Spacecraft attitude position data reaffirm the stability of the thrust vector. Thermal transients, such as start-up and shut-down, do not result in distortion or positional shift of the accelerator and screen grids, and hence do not cause changes in the thrust vector.

3) Flight data also verified that if accurate control of the c.m. is maintained, and the mechanical alinement of the thruster accelerator plate to c.m. is controlled, the resulting thrust misalinement error will be small, and gimbal positioning is not required. For SERT II the c.m. was held to a theoretical position accuracy of ± 0.5 in. The accelerator plate was alined through the c.m. to within $\pm 0.25°$. Resulting maximum thrust vector misalinement from flight data was calculated to be 0.53° in roll for one thruster, and 0.24° in pitch for the other thruster. Yaw measurements were not obtainable.

The mechanical and thermal interfaces between the propulsion system and the spacecraft that required special consideration were:

a) The thermal design had to recognize that the power conditioner (P/C) will dissipate a large (by spacecraft standards) amount of heat. For example, with a 1-kw input, the SERT II P/C rejected from 125 to 150 w continuous, depending on the input voltage. The large radiator area required also presents design problems when the P/C is not operative; components can become too cold. Proper surface finishes, mounting torques and thermal interface material must be specified between the P/C, thruster and spacecraft and evaluated through thermal-vacuum test.

b) Storage of propellant (Hg) presents integration problems that are unique to electric propulsion. For a multiengine installation, a choice must be made between totally self-contained thruster feed systems, as flown on SERT II, or a multifeed single tank installation. Because the thruster operates at high voltage, the propellant storage and feed system must be either isolated electrically from the spacecraft or from the thruster. 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-propellant density presents other problems. Dynamic loads presented by this concentrated mass as well as its effect on the location of the spacecraft c.m. (and any shift with time) must be considered. These problems were resolved for SERT II by making the thrust vector pass through the center of the propellant tank and locating the tank so as to minimize dynamic loads with relationship to the gimbal mount.

Table 1 Sert II spacecraft environmental qualification specification

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

Table 2 Sert II spacecraft allowable disturbance torques

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

Thermal design layout of the propellant feed system must recognize that it is possible to either 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. Nonoperating systems can, if not properly thermally integrated, freeze. Multithruster installations also pose a major thermal problem. The thermal design of clustered thrusters must consider the effects of increased temperatures on the vaporizer control for both the main and neutralizer feed systems.

Electrical Requirements

The single most important electrical design consideration faced by the spacecraft designer is the containment of high-voltage electrical breakdowns, which can be classified as those that are expected to occur as a characteristic of thruster operation, and those which must be contained by design.

Outgassing must be controlled such that the combination of critical electrode spacing, potential difference and gas pressure, which will permit high-voltage breakdown, is never realized. The materials used must have a very low vapor pressure at maximum operating temperatures that they will experience. Component enclosures must be designed to allow ready egress of evolved gases. For SERT II, the thermal design provided the P/C with an environment such that, in the off-state, except just prior to turn-on, the P/C 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 P/C's in the off-state.

For the high-voltage wiring, particularly that used in the thruster and interconnections between thruster and the P/C, the standard wire selected was MIL-W-81381(AS) Navy, "Wire, Electric, Polyimide-Insulated, Copper and Copper Alloy." Of particular concern were dielectric strength at relatively high temperatures, radiation resistance when exposed to the space environment, outgassing characteristics, cut-through resistance, life in vacuum under stress, and basic construction. The aforementioned wire, with a 6.5-mil insulation, 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 P/C's. The greater than 4000 hr of spacecraft and 10,000 hr of thruster system thermal-vacuum testing resulted in no electrical breakdown failures of the polyimide insulated wire. In addition, grounded shields were used over the harness between the thruster and P/C. The shield contains the electric fields adjacent to the insulation and provides a controlled breakdown path to spacecraft ground in the event of a breakdown in one of the leads.

