Pulsed Plasma Microthruster Propulsion System for Synchronous Orbit Satellite

William J. Guman*
Fairchild Hiller Corporation/Republic Aviation Division, Farmingdale, N.Y.,
AND
David M. Nathanson†
Lincoln Laboratory,‡ Massachusetts Institute of Technology, Lexington, Mass.

A pulsed plasma microthruster system was designed, built, flight qualified, and delivered to the Massachusetts Institute of Technology (MIT) Lincoln Laboratory for installation in the LES-6 satellite. The system was designed to meet the requirements for east-west stationkeeping of the spin stabilized, synchronous orbit satellite. Four valveless pulsed plasma microthrusters using solid Teflon as the propellant are spaced on the satellite periphery. The flight qualification and acceptance program was successful in meeting environmental constraints and indicated that RFI was not a serious problem in the application of the thruster to the communication satellite. East-west stationkeeping was functionally performed in more than 16 months of space operation.

Introduction

THE LES-6 satellite, which was launched into a synchronous equatorial orbit Sept. 26, 1968 is spin stabilized at ~10 rpm. The satellite configuration is ~48 in. in diameter and 66 in. long and is covered with solar cells as shown in Fig. 1. An electronics platform spans the cylinder approximately at the mid-plane and contains optical sensors and four pulsed plasma microthrusters for east-west stationkeeping. The satellite operates in a synchronous, equatorial orbit and is spin stabilized at ~10 rpm about the cylindrical axis which is nominally parallel with the earth polar axis. Among the more significant new technical features of the LES-6 satellite is the first completely automatic, self-contained stationkeeping system using cold gas and pulsed plasma microthrusters for propulsion. Each thruster housing also serves as a main structural member of the satellite.

Since the satellite is spinning, the pulsed plasma thruster was a logical propulsion system to consider. The thrust axis of each thruster acts through the axis of rotation of the satellite. Impulse bits are required for an extremely short time of the spin period. The valveless solid propellant pulsed plasma microthruster[1] remains completely passive when either not used or in the event of a failure. With such a system it is not possible to impress undesirable forces or moments onto the satellite such as may possibly occur with valved gaseous systems if the valve should leak, remain open, or, from leaks in any other part of the gaseous propellant system. The instantaneous start-up time, requiring essentially no standby power, was also an advantageous feature. During the 5-yr mission life it is anticipated that each thruster will be called upon to provide roughly 1.2×10^7 impulse bits of ~6 μlb-sec each. This total number of impulse bits was considered to be within the state-of-the-art of the pulsed plasma microthruster.

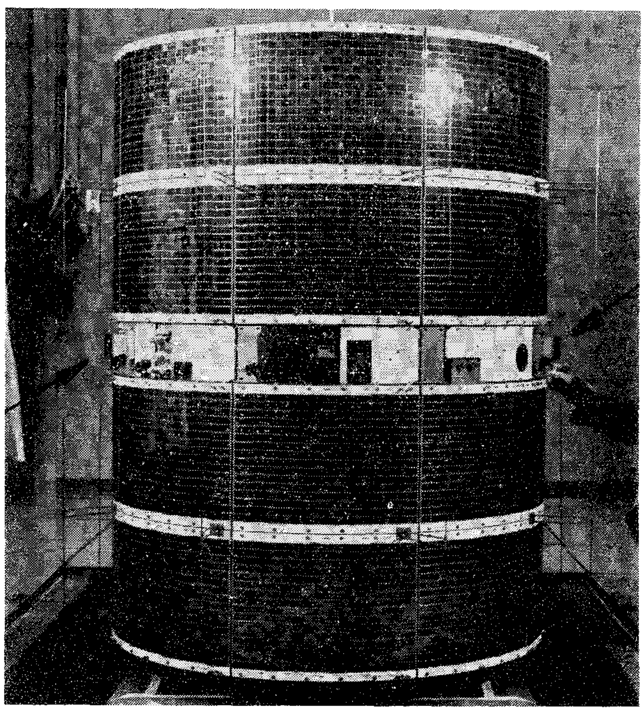

Fig. 1 LES-6 satellite.

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* Chief of Propulsion, Power Conversion Department. Member AIAA.
† Staff Member. Member AIAA.
‡ Operated with support from the U.S. Air Force.

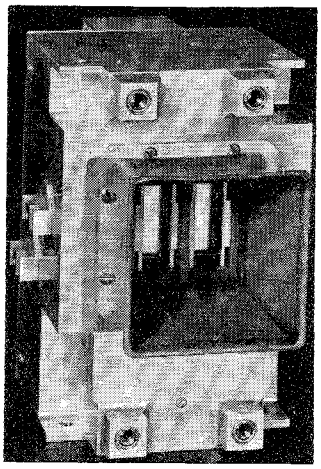

Fig. 2 Flight thruster.

