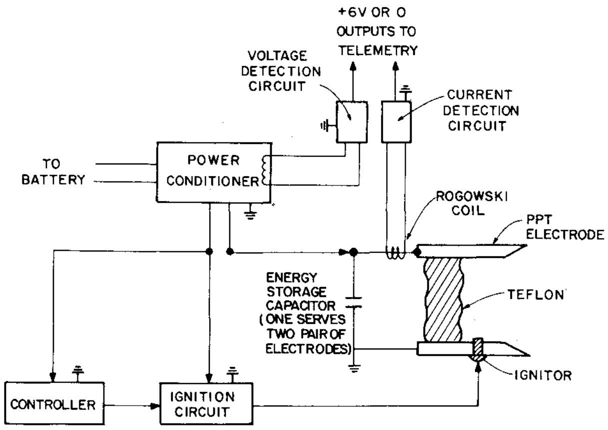

Fig. 1 MDT-2A schematic diagram.

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 onboard 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.

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Aspects of Nonseparating Apogee Motors

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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

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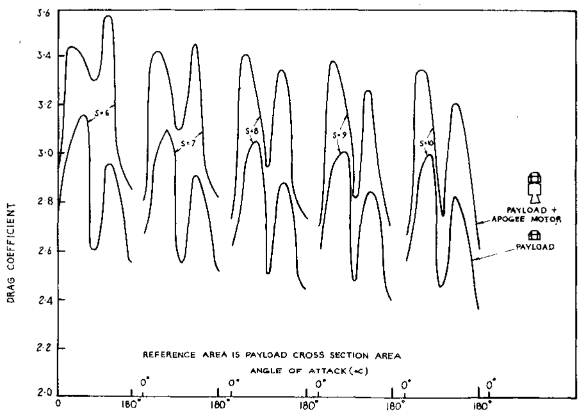

Fig. 1 Drag coefficient for payload and apogee motor and payload alone.

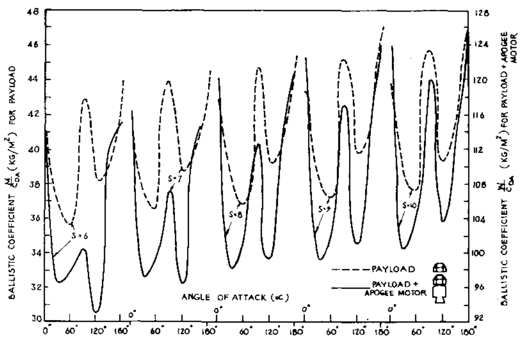

Fig. 2 Ballistic coefficient for payload and apogee motor and payload alone.

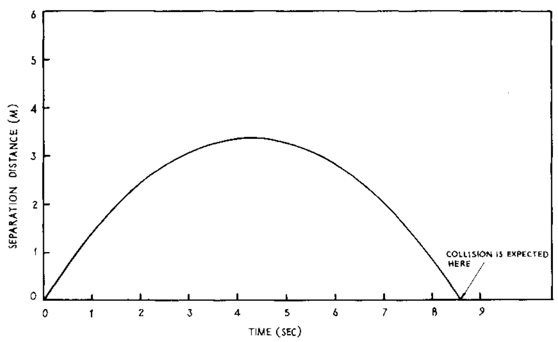

Fig. 3 Time history of separation distance of payload from apogee motor.

satellite has been determined by using Sterne's small perturbation analysis.[2,3]

At altitudes of satellite motion, the intermolecular collisions can be neglected and free molecular flow theory can be used to derive the drag coefficient for geometrically shaped satellites. For a satellite which is made up of plates, the drag coefficient can be computed, taking into account the plate shape combination. The angle of attack (α) is the angle between the satellite velocity vector and satellite body axis.

The drag coefficient for a typical polyhedron shaped satellite when the apogee motor has not been separated from the satellite is shown in Fig. 1. It can be seen that the drag coefficient for the combined apogee motor and satellite has been increased by 12% to 38% compared to that of the satellite alone (using the same drag reference area). This is due to the fact that the contribution from the cylindrical apogee motor is much less than that of the polyhedron shaped satellite.

Table 1 Satellite lifetime prediction

	Satellite	Satellite with apogee motor
Weight, kg	35	110
Lifetime, days	246	578

	Apogee	Perigee
Orbit altitudes, km	672.9	289.5

Table 2 Payload and apogee motor mass, c.g., and moment of inertia

	Apogee motor	Payload
Mass, kg	75	35
c.g. location, measured from common c.g., m		
X	−0.2470	0.5060
Y	−0.0025	0.0751
Z	−0.0025	0.0051
Moment of inertia, kg/m^2		
Ixx	3.6890	1.4522
Iyy	27.3570	1.3222
Izz	27.3570	1.3523
Ixy	0.00	−0.0294
Iyz	0.00	−0.0022
Izx	0.00	−0.0022

The ballistic coefficients for the same configurations, i.e., the configurations for which the drag coefficients have been shown in Fig. 1, are shown in Fig. 2. One can see in Fig. 2 that the ballistic coefficient for the apogee motor and satellite together is 2.28 to 2.81 times higher than that of the satellite alone. This is due to the fact that the weight increases for the satellite and apogee stages together by 3.14 times, while the drag coefficient increases by only 12 to 38%. Since the orbital decay rate is inversely proportional to the ballistic parameter, the orbital decay of the combined satellite and apogee motor will be about 60% less than that of the satellite alone.

The lifetime of a polyhedron shaped satellite alone and in combination with the apogee motor (using the 1966 U.S. standard atmosphere and the drag coefficient shown in Fig. 1) are presented in Table 1. It can be seen from this table that the lifetime increases by 2.36 times if the apogee motor is not separated from the payload.

