Performance of a Fifteen-Centimeter Ion Thruster with Reliable Restart Capability

Paul J. Wilbur*
Colorado State University, Fort Collins, Colo.

Experimental evaluation of a modified SERT II thruster with dished grids capable of producing a 600 ma ion beam at 2690 sec specific impulse and achieving rapid, reliable startup is described. In addition to steady-state performance curves, thruster performance is examined over the duration of rapid startup and one hr operating sequence. Rapid startup utilizing high voltage spark ignition of the main and neutralizer cathodes is described. Minimal deterioration of the cathodes and the ignition system during 1000 neutralizer cathode startups and 2000 main cathode startups is demonstrated. The need for heaters and a vaporizer control system to facilitate more rapid startup is identified.

Introduction

APPRECIABLE cost or weight savings can be achieved in synchronous orbit satellite systems if ion thrusters are used to produce the associated North-South stationkeeping requirement instead of a chemical propellant thruster.[1] For a typical synchronous communication satellite having a mass of 700 kg, the 46 m/sec annual North-South stationkeeping velocity increment can be met by a mercury ion thruster operating at a 600 ma beam current level and 1000 v thruster potential on a duty cycle of 230 hr per year. Assuming 60% over-all electrical efficiency, the thruster could satisfy this requirement by operating for less than an hour a day from a 1 kw-hr battery which could be charged continuously from a small solar array ($\sim$42 w output). For such a thruster system to perform effectively, however, it must be capable of repeated startup which is both rapid and reliable and be able to produce a 600 ma ion beam at 1000 v thruster potential with acceptable propellant utilization and electrical efficiencies. Electron bombardment ion thruster startup has required heating of the hollow cathode tip to a sufficiently high temperature to permit spontaneous ignition of the keeper discharge. This mode of startup is frequently unreliable because the cathode tip temperatures and times required to start the discharge tend to vary with the history of the cathode. Application of a high voltage discharge system developed by Wintucky[2] to ignite the keeper discharge is proposed as a means of correcting this reliability deficiency. A SERT II thruster[3] modified to employ high perveance dished grids is proposed as a thruster capable of producing the necessary ion beam current at 1000 v thruster potential.

Steady-State Thruster Performance

In order to facilitate laboratory testing and evolution of a thruster suitable for the aforementioned mission, a standard SERT II thruster[3] was modified to include independent main and hollow cathode flow control and measurement systems, electromagnets, a solid disk tantalum baffle, a $\frac{1}{8}$ in. diam. hollow cathode with a 0.05 cm diam. orifice and dished grids. This cathode orifice which is larger than the one used on SERT II was required to facilitate sustained operation at the high arc current required to achieve the high beam current capability of the dished grids. The dished grids used in the study had 0.20 cm diam. apertures in both the accel and screen grids on 0.254 cm centers. They were separated by 0.079 cm and dished outward 0.5 cm at their centers. These grids have a 67% open area and for all tests described herein the screen and accel grid potentials were held at 1000 v and -500 v, respectively; the screen potential being selected as sufficiently low to effect a small thruster-power system mass and the accel voltage being sufficient to prevent electron backstreaming without causing excessive ion beamlet defocusing at this screen grid potential. A discussion of the relative benefits of these modifications and performance comparison with the unmodified SERT II thruster is presented in Refs. 4 and 5.

Steady-state testing of the thruster yields the performance curves presented in Fig. 1 for the total flow rates expressed in equivalent ma and designated by the symbol $\dot{m}_T$. The cathode flow rates designated by the symbol $\dot{m}_c$ are constant along each curve. The indicated values were selected at each total flow rate to produce an arc voltage in the range 36–37 v (designated by the solid symbols) near the knee of each performance curve. For the purposes of this paper the high flow rate case is of primary interest as it results in an ion beam current of 630 ma at 85% utilization. For the screen grid voltage of 1000 v this corresponds to a specific impulse of 2690 sec.

