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SUMMARY

The SERT I (Space Electric Rocket Test) spacecraft was launched at 0553:15 EDT on July 20, 1964, from Wallops Island, Virginia. Two ion thrustors were programed to operate sequentially during the flight. One of these, a contact ion thrustor, could not be started because of a high-voltage electrical short circuit. The other thrustor, which was of the electron-bombardment type, operated as expected for a total of 31 minutes and 16 seconds during the flight. The primary flight objective of confirming neutralization of a high-perveance ion beam was attained. Ion beam thrust was measured with three independent thrust measuring systems. All voltages, currents, and other operating characteristics of the electron-bombardment thrustor were close to the nominal values obtained in previous vacuum tank tests.

Telemetry data indicated no noticeable radio-frequency interference generated by the ion beam. Furthermore, all commands between the ground transmitter and spacecraft were transmitted and executed without interference.

INTRODUCTION

The SERT I (Space Electric Rocket Test) spacecraft was launched on a 46-minute 58-second ballistic trajectory on July 20, 1964. Two ion thrustors were programed to operate sequentially during the flight; one of these was a contact ion thrustor and the other was an electron-bombardment ion thrustor. The contact ion thrustor could not be started because of a high-voltage electrical short, but the electron-bombardment ion thrustor was operated for a total of 31 minutes and 16 seconds. This was the first time that an ion thrustor of any type had been operated in space. The performance data of the ion thrustor system obtained during the flight are presented herein, and these data are compared with similar data obtained in vacuum tank tests of the same ion thrustor system.

The primary objective of the SERT I flight experiment was to determine whether high-perveance ion beams can be neutralized in space. The secondary objectives of the flight were (1) to determine whether any unforeseen differences in ion thrustor performance occur in space as compared with their operation in vacuum tanks, (2) to determine whether different types of neutralizers are effective, and (3) to determine whether radio-frequency signals can be transmitted to and from an ion-propelled spacecraft.

Neutralization of ion beams had been effected in vacuum tank tests with various neutralizers, all of which used a heated filament as a source of electrons. A flight experiment, however, was required to provide conclusive proof of the feasibility of ion beam neutralization because of several uncertainties inherent in vacuum tank tests. These uncertainties exist because of (1) the secondary electrons produced when the ions strike metal surfaces, (2) the dilute plasma that fills the vacuum tanks during ion thrustor operation, and (3) the possible capacitive coupling between the ion beam and the vacuum tank walls.

In planning the SERT I flight, it was decided that neutralization of the ion beams could be demonstrated most conclusively by proving the existence of thrust produced by the ion beams. Accordingly, the two ion thrustors were mounted so that their thrust would change the spacecraft spin rate, and three independent systems for spin rate measurement were installed on the spacecraft. In addition, extensive ion thrustor instrumentation was included on the spacecraft to permit a direct comparison between flight and vacuum tank test results.

A lengthy development program was required before the ion thrustors, power supplies, and spacecraft components were made compatible and sufficiently reliable to assure a high probability of success in the flight. During this development program, performance data were obtained that provide the basis for comparison between vacuum tank and flight test results. In addition to the flight data, this report includes the performance data of the electron-bombardment ion thrustor obtained in two flight simulation tests that were conducted with the flight spacecraft in a 15-foot-diameter vacuum tank at the Lewis Research Center.

APPARATUS AND INSTRUMENTATION

Spacecraft

Figure 1 shows the SERT I spacecraft. A description of the design and development of this spacecraft can be found in reference 1.

Electron-Bombardment Ion Thrustor and Power Supplies

Figure 2 shows a flight-model electron-bombardment thrustor. A detailed descrip-

TABLE I. - THRUSTOR OPERATING CONDITIONS FOR SERT I FLIGHT

Component	Potential, V, volts	Current, J, amperes	Nominal operating power, watts	Operating point
Boiler	0 - 40 ac	0.60 - 2.61	10 (as required)	433° to 479° K
Magnetic field	6 dc	15 - 26	180	30 G
Cathode	9 - 10 ac	21.9 - 22.9	220	Cathode temperature set to give 5-A discharge
Discharge	46 dc	2.08 - 5.45	250	
Neutralizer heater	9 - 10 dc	22 - 26	200	$I_{neutralizer} = I_{beam}$
Positive acceleration (screen) potential	2500 dc	0.100 - 0.377 ($J_{source} \cong J_{beam}$)	725	2500 V
Negative acceleration (accelerator electrode) potential	-2000 dc	0.003 - 0.013	10	-2000 V

tion of the general thrustor design and principle of operation can be found in references 2 to 4. Nominal thrustor operating parameters are shown in table I. Figure 3 is an electrical schematic identifying the location of pertinent thrustor voltage and current measurements and the corresponding significance of these measurements. The neutralizer was a tantalum filament heated electrically to a temperature sufficient to yield a space-charge-limited electron current. All the flight and vacuum tank test data presented herein were obtained with the same electron-bombardment thrustor and the same power supplies.

The positive and negative high-voltage power supplies for the thrustor ion source and accelerator electrode were operated from a 56-volt battery with a 28-volt tap. Power output transistors in each of these power supplies received pulse-width-controlled drive from drivers and a master oscillator. The positive supply was current regulated, and the negative supply had voltage and current regulation. Current overload sensors in the output of both of these power supplies turned off all outputs when overloads exceeded 10 to 20 milliseconds of duration. The overload sensors were set below the current limits of the power supplies.

Thrust Detection Systems

The SERT I spacecraft was designed to measure thrust by determining changes in the spin rate and, hence, angular momentum of the spacecraft. The instrumentation was of high accuracy and wide response range in order to assure rapid measurements of change in spin rate in the event of deviation from expected thrustor performance or failure of thrustor or spacecraft after a short interval of time.

The spin rate was measured by three independent systems. Two of these systems were photovoltaic solar-cell spin-period detectors. The third system used an accelerometer mounted so that its sensitive axis was perpendicular to and intersected the spacecraft spin axis.

Each solar-cell system utilized a separate telemetry link. The received pulses were fed into clock-controlled electronic counters for period measurements. For pure spin, that is, no energy transfer between axes, the system error in spin-period measurement was calculated to be less than 0.1 percent for a single revolution.

The output of the accelerometer contained information concerning motion about all three principal axes. Therefore, the accelerometer provided necessary information concerning spacecraft precession if motion about the transverse axes is large and if there is energy transfer from the transverse axes to the spin axis.

Hot-Wire Calorimeter

Figure 4 shows the flight-model hot-wire calorimeter probe. The theory of operation of this instrument can be found in reference 5. The hot-wire calorimeter probe was pivoted through the ion beam at a location 7 inches down beam from the accelerator electrode to measure the distribution of ion beam power density (see fig. 1). Power density contour maps were constructed and graphically integrated to obtain total beam power. From this value of beam power and the measured accelerating potentials, a value of ion beam current was then calculated.

E-Field Meter

The E-field meter was mounted on the contact thrustor control box. It consisted of two sets of thin gold-plated vanes, 2 inches in diameter. The inner set of three vanes was stationary and connected to vehicle ground through a precision resistor. The outer set of three vanes rotated at a speed of 8000 rpm and intermittently shielded the inner vanes. If an electrostatic field were present, the inner vanes would generate a 400 cps signal, which was fed into a solid-state alternating-current signal amplifier. The amplifier output was then rectified and fed into the telemetry signal conditioning. The amplifiers were adjusted to provide data on electric fields up to a maximum of 120 volts per centimeter.

TABLE II. - SERT I FLIGHT HISTORY FOR JULY 20, 1964

(a) Programed

Event	Programer time, min
Activate payload and start programer	0
Lift-off	0:53
Contact thrustor switched to full heat mode	1:59
Separation	5:31
Engine deployment	5:33
Contact thrustor pod door opening	5:35
High voltage applied to contact thrustor	5:40
Study of contact thrustor switched to trap neutralizer	20:20
Study of contact trap neutralizer completed	26:34
High voltage removed from contact thrustor	26:39
Contact thrustor turned off	27:38
Electron-bombardment thrustor turned on	27:44
Thrustor neutralizer turned on	27:45
Boiler heating, command being sent from ground station to open magnetic field coil winding to trigger thrustor start	27:46
Study of hot-wire calorimeter probe started	39:44
Study of neutralizer voltage control started	41:43
Study of neutralizer voltage control completed	47:33
Study of hot-wire calorimeter probe started (neutralizer off)	47:34
Study of hot-wire calorimeter probe started (neutralizer on)	49:34

Lewis 15-Foot Vacuum Tank Facility

Vacuum tank tests of the spacecraft were conducted in a 15-foot-diameter by 60-foot-long chamber at pressures of the order of 5×10^{-6} torr. Figure 5 shows the flight spacecraft installed in the vacuum facility. The spacecraft was mounted on a spin table capable of spinning at the rate expected during flight, nominally 90 rpm. The spacecraft could be electrically insulated from ground.

RESULTS AND DISCUSSION

The data to be presented are divided into three parts:

(1) The test results of the ion-bombardment thrustor system during the 31 minutes and 16 seconds of operation for the ballistic space flight of July 20, 1964

TABLE II. - Concluded. SERT I FLIGHT HISTORY

FOR JULY 20, 1964

(b) Actual

Event	Flight time, min after launch
Spin-up of payload to ~109 rpm	2:48
Payload separation, precession dampers released, YO system timers activated	4:38
Both thrustors and hot-wire probe deployed	4:40
Electron-bombardment thrustor turned on	13:51
Thrustor producing thrust	17:49
Study of hot-wire calorimeter probe started (neutralizer on)	25:52
Study of hot-wire calorimeter probe completed	27:06
Study of neutralizer voltage control started	27:33
Study of second hot-wire calorimeter probe started (neutralizer off)	33:26
Study of second hot-wire calorimeter probe completed	34:34
Study of third hot-wire calorimeter probe started (neutralizer on)	35:25
Study of third hot-wire calorimeter probe completed	36:36
Thrustor command off	36:51
Electron-bombardment thrustor command on restart	38:42
Second thrust period and flight completed	46:58

(2) The test results of the electron-bombardment thrustor system for the two system tests of the flight spacecraft in the 15-foot-diameter vacuum chamber on June 7 and June 9, 1964

(3) The comparison of vacuum tank and space-flight test data

Space-Flight Test Data

The spacecraft programer sequence is given in table II(a), and the actual flight events are given in table II(b). The early portion of the flight was devoted to an attempt to operate the contact ion thrustor. This was unsuccessful due to a high-voltage electrical short circuit. The differences between the planned (table II(a)) and the actual (table II(b)) flight events resulted from command changes to the programer deemed advisable after the high-voltage short in the contact ion thrustor system.

Figures 6 and 7 show photographs of the real-time data recordings of the principal electron-bombardment-ion-thrustor operating parameters during the flight. The ordi-

TABLE III. - ELECTRICAL BREAKDOWNS DURING SERT I FLIGHT

Trip	Flight time	Operating conditions
1	19:45	Constant voltages
2	20:14	
3	20:42	
4	21:47	
5	22:35	
6	23:47	
7	25:28	
8	25:55	
9	26:56	
10	29:09	Study of neutralizer step voltage
11	29:17	
12	29:39	
13	29:49	
14	30:15	
15	30:18	
16	31:27	
17	31:47	
18	32:18	
19	32:26	
20	32:49	
21	32:59	
22	33:21	
23	33:28	
24	33:34	
25	33:44	Neutralizer off
26	33:50	
27	33:57	
28	34:03	
29	34:15	
30	34:23	
31	34:34	
32	34:47	
33	34:59	
34	35:40	
35	40:04	Constant voltages - second thrustor operating period
36	41:17	
37	41:21	
38	41:26	
39	42:15	
40	42:20	
41	42:49	
42	43:00	
43	43:05	
44	43:23	
45	43:34	
46	43:59	
47	44:29	
48	44:34	
49	45:48	
50	46:25	
51	46:30	
52	46:38	
53	46:53	

nate in figures 6 and 7 is the telemetry output voltage. The calibration curves for the various data channels are not linear; therefore, no absolute values are shown on the figures.

Figures 8 to 11 show quantitative values of the thrustor parameters monitored during the space flight test of the electron-bombardment ion thrustor. The ion-chamber discharge initiated at 3:50 minutes after the thrustor system was turned on (flight time, 17:49 min). The ion thrustor startup was normal.

The electron-bombardment ion thrustor was turned off by command at a flight time of 36:51 minutes and was turned on again at 38:42 minutes. In this time interval a second attempt was made to operate the contact ion thrustor. During the second period of operation of the electron-bombardment ion thrustor (flight time, 38:42 to 46:58 min), the vaporizer overheated slightly, which gave higher than nominal discharge and beam currents as indicated in figure 9. This overheating can be attributed to the departure from the planned flight test sequence and the resulting increased total test time on the electron-bombardment ion thrustor. A thermistor on the propellant vaporizer turned off the vaporizer heater power toward the end of the flight, but heat flux from the ion source was sufficient to cause the overheating.

Many electrical breakdowns that caused automatic shutdown of the power supply are indicated in the figures by vertical lines superimposed on the curves. The time of occurrence of each electrical breakdown is shown in table III. No change in telemetry signal strength was observed between those times when the thrustor was operating and those times when it was not operating; it can therefore be concluded that the ion beam did not interfere with radio-frequency transmission from the spacecraft.

During the total thrustor operating time of 31 minutes and 16 seconds, there were 53 electrical breakdowns in the thrustor system. Eight of these electrical breakdowns occurred in the initial 11 minutes and 53 seconds of operation. In this time interval the voltages were maintained constant and the thrustor gradually warmed up to its design operating conditions. Some electrical breakdowns would be expected in this period because of high outgassing rates during the warmup of the thrustor.

One electrical breakdown resulted in the course of the hot-wire calorimeter probe studies for a flight time of 25:52 to 27:06 minutes. An arc between the thrustor and the probe is a likely possibility, since such arcs have frequently occurred for similar probe surveys in vacuum tank tests.

