LEASAT Liquid Apogee Motor Subsystem Performance

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This paper describes the performance of the bipropellant propulsion subsystem for the LEASAT spacecraft. Two LEASAT spacecraft have been successfully Shuttle-launched and are currently providing satellite communications services to the United States Navy. Spacecraft propulsion incorporates three separate subsystems: A solid perigee motor, a bipropellant subsystem, and a monopropellant reaction-control subsystem. The solid perigee motor provides impulse to raise orbital apogee from the Shuttle parking orbit. The bipropellant subsystem, which utilizes monomethylhydrazine and nitrogen tetroxide, provides the impulse required for apogee augmentation and injection into synchronous orbit at apogee. The monopropellant reaction control subsystem supplies the impulse required for spacecraft spinup, apogee augmentation, attitude and spin speed control, and other required on-orbit maneuvering impulse throughout the operational life of the spacecraft. The LEASAT spacecraft were launched in August and November of 1984. The flight data obtained support the results of analytical modeling and subsystem level test data as well as performance data supplied by vendors.

Nomenclature

A_o	= orifice area
A_s	= exposed propellant surface area
C	= flow coefficient
D_o	= orifice diameter
g_e	= effective acceleration due to spacecraft spin
h_i	= distance between fluid level and equilibrium level
H	= total fluid head
K1-K4	= constants derived from thruster data
P_c	= chamber pressure
P_{in}	= thruster inlet pressure
Q	= volumetric flow rate
Re	= Reynolds number
$\dot{w}$	= propellant flow rate
μ	= viscosity
ρ	= density
τ	= time constant
Φ	= sun angle

Subscripts

f	= fuel
ox	= oxidizer
t	= total

Introduction

THE LEASAT series of spacecraft (Fig. 1) represents the first synchronous altitude satellites designed specifically for optimal use of the space transportation system (STS). The spacecraft length and weight have been designed to achieve the optimum payload cost, thus minimizing overall mission cost.[1]

The LEASAT spacecraft is spin-stabilized and utilizes a despun communications platform. The large-diameter design allows for all orbital injection components to be self-contained, internal elements of the spacecraft (Fig. 2). This shorter, larger diameter design is passively stable, allowing for a relatively simple attitude-control system.

The unique arrangement of the spacecraft in the payload bay (Fig. 3) required the design of an innovative method of deployment. This design uses a "frisbee" technique in which a cradle-ejection mechanism imparts both a linear separation velocity of about 2 ft/s and a spin speed of 2-3 rpm to provide spin stabilization. The deployment sequence has been performed successfully for two spacecraft (F-1 and F-2) placed into orbit during Shuttle missions 51-A and 41-F in 1984.

The spacecraft deployment occurs at the Shuttle parking orbit of 160 n.mi. (296 km). Since a geostationary satellite orbits at 19,300 n.mi. (35,744 km), a relatively large impulse must be imparted to achieve this altitude. This impulse is supplied by two separate subsystems: A solid perigee motor (PKM) and a bipropellant liquid apogee motor (LAM). A third monopropellant (hydrazine) reaction-control system (RCS) provides on-station attitude and spin-speed control, since the LAM has completed its function when the spacecraft reaches synchronous orbit.

The Minuteman III solid motor was chosen to transfer the spacecraft from the circular low Earth orbit to an elliptical subtransfer orbit with an apogee altitude near 8200 n.mi. (15,200 km). The LAM is then activated to raise the apogee to geosynchronous altitude. A propellant budget is shown in Table 1. The apogee augmentation is obtained through a series of three burns of 650-700 s duration centered about perigee. At the end of these perigee firings, the spacecraft is in a highly elliptic orbit with an apogee at 19,300 n.mi. (35,744 km) and a perigee at the Shuttle parking altitude of 160 n.mi. (296 km) (Fig. 4).