Design Constraints

The program called for the delivery of eight flight qualified thruster systems and two flight qualified power conditioning subsystems, i.e., two complete propulsion systems, subject to a number of design constraints. A flight thruster is shown in Figs. 2 and 3, respectively.

The program was pursued by two essentially parallel efforts. One of these efforts concerned itself with obtaining accelerated life test data of laboratory prototype thrusters representative of final flight hardware, while the second effort concerned itself with engineering design, procurement, manufacturing, qualification, and acceptance testing of the flight hardware.

Since the satellite was in an advanced state of readiness at the time the electric thruster system program was initiated, stringent weight, volumetric and power constraints had to be adhered to. Minor acceptable dimensional changes in the exterior dimensions of the thruster housing envelope were made in order to accommodate the quantity of propellant necessary for the mission yet retaining the inherent simplicity of the linear Negator propellant feed system described in Ref. 1. To retain over-all simplicity, increased reliability, and remain within volumetric constraints two accelerator nozzles instead of one were used in each of the four thrusters. Half of the propellant of each thruster was used in conjunction with each accelerator nozzle. By this technique, each thruster became effective two thrusters thereby achieving redundancy in essentially all major elements of the thruster. Because of the relatively heavy weight (220 g) of the derated thruster capacitor, only one capacitor was used to sequentially energize each of the two accelerator nozzles of a thruster. The over-all redundancy incorporated increased the thruster system weight by roughly 300 g but was believed warranted in terms of the trade-off that was gained in system reliability.

Original estimates indicated that the thruster system would weigh about 1588 g. This figure included the thruster, its discharge initiating electronics, the propellant and its feed system, RFI suppression filtering, and the relatively heavy thruster enclosure which also serves as a structural member of the satellite. The actual weight of each of the nine flight thrusters ranged from 1379 to 1391 g, which was below the original estimate.

The electric power allocated to the thruster system was such that a total energy of 1.85 joules would be available at each thruster every time a thruster would be called upon to generate an impulse bit. Prior experience with the solid Teflon pulsed plasma microthruster showed that for a given power level of operation optimum thruster performance is achieved by operating it at the highest tolerable discharge energy level (i.e., lowest pulse frequency). Therefore, it was decided to use the available power at the thruster to deliver only one impulse bit per thruster per satellite revolution each time power was allocated to the thruster instead of a sequence of lower energy bits having an equivalent total power. Experience from a large number of prior tests also indicated that the discharge energy level of 1.85 joules would completely depolymerize the Teflon propellant. Operation at the largest tolerable energy level per discharge also results in the smallest total number of impulse bits demanded of the thruster during the mission life which effectively increases system reliability.

Environmental constraints imposed upon the system were that it meet the launch environment of a Titan III-C vehicle, be able to withstand temperatures from $-40°C$ to $+60°C$ with a most probable temperature range of operation of $+10°C$. Furthermore, it was required that the system be able to be functionally checked in a humidity environment of $80 \pm 10\%$ at sea level. In general the system was to be compatible with the thermal-vacuum-radiation environment to be encountered at a synchronous altitude and a 5-yr lifetime requirement.

No difficulties were encountered during qualification and acceptance tests of flight hardware either structurally or with respect to performance in meeting environmental constraints.

Propulsive System Description

The electric propulsion system is comprised of four subsystems: the VAT (Vacuum Arc Thruster) driver, the power conditioner, the thrusters, and the discharge initiating circuitry, respectively. The VAT driver designed by MIT Lincoln Laboratory generates the triggering pulses to sequentially activate and discharge each of the four thrusters. The power conditioner accepts low voltage and provides the high-voltage input requirements for each of the four thrusters. The power conditioner also has telemetry outputs for monitoring the voltages of the thrusters and input current to the power conditioner. The thrusters are solid-propellant pulsed plasma microthrusters using Teflon as the propellant and generate the required impulse bits.

A. Thruster and Propellant Feed

The thruster system is of the same principle that has been described elsewhere.[1] Solid Teflon is used as the propellant. The addition of electric energy depolymerizes the Teflon and converts the products to plasma. The plasma is subsequently accelerated through the accelerator nozzle to deliver an impulse bit. A schematic representation of one of the dual parallel rail accelerator nozzles is shown in Fig. 4.

