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SERT II 1979-1981 Tests: Ion Thruster Performance and Durability

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The Space Electric Rocket Test-II (SERT II) spacecraft, launched in 1970 for a 1-yr mission as a test bed for an ion thruster system, continued to function as a working spacecraft through May 1981. As a result, an opportunity existed to obtain extensive thrust system test data over a period of more than a decade. Ion thruster testing continued until the last mercury propellant tank was empty in April 1981. This paper presents the thruster system efficiency, lifetime, and restart capability demonstrated over the last 2½ years of the flight period. The operation characteristics of thruster system components, such as hollow cathodes, electrical insulators, and propellant tanks, are also described.

Introduction

THE SERT II spacecraft was launched with a primary objective of demonstrating long-term operation of a space electric thruster system. Progress toward that objective was completed late in 1970 when each of two ion thruster systems onboard developed a high-voltage grid short. The spacecraft, though, was still functional and cathode restart/propellant system tests were performed in 1973-74. A spacecraft spinup maneuver in 1973 caused the grid short of one thruster to clear, and subsequent tests in 1974 showed normal operation of that thruster. Tests in the mid 1970s, however, were limited to brief test periods because the Earth's shadow eclipsed the spacecraft solar array during each orbit. In early 1979 through May 1981, the orbit became a continuous sunlight orbit again, continuous testing exhausted the ion thruster system propellant tanks, and no more thruster operation was possible. The spacecraft continued to function until its transmitters were turned off by ground command in June 1981.

This paper describes the performance of a 15-cm-diam mercury ion thruster system tested for electrical efficiency and lifetime in space, analyzes critical thruster component behavior, and gives design recommendations for future ion propulsion applications. The following extended mission objectives were accomplished: 1) clearing the high-voltage short from thruster system 2 and reestablishment of normal thruster operation and efficiency in 1979 and 1980 after ten years of space storage, 2) successfully demonstrating 300 restarts without difficulty, and 3) operating discharge chambers for nearly 10,000 h (14 months) in space.

A companion paper presents the results of operating the thruster in a plasma-thrust mode and the measurements of the ion beam plasma with various sources of neutralizer electrons.[1]

Ion Thruster Systems

Two identical mercury electron bombardment ion thruster systems were located on the SERT II spacecraft. Each thruster was capable of producing a 250-mA beam of mercury ions at an energy of 3000 V, giving a thrust of 28 mN at 4200-s specific impulse and using 850 W of power to the thruster. Throttle operation was possible at 200 and 85-mA beam current using the same beam voltage. Figure 1 is a cutaway drawing of a thruster and propellant tanks. Complete description of the thruster system and its preflight ground testing is available elsewhere.[2,3] The information below gives the operation of each thruster system applicable to the results of this paper.

Thruster system 1 (T/S-1) data were taken with the neutralizer discharge only lit, and with both the main and neutralizer discharges lit. The screen (V5) to accel grid (V6) short developed in 1970 remains, and prevented these supplies from turning on without current overload. A schematic wiring diagram of thruster power supplies is shown in Fig. 2 of the companion paper.[1]

Thruster system 2 (T/S-2) functioned as a normal thruster (no V5-V6 short). The beam current of T/S-2 was maintained at 85 ± 3 or 200 ± 3 mA by closed-loop control. Normal degradation of available solar array power after ten years, limited steady-state operation to 200-mA beam current. Operation at 250 mA was attempted several times, but running times were less than 1 min because the total power required just happened to equal the maximum power of the degraded solar array. Small control loop perturbations caused power demands slightly greater than maximum power with

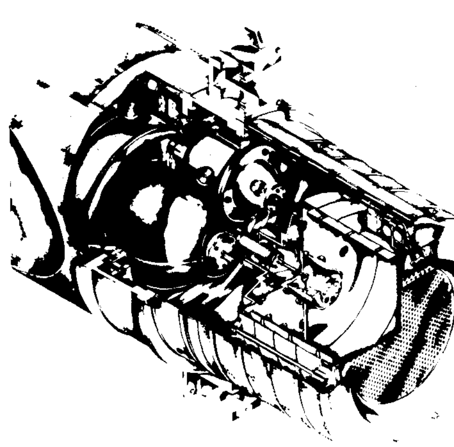

Fig. 1 SERT II ion thruster.