Fifteen electrical breakdowns occurred during the neutralizer step voltage transient studies for a flight time of 27:33 to 33:39 minutes. Ten electrical breakdowns occurred in the flight time interval from 33:26 to 34:34 minutes while the ion thrustor neutralizer was turned off. Numerous breakdowns during some portions of the neutralizer step voltage transient study and during the time the neutralizer was off would be expected. Electrons were not available to neutralize the ion beam when the neutralizer voltage was too high or when the neutralizer heater current was turned off. Under these conditions ions could not leave the thrustor as a beam. Instead, the ions traveled to nearby low-voltage surfaces and were neutralized on contact. The resulting concentration of ions and propellant gas inside and adjacent to the thrustor would then be expected to cause frequent electrical breakdown.

The remaining 19 electrical breakdowns occurred in the course of the second operating period from a flight time of 38:42 to 46:58 minutes (end of flight). These 19 electrical breakdowns for almost steady-state operation were more frequent than were usually observed in the vacuum tank test program; the probable reason for this was the overheating of the propellant vaporizer, as will be discussed in the section Comparison of Vacuum Tank and Flight Test Experimental Data.

Figure 10 shows neutralizer current and beam current as functions of time. The two currents are equal, as expected, within the accuracy of measurement, except at flight times of 27:00 and 36:30 minutes for the hot-wire calorimeter probe surveys and 38:42 to 46:58 minutes for the final 8 minutes of thrustor operation. The difference in neutralizer current and beam current for the hot-wire calorimeter probe surveys can be attributed in part to ion interception and in part to secondary electron emission from the hot-wire calorimeter probe. These secondary electrons contribute to neutralization of the ion beam and do not show up as a measured current from the neutralizer. The difference in neutralizer current and beam current during the final part of the flight (beyond a flight time of 38:42 min) can probably be attributed to the unsteady thrustor operation. Frequent electrical breakdowns occurred throughout this part of the flight, and current measurements were varying rapidly with time immediately before and after each breakdown. The neutralizer and beam currents were commutated measurements and were therefore not made at the same instant in time.

Figure 10 shows the results of the neutralizer step-voltage transient study, which was repeated twice. Each study consisted of five discrete bias potentials applied to the neutralizer filament. This was accomplished by successively placing five different values of resistance in series with the neutralizer filament. The voltages applied to the neutralizer filament for the first half of the study were 190, 360, 650, 790, and 1030 volts positive relative to the spacecraft. Neutralizer filament potentials of 190 and 360 volts resulted in stable ion thrustor operation, which is indicative of effective ion beam neutralization. The neutralizer and beam currents decreased between 11 and 14 percent during these two neutralizer voltage steps (see fig. 10). At the three highest neutralizer voltages the thrustor became unstable; frequent electrical breakdowns occurred and the accelerator electrode current increased rapidly each time the ion source current rose above zero. This behavior is typical of an ion thrustor operation without adequate neutralization of the beam. For the second half of the study where the neutralizer potential was varied, the voltages applied to the neutralizer were 210, 375, 605, 886, and 1065 volts positive relative to the spacecraft. The same phenomena occurred as during the first half of the study; that is, the thrustor operation was stable at the two lower voltages, the neutralization and beam currents decreased about 12 percent, and the thrustor was unstable at the three highest voltages. Reasons for this decrease in neutralizer and ion beam currents are discussed in the section Comparison of Vacuum Tank and Flight Test Data.

Figure 11 shows the ratio of accelerator current to ion beam current as a function of time. The accelerator current increases monotonically, as expected, because of charge exchange. Figures 10 and 11 indicate the beam current and accelerator current to be zero during the time the neutralizer was turned off; however, close inspection of the data of figures 6 and 7 indicates that the ion source and accelerator currents increased momentarily each time the anode-cathode discharge occurred for a brief interval. The time intervals are so short that they are not shown in figures 10 and 11. For the time interval when electrons were not available to neutralize the ions, the ions could not leave the thrustor as a beam. Instead, the ions were drawn to the accelerator electrode and possibly other metal surfaces, which caused automatic shutdown of the power supply. This is expected to occur when the ion beam is not neutralized.

Figure 12 shows the ratio of accelerator current J_A to ion beam current J_B plotted as a function of J_B. A linear plot would be expected here if the propellant utilization efficiency were constant. The fact that J_A/J_B increases at greater than a linear rate indicates that the utilization efficiency decreased as J_B was increased.

The upper curve in figure 13 shows the thrust calculated from ion beam current and voltage measurements. The two lower curves in figure 13 show the thrust corrected for doubly ionized mercury ions and ion beam spreading.

Reference 6 indicates that for an ion-chamber discharge voltage of 46 volts, 6.5 percent doubly charged ions (13 percent of the current) can be expected. Allowing for these doubly charged ions produces a 4-percent reduction in thrust calculated from current and voltage measurements.

Ion beam spreading causes a reduction in thrust proportional to the cosine of the ion beam spreading angle. The ion beam current profile as a function of radius was obtained from the four hot-wire calorimeter probe surveys. It was then assumed that the same profile existed at the exhaust of the ion thrustor. The average beam spreading angle was computed by using this assumption. The resulting average reduction in thrust using this approximate method was about 1 percent.

Figure 14 is a plot of the delivered thrust as a function of time as measured by the two sun sensor systems and the accelerometer system. The thrust as computed from these three measurements agrees to within a few percent. The thrust increased monotonically with the ion beam current. Throughout each of the hot-wire calorimeter probe surveys at flight times of 25:52 and 35:25 minutes the thrust measured by the sun sensor and accelerometer systems decreased. The hot-wire calorimeter probe projected frontal area is 3.5 square inches. When a beam spread angle which yields a 1-percent thrust reduction is assumed, the probe body intercepts 27 percent of the ion beam at the midpoint of its traverse; consequently, a thrust reduction of approximately 27 percent would be expected at that moment in time. The measured thrust reduction was 20 and 24 percent, respectively, for these surveys. The nonuniform current density distribution from the thrustor might account for this small discrepancy. The sun sensor and accelerometer data show no measurable change in spin rate (consequently, no thrust) for the portion of the flight when the neutralizer was turned off; this confirms the results of all other measurements during this time period.

In figure 15, the thrust calculated from current and voltage measurements is compared with the thrust calculated from change in spacecraft spin rate as measured by the sun sensors and accelerometer. In the discussion that follows, "measured thrust" is defined as the value computed from changes in spacecraft spin rate; "calculated thrust" is defined as the value calculated from measurements of ion beam current and net accelerating voltage with correction for doubly charged ions and beam spreading. The calculated thrust is about 25 percent above the measured thrust throughout most of the flight. For example, at flight times of 25:00 and 43:00 minutes the calculated thrust is 22.3 and 26.1 percent, respectively, above the measured thrust. The measured thrust is believed to be accurate within ±1 percent. This accuracy is based on the allowance for transfer of spacecraft momentum from precession into spin, the estimated accuracy of the measurement of spacecraft moment of inertia and moment arm of the thrustor about the spin axis, and the accuracy of transmission and recording of the digital data from the sun sensors. It must then be concluded that the calculated thrust is erroneously high by approximately 25 percent. This error is attributed to a backstreaming electron current originating at the neutralizer, following some path around the outside of the thrustor, and terminating at the high-voltage ion source. The solution to the problem of backstreaming electrons in all vacuum tank tests of ion thrustors has been to completely shield the high-voltage portions of the thrustor with a screen. The flight thrustor had such a screen biased at spacecraft potential. The screen can be seen in figure 2. The screen was tested in a small vacuum tank prior to flight and was found to be completely effective. However, additional tests conducted subsequent to the flight showed the screen to be inadequate in a larger vacuum tank. It can, therefore, be concluded that the screen was also inadequate to prevent backstreaming electrons during flight. The details of the vacuum tank tests of this screen are further discussed in the section Comparison of Vacuum Tank and Flight Test Data.

As shown in figure 15, for those portions of the neutralizer step voltage transient study where stable thrustor operation was obtained (flight times of 27:41 to 27:57, 28:13 to 28:29, 30:55 to 31:11, and 31:27 to 31:43 min), the calculated thrust decreased 17.2, 16.9, 12.3, and 18.1 percent. For the same four time intervals, the measured thrust decreased 3.5, 6.5, 3.2, and 4.3 percent, while the calculated thrust was 2.5 to 11.0 percent above the measured thrust.

The expected thrust reduction for the previous four time intervals was 4.0, 7.4, 4.2, and 7.8 percent, respectively. These values are calculated from the change in net accelerating potential, that is, the ion source potential minus the neutralizer potential. The reductions in measured thrust are in excellent agreement with the theory except for the last of the four time intervals discussed. An electrical breakdown had caused power supply shutdown at a flight time of 31:27 minutes, and the thrustor was not operating during approximately the first half of the neutralizer voltage step that was applied over the flight time interval from 31:27 to 31:43 minutes. The time available to measure thrust during this voltage step may have been too short to yield accurate data.

The reduction in calculated thrust during each of the four time intervals is due only in part to the reduced net accelerating potential discussed previously. Most of the reduction in calculated thrust is due to the reduced apparent ion beam current (see fig. 10). The reduced apparent ion beam current is, in turn, due to the elimination of the backstreaming electron current when the engine screen was biased at negative potential relative to the neutralizer.

From the foregoing discussion, it is apparent that the neutralizer and ion beam currents were in error at all times throughout the flight except for the four brief intervals when steady thrustor operation was obtained with the neutralizer biased at a positive voltage relative to the spacecraft. Hence, it is only for these four time intervals that agreement can be expected between calculated and measured thrust. As noted previously, the calculated thrust was 2.5 to 11.0 percent above the measured thrust for these four intervals.

The spacecraft spin rate increased from 85 to 94 rpm during the flight due to ion thrust. The data obtained from the accelerometer (ref. 1) showed the spacecraft angular motion to be almost pure spin. The total spacecraft angular momentum in precession at the start of ion thrustor operation was less than 1 percent of the total angular momentum imparted to the spacecraft by the ion thrustor. The total spacecraft angular momentum in precession decreased gradually throughout the flight, and at the end of the flight it was approximately one-half its initial value.

Figure 16 shows the recordings of the five hot-wire calorimeter probes for the three separate surveys. In each survey, the probe was moved across the ion beam and then back to its original position, thus producing two sets of data for each survey. The second survey was with the neutralizer turned off and verifies that no ion beam existed at that time. The ion beam power density profiles obtained from the hot-wire calorimeter probe during the first and third surveys are shown in figures 17 and 18, respectively. A value of total ion beam power was obtained by graphical integration of the data in these figures. In the first survey the values of total beam power were 492 and 482 watts for the first and second probe sweeps, respectively. In the third probe survey, the values of total beam power were 618 and 643 watts.

Figure 7 shows a photograph of the real-time oscillograph trace of the output of the rotating vane electric field meter. The oscillations in the E-field meter output throughout thrustor operation are at the beat frequency resulting from the arithmetic difference between the telemetry commutating rate (120/min) and the spin rate of the spacecraft (85 to 94 rpm). This beat frequency varies between 35 and 26 cycles per minute during the flight. It can, therefore, be concluded that the apparent E-field varies with a frequency equal to the spacecraft spin rate. If the E-field meter readings are interpreted to truly indicate a surface electric field strength, the minimum and maximum portions of the E-field data are 4 and 15 volts per centimeter, respectively. These values of E-field strength, for reasonable assumptions of electron density and temperature, indicate a spacecraft potential between 40 and 150 volts. There are, however, other possible explanations for the observed behavior of the E-field meter. These include photoemission from the vanes of the meter and interaction between the moving spacecraft and the ambient plasma.

For the two lowest neutralizer voltage steps (190 V at 27:41 to 27:57 min and 360 V at 28:13 to 28:29 min), the E-field meter output ceased its oscillation and gave an apparent E-field magnitude of 2.5 and 3.3 volts per centimeter, respectively. Similar results were obtained when the two lowest neutralizer voltage steps were repeated (210 V at 30:29 to 30:39 min and 375 V at 30:55 to 31:11 min). Again, the E-field oscillations ceased; the indicated field strength was 2.5 and 3.1 volts per centimeter, respectively.

The meter does not provide a direct indication of the polarity of the E-field. It may be significant that the E-field oscillations ceased for the time intervals when the neutralizer was biased at a positive potential; the presence or absence of E-field oscillations may indicate a difference in polarity. No definite conclusions can be made regarding the significance of the E-field data until the data analysis and further tests of a rotating-vane E-field meter are completed.

Vacuum Tank Test Data

This section will report electron-bombardment thrustor performance data obtained in two flight simulation tests that were conducted with the flight spacecraft in the 15-foot-diameter vacuum facility at Lewis Research Center on June 7 and June 9, 1964.

Figures 19 to 22 show thrustor data obtained in the vacuum tank test of June 7, 1964. The ion chamber discharge was initiated at 3:30 minutes. No hot-wire calorimeter probe surveys or voltage step transient studies were attempted in this test. The ion beam current reached 190 milliamperes (fig. 21); this low ion beam current was the result of a low setting on the vaporizer temperature control. Figure 23 shows the thrust calculated from the ion beam current and net accelerating voltage with no corrections for doubly charged mercury ions and ion beam spreading.

Figures 24 to 27 show thrustor parameters for the test run on June 9, 1964. All operating parameters were normal (see table I). The ion chamber discharge was initiated at 4:45 minutes. The ion beam current reached 280 milliamperes (fig. 25).

During the step voltage transient study with the spacecraft isolated from ground, the bias potentials on the neutralizer were 196, 360, 630, 830, and 994 volts with respect to the spacecraft. Figure 27 shows that neutralizer potentials of 630, 830, and 994 volts resulted in an abrupt increase in the accelerator impingement current; this would be expected for an unneutralized ion beam. For neutralizer potentials of 196 and 360 volts, the accelerator impingement current did not change. These results are in good agreement with the flight test results.

Figure 28 shows the thrust calculated from the ion beam current and the net accelerating potential with no corrections for doubly charged mercury ions and ion beam spreading.

Figure 29 presents the hot-wire calorimeter probe data. The graphically integrated ion beam power obtained from the contour maps is 326 and 375 watts for the first and second probe sweeps, respectively.

