Engineering Notes

ENGINEERING NOTES are short manuscripts describing new developments or important results of a preliminary nature. These Notes cannot exceed 6 manuscript pages and 3 figures; a page of text may be substituted for a figure and vice versa. After informal review by the editors, they may be published within a few months of the date of receipt. Style requirements are the same as for regular contributions (see inside back cover).

Space Flight Test of Electric Thruster System MDT-2A

Shi-Ming An,* Han-Ji Wu,† Xue-Zhang Feng‡
and Wen-Xi Liu‡
SSTC Academia Sinica, Beijing, China

Introduction

DEVELOPMENT of a pulsed plasma thruster (PPT) system suitable for use in the Chinese space program began in about 1970. Preliminary work leading to the establishment of the necessary test facilities, basic research on the PPT concept, and development of a flight-qualified PPT system were accomplished during the first ten years of the project.[1] Specific work conducted during this period included the development of the required vacuum test facilities; basic studies on the thruster, energy storage, instrumentation, and power circuitry subsystems; and development of the prototype system and acceptance testing of this system. The thruster that resulted from the program utilizes components that are available in China and has capabilities that are consistent with the objectives of the Chinese space program. A test of the PPT system, to be conducted on a rocket vehicle launched on a ballistic trajectory, was proposed in 1976 and carried out in 1981. The flight test was designed to demonstrate that: 1) the thruster could tolerate the launch environment, 2) its electrical performance was the same in space as in the laboratory and 3) commands passed to the rocket and telemetry signals from it were not affected adversely by PPT operation.

Two pulsed plasma thruster modules were tested on a ballistic rocket mission on December 7, 1981. During the 37-minute space test in which the rocket reached an altitude of 3400 km, telemetry signals revealed that the thruster systems survived the launch and then performed in space as they did during ground-based tests. No evidence of interference from the thruster systems on spacecraft control and communication systems was observed.

System Description

The basic principles of operation of the MDT-2A pulsed plasma thruster, its performance characteristics as measured in the laboratory, and justification of the design have been presented previously.[1] The key elements of the flight system are summarized in Fig. 1. The power conditioner in this system, which is operated from a general service battery, charges the energy storage capacitor continuously. This capacitor is connected in parallel across two sets of PPT electrodes. Either PPT can be fired by triggering the appropriate ignitor to discharge the capacitor. In practice the controller triggers the ignitors in sequence, so the two PPTs' nozzles connected to one capacitor are fired in an alternating sequence. The total system placed on the spacecraft, therefore, contained two power conditioners, two energy storage capacitors, and two thrusters (four sets of electrodes). The power conditioners were operated at a constant power level and had an efficiency greater than 80%.

The capacitors have a capacitance of 2 μF and are charged to a voltage of 2 kV. The pulsed plasma thrusters utilize teflon propellant, which is fed into the thruster using a constant force spring. The area of the propellant exposed to the discharge is 10 mm by 25 mm and the length of the electrodes along which the teflon plasma is accelerated is 15 mm. During operation, a discharge current having a peak value of 10 to 12 kiloamperes is produced and the discharge duration is 5 to 10 μs. Each capacitor is discharged at a rate of 1 pulse per second; therefore, each discharge nozzle is operated at a rate of one pulse every two seconds. This result is a total mean power consumption of 5 watts for each thruster.

The ignition circuit, shown in Fig. 1, has a silicon controlled rectifier which works as a switching element in the circuit and discharges the 10 μF capacitor charged to 150 V under control of a command impulse from the controller. At this voltage, commercially available silicon controlled rectifiers can be used and acceptable lifetimes are realized.

Figure 1 suggests two measurements that are used to characterize the performance of each pair of pulsed plasma thrusters and the associated capacitor charging circuit. Thruster operation is sensed by a Rogowski coil that detects the current flowing through either PPT electrode pair connected to a given capacitor. Charging circuit operation is detected by a coil on the high voltage transformer that senses the output of the power conditioner supplying the energy storage capacitor. These sensors produce small currents that are transmitted to and charge small capacitors in the current and voltage detection circuits. It is the voltage developed across these small capacitors that is sensed, converted into a digital signal, and transmitted using on-board telemetry equipment. Four signals are telemetered to the ground, one for each of the two power conditioners and one for each of the two PPTs.

System Characterization

System characterization tests included performance tests, launch qualification tests, and life tests. Performance tests conducted far in advance of the actual space launch were executed in a vacuum chamber equipped with a thrust balance. Measurements of the thrust, power consumption, and propellant mass loss over a certain number of pulses were used to determine the performance condition identified in Table 1. Launch qualification tests included mechanical shock

Table 1 Characteristics of MDT-2A pulsed plasma thruster system

Average impulse bit	63 μN-s
Average specific impulse	280 s
Thruster efficiency	2%
Total power consumption[a]	5 W
Total weight[a]	2.75 kg

[a] For one thruster system composed of one power conditioner, one controller, one capacitor, two ignition circuits, two telemetry output devices and two sets of electrodes.

Presented as Paper 82-1874 at the AIAA/JSASS/DGLR 16th International Electric Propulsion Conference, New Orleans, La., Nov. 17-19, 1982; received Dec. 10, 1982; revision received Nov. 29, 1983.

*Associate Professor, Electric Propulsion Laboratory.
†Assistant Scientist, Electric Propulsion Laboratory.
‡Engineer, Electric Propulsion Laboratory.