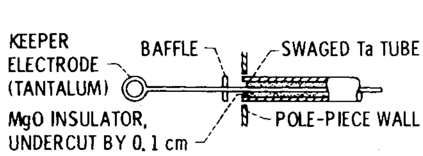

Fig. 2 Main keeper assembly.

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Table 1 Summary of operating hours for SERT II thrusters

Year(s)	Thruster 1, hr				
	250 mA beam	200 mA beam	85 mA beam	Discharge only w/neut	Discharge only w/o neut
1969	63	8	8	4	-----
1970	3794	3	3	33	-----
1971–72	----	--	---	42	-----
1973–78	----	--	---	36	-----
1979–80	----	--	---	[a]1003	[b]2940
Total beam	3879				
Total neut.	4997				
Total discharge	[b]7937				

Year(s)	Thruster 2, hr				
	250 mA beam	200 mA beam	85 mA beam	Discharge only w/neut	Discharge only w/o neut
1969	43	8	8	4	-----
1970	2017	2	2	8	-----
1971–72	----	--	---	20	-----
1973–78	0	0	1	164	-----
1979–80	0.01	58	744	[a]1005	[c]5948
Total beam	2877				
Total neut.	4078				
Total discharge	[c]10,026				

[a]Neutralizer Hg tank empty at end of test hours. [b]Main-1 Hg tank empty, Dec. 1, 1980. [c]Main-2 Hg tank empty, April 19, 1981.

Table 2 High-voltage shorts on SERT II flight thrusters

Short No.	T/S-1 Beam, h	T/S-1 Year	T/S-2 Beam, h	T/S-2 Year
1	2385	1970	2017	1970
2	3794	1970	2561	1979
3			2626	1979
4			2877[a]	1980

[a]This short between 2V5 and thruster (spacecraft) ground, all other shorts between V5 and V6.

subsequent drop in array voltage to a point of automatic system shutdown. Detailed electrical thruster data may be found in Table 2 of Ref. 4.

Ion Beam Thrusting

A summary of operating hours for both thruster systems is presented in Table 1. T/S-2 was operated 42 times in 1979-81 for a total of 606 h at 85-mA beam and 58 h at 200-mA beam. T/S-2 thrust data taken in 1980 agreed well with those measured in 1979 and in 1970. The thrust values were measured from the resulting changes in spacecraft spin rate data. Details of this type of thrust measurement may be found in the appendix of Ref. 5.

There has been no change in thruster performance over the 11½-yr period of space operation that included 4078 h of operation (2877 with beam) and 261 restarts with T/S-2. An additional 110 h of beam current were logged after the 4078-h (next tank empty) point during distant-neutralizing tests described later. The main discharge of T/S-2 ran an additional 5948 h after the 4078-h point before the main tank was empty in April 1981. The total operating time was 10,026 h.

High-Voltage Shorts

Table 2 summarizes high-voltage shorts that have occurred on SERT II flight thrusters. From telemetry analysis of the V5 and V6 supplies it was possible to conclude that all shorts except No. 4 on T/S-2 were between V5 and V6, i.e., the screen and accelerator grid. Also concluded was that the value of the short was below 10 kΩ.

Short 1, T/S-1, was removed by a single shutdown and normal preheat period. Short 2 still exists in T/S-1 after 300 thermal recycles and 20 attempts to sustain high voltage. T/S-2, short 1, occurred in 1970 and was not cleared until 1974 following a spacecraft spin maneuver which placed the thruster in a small artificial gravity of about 0.01g. T/S-2, short 2, occurred during a test in which high voltage was applied with no discharge to measure grid insulator leakage. No leakage was found, but a high-voltage grid short developed, possibly from a higher-voltage stress on the grids. Short 2 was cleared by a cold restart application of high voltage after several hot restarts failed. T/S-2, short 3, occurred during another test of high-voltage application with no discharge. In this case the main discharge lighted, produced a beam current for 2-15 s, and then T/S-2 developed a V5-V6 short. Several hot and cold restarts of high voltage were tried before the short finally cleared following a long heated period. The last short of T/S-2 was a short of <10 kΩ between V5 and thruster ground. This short occurred after 1½-h steady-state operation at 85-mA beam with neutralization from neut-1/main-1, and also one day after operating T/S-2 for 53 min with no neutralizer source. Short 4 still remains despite 16 thermal recycling efforts to clear it.