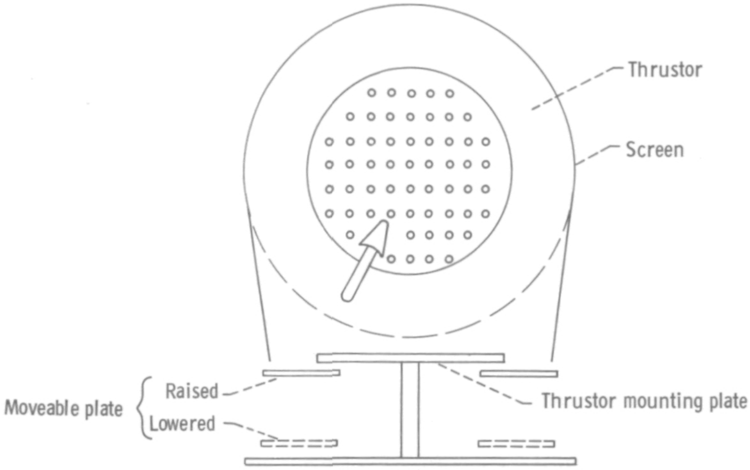
In the vacuum tank tests, no quantitative thrust data could be obtained from either the sun sensor or the accelerometer system. The sun sensor and accelerometer systems were both functionally checked out during these tests.

Comparison of Vacuum Tank and Flight Test Data

Figures 17, 18, and 29 provide a comparison of beam spreading in flight and in a vacuum tank test. The beam current in flight was 44 percent higher than in the vacuum tank tests. Nevertheless, the ion beam periphery, as determined by the hot-wire calorimeter probe, was at almost identical locations for the two tests. This result indicates that substantially all of the ion beam spreading is due to ion optics and not to the effect of space charge on the ion beam.

As noted earlier, seven electrical breakdowns occurred during the first 11 minutes and 53 seconds of the flight, and 19 electrical breakdowns occurred during the last 8 minutes and 16 seconds of the flight. In the course of these two time intervals there were no probe surveys or neutralizer voltage changes that might cause the electrical breakdowns. For comparison, there were no electrical breakdowns for the vacuum tank test of July 7. There was a total of three electrical breakdowns for the vacuum tank test of June 9, but all three of these occurred for the neutralizer voltage study when the neutralizer was biased at voltages more than 400 volts above spacecraft potential.

Extensive testing in vacuum tanks has demonstrated that ion thrustors which have not been operated for several weeks will exhibit frequent electrical breakdowns for several minutes to an hour, and thereafter operate with few, if any, electrical breakdowns. The phenomenon has always been attributed to the removal of adsorbed gas and dust from the high-voltage electrodes. The flight thrustor had not been operated for 41 days prior to operation in space. Therefore, the frequent electrical breakdowns that were encountered in the flight were expected. Conversely, since the thrustor had been operated on June 6, the absence of electrical breakdowns in the two vacuum tank tests on June 7 and June 9 was also expected.



Another probable cause of the frequent electrical breakdowns during the later portion of the flight is the high vaporizer temperature for this portion of the flight as compared with the vaporizer temperature in vacuum tank tests (figs. 8 and 23). This higher vaporizer temperature, for fixed thrustor conditions, would be expected to produce a lower propellant utilization efficiency. Figure 12 shows the ratio J_A/J_B is increasing at greater than a linear rate with increase in J_B. This is an indication of a reduction in propellant utilization efficiency with increase in vaporizer temperature and beam current. An increased frequency of electrical breakdowns is typically observed for vacuum tank tests where a low utilization efficiency is obtained; this is attributed to the increased densities of neutral propellant atoms and charge exchange ions in the vicinity of the high-voltage parts of the thrustor.

To check the adequacy of the thrustor screening for preventing backstreaming electrons, a flight-model electron-bombardment ion thrustor was installed in the 15-foot-diameter vacuum tank at NASA Lewis Research Center. These tests were conducted after the flight test. The largest opening in the thrustor screens was around the edges of the thrustor mounting plate (see above sketch). A small moveable plate was therefore added to the system as shown in the sketch. This plate could be raised to block most of the opening between the thrustor screen and the mounting plate, or it could be lowered to provide essentially the same screen arrangement that was used in flight. With the thrustor operating at conditions giving an indicated 380-milliampere ion beam, the plate was raised to observe the reduction in indicated beam current, and the thrustor screens were then biased at negative voltages to further reduce the indicated beam current. The results are tabulated in table IV in the order in which the tests were conducted.

For these tests, a low-mass conical thrust target was used to indicate relative values of thrust. During all of the tests there was no change in the thrust target reading, which provided further evidence that only backstreaming electron current was

TABLE IV. - THRUSTOR SCREENING RESULTS FOR PREVENTING BACKSTREAMING ELECTRONS

Indicated beam current, mA	Test conditions
380	Plate lowered; no screen voltage
360	Plate raised; no screen voltage
330	Plate raised; -100 V on screen and plate
320	Plate raised; -300 V on screen and plate
320	Plate raised; -400 V on screen and plate
320	Plate raised; -450 V on screen and plate
360	Plate raised; no screen voltage
380	Plate lowered; no screen voltage

TABLE V. - COMPARISON OF RESULTS OF ION BEAM FOR BOTH FLIGHT AND JUNE 9 TANK TEST

Type of test	Probe survey	Sweep	[a]P_P, W	[b]P_J, W	[c]P_F, W	P_P/P_J	P_P/P_F
Flight	1	1	492	537	461	0.92	1.07
Flight	1	2	482	556	466	0.87	1.03
Flight	3	1	618	632	552	0.98	1.12
Flight	3	2	643	656	563	0.98	1.14
Vacuum tank (June 9)	1	1	326	444	---	0.74	----
Vacuum tank (June 9)	1	2	375	471	---	0.80	----

[a]Total ion beam power from graphical integration of beam profiles obtained from hot-wire calorimeter probe data.

[b]Total ion beam power calculated from measured ion beam current and net accelerating voltage with a 16-percent correction for backstreaming electrons.

[c]Total ion beam power calculated from measured thrust and net accelerating voltage.

being altered and no change in true ion beam current was occurring. The 60-milliampere reduction in indicated beam current is 16 percent of the original indicated beam current. These data are in reasonable agreement with the flight data of figure 10, where a reduction is shown in neutralizer current and apparent ion beam current of about 12 percent for the four brief time intervals when the neutralizer was biased at a positive potential relative to the spacecraft (and screen).

A 16-percent error in indicated ion beam current due to backstreaming electrons can be assumed for the flight data. This error should exist throughout the flight, except for the brief time intervals when the neutralizer was biased at a positive voltage. Correction for the assumed error in beam current results in a reduction in calculated thrust equal to 20 percent of the measured thrust. Since the calculated thrust was initially about 25 percent above the measured thrust, the previous correction gives agreement between the two thrust values within about 5 percent. A 5-percent error in sensing and recording the beam current is possible.

A comparison of the ion beam power obtained by different methods in both the flight and the June 9 vacuum tank tests is shown in table V.

The flight data show good agreement between the hot-wire calorimeter probe and other methods of measurement. In the vacuum tank test, the beam power obtained with the probe was between 70 and 80 percent of the power calculated from current and voltage measurements; this result is typical of similar probe studies in vacuum tanks.

SUMMARY OF RESULTS

The following major results were obtained from the SERT I flight of July 20, 1964:

1. The first successful flight test of an ion thrustor was achieved. The electron-bombardment ion thrustor was operated for 31 minutes and 16 seconds during the flight. The contact ion thrustor could not be operated because of a high-voltage electrical short circuit.

2. Effective neutralization of the ion beam was obtained; this was the primary objective of the flight.

3. No major differences were observed between ion thrustor performance in space and in vacuum tanks.

4. Only one type of neutralizer was tested during the flight, but a significant variation in its effectiveness was obtained by biasing the neutralizer at various positive potentials relative to the spacecraft. This particular neutralizer remained effective up to voltages of approximately 400 volts; it was ineffective at higher voltages. This same result was obtained in similar neutralizer voltage studies in a vacuum tank; it is therefore concluded that meaningful beam neutralization experiments with other type neutral-

izers can be conducted in vacuum tanks.

5. There was no indication of the ion beam causing radio-frequency interference with signal transmission to and from the spacecraft.

6. All instrumentation functioned during the flight. Excellent agreement was obtained with two independent sun sensors and a radial accelerometer for measuring changes in spacecraft spin rate. Values of ion beam thrust determined from change in spacecraft spin rate have an estimated accuracy of ±1 percent. Ion beam thrust calculated from ion beam current and voltage measurements agreed within 5 percent with the other thrust measurements after corrections were made for backstreaming electrons, doubly charged ions, and beam spreading.

7. Measured ion beam current and neutralizer current were in good agreement during those portions of the flight when the thrustor was operating near design conditions. On the several occasions when the ion beam was turned off, the neutralizer current also immediately dropped to zero.

8. Frequent electrical breakdowns caused automatic shutdown of the power supply during the flight. However, the frequency of these electrical breakdowns did not exceed that expected in vacuum tank tests under similar conditions.

9. Spacecraft potential was indicated by a rotating vane electric field meter. A time-varying E-field was indicated with a frequency equal to the spacecraft spin rate and an amplitude of 4 to 15 volts per centimeter. Doubt remains as to the interpretation to be placed on the E-field data; further analysis is under way.

10. Ion beam spreading was measured at a location 7 inches down beam from the accelerator electrode by means of a 5-sensor hot-wire calorimeter probe. The amount of beam spreading was almost identical with that observed in vacuum tank tests.

11. The spacecraft angular motion consisted almost entirely of rotation about the design spin axis. The total angular momentum about transverse axes (precession) was less than 1 percent of the total angular momentum imparted about the spin axis by ion beam thrust.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, January 13, 1965.

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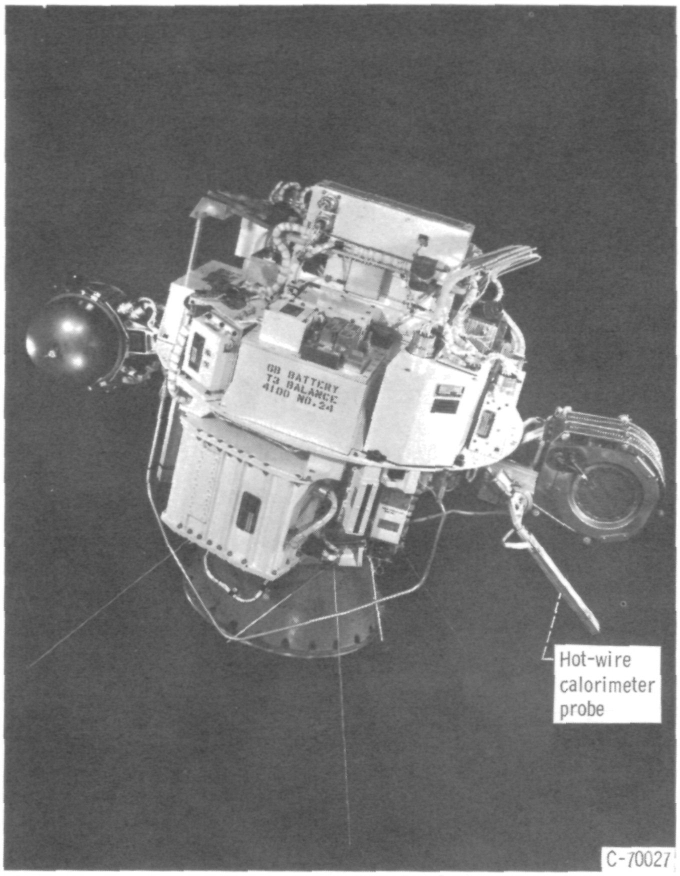


Figure 1. - SERT I flight-model spacecraft.

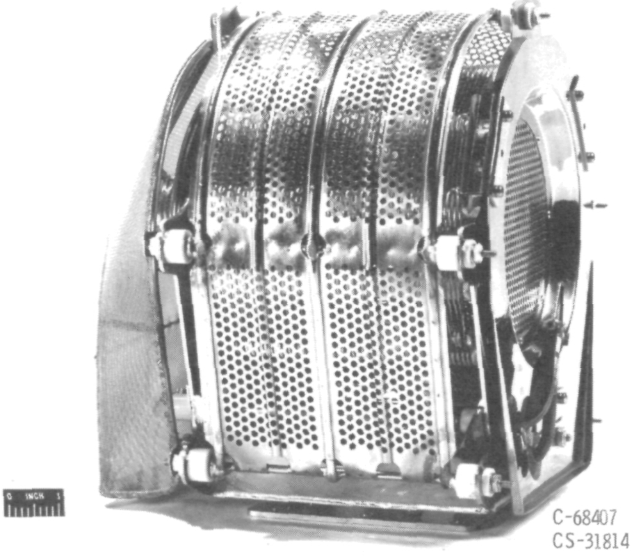


Figure 2. - Flight-model electron-bombardment thrustor.

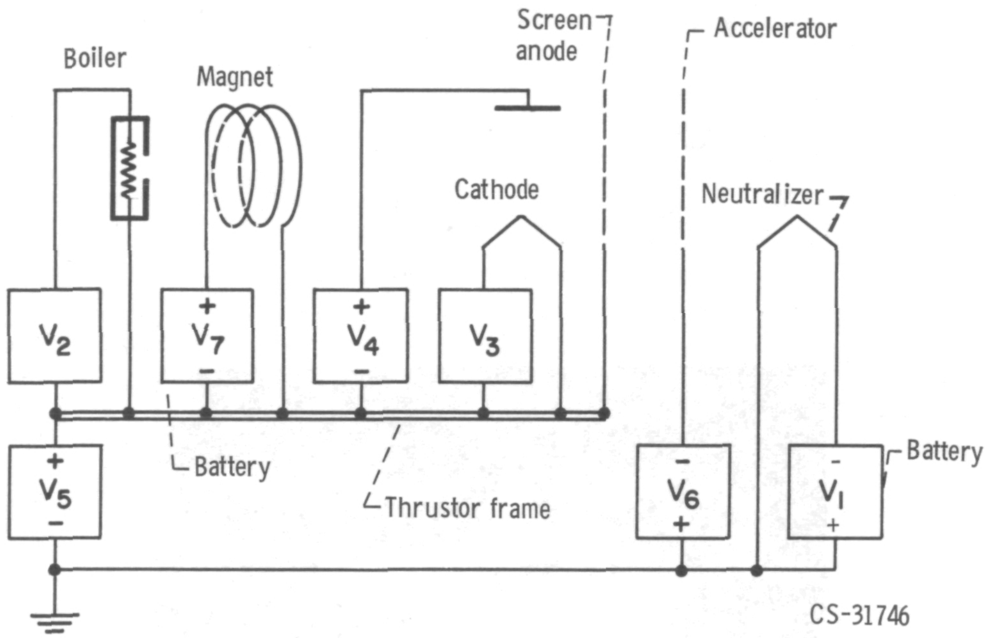


Figure 3. - Electrical schematic of electron-bombardment thrustor.