The terminals of the energy storage capacitor are connected by a low inductance, low resistance path to the anode and to

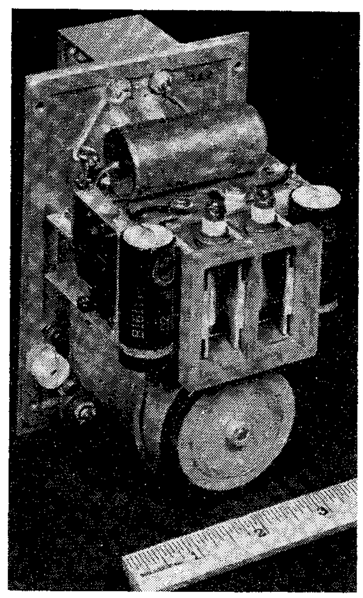

Fig. 3 Flight thruster removed from housing.

the cathode of the thruster nozzle, respectively. A fuel retaining shoulder is machined into the anode in order to properly position the Teflon propellant rod with respect to the thruster nozzle. A constant force steadily provided by a Negator spring assures that an edge of the propellant is held against the fuel retaining shoulder independent of thruster attitude, thermal environment, absence of gravity or duration of thruster operation.

To generate an impulse bit, the discharge initiating capacitor and the energy storage capacitor are charged to operating voltages of 550 v and 1360 v, respectively, starting about 1.5 sec before firing. These voltage levels appear across both thruster-nozzles of the dual nozzle thruster. Because a vacuum exists in the interelectrode spacing, the applied voltage is retained until a command signal from the satellite logic and VAT Driver arrives at one of the dual discharge initiating circuits of the thruster to be operated. The particular discharge initiating circuit which is triggered generates a microdischarge by means of a discharge initiating plug in the thruster-nozzle to which it is assigned to activate. This "microdischarge" occurs at roughly 1650 v at the plug and allows the main thruster energy storage capacitor to deliver energy into the corresponding thruster nozzle. The other seminozzle remains passive until it is activated when the satellite assumes the same orientation one complete rotation later. The energy discharged into each thruster nozzle depolymerizes surface layers of Teflon which in turn become energized and ejected as a plasma through the thruster nozzle.

Since the Teflon also depolymerizes behind the fuel retaining shoulder, the Negator spring replenishes the consumed propellant by moving it into the region depleted by the previous discharge. This propellant movement is essentially imperceptible to the eye.

A flip-flop circuit in the VAT Driver circuit alternates the thruster-nozzle being discharged in each of the four dual nozzle thrusters, one each satellite revolution. In the event of a failure of one of the seminozzles, the total thrust level generated by the four thrusters is reduced by only $\frac{1}{8}$ of the system total thrust capability.

Since the thruster is positioned in a housing which also serves as a main structural member of the satellite, the entire thruster and its discharge initiating electronics are mounted off the rear cover of this housing (see Fig. 3). In this manner the thruster can readily be removed from its housing for inspection. A support assembly at the front face of the thruster housing takes up any movement and loads that may be induced on the thruster assembly during shock and vibration loading. Particular attention was given to the transition from the thruster nozzle exit into the thruster housing exhaust cone to preclude plasma from contacting the thruster housing. This precaution prevents the electric discharge from generating a ground loop and possible RFI. Similarly, the entire thruster is isolated from the housing with the only ground connection of the system occurring through an inductor. This inductor is seen in Fig. 3 as a white cylindrical button in the lower left corner of the rear cover. Even though inadequate laboratory data exists to indicate whether such severe isolation is required, the extra precautions introduced preclude any likely interference from the electric thruster system with other electronic equipment on the satellite via ground loops or electrical conduction paths.

B. Power Conditioner

In order to keep volume and weight at a minimum, one power conditioner with four output channels is used to charge the thruster capacitors. The simplest conceivable power conditioner is one in which all four thruster capacitors are always kept on charge with each discharged thruster capacitor being immediately recharged. Simple calculations show that for such an operation the required d.c. life of the capacitor over a 5-yr mission would be significantly greater than provided by state-of-the-art capacitors. Therefore, the power conditioner was designed to charge only the thruster capacitor that was about to be discharged. The other three thruster capacitors remain uncharged in the interim. With this latter feature it is also a simple matter to bypass a thruster system in the event of complete failure of one of the four thrusters.

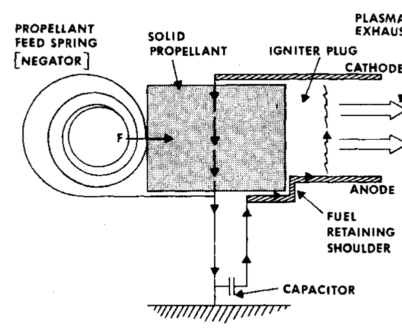

Fig. 4. Thruster nozzle and propellant subsystem schematic.

