NASA TECHNICAL MEMORANDUM

NASA TM X-1077

NASA TM X-1077

N65-19272
(ACCESSION NUMBER)
19
(PAGES)
(NASA CR OR TMX OR AD NUMBER)
(THRU)
1
(CODE)
31
(CATEGORY)

DESCRIPTION AND OPERATION OF SPACECRAFT IN SERT I ION THRUSTOR FLIGHT TEST

by Harold Gold, Raymond J. Rulis, Frank A. Maruna, Jr., and William H. Hawersaat

Lewis Research Center
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • MARCH 1965

NASA TM X-1077

DESCRIPTION AND OPERATION OF SPACECRAFT IN SERT I ION THRUSTOR FLIGHT TEST

By Harold Gold, Raymond J. Rulis, Frank A. Maruna, Jr., and William H. Hawersaat

Lewis Research Center
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

For sale by the Office of Technical Services, Department of Commerce, Washington, D.C. 20230 -- Price $1.00

DESCRIPTION AND OPERATION OF SPACECRAFT IN SERT I ION THRUSTOR FLIGHT TEST*

by Harold Gold, Raymond J. Rulis, Frank A. Maruna, Jr., and William H. Hawersaat

Lewis Research Center

SUMMARY

19272

The SERT I spacecraft was flown into a ballistic trajectory on July 20, 1964, by the Scout launch vehicle. The trajectory provided an experimental period of 47 minutes, during which the altitude of the spacecraft was above 250 nautical miles. The spacecraft carried two small ion thrustors and telemetered measurements of all major thrustor operating parameters. One of the thrustors operated for approximately 30 minutes, during which time the measured thrust verified the establishment of a neutral ion beam.

author

INTRODUCTION

The SERT I flight was the first space test in the ion propulsion development program of the National Aeronautics and Space Administration. The primary objective of the flight was to verify the achievement of a neutralized ion beam and the consequent development of thrust.

The fundamental processes in ion thrustors are production of propellant ions, coherent acceleration of ions by electric fields, and neutralization of the discharge beam. Of the first two processes, much was known 6 years ago at the outset of thrustor development. Furthermore, the problems involved in these processes, such as ionization

---

*This report was given limited distribution in August 1964. The flight data are presented in detail in NASA Technical Note D-2718, "Results from SERT I Ion Rocket Flight Test," by Ronald J. Cybulski, Daniel M. Shellhammer, Robert R. Lovell, Edward J. Domino, and Joseph T. Kotnik.

efficiencies and electrode life, can be experimentally investigated in vacuum chambers with reasonable certainty of results. The final and critical process of ion beam neutralization was not investigated prior to the development of ion thrustors. Although beam neutralization has been achieved in vacuum chambers, the uncertain effects of electron emission from chamber walls and of residual gas molecules on the neutralization process has made these results subject to doubt. The status of ion propulsion technology at the time of the flight reported herein is given in references 1 and 2.

Development of the SERT I spacecraft was begun in the middle of 1961 under the management of the Marshall Space Flight Center. At the end of 1961, management was transferred to the Lewis Research Center. During the continuous period from January 1963 to June 1964 extensive tests of the ion thrustors, their high-voltage power converters, and all spacecraft systems were carried out in the Lewis 15- by 60-foot vacuum chamber. Final qualification tests of the flight spacecraft were also performed in this chamber.

The purpose of this report is to present at an early date those significant results that could be interpreted without detailed analysis or data reduction. The data will be refined subsequently and a detailed analysis made.

The solution to the many technical problems that arose during the nearly 3-year period of development of the SERT I spacecraft has required the efforts and skills of many groups and individuals. The members of the Lewis staff who contributed significantly and the principal area to which they contributed are tabulated as follows:

Contributor	Area
James F. Bell	Data Reduction
Martin J. Conroy	Ground Support
Ronald J. Cybulski	Thrustor Systems
Edward J. Domino	Telemetry
Harold Gold	Project Manager
Guy S. Gurski	Mechanical Systems
William H. Hawersaat	Test Director
Robert M. Jabo	Environmental Test
Charles W. Knoop	Power Converters
Robert H. Kuhnapfel	Spacecraft Reliability and Quality Assurance
J. Thomas Kotnik	Power Converters
Vincent R. Lalli	Component Reliability and Quality Assurance
Robert R. Lovell	Thrust Detection Systems

James J. Pelouch, Jr.	Component Reliability and Quality Assurance
Frank A. Maruna, Jr.	Spacecraft Contract Technical Monitor
William C. Nieberding	Thrust Readout System
Raymond J. Rulis	Spacecraft Manager
George R. Sharp	Mechanical Design
Daniel M. Shellhammer	Beam Probe
Joseph B. Talbot	Vehicle Coordination
David R. Van der Cook	Telemetry
Ralph J. Zavesky	Mechanical Design

SPACECRAFT

The general configuration of the spacecraft during launch and in free flight is shown in figures 1(a) and (b), respectively. As depicted, the spacecraft is separated from the Scout fourth stage after trajectory insertion, and the thrustors are deployed outward. The spacecraft is spin stabilized with the spin induced by the fourth-stage rocket. The thrustors, which are oriented to apply a torque about the spin axis, are operated alternately, and thrust is detected from a measurement of the changes in the spacecraft total angular momentum. A photograph of the spacecraft in the free-flight configuration is shown in figure 1(c).