The ion beam profile measured at 85% utilization, a 630 ma beam current condition and with the hollow cathode neutralizer operating at this same emission current is shown in Fig. 2. Beam current densities in this figure were measured using a Faraday probe located 5 cm downstream of the accel grid. Integration of the current density profile shown over the beam cross-sectional area yields a beam current which agrees with the measured beam

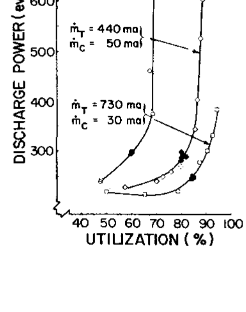

Fig. 1 Dished grid—15 cm thruster performance.

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* Associate Professor, Mechanical Engineering Department. Member AIAA.

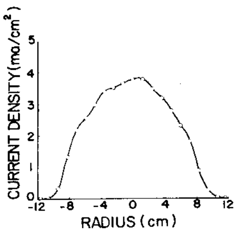

Fig. 2 Ion beam current density profile (630 ma beam current).

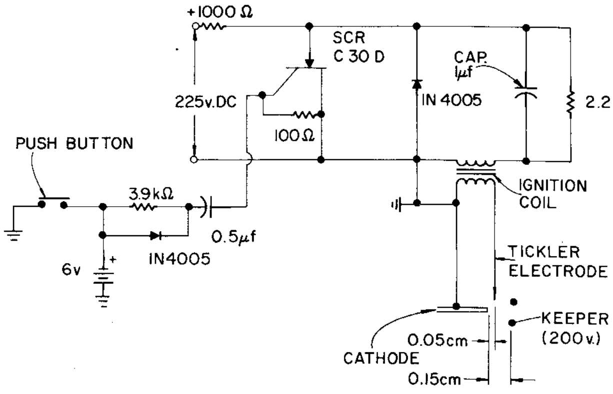

Fig. 3 Spark ignition system circuitry.

current to within 5%. This plot shows a significant current density in the region outside of the thruster radius (7.5 cm). This ion beam divergence, caused by electrostatic deflection of the ion beamlets in an improperly compensated dished grid system, has been eliminated by improved grid fabrication techniques.[6]

High Voltage Ignition System

In order to achieve the rapid, reproducible startup necessary for a thruster on a North-South stationkeeping mission the high-voltage ignition system designed by Wintucky[2] was installed. This system, shown schematically in Fig. 3, utilizes a 1 μf capacitor which discharges through the secondary winding of a standard 12 v automobile ignition coil to produce a high voltage pulse. This pulse is applied to a tickler electrode located between the cathode and keeper electrodes across which the discharge is to be ignited. Actuation of the pulse occurs when the push button circuit is closed thereby triggering the silicon controlled rectifier (SCR) C30D shown in Fig. 3. Details of the mechanism of operation and advantages of this system as well as a study of effects of parametric variations of the system are presented by Wintucky.[2]

The tickler used in these experiments, a 1 mm diam, 1% thoriated tungsten welding electrode with its tip ground to a sharp point, was installed perpendicular to the cathode axis, 0.05 cm from the cathode tip. Its installation resulted in no measurable change in thruster performance from that shown in Fig. 1. Preliminary studies using a cathode-keeper assembly at a cathode flow rate of ~30 ma demonstrated: 1) a 225 v d.c. charging capacitor voltage was sufficient to insure electrical breakdown between the tickler and the cathode, 2) a 200 v d.c. keeper potential was sufficient to insure subsequent current flow between the cathode and keeper and 3) a cathode temperature greater than 400°C resulted in an arc rather than a glow discharge between the cathode and keeper.

The only change in the circuitry shown in Fig. 3 from that suggested by Wintucky is the addition of the 1000 Ω resistor shown in the power line supplying the discharge capacitor. This resistor was found to be necessary to limit the rate at which the discharge capacitor was recharged after the SCR was tripped. Without the resistor the capacitor would frequently not remain discharged for a sufficiently long period of time to allow the SCR to reset and in this state the SCR represented a permanent short circuit across the power supply.

Testing of the system under conditions where breakdown did not occur at the tickler electrode resulted in high-voltage breakdown through the Bakelite between the primary and secondary terminals on the ignition coil and demonstrated the lowest resistance to high-voltage breakdown at this point in the system.