With the perigee firings completed, the spacecraft is ready for insertion into geosynchronous orbit. The orbit is circularized by two LAM firings occurring at apogee with a combined duration of approximately one hour. A short apogee firing followed by reorientation maneuvers places the spacecraft in the desired geosynchronous orbit. Once the satellite is on station, the RCS system provides the stationkeeping impulse during the nominal seven-year life. However, if adequate propellant residuals remain in the LAM system, it can continue to perform the propellant intensive north-south maneuvers.

Liquid Apogee Motor Design

The LAM, shown schematically in Fig. 5, is a bipropellant system which utilizes the hypergolic combination of monomethylhydrazine (MMH) and nitrogen tetroxide (N_2O_4). The system incorporates two separate half-systems sharing a common high-pressure helium-pressurization system.

The centrifugal force associated with a spinning spacecraft acts to settle the propellants at the outlet of each tank. This eliminates the need for mechanical expulsion devices to assure bubble-free propellant flow.

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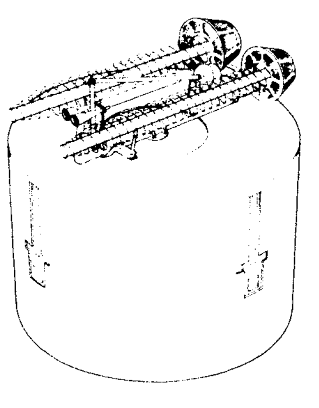
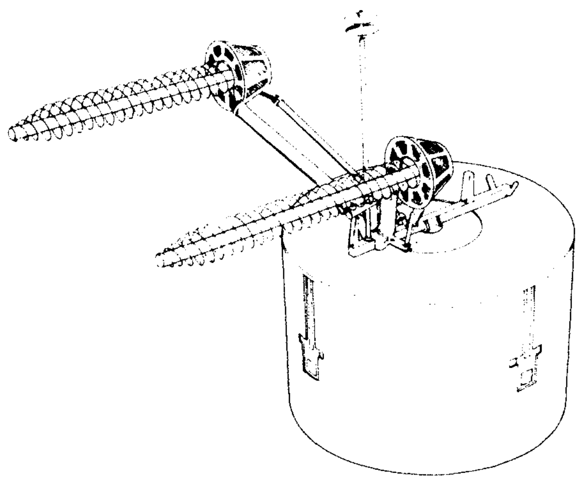


LAUNCH CONFIGURATION



ON-ORBIT CONFIGURATION

Fig. 1 LEASAT spacecraft.

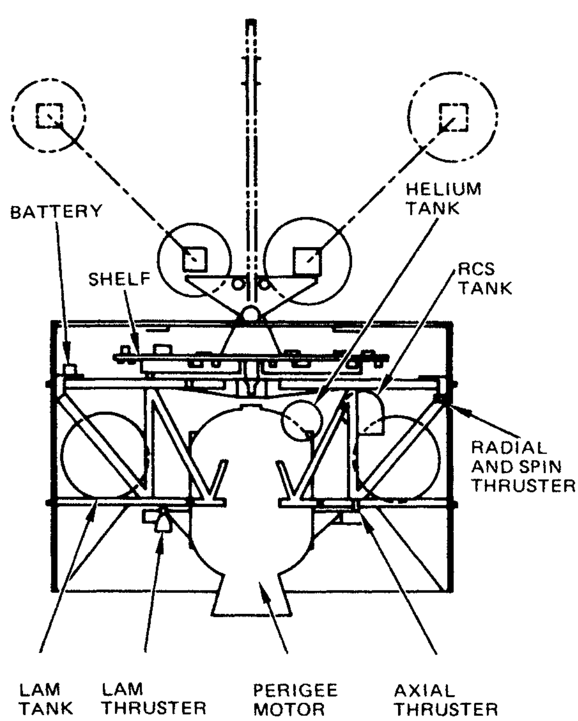


Fig. 2 LEASAT internal configuration.