The power conditioner was designed and built by Wilmore Electronics Inc., according to specifications provided by MIT Lincoln Laboratory and Fairchild Hiller/Republic Aviation Division. Details of the principle of operation of the power conditioner are presented elsewhere.[2] Basically, with the constant power or constant rate of energy transfer technique that is being used, small energy bits are switched at a rate of 5 Kc into the primary of an energy storage transformer. The secondary winding of this transformer, the "flyback" circuit, appears as a current source rather than a voltage source, thus enabling it to deliver energy efficiently to the thruster energy capacitor. Since the load and the source are never directly coupled to each other the primary circuit becomes protected from load short circuits such as occurs during operation of the thruster. Four main outputs are provided to sequentially charge the 2 μ f thruster capacitors to 1360 v $\pm 5\%$ in less than 1.2 sec. Four additional auxiliary outputs are provided to sequentially charge the 1 μ f discharge initiating capacitor for each thruster discharge initiating circuit to 550 v $\pm 9\%$. Each of the 550-v outputs is charged in unison with its corresponding 1360-v output. A fuse located in each of the four channels of the power conditioner will isolate any given channel from solar bus power in the event of a power transistor failure in that channel. Telemetry outputs are provided for each of the eight outputs in order to be able to monitor the thruster circuit and discharge initiating circuit of each of the four thrusters. The final weight of the power conditioner is 873 g.

The allowable input voltage to the power conditioners can range from 18 to 26 v d.c. The power conditioner will automatically be disabled by an under-voltage cut-off activated below 14 v and the power conditioner can survive operation as high as 40 v.

An input current monitor and output telemetry signal is also provided to indicate power conditioner current consumption from the solar bus.

On the basis that once the entire electric thruster system is activated, the four thrusters will generate 6 μlb-sec impulse bits once every 1.5 sec (i.e., each of the four thrusters will operate once every 6 sec). The calculated power budget for maintaining the four thruster systems active and providing telemetry outputs is presented in Table 1. Actual power input consumption measurements of each of the three power conditioners that were built are also presented in Table 1. It is seen that actual power consumption falls below calculated values. For an overall input power of 2.5 w (including telemetry, etc.) the four thrusters generate an average steady-state thrust of 4 μlb.

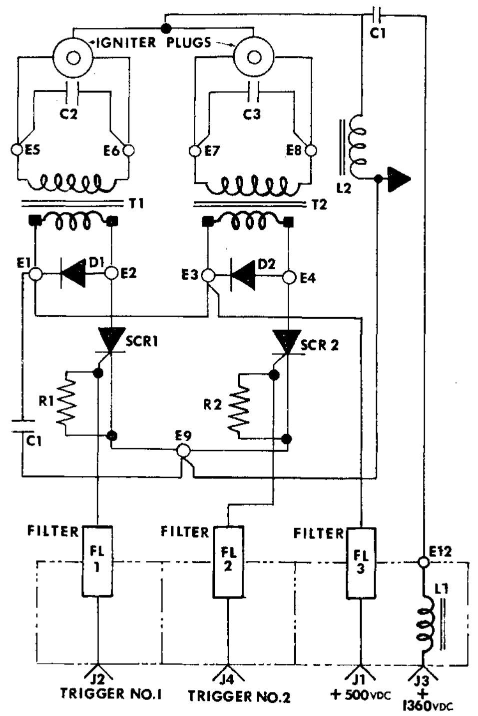

Fig. 5 Discharge initiating circuit schematic.

C. Discharge Initiating Circuit

As indicated previously, the function of the discharge initiating circuit is to energize a discharge initiating surface ignitor plug which in turn discharges the main thruster capacitor. In order to minimize over-all power consumption the discharge energy level was selected to be such that adequate derating of all components of the circuit would be achieved and yet the energy level should be capable of initiating the discharge of each thruster during its intended life. Figure 5 presents a schematic of the circuit that it used to discharge either of the two accelerator nozzles of a given thruster. The components of this circuit are mounted on a chassis and surround the two Teflon propellant rods of the thruster as can be seen in Fig. 3. In prototype tests of the thruster system, failure of an SCR occurred because of inadequate isolation from the chassis. A machined Teflon bushing design subsequently used in the flight hardware eliminated the possibility of the recurrence of such a failure. Upon completion of assembly, the entire discharge initiating chassis is conformal coated, black light inspected and subjected to an overvoltage check at several voltage levels above the operational voltage.

VAT Driver and Associated Logic

The thruster system is designed to operate automatically or on ground command. A system of optical sensors and timing devices monitor the relative position of the satellite and turn on the thruster system as the satellite drifts through a predetermined angle. When operation of the thrusters is required an "enable" signal is generated by Earth sensors 90° of satellite rotation before thrust. This enable signal activates the main stationkeeping logic which generates a 6-v turn-on signal activating the power conditioner, and four distinct enable signals to the VAT Driver that contains the thruster logic, as shown in Fig. 6.