Repetitive Startup Cycle Test Results

In order to demonstrate the suitability of the thruster and ignition system for repeated startup/shutdown cycle operation, two tests were conducted. The first one involved establishing all mercury flow conditions and the neutralizer discharge and then performing 1000 startup and shutdown cycles of the main discharge while the second series of tests involved establishing all flow rate conditions and then performing 1000 cycles of neutralizer followed by main discharge startup with subsequent termination of both discharges.

The main cathode used in these tests was first installed in the thruster several months before the tests mentioned above. The cathode insert had been dipped in triple carbonate mixture (R-500) before cathode installation and no additional carbonate was added after this initial coating. The cathode had been operated for about 130 hr at the time the first test sequence began. The neutralizer cathode is a standard SERT II model; it had been operated for over 250 hr and had not been supplied with carbonate material beyond that applied at the time of fabrication for the SERT II flight program.

In conducting the first series of tests the main, cathode, and neutralizer flow rates were established at 700 ma, 30 ma, and ~ 20 ma, respectively, and the neutralizer discharge was established at 0.2 amps. A keeper voltage was then established at 200 v and the arc power supply was set to limit the arc current at about 3.9 amps once the arc discharge started. The cathode tip temperature was next raised to 500°C, a value selected as sufficiently far above the necessary temperature for startup (400°C) to assure a high startup reliability. The spark ignition system was actuated next, the keeper and arc discharges both established themselves and the keeper current was then adjusted to 0.3 amps. The high voltage was turned on next to establish a beam current and the beam current, arc voltage, arc current, keeper voltage, and cathode tip temperature were recorded. The high voltage, keeper, and arc power supplies were subsequently turned off and when the cathode tip had cooled to 500°C, the spark ignition system was actuated again and the sequence was repeated until 1000 cycles had been completed. Two typical startup-shutdown cycles showing the times required to complete each portion of the cycle are presented as Fig. 4, which shows

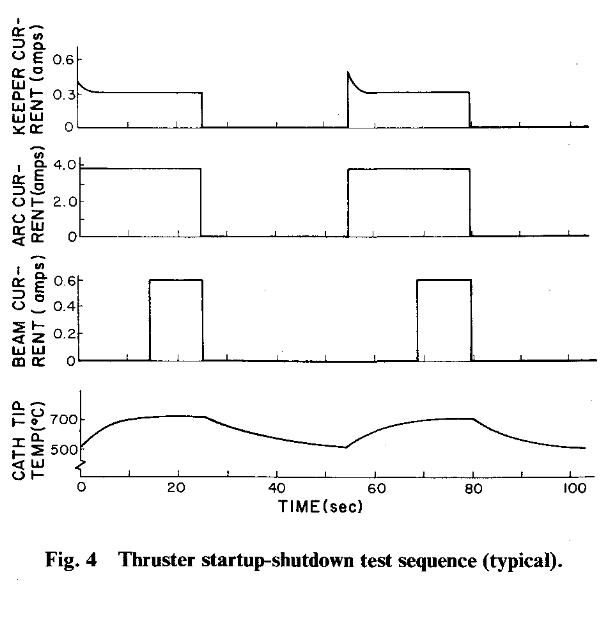

Fig. 4 Thruster startup-shutdown test sequence (typical).

