NASA TECHNICAL MEMORANDUM

NASA TM X-71651

NASA TM X-71651

(NASA-TM-X-71651) SERT 2 THRUSTER SPACE RESTART, 1974 (NASA) 16 p HC $3.25 CSCL 21H

N75-15731

G3/20 Unclas 07762

SERT II THRUSTER SPACE RESTART - 1974

by William R. Kerslake and Robert C. Finke
Lewis Research Center
Cleveland, Ohio 44135

TECHNICAL PAPER to be presented at Eleventh Electric Propulsion Conference sponsored by American Institute of Aeronautics and Astronautics
New Orleans, Louisiana, March 19-21, 1975

SERT II THRUSTER SPACE RESTART - 1974

William R. Kerslake and Robert C. Finke
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio

E-8229

Abstract

The results of testing the flight thrusters on the SERT II spacecraft during the 1974 test period are presented. The most notable result was the clearing of the high voltage short from thruster 2 and the successful stable operation of its ion beam. Test periods were limited to 70 minutes or less by Earth eclipse of the spacecraft solar array and by ground station coverage limitations. Thruster 2 was restarted 26 times with an ion beam produced 21 times. The high voltage short remains in thruster 1, but the cathodes were restarted 12 times to demonstrate continued restart capability. The propellant feed systems, power processors, and spacecraft ancillary equipment were demonstrated to be functional after $4\frac{1}{2}$ years in space. In addition to the thruster tests, a neutralizer cathode was operated separately to demonstrate that the potential level of a spacecraft could be controlled by the neutralizer alone.

Introduction

The SERT II spacecraft was launched in February 1970 with a goal of demonstrating long-term operation of an ion thruster in space. The spacecraft contained two 15-cm diameter mercury electron bombardment ion thrusters designed to operate at a nominal one kilowatt power level. In 1970 thruster 1 was operated for $5\frac{1}{2}$ months and then thruster 2 was operated for 3 months.(1) In each case, thruster operation was terminated by a high-voltage short. Analysis of data and comparison with ground life tests indicated that the short was due to an eroded web of the accelerator grid which was lodged between the grids. Since ground tests indicated that such an eroded web would be very lightly spot welded by the thruster power processor, a series of thruster turn-on tests were conducted in 1971 in an attempt to clear the short. These tests were unsuccessful and the spacecraft was placed in a storage mode.

By 1973 proposed electric propulsion missions included a need to restart thruster many times. Therefore, the stored SERT II spacecraft was activated to demonstrate both multiple restart capability and the integrity of thruster components, propellant feed system, power processor, and other spacecraft ancillary equipment after long-term space storage. Although the original SERT II spacecraft and thrusters were not designed for automatic cathode restarting, it was possible to manually command both the ignition of the cathodes and the subsequent turnoff. Such procedures were limited to real time while the spacecraft passed over a ground tracking station. During 1973, 112 successful restarts of each thruster were so demonstrated.(2) The 1973 test program ended, based on priorities for the ground-support equipment.

The 1970 launch, initial sun-synchronous, polar orbit of the SERT II spacecraft had precessed such that in 1973 the sun angle was oblique and only marginal power was available to operate the cathodes. Inadequate spacecraft power was predicted for 1974. Therefore, at the end of the 1973 test program a new spacecraft orientation was proposed(3) and executed to give a more direct sun angle and hence more spacecraft power for testing in subsequent years. (The new orientation is described in a following section.) In August 1974 the SERT II spacecraft was again reactivated for thruster testing. The results of these 1974 tests, which include clearing of the high-voltage short from thruster 2, return to normal operation of thruster 2, multiple restarts of both thrusters, and electrical potential control of the spacecraft by the neutralizer cathode are presented in this paper.

SERT II Spacecraft, Orbit, and Orientation

The SERT II spacecraft (Fig. 1) was launched into a polar, sun-synchronous orbit. The spacecraft was gravity-gradient stabilized with the ion thrusters pointed towards Earth. The solar array panels were in the plane of the orbit and directly (zero angle of incidence) faced the sun. The precession of the polar orbit is about 341 degrees per year, so that the sun angle of incidence on the solar array changes by approximately 19 degrees per year. In September 1973 the incident sun angle was estimated(4) to be 65 degrees. Thus, in addition to a reduced solar flux on the arrays, part of each orbit is in shadow. Within one more year at the given precession rate the SERT II spacecraft would have lost power and experiment capability.

In September 1973, the following spacecraft maneuvers were performed using the cold-gas backup attitude control system. First, the spacecraft pitch axis (Fig. 1) was misaligned from normal to the orbit plane to a direction nearly pointing towards the sun. Then the spacecraft was spin stabilized about the pitch axis to maintain this pointing direction. Thus, the sun angle was returned to a smaller angle of incidence giving more solar power, and once in each future year,(3) the pitch axis will again point in a similar direction towards the sun giving maximum solar power. The period of useful power was predicted(3) to be about three months, centered about September of each following year.

In August 1974 the SERT II spacecraft was again commanded on to perform thruster cathode restart experiments. At this time it was determined that there was adequate power available for experiments, but that this power was cyclic over a 3-week period. The spin rate about the pitch axis was 40 revolutions per hour and was not sufficient to hold a fixed pointing direction for the pitch axis. A combination of perturbing forces resulted in the pitch axis describing a coning angle.(4) This coning angle was much like the wobble of a slowly spinning top. The period of the wobble was 23 days and the solar angle of incidence varied continuously during the period.

Maximum solar power (about 750 W) was

achieved when the pitch axis was most closely aligned with the sun. Sufficient power was available for experimentation approximately 5 days on either side of the maximum. All thruster experimentation reported herein was conducted during three such 10-day periods in late August, mid-September, and early October of 1974. The ground tracking facilities were not available past October 1974, so the spacecraft was again deactivated.

Command and operation of the SERT II spacecraft in its present spinning mode is possible until late 1975 or 1976. By that time the continued orbit precession (341^o/yr) will result in the orbital-plane-solar angle of incidence passing through 90 degrees (parallel to orbit plane) and the solar incidence will be on the other side of the orbit plane. It may be possible to then despin the spacecraft and realign the pitch axis to be normal to the orbit plane with the solar array facing the sun. Sufficient power would be available to operate a thruster, but part of the orbit will be in shadow. By late 1978, however, the orbit should be free of shadow which would allow continuous thruster operation. It may be possible to check the thruster operation once every year until late 1978 and then have a period for continuous thruster operation in 1979.

SERT II Spacecraft Apparatus

Figure 2 is a photograph of the payload section of the SERT II flight spacecraft installed in a vacuum tank for flight acceptance testing. Thruster 1 is to the right in the photograph and thruster 2 to the left. The performance history of each thruster may be found in Reference 1 and Figure 3 shows thruster details in a cutaway drawing. Each thruster shown in Figure 2 has a hot-wire beam probe and probe actuating box attached to it, on the side facing inward to the center of the spacecraft.[5] Figure 4 shows the circle traveled by the beam probe tip during a single 1-minute sweep. Each hot wire probe was operational and returned data on the electrical potential of the plasma in which it was immersed. The probe arm on thruster 1, however, was jammed in its start position and was not swept in the present set of tests. The probe on thruster 2 functioned normally. A space probe, mounted on a 1.5-meter long boom, used a hot wire filament that turned continuously. Its design life was one year and it burned out in 1971. The results of the 1970 SERT II probe measurements may be found in reference 5. During 1974 the hot-wire probes of thrusters 1 and 2 were operated at various times with and without a thruster on and with and without neutralizer bias.