Since the reliability of a capacitor is a function of voltage stress and time-on stress it was decided early in the program to leave the capacitors of each thruster uncharged until immediately before firing. Four logic lines were required then to sequentially charge and discharge each individual thruster.

The logic circuitry for charging and discharging the thrusters was packaged separately in the VAT Driver to prevent RFI generated by the thruster from reaching the main stationkeeping logic. Feed-through filters were used on the output of the trigger lines from the VAT Driver and also at each trigger input to the thrusters. Semirigid coax wiring was used for the trigger and charging lines while No. 26AWG wire was used for the logic lines.

The enable signals from the main stationkeeping logic to the VAT Driver consist of 4 low-level sequential pulses each lasting for a period equivalent to 90° of satellite rotation. These signals are amplified by the VAT Driver and sent to the respective channels of the power conditioner which in turn sequentially charge the thrusters. Since each thruster contains two nozzles (a and b) that are fired alternately on successive firings it was necessary to include a flip-flop device in the logic circuitry. Figure 7 shows the form of the logic pulses from the VAT Driver to the power conditioner and the

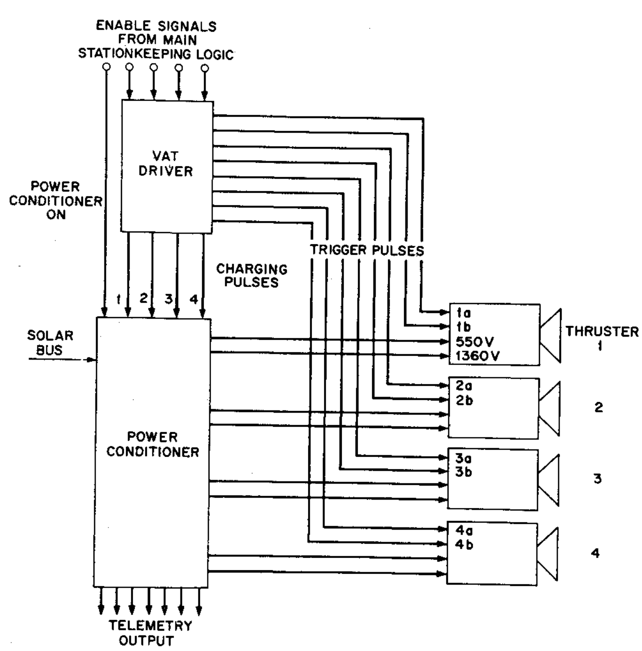

Fig. 6 Thruster system block diagram.

Table 1 Power allocation of power conditioner

Output load	Usable power out, w (nominal)	Nominal conversion efficiency	Nominal power required, w	Maximum power required, w
4-2.0 μf capacitors (charged to 1360v)	1.233	75%	1.64	2.47
4-1.0 μf capacitors (charged to 550v)	0.0833	50%	0.1667	0.208
550v leakage power	0.001	50%	0.02	0.20
Telemetry power				
1360v telemetry	0.0544 (max)	75%	0.0725	0.0725
550v telemetry	0.02 (max)	50%	0.04	0.04
Total power conditioner loading	1.401	...	1.939	2.99

Input power measurements

Temperature, °C	Serial no.	Power, w
+40	1, 2, 3	2.574, 2.504, 2.596
+25	1, 2, 3	2.488, 2.472, 2.444
−20	2, 3	2.434, 2.468
−40	1, 2	2.421, 2.36

trigger pulses to the thrusters. Trigger pulses of 30 μsec duration were generated at the leading edge of the successive charging pulse to fire the respective nozzle of each thruster. The flip-flop is activated at the end of the 4th charging pulse to convert the trigger pulse on the next firing cycle to nozzle *b* of each thruster.

RFI Suppression and Measurements

It was desired to limit the thruster system rf signal to -110 dbm/Hz in the UHF band relative to the receiver input terminals. Signals greater than 50 μsec duration above -169 dbm/Hz would have caused severe interference. It was not known whether these requirements could be met by a pulsed microthruster operating at high voltage.

A prototype thruster was built early in the program to provide RFI information. The engine was mounted in a drawn aluminum case having dimensions similar to the final thruster housing. The engine construction was representative of the final flight unit although some electrical components were later changed and the igniter plug configuration modified to improve reliability in the flight unit.

To supplement information derived by RFI tests, a calibrated RCA-1P42 vacuum photo tube and a high speed Tektronix voltage probe were used to examine the duration of the luminous discharge and the duration of discharge voltage variation, respectively. The photo tube was positioned to look along the thrust axis directly into the thruster nozzle. The voltage probe output and the photo tube output were displayed simultaneously on a dual beam oscilloscope. The capacitor was found to be fully discharged in 2.2 μsec, while the plasma persisted a little over 10 μsec with its peak light intensity occurring roughly 1.45 μsec after discharge initiation.

