SERT II: Mission and Experiments

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The SERT II mercury electron bombardment ion thruster produces a neutralized 0.25-amp beam of mercury ions accelerated through +3000 v. The thrust is 6.2 mlb at an effective specific impulse of 4450 sec. The primary mission of SERT II is a 6-month space test of an electric thruster, using a space power source (solar array) and an efficient, light-weight, power conditioner. The primary mission and auxiliary experiments are described.

Introduction

THE SERT II (Space Electric Rocket Test II) thruster system received intensive development at the NASA Lewis Research Center starting in August 1966. At that time the thruster was defined as a nominal 1-kw power level Hg-bombardment thruster. A bombardment thruster was chosen because in the range of power and specific impulse desired for primary propulsion, it has a high over-all efficiency.[1] The 1-kw level was chosen because it is large enough to be representative of such missions (through multiple modules) and yet low enough to operate from an established solar array on an Agena vehicle.

The SERT II is scheduled to be launched for a 6-month space test in late 1969 or early 1970. The components that compose the flight thruster system are the thruster, the neutralizer, the propellant feed systems, and the power conditioning. Even though the major technology for these components was on hand, the extensive development necessary to produce a practical flight system resulted in many increases in performance and knowledge of these components.[2-6] The performance goals of the SERT II thruster system are: a design lifetime of 10,000 hr, a thruster power efficiency of 79%, and a thruster propellant efficiency of 85%. This paper and the two following companion papers[2,3] summarize the development program and the resulting typical performances of components. The present paper briefly describes the SERT II mission and auxiliary space experiments to be performed. The second paper[2] describes the thruster system and documents the performance of a flight prototype unit, and, the third[3] gives the results of hollow cathode durability tests and speculates on the theory of cathode operation.

SERT II Mission

The SERT II is to be launched by a Thorad/Agena booster into a 1000-km circular polar orbit (Fig. 1). The final space vehicle will be oriented as shown in Fig. 2. The second-stage Agena is part of the SERT II spacecraft. The 1.5-kw solar array will be unfolded from its position near the Agena rocket nozzle and furnish power to operate one of two complete thruster systems. The second thruster system, including its separate power conditioning, will be held in "stand by" should difficulty occur with the first system.

The two solar panels, which are each 1.5 by 5.8 m in size, will be oriented in the orbit plane, which has an initial inclination of 15° with the sun perpendicular. As the mission proceeds, the sun angle will change due to precession of the orbit and disturbing torques on the spacecraft. The launch window will be chosen such that no inclination angle greater than 15° occurs. Attitute control is maintained about two axes by gravity-gradient stabilization through the length of the Agena vehicle. Control moment gyros are necessary on the third axis to overcome disturbing torques too large to be controlled by crossed forces available from the other two axes. After 6 months the orbit will pass through shadow periods during which no electric power will be available to operate the thruster. The requirement of 6 months continuous sunlight allows two launch windows per year, one in the spring and one in the fall.

The two thruster systems including propellant tanks and power conditioning will be mounted to a 1.5-m-diam spacecraft and support structure which is bolted to the Agena vehicle (Fig. 2). The standard Agena shrouds will protect this area during launch and boost operation. Once in orbit the Agena vehicle will be positioned such that the electric thruster will be firing directly at the earth. The altitude of the orbit will be gradually raised during the mission by the electric thruster from an initial orbit of 1000 km to a final height of about 1120 km.§ The raising of the orbit is one direct measurement of thrust. An accelerometer may be used to obtain an additional direct thrust measurement. A thrust will be calculated from electrical measurements of ion beam current and voltage and compared with the direct thrust measurements.

Auxiliary Space Experiments

Plasma Potential Measurements

Two hot-wire emissive probes will be used to measure plasma potentials. One probe will be far removed from the spacecraft or the ion beam and will measure the potential of the ambient space plasma relative to the spacecraft. This potential should indicate any net charge acquired by the spacecraft. The second emissive probe will be located on a movable arm that can be rotated through the ion beam about 20 cm downstream of the thruster accelerator grid. Both of these hot-wire probe measurements can be used to measure an effectiveness of the neutralization system and to correct thrust calculations.

During the mission, upon ground command, the entire neutralizer system can be biased negatively or positively with respect to the spacecraft. Small bias values up to plus or minus 50 v will be applied to note any changes in the spacecraft or ion beam potential.

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§ The combination of low thrust/mass and radial thrust vector results in a relatively small orbit change in 6 months. Future electric spacecraft designed for orbit raising will have much greater mission performance than this experiment (Associate Editor).