The basic support structure consists of a flat, circular baseplate supported on a cylindrical pedestal. The baseplate has a ribbed understructure and is machined from a forged magnesium billet; the supporting pedestal is also machined from a magnesium billet. The pedestal is clamped to a conical magnesium adapter that mates the spacecraft to the Scout fourth stage. The adapter-pedestal clamp is opened by firing explosive bolts to separate the spacecraft from the Scout fourth stage.

A welded aluminum box frame is mounted on the top center of the baseplate. In this structure and in the pedestal below it are mounted the basic spacecraft gear: the programer, the power distributor, the telemetry signal conditioning and switching gear, and the command receiver. The heavy components, such as batteries and power converters, are mounted on both sides of the baseplate beside the central frame and pedestal. This mass distribution provides a dominant roll axis moment of inertia. The thrustor mounting arms are hinged near the outer edge of the baseplate. The deployment linkage is locked to the central pedestal and is released by an explosively actuated latch to permit outward deployment of the thrustors. The deployment is centrifugally actuated with the rate limited by hydraulic dampers. The weight of the spacecraft is 375 pounds.

A small separation velocity between the spacecraft and the vehicle is imparted by a helical spring that is coaxial with the spacecraft spin axis and is located in the conical adapter. Cancellation of this velocity differential by motor thrust tail off is prevented by a thrust misalining device that consists of a single weighted cable that is wrapped to provide motor casing despin and tumble upon release. The cable is released a few seconds after spacecraft separation. Precession due to separation disturbances is suppressed by sliding weight dampers.

ION THRUSTORS

Basic Characteristics

The nominal thrustor characteristics and operating parameters of the two thrustors are shown in the following table:

	Electron-bombardment thrustor	Contact-ionization thrustor
Ion-beam outside diameter, in.	3.9	3.2
Overall thrustor diameter, in.	7.5	4.0
Thrustor weight, lb	9.3	14
Propellant	Mercury	Cesium
Specific impulse, sec	4900	8050
Total input power, w	1400	610
Power efficiency, percent	48.5	32.2
Propellant-utilization efficiency, percent	80	96
Thrust, mlb	6.4	1.25
Beam current, ma	275	45

Contact-Ionization Thrustor

A schematic diagram of the contact-ionization thrustor is shown in figure 2(a). This thrustor was developed under NASA contract by the Hughes Research Laboratories. Cesium vapor flows from an electrically heated boiler through an electrically heated porous tungsten ionizer. The porous tungsten also serves as the propellant-flow-control restriction. Propellant feed control is accomplished through regulation of the boiler temperature. A solenoid valve, which is between the boiler and the ionizer, is employed for turnon or turnoff of the propellant flow. The electrode array consists of focus, accelerator, and decelerator electrodes. The focus electrode is held at a positive potential of 4500 volts above the spacecraft potential in common with the ionizer. The accelerator electrode is held at 2000 volts below the spacecraft potential, and the decelerator electrode remains at the spacecraft potential.

The thrustor carries two beam-neutralizing systems that are programed to operate alternately. One neutralizing system consists of a tantalum filament that thermally emits electrons into the ion beam just downstream of the decelerator electrode. The second neutralizing system consists of an electrode controlled electron gun that injects electrons into the ion beam downstream of the accelerator electrode. During operation of this system, the gun-emitter potential with respect to the ion-beam decelerator potential and the control-electrode-to-emitter potential are slowly varied over a range of 0 to 50 volts. The object of this variation is to map the conditions over which an electron trap can be established.

In order to reduce the entrainment and adsorption of gases in the propellant feed system and in the tungsten ionizer during the launch period, the thrustor assembly is mounted to the spacecraft in an evacuated pod. The pod is opened in space by ejection of the pod cap at the electrode end. A photograph of the thrustor taken during vacuum-chamber operation that shows the open pod configuration is presented in figure 2(b). The pod enclosure permits the thrustor to operate very early in the flight, and consequently this thrustor is programed to operate during the first half of the flight period.