Other SERT II experiments were reactivated in 1974 with the following results: the RFI experiment continues inoperative; the reflector erosion experiment (REX) continues to show slowly decreasing temperature with time, but an accurate analysis of the data is difficult because of thermal lag of the sensor and the present spinning spacecraft configuration; the miniature accelerometer (MESA) experiment is impossible to check because the spinning spacecraft causes an acceleration beyond the maximum range of accelerometer operation; the contamination sensor and other spacecraft surface thermistors continue to give data and the results are the subject of a companion paper.[6]

Results of 1974 Testing

The SERT II spacecraft was located and activated on August 15, 1974. On August 19 both thruster discharges were turned on for the first time since August 27, 1973. Thruster testing continued until October 19. On October 31, 1974 the SERT II spacecraft was turned off. During this thruster testing period, thruster 1 cathodes were both restarted 12 times for a mission total of 156 restarts with 3889 hours of cathode operation total. Thruster 2 cathodes were both restarted 26 times for a mission total of 214 restarts and 2175 hours of cathode operation. An ion beam was produced by thruster 2 on 19 different occasions at beam currents of 0.068 to 0.227 amp. The beam on time varied from a few seconds to 40 minutes and the total beam on time for 1974 was 128 minutes. The hot-wire plasma potential measuring probes were turned on at various times to determine electrical potential correlations between conditions of quiet spacecraft (no discharges on), neutralizer cathode operating, and ion thruster operating. At appropriate times a bias voltage supply was activated between the neutralizer cathode and the spacecraft ground.

The results of these tests are presented, compared and discussed in the following sections.

High-Voltage Short Clearing

The first operation of thruster 2 in 1974 was to verify the restart of its hollow cathodes and no high voltage turn-on was attempted. The cathodes restarted and operated normally for about 10 minutes (limit of pass). The second operation of thruster 2 on the following day was also a normal start of the cathodes, followed by high-voltage turn on. At the first application of high voltage, there was no overload and high voltage was maintained for about 0.2 minutes. The high voltage then tripped off and reset automatically about 5 to 6 times in the next two minutes. Beam current was indicated for periods of 0.1 to 0.5 minutes between high voltage trip offs. The test was concluded at the end of these two minutes by the time limit of the ground station pass. On the following day another thruster 2 restart was attempted and 4.5 minutes of stable beam current was indicated before the pass ended. There was one recycle of the high voltage during the 4.5 minutes of stable beam current. On 16 subsequent tests, high voltage remained on (except for occasional 0.1 sec recycles) until shut down by ground command or by undervoltage (solar array power overload) to the power processor. The tests were not long enough to give a reliable indication of thruster arcing. Some tests had an arc or recycle once every 2 or 3 minutes. Other tests had no arcs in the entire (up to 40 min) thruster test duration.

It is theorized that the clearing of the high voltage short was indirectly due to the new, spinning mode of the spacecraft. The direct force on the short-causing eroded web was only 0.001 "g" ("g" is force of Earth's gravity) due to the spin and this force was probably too low to break away a lightly held eroded web. This 0.001 "g" force is, however, 2000 times greater than the very slight gravity gradient force normally existing prior to September 1973. Thus prior to September 1973, if the eroded web ever became free, it would have moved very slowly away from its position and when high voltage was cycled on, the electrostatic force could have pulled the web back into a shorted position. In the spinning spacecraft configuration of 1974, however, a loose web could travel about 5 mm in one second and may move away from the influence of the electrostatic field between the grids.

A possible model is as follows. The eroded web of the accelerator grid originally remained fixed in an unshorted position until it can undercut by erosion at its attached end. Just before the undercutting eroded through, electrostatic force pulled the weakened web to the screen grid, leaving the eroded web "hinged" to the accelerator grid and bent over to touch or short to the screen grid. In the spinning spacecraft configuration a steady 0.001 "g" and alternative electrostatic force would act on the "hinge" every high-voltage recycle. Perhaps the "hinge" was weakened by fatigue and eventually broke free. Now both ends of the eroded web were unattached and the web might fall free under the influence of the 0.001 "g" field.

An alternate theory to explain removal of the short is that in 1973 the shorted web was broken completely free of the accelerator grid. It was, however, held in a shorted position by a weak spot weld at either the screen grid or accelerator grid until the spacecraft was put in a spinning orientation. The spinning centripetal force, although weak, was 2000 times greater than prior gravitational forces and strong enough to pull free the weakly held web.

Thruster Ion Beam

Figure 5 shows a plot of various thruster parameters during a successful start and stable operation of thruster 2 at 0.083 amp beam current. The preheat command (time base zero) turned on cathode and propellant vaporizer heaters (I2, V3, V7). (All power supply numbers are listed in Table 1.) In 5 minutes, the neutralizer cathode was heated to starting temperature and sufficient mercury flow had been established to light that cathode. The light was indicated by a sharp drop in V8, the neutralizer cathode keeper (and starter) voltage. Neutralizer vaporizer heating continued which increased the flow and drove V8 to its set point of 28V. At 11 minutes V7, the neutralizer heater, began to cut back indicating control of V8.

At 12.6 minutes the main cathode lit as indicated by the increase of I4, the main discharge current. The main cathode heater, V3, was programmed to cut back at I4 levels above 0.5 amp because the cathode is primarily self-heated by the discharge once the main discharge is lighted. For the next four minutes the thruster was in a programmed control mode where the main discharge level was controlled at 1.6 amp by the closed-loop control of I2, the main flow vaporizer. This was done to set the main flow rate at a proper level such that when high voltage is applied the desired beam current level is correct. If the flow is too high, too much beam will be produced resulting in an overload of the high-voltage power supplies.

At 17 minutes the high voltage was turned on. The ion beam I5 overshot slightly in the first minute, but then reached its control set point of 83 mA. There was a slight cutback of I2 as control was established. The thruster then operated in a normal, controlled fashion until 57 minutes (not shown on Fig. 5). At this time the spacecraft passed into the Earth's shadow and the thruster power processor shut down due to a solar array under voltage.

Of the 19 times that an ion beam was produced, seven times were similar to Figure 5 and the stable on time of the beam was 3 to 40 minutes. During one test, a stable beam was produced for 7 minutes at 0.198 amp. In each case the thruster was shut down either due to power processor undervoltage as the solar array passed into the Earth's shadow or by command from the ground. Ground command shut down was used at the end of a ground station pass when the on-board tape recorder was not able to record data subsequent to loss of real-time data at the end of the pass coverage.

The remaining 12 times the ion beam was on for 1 minute or less. In each case, the ion beam current became too high, too much current was drawn from the solar array, and its voltage dropped below the undervoltage shut off value (48 V) of the power processor. This type of shutdown resulted from two factors. One factor was a small power margin between the thruster load and solar array output. (The maximum solar array power available on any day varied between 100 and 700 W in the 23-day cycle previously described, while the thruster load was 500 to 650 W.) The other factor was insufficient real time over a ground station to preheat, light the cathodes and stabilize the main vaporizer in the 15 minutes typically available. The SERT II thruster, when originally developed, planned for 1.5-hours preheat and 0.5-hour main discharge heating to stabilize the thruster and vaporizer temperatures. To attempt the quicker 1974 thruster restarts, the main vaporizer, I2, was turned on early and timed to be near the correct flow when the main cathodes lighted. If the time were guessed incorrectly, or the thruster not warm enough, excess mercury would be present due to either excessive vaporizer temperature or condensed mercury in propellant flow passages. The time response (about 1.5 min time constant) of the main vaporizer in these cases was too slow to prevent beam overshoot and consequent undervoltage shutdown of the power processor. In seven other tests, no high voltage was commanded on. These tests were either time-limited during attempts to produce an ion beam, or they were intended to study the effect of neutralizer bias on a spacecraft in the absence of an ion beam.

The SERT II mission was to endurance test thrusters and only a few thruster restarts were envisaged with turn on times of 1.5 to 2 hours acceptable. By proper thermal design, future thrusters can be built to start from cold storage in about 15 minutes. If the thruster body and propellant flow passages are warm, the starting time will be about 2 minutes. An instantaneous thrust, may, if desired, be produced by prestarting the thruster discharges and subsequent turn-on of the high-voltage supplies to produce thrust.