Connections are a special problem. It is recommended that, for the sake of reliability, a decision should be made early in the spacecraft design process 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 exist. Modified connectors, which are vented, have reportedly been successfully flight tested.[2] The design that was successfully implemented in both the P/C and thruster system designs is shown in Fig. 2. 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 P/C high-voltage terminations are protected from the thruster and space plasma by stainless steel screen, which also permits ready egress for outgassed material.

It is strongly recommended that those in the chain of design approval authority acquire, as a minimum, a working knowledge of the causes of electrical breakdowns and of the solutions that have been generated by other designers. Reference 2 is an excellent current summary on this area. 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. Details of the design execution of the SERT II P/C are presented in Ref. 5, which includes a discussion on the use of lightweight dielectric barriers.

Electrical Transients

Both the SERT I and SERT II spacecraft encountered problems that resulted from thruster arcing. The thruster telemetry system was particularly susceptible to transient voltage damage. Some ground rules that evolved from the SERT effort are:

1) Current-limiting of all high-voltage outputs from the P/C should be mandatory to suppress the transient that results from a breakdown in the thruster. Both resistive and inductive limiting were incorporated in the SERT II P/C.

2) All telemetry outputs 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 P/C to limit transient voltages.

3) Specifications should be imposed on all spacecraft systems and experiments for electromagnetic compatibility, and all components and systems should be tested thoroughly in this respect.

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 effects of high-voltage transients.

Thruster-Power Conditioner System Requirements

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. Again, the SERT II experience is detailed in Ref. 5.

For electric propulsion systems using power from large solar arrays, the P/C design must provide a capability to operate over a fairly large voltage excursion if the supply output is fed directly to the P/C. For SERT II, a voltage swing from 75 to 50 v—no load to end of mission life—was specified. In addition, under-voltage protection must be provided in flight for loss of solar array power due to attitude orientation and solar eclipse encounters.

The SERT II system utilized so-called "blink-off" and an "overload shutdown" circuit to provide for arc suppression and to permit automatic shutdown in flight in the event repeated arcs were encountered.[5] Specifications required that all supplies be so rated that a continuous overload or short-circuit between supplies would be tolerated.

Presently, there appear to exist two basic philosophies of design for achieving high reliability: use of minimum components and use of redundancy by functional modular construction. Regardless of the design approach utilized, the mainstays of good design practice, 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.

For long missions, the control system should 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 multithruster installations, should also be functions for which ground override command capability should exist.

Sufficient instrumentation should be provided 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. For reliability, 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.

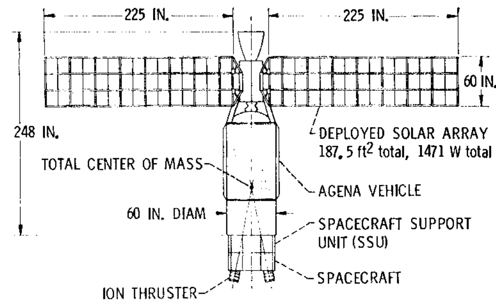

Fig. 1 SERT II spacecraft.

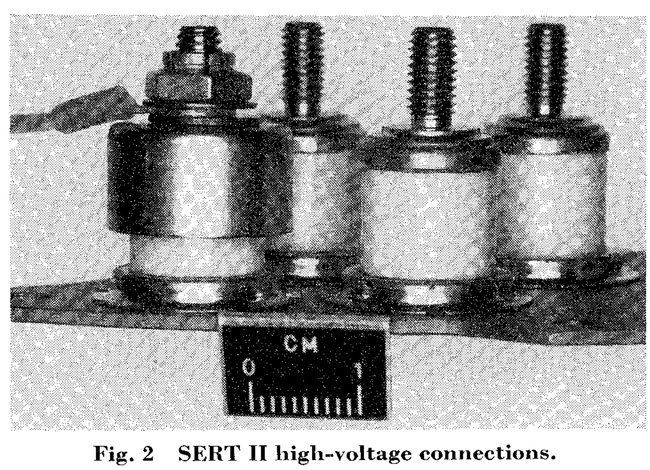

Fig. 2 SERT II high-voltage connections.