Electron-Bombardment Thrustor

A cutaway model of the electron-bombardment thrustor is shown in figure 3(a), and an electrical schematic diagram is shown in figure 3(b). This thrustor was invented by Harold R. Kaufman of the NASA Lewis Research Center. Mercury vapor flows from an electrically heated boiler into the ionization chamber. The rate of propellant feed is controlled by a porous stainless-steel plug through regulation of the boiler temperature. A circular baffle plate is located just downstream of the plug to induce uniform distribution of the mercury vapor. The bombarding electrons are emitted from a tantalum filament cathode and are attracted to a cylindrical shell anode that is 50 volts positive with respect to the cathode. The anode shell and the cylindrical ionization chamber wall are coaxial with the thrust axis. A coaxial magnetic field is generated by a coil that is wound around the ionization chamber. The magnetic field causes electrons to move from the cathode to the anode in a complex path and thereby increases the probability of collision with mercury atoms. A perforated screen electrode covers the downstream end of the ionization chamber. The ionization chamber and the screen electrode are maintained at a potential of 2500 volts above the spacecraft potential, and the accelerator electrode is maintained at 2000 volts below the spacecraft potential. As in the case of the contact-ionization thrustor, the boiler temperature is regulated to maintain the propellant feed rate slightly below the value that would produce a space-charge-limited ion-beam current.

The beam neutralizing system consists of an electrically heated tantalum filament that is partly immersed in the beam just downstream of the accelerator electrode.

As shown in figure 3(b), the neutralizer filament is heated by current from a low-voltage battery. In the normal operating mode, the battery is connected to the spacecraft ground. Resistance is inserted into the ground line at programed intervals so that the effect of neutralizer potential may be studied.

The flight model of the thrustor is shown in figure 3(c). The external screening, which is grounded to the spacecraft, functions to block the flow of electrons from the neutralizer filament to the walls of the ionization chamber.

Power Supplies

The electrical power system for thrustor operation consists of one storage battery and a separate converter for each thrustor. The power converter for the contact-ionization thrustor supplies all power to the thrustor. The electron-bombardment thrustor utilizes two additional batteries, which feed the magnetic field coil and the neutralizer filament directly. The telemeter system is powered by a separate battery. All batteries are multicell silver-zinc. The thrustor battery is a 56-volt assembly housed in two sealed magnesium cases. The magnetic field and the neutralizer filament batteries are 6- and 10-volt assemblies, respectively. Because of the high voltage level upon which the two direct-feed bombardment-thrustor batteries float, these assemblies consist of sealed cells housed in fiber-glass cases.

The two thrustor power converters, which are similar in method of operation, use transistor choppers, transformer voltage amplification, and solid-state rectifiers. The chopper transistors are driven by control oscillators through which voltage and current are regulated. The battery is directly connected to the chopper transistors. Converter turnon and turnoff are controlled by switching power to the control oscillators. The two direct-feed bombardment-thrustor batteries are switched by high-voltage relays that are encapsulated in the fiber-glass cases.

For arcing transient protection, the converters incorporate electrostatic shielding in transformers, breakdown-diode overvoltage protection for chopper transistors, and output current limiting through feedback and through current limiting resistors between the converter and thrustor. For protection against arcing and corona discharge within the converters, all high-voltage sections are pressurized. Conductors terminating at high-voltage points on the converters are covered with molded insulation at the junction Wire connections to the contact-ionization thrustor are made through bare sleeve-to-pin joints, which are mechanically held in structures that permit rapid outgassing. Wire connections to the electron-bombardment thrustor are made through bare terminal boards. Conventional connectors are incapable of outgassing rapidly enough.

INSTRUMENTATION

Electrical Parameters

Spacecraft performance monitoring follows conventional telemetry practice. High-voltage converter parameters are measured in alternating current to permit transformer voltage reduction. For transmission of parameters, such as the magnetic field current in the electron-bombardment thrustor, which is a direct current of approximately 15 amperes flowing in a low-resistance circuit that is grounded at 2500 volts above telemetry ground, current transducers are used. Special instrumentation for diagnosis of thrustor performance consists of a hot-wire probe that sweeps across the discharge beam of the electron-bombardment thrustor to obtain the beam profile and an electric field meter to detect the presence and strength of electric fields surrounding the spacecraft.

Thrust-Measurement Systems

Three independent systems are employed for thrust measurement through spin-rate detection. Two of the systems are independent solar-cell spin-period detectors. The third system utilizes an accelerometer that is so mounted that it senses radial acceleration along an axis that passes through the spacecraft center of gravity.

Each solar-cell system employs a silicon photocell that is housed behind a narrow slit. The cells are mounted 180° apart on the periphery of the spacecraft with the slit parallel to the spin axis. Each solar cell generates a pulse per revolution that is transmitted through independent telemeter links. The received pulses are fed into clock-controlled electronic counters for period measurement.

The accelerometer system provides a frequency modulated output with a center frequency of approximately 200 cps that is fed directly to each of the two transmitters on board. At the receiving stations, the accelerometer frequency is extracted by a linear phase shift filter and fed into preset counters.