Table 1 compares the values of each of the flight-measured parameters for thruster 2 at three different times; early in the mission, 1970; cathode restart conditions in 1973; and thruster operation in 1974. There is good agreement within telemetry uncertainty between all parameters covering each mode of thruster operation over the nearly 5-year time period. Differences in high voltages (V5, V6, V10) result because the power supplies are unregulated with respect to solar array voltage input variation to the power processor. Controlled parameter set points (I5, I8) are a minor function of solar array voltage. The vaporizer heater powers (supplies 2 and 7) were higher in 1973 and 1974 than 1970 because of a colder thruster thermal environment in 1973 and 1974. The solar array voltage was lower in 1973 and 1974 primarily due to a higher solar array temperature which was caused by increased Earth albedo thermal flux to the array.

Cathode Restarts

Figure 6 chronologically shows the number of cathode restarts, storage time between restarts, and total hours of operation. The start up history prior to 1974 is presented in Reference 2. Since 1973, the thrusters, cathodes, and propellant supply systems were dormant for 326 days, used for approximately 2 months, and then turned off. When reactivated in August 1974, all systems were unaffected by the storage period and operated correctly. Due to the 1974 spacecraft orientation, however, the thruster thermal environment was both cooler and more variable than in 1973. This thermal environment led to a wider range of cathode starting times in 1974 than in 1973. For example, in 1973 the neutralizer cathode for thruster 2 normally required 5 to 6 minutes to start, while in 1974 the range was 3.9 to 7.2 minutes. Also the main cathode lighted in 6 to 13 minutes in 1973, but required 6 to 22 minutes in 1974. Table II presents representative times to ignite each of the four flight cathodes from preflight qualification tests to the present.

Figure 7 presents a correlation between the start time for neutralizer cathode 2 and the neutralizer propellant tank temperature. The thermistor on the neutralizer tank was the best flight measurement available to determine the thermal state of the thruster. When the thruster was cold, it required longer to light than when it was warm. As shown on Figure 7, the neutralizer cathode starting can be predicted from the neutralizer propellant tank temperature. The root-mean-square deviation in the data of Figure 7 is only 0.3 minute and the maximum deviation is 0.7 minute. In addition to the data of Figure 7, the neutralizer cathode was restarted six different times after prior operation in the same orbit; that is, it did not have time to cool down to its usual initial temperature. These six restarts were accomplished in less time than those shown on Figure 7: the range being 2.2 to 3.2 minutes. The conclusion of these observations is that the cathode starting time primarily is a function of the initial temperature of the thruster; and that to date, there has been no observable change in the starting ability of the neutralizer cathodes after several thousand hours operation and over two hundred restarts. Insufficient flight data exists to predict whether the increased starting time for the main cathode is due to the cooler thermal environment in the 1974 opportunity or to the number of multiple restarts performed. Once started, the equilibrium value of the main keeper voltage has remained unchanged since 1970, indicating little deterioration of the main cathode at steady-state running.

Thruster System Component Status

There is little or no apparent change in any of the heaters of the SERT II thrusters. Table 2 presents representative values of heater currents and voltages for the point of maximum heating time at full power. As can be seen from Table 2 and within flight data accuracy, all heaters continue to operate at constant values. The heater resistance, as indicated by the heater voltage divided by the current, remained constant over the 5-year period from preflight qualification tests to the present.

There is no apparent electrical leakage across any insulator in thruster 2. This includes the insulators between the accelerator grids (+2960 to -1480 V), between thruster to spacecraft (+2960 to 0 V), and between cathode keeper to cathode (+371 to 0 V).

The propellant feed system remains completely functional for each thruster. Mercury is supplied upon command from each of four vaporizers. In spite of a different ambient thermal environment than originally designed for, the vaporizers maintain flow control well within the limits of their heaters. The pressure of the nitrogen blow-down gas behind the rubber bladders of both the neutralizer propellant tanks remained constant without leaking during storage periods of over a year. The pressure in September 1973 and August 1974 was 12.2 and 14.4 N/cm^2 for tanks 1 and 2, respectively. No flight pressure transducers were installed on the main propellant tanks. At present, thrusters 1 and 2 have operated their vaporizers for 3889 and 2175 hours, respectively. The design value of the propellant tanks provides for 6000 hours of flow, so thruster 2 (presently operational) has propellant remaining for nearly 4000 hours more flow.

The power processors continue to function without malfunction or noticeable degradation after 5 years in space. Each individual power supply output current and voltage agrees with its original response curve as measured in preflight qualification testing. The output voltage of the high voltage supplies, V5 and V6, and the keeper supplies, V8 and V10, varied directly with the voltage input from the solar array. All thruster set points (3 levels of beam current, 3 values of main discharge voltage, and 2 neutralizer keeper voltages) are functional and vary slightly as predicted by original response to load or solar array input voltage. The high-voltage overload shutdown and automatic recycling continues to perform normally. All power supply telemetry outputs (also part of the power processor) remain operative.

Spacecraft and Plasma Potential Level Experiments

Future spacecraft designs may require both electrostatic cleanliness and control to perform particle energy experiments or electron emission to control spacecraft charging. The initial SERT II experiments[5] demonstrated that the spacecraft potential level could be controlled by controlling the potential bias level of the neutralizer of an operating ion thruster. The objective of tests in 1974 was to see if a neutralizer operating alone could similarly control the spacecraft potential level. An operating neutralizer should, in addition, be capable of emitting sufficient electrons needed in the control of spacecraft charging, although the level needed (about 1 mA), would be below the sensing capability of the existing spacecraft instrumentation. A second objective was added after thruster 2 became operational, namely, perform a neutralizer bias experiment with an operating thruster at a beam current level not previously tested, and compare the results with those taken at a higher beam current level in 1970.

Experiment description. For the following experiments and figure discussion, the potential of space is used as reference and is assumed to be zero. The space probe, which was designed to measure the difference between spacecraft and space plasma potential, had an open emitter wire. An alternate measurement was therefore made by use of either or both of the hot-wire beam probes. Reference 5 indicated that for a quiet spacecraft (no power to the thruster or neutralizer) the spacecraft potential measured by the space probe or the beam probe agreed within one telemetry count (2.5 V). The beam probe data taken in 1974 indicated identical quiet spacecraft potential from either probe, and a level equivalent to data reported in Reference 5. This was true for beam probe 1, which was jammed in its start position, and for beam probe 2 over its entire sweep range.

The neutralizer cathode of each thruster was electrically isolated through a bias supply. The low side of the bias supply was connected to spacecraft ground. The high side could be varied to make the neutralizer cathode a nominal ±25 and ±50 volts different than the spacecraft potential. For example, applying a +25 volt bias made the neutralizer cathode 25 volts positive with respect to spacecraft ground.

The SERT II spacecraft solar array was designed to be reconfigured such that the normally separate array portion dedicated to housekeeping power could be switched in parallel with the negative half of the main array. This reconfiguration was made for both the 1973 and 1974 test opportunities to supply additional housekeeping power at periods of low total power available. The main array configuration (±30 V nominal with center tap ground) remained in its original configuration. The effect, if any, of the reconfigured housekeeping solar array on spacecraft potential levels could not be deduced from available data. In 1974, the main solar array output voltage varied from 81 to 48 volts during thruster or neutralizer tests. The usual voltages were 70 to 60 for neutralizer only tests and 60 to 50 when the thruster operated.

