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Development of a 5-cm Flight-Qualified Mercury Ion Thruster

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A 5-cm structurally-integrated ion thruster (SIT-5) has been advanced from a predesign concept to a flight-qualified system for application to attitude control and stationkeeping of synchronous satellites. All elements of the system are structurally and thermally matched with one another and also with respect to a spacecraft with which the system will ultimately be associated. A thorough program of component and system testing has qualified the SIT-5 for operation at a thrust $T = 2.1$ mN with a specific impulse $I_{sp} = 3040$ sec and a total elecrical power consumption $P_T = 72.2$ w. With its vectorable beam-extraction system, thrust can be deflected electrostatically by up to 10° in any azimuthal direction. The system is 31.2 cm long by 12.1 cm in diameter. It weighs 2.2 kg including tankage for 6.8 kg of mercury propellant which is sufficient for over 25,000 hr of full thrust operation. A launch-enviroment test has structurally qualified the system for shock (30 g), sinusoidal (9 g) and random (19.9 g rms) vibrations.

Introduction

MISSION and system analyses have demonstrated the suitability of electron-bombardment ion thrusters for attitude control and stationkeeping of synchronous satellites where low system mass and long life are major requirements.[1–3] In initial research investigations with mercury bombardment thrusters at the NASA Lewis Research Centre (LeRC), a thrust level of 2.2 mN was chosen as being representative of a number of applications for satellites weighing 220 kg to 680 kg. At this thrust level, the duty cycle to provide the North-South stationkeeping function is estimated[2] at about 0.5 for each of two thrusters operated back-to-back, and at 0.01 for the attitude control function. With these duty cycles, a satellite lifetime of 3 yr requires an operating lifetime for the stationkeeping thruster of about 13,000 hr and includes 1000 on-off cycles.

Thruster operation at LeRC reported in 1966 by Kerslake et al.[4] demonstrated the suitability of an experimental thruster of 5-cm anode diameter. With an oxide-coated brush cathode, the thruster system was endurance tested for 1553 hr. At a specific impulse $I_{sp} = 3050$ sec, this system generated a thrust $T = 2.9$ mlb with an over-all power-to-thrust ratio of 50 w/mN.

In subsequent investigations at LeRC by Reader et al.,[5] somewhat more efficient operation was demonstrated at lower specific impulse ($I_{sp} = 1800$ sec) with a 5-cm thruster which utilized technology developed for the NASA Space Electric Rocket Test II (SERT II),[6,7] including the use of hollow cathodes[8] for both the discharge-chamber and neutralizer electron sources. Both the discharge and neutralizer emitters were of the enclosed hollow cathode type.[9] The neutralizer cathode was operated successfully at an equivalent mercury flowrate below 2 ma while coupled to an operating thruster beam. The total discharge-chamber flow was directed through the discharge cathode, and post-cathode propellant deversion was provided by ports passing through the walls of the cathode-cup polepiece. All thruster tests were conducted with a single glass-coated accelerator grid.[10]

Based on the encouraging results of the LeRC research program, a development effort was initiated at Hughes Research Labs. (HRL) to advance the 5-cm mercury-bombardment thruster subsystem to flight-qualified status. Progress on this effort was first reported by Nakanishi et al.[11] in conjunction with the simultaneous investigation at LeRC of a 5-cm prototype thruster of different design. The intent of the present paper is to report on development by HRL of the 5-cm flight qualified unit which has now been completed.

Technical Program

A 5-cm structurally integrated ion thruster (SIT-5) has been advanced from a predesign concept to a flight-qualified system for application to attitude control and stationkeeping of synchronous satellites. In design of the SIT-5 system, special

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emphasis has been placed on both structural and thermal integration of all components to achieve minimum size, maximum structural strength, and maximum over-all efficiency. As opposed to larger thruster systems where discharge-power losses predominate, all power losses must be carefully minimized in satellite-control thrusters to achieve high electrical efficiency.

Extensive analysis accompanied design of this thruster system to ensure compatibility of the propulsion device both with the launch environment and with the thermal and mechanical environments imposed by a host spacecraft. The completed thruster system underwent a comprehensive test program, including vibration testing, to demonstrate structural integrity and performance testing both with conventional and with thrust-vectoring beam extraction systems. Thruster tests were complemented by an accompanying program of component testing, including separate tests of the main-cathode subassembly, the neutralizer subassembly and the mercury propellant reservoir subassembly.

