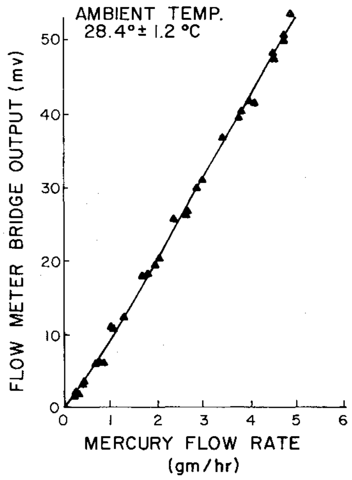

Fig. 3 Flow meter calibration curve.

Fig. 3. It shows a nearly constant sensitivity of about 10 mv/g/hr over a 0.2–5.0 g/hr flow rate range and it appears higher flow rates could be indicated. By increasing the sensitivity of the readout equipment the flow meter appears to perform satisfactorily at low flow rates (<0.3 g/hr). The scatter in the data points of Fig. 3 are considered to be due to variations in ambient temperature and scatter in flow rate measurements obtained from the glass tube.

One drawback with this device is its sensitivity to variations in ambient temperature (about −0.4 mv/°C) which could introduce significant flow rate measurement errors. This could be eliminated by holding the mercury entering the system at a fixed temperature by using a temperature controller as Pye does.[3] Theory[5] suggests however that the use of matched thermistors in the system should also eliminate the drift and since this approach would yield the simplest system, it is being pursued at the present time.

Several months of operating experience with the flow meter on an operating thruster have demonstrated that it is very useful as a tool for indicating flow rate trends. This usefulness is illustrated in Fig. 4 which displays flow meter output as a function of time for flow rate initiation—stabilization—termination cycle. After the flow meter heater had been allowed to stabilize and the bridge circuit had been zeroed, the vaporizer was energized. The figure shows a large negative flow rate associated with expansion of the mercury occurs immediately. This is followed by the establishment of a positive flow rate about 6 min after vaporizer energized, an increase in flow rate, associated with contraction ment which was necessary to achieve the desired flow rate, stablization at a steady flow rate is observed (about 15 min after vaporizer heatup began). When the vaporizer heater is de-engerized, an increase in flow rate, associated with contraction of the mercury as it cools, is observed. Subsequently the output

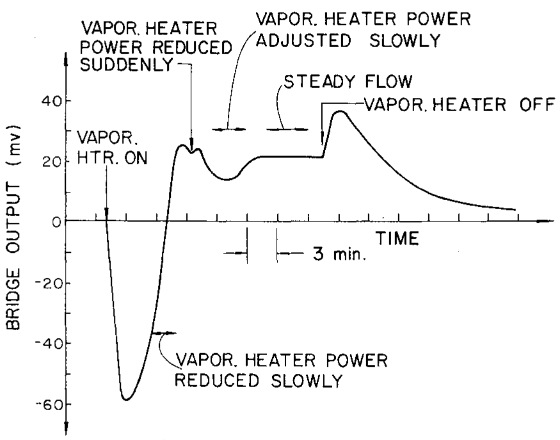

Fig. 4 Typical flow meter output in thruster feed line.

drops to zero exponentially as the flow meter temperatures stabilize back to their no-flow values.

Conclusion

A continuously indicating thermal flow meter has been made which indicates mercury flow rates in the range of interest for ion thruster testing. This device, which employs a small bore Teflon flow tube and thermistor sensors in a bridge circuit has been operated over a flow rate range from zero to 5.0 g/hr, has shown a sensitivity of about 10 mv/g/hr and a time constant of the order of a minute.

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Pulsed Plasma Microthruster for Synchronous Meteorological Satellite

WILLIAM J. GUMAN*
Fairchild Republic Company, Farmingdale, N.Y.
AND
THOMAS E. WILLIAMS†
NASA Goddard Space Flight Center, Greenbelt, Md.

Introduction

MAJOR advances in pulsed plasma microthruster system technology have been made since the LES-6 flight launched 1968.[1,2] One of these latter thrusters accumulated about 8900 hr of thruster operation at synchronous altitude during East-West stationkeeping.