Through the associated changes in radial acceleration, thrust causes a drift in the accelerometer frequency, and spacecraft precession causes a sinusoidal frequency variation. For real time readout, an FM discriminator is used along with the preset counters. The output of the discriminator is displayed on a pen recorder.

TELEMETRY SYSTEM

The telemeter system for the SERT I spacecraft consists of two independent FM/FM systems, each transmitting at an output power of 10 watts. Each transmitter receives signals from three subcarrier oscillators and from the accelerometer. The subcarrier center frequencies are 1.7, 7.35, and 10.5 kilocycles. The subcarrier channels are utilized as follows:

Transmitter	Frequency, Mc	Subcarrier frequency, kc	Parameter
1	240.2	1.7 7.35 10.5	Command system data Solar-cell pulse Commutated (45 segments; 2 frames/sec)
2	244.3	1.7 7.35 10.5	Programer data Solar-cell pulse Commutated (45 segments; 2 frames/sec)

Critical thrustor electrical data are carried on both telemeter links.

COMMAND SYSTEM

The command system utilizes a 1700-watt AM transmitter feeding the Wallops Island Tiros-Kennedy antenna. Command receiver relay closure is provided upon reception of two audio tones, 1 second apart. Ten separate relays are provided. The command system provides programer function backup and modification and on-off control of the following thrustor subsystems:

(1) Contact-ionization thrustor: feed valve and boiler heater

(2) Electron-bombardment thrustor: magnetic field

RESULTS

Launch Vehicle and Spacecraft Performance

Trajectory. - The launch vehicle provided a trajectory that was close to the predicted one and a test time at altitudes above 250 nautical miles of 47 minutes. Spacecraft separation from the fourth stage occurred as programed, 2 minutes after burnout. Preliminary radar information indicated that following separation the fourth-stage motor casing followed a trajectory sufficiently different from that of the spacecraft to conclude that the thrust misalining device performed its function.

Spacecraft motion. - The spin rate of the spacecraft at separation was 106.2 rpm. Thrustor deployment reduced the spin rate to 87 rpm without a perceptible increase in precession angle. Throughout the flight, precession damping was evident through a reduction in precession angle of about 50 percent at the end of the flight.

Power supplies. - All battery power sources for both thrustors functioned normally throughout the flight and indicated ample reserve at the end of the experiment. The internal temperature of the power converter for the electron-bombardment thrustor showed a normal rise from 65° to 100° F at the end of the flight. A slight rise in internal pressure from 24.5 to 25 pounds per square inch absolute reflected the increase in operating temperature. The operating portions of the converter for the contact-ionization thrustor showed temperature rises that would have been within operational limits had a normal flight sequence been followed.

Thrust-measurement system. - All three independent systems for thrust measurement and the supporting ground station equipment, at both Wallops and Bermuda Stations functioned without apparent malfunction or drift throughout the flight. The output of the accelerometer FM discriminator as recorded in real time is shown on the sample trace of figure 4. As illustrated in the figure, the thrustor-off period in which the spin rate is constant results in a constant discriminator output voltage and hence a vertical line on the recorder. The angular acceleration of the spacecraft caused by the production of thrust is clearly evident in the sloping line on the recorder. The precession of the spacecraft is evident as a small sinusoidal signal on the recorder trace. In order to utilize a very high recorder gain for maximum thrust resolution, the precession signal was attenuated by a low pass filter with a break frequency of 0.1 cps. The response of this filter and of the recorder accounts for the apparent lag in response of thrust to ion-beam current. The actual response is essentially instantaneous.

Calculations performed to date show good agreement in spin rate and angular acceleration between sun sensor and accelerometer data.

Telemetry system. - The telemetry receiving antenna system used at Wallops Station during the flight consisted of a high-gain (29 db) and a medium-gain (17 db) antenna. Initially the 240.2-megacycle link utilized diversity combining and the 244.3-megacycle link utilized two receivers that were fed from a right- and a left-hand circularly polarized antenna, respectively. From the launch pad the received signal strength was 10,000 microvolts. The signal strength dropped to as low as 200 microvolts during first- and second-stage burning, and the signal was lost during third- and fourth-stage burning. At payload separation the signal level increased approximately 10 decibels and increased an additional 4 decibels after thrustor deployment. At approximately 7 minutes after lift-off all receivers were switched to right-hand circularly polarized antennas through which the best telemetry performance was obtained. Signal strength held at approximately 350 microvolts on 240.2 megacycles and 100 microvolts on 244.3 megacycles. There were no indications of signal degradation due to ion-thrustor operation.