Neutralizer only tests. Figure 8 is a plot of 1974 measurements of the SERT II spacecraft potential as a function of latitude. The quiet spacecraft floated at potential levels of near zero to -22 volts, depending on time of day (longitude, latitude, and perhaps local anomalies in space plasma.) When either thruster neutralizer was turned on, however, the spacecraft potential was held between zero and -5 volts irrespective of spacecraft position. The spacecraft potential was thus driven to near zero by an unbiased neutralizer cathode without the need of an ion beam to assist in coupling the neutralizer electrons to space.

The hollow cathode therefore is a candidate cathode to perform long-term, reliable spacecraft potential control. The SERT II neutralizer cathodes have operated in space for over 5 years with operating times of 3889 and 2175 hours. Recent ground tests of similar cathodes have accumulated 20 000 hours operating time on a single cathode without failure.(7)

In addition to the above tests, each thruster neutralizer was turned on and the neutralizer bias voltage was set at -45, -23, and 0 volts. (The positive bias supply voltage was unavailable due to a bias supply design feature which required a net neutralizer emission current for the supply to generate a positive voltage.) The result at zero bias was a spacecraft potential of 0 to -5 volts as shown on Figure 8. The result at negative bias indicated a small increase in spacecraft potential in the range +2 to +5 for -23 volt bias and +2 to +10 for -45 volt bias.

Next, beam probe 2 was used to measure plasma potential variations downstream of thruster 2 with only its neutralizer on. During these tests, the following results were obtained at various bias settings: for zero bias, the plasma potential near the thruster (±20° probe position) was +5 volts while the outside or wing area was at zero potential. For negative bias, the plasma potential near the thruster was near zero and the wing area was -2 to -10 volts and -10 to -15 volts for bias voltage of -23 and -35, respectively. The neutralizer emission was 0.080 and 0.325 amp for bias voltage -23 and -35, respectively. For one probe sweep only, data was taken with both neutralizer 2 and main discharge 2 on (but no H.V. extraction). In this case the wing area plasma dropped to -30 volts.

Thruster test with bias. Once, during the 1974 test opportunity, thruster 2 was turned on, stabilized at 0.083 amp beam current, and the neutralizer cathode was biased at nominal ±25 and ±50 volts. Beam probe 2 was swept through the beam at each bias and at zero bias. The results of beam plasma potentials (probe 2 - probe 1 voltage reading) were plotted on Figure 9. Also included on Figure 9 was a table of the actual bias voltages, resulting spacecraft voltages, and various coupling voltage differences. The beam probe was only in the ion beam approximately ±20 degrees about the thruster centerline which coincided with the midpoint of the probe sweep. The balance of the sweep measured the plasma potential in the fringe or wing area of the beam plasma.

The data shown on Figure 9 agree with the results of Reference 5, that is, the spacecraft can effectively be biased to negative levels, but positive levels are difficult because the neutralizer emission current flows to the thruster ground screen (a more convenient anode) rather than coupling to space plasma. The coupling voltage between the beam center and neutralizer varied little (37 to 44 V) between the +44- and -23-volt bias levels. The beam center potential monotonically decreased with decreasing bias voltage, but this decrease was much less than the bias voltage decrease. The beam center decreased only 27 volts while the bias decreased 90 volts.

The only significant difference in the data of Figure 9 and Reference 5 was in the plasma potential of the wings. Reference 5 tended to have a flatter wing profile without the presence of the negative wells shown in Figure 9 for negative bias sweeps. As no data was taken by Reference 5 for the 0.083-amp beam current level of Figure 9, and as the data of Figure 9 were only attempted once, no conclusive comparative statement can be made. While beam probe 2 was being swept to obtain the data of Figure 9, beam probe 1 was on and sensing spacecraft potential. During the negative bias sweeps there was no change in beam probe 1 reading, and during the zero and positive bias sweeps the probe voltage was constant to within ±2.5 volts (±1 count).

In summary, the SERT II spacecraft tended to float at 0 to 20 volts below space potential with no thruster or neutralizer on. With a thruster or neutralizer on the spacecraft could be maintained near zero potential or biased negatively. Positive bias of the spacecraft was ineffective because the neutralizer emission current was preferentially drawn to the spacecraft rather than space plasma. (The authors thank N. J. Stevens and V. W. Klinect of Lewis Research Center for their help in the attaining and preparation of the neutralizer bias data.)

Concluding Remarks

The SERT II spacecraft, designed for 1 year life, remains functional after 5 years in space. Opportunity exists therefore to check the long-term operational status of the on-board ion thruster components, power processors, and other spacecraft ancillary equipment. During the 1974 test opportunity reported in this paper, the high-voltage short was clear on thruster 2, and in addition to restarting cathodes and demonstrating the continued functioning of the propellant supply systems, complete operation of thruster 2 was demonstrated. Both power processors continued to function without fault after 5 years in space and 3889 and 2175 operating hours, respectively. In addition to the thruster tests, a neutralizer cathode was operated separately to demonstrate that the electric potential level of a spacecraft could be controlled by the neutralizer alone. Orbital mechanics predict a continuous sun-lighted orbit in late 1978. If spacecraft reorientation maneuvers are performed, it could be possible to operate thruster 2 continuously in a 1979 test opportunity with the propellant remaining in the thruster reservoirs.

References

1. Kerslake, W. R., Goldman, R. G. and Nieberding, W. C., "SERT II: Mission Thruster Performance and In-Flight Thrust Measurements," Journal of Spacecrafts and Rockets, Vol. 8, No. 3, March 1971, pp. 213-224.

2. Kerslake, W. R. and Finke, R. C., "SERT 2 Hollow Cathode Multiple Restarts in Space," Journal of Spacecraft and Rockets, Vol. 11, No. 9, Sept. 1974, pp. 669-671.

3. LeRoy, B. E., unpublished analysis, NASA Lewis Research Center, Cleveland, Ohio, 1973.

4. Ignaczak, L. R., Stevens, N. J. and LeRoy, B. E., "Performance of the SERT 2 Spacecraft after 4 and One-Half Years in Space," TM X-71632, 1974, NASA.

5. Jones, S. G., Staskus, J. U. and Byers, D. C., "Preliminary Results of SERT II Spacecraft Potential Measurements Using Hot Wire Emissive Probes," AIAA Paper 70-1127, Stanford, Calif., 1970.

6. Stevens, N. J., "Ion Thruster Efflux Deposition on Spacecraft Surfaces from SERT II Flight Data," AIAA Paper 75-356, New Orleans, 1975.

7. Wintucky, E. G., "A 20 000-Hour Endurance Test of a Structurally and Thermally Integrated 5-Cm-Diameter Ion Thruster Main Cathode," AIAA Paper 75-368, New Orleans, 1975.

Table 1. - Performance of Flight Thruster 2.