The demonstrated success of the pulsed plasma system suggested possible application of a pulsed plasma propulsion system to perform East-West stationkeeping as well as satellite fine pointing which is realized by spin axis precession control of the synchronous meteorological satellite (SMS). The new system required a 25-μ lb-sec impulse bit capability with these impulse

Presented as Paper 73-1066 at the AIAA 10th Electric Propulsion Conference, Lake Tahoe, Nev., October 31–November 2, 1973; submitted November 14, 1973; revision received June 18, 1974. This work was carried out under NASA Goddard Space Flight Center Contract NAS5-11494. The authors wish to acknowledge the invaluable contributions made by a number of individuals during the course of the program. At Republic, J. Chung completed the final mechanical design with many contributions made by M. Katchmar throughout the program. The earlier design effort was carried out by G. Schweitzer. The thruster logic design was performed by R. Gelbman and D. Conroe and W. Gier performed the reliability calculations. L. Brown supervised the laboratory effort. The Power conditioner design at Wilmore Electronics, Inc. was under the direction of E. T. Moore. The efforts of Republic's laboratory personnel also have contributed to successfully fabricating, assembling, and testing the system. At NASA Goddard Space Flight Center, the laboratory tests were performed by T. Carolla, D. McHugh, and J. Mullins.

Index categories: Electric and Advanced Space Propulsion; Spacecraft Attitude Dynamics and Control.

* Chief of Electric Propulsion.
† Aerospace Technologist.

bits to be delivered either on a single pulse basis or over the frequency range of 50 to 110 ppm. The required total impulse capability was 400 lb-sec. The flight prototype propulsion system developed and delivered to NASA Goddard Space Flight Center is therefore capable of providing either singular impulse bits upon command or having its thrust throttleable at constant specific impulse and efficiency over the range 20.8 to 45.8 μ lb-sec. Besides these propulsion performance requirements, the microthruster was required to operate as specified under a sustained acceleration of up to 13 g's for a period of 5 yr with full capabilities at a possible 20 g load condition.

System Details

Results of the extensive design analysis are reported in Ref. 3. Reference 4 presents details of the system design, whereas Ref. 5 presents the results of system tests.

Operational capability of the solid propellant pulsed plasma system under high g environmental conditions was demonstrated under a simulated environment by installing a negator spring into an operational thruster which forced the solid Teflon propellant rod against the fuel retaining shoulder at a constant force of 17 lb. The thruster was operated continuously for 1293 hr (8, 578, 289 pulses) at a thrust level of 41.4 μ lb. The simulated g environment encompassed the range 18.8–33 g with no adverse effect on thruster operation or performance. Besides this simulated high g test, another endurance test of 2080 hr (13, 788, 327 pulses) was carried out at an average impulse bit level of 27.62 μ lb-sec. A total impulse capability of 380.8 lb-sec was verified.

A unique power conditioner subsystem was specifically developed to realize the variable thrust throttling requirements without adversely stressing the thruster capacitor. Since the satellite spin rate can vary from 50–110 rpm, it is necessary to charge the capacitor in 0.5454 sec which is compatible with the period of the higher spin rate. If, on the other hand, the satellite should spin at the lower rate, one finds that under worse-case conditions an accumulated time of as much as 3120 hr at peak voltage could be impressed upon the capacitor. The power conditioner‡ charges the two 4-μf thruster capacitors to 1450 v$\pm$1% in 500$\pm$40 msec independent of the satellite fire command signal pulse rate such that at the spin rate of 110 rpm capacitor recharging occurs immediately after each thruster firing. At lower pulse rates initiation of the fixed duration capacitor recharge cycle is delayed with the delay time at zero capacitor voltage being a function of the satellite spin rate. This charging technique was successfully implemented and has reduced the d.c. voltage stress impressed upon the capacitor to only 179 hr over the entire mission life.

Based upon the experience gained during the LES-6 flight system program, it was decided not to use high voltage connectors in interfacing the power conditioner to the thruster. Instead, the output leads of the power conditioner were hard-wired directly to the thruster terminals thereby eliminating bulkhead standoffs or connectors.

System operation requires only three inputs: 1) a 29.4$\pm$0.2 VDC bus power input, 2) a 28$\pm$2 VDC enable signal, and 3) a 50-msec$\pm$5-msec long 5-v amplitude command fire signal. This latter signal can be provided at any desired pulse rate in the range 50–110 ppm to produce either a sequence of thruster impulse bits or an equivalent steady-state thrust level. Both the 29.4-v primary power and the 50-msec long command fire signal imputs can be impressed continuously upon the propulsion system without drawing power. The system will be "off" until the 28-v enable signal is applied. Power is used only after the enable signal is applied. Depending upon the rate of the satellite supplied command fire signal, the first impulse bit will be provided in the range of 0.55–1.2 sec after enabling the system. Removal of the enable signal instantly shuts down the system without further power being consumed by the system. Figure 1 shows the complete propulsion system ready for installation.