The RF system maintained frequency and deviation throughout the flight. Carrier deviation of 50 kilocycles was employed on each link. Linear preemphasis was applied to all subcarrier signals except the accelerometer signal. The accelerometer signal deviated the transmitter 15 kilocycles. Under these conditions prelaunch checks indicated noise of 2 percent of full scale. This noise level was not exceeded in flight. The three subcarrier oscillators on each link maintained frequency stability and deviation within 4 percent. The commutators, both of which were mechanical, maintained speed within the tolerance, and the noise level was less than 2 percent of full scale. All instrumentation with the exception of two transducers operated properly throughout the flight. The loss of the two transducers, which was detected during the countdown, had no effect on the flight or on the interpretation of the data.

Command system. - The ground command transmitter carrier was activated prior to lift-off of the vehicle. The spacecraft command receiver was captured by this carrier and produced an automatic gain control (AGC) voltage which was read out through telemetry.

In flight, the spacecraft spin rate was detectable through modulation of the AGC voltage. During ion-thrustor operation, this spin modulation diminished, and the AGC voltage increased by approximately 20 percent of full scale. Preliminary analysis indicates that a noise voltage of unknown nature was produced at the time of thrustor operation, which predominated in producing the AGC voltage. Despite this noise voltage, however, there were no inadvertent commands. Throughout the flight the command system was activated 32 times and confirmed through telemetry in all instances.

Thrustor Performance

Contact-ionization thrustor. - Preheating of the tungsten ionizer functioned properly during a prelaunch phase and during boost. When the high voltage was turned on following pod opening, the 4500-volt positive potential supply indicated a high-voltage breakdown. In response to the breakdown, the high-voltage supply automatically turned the system off and 1 second later automatically restored the high voltage. Each time the high voltage was turned on, breakdown of the positive potential occurred at approximately 2000 volts. This cycling continued with approximately a 1-second period until the system action was terminated by ground command after 9 minutes and 11 seconds of attempted operation.

Because of the possibility that the cause of the high-voltage breakdown might have been eliminated after prolonged outgassing, the contact-ionization thrustor system was turned on for a second time, by ground command, after a period of 23 minutes. High-voltage breakdown was again indicated and the turn-on attempt was terminated after 2 minutes.

Electron-bombardment thrustor. - This thrustor-system operation was initiated through programer advance by ground command at 830 seconds after lift-off. All initial voltages and currents were normal. The boiler temperature rose at the expected rate and reached 265° F after 134 seconds following the turn on. At 134 seconds after the turn on, the magnetic field coil was momentarily deenergized by ground command in an attempt to achieve an ion beam as early as possible. The magnetic field coil was momentarily deenergized for four times prior to the beam initiation at 233 seconds after system turn on. At beam initiation the beam current was 100 milliamperes. An increase in spacecraft spin rate was detected through the radial accelerometer real time readout within seconds after the indication of beam current. For a period of 122 seconds after the indication of ion-beam initiation, the thrustor operated without interruption, during which period the beam current continuously increased as a consequence of a continuously increasing boiler temperature and propellant flow. At the end of the 122-second period, the thrustor system automatically shut down as a result of a momentary voltage breakdown. Following the shutdown, the beam was reinitiated in 7 seconds by automatic turn on and a momentary magnetic field interruption through ground command. The thrustor operated with all conditions fixed other than the rising boiler temperature for a total period of 14 minutes. During this period there were 10 automatic shutdowns as a result of voltage breakdowns. The shutdown times varied from 2 to 16 seconds and the total shutdown time was 55 seconds.

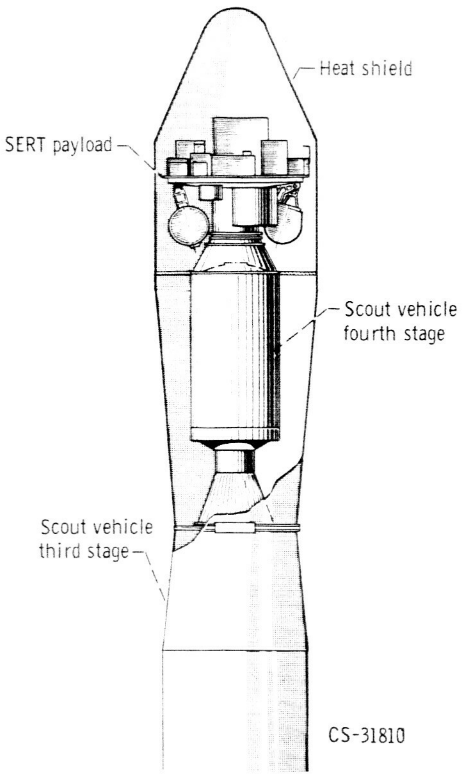
Following the 14-minute operating period, the programed neutralizer voltage and neutralizer turn-off studies and the beam probe surveys were performed. The neutralizer was turned off for a period of 2 minutes, during which time the system automatically cycled between voltage turn on to voltage breakdown for a period of 120 seconds without the establishment of an ion beam. The beam was quickly restored following neutralizer turn on at the end of the 120-second period.