		Preheat			Propellant, no beam			30% beam		80% beam		Telemetry uncertainty (rss)
	Year	1970	1973	1974	1970	1973	1974	1970	1974	1970	1974	
	Day	2/11	6/1	10/7	2/11	6/14	8/23	2/11	9/10	2/11	9/11	
	Restart number	10	80	213	10	86	195	10	198	10	200	
Main vaporizer heater	V2,v	0	0	0	[g]1.63	[a]1.49	1.85	[g]1.63	1.70	1.70	1.85	±0.07
	I2,a	0	0	0	[g]1.41	[a]1.32	1.70	[g]1.51	1.77	1.70	1.95	±0.08
Main cathode heater	V3,v	16.0	15.6	15.6	8.7	9.5	9.1	7.9	8.7	8.3	8.7	±0.35
	I3,a	2.86	2.81	2.81	1.54	1.57	1.57	1.54	1.57	1.54	1.57	±0.05
Main discharge	V4,v	>50	>50	>50	39.9	39.7	40.4	42.2	42.4	41.5	41.4	±0.2
	I4,a	0	0	0	2.0	2.3	1.7	0.7	0.6	1.2	1.1	±0.05
Beam voltage	V5,v	0	0	0	0	0	0	[d]3490	[d]2960	[d]3160	[d]2630	±65
Beam current	I5,a	0	0	0	0	0	0	[d]0.088	[d]0.083	[d]0.203	[d]0.198	±0.005
Accelerator grid	V6,v	0	0	0	0	0	0	[d]-1730	[d]-1480	[d]-1640	[d]-1330	±50
	I6,ma	0	0	0	0	0	0	1.1	0.9	1.4	1.4	±0.1
Neutralizer heater	V7,v	[g]7.7	8.8	8.6	[g]7.7	[a]10.4	8.4	[g]6.6	8.1	[g]6.4	7.5	±0.25
	I7,a	[g]2.3	2.6	2.5	[g]2.3	[a]3.0	2.4	[g]2.0	2.3	[g]1.9	2.2	±0.05
Neutralizer keeper	V8,v	28.5	28.5	27.8	28.5	[a]32.3	28.5	27.8	27.8	[c]24.0	[c]27.8	±0.7
	I8,a	[d]0.226	[d]0.183	[d]0.191	[d]0.199	[a]0.175	[d]0.179	[d]0.215	[d]0.175	[d]0.206	[d]0.167	±0.004
Spacecraft voltage	v	-6	(f)	-3	-9	(f)	-4	-17	-8	-17	(f)	±2
Neutralizer emission	a	0	0	0	0	0	0	0.087	0.080	0.201	0.195	±0.006
Main cathode keeper	V10,v	[d]>416	[d]363	[d]371	12.3	9.9	10.8	20.4	20.0	13.9	13.1	±0.5
	I10,a	0	0	0	[b]0.289	[b]0.283	[b]0.282	[b]0.282	[b]0.272	[b]0.283	[b]0.272	[b]±0.003
Solar array voltage	v	70	62	65	68	61	60	68	59	63	52	±1.0

[a]Value changing in response to control signal.

[b]I10 value estimated from V10 value and power supply response characteristic curve.

[c]V8 values due to different set points.

[d]Difference in values due to different solar array voltages input to power processor.

[f]Data unavailable.

[g]Heater power lower due to higher thermal background.

Table 2. Representative heater values[c] and cathode starting times

Thruster	Start number	Date	Main vaporizer			Main cathode			Neutralizer cathode			Cathode start time		Total cathode on time,[d] hr	Neutralizer reservoir temperature, °C
			I2, A	V2, V	V2/I2, Ω	I3, A	V3, V	V3/I3, Ω	I7, A	V7, V	V7/I7, Ω	Neutralizer cathode, min	Main cathode, min		
1	1	12/9/69	2.80	(a)	(a)	2.80	>15	>5.3	2.78	(a)	(a)	$8.5^{+0.0}_{-0.0}$	$0.3^{+0.0}_{-0.0}$	----	(a)
	4	12/28/69	2.81	(a)	(a)	2.92	15.7	5.4	2.79	9.9	3.6	$6.2^{+0.0}_{-0.0}$	$0.4^{+0.0}_{-0.0}$	----	(a)
	5	2/14/70	2.81	2.74	0.98	2.88	15.7	5.5	2.90	10.3	3.6	$3.3^{+0.4}_{-0.6}$	$0.3^{+0.7}_{-0.0}$	0	(a)
	6	3/8/70	2.89	(a)	(a)	2.88	15.3	5.3	2.90	10.6	3.7	$4.2^{+0.1}_{-0.6}$	$0.3^{+0.7}_{-0.0}$	508	83
	7	5/21/70	(a)	2.67	(a)	2.88	15.3	5.3	2.90	10.8	3.7	$4.3^{+0.4}_{-0.6}$	$0.7^{+0.3}_{-0.1}$	2283	78
	8	7/23/70	2.89	2.67	0.93	2.88	15.3	5.3	2.90	10.6	3.7	$4.3^{+0.4}_{-0.6}$	$1.0^{+0.3}_{-0.3}$	3763	18
	14	10/26/70	2.89	2.60	.90	2.88	14.1	4.9	2.90	10.8	3.7	$4.2^{+0.4}_{-0.6}$	[b]$4.4^{+0.7}_{-0.3}$	3794	47
	20	2/11/71	2.89	2.67	.93	2.88	15.7	5.5	2.90	10.3	3.6	$4.2^{+0.0}_{-0.6}$	$0.3^{+0.7}_{-0.1}$	3835	83
	32	1/21/72	(a)	(a)	(a)	2.88	15.7	5.5	2.79	10.1	3.6	$6.2^{+0.0}_{-0.6}$	(a)	3868	29
	33	5/25/73	2.81	2.74	0.97	2.82	15.3	5.4	2.90	10.6	3.7	$6.6^{+0.4}_{-0.4}$	[b]$6.4^{+0.4}_{-0.3}$	3869	(a)
	53	6/20/73	2.89	2.67	.93	2.82	15.3	5.4	2.90	10.8	3.7	$5.8^{+0.1}_{-0.6}$	[b]$6.9^{+0.1}_{-0.6}$	3873	(a)
	82	7/16/73	2.89	2.74	.95	2.82	15.3	5.4	2.90	10.8	3.7	$6.0^{+0.0}_{-0.4}$	[b]$8.0^{+0.0}_{-0.4}$	3881	(a)
	144	9/27/73	2.89	2.88	.99	2.88	15.3	5.3	2.90	10.1	3.5	$5.4^{+0.0}_{-0.6}$	$.6.4^{+0.0}_{-0.1}$	3884	(a)
	145	8/19/74	2.89	2.74	.95	2.82	15.3	5.4	2.90	10.8	3.7	$6.3^{+0.4}_{-0.6}$	$7.4^{+0.4}_{-0.3}$	3885	(a)
	149	9/30/74	2.81	2.74	.98	2.82	15.7	5.6	2.90	10.6	3.7	$6.6^{+0.0}_{-0.4}$	$6.0^{+0.0}_{-0.5}$	3887	(a)
	156	10/9/74	2.81	2.74	.98	2.82	15.7	5.6	2.90	10.3	3.6	$6.8^{+0.0}_{-0.6}$	$9.5^{+0.0}_{-0.5}$	3889	(a)
2	1	11/29/69	2.89	(a)	(a)	2.78	>15	>5.4	2.94	(a)	(a)	$10.0^{+0.0}_{-0.0}$	$1.0^{+0.0}_{-0.0}$	----	(a)
	4	12/21/69	2.90	(a)	(a)	2.77	16.0	5.8	2.86	(a)	(a)	$6.3^{+0.0}_{-0.0}$	$1.0^{+0.0}_{-0.0}$	----	(a)
	10	2/11/70	2.88	2.77	0.96	2.86	16.0	5.6	2.97	10.2	3.4	$3.2^{+0.2}_{-0.6}$	$0.4^{+0.9}_{-0.3}$	0	97
	11	7/24/70	2.97	2.70	.91	2.86	16.0	5.6	2.97	10.2	3.4	$3.2^{+0.1}_{-0.4}$	$0.9^{+0.9}_{-0.4}$	38	97
	12	9/2/70	2.97	2.70	.91	2.81	15.6	5.6	2.97	10.4	3.5	$3.7^{+0.1}_{-0.6}$	$0.9^{+0.9}_{-0.4}$	934	65
	15	10/20/70	2.97	2.70	.91	2.81	15.6	5.6	2.97	10.4	3.5	$2.8^{+0.1}_{-0.6}$	$0.9^{+0.9}_{-0.4}$	2011	73
	33	10/30/70	2.97	2.70	.91	2.81	15.6	5.6	2.97	10.4	3.5	$3.1^{+0.0}_{-0.6}$	$0.5^{+0.9}_{-0.0}$	2053	73
	53	11/13/70	2.97	2.70	.91	2.81	15.6	5.6	2.97	10.4	3.5	$2.8^{+0.1}_{-0.6}$	$0.9^{+0.9}_{-0.4}$	2094	69
	67	2/26/71	2.97	2.70	.91	2.86	16.0	5.6	2.97	10.4	3.5	$2.7^{+0.1}_{-0.6}$	$0.4^{+0.9}_{-0.3}$	2126	115
	76	1/21/72	(a)	(a)	(a)	2.86	16.0	5.6	2.97	10.4	3.5	$5.3^{+0.3}_{-0.6}$	(a)	2149	33
	77	5/25/73	2.97	2.77	0.93	2.81	15.6	5.6	2.97	10.4	3.5	$5.3^{+0.4}_{-0.6}$	[b]$7.9^{+0.4}_{-0.3}$	2150	27
	97	6/20/73	2.97	2.70	.91	2.81	16.0	5.7	2.97	10.4	3.5	$5.0^{+0.1}_{-0.4}$	[b]$9.9^{+0.1}_{-0.4}$	2154	23
	126	7/17/73	2.97	2.70	.91	2.81	16.0	5.7	2.97	10.4	3.5	$5.2^{+0.0}_{-0.4}$	[b]$8.2^{+0.0}_{-0.4}$	2162	22
	188	9/28/73	2.88	2.77	.96	2.81	15.6	5.6	2.97	10.2	3.4	$8.2^{+0.0}_{-0.4}$	$11.1^{+0.0}_{-0.4}$	2165	15
	189	8/19/74	2.97	2.70	.91	2.81	15.6	5.6	2.97	10.4	3.5	$5.4^{+0.0}_{-0.2}$	$10.5^{+0.1}_{-0.1}$	2166	43
	203	9/12/74	2.97	2.70	.91	2.81	15.6	5.6	2.97	10.2	3.4	$6.1^{+0.0}_{-0.4}$	$22.5^{+0.0}_{-0.0}$	2169	40
	211	10/2/74	2.97	2.70	.91	2.81	15.6	5.6	2.97	10.2	3.4	$6.8^{+0.0}_{-0.6}$	$12.7^{+0.0}_{-0.4}$	2175	35