SIT-5 Design

Structural design of the SIT-5 system centers about the concept that the thruster system should be supported close to the point of maximum mass concentration. As shown in Fig. 1, the propellant reservoir is supported by a mounting flange near the equator of the spherical mercury tank. The thruster itself is cantilevered from the location by a main connecting structure and insulated from that structure by four ceramic supports. The neutralizer is attached to the ground screen enclosing the thruster assembly which is also cantilevered from the main connecting structure. Type 304 stainless steel was chosen for fabrication of most structural elements, because it is known to resist attack by mercury and can readily be welded to the soft-iron components of the thruster. No mass penalty was associated with the use of stainless steel, because it may by used in thin sections while still maintaining structural integrity. Over-all system length is 31.2 cm, with a maximum diameter of 12.1 cm except for the mounting flange which is 13.4 cm in diameter. System mass is 2.2 kg including tankage for 6.8 kg of mercury propellant which is sufficient for over 25,000 hr of full thrust operation.

The main cathode is designed as part of an integral combination with a high-voltage isolator and main propellant vaporizer with minimum separation provided between the various elements. This allows the cathode, isolator, and vaporizer to be coupled closely to one another, thereby minimizing the over-all extent and also permitting heat transfer between the several elements. In this way, cathode heat (which is generated by the discharge) is used effectively to maintain the downstream end of the isolator assembly at the temperature required to avoid mercury-vapor condensation. Similarly, the heat rejected from the main vaporizer assembly is conducted toward the upstream end of the isolator so that no separate heater is required for the isolator assembly as was necessary in earlier systems.[12] The same philosophy is used in design of the neutralizer subassembly. The vaporizer is placed as close to the neutralizer cathode as permissible within the constraint of maintaining control over vaporizer temperature by the vaporizer heater and not having its temperature rise uncontrollably in response to thermal coupling with the cathode.

A magnetic field is generated inside the discharge chamber by permanent magnets which span the gap from the discharge-chamber end-plate to the collar polepiece. As indicated in Fig. 2, the field is directed axially for the most part, but diverges at the downstream end in a manner determined by the shape of screen and cathode polepieces. All elements of the magnet circuit are fabricated of type 1018 mild steel. General aspects of this magnetic design are typical of all ion thrusters which are adapted from the SERT II configuration.

The beam-extraction system provides an electrostatic thrust vectoring capability of 10° from the thruster axis in any azimuthal direction. Individual beam deflection elements consist of thin, flat molybdenum electrodes which are slotted to form an interlocking structure which functions as the accel electrode. A conventional drilled molybdenum plate serves as the screen electrode. The accel elements are supported at each end by fingers extending from four molybdenum spring-tension strips which are brazed to long rectangular accel-support insulators which are bolted to each edge of a rectangular extension of the collar polepiece. Small pins brazed to the ends of the accel elements lock into recesses provided on the bent tab of the spring fingers. The 89 circular beam

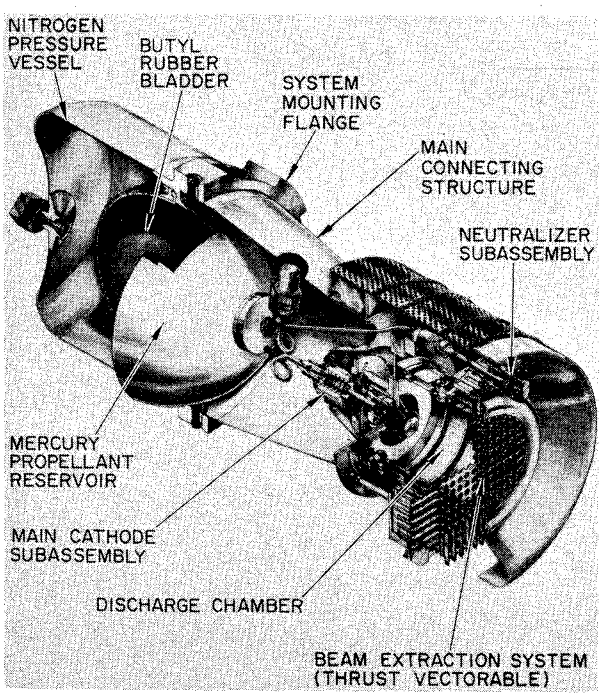

Fig. 1 SIT-5 ion thruster.