Radio Frequency Interference

Although there is little doubt that broadcast commands can effectively be sent to control the SERT II mission, future deep space probes using an electric thruster may have enough rf noise generated by the ion beam that weak radio commands from earth may be lost in the beam noise. Therefore, an experiment planned for SERT II is to place an antenna near the thruster and in a plane perpendicular to the long axis of thruster. This antenna is coupled to a tuner network that will scan a range of radio frequencies and measure the strength of the noise present in the ion beam. Such measurements can be used to predict the strength of command signals that would be required for future electric propulsion missions at various distances from the earth.

Accelerometer

A novel type of electrostatic accelerometer, in which electrostatic forces suspend and restrain the inertial element, is being considered for direct measurement of thrust.[7] The displacement of a cylindrical element is sensed as a change in capacitance by "pickup" electrodes. This type of accelerometer is rugged enough to withstand launch vibrations and has been accurately tested at accelerations of 10^{-4} g's. The present design should be capable of sensing 1×10^{-6} g's with 3% accuracy, although ground testing at this acceleration level is difficult because of the 1-g background field. The expected acceleration of the SERT II spacecraft is 4×10^{-4} g.

Surface Contamination

Small solar panels will be located at various positions in the vicinity of the thruster, as shown in Fig. 3, but not directly in the ion beam. The output of these cells will be measured periodically and compared with the output of the main solar array. Thus any net changes due to ion thruster efflux (either ions, neutral propellant, or condensable sputtered atoms) can be measured. Calculations of efflux patterns indicate that negligible degrading of the solar cells should occur because the ion density and energy are too low to damage the cells except if placed in the direct beam.[8] The solar cell temperature (55°C) is high enough to prevent mercury condensation, and the amount of sputtered condensable material is zero upstream of the accelerator grid plane and diminishes approximately as a cosine law in the downstream direction. A negative experimental result (that is no change in the solar cell output) will serve to prove the compatibility of solar panels and mercury-bombardment thrusters for future solar-electric space missions.

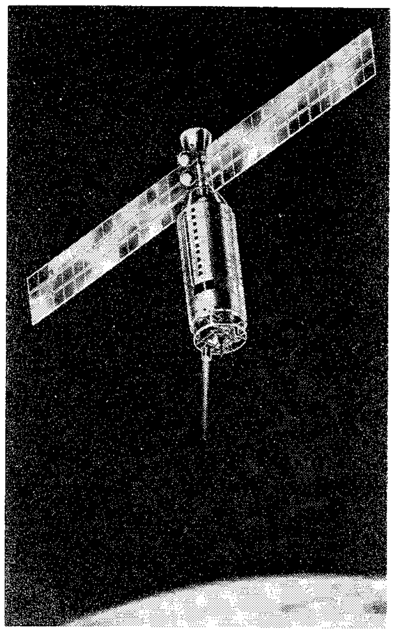

Fig. 2 SERT II spacecraft and solar cell array in orbit.

Surface Erosion

Existing experimental micrometeoroid flux data differ for near earth orbit locations by as much as four orders of magnitude. An experiment with a long duration in near earth orbit, such as SERT II, and a sensitive meteoroid transducer are needed to resolve the flux data spread. The transducer consists of a thin stainless-steel disk coated with a mirror finish of aluminum. The disk, thermally isolated in a box and pointed away from any spacecraft contaminant source, changes temperature as its exposed aluminum surface changes emissivity. The change in emissivity as a function of impinging meteoroid flux is known through ground calibrations[9] using a hypervelocity particle shock tube.

Hollow Cathode

The SERT II thruster system, shown in Fig. 4, incorporates a hollow cathode. Although the use of a plasma-bridge cathode for the main discharge of a mercury bombardment thruster was first accomplished as a part of the SERT II

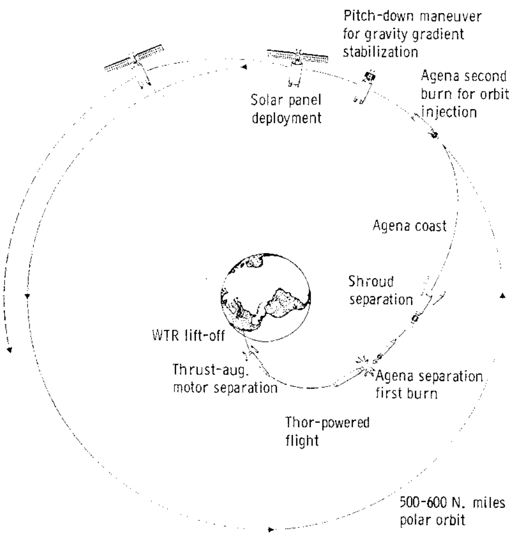

Fig. 1 Representation of SERT II flight sequence.