[a]Data not taken or unavailable.

[b]No preheat used.

[c]Quantizing and calibration error, ±3%, root-sum-square.

[d]Includes heating time in space only; ground time, thruster 1 - 83 hr, thruster 2 - 91 hr.

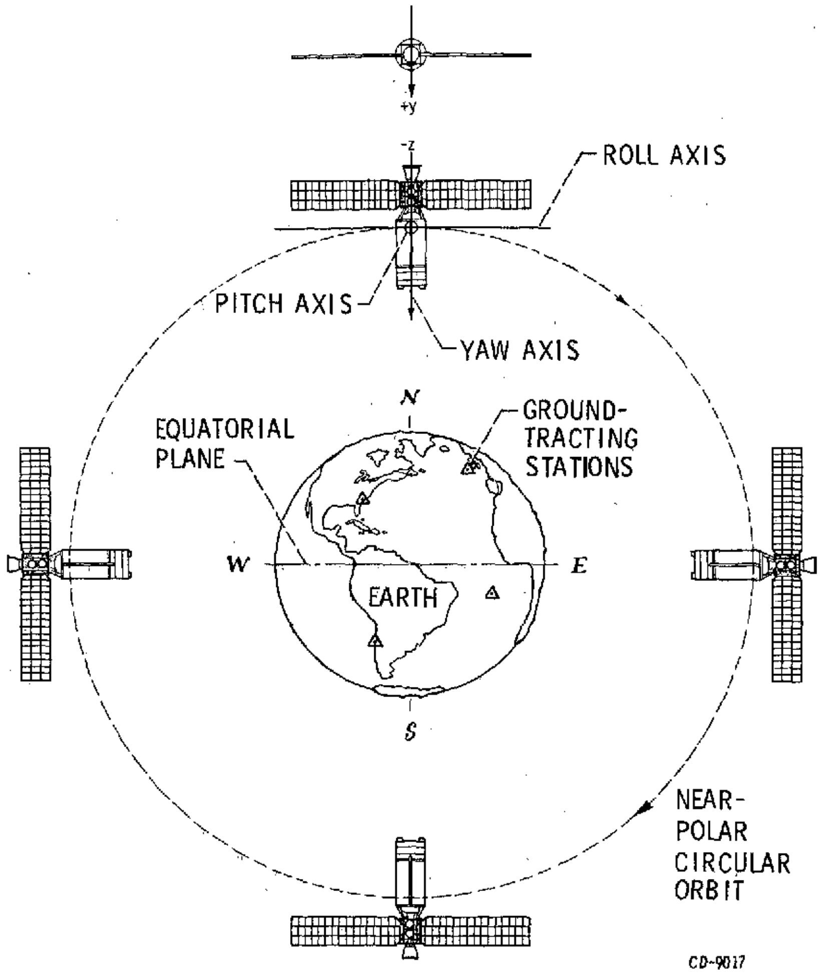


Figure 1. - SERT II vehicle coordinate system in orbit viewed from Sun for spring launch and sunset orbit injection.

E-8229

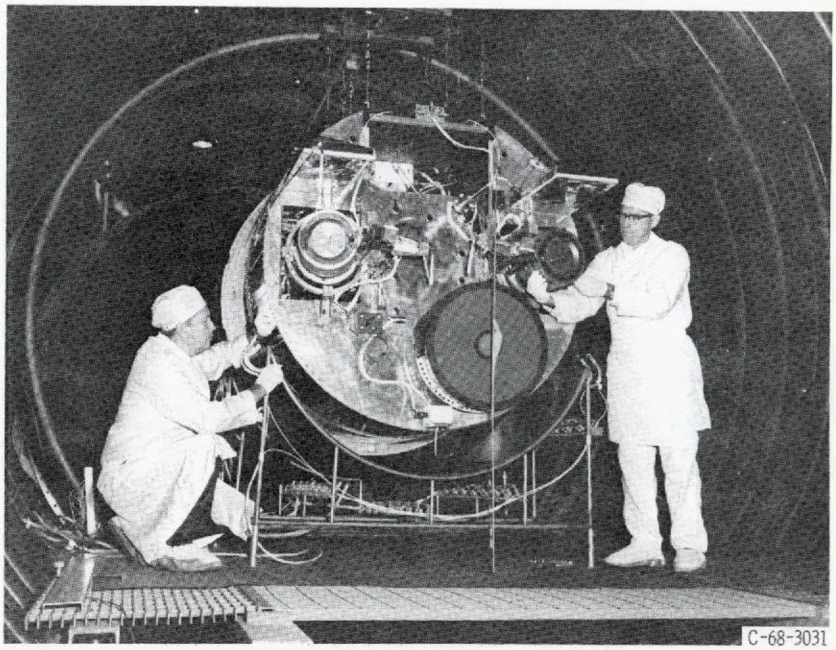


Figure 2. - SERT II flight spacecraft.