Following the second attempt to start the contact-ionization thrustor, the electron-bombardment thrustor was turned on for the second time. The thrustor produced an ion beam very rapidly and continued to operate with short shutdown periods for 8 minutes, after which the flight was terminated by reentry into the atmosphere. The final recorded beam current was 377 milliamperes with a thrust of 0.0055 pound. Ion-beam current and thrust corresponded in good agreement with theory throughout the entire thrustor operating period. The precision of this correspondence will be determined through data processing. Operation of the electron bombardment thrustor during the flight period, as illustrated by the variation of thrust with time, is presented in figure 5.

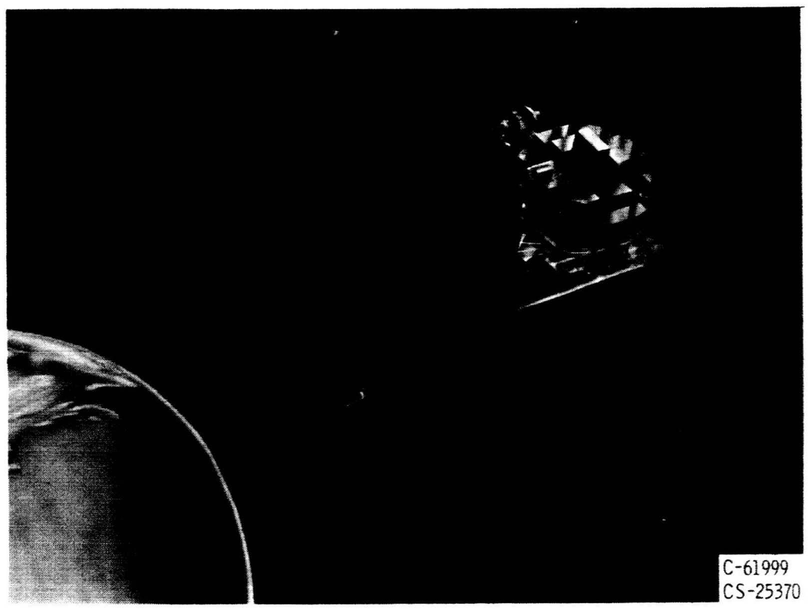
Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, August 10, 1964.

REFERENCES

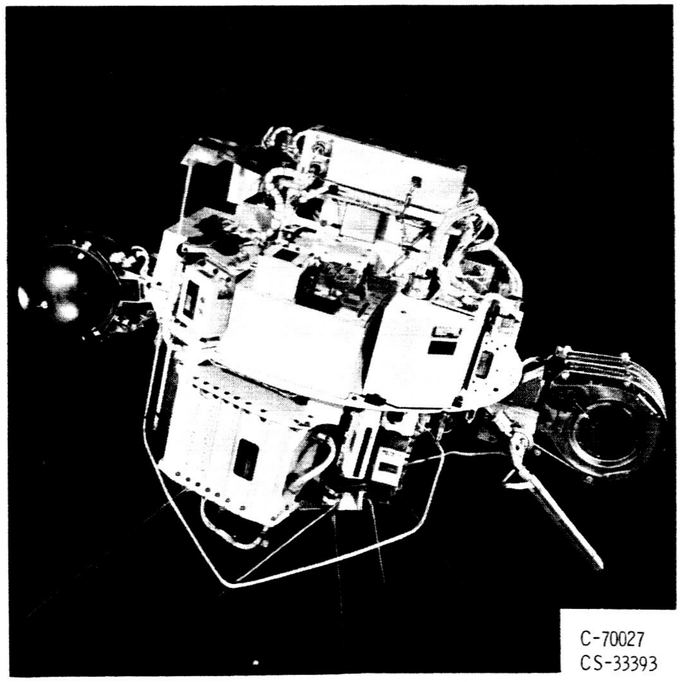
1. Mickelsen, William R., and Kaufman, Harold R.: Status of Electrostatic Thrustors for Space Propulsion. NASA TN D-2172, 1964.
2. Stuhlinger, Ernst: Ion Propulsion for Space Flight. McGraw-Hill Book Co., Inc., 1964.



(a) Launch configuration.