The V5-V6 shorts were believed caused by neutralizer discharge ions striking the accel grid, causing erosion and web fragments to be produced. The web fragments would be electrostatically drawn to and short to the screen grid. The V5-ground-short location was not as apparent. The most probable area was a plate attached to the thruster ground screen, underneath the neutralizer cathode. This plate was in close proximity (~ 3 mm) to a surface of the thruster body (V5 potential).

Tests were made on T/S-2 in late 1979 in which high voltage was applied to the grids before the thruster discharges had lighted. In all cases, there was no measurable leakage in either I5 (<1.5 mA) or I6 (<0.1 mA) for V5 of 4020 V and V6 of -1650 V. This result of no measurable insulator degradation agrees with results from thruster life tests in vacuum chambers and confirms the insulator design. The insulators were Al_2O_3 balls with double-cup, line-of-sight, shields. The grid insulators of T/S-1, of course, could not be tested because of the high-voltage grid short that was present.

Durability Testing

Main Cathodes

The main cathode of T/S-1 operated for 7937 h and 240 restarts before the main propellant tank became empty. During this period there was no noticeable change in cathode performance. Cathode starting time varied from 0.3 to 9.5 min depending on initial system temperature. (Design specification was to light in 90 min of preheat.) Furthermore, as shown in Table 5 of Ref. 4, no change occurred in cathode tip heater resistance over the operating period. The main cathode of T/S-2 accumulated 10,026 h of running with 300 restarts without demonstrating any detectable change in its main and keeper discharges, or in its tip heater resistance.

The successful operation of the main discharge chamber for over 10,000 h in space has a bearing on the concerns evidenced in Ref. 6. Reference 6 addressed sputtering in mercury bombardment thrusters and the effect of reduced sputtering resulting from adsorbed vacuum tank gases. Reference 6 presented sputtering data over a range of vacuum tank pressures, including operation down to the 10^{-7} Torr range, which is difficult to obtain. For lower pressures such as exist in space, the authors of Ref. 6 had no data. They concluded,

Table 3 Summary of neutralizer cathode totals

System	Total, h	Restarts	Restart time, min
T/S-1	4997	222	3-7
T/S-2	4078	300	3-7

however, that no significant increase in sputtering would occur in extrapolating sputtering rate data to the low pressures of space. The absence of any significant change in the SERT II discharge chamber operation is evidence that no significant sputter erosion occurred in space operation of discharge chambers; thus this absence of change supports the extrapolation and conclusions made in Ref. 6.

Neutralizer Cathodes

The performance of the neutralizer cathodes equals that of the main cathodes; i.e., there was no noticeable change in discharge characteristics, restarting was consistent, and tip heater resistance was unchanged over the full test time. Table 3 summarizes the totals for the neutralizer cathodes.

Main Keeper Insulator

Figure 2 shows some of the construction details of the main cathode keeper electrode design. This design has proven satisfactory for 7937 h of operation and 300 restarts over an 11-yr period in space. Features of the design included a solid tantalum electrode and a swaged insulator support tube. The electrode was made of a refractory metal to resist melting. The normal operating discharge heat load was only several watts, but during starting transition the heat load may be 35 W. The swaged insulator was undercut to improve resistance to surface contamination, and a line-of-sight baffle was located, as shown in Fig. 2, to reduce the flux of sputtered metal on the exposed end insulator.

The resistance of this design to contamination can be inferred by the data of Fig. 10 of Ref. 4, which indicated 2-mA insulator leakage current at beginning of life and 10-mA leakage at end of testing.