‡ Developed by Wilmore Electronics, Inc. of Durham, N.C., to Fairchild Republic specifications.

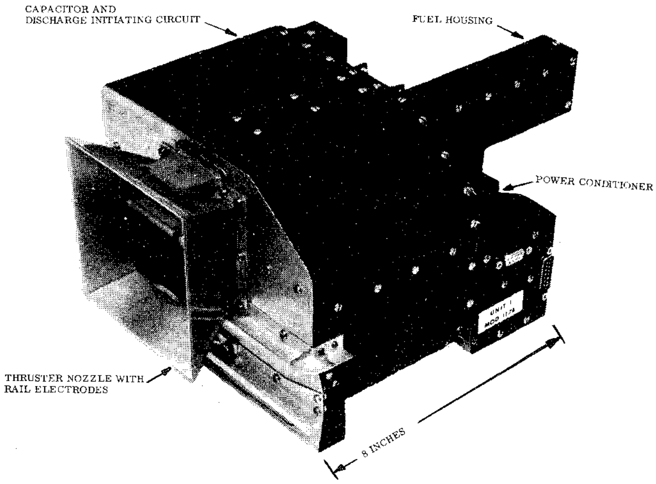

Fig. 1 Integrated SMS propulsion system.

A reliability analysis of the system was performed. This analysis consisted of piece-part component stress analysis, failure rate predictions, and total system reliability computations. The stress ratios of each piece-part were evaluated under worse case conditions of system operation at $T = 60°C$ and a pulse rate of 110 ppm. A 60°C environment was used to include a possible 10°C system temperature rise above the 50°C maximum spacecraft temperature under maximum thruster firing conditions. MIL-HDBK-217A was used for the source of failure rate deviation for semiconductors and for other devices in which failure rate deviations were not covered in MIL-STD-198 and MIL-STD-199. In the final system reliability calculations, a series reliability model was assumed; i.e., the failure of any part is defined as a systems failure. The final system reliability was calculated from the relation

$$R = \exp\left(-\sum_{i=1}^{n} \lambda_i t\right)$$

where λ_i is the application failure rate of the ith part evaluated and "t" the total system operating time during the 5-yr mission. This latter was taken as 3030 hr. The predicted reliability of the system under 100% worse case conditions for the 5-yr mission was found to be 0.92. Less the main energy storage capacitor, this value became 0.98. The analysis showed the system reliability to be strongly dependent upon the main energy storage capacitor. It was also found that the numerical failure rate in many cases penalized the system because the applied operating stresses were often considerably below minimum values listed in the failure rate deviations source. Extrapolation of the failure rate curves was not allowed and the lowest tabulated stresses were used in the calculations.

Test Results

Predelivery tests of two complete systems performed at Fairchild Republic showed that excellent reproducibility of the average impulse bit amplitude can be obtained from systems built to the same drawings and specifications. Extensive testing was subsequently carried out at NASA Goddard Space Flight Center. One unit, designated the qualification thruster, was used for vibration tests, thermal-vacuum tests, and life tests.

An anomalous noise spike was observed on each of the four telemetry output lines during each main thruster discharge. While the spike varied somewhat from one firing to the next, it was typically one to two volts in amplitude and approximately 5 msec in duration which is considerably longer than a thruster discharge. Various tests carried out to locate the source of this noise have been unsuccessful and the problem has not been corrected. It should be noted, however, that this condition has caused no problems in obtaining the desired thruster perform-

Table 1 **Endurance test results**

Test average impulse bit	27.5 μ lb-sec
Total propellant used	9.74 in. or 0.888 lb
Test average I_{sp}	405 sec
Total impulse delivered in this test	359 lb-sec
Anticipated total impulse capability	387 lb-sec

ance data. Furthermore, operation of the thrusters in conjunction with a spacecraft type telemetry system has been carried out without any consequence. For this reason, the elimination of this noise has had a low priority.