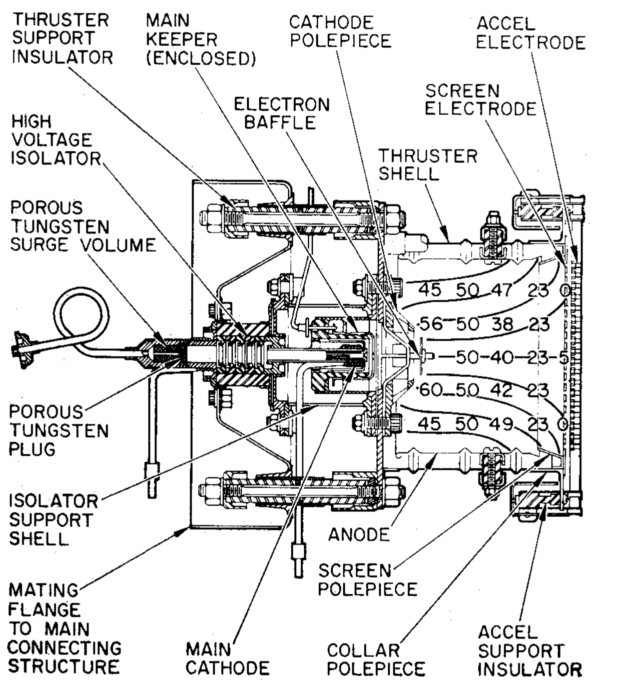

Fig. 2 Discharge chamber with main-cathode subassembly attached. The magnetic field shape is shown with values of the axial field component indicated in gauss.

forming apertures of the screen electrode are each 0.317 cm in diameter and are arranged in a square pattern with a 0.445-cm center-to-center spacing. This provides a screen open-area to total-area ratio of 36%. The individual beam-deflection elements are each 0.254-cm high and separated so as to form an array of square apertures which are 0.266 cm on a side.

A unique design feature is incorporated into the main and neutralizer vaporizers to prevent liquid mercury from being driven through the pores of the vaporizer plug by the high dynamic pressures generated during booster launch into Earth orbit. Under the maximum shock loading of 30 g anticipated by the predesign analysis, a total hydrostatic pressure of 110 N/cm^2 is generated at the neutralizer location which tends to cause mercury intrusion through the 1.8 μm pores of the vaporizer plug. To prevent such intrusion, a 0.65-cm high porous tungsten column is enclosed inside both of the mercury feedlines, upstream of the vaporizer disk as shown in Fig. 2. This column acts as a surge volume to absorb the total quantity of inertia-limited mercury flow which passes through the feedlines during the shock loadings anticipated in the design analysis. At a total hydrostatic pressure of 28 N/cm^2, the flow begins to penetrate the 4.5 μm, diameter pores of the tungsten column, and this serves to limit the pressure below the value that would cause intrusion into the vaporizer plug. A channel is provided through the axis of the column to permit free transmission of mercury liquid to the vaporizer plug during normal operation. This opening does not compromise the pressure-limiting feature, since the necessary flow constriction occurs at the feed lines and not within the surge volume.

Following the research studies of Hall et al.[9] and Nakanishi et al.,[11] an enclosed keeper geometry was chosen for use in the SIT-5 system both for the main discharge and neutralizer cathode application. The vapor flow impedance afforded by the enclosed keeper configuration has permitted satisfactory neutralizer coupling with the ion beam to be achieved with neutralizer vapor flowrates as low as 1.0 ma equivalent. The neutralizer cathode is located so that the aperture of the neutralizer-keeper electrode is placed at a position 2.71 cm downstream and 2.71 cm radially outward from the outermost beam aperture of the ion extraction system. The axis of the neutralizer is directed straight downstream parallel to the thruster axis.

Figure 2 shows in detail the manner in which the main-cathode subassembly and discharge chamber are supported by a conical element which attaches directly to the main connecting structure. The alumina isolator body and all alumina thruster-support insulators are held in compression during assembly and under peak-load conditions; this increases the strength of the ceramic supports by an order of magnitude. The four thruster-support insulators are placed in compression by pressing them (by means of coaxial stainless-steel lugs) against a second set of four insulators which are inserted from the opposite side of Kovar bushings. With all four thruster support insulators placed in compression by this expedient, it is possible (in the final assembly) to ensure that the isolator ceramic is also placed in compression without placing the support insulators in tension.

Dimensions for the discharge-chamber were obtained by scaling from the 15 cm SERT II design by the geometric ratio of thruster diameters except for several important exceptions which are discussed below. The discharge-chamber length-to-diameter ratio was determined by direct optimization and confirmed by inference from the ratios chosen for other highly optimized systems which employ the SERT II magnetic-field configuration. The length-to-diameter L/D ratio for five discharge chambers is shown in Fig. 3. Although each thruster was optimized independently at one of three different laboratories (HRL[13,14], LeRC[6,15] and the Jet Propulsion Lab., JPL[16,17], the values of L/D for all of the thrusters fall close to a common curve.