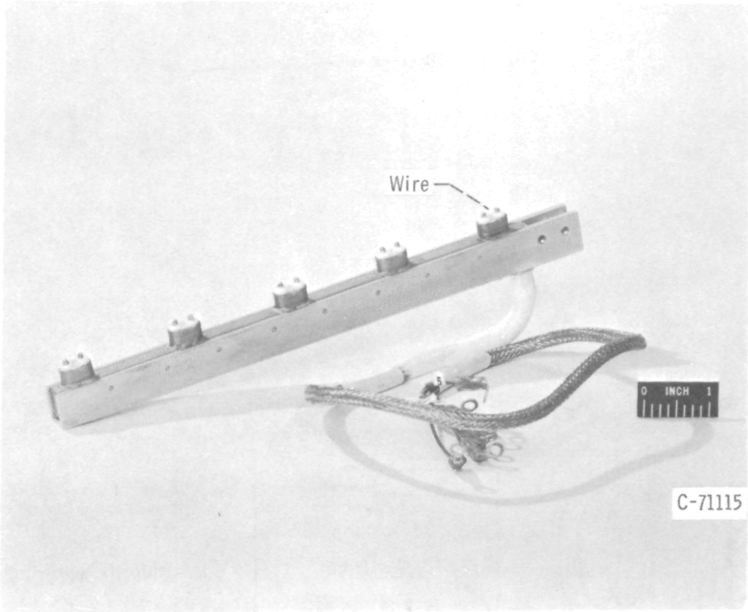


Figure 4. - Flight-model hot-wire calorimeter probe.

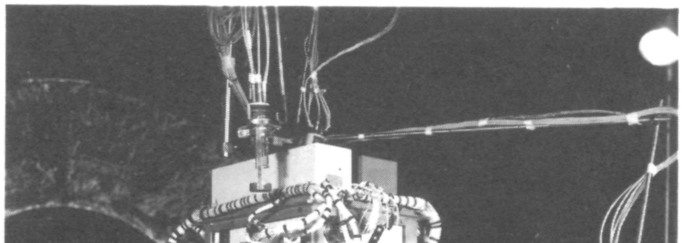


Figure 5. - SERT spacecraft in Lewis Research Center 15-foot-diameter vacuum tank.

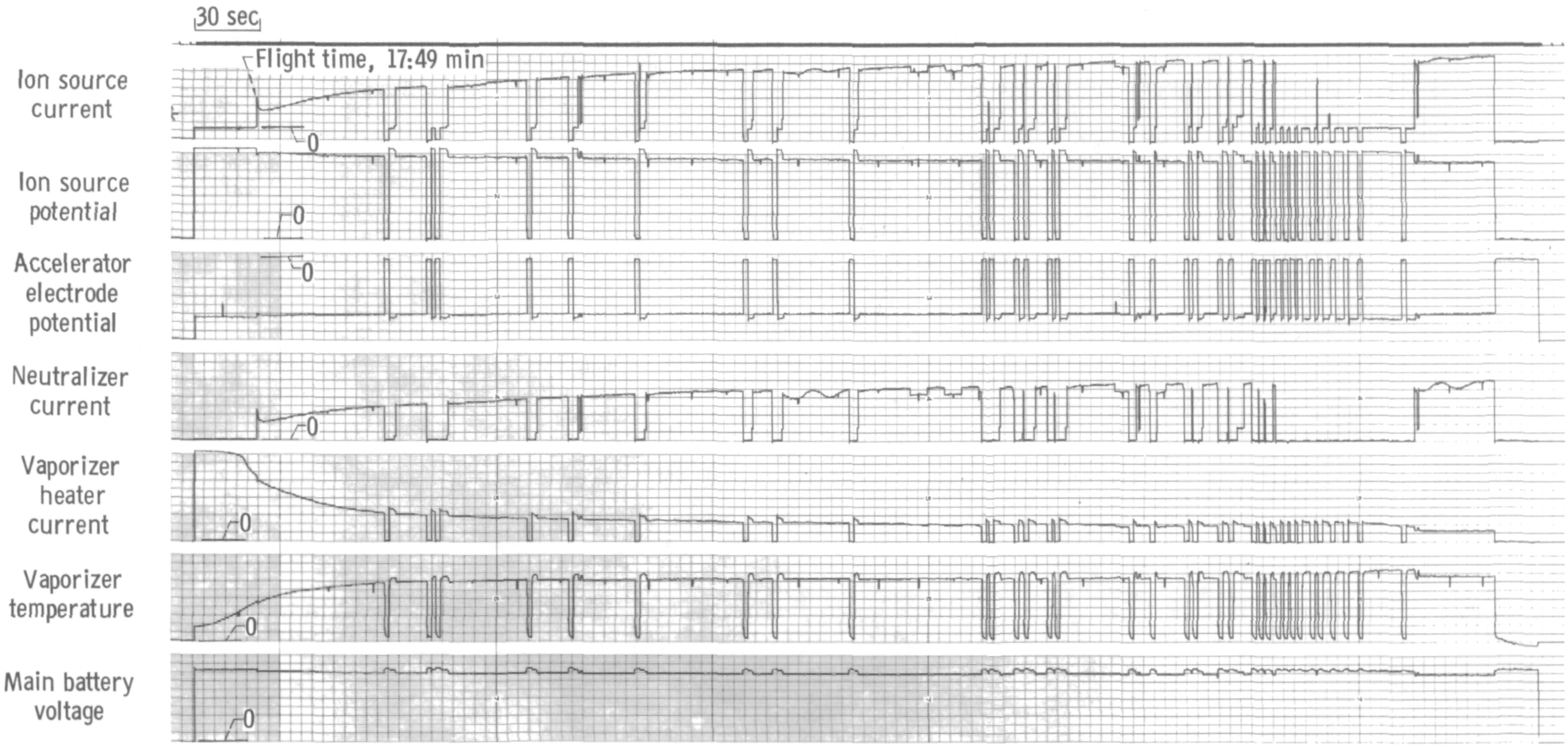


Figure 6. - Real time oscillograph 1 data traces. SERT I flight.

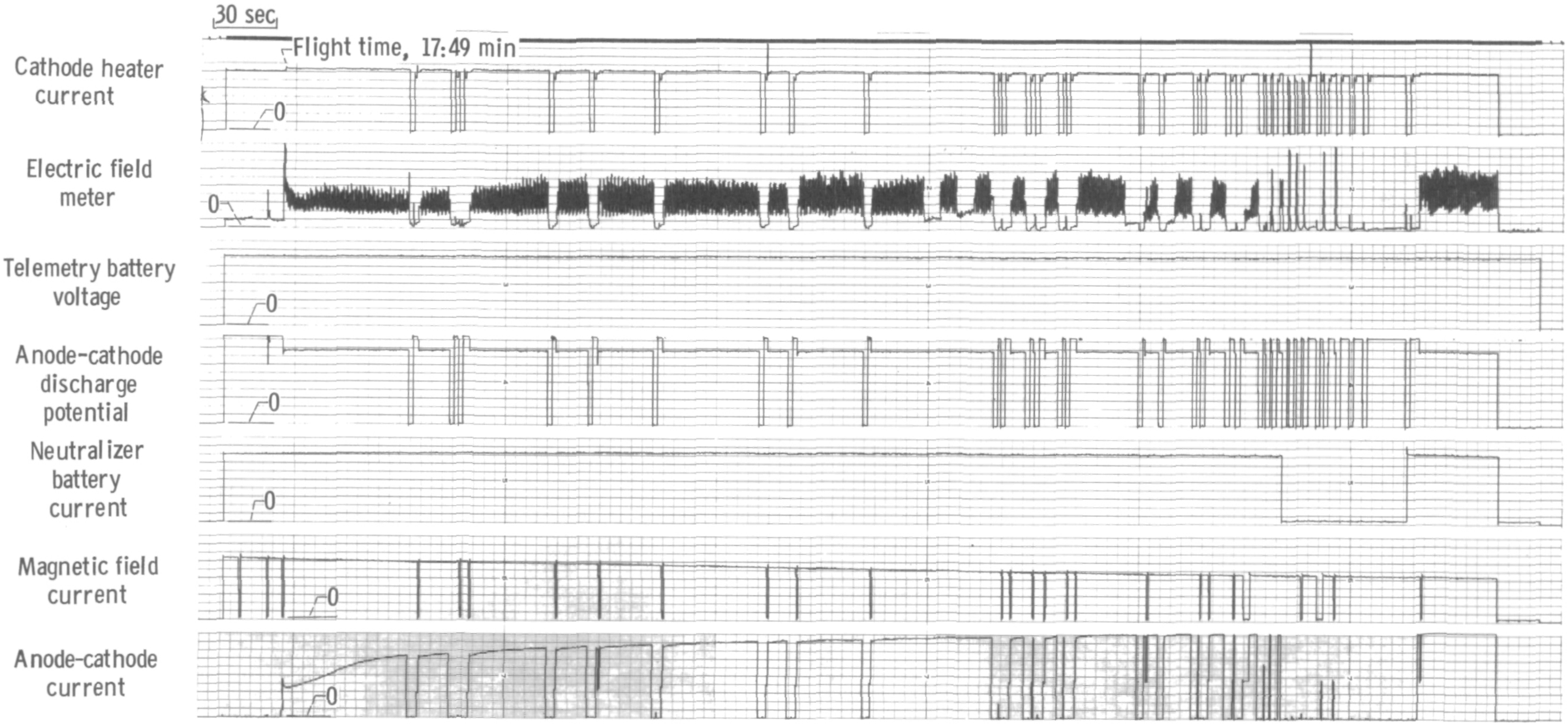


Figure 7. - Real time oscillograph 2 data traces. SERT I flight.

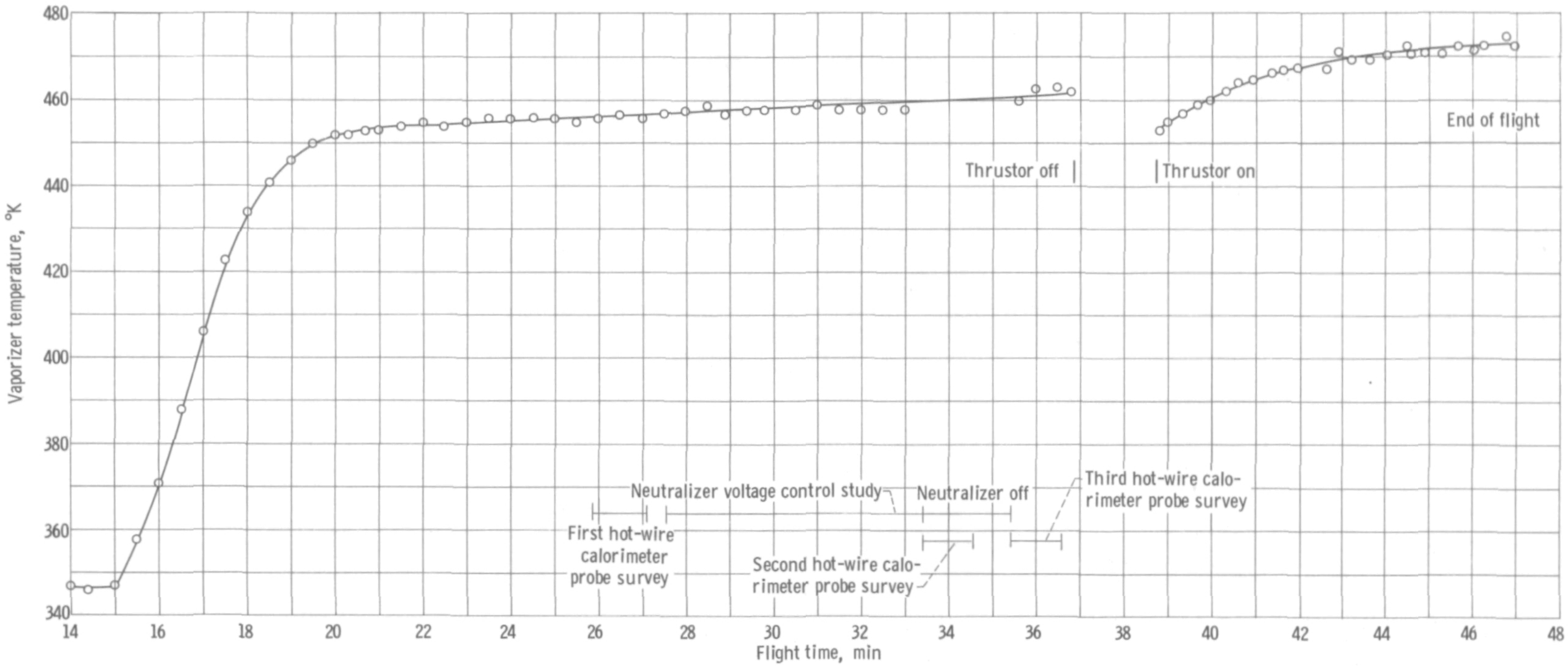


Figure 8. - Propellant vaporizer temperature as function of time. SERT I flight.