The conclusions from leakage measurements follow: 1) no problem resulted from the small amount of leakage current that occurred; 2) this current increased with time, probably owing to surface buildup at the exposed end of the swaged insulator; 3) future insulator designs should be at least equivalent to that used; and 4) the roll-off design of the keeper voltage supply should provide adequate starting voltage for end-of-life leakage currents as high as 20 mA.

Neutralizer Keeper Insulator

Figure 3 shows some of the construction detail of the neutralizer keeper electrode design. The sketch is to scale with the Al_2O_3 insulator being 0.6 cm across. Insulator shielding design included the keeper electrode itself and a spacer washer between the keeper and Al_2O_3 block to minimize contact between the two. (In retrospect, a spacer washer probably should have been used under the screw head, also.) The electrode was tantalum to avoid melting for the same reasons noted above for the main cathode keeper.

Current leakage data (shown in detail in Fig. 11 of Ref. 4) varied from a few milliamperes at beginning of life to 10 and 30 mA at end of testing for T/S-2 and T/S-1, respectively. The increase of this current with time probably was caused by a buildup of condensed sputtered metal on the Al_2O_3 insulator surfaces. This buildup may have been sputtered accelerator grid material (molybdenum).

The conclusions reached from the leakage data follow: 1) The neutralizer keeper insulator shielding design was adequate, but not as good as that for the main keeper. 2) The insulator surface buildup resulted in leakage currents of 10-30 mA during neutralizer cathode lighting attempts, and a resulting fall-off of keeper starting voltages from 420 V to a range of 250-300 V. 3) Even at the reduced keeper starting voltage, however, the neutralizer continued to relight upon command. 4) More attention should be given to future insulator shielding designs to prevent sputtered metal from reaching critical insulator areas. 5) The design keeper voltage "roll-off" should provide for a leakage current margin of 10-20 mA at the minimum starting keeper voltage.

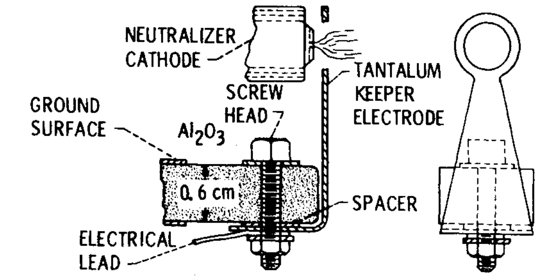

Fig. 3 Neutralizer keeper assembly.

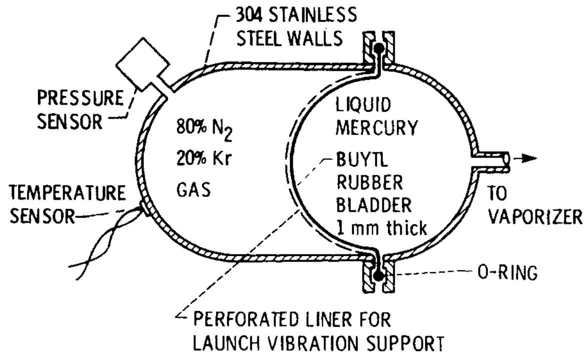

Fig. 4 Schematic of neutralizer propellant tank construction design.

Neutralizer Propellant Tanks

The neutralizer propellant tanks of each thruster system were operated until they became empty of mercury. This occurred on day 203, 1980 for thruster system 1 and day 122, 1980 for thruster system 2. The purpose of this section is to present detailed performance data of the neutralizer propellant tanks and compare this performance with design predictions.

A schematic cross section of a neutralizer tank is shown in Fig. 4. The tank consisted of two nearly equal volumes. One contained liquid mercury and the other contained a pressurizing gas (80% N_2, 20% Kr). (Krypton gas was added as a tracer gas for leak detection ground tests.) The two volumes were separated by a butyl-rubber bladder which terminated in an "O-ring" shape. This O-ring formed a seal between the two halves of the tank. A pressure transducer was mounted on the gas volume and a temperature sensor was located externally on the essentially isothermal tank.