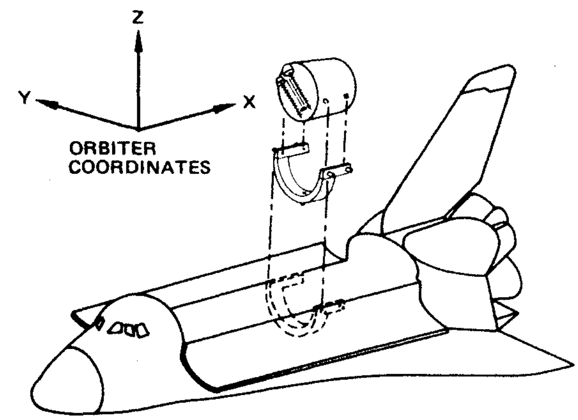


Fig. 3 Spacecraft installation in STS.

Table 1 LAM propellant budget

Maneuver	Requirement	Propellant mass
Perigee velocity augmentation	2050 ft/s (625 m/s) – 3 burns –	1380 lbm (626 kg)
Apogee velocity augmentation	5750 ft/s (1753 m/s) – 2 burns –	2590 lbm (1175 kg)
Postburns and reorientations		73 lbm (33 kg)
	Propellant residual	37 lbm (17 kg)
	Total loaded propellant	4080 lbm (1851 kg)

The pressurization system includes two spherical helium tanks initially charged to 4200 psi (290 bars). While in the Shuttle payload bay, these tanks are isolated by a normally closed squib valve. Following the solid motor burn, the squib valve is opened, thus activating the pressurization system.

A set of two series-redundant regulators act to maintain a constant propellant tank pressure. However, midway through the second apogee motor firing, the helium tank pressures drop to the level where regulated operation can no longer be sustained. From this point, the system is operating in a blowdown mode.

During regulated operation, the propellant tank pressures are maintained at approximately 212 psia (14.6 bars). This is the combined effect of the regulator characteristics and the check valve and line pressure drops between the regulators and the propellant tanks. The check valves act to prevent propellant migration and account for a nearly 7 psia (0.5 bar) pressure drop. At the end of LAM life, the propellant tanks have blown down to near 185 psia (12.7 bars).

Thrust is provided by two bipropellant rocket engines, each providing 100 lb (445 N) of thrust.[2] Since the propellant combination is hypergolic, no ignition system is required. The thrusters are isolated from the propellant tanks by both dual-seat latching valves and the engine oxidizer and fuel valves. This combination satisfies the STS launch-vehicle safety-design requirements.[3]

The redundant half-system design incorporated in the LAM allows for all LAM maneuvers to be performed by a single thruster in the unlikely event that one thruster should become disabled. The liquid-manifold design incorporates interconnecting latch valves which allow all of the system propellant to be available to the redundant thruster. The interconnecting latch valves can also be used during the mission to equalize fuel or oxidizer imbalances that result from variations in thruster flow rates.