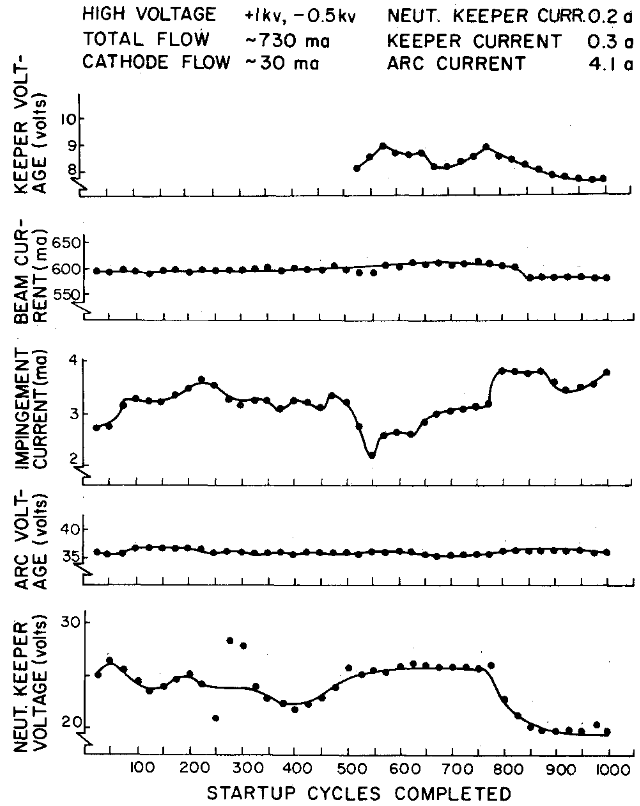

Fig. 5 Effect of start cycles on thruster parameters.

that the typical cycle was completed in less than one minute. This test sequence demonstrated the main discharge startup probabilities shown as follows: frequency of startup on first actuation—99.0%, frequency of startup on second actuation—0.8%, and frequency of startup on third actuation—0.2%.

The second startup test sequence involved starting the neutralizer with the high voltage ignition system at a neutralizer flow rate of ~ 20 ma a few seconds before the main neutralizer was started and then proceeding in the manner suggested by Fig. 4. The neutralizer ignition system was identical to the one used with the main cathode. The frequency of discharge initiation over a 1000 cycle neutralizer/main discharge startup and shutdown sequence is shown in Table 1.

Table 1 Frequency of discharge initiation

	Neutralizer discharge	Main discharge
Frequency of startup on first actuation	84.7%	95.0%
Frequency of startup on second actuation	5.6%	3.1%
Frequency of startup requiring three to five actuations	4.4%	1.3%
Frequency of startup where more than five actuations were required	5.3%	0.6%

When more than five spark ignition actuations were required to start the discharge a component or operator malfunction could generally be identified. These malfunctions included: 1) improper positioning of a switch used to connect either the main or neutralizer tickler to the ignition coil output, 2) low cathode tip temperature or flow rate, and most frequently, 3) failure of the SCR to reset following the previous ignition. This failure of the SCR to reset was corrected by the installation of a 1000 Ω resistor in the line from the power supply used to charge the discharge capacitor as discussed previously.

The variations in neutralizer keeper voltage, keeper voltage, impingement current, beam current, and arc voltage achieved after each startup during this test are shown in Fig. 5 as a function of the number of startup cycles executed. These data show some variations in impingement current, keeper voltage, neutralizer keeper voltage, and beam current. The variations are considered to be due to corresponding variations in flow rates which were observed to vary as much as ± 15 ma for main flow and ±5 ma for cathode flow rates during the test.

Photographs were made of the cathodes after each test sequence.[5] They revealed minor erosion of the edge of the main cathode orifice plate where the tickler passed over it after the first 1000 cycles. The tickler showed an accumulation of metal at this location. No additional erosion of the main cathode was observed during the second 1000 cycles. The neutralizer also showed no evidence of erosion.

A comparison of the data on startup reliability of the main cathode for the two tests suggests the main cathode was more difficult to start during the second thousand tests. After the second thousand cycles both cathodes were sent out to be photographed and when they were returned neither could be started at the flow rates and temperatures used in these tests. Both cathodes were subsequently coated with triple carbonate mixture and they restarted readily. Although variations in cathode startup characteristics are common,[7-9] it is believed in this case that exposure to the aromatic chemical vapors in the photographic laboratory where the cathodes were stored for about a week while they were waiting to be photographed after each test series caused a degradation in startup reliability.

Rapid Startup Study

In order to prevent excessive propellant loss during a thruster duty cycle, it is necessary to be able to achieve rapid startup and shutdown of the system. Shutdown is not considered a particularly severe problem because the thruster can continue to operate and produce thrust after the vaporizers have been shut down and until the flow rates drop to rather low values. Startup on the other hand requires a substantial flow rate to facilitate ignition, and propellant is lost during the time period when this flow is being established.