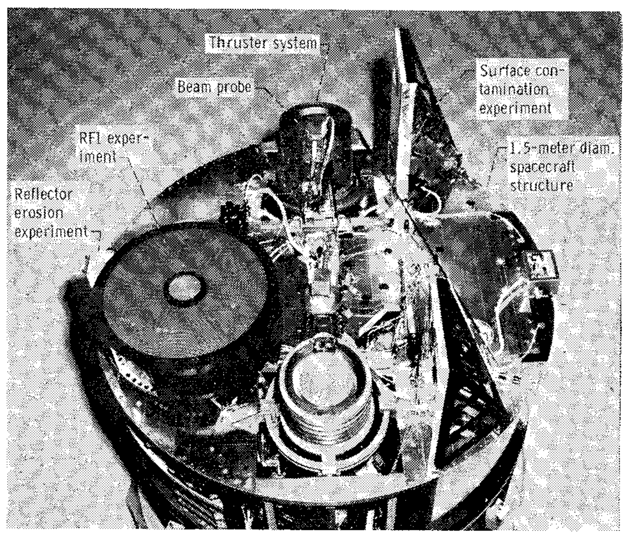

Fig. 3 SERT II spacecraft and experiments.

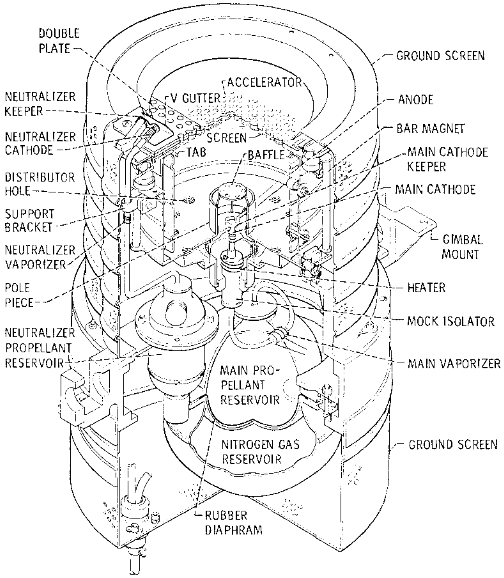

Fig. 4 SERT II thruster system, anode 15-cm diam.

program, some of the first data at the necessary emission levels of several amperes was obtained as part of a neutralizer development program.[10] This type of hollow cathode replaced the initial choice of an oxide magazine cathode,[11] because it can be ground-tested before launch, exposed to air for an indefinite period of time, and still perform the SERT II mission. The savings in cathode heating power (from 110 to 30 w), although considerable, was incidental to the replacement decision.

The steady-state power to heat the cathode tip is about 15 w. The total power charged against the hollow cathode includes the sum of the tip power, the keeper-electrode power, and propellant-line-heating power (necessary because of the high propellant feed pressure). The total cathode power is about 30 w during steady-state operation.

The many details of the development of the hollow cathode are presented elsewhere.[3,5,6] Briefly, when first tried in a thruster, the cathode produced extremely poor thruster performance (discharge power loss of 1000 ev/ion and maximum propellant utilization of 50%). Under continued development (including a discharge baffle[12]) the thruster performance was improved to approximately that obtained when using an oxide-magazine cathode, about 250 ev/ion at greater than 80% propellant utilization. After initial endurance tests of several hundred hours showed no significant erosion or performance deterioration, the hollow cathode was selected to replace the oxide magazine cathode.

Concluding Remarks

The basic SERT II thruster system design was frozen April 15, 1967. This and companion papers have presented the development history leading to this design, a description of the components, and the ground testing of thruster performance. A continuing effort is being made to check prototype and flight thruster systems through the launch date. Only small changes have been needed to solve minor problems as the flight readiness status is approached.

The improvements in technology and thruster performance, as a result of developing the SERT II thruster system, are substantial. The reductions obtained in major thruster power losses, such as in the discharge chamber power losses, should scale to larger thrusters. The fixed losses, such as those associated with the neutralizer, will become proportionately smaller because the same size neutralizer should serve a thruster much larger in size. Hence, for the same specific impulse, larger power thrusters or arrays of thrusters can be expected to have thruster efficiencies substantially greater than the SERT II thruster system.

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