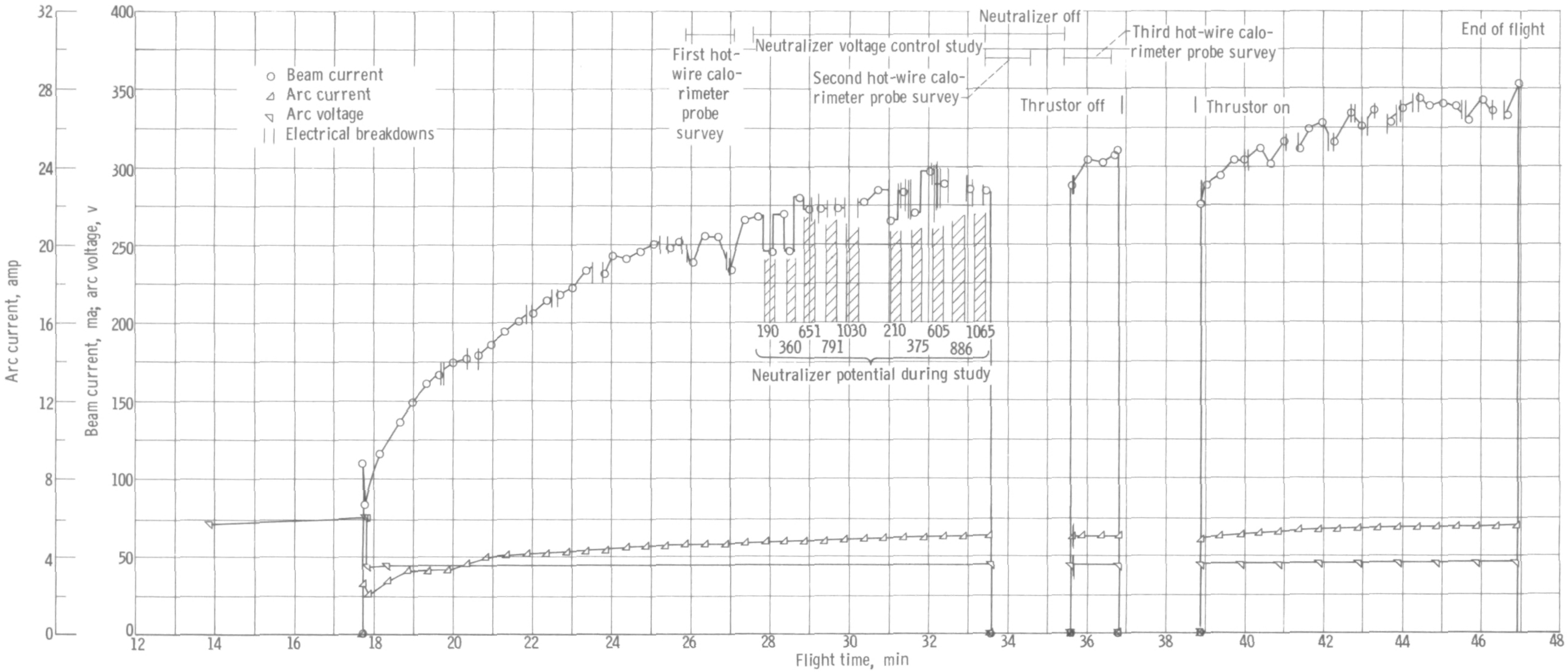


Figure 9. - Arc current and voltage and ion beam current. SERT I flight.

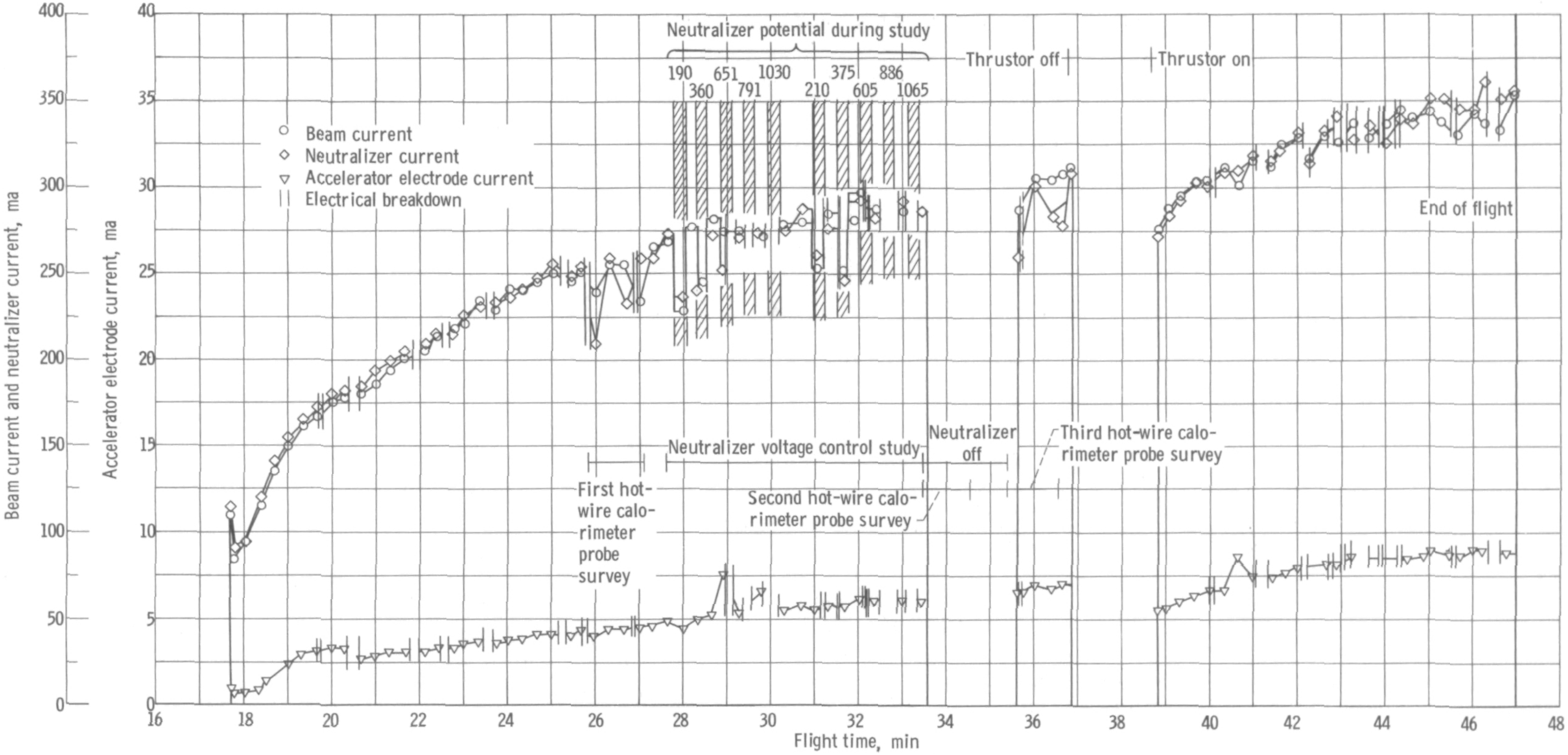


Figure 10. - Beam current, neutralizer current, and accelerator electrode current. SERT I flight.

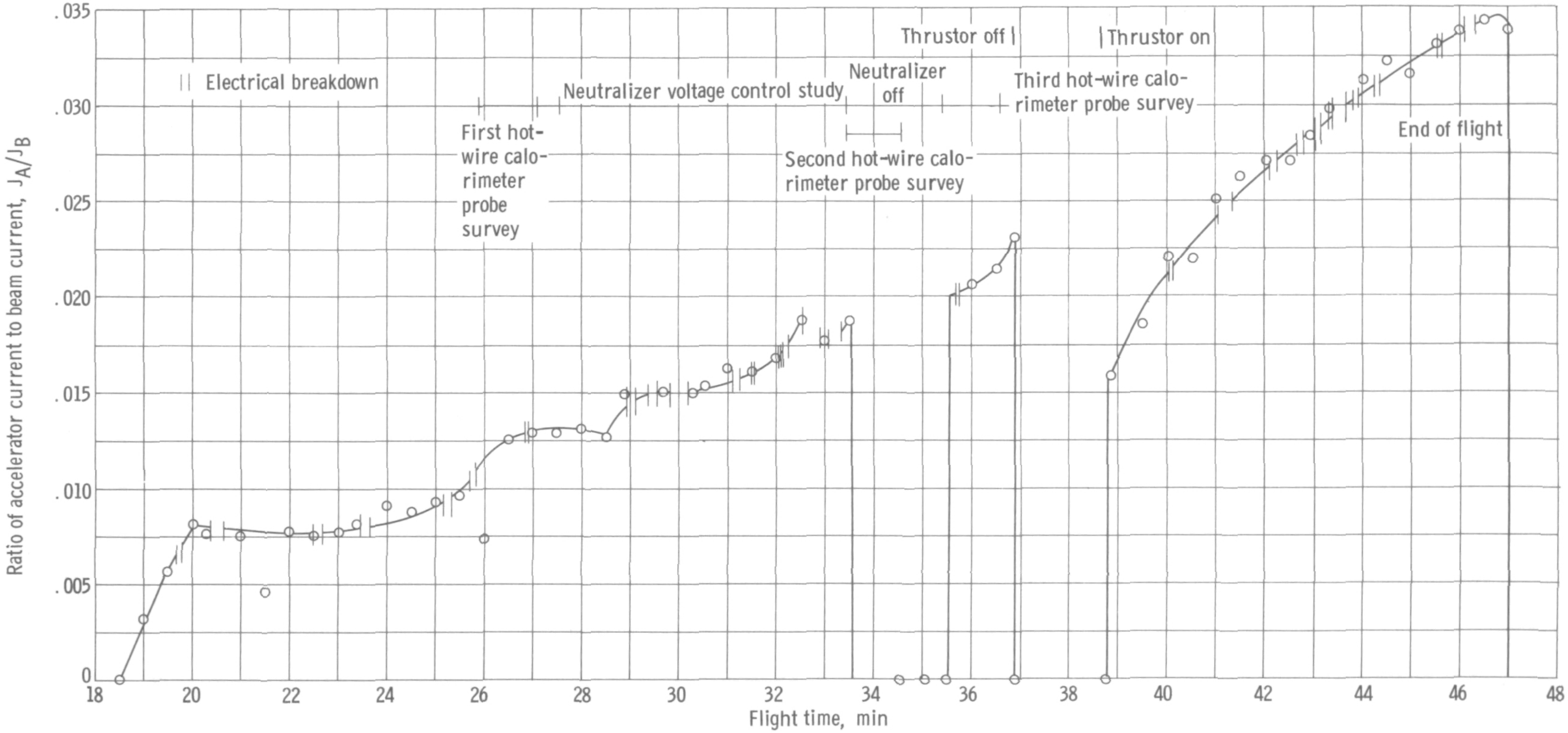


Figure 11. - Ratio of accelerator current to ion beam current. SERT I flight.

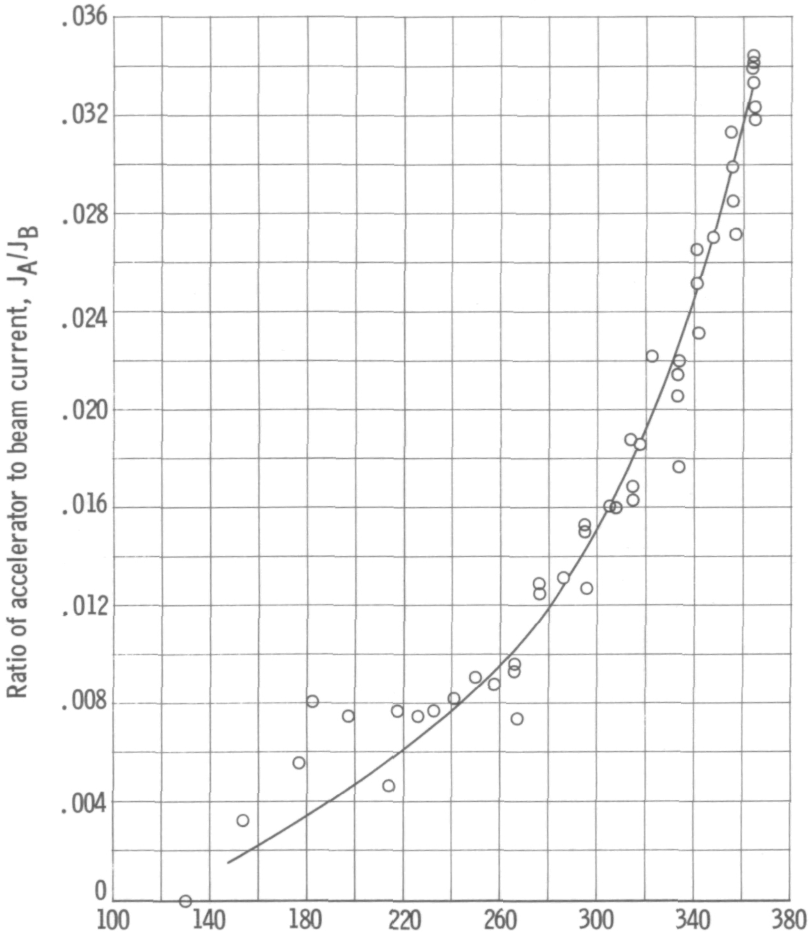


Figure 12. - Ratio of accelerator to ion beam current as function of ion beam current. SERT I flight.