Obtaining meaningful RFI information is difficult because actual flight antenna coupling cannot always be accurately simulated and also because reflections from the environment often invalidate the measurements. Preliminary RFI measurements were made in a vacuum bell jar 18 in. in diameter by 30 in. high. The flight antenna system had not been completed at this time. Measurements were taken using a mono-pole antenna, a Hewlett Packard Spectrum Analyzer having a wide dispersion, and a low-noise preamplifier. Measurements were taken referenced both to a 100 kHz bandwidth and a MHz bandwidth with the antenna directly in front of the exhaust. It was found that the power spectrum was flat within these bandwidths. Measurements were taken with the antenna both inside and outside of the bell jar. No evidence of antenna and plasma interaction was observed.

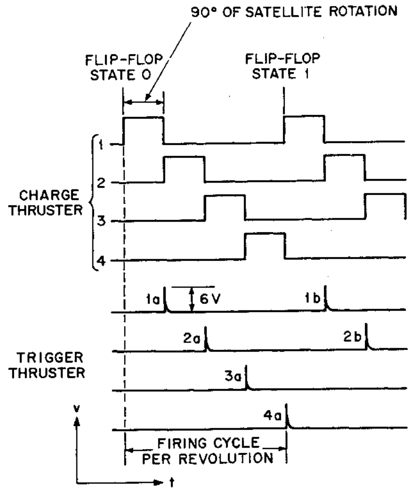

Fig. 7 Pulse waveforms from VAT driver.

The maximum rf signal measured in the bell jar was approximately -136 dbm/Hz. It was not possible to accurately determine the time duration because of reflections from boundaries of the test facility. It was noted at this time that successive firings produced varying spectra. A number of measurements were taken through successive firings to determine the maximum intensity.

Tests were later conducted using the flight antenna system. A flight model of the satellite was available so that the thruster was mounted to reproduce the actual antenna coupling. Tests were conducted in a vacuum chamber 8 ft in diameter by 12 ft long with an absorbing screen separating the thruster/antenna system from the tank walls. The maximum rf signal measured with the flight (dipole) antenna system was -143 dbm/Hz. The absorbing screen did not eliminate all reflections and the actual pulse duration could not be determined accurately.

In order to obtain a meaningful measurement of the time duration of the RF signal the thruster was mounted in a 4-ft vacuum chamber lined with ferrite absorber material. The lining was successful in eliminating reflections. Using a monopole antenna attached to a ground plane mounted inside the chamber, the maximum time duration above -169 dbm/Hz was found to be 15 μsec. Experiments were conducted to determine the effectiveness of the epoxy coating on the inside surfaces of the thruster exhaust transition in suppressing RFI. An uncoated cone produced RFI signals 12 db higher than in the coated cone.

Peak pulse power measurements indicated that the plasma RFI coupling to the monopole antenna was the same as that

Table 2 Exploratory laboratory life tests

Test no.	Testing period	Pulse frequency, Hz	Impulse per discharge, μ lb-sec	Thrust level, μ lb	Specific impulse,[a] sec	Discharges of test	Hours of operation, hr
96-A	10/27/67– 11/21/67	1	5.86	5.86	215	2,125,424	590
96-B	11/22/67– 12/19/67	1	7.1	7.1	251	2,270,898	630
99-A	12/12/67– 1/03/68	1 increased to 1.5	4.55	4.55 at 1 Hz	319	(2,076,348)	506
99-B	1/05/68– 1/22/68	1.5	[b]	...	...	4,339,132 Δ(2,262,784)	926 Δ(420)
96-C	1/08/68–	4	[b]	...	...	2,423,292	168
			Flight prototype life tests				
96-D	1/16/68– 2/23/68	4	[b]	...	...	11,013,985	760
101-A	2/21/68	1.5	5.36	8.04	308	8,744,277	1600

[a] Specific impulse is calculated using the test averaged thrust and the test averaged propellant weight flow rate.
[b] Thrust stand not available because of other contractual commitments.

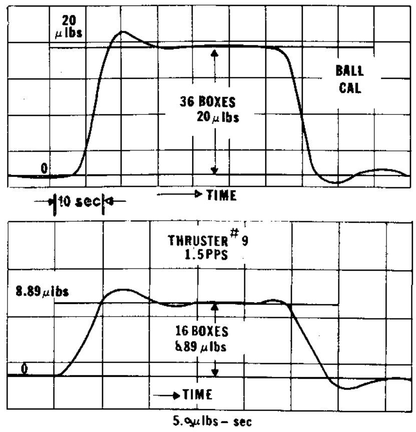

Fig. 8 Typical calibration and thrust level traces.

to the satellite antenna system. It was concluded that the RFI time duration above −169 dbm/Hz of 15 μsec and a peak power density of −143 dbm/Hz is what the satellite receiver would see in a reflectionless environment.