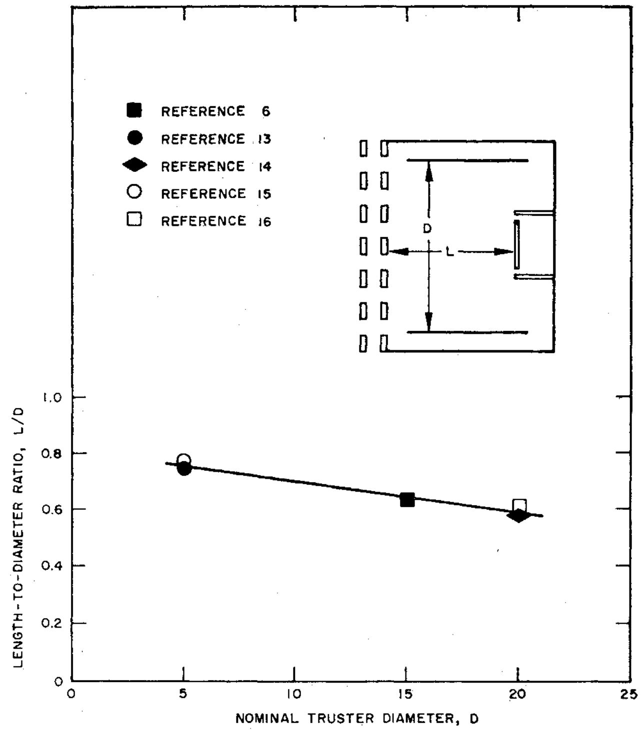

Fig. 3 Length-to-diameter ratio L/D for optimized discharge chambers of SERT II-type magnetic field geometry.

In other hollow-cathode thrusters of the SERT II type, the major fraction of propellant flowrate is introduced directly into the discharge chamber by means of a propellant delivery which is separate from the one associated with the main cathode.[6,12,16] Contrary to this practice of precathode propellant diversion, in the SIT-5 thruster all discharge-chamber propellant is introduced through the hollow cathode and passes through the enclosure formed by the electron baffle and cathode polepiece. With this design, there is a tendency for the discharge voltage to decrease as propellant flow through the cathode increases, if the magnetic-field configuration and geometrical parameters of the discharge chamber are held constant. In the SIT-5 thruster, independent control over discharge voltage has been achieved through a technique of postcathode propellant diversion which was first established at HRL in development of the LM cathode thruster.[18] Following the design suggested by Ref. 11, ports are located in the walls of the cathode polepiece which permit reduction of the neutral-particle density below the value which obtains when these ports are absent. For a given level of beam current, discharge voltage can be regulated by changing the transmission of the propellant diversion ports by the use of wire mesh of varying transparency. This permits independent adjustment of the flow conductances (between the cathode-polepiece region and discharge-chamber region) for electrons on the one hand and for the propellant atoms on the other; the electrons pass predominantly through the radial gap between cathode polepiece and the electron baffle, while the propellant escapes predominantly through the screened ports. With this modification, it has been possible to adjust flow conditions to achieve the optimum values of particle density within the cathode polepiece and its openings while operating the thruster at the required values of propellant flowrate and discharge current.

Design Analysis

Because of the broad based nature of the design and analysis, the SIT-5 system is of flight-type status and ready for system integration from the standpont of structural integrity and firm control over the operation of each component. All elements of the thruster system are structurally and thermally matched with one another and also with respect to a spacecraft with which it will be ultimately associated. During the design phase of the SIT-5 system, elements of the thermal design were checked against a comprehensive computer-aided thermal analysis, in which the temperatures of the thruster were calculated under varying environmental conditions, including exposure to full sunlight or spacecraft shadow, operation at full power or with discharge extinguished, and location of thruster near or within a spacecraft as opposed to far from a spacecraft.

A total of eight cases (representing a wide variety of operating conditions) were analyzed.[19] These are listed in Table 1. Particular attention was directed toward achieving thermal control of the two critical areas in the thruster system. 1) Maintaining optimal thermal control of the vaporizers (which are heated partially by cathode-discharge power) in order to minimize vaporizer power requirements while precluding thermal runaway. 2) Maintaining the temperature of the propellant reservoir above the freezing point of mercury (−39°C) and below the maximum permissible bladder temperature (120°C).