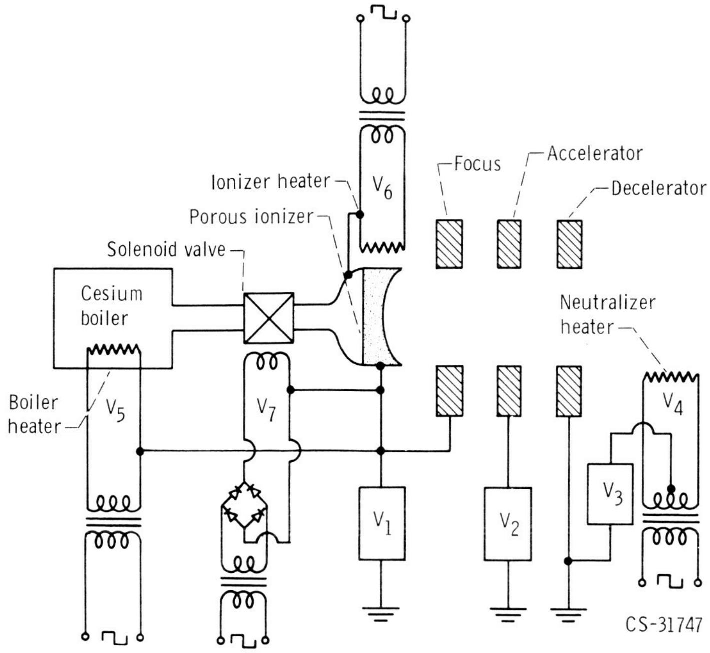


(b) Drawing of free-flight configuration.



(c) Photograph of free-flight configuration.

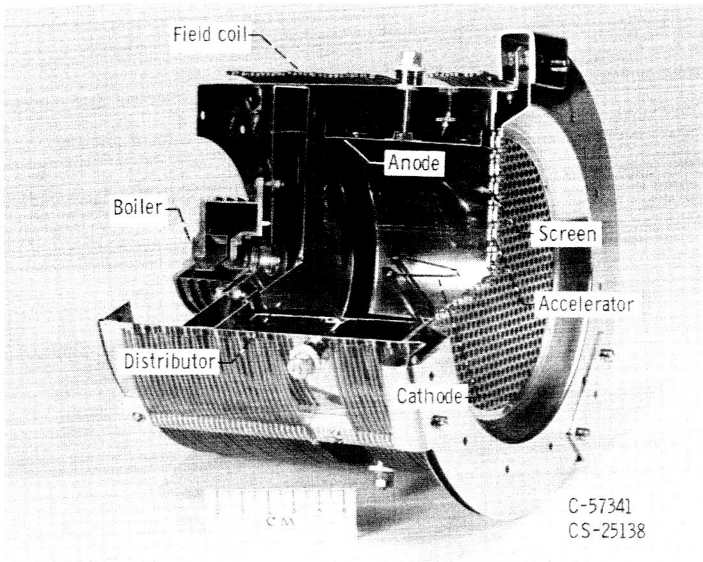
Figure 1. - SERT I spacecraft.



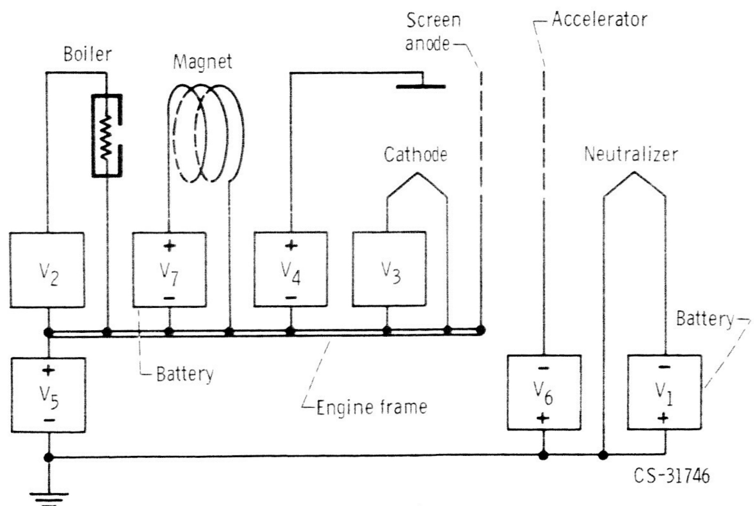
(a) Schematic diagram.

(b) View during vacuum-chamber operation.

Figure 2. - Contact-ionization thrustor.

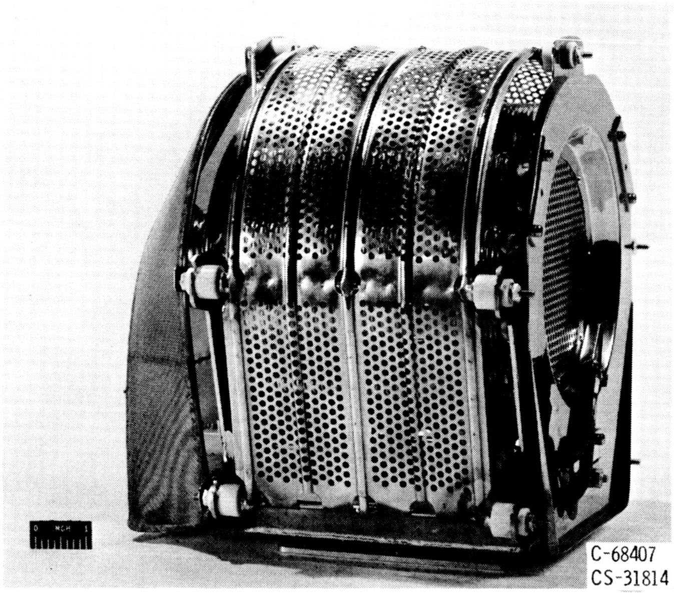


(a) Cutaway laboratory model.