A preliminary study of the cold startup to operation characteristics of the modified SERT II thruster indicated the neutralizer and main cathodes could be started and then the arc discharge and the beam current could be established within four minutes after power was first supplied to the thruster. Within about three minutes after startup, the thruster was operating at acceptable beam and arc current conditions, but it was not until 12 min after startup, when flow rates had reached more nearly correct values, that performance parameters reached their steady-state values. Flow rates were controlled manually during this test, and it is anticipated that an automatic control system possibly coupled with higher power heaters capable of raising vaporizer and cathode temperatures more rapidly will facilitate more rapid startup. Additional testing to determine propellant utilization over the startup-shutdown cycle is required.

Conclusion

A 15-cm mercury ion thruster operating with dished grids is capable of producing a 600 ma ion beam at a specific impulse of 2690 sec. Rapid, reliable ignition of the neutralizer and main discharges can be achieved at low cathode temperatures and without cathode propellant overflow using a spark ignition system. At these low cathode temperatures the loss of low work function oxides from the cathode should be negligible. Use of this ignition system over a few thousand cycles resulted in no significant degradation of the thruster components. Additional attention is needed in the area of rapidly acting vaporizers and a control system capable of bringing flow rates to near the desired values quickly in order to minimize propellant loss during startup from a thruster operated periodically.

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SEPTEMBER 1974 J. SPACECRAFT VOL. 11, NO. 9

Extendible Exit Cone Development for the C4 Third-Stage Motor

LEE CAREY*
Bell Aerospace Company, Buffalo, N.Y.

AND

RUSSELL ELLIS†
United Technology Center, Sunnyvale, Calif.

This paper is a report of the Extendible Exit Cone design approach and test results. Design considerations for this actuator driven rolling metal convoluted nozzle extension are discussed. Silicide coated columbium was selected for operation at 3200°F. Simple open loop deployment control was selected and two working fluids (hydraulic oil and N_2 gas) were evaluated. Five development units were successfully tested. Two nozzles were hydraulically deployed (in bench tests) and two nozzles were deployed with N_2 gas driven actuators. The fifth unit was installed on a third stage development motor, deployed with N_2 gas and fire tested for full duration.

Introduction

THE Extendible Exit Cone (EEC) consists of a Convoluted Nozzle and a deployment system of three or more actuators. The Convoluted Nozzle (U.S. Patent 3,711,027) is an otherwise conventional sheet metal nozzle extension that is formed with a portion of the nozzle convoluted (i.e., turned inside out) to reduce the installed length of the nozzle extension to approximately one third of the deployed or operating length. The nozzle is extended by simple roll through.

The prototype design and development of the Convoluted Nozzle was reported in Refs. 1 and 2. This original effort was directed to Liquid Rocket Motor applications in upper stages and spacecraft. The optimum flight configuration of these Convoluted Nozzles were self-actuated by internal pressure retained by a jettisonable exit closure. The exit closure in effect transforms the Convoluted Nozzle into its own actuator and deployment is intrinsically self aligning. Deployment in flight would take place during a programed interval (e.g., 1 sec) between staging and upper stage or spacecraft motor ignition.

Convoluted Nozzle development since this prototype work has been directed to solid rocket motor applications in ballistic missile upper stages. In these ignition staged applications the self-actuated Convoluted Nozzle is precluded by the requirement to deploy the nozzle extension after ignition staging. These applications therefore require an extendible nozzle system composed of a Convoluted Nozzle extension and external deployment subsystem. The EEC abbreviation has been adopted herein to denote the complete system of extendible exit cone (i.e., convoluted nozzle) and deployment subsystem.

For the short burn time of solid rocket motors, design studies and tests have shown the feasibility of insulated stainless steel Convoluted Nozzles (with flexible liners). However, the radiation cooled refractory metal Convoluted Nozzle has continued to be the main line of development on the basis of its simplicity and light weight. The radiation cooled unit also retains applicability to the long burn duty cycles of liquid rocket motors.

After Ref. 2 was published, the last Prototype Convoluted Nozzle was fabricated by a shear spinning process to produce

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* Program Manager/Technical Director. Member AIAA.

† Advanced Project Manager. Member AIAA.