By factoring the results of thermal analysis directly into thruster design, acceptable temperature limits were established for operation of the thruster system under all normal conditions. For thruster operation in shadow far from a spacecraft, mercury freezing at the reservoir is prevented by employing a reservoir heater which consumes 5 w of electrical power. When the reservoir subsystem is exposed directly to sunlight, a thermal control coating is necessary to keep the butyl rubber bladder at a temperature which is acceptably below the maximum allowable temperature of 120°C.

Thermal transients were analyzed also to determine the critical thermal relaxation times of various components as a function of time after the system enters a zone of solar eclipse subsequent to normal operation (as a worse case, it is assumed that normal thruster operation would be suspended during that period). Results of the transient analysis show that the shortest time constant occurs at the feed line adjacent to the neutralizer. It has a time constant of 52 min to reach the freezing point of mercury with no input power. However, a minimal heat input to the neutralizer cathode of only 1 w is sufficient to prevent freezing. The next most critical area is the main vaporizer feed line, which has a slightly longer time constant of 1.2 hr. The reservoir flange has a time constant of 2.1 hr, and the reservoir has a time constant of about 10 hr.

Comprehensive mechanical and dynamic analysis was also carried out. The analysis was intiated by a mathematical model of the thruster system in which it is represented as a matrix of lumped masses joined by spring tensors. This model was subjected mathematically to the dynamic environment of thruster launch. System responses to the dynamic environment are generated by this program and expressed in terms of amplitude displacement. Displacement amplitudes in turn are related to material stresses, which permit stress analysis of the thruster system subject to the dynamic environment. Where stuctural weaknesses were identified by this procedure, corrective action was taken prior to design completion. Similarly, regions that appeared to be over-designed were reduced in size in order to minimize over-all system mass. The final proof of structural integrity was indicated by a dynamic test to simulate the launch environment in which the actual thruster was subjected to shock (30 g), sinusoidal (9 g), and random (19.9 g rms) vibrations. This test demonstrated a high degree of structural integrity and only minor changes were required to ensure a satisfactory launch capability.

Table 1 Cases analyzed thermally

Conditions	Case no. 1	2	3	4	5	6	7	8
Full sunlight	×	×	×	×				
Full shadow					×	×	×	×
Propellant reservoir full	×	×	×					
Propellant reservoir empty				×	×	×	×	×
Maximum power operation[a]	×	×	×					
Minimum power operation[b]				×			×	×
Cathode power = 10 w					×			
Neutralizer power = 6 w					×			
Reservoir power input = 5 w						×	×	
Main cathode power = 5 w, Neutralizer power = 1 w						×		
Near spacecraft	×							
Far from spacecraft		×		×	×	×	×	×
Inside of spacecraft			×					
Solar radiation on optics			×					

[a] Maximum power operation is with 7.55 w to the main cathode, 2.25 w to the cathode keeper, 2 w to the main vaporizer, 5.3 w to the anode, 2.4 w to the thruster endplate, 0.9 w to the grid, and 14.8 w to the neutralizer subassembly.
[b] Minimum power operation is with 6.2 w to the main cathode, 1.8 w to the cathode keeper, 2 w to the main vaporizer, 4 w to the anode, 1.8 w to the thruster endplate, 0.66 w to the grid, and 10.9 w to the neutralizer subassembly.

Subassembly Tests

Prior to assembly of the integrated thruster system, separate testing of various subassemblies was carried out as a means of ensuring against deficiencies in system operation which might otherwise delay program progress. The main-cathode, neutralizer, and propellant reservoir subassemblies were tested to establish their performance, and generally, satisfactory results were demonstrated.[19] The main-cathode subassembly demonstrated a capability for operation throughout the range of parameters anticipated in thruster operation. Temperature profiles for the cathode feed tube, the isolator, and the vaporizer were generally satisfactory with no evidence of cold spots and an over-all power consumption close to the minimum anticipated values.

Similar satisfactory performance was obtained with the neutralizer subassembly. A general characteristic was identified, for a given mercury-vapor flowrate, which indicated a decreasing coupling voltage between the neutralizer cathode and a screened collector electrode (to simulate the ion beam) as a function of increasing discharge current to the neutralizer keeper electrode. For a given value of keeper current, both the keeper and collector coupling voltage were found to decrease with increasing mercury-vapor flowrate.

Testing the propellant reservoir subassembly served to establish the pressure retention capability of the positive-expulsion reservoir and to establish the long term capability for vaporizer phase separation under steady-state and cyclic conditions. The same mercury-propellant-reservoir feed line and vaporizers were used in this test as had earlier been subjected to the dynamic environment imposed by the structural integrity tests.