As mercury flows out of the tank, the gas volume increased and the pressure decreased. A plot of this pressure decrease with operating time is shown in Fig. 5 for neutralizer tank 1. (The pressure values were at discrete levels owing to the telemetry count system used.) The use of the pressure change with time and the ideal gas law permitted a calculation of the change in gas volume. The gas volume change was equal to the change of the mercury volume, and the mercury flow rate thus could be calculated. A second way to calculate the mercury flow rate was to integrate all flow periods after the tank is empty and divide into the total mercury loaded. Table 6 of Ref. 4 gives major flow periods for each neutralizer tank and respective data about flow rates. The flow rates based on

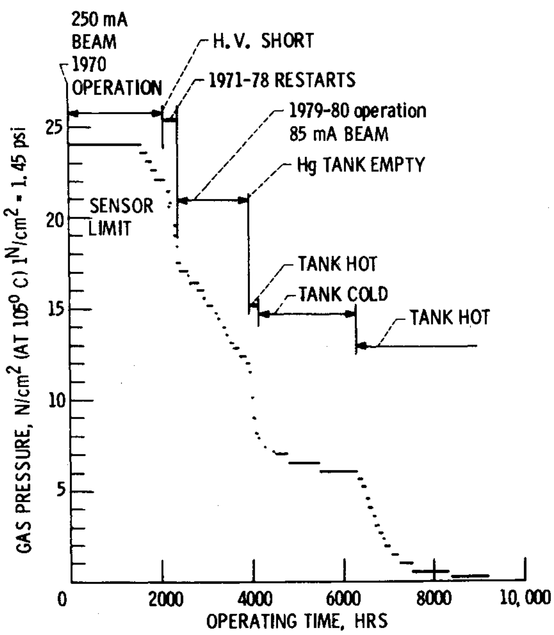

Fig. 5 History of neutralizer tank-1 pressure.

the integrated total flow are in good agreement with ground-based flow rate taken with "flow tubes" before launch. Flow rates earlier estimated,[1] based on the ideal gas law, were 10-20% higher than the integrated total flow rates. An error in the ideal gas law calculations was caused by not accounting for diffusion or leakage loss of pressuring gas through the rubber bladder.

A complete analysis of gas diffusion was somewhat involved, but will be presented here because of possible impact on future propellant system design. There were two gas diffusion paths of interest. One was straight through the bladder from the gas side to the mercury side. This path was of no importance in the SERT II design, but may be significant in other designs. Any gas diffusing to the mercury side will become trapped, setting up a near equal back pressure and reducing the net diffusion rate. The only way for that gas to escape would be past the O-ring seal or past the natural mercury seal formed by liquid mercury in the tube between the tank and the vaporizer. Neither of these escape paths were probable in the SERT II design. Future designs, however, should confirm that a natural mercury seal exists; that is, there is no surface roughness or grooves on the tube inside wall that would allow gas to slip past the liquid seal. The second diffusion path of interest was the following: gas entered the rubber bladder and traveled sideways in the rubber until it reached the O-ring. It continued through the rubber of the O-ring and escaped past the unsealed outer joint between tank-half flanges. Whereas this path was very tortuous, it did constitute the major path of pressurizing gas loss. The rate of pressurizing gas lost during storage periods of 1971-1978 was able to be measured (see Fig. 5). The measured loss rates agreed within 20% with the calculated diffusion loss rate through the O-ring material.

During the storage periods the tank temperature was cooler (35-60°C) than when the thruster was operating (98-105°C). Literature values of butyl rubber diffusion rates[7] were used to extrapolate diffusion low rates during thruster operation at high temperatures. The diffusion rate was 25 times greater at 105 than at 35°C. The higher temperature diffusion rate value was used to correct the pressure decrease in the ideal gas low flow calculations. When this was done, the flow rates thus calculated were within experimental error of those flow rates calculated from integrated total flow data.

After the mercury was empty, the gas reservoir pressure dropped rapidly. This drop in pressure was caused by gas diffusion directly through the butyl-rubber bladder. The gas that diffused across the bladder was now more free to flow through the empty mercury propellant line, through the porous tungsten vaporizer plug and out through the neutralizer cathode orifice to space. The gas pressure decay curve was used to calculate a gas flow rate. This flow rate agreed with a flow rate calculated from butyl rubber diffusion rates, rubber thickness, and surface area.