E-8229

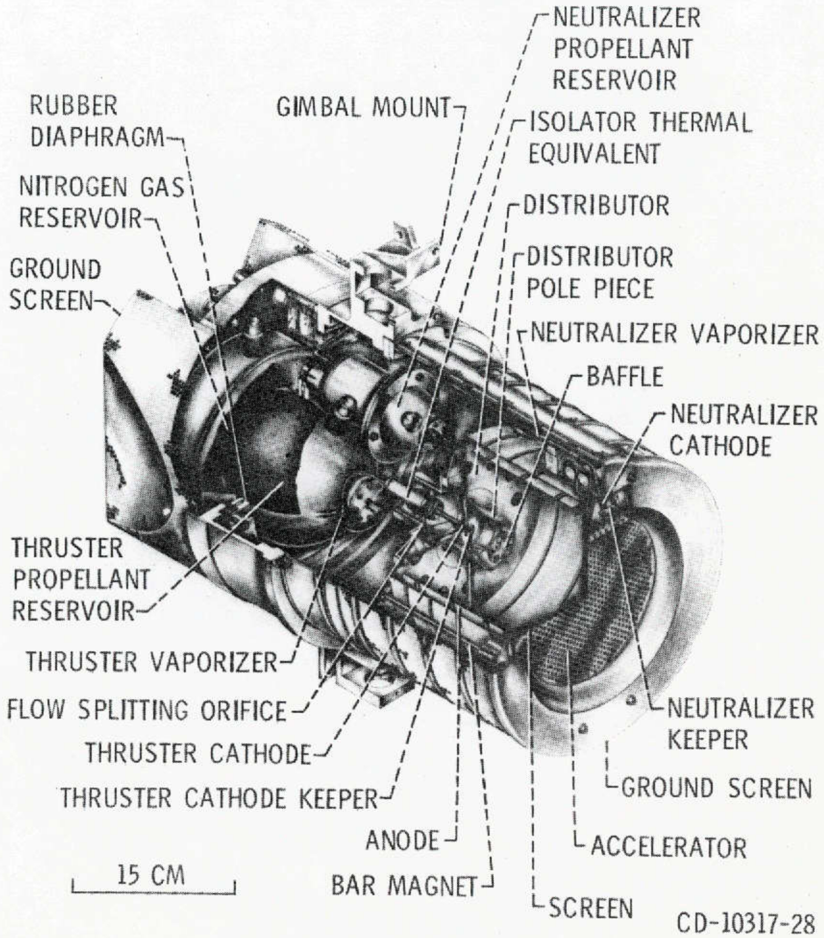


Figure 3. - SERT-II thruster.

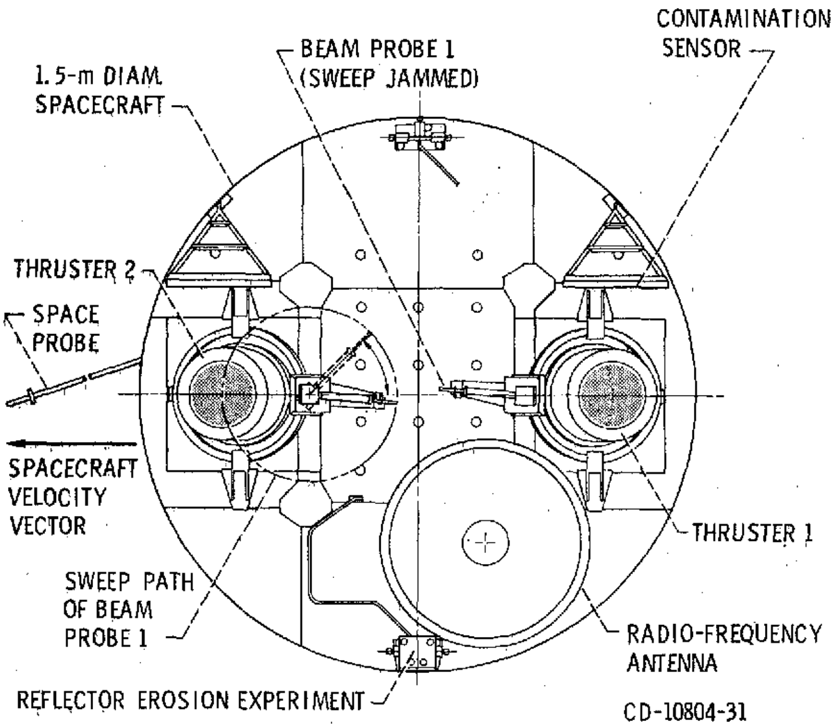


Figure 4. - Bottom view of SERT II spacecraft showing position of thrusters, probes, and experiments.

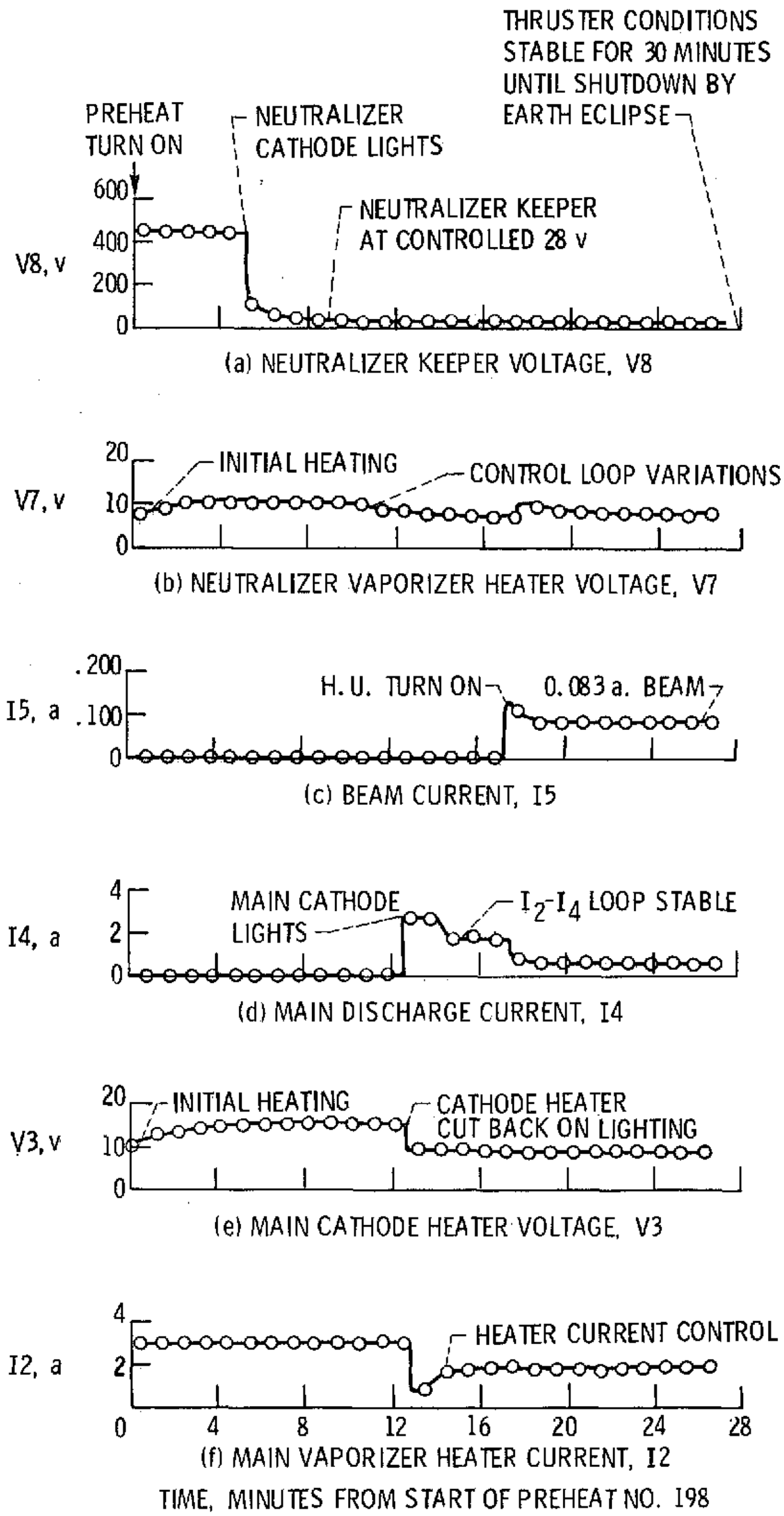
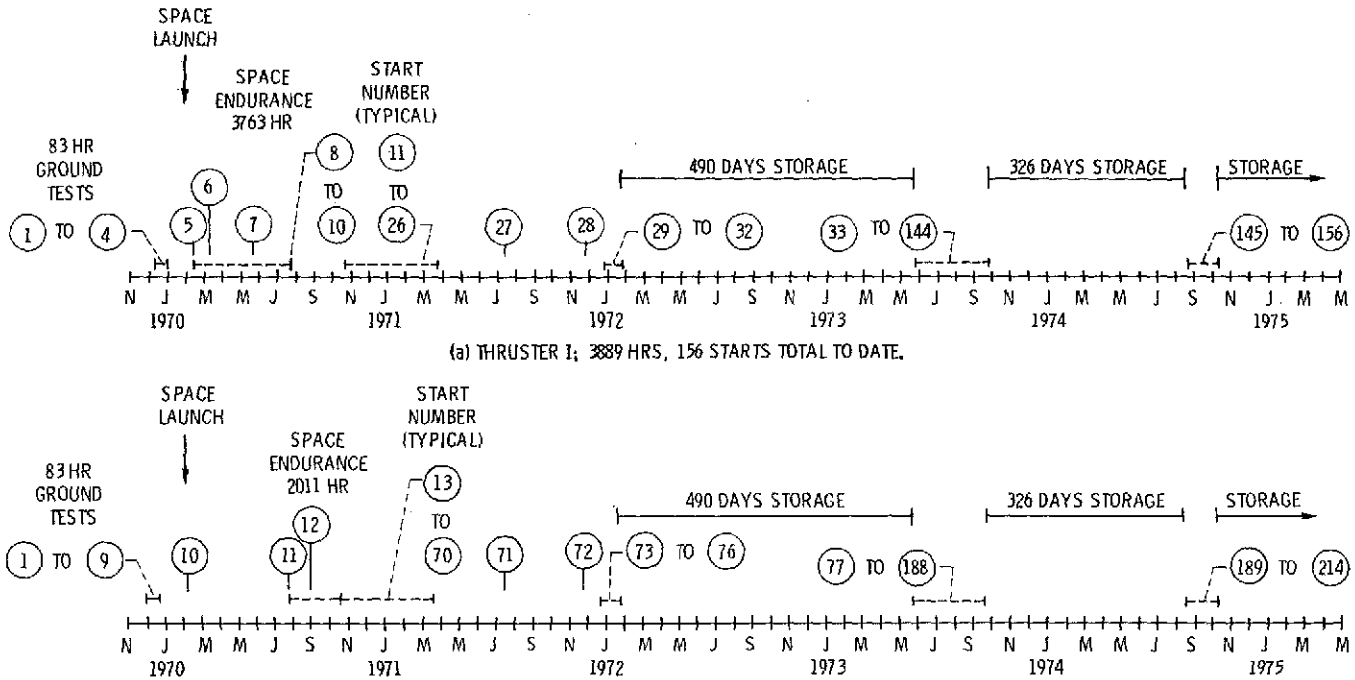