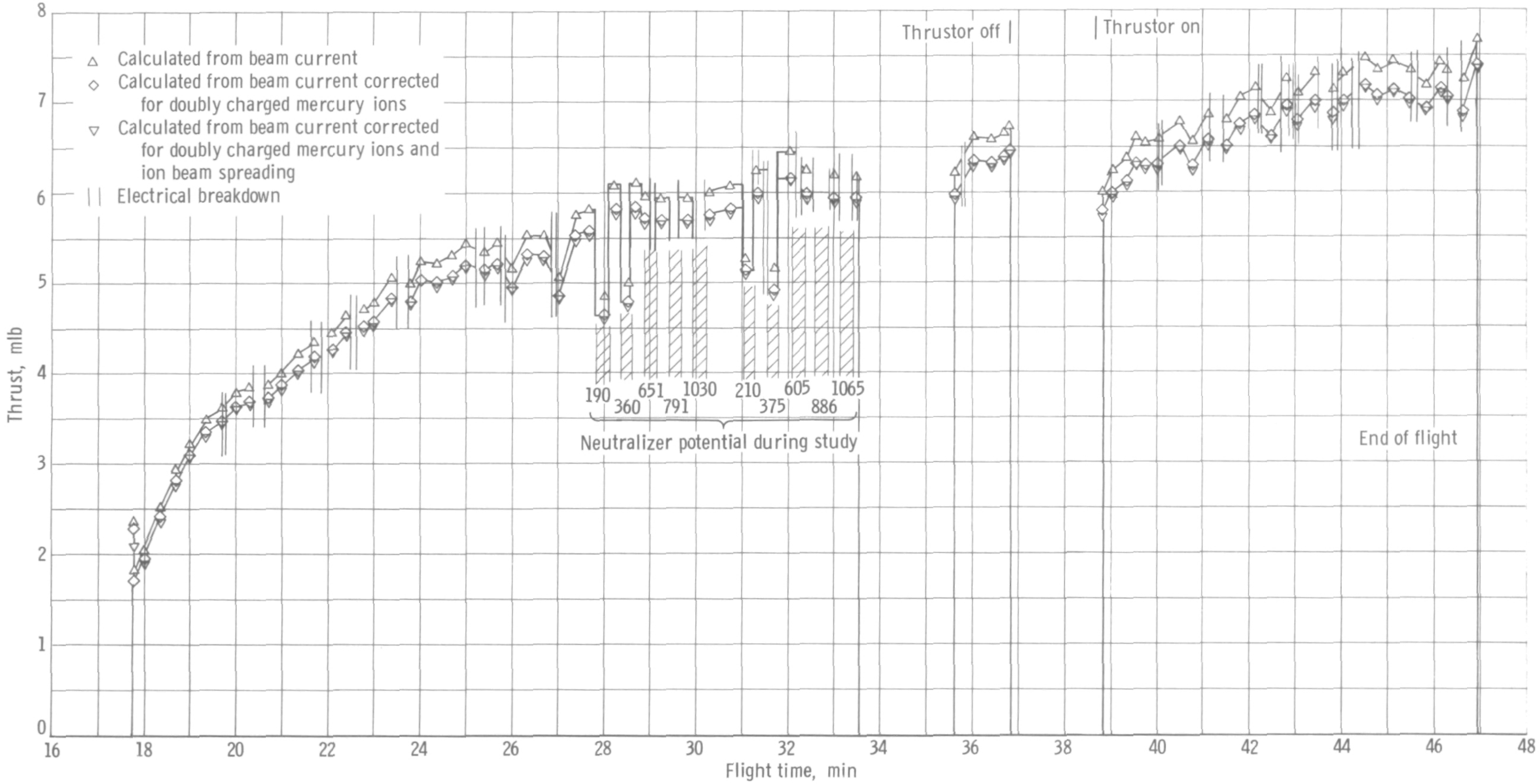


Figure 13. - Thrust computed from ion beam current and voltage measurements. SERT I flight.

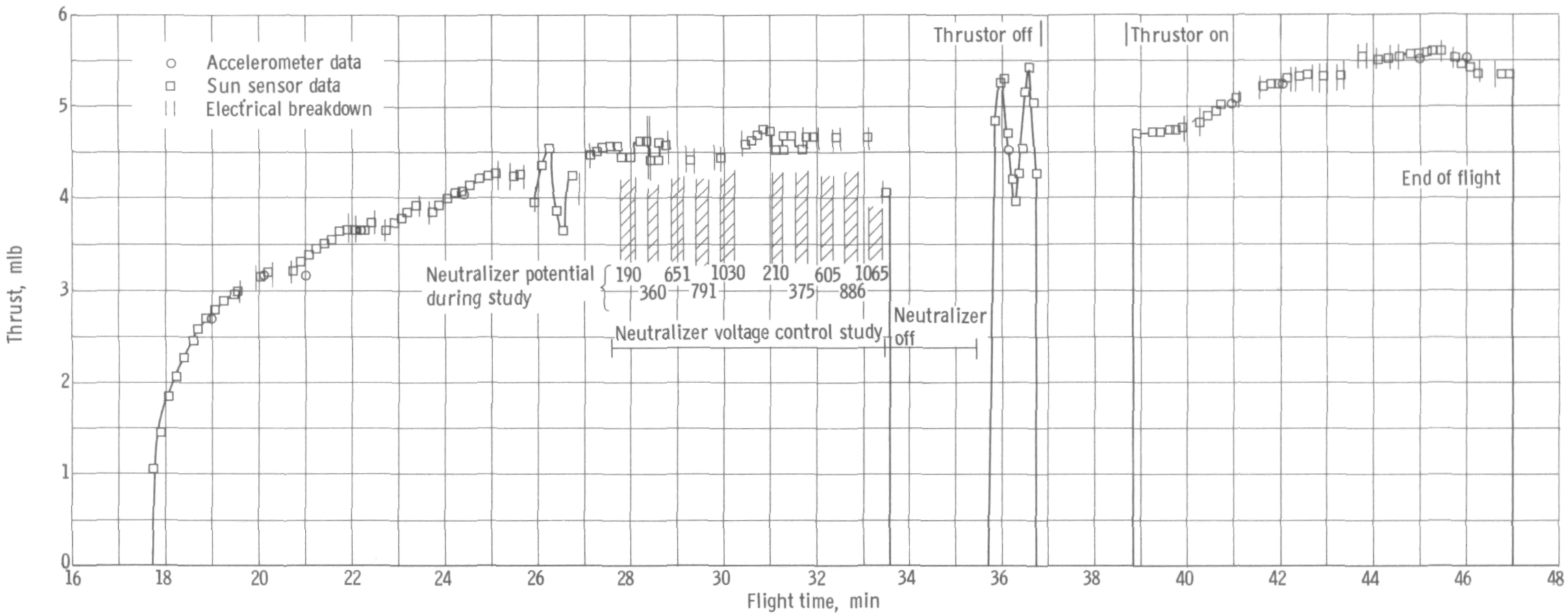


Figure 14. - Thrust computed from sun sensor and accelerometer data. SERT I flight.

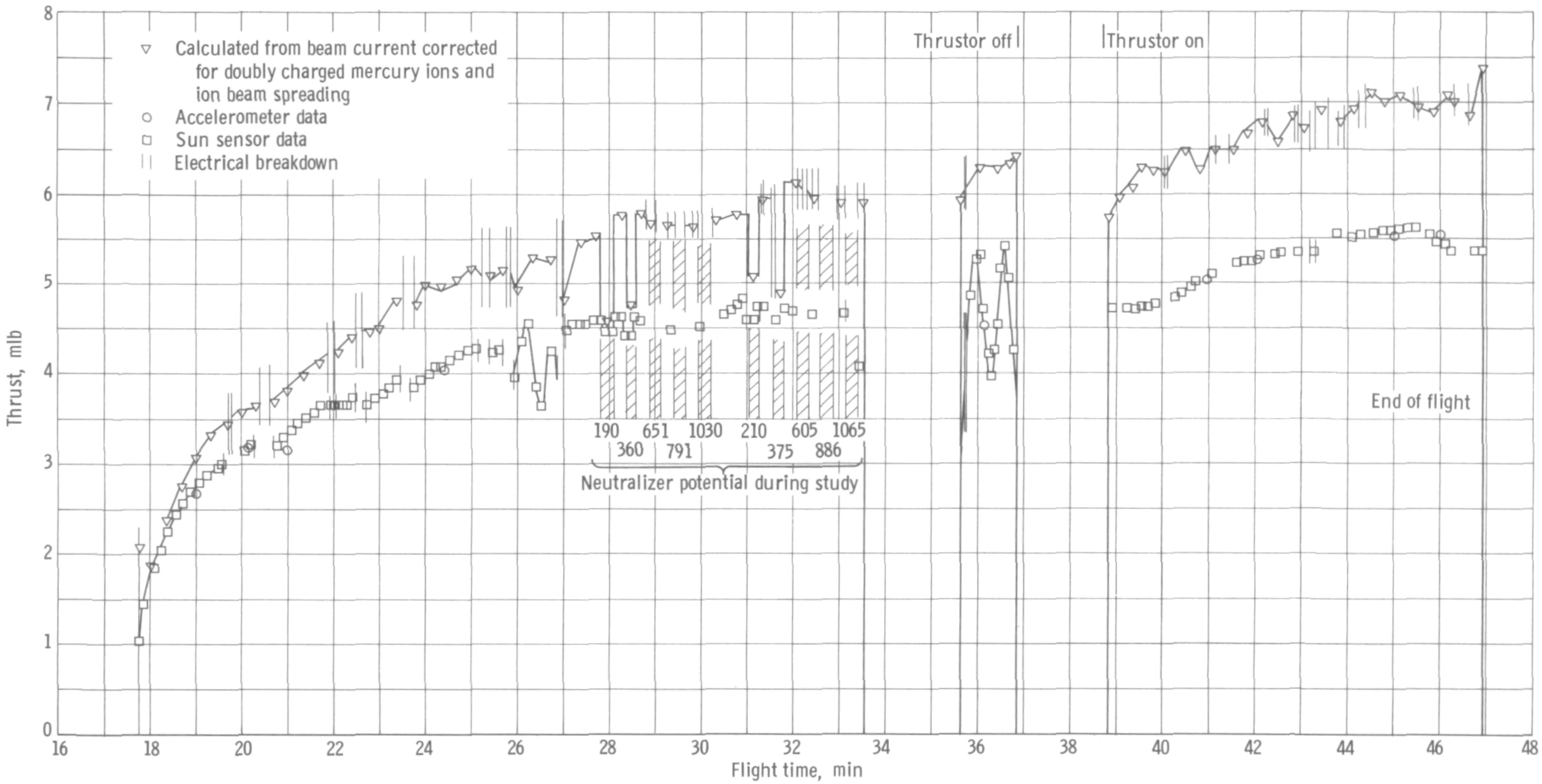


Figure 15. - Thrust computed from ion beam current and voltage measurements and sun sensor and accelerometer data. SERT I flight.

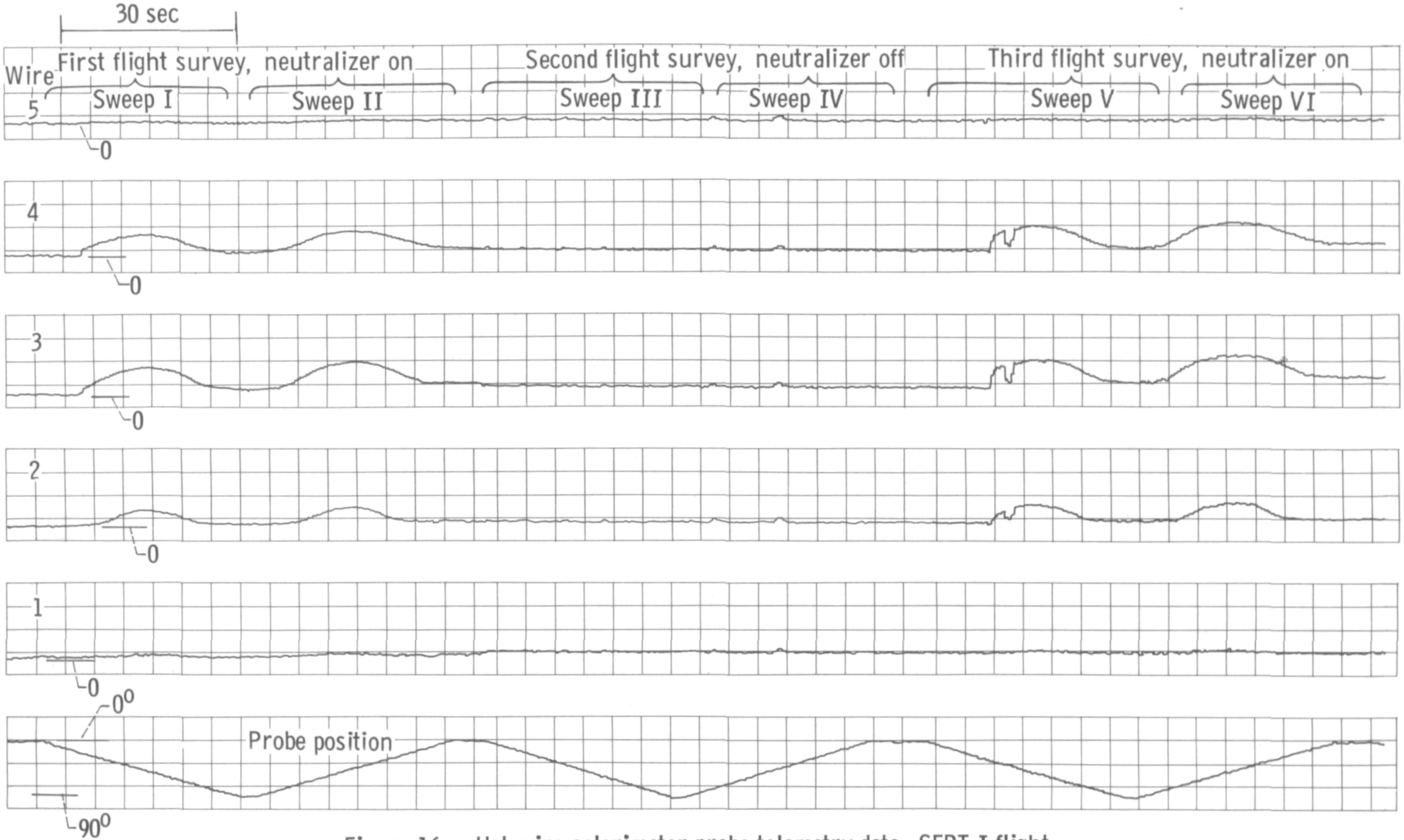
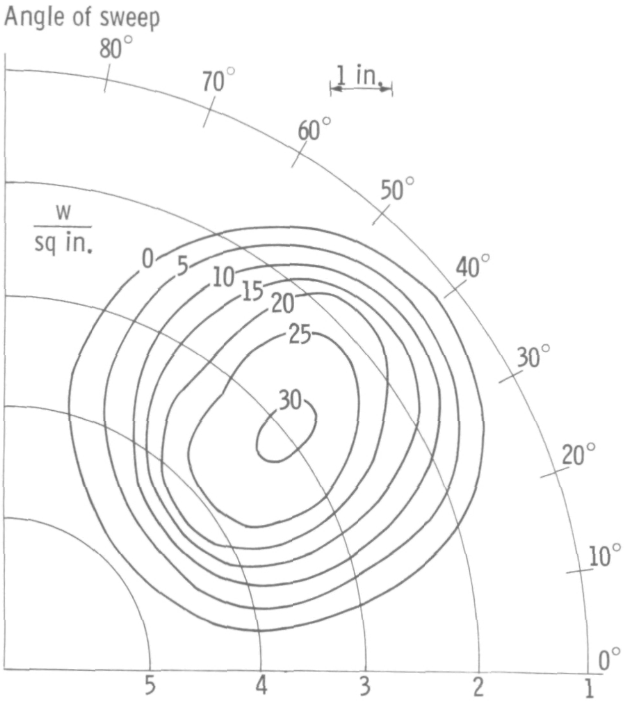
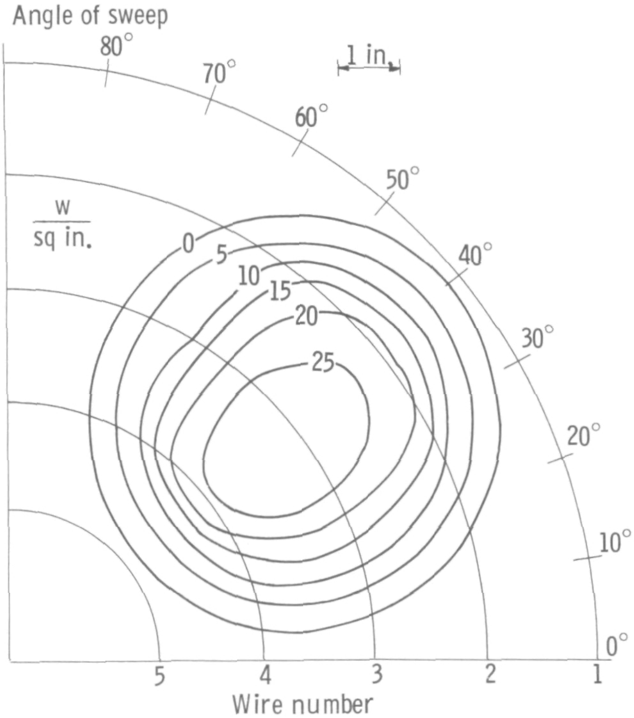


Figure 16. - Hot-wire calorimeter probe telemetry data, SERT I flight.