Thruster Tests

An extensive program of thruster testing has confirmed design expectations with regard to steady-state and cyclic performance capabilities. Qualification tests were preceded by a program of optimization to adjust elements of the discharge-chamber geometry for peak thruster performance. In its optimized configuration, the thruster was operated under steady-state and cyclic conditions to qualify the system for its intended application for altitude control and stationkeeping of synchronous satellites.

Discharge-Chamber Optimization

On the basis of scaling criteria discussed earlier, flight-type components were constucted for SIT-5 thruster assembly and a program of final discharge-chamber optimization was initiated. By choosing to proceed directly to the flight-type construction rather than a more easily modified laboratory configuration, many of the geometrical parameters were difficult to change and no attempt was made toward further optimization, however, the design permitted ready modification of those components expected to play a major part in discharge-chamber optimization. These included the following discharge-chamber elements: screen polepiece shape, collar polepiece length, electron baffle size and position, transmission of propellant diversion ports, keeper aperture size, and main hollow-cathode type (open or enclosed). To permit the magnetic field to be varied as a parameter for optimization, the optimized thruster was equipped initially with a set of 8 rod electromagnets.

For all optimization testing, mercury flowrate to the main-cathode and neutralizer subassemblies were separately measured with an accuracy of about 1%. Feed tubes from the two vaporizers were connected to separate liquid-mercury reservoirs. For the main cathode, mercury was supplied to the vaporizer from a cylindrical reservoir in which a piston is pressed against the mercury surface to provide the propellant driving force. The piston position is indicated by a dial indicator (calibrated to 0.00025 cm) which contacts the top of the piston shaft and permits accurate determination of the rate of mercury consumption. For the neutralizer cathode, a more sensitive flowmeter was employed which uses a capillary tube (open to atmosphere at the upstream end) to measure mercury consumption. In this system, mercury is supplied to the vaporizer from a 0.05 mm precision bore capillary burette calibrated in 0.001 cm^3 increments. Displacement of the mercury meniscus as a function of time permits accurate determination of the mercury flowrate. Both feed systems were filled under vacuum in order to eliminate introduction of gas bubbles.

Discharge-chamber optimization of the SIT-5 system was impeded somewhat by the fact that the beam-extraction system was developed under a separate and concurrent program.[20] (Discharge-chamber performance is affected strongly by the specific nature of the ion-extraction system with respect to the geometrical open-area ratio, the extraction fields, etc.[16,17,18,21]) To minimize this conflict, a beam-extraction system was constructed in a manner which simulated the anticipated geometry of the beam-vectoring system by use of an identical screen electrode in combination with a conventional accel electrode with circular apertures of diameter equal to the separation between vector elements.

In discharge-chamber optimization, the thruster was operated almost exclusively at a discharge voltage close to the maximum acceptable value, and the propellant-utilization efficiency was determined at that operating point alone. The data were sufficient to characterize optimal thruster performance, because both the discharge voltage and propellant utilization efficiency rise steeply together as a function of discharge current.† The value of discharge voltage was restricted to $V_D \lesssim 45$ to limit the rate of discharge-chamber erosion due to ion-bombardment of cathode-potential surfaces.[22] While preference was given to configurations where high utilization could be achieved with low values of discharge power, only the maximum value of propellant utilization (obtained at maximum discharge power) was considered to be of primary importance, since discharge losses constitute only about 30% to 40% of the total power losses.

Steady State Performance

On the basis of selective modifications of the discharge-chamber elements available for optimization, a performance peak was identified for operation with the configuration described in Table 2. This configuration was selected initially for operation with the rod electromagnets at a peak axial magnetic-field strength of 53 gauss, one-half the intensity reported for operation with 8 permanent magnets chosen on the basis of scaling from the SERT II geometry.