After initial thrust measurements, thruster Unit B was subjected to the SMS qualification level vibration and acceleration tests. This consisted of a sinusoidal vibration schedule in each of the three principal axes, a random vibration schedule in each axis, and two steady-state acceleration tests. After each of these individual tests, power was applied to the thruster at atmospheric pressure to verify proper voltage levels and igniter sequencing. No problems of any kind were encountered. During the sinusoidal tests a stroboscope was used to look for any distortions or resonance that might occur. None could be seen. After the series of vibration and acceleration tests were completed, thruster Unit B was retested for thrust and electrical performance. The values obtained during these tests agreed with those taken prior to vibration.

Performance of the thruster through the scheduled $-40°C$ and $-20°C$ thermal-vacuum tests was without incidence. Instrumentation revealed that the thruster electrical performance was excellent throughout the entire range of duty cyles, input levels, and firing rates. However, on the last operational check prior to increasing the temperature to $+50°C$ a failure occurred. Upon disassembly of the thruster it was found that one of the two energy storage capacitors was shorted. As a result of the low temperature environment, radial cracks developed in the hard epoxy end seals on the capacitor. This permitted air, entrapped between the layers of foil and mylar during manufacture, to slowly be pumped out. Thruster operation continued normally with no indication of a fault until the pressure was reduced to a critical value and a Paschen breakdown occurred within the capacitor.

The immediate problem was solved by replacing the hard epoxy end-seals with a flexible polyurethane potting compound that was compatible with the capacitor material and temperatures. After replacing the capacitors the thermal-vacuum test was begun anew. This time the thruster performed satisfactorily under all combinations of temperatures and input signal variations without anomalous deviations. Perhaps the most important observations made are that the fully charged voltage (1450 v) of the 2-μf energy storage capacitors remaining essentially constant ($\pm 0.2\%$) under all test conditions; and the capacitor temperature did not exceed 150°F.

During additional tests, the thruster was operated for 2×10^6 pulses at 100 ppm and a 90% duty cycle to simulate its spacecraft usage. This was accomplished by energizing the thruster "Enable" input at the desired duty cycle while leaving the "Input Power" and the "Fire Command" inputs on continuously. The number of on/off cycles exceeded what would be required for daily use in a 5-yr mission.

After thrust measurements, the test was continued. All electrical performance data remained very stable and indicated normal operation. After 2.7 million pulses, however, a failure occurred within the thruster. Instead of completely shutting down during the "Off" period the high-voltage converter portion of the thruster remained on and kept the capacitors charged. When fully on, however, thruster performance was not affected. Because of this and the fact that the required number of on/off cycles had already been exceeded, the test was continued. "Normal" thruster system operation could be realized under these conditions merely by applying or removing the 29.4-v input power in lieu of the enable signal. Thruster operation was resumed and continued to 13.1 million pulses at which time the test was terminated to ensure that sufficient propellant would be available for additional thrust measurements. Throughout the test, all telemetry and voltage monitors remained normal and very stable.

Thrust measurements carried out after the endurance test yielded an impulse bit of 27.7 μ lb-sec. Previous work indicates that for a given configuration the product of $I_{bit} \times I_{sp}$ is constant. This factor in conjunction with the propellant consumption rate, which was measured throughout the endurance test, was used to calculate the test average impulse bit, as shown in Table 1. This value was then used to calculate the test average I_{sp} and total impulse. The anticipated total impulse capability shown in Table 1 was based on the propellant used before the endurance test and the available propellant at completion.

The results of Table 1 are in excellent agreement with earlier results obtained at Fairchild Republic.

Conclusions and Recommendations

1) The thrusters structural design is sound and is capable of satisfying the SMS qualification vibration and acceleration levels.

2) No thermal problems were encountered with the basic thruster design. The capacitor failure is considered a component problem in which more developmental work is recommended.

3) The basic thrust and electrical performance was very good but additional effort is needed to locate the source of and to eliminate the noise spikes on the telemetry lines.

4) A failure analysis of the "Enable" circuit has not yet been carried out so no recommendations can be made as to how it might be fixed.

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Optimal Scheduling of a Satellite Control Network Using Zero-One Linear Programing

L. C. WESTPHAL*
University of California, Los Angeles, Calif.

I. Introduction

AT any given time this country has a large number of satellites in Earth orbit. These are sponsored by a number of agencies for purposes ranging from weather observation to defense and hence their characteristics and orbits vary widely.

Received February 11, 1974; revision received April 22, 1974. Portions of the work reported here were performed while the author was employed by the Aerospace Corporation, El Segundo, Calif.

Index categories: Aerospace Management; Earth Satellite Systems, Unmanned.

* Assistant Professor, School of Engineering and Applied Sciences.