Design Considerations Resulting from Flight Experiment Data

Results from the SERT II flight experiment on efflux contamination confirm pre-flight ground test results[6] and indicate that the sputtered molybdenum, from the accelerator and screen grids, is a major source of contaminant which causes serious coating of cells, optical surfaces and thermal control surfaces.[1] Mercury propellant efflux does not appear to be a problem. The spacecraft designer can avoid this problem by judiciously placing the thruster, or critical components, so no line-of-sight view of the accelerator plate exists, or by shielding.

Unfortunately, the SERT II experiment on *radio frequency radiation* generated by ion thrusters failed, apparently because the 400-700-MHz measurement section suffered a failure before turn-on. 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. Issuance of design guidelines for communication systems based on results from this experiment must await the completion of the data analysis.

Results from thrust measurements in flight are discussed in detail in Ref. 7. 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. It was demonstrated in flight that the spacecraft potential could be varied be means of a bias power supply between the spacecraft ground and the neutralizer[1]; thus, it can be adjusted so as to minimize undesirable interactions.

Reliability through Testing

From the onset of the SERT II program, a basic philosophy was established to achieve in-flight reliability by first "flying the spacecraft on the ground." When it was felt that the thruster design was firm, three life tests were initiated: two with the integrated P/C and one utilizing the total spacecraft systems.[7] At that time of flight go-ahead, in excess of 10,000 hr had been obtained on the basic thruster system, while the flight-type P/C had more than 5000 hr. The tests provided information on variation of thruster system characteristics with time and proved the compatibility of the integrated system in such areas as materials and thruster-P/C control stability.

In the area of total system integration testing, the prototype spacecraft received >3200 hr of thermal-vacuum testing, of which more than 2400 hr was with operational thrusters. 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. Evaluate, e.g., thruster arcing problems.

2) Confirm the total thermal design. For SERT II, the passive thermal control system demanded a thorough evaluation program. The effectiveness of the radiator, which controls the P/C temperature, had to be established. Thruster components, such as the neutralizer system, which might see direct space exposure, warrant particular attention.

3) Evaluate the integrated spacecraft under a simulated launch environment as early as possible in the program.

4) Assess mission reliability. Long missions with active propulsion systems demand long-duration ground tests.

5) Evaluate in-flight control procedures, including effectiveness and response of the ground control personnel for both the normal planned events and all possible emergency procedures. For the SERT II mission, the prototype spacecraft, with all systems and experiments, was "flown on the ground" from the flight control center for over six months.

Launch Base Activities and Launch Vehicle Integration

Inasmuch as the ion thruster incorporates a 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. Cleanliness should be assured to avoid contamination of high voltage insulators from grease, oil, etc. The high standards of cleanliness normally associated with the thruster and spacecraft assembly are not found at most launch bases. Humidity and temperature control 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 ion thruster cannot be given a final operational check immediately before flight. Its "health" is determined in the final thermal-vacuum testing of the thruster. 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.

Unlike the thruster system, the P/C and control system can receive a final operational verification test shortly before flight. The basic integration layout of the spacecraft must provide ready access to the interconnections between the P/C and thruster for test measurements and for inspection. From a reliability point of view, a good rule is "do not break electrical connections unless the original integrity can be re-established." The high-voltage connectors, previously described, readily permit thorough inspection and the desired reliability assessment to be made if connections must be broken.

Once the final checkout of the P/C is made with a simulated load, no further testing is possible. This makes for a very simple countdown with the launch vehicle. SERT I and SERT II have shown that the final checkout and launch readiness determination of an electrical propulsion system is relatively straightforward, requiring a minimum of ground support equipment and personnel.

Conclusions

The control of the thrust vector is as an important consideration with electric propulsion as it is with other propulsion systems. Effective containment and control of high-voltage systems are prime design considerations facing the spacecraft designer.