Table 2 Key parameters for SIT-5 thruster operation with simulated beam deflection system

Main keeper aperture	
Diameter	0.475 cm
Thickness	0.051 cm
Neutralizer keeper aperture	
Diameter	0.081 cm
Thickness	0.025 cm
Neutralizer position	
Neutralizer pointing angle with respect to the thruster axis	0°
Distance from the keeper aperture from the outermost beam aperture	
Downstream	2.71 cm
Radially outward	2.71 cm
Baffle	
Diameter	0.952 cm
Radial gap	0.159 cm
Axial gap	0.159 cm
Transmission of mesh covering propellant diversion ports	91%
Magnetic field generation	4 magnets (0.450 cm dia) equally spaced
Length of collar polepiece	0.952 cm

For subsequent thruster tests, a magnetic-field pattern similar to the one generated with the electromagnets was provided by substitution of four permanent magnets (0.450 cm in diameter) equally spaced at every second position identified in the design for the eight-magnet configuration.‡ In the final SIT-5 design, the four sets of unused magnet-retainer clamps were left in place and stainless steel rods were placed at the unused magnet locations to preserve azimuthal structural symmetry. In this configuration, the thruster was operated for over 46 hr at the performance level indicated in Table 3 with a beam current $I_B = 25.6$ ma, a discharge-chamber propellant-utilization efficiency $\eta_u' = 82.5\%$ and an electrical efficiency $\eta_E = 56\%$ at a discharge voltage $V_D = 45$ v. Similar electrical performance was demonstrated at a discharge voltage $V_D = 42$ v with $\eta_u' = 79.9\%$.

The SIT-5 thruster was operated with the beam-deflection system only after it was combined with its flight-type propellant feed system. In this configuration, electrical performance was virtually unchanged, and no significant change in propellant-utilization efficiency was detectable from mass measurements conducted before and after that operation. Deflections of $\pm 10°$ were attained repeatably in the X and Y directions with accel currents remaining relatively constant at 0.08 ma. A complete mapping of thrust vectoring data are reported in Ref. 20 where a linear deflection rate of about 18°/kv is indicated. During that operation, however, propellant measurements were carried out with a burette feed system, and a significant decrease in propellant utilization was reported. This discrepancy has not been resolved under the subject effort, and the need for further study is indicated.

Cyclic Performance

Two tests of thruster performance were undertaken to establish the system's capability for cyclic operation. The first test included temperature measurements of a number

†At a given value of ion-beam current, discharge-chamber propellant utilization and discharge voltage are uniquely determined by the discharge current for a particular magnetic and geometric configuration. This is true with all thrusters where total discharge-chamber propellant flow passes through the main cathode.[11,14,18]

‡This arrangement does not place the four magnets in position of azimuthal symmetry with respect to the thrust-vectorable ion-extraction system.

Table 3 SIT-5 system performance profile with simulated beam deflection system

Nominal operating parameters	Operating values
Beam voltage,[a] v	1600
Beam current, ma	25.6
Accel voltage,[a] v	−800
Accel drain current, ma (at 0° deflection)	0.075
Discharge voltage, v	45
Discharge current, ma	260
Discharge power, w	11.7
Cathode	
Keeper voltage, v	18
Keeper current, ma	365
Keeper power, w	6.6
Heater power, w	0
Vaporizer voltage, v	3.2
Vaporizer current, amp	1.3
Vaporizer heater power, w	4.2
Neutralizer	
Keeper voltage, v	17
Keeper current, ma	365
Keeper power, w	6.2
Heater power, w	0
Vaporizer voltage, v	2.3
Vaporizer current, amp	1
Vaporizer heater power, w	2.3
Coupling voltage, v	11
Output beam power, w	40.3
Total input power, w	72.2
Thruster propellant flow Equivalent, ma (cathode flow)	31.1
Neutralizer propellant flow Equivalent, ma	2.6
Propellant utilization efficiency (including neutralizer), %	76
Electrical efficiency, %	56
Over-all efficiency, %	43
Discharge loss, ev/ion	465
Thrust, mN	2.1
Specific impulse, sec	3040
Power-to-thrust ratio, w/mN	34.4

[a] For experimental convenience, data for this test were actually generated at a beam and accel voltage of 1200 v each rather than the design point values listed above. Subsequent operation of the SIT-5 system at the design point demonstrates that no significant error was introduced by this expedient.

of important thruster components which confirmed the predictions of thermal analysis. Although power losses in actual thruster operation differed in detail from the distribution assumed in the analysis (see notes a and b of Table 1), over-all power losses corresponded closely to the minimum-power case and predicted temperatures corresponded closely to those measured experimentally. The second test was undertaken to demonstrate a long-term system capability to endure thermal and mechanical stresses associated with thruster start up and shut down.