Separation of Payload From Apogee Motor

In the case of satellite separation, the apogee motor may collide with the satellite after separation because tail-off thrust has been acting on the apogee stage for a long time in the near-vacuum conditions existing at the satellite separation altitude. The analysis of a satellite separation has been made by simulating the six-dimensional trajectory of motion of the separating bodies, taking into account the forces and moments due to gravity, thrust, and separation.[5]

The mass and inertia properties of the payload and apogee motor are given in Table 2. The tail-off thrust is assumed to be $T = 3e^{-0.001t^*}$ where t^* is the time in seconds measured after apogee motor burnout. The relative distance between the payload and apogee motor after separation is shown in Fig. 3. It can be seen that collision between the payload and apogee motor will occur at about 8.5 s after separation.

For this particular case, the separation must be delayed sufficiently so that the tail-off thrust dies down significantly. The delay of separation will often create unusual operational problems for the ground station. On the other hand, if the apogee motor is not separated from the payload, this problem will not exist, and payload weight gain equal to the separation system weight can be achieved.

Conclusion

There are distinct advantages of nonseparation of a payload from the apogee motor. However, certain unfavorable implications of this option should also be taken into account. One major problem that has been pointed out is that the apogee motor case may lose its rigidity, and this may lead to coning angle and finally a flat spin. Moreover, this may lead to satellite antenna dip and to thermal imbalance.

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A Shuttle Derived Utility Vehicle for Delivery of Small Payloads to Orbit

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Introduction

A DERIVATIVE of the Space Shuttle that results in a vehicle having less capability, but at a reduced cost per flight, is proposed. In this design, the two Solid Rocket Boosters (SRB's) are removed (Fig. 1). The two Shuttle SRB's weigh a total of approximately 2.6 million lb and constitute 57% of the launch weight. After removal of the SRB's, three Space Shuttle Main Engines are added at the rear of the External Tank (ET), and additional main engine propellant tankage is added in the cargo bay. This modified Shuttle is referred to herein as a utility vehicle. It should be noted that the three tank-mounted engines are modified for an expansion ratio of 40 to 1 compared to 77.5 to 1 for the present Shuttle. Only one tank-mounted engine is visible in Fig. 1, since the engines are in line in the side view.

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Mission Description

All six LOX/LH_2 engines are operated at liftoff, giving a thrust-to-weight ratio of 1.3 (Fig. 2). This value varies slightly, depending on the payload and amount of propellant stored in the payload bay. Throttling of the three tank-mounted engines is initiated 93 s after liftoff. Throttling and sequential shutdown of the engines continues until 245 s after liftoff, when all tank-mounted engines are shut down. At this time, throttling of the engines on the Orbiter is initiated while the electrical umbilicals and propellant lines on the tank-mounted engines are separated (a 30 s interval is available for this procedure). At 275 s into the flight, the ET is separated from the Orbiter and is allowed to reenter (Fig. 1). The Orbiter continues on the internal propellants stored in the payload bay. Throttling and sequential shutdown of the engines on the Orbiter proceeds until orbital insertion. After achieving orbit and delivering the payload, the hydrogen tank in the payload bay is discarded to reenter from low-Earth orbit.

The three ET-mounted engines are then retrieved with an extended manipulator arm which is stored in the payload bay. The ET thrust structure is then either collapsed and returned for re-use or expended. The Shuttle returns with the three engines and the cargo bay LOX tank.

In the event of an abort during ascent, the stored propellants in the Orbiter can be burned to depletion with the

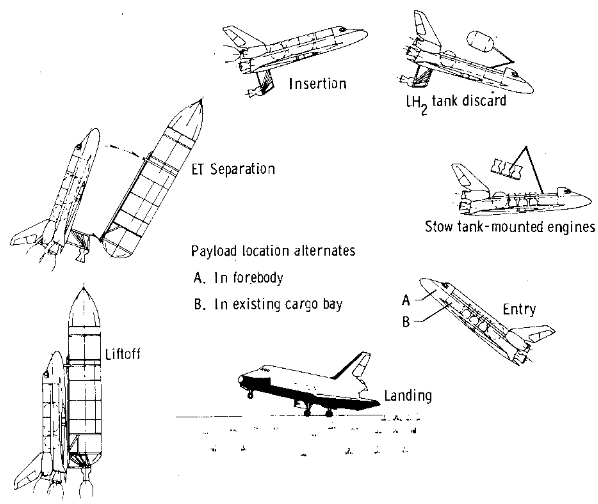

Fig. 2 Thrust, weight, and thrust-to-weight ratio vs time.

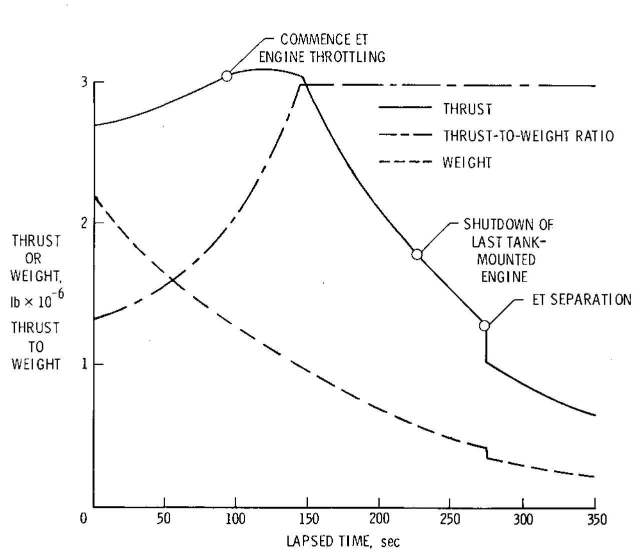

Fig. 1 Utility vehicle configuration and mission sequence.