Prototype Thruster Accelerated Life Tests

If each thruster was to be laboratory life tested at the design pulse repetition rate and at 100% duty cycle, the life test would last roughly two years. An accelerated rate of thruster operation was therefore necessary to learn at an early date in the program any operational difficulties or failures likely to be encountered by the flight system in meeting the required total number of discharges of the mission. The accelerated pulse rate of any life test should be meaningful in that it must reasonably simulate conditions encountered at the pulse repetition rate of the flight hardware.

Two criteria were considered to be meaningful: 1) the system temperature during the accelerated life test should remain reasonably close to the temperature of the system during unaccelerated testing; 2) the pulse repetition rate during accelerated testing should be low enough to prevent possible environmental dependency between discharges. Over-all thruster pulse repetition rates of from 1 to 4 Hz were considered useful during these accelerated tests. At these rates, each thruster nozzle would operate at a rate varying from 0.5 to 2 Hz, respectively.

Table 2 shows the results of these accelerated tests. The first series of tests denoted by 96A, B, C, and 99A, B, were all carried out with the arc discharge initiating spot permitted to occur at random over a relatively wide area of the cathode. Under these circumstances it was found that the arc spot would tend to move to a new location after remaining very nearly fixed at one location for some length of time. Whenever one side of the dual nozzle thruster became intermittent it was invariably found that the arc discharge initiating spot was covered by the propellant. A minor redesign was carried out to limit movement of the arc spot. The first exploratory test (test 96-C) with this modification proved inconclusive since vacuum was lost in the test chamber over the weekend and resulting electrical shorts necessitated repeating the test. A repeat of the test (test 96-D) revealed that the design goal of 1.2×10^7 discharges was essentially capable of being attained by the thruster by limiting the arc spot movement. After 760 hr of operation at a pulse rate of 4 Hz the nonflight capacitor failed after having been subjected to 14,665,281 discharges. In order to confirm the results at a lower pulse rate, the test was repeated at a rate of 1.5 Hz. After 1600 hr of operation, corresponding to 8,744,277 consecutive discharges, the test had to be stopped because the test facility was needed for qualification and acceptance testing of the flight hardware. The over-all test average specific impulse was measured to be 308 sec with the test average impulse bit 5.36μlb-sec.

During accelerated life tests it was learned that a chemical reaction apparently takes place between the plasma and backstreamed diffusion pump oil and that this reaction adversely affects thruster life by forming conducting deposits in undesirable locations. None of the tests were carried out in vacuum chambers having liquid nitrogen cold traps. A similar reaction was also found to occur when recently cured epoxy was in contact with the plasma. In retrospect, it would appear highly desirable to perform laboratory life tests in an ion pumped vacuum chamber having sorption roughing to reliably establish maximum system life capability. Furthermore, it was necessary to cure epoxy for several days after which it was vacuum cured and vacuum baked.

Flight Qualification and Acceptance Tests

Qualification and acceptance tests of the thruster subsystem and of the power conditioner were carried out in accordance with MIT Lincoln Laboratory Environmental Test Specification ETS-4, Lincoln Experimental Satellite, LES-6. Quite generally, this document spells out the qualification and acceptance shock and vibration tests of a Titan III-C launch environment, as well as vacuum temperature tests and an ambient humidity test. In addition to the test requirements of the latter document, detailed performance and functional tests were carried out of the power conditioner, of the thruster, and of individual components of the thruster subsystem.

The qualification thruster and power conditioner were subjected to shock and vibration qualification tests without evidence of any physical damage. The shock tests were performed on a modified Barry Shock machine to produce a $\frac{1}{2}$ sine wave pulse of 45 g's for a duration of 4 msec. This shock is equivalent to the energy produced by 850 g's for 0.2 msec. The qualification units were subjected to three drops in each of the six orthogonal directions.

All qualification and acceptance vibration tests were performed on an MB-C-70 electrodynamic shaker using a magnetic tape input. The random motion spectrum was shaped to the specification set forth by Lincoln Laboratory document ETS-4. A spectral analysis of the input and output was made to assure compliance with specification.

The qualification vibration test level was 18.8 g's rms for 3 min while the acceptance test level was 11.9 g's rms for 1-min duration. All thrusters and power conditions were tested in each of three orthogonal axes.