The objective of the first cyclic test was to provide performance data for the SIT-5 thruster to go from a low-level standby condition defined in Table 4 to a state of full power and return to the low-level condition. The low-level condition, with a standby power of 10.3 w and total mercury flowrate equivalent of 3.8 ma, was consistent with low-power and low-propellant consumption while maintaining the main cathode-to-keeper and neutralizer cathode-to-keeper discharges. Both keeper discharges operated without a single extinction during the 19 hr standby operation prior to the start of the cyclic test and the 9 hr required for the cyclic test. The average time per cycle was 47 min, and it took approximately 10 min to advance from the standby condition to full beam operation. A temperature profile taken during the sixth cycle of the test is shown in Fig. 4. As expected, temperatures of the neutralizer vaporizer, neutralizer body, main vaporizer, and isolator downstream end increase significantly as the full power condition is approached. There is only a

Table 4 Values for standby thruster operation

Thruster parameter	Operating value: Discharge cathode	Operating value: Neutralizer cathode
Keeper voltage, v	24.0	16.0
Keeper current, ma	100.0	100.0
Keeper power, w	2.4	1.6
Vaporizer heater voltage, v	2.8	2.5
Vaporizer heater current, amp	1.25	1.1
Vaporizer power, w	3.5	2.75
Vaporizer temperature, °C	139.0	248.0
Mercury flowrate equivalent, ma	2.42	1.42
Total power, w	5.9	4.35

slight temperature rise on the discharge-chamber endplate, shell, accel, and ground screen, while temperatures of the feed lines and reservoir are virtually constant throughout the cycle.

As a second demonstration of cyclic operation, the thruster was cycled from the power-off to the full beam-on condition repetitively over various periods[23] to establish its capability to function in the mode anticipated for North-South stationkeeping of a synchronous satellite. In a final 500-hr accelerated duty cycle test, the thruster was cycled every 30 min to establish its capability for a lifetime in excess of 1000 cycles. The chosen cycle included a 10 min off time, a 10 min startup time, and a 10 min neutralizer and beam-on time. Throughout the test, the cycle was completed in ½ hr. The off time was always 10 min. Occasionally, the startup time took longer than 10 min and, if this were the case, the neutralizer and beam-on time was less than 10 min. This sequence was followed for 594 cycles at which time an open circuit occurred in the main-vaporizer heater which precluded the possibility of main-cathode ignition. The test was nonetheless continued for the full 1000 cycles with neutralizer ignition only, but with the main cathode heated throughout each cycle with a full ignition power of 18 w. After completion of the 1000 neutralizer-ignition cycles, the main-vaporizer heater was replaced and the thruster was operated for several additional cycles to demonstrate that normal operation had been reestablished. An improved heater design is described in Ref. 23 which avoids the high thermal stresses which obtained with the main-vaporizer heater used in this test.

System Lifetime

No extended demonstration of thruster lifetime has been included as part of the system development at HRL. However, an extensive test of SIT-5 thruster performance has recently been reported by Nakanishi at LeRC[24] using a thruster provided under contract by HRL. As of Oct. 1, 1972, 8000 hr of consecutive operation had been logged by one of

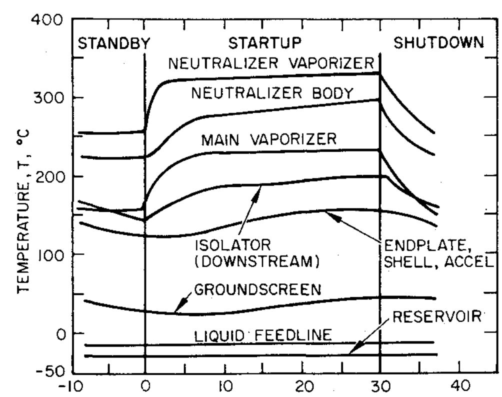

Fig. 4 Thruster temperature history.

the thrusters. For the initial 2023 hr, the system was operated with a translating-screen beam-deflection system, but subsequent operation has been logged with the electrostatic vectoring system described in Ref. 24 and this report. Several details of thruster geometry were modified by LeRC, as required for each of the tests, and do not correspond exactly to the configuration described in Table 2. For this operation, the discharge voltage was held in a range 37 v $< V_D <$ 40 v.

Summary of Results

Under the subject program, the SIT-5 propulsion system has advanced from a predesign concept to a performance qualified prototype device. Dynamic analysis and vibration testing have confirmed the structural integrity of the system design. The thruster shown in Fig. 1 has been qualified for shock (30 g), sinusoidal (9 g), and random (19.9 g rms) vibrations which establishes the system's capability to survive booster launch. Performance testing with a conventional beam-extraction system has demonstrated operation at specific impulse $I_{sp} =$ 3040 sec with total efficiency $\eta_T = 43\%$. Similar performance with a high-angle (10°) electrostatic beam-deflection system demonstrates the thruster's capability for attitude control and stationkeeping of synchronous satellites.

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