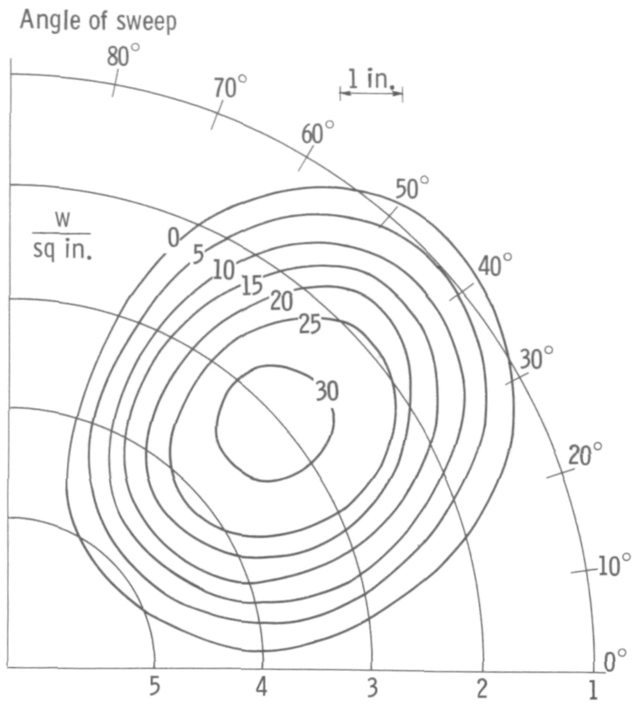


(a) First half of first probe survey.

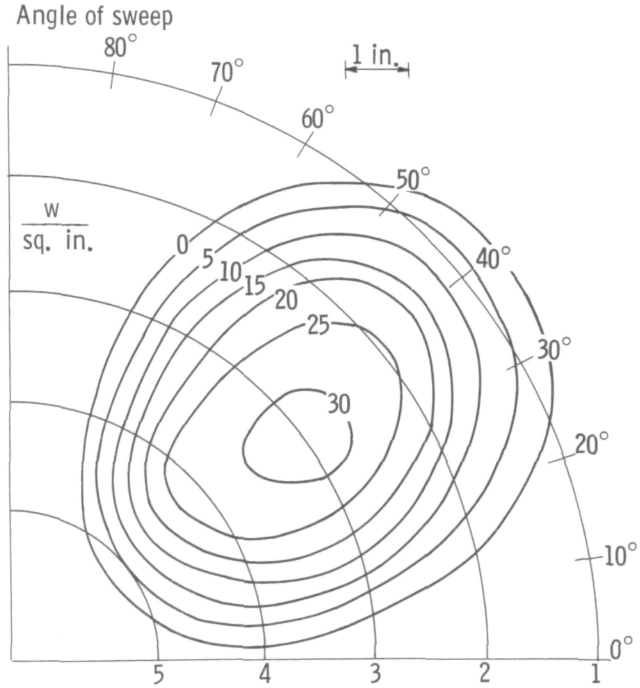


(b) Second half of first probe survey.

Figure 17. - Power density distribution obtained with hot-wire calorimeter probe during first survey. SERT I flight.



(a) First half of third probe survey.



(b) Second half of third probe survey.

Figure 18. - Power density distribution obtained with hot-wire calorimeter probe during third survey. SERT I flight.

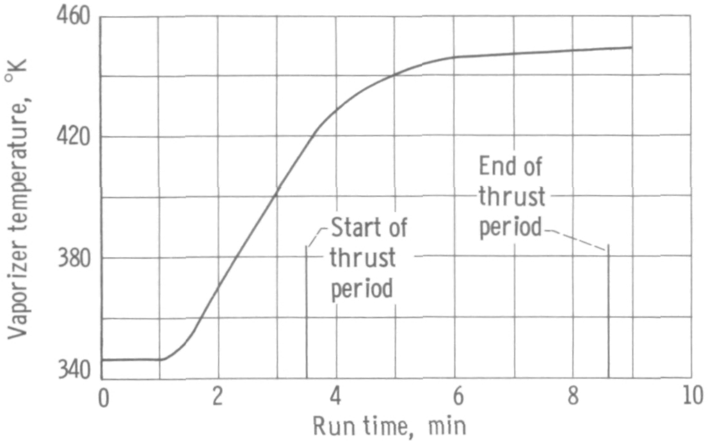


Figure 19. - Propellant vaporizer temperature. Tank test of 6-7-64.

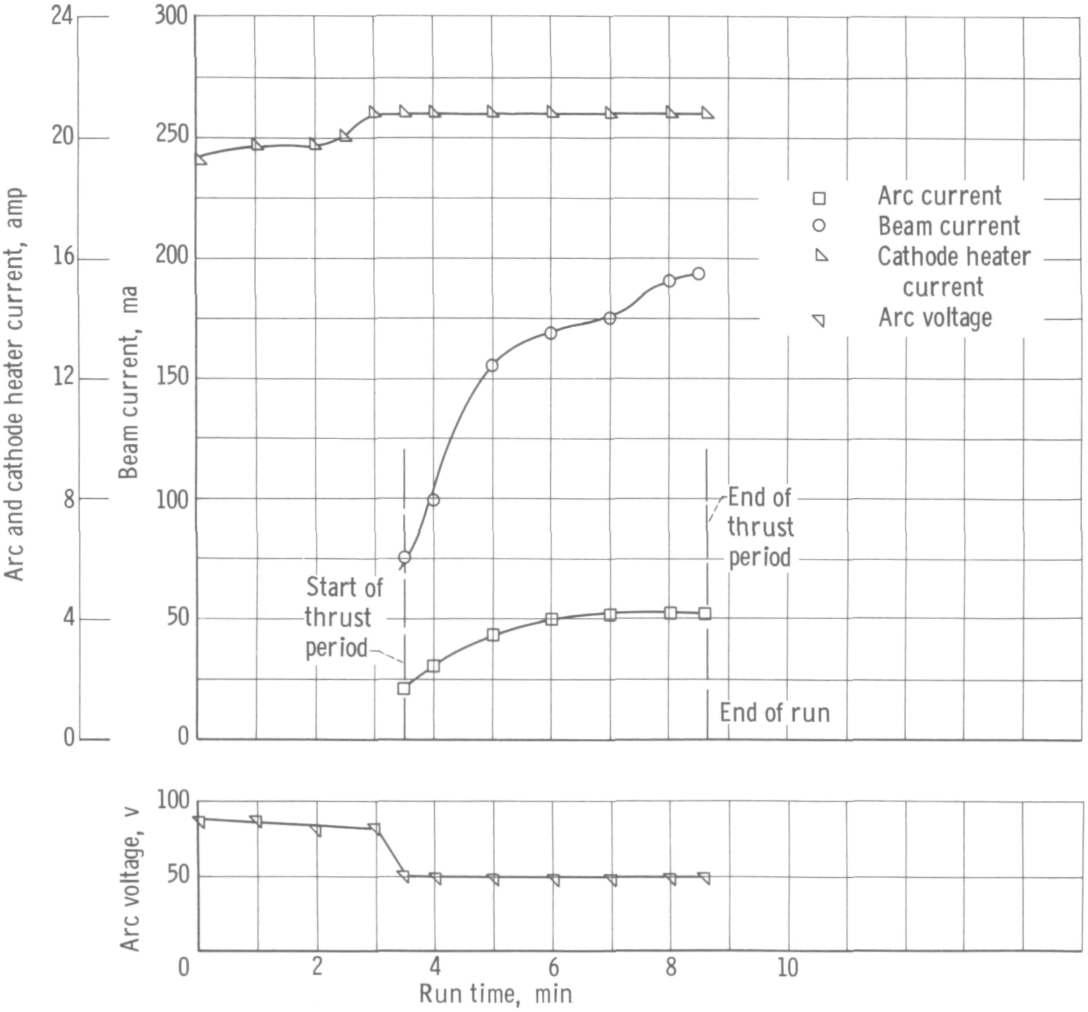


Figure 20. - Ion thrustor voltage and currents. Tank test of 6-7-64.

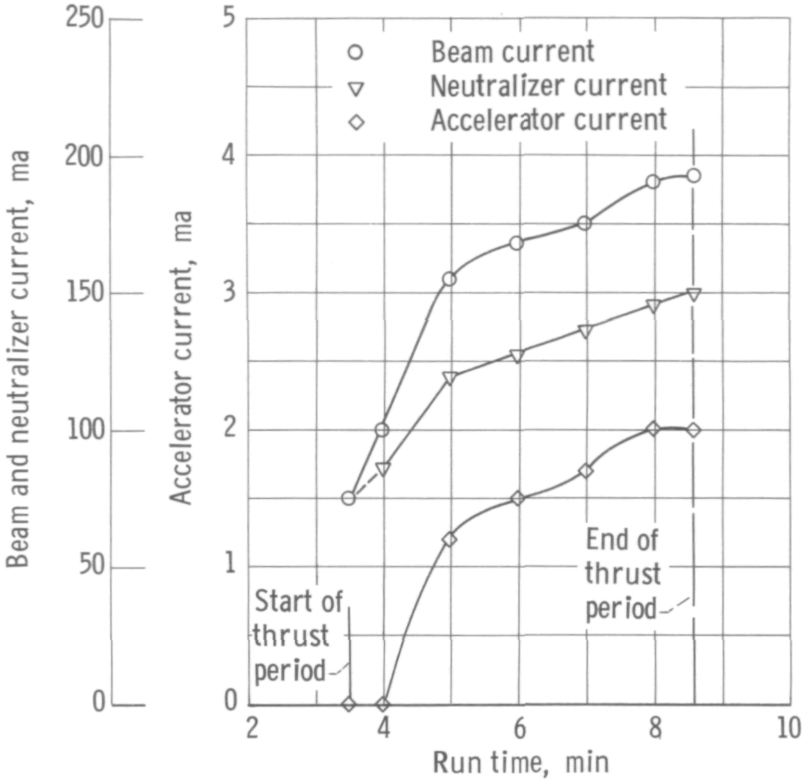


Figure 21. - Beam, neutralizer, and accelerator currents. Tank test of 6-7-64.

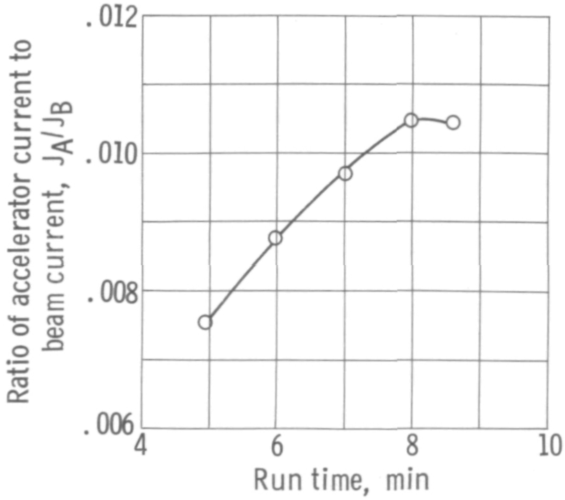


Figure 22. - Ratio of accelerator impingement current to ion beam current. Tank test of 6-7-64.

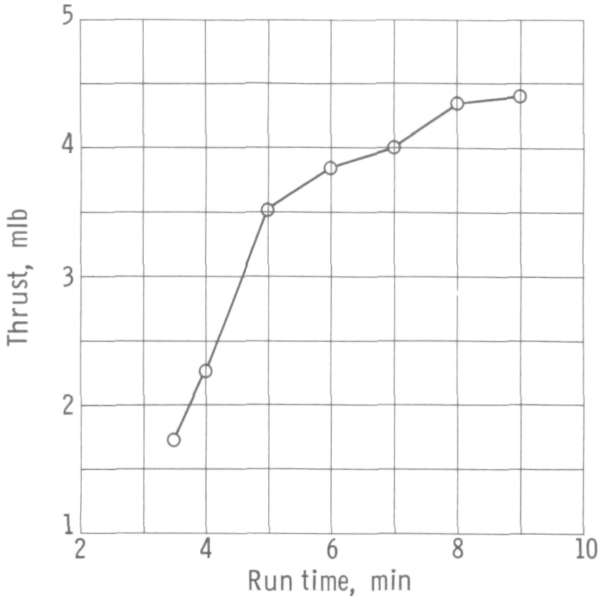


Figure 23. - Ion beam thrust calculated from beam current and net accelerating voltage. Tank test of 6-7-64.