The following are conclusions based on neutralizer tank pressure data.

1) Neutralizer flow rate performance in space was the same as for ground vacuum chamber thruster operation over all conditions tested.

2) The design of a butyl-rubber bladder blow-down tank was validated for a 10-yr period with the following constraints: thermally design the tank to cool temperatures (20°C) where gas diffusion through the bladder is low enough for mission life, or depend on the liquid mercury to seal or trap the diffused gas.

3) Under normal flow operation (some mercury remaining in tank) the loss of pressurizing gas was negligible. The total of any leakage plus diffusion measured over an 8-yr storage period was only 1.7×10^{-4} cm^3/h (STP). A small excess (10-15%) of pressuring gas could provide for this loss, even over a 10-yr system life.

4) There were no known materials compatibility problems. The neutralizer tank provided pressurized liquid mercury to the vaporizer for the full life of the tank capacity. There were no known leaks of mercury and the tank capacity was exhausted when anticipated. The leakage of pressurizing gas was less than specified for typical gas-tight construction and had no impact on the flow life of the system.

Main Propellant Tanks

The design of the main propellant tank followed the same philosophies as for the neutralizer propellant tank, but the main tank was constructed in a larger size to hold 14 kg of mercury. Because the main tank was at high voltage, no pressure transducer was used to measure change of pressurizing gas with time. Hence there was no way to estimate flow rate as the mission progressed. Once the main tank is empty, however, the estimated flow rates could be integrated and compared with the total used to give a measure of confirmation of the actual flow rates.

The main propellant tank of T/S-1 became empty on December 1, 1980. Tables in Ref. 4 summarize the operating hours and estimated flows for main tank-1. The estimated integrated total flow was 13,830 g which was only 1.6% less than the useful mercury, 14,050 g, loaded into the tank (50 g of additional mercury was loaded into the line between the tank and vaporizer and was considered to be unavailable for use).

The main propellant tank of T/S-2 became empty on April 19, 1981 after a total of 10,096 h of flow. Table 4(b) of Ref. 8 summarizes the operating hours and estimated flows for main tank-2. The estimated integrated total flow was 14,103 g which was only 0.4% more than the useful mercury, 14,050 g, loaded into the tank.

The (−)1.6% and (+)0.4% differences for tank-1 and tank-2, respectively, constitute excellent confirmation that mercury bombardment thrusters operate at essentially the same propellant utilization in space as measured in laboratory vacuum tanks on the ground.

Vaporizers

Because the vaporizers caused absolutely no trouble, their performance tended to be overlooked. Vaporizers were made of porous tungsten (2.4-μ bore diameter, 70 and 76% dense) electron beam welded into tantalum housings. Vaporizer design and flow information is documented elsewhere.[9] The SERT II flight vaporizers withstood flight qualification and launch vibration, 11 yr in space, 5000-10,000 h of operation, and 300 restart cycles, all without failure. The dynamic head (distance from tank to vaporizer) was kept small by design, so that no valve was necessary to withstand launch vibration pressures that might force liquid mercury through vaporizer pores. The vaporizer operating power with time (see Table 2 or 5 in Ref. 4) was nearly constant, and the small differences that were measured were probably a result of a varying thermal (sun-angle) environment more than anything else.

Conclusions

The testing of SERT II ion thrusters proceeded for over 11 yr before the mercury propellant tanks were finally exhausted. During this period thruster systems 1 and 2 accumulated 8000 and 10,000 h of operation, respectively. The high-voltage grid short, which stopped high-voltage beam operation in 1970, was cleared for thruster system 2, and 800 h of beam operation were logged in 1979-1981. The thrust and electrical performance of this thruster after 11 yr in space was unchanged. Restartability was demonstrated over 300 times and was never a problem. Thruster system component life was excellent with no change in the operation of hollow cathodes, vaporizers, discharge chambers, electrical insulators, or propellant tanks. The sum total of all these results is confidence that mercury ion bombardment thruster systems can be built and operated in space on a routine basis with the same lifetime and performance as measured in ground testing.

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