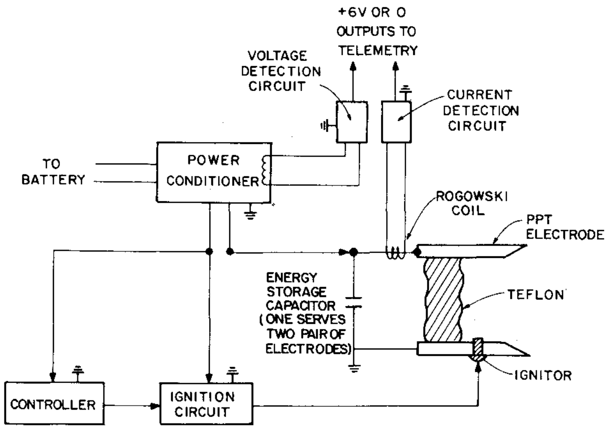

Fig. 1 MDT-2A schematic diagram.

tests, vibration tests, thermal vacuum tests and exposure to the humidity conditions that might be expected during the launch and in the pre-launch environment. The necessary tests were conducted and the PPT system was determined to be suitable to meet the launch and pre-launch environments.

Life tests were conducted on four different thrusters, each test being on the order of 1,000 h long, and conducted over the period 1978 through 1981. In each case, the thruster operated without failure until it was terminated by the operators. Since the launch, additional testing of one thruster has extended to over 3,500 h.

Preflight Testing

In order to insure that the thruster systems to be operated in space were performing properly, two preliminary tests were conducted on the ground. The first was conducted in the vacuum chamber wherein power and thrust measurements could be made at the same time telemetry signals were being examined. This facilitated the establishment of benchmark telemetry signals from the high-voltage transformer and Rogowski coil sensors that had been digitized and transmitted through telemetry links. These signals could be used to indicate normal operation of the thruster system. After these tests had been concluded, the thrusters, along with the power system, were mounted on a rocket and compatibility of the thrusters with the other systems on the rocket was investigated. In this test, the thrusters were maintained at atmospheric pressure and were discharged at the normal repetition rate. Telemetry links were operated to activate the attitude control system on the rocket. In addition to the on-board computer, the communication system and the measurement system were operated. With the pulsed plasma thruster operating in this atmospheric pressure environment, each of these systems was observed to operate properly. Telemetered signals from the thruster system were the same as those observed during system characterization tests. These tests suggested that the electromagnetic interference from the pulsed plasma thruster system should not affect operation of the vehicle adversely.

Flight Test Results

On December 7, 1981, a rocket vehicle with two pulsed plasma thrusters on board was launched on a ballistic trajectory. It achieved an apogee altitude of 3,400 km. The thruster system was turned on at an altitude of 600 km and remained on for 37 minutes while the vehicle rose to apogee and descended to an altitude of 400 km. During this period of operation, the telemetry signals received from the PPT system corresponded precisely to those observed during the two sets of ground-based tests. This indicated that the signals from the Rogowski and transformer coils were also the same for each of these tests. The fact that the telemetry signals were the same in all cases suggests that the pulsed plasma thruster operating conditions were the same for each of the three tests conducted, i.e., the test in the vacuum facility, the test on the rocket at atmospheric pressure, and the space test. Throughout the space test the vehicle responded normally to commands from the ground, thus suggesting that electromagnetic interference from the pulsed plasma thruster did not interfere with any of the normal communication telemetry links or the on-board computer system.

Conclusion

Ten years of development effort resulted in the successful launch of two pulsed plasma thrusters. These thrusters performed in the same way in space as they did in ground-based tests. Electromagnetic radiation from the thrusters did not interfere with communications with the vehicle, the on-board computer system, any of the chemical rocket ignition systems, or the attitude control systems. The pulsed plasma thruster technology resulting from this work will be considered ready for application in space when a life test of the same duration as that intended for the space test has been completed.

Reference

[1]S.M. An and H.J. Wu, "MDT-2A Pulsed Plasma Thruster," AIAA Paper 81-0743, 1981.

Aspects of Nonseparating Apogee Motors

K.K. Biswas*
Vikram Sarabhai Space Center, Trivandrum, India

Introduction

LOW-Earth orbit of a satellite is often difficult to achieve beause of the weight penalty of stage motors and possible increase of payload weight from the design value. In addition to this, the payload separation from the apogee stage may create a collision problem.

A study related to nonseparation of the apogee stage from the payload has been made with specific reference to the first Indian launch vehicle, SLV. It was found that the orbital lifetime of a low-Earth satellite of polyhedron shape will be increased by a factor of 2.3 if the apogee motor is not separated from the payload. In addition, if there is no separation of the payload from the apogee stage, then the separation system becomes unnecessary and a payload saving equal to the weight of the separation system can be achieved. This will further enhance the orbital performance of the satellite. Note that the first Russian satellite, Sputnik 1 was attached to its apogee stage for the first 60 days (Sputnik 1 had a 93-day lifetime).[1]

Orbital Lifetime Analysis

Estimation of satellite lifetime is necessary for planning scientific experiments on a satellite. A number of factors affect the lifetime of a satellite. Among them are the drag coefficient, atmospheric density and its variation (diurnal and solar cycle effects), and satellite mass. The lifetime of a

Received May 16, 1983; revision received Jan. 20, 1984.

*Manager, Software Systems, Aerodynamics Division.