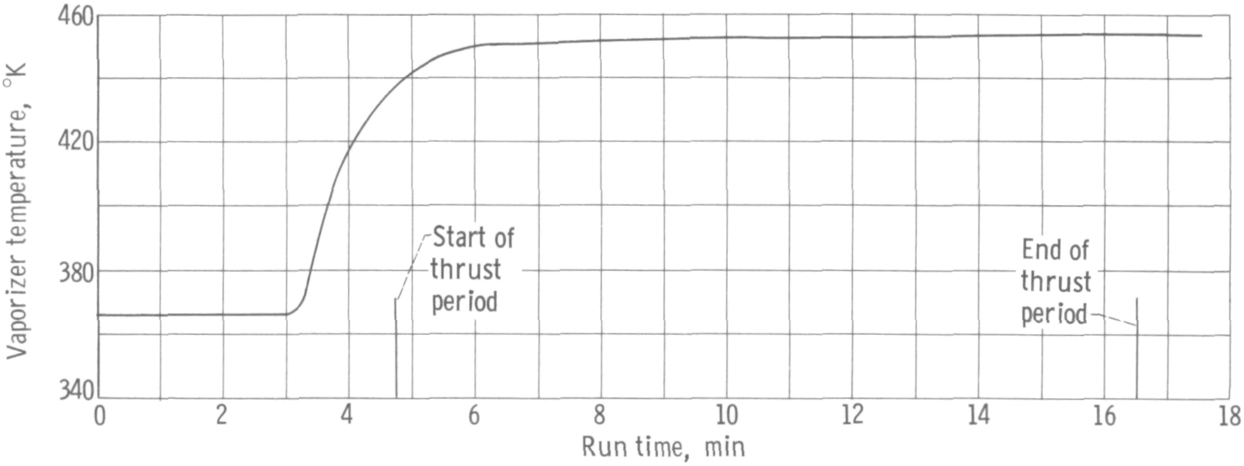


Figure 24. - Propellant vaporizer temperature. Tank test of 6-9-64.

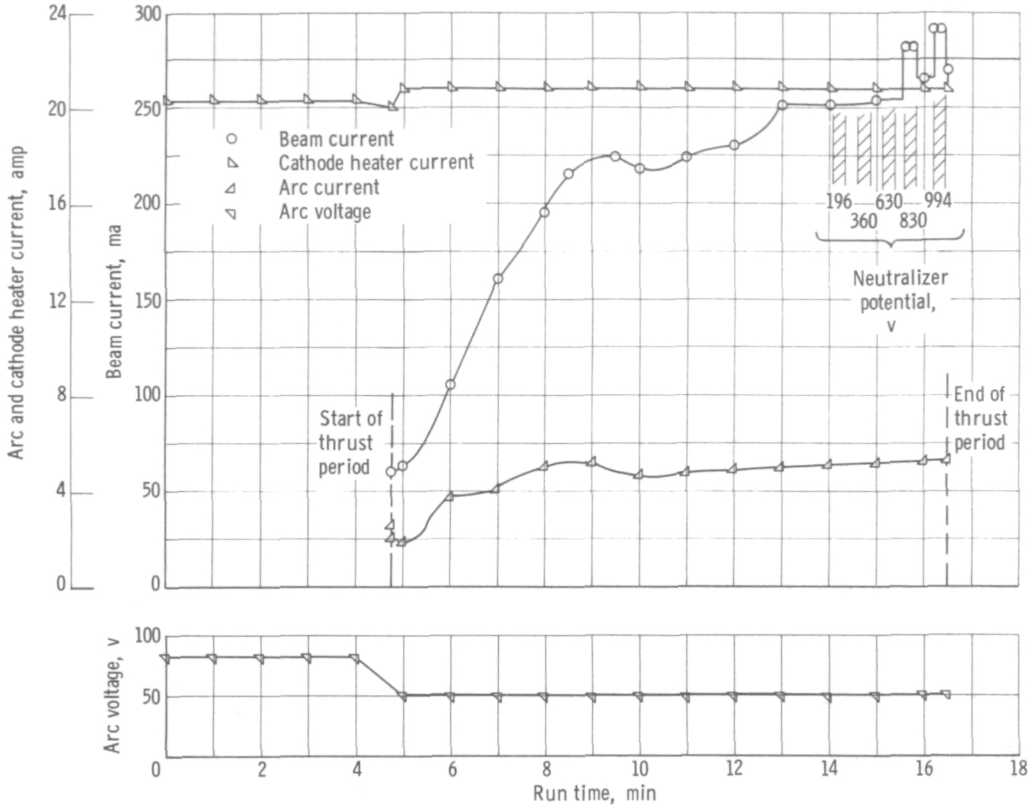


Figure 25. - Ion thrustor voltage and currents. Tank test of 6-9-64.

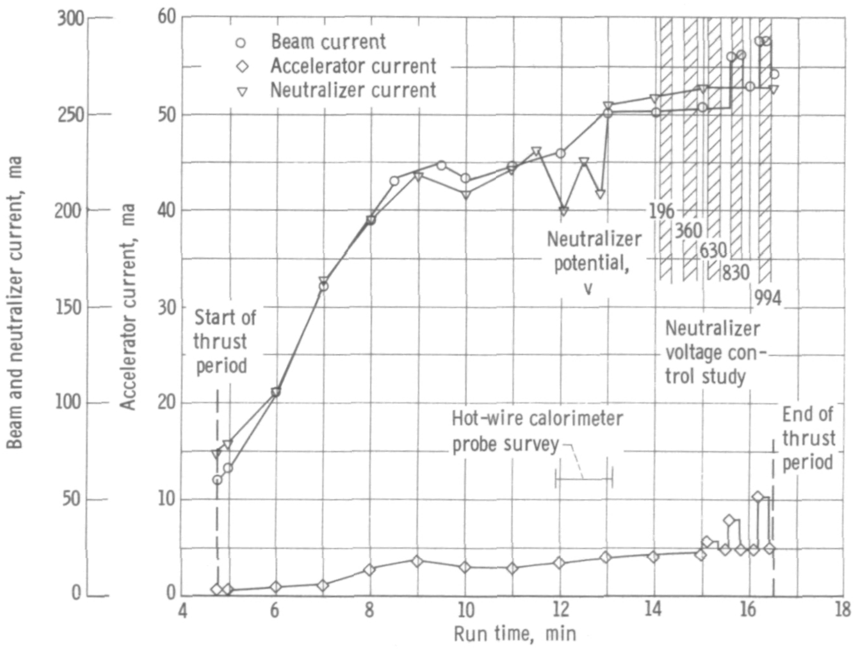


Figure 26. - Beam, neutralizer, and accelerator currents. Tank test of 6-9-64.

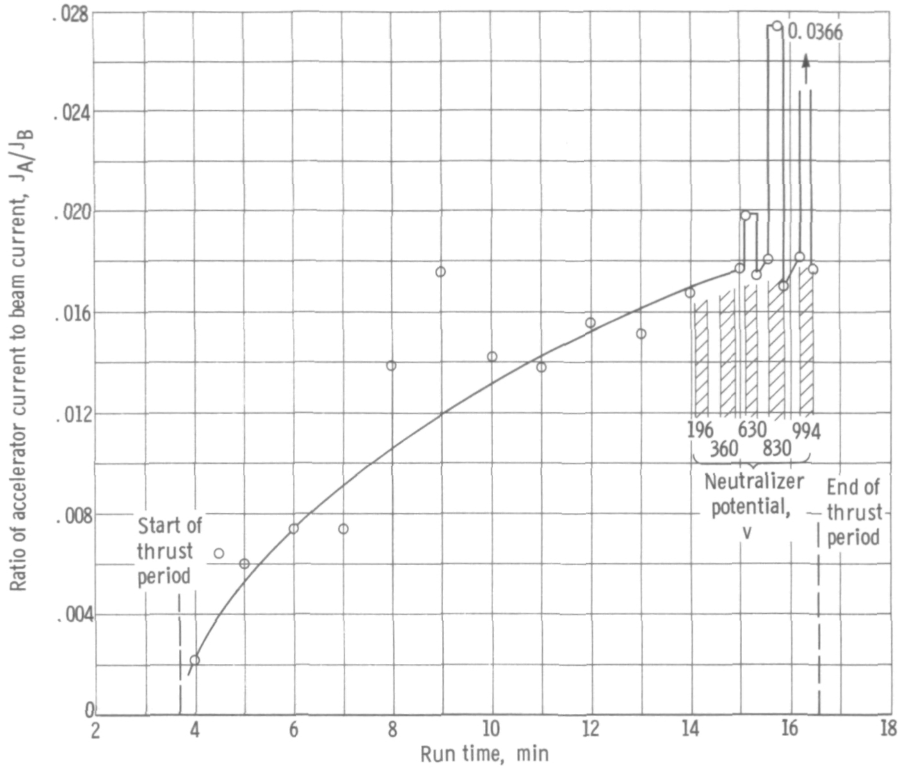


Figure 27. - Ratio of accelerator impingement current to ion beam current. Tank test of June 9, 1964.

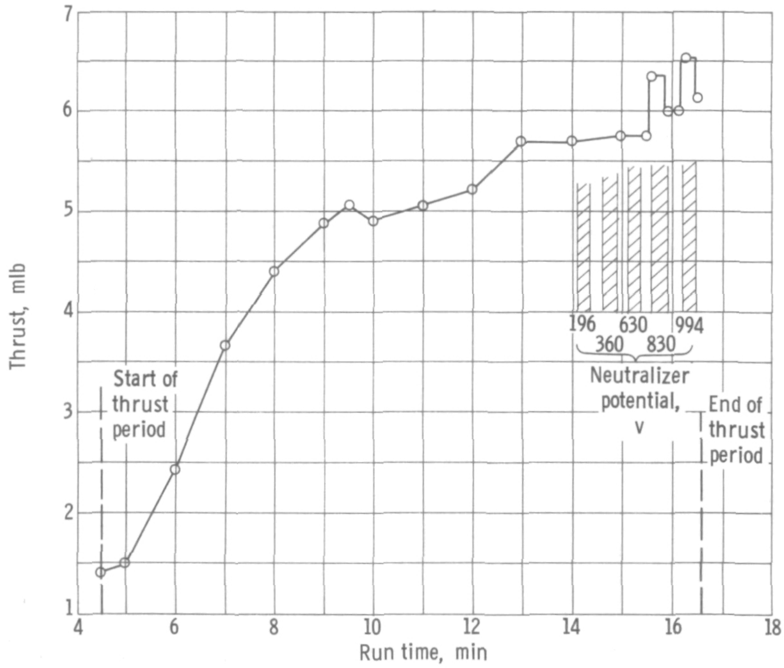
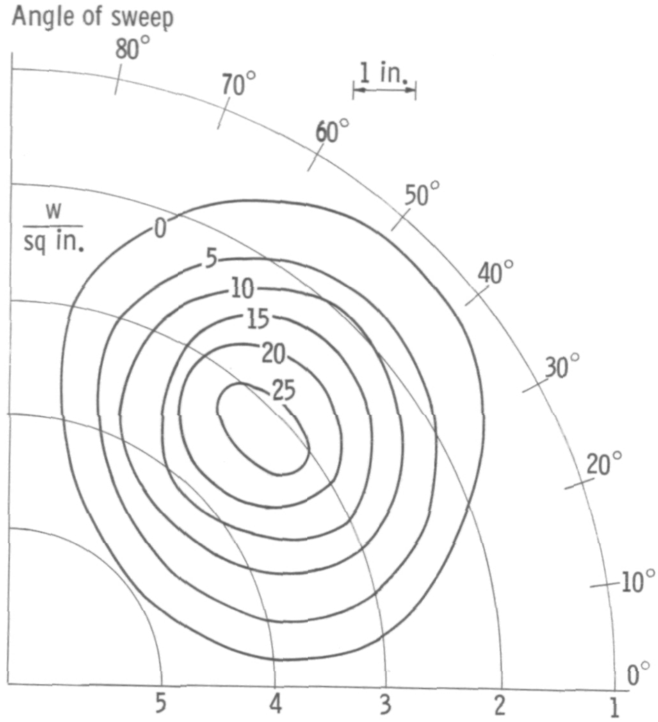
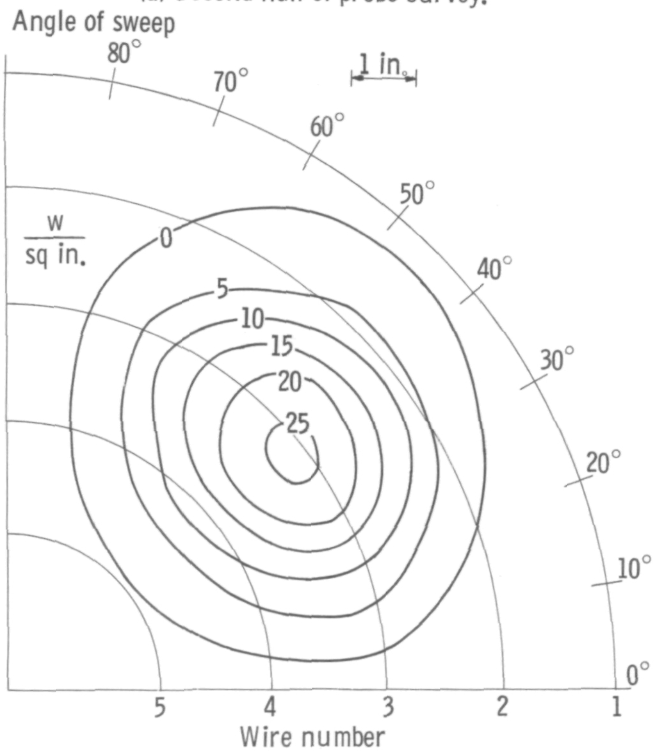


Figure 28. - Ion beam thrust calculated from beam current and net accelerating voltage. Tank test 6-9-64.



(a) Second half of probe survey.



(b) First half of probe survey.

Figure 29. - Power density distribution obtained with hot-wire calorimeter probe during tank test of 6-9-64.