(b) Schematic diagram.

Figure 3. - Electron-bombardment thrustor.



(c) Flight model.
Figure 3. - Concluded. Electron-bombardment thrustor.

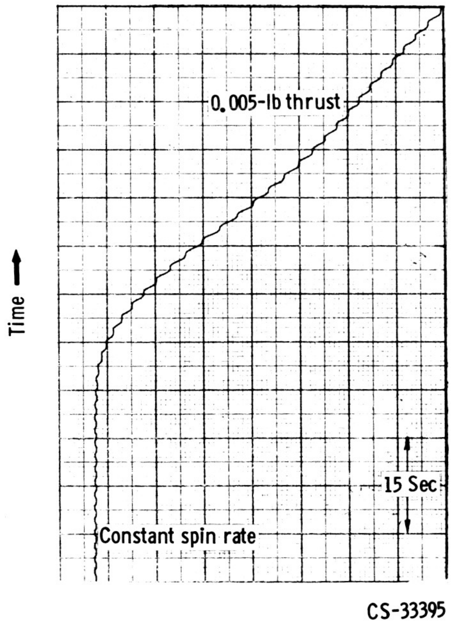


Figure 4. - Sert I spacecraft radial accelerometer real-time record.

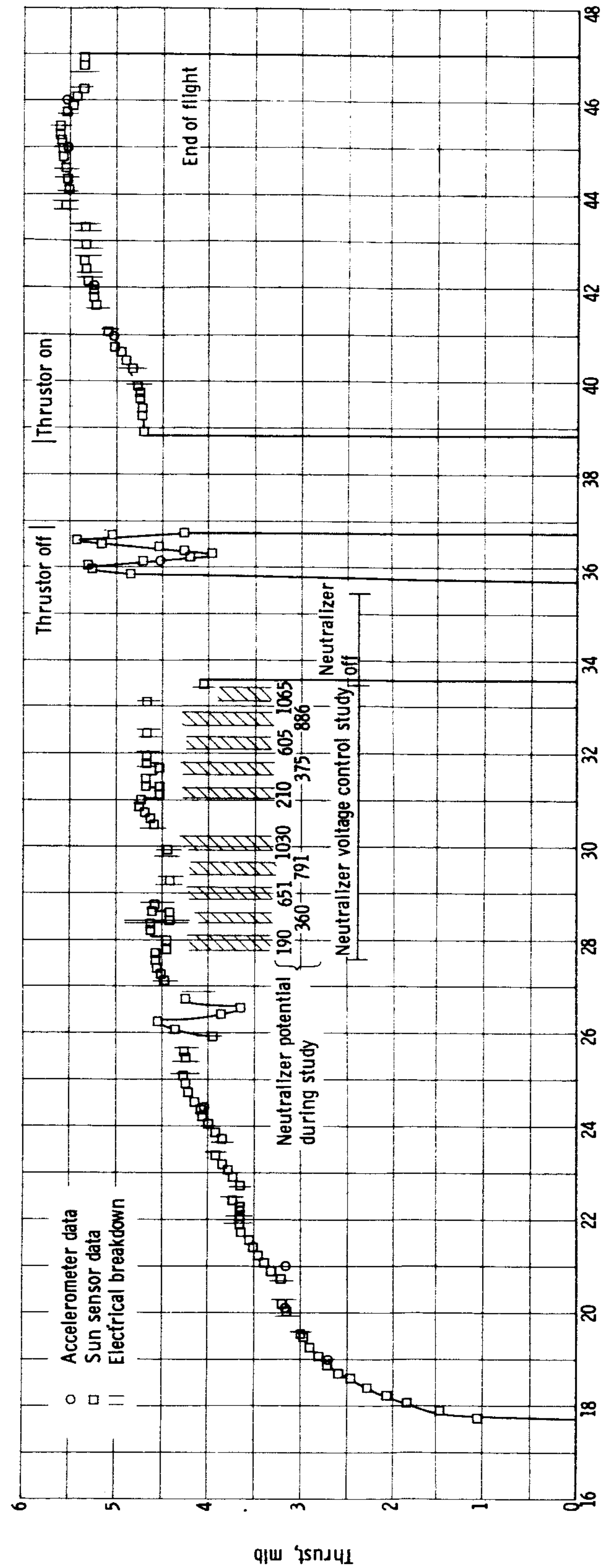


Figure 5. - Thrust computed from sun sensor and accelerometer data. SERT I flight.