Figure 5. - Start and operation of SERT II flight thruster at 0.083 a beam current, September 10, 1974.



(a) THRUSTER 1; 3889 HRS, 156 STARTS TOTAL TO DATE.

(b) THRUSTER 2; 2175 HRS, 214 STARTS TOTAL TO DATE.

Figure 6. - Chronological representation of SERT II thruster preheats (starts), endurance running, and storage periods.

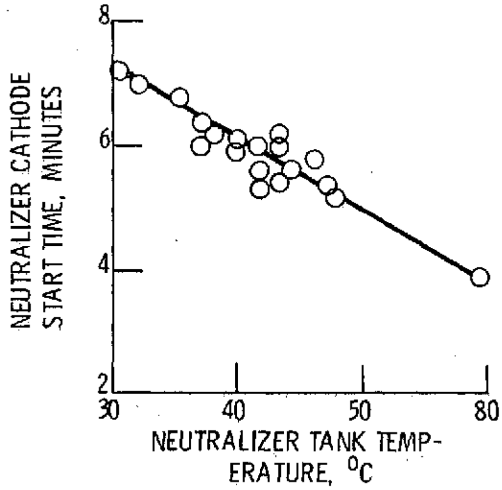


Figure 7. - Variation of thruster 2 neutralizer cathode start time with ambient thermal environment.

E-8229

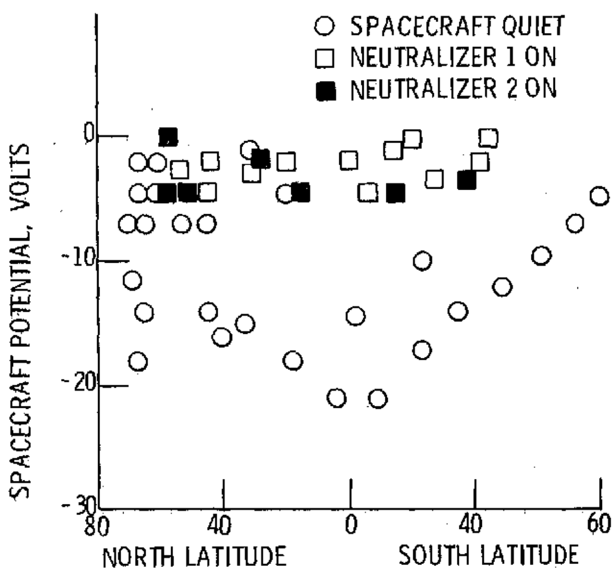


Figure 8. - SERT II spacecraft potential as a function of latitude. Measurement made by hot-wire beam probe 1 and/or 2. (Spacecraft potential is probe reading times minus one. Neutralizer cathode is at zero bias.)

	NEUT. BIAS,	SPACE-CRAFT	ΔV_{BS} v	ΔV_{BN} v	LATI-TUDE	NEUT. CATHODE EMISSION a	NEUT. CATHODE POTENTIAL v
○	+44.5	-43±2	86	41	28S	0.083	+2
□	+22.1	-28±1	59	37	34S	0.083	-6
△	0	-8±1	38	38	20S	0.083	-8
◇	-22.8	0±1	21	44	38S	0.130	-23
◺	-45.2	0±1	0	59	42S	0.195	-45

ΔV_{BS} - DIFFERENCE, BEAM CENTER-TO-SPACECRAFT
ΔV_{BN} - DIFFERENCE, BEAM CENTER-TO-NEUTRALIZER

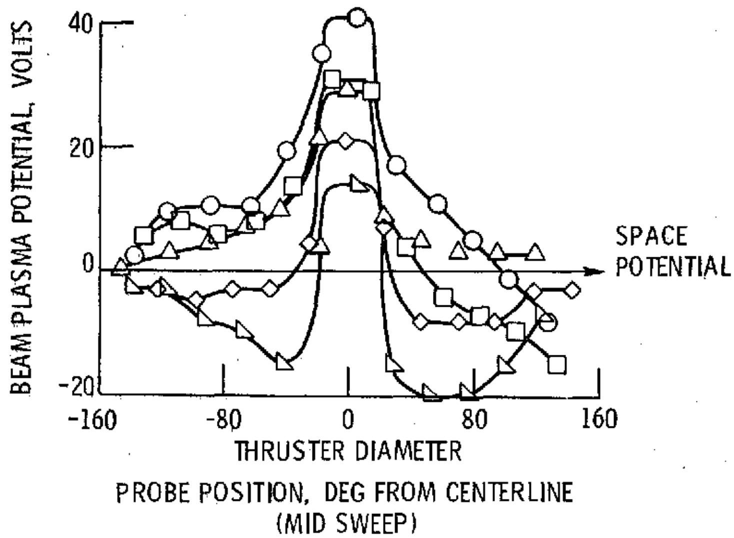


Figure 9. - Beam plasma potential profiles at various neutralizer bias potentials for thruster 2 at 0.083 a. beam current. Spacecraft potential measured by hot-wire beam probe 1. Spacecraft, neutralizer cathode, and beam plasma voltages are relative to space plasma voltage which is assumed zero.