Since the thruster can operate only in a vacuum and will not be operated during the launch, thruster performance tests in a vacuum were carried out only before and after shock and vibration testing. Each thruster was operated on a thrust balance located in a vacuum chamber for 4 hr before and another after shock and vibration at a pulse rate of 1.5 Hz instead of the design pulse rate of 0.166 Hz. This increased

Table 3 Measured thrust levels

Test condition	Serial no.	Thrust (at 1.5 Hz) before/after μlb	Impulse bit before/after μlb-sec
Qualification shock and vibration	1	7.6/7.6	5.05/5.05
Acceptance vibration	2	8.9/8.55	5.95/5.7
	3	8.55[a]/8.8	5.7[a]/5.9
	4	8.95/9.1	5.98/6.05
	5	9.1/9.1	6.05/6.05
	6	9.1/9.1	6.05/6.05
	7	8.8/9.1	5.87/6.05
	8	8.45/8.8	5.62/5.9
	9	8.55/8.89	5.7/5.9

[a] Retest; first test terminated because connector failed.

pulse rate was used in order to obtain reliable and more accurate thrust measurements at the correspondingly higher thrust level than possible at the 0.166 Hz pulse rate. Figure 8 shows a typical 20μlb calibration trace and thrust stand recording of the thrust level of thruster No. 9 after vibration. Table 3 presents measured thrust levels of the thrusters before and after these environmental tests. All changes in performance before and after shock and vibration is within the reading accuracy and the accuracy to which the pulse repetition rate can be held. The lower thrust level of thruster No. 1, the qualification thruster, is believed due to a slightly lower pulse rate that may have been used in carrying out the test.

Temperature vacuum tests were conducted on the thruster and the power conditioner at temperatures of $-40°$F and $+140°$F. During operation in this vacuum temperature environment, performance checks were made of the power conditioner for normal sequential commanding and loading and of arbitrary commanding and loading. It was found possible to perform these tests within the specified temperature range whenever the power conditioner was tested by itself and operated into simulated loads. Because the surface emittance of the thruster was vastly different from that of the power conditioner it was not always possible to maintain both units within allowable temperature during concurrent testing in the vacuum chamber. During overvoltage survival checks the thruster reached and was operated at a temperature of as low as $-70°$F in order that normal temperature could be maintained on the power conditioner. No problems in thruster or power conditioner operation were encountered in any of the thermal vacuum tests.

Besides the interaction of plasma with either back streamed diffusion pump oil or with freshly cured epoxy, the only serious problem area that was encountered during qualification and acceptance testing was with miniature high-voltage coaxial connectors (up to 1500 v) designed for long time vacuum operation. A high voltage breakdown occurred in several of the miniature high-voltage connectors that were used. The technique adapted to overcome this problem area was to provide bleed holes to release trapped air whenever feasible and also to subject the system for at least 4 hr to a high vacuum before activating the system.

Thruster System Integration and Testing

All thrusters and power conditioners were vacuum baked for 8 hr at a pressure of 1×10^{-5} torr and a temperature of 40°C to insure that the depolymerization process would not be affected by contaminants due to outgassing.

Extreme care was taken in preparing the thruster system for installation onto the flight platform. As a safety precaution shorting plugs were made and attached to the high-voltage connectors during storage and installation to insure that no residual voltage remained on the capacitors. Protective covers were made to fit over the exhaust area during final assembly of the satellite.

System checkout was conducted using specially constructed high voltage probes that were designed to fit each thruster nozzle. The charging sequence and voltage levels were checked using the probe while a visual and audible check was made to determine whether the discharge initiating circuitry and igniter plug discharge were consistent with the logic sequence.

Thorough systems testing indicated the need for a minor modification of the thruster logic. The major difficulty encountered during vacuum testing, however, was high voltage breakdown at the 1360 v connector. A similar failure was experienced during acceptance testing and at the time was attributed to a faulty connector and considered a singular occurrence. Individual connectors had been tested to 5000 v without any sign of breakdown.

These failures resulted in carbonization on the connector and mating pins and occurred after holding vacuum at 1×10^{-5} torr for as long as 5 hr before firing the thrusters. It was felt that the failure was due in part to potting compound on the back side of the connector causing air entrapment at the interface of the connector pin and insulation bushing. The potting compound which had been applied primarily to insure against arcing, was removed. In addition, small vent holes were drilled in the cover plates of the high voltage compartments. After these modifications vacuum was held for approximately 8 hr after which all thrusters were fired successfully. There was no evidence of arcing and it was found that RFI generated from the thruster system was not detrimental to any of the onboard experiments or systems.

The LES-6 experimental communications satellite was launched into synchronous orbit September 26, 1968. The four pulsed plasma microthruster propulsion system was first activated in space October 15, 1968. In more than 16 months of operation the complete system has operated flawlessly without interfering with either the telemetry or the communications system of the satellite. The thrusters have also operated normally during the strong solar flare activity occurring during the last week of October 1968. From orbital data[3] it has been established that the thrusters have functionally performed east-west stationkeeping. These results represent major milestones in electric propulsion technology.

References

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