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. The integrated spacecraft potential can be controlled by biasing the neutralizer, if desired, and rf generation from the beam does not appear to be a problem area.

Reliability can be established by conducting an extensive ground test program on the thruster system, power conditioner, and the total integrated spacecraft. An integrated electric propulsion system offers relatively few problems to consider during the final flight readiness verification of the system.

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SERT II: Mission, Thruster Performance, and In-Flight Thrust Measurements

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The SERT II spacecraft, containing two 15-cm-diam mercury electron bombardment ion thrusters, was launched February 3, 1970. One thruster operated for 5 months of the 6-month mission goal. The over-all thruster efficiency 0.68 remained constant; the specific impulse was about 4200 sec; and the thrust was 28 mN (6.3 mlb) for a thruster input power of 850 w. A second flight thruster operated 3 months in space. Both thrusters failed due to sudden shorts between high voltages which have since been determined as resulting from the neutralizer location. This paper compares the thrusters performance with corresponding preflight ground testing. The thrust in flight, determined by three different methods, is presented. In addition, the SERT II spacecraft and mission are described.

Introduction

SPACE Electric Rocket Test I (SERT I) in 1964 verified the production of thrust and the neutralization of an ion beam in space.[1] SERT II was launched February 3, 1970, for the purpose of demonstrating 6-month space operation of either one of two ion thruster systems on board. The spacecraft is powered by a nominal 1.5-kw solar cell array.[2] Companion papers describe the auxiliary experiments performed, and the development of the power conditioner.[3–5] The extensive developmental thruster ground testing, which established the confidence necessary for the flight, has been reported.[6–8]

Constant sunlight during the mission was achieved by selecting a polar orbit (Fig. 1) with an altitude (1000 km) and inclination such that the oblateness of the Earth precesses the orbit plane approximately equal to the angular rate at which the Earth moves above the sun. The orbit altitude was an optimization of gravity gradient torques, aerodynamic drag, solar cell degradation, required constant sunlight, and launch vehicle limitations. The Thorad-Agena boost vehicle launched from the Western Test Range (Vandenberg Air Force Base, Calif.), put the satellite into an elliptical transfer trajectory to 1000 km where a second Agena burn circularized the orbit. Once orbit was achieved, the Agena performed several key functions: 1) maintained active control of the satellite while it performed initial steps such as solar cell deployment; 2) mounting platform for the spacecraft and spacecraft support unit (SSU); 3) mounting base for the solar array; 4) gravity gradient orientation structure for entire mission; and 5) provided horizon scanners to determine satellite attitude.

The ion thrusters were offset 10° from the vertical, thus producing enough tangential thrust to raise or lower the orbit depending on the thruster selected. The orbit changing provided a direct measurement of thrust. Real-time thrust was measured by a miniature electrostatic accelerometer (MESA) and calculated from electrical measurements of the ion beam.

The use of flight-proven hardware (except ion thrusters) was an important ground rule. Table 1 lists equipment derived from previous missions. In many cases items were purchased to original specifications; in others, surplus hardware was used. Thermal control of the spacecraft was obtained by entirely passive means using thermal coatings on outer surfaces of the spacecraft and SSU.

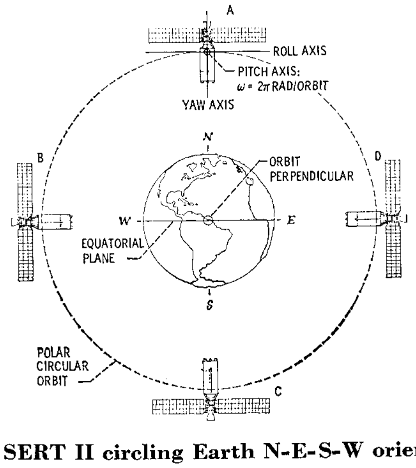

Fig. 1 SERT II circling Earth N-E-S-W orientation.

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