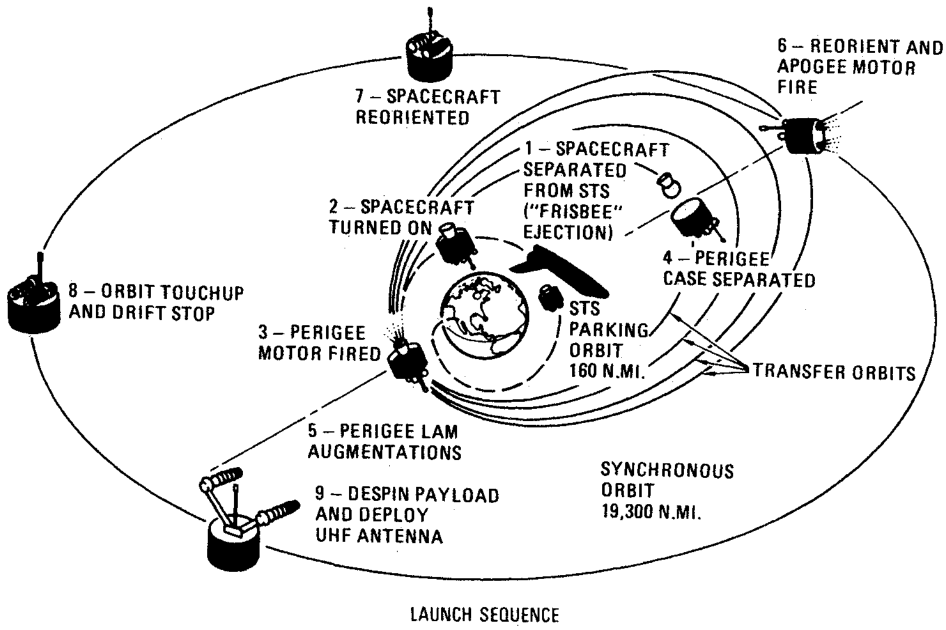


Fig. 4 LEASAT transfer-orbit profile.

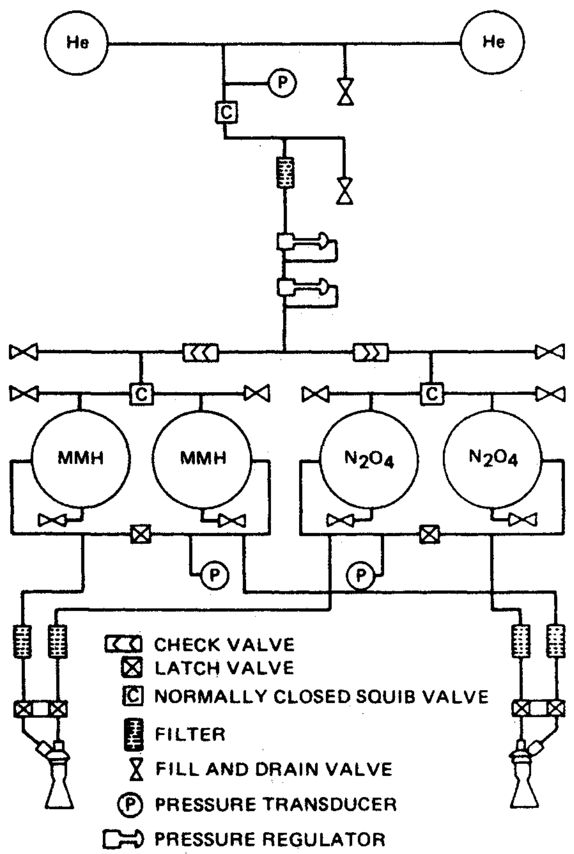


Fig. 5 LAM subsystem schematic.

LAM Component Performance

The F-1 and F-2 LAM systems have performed nominally. Both spacecraft are in the desired orbit with more propellant remaining than required for nominal life.

During the F-1 transfer orbit operations, the oxidizer and fuel propellant tank pressures differed by 6 to 10 psi (0.4-0.7 bar) both during regulated and lockup conditions. These differences were not observed on F-2, which was launched before F-1. Since the pressurization-system design requires the regulated pressure upstream of the check valves to be equal at all times, the difference in measured pressures could be caused by non-nominal operations of the check valves or pressure transducer error or both.

Because preflight check-valve test data are inconsistent with the inflight data, incorrect pressure transducer readings are suspected. The most probable cause is calibration shifts within the pressure transducer. These changes could be caused by oscillations occurring during launch vibration, water-hammer effects, or feedback from the thruster combustion oscillations. The inflight data deviations, while of interest in evaluating component performance, were not large and did not affect spacecraft transfer-orbit operations.

Any differences in tank pressures as well as variations between the performance and flow rates of individual thrusters can result in propellant imbalances. By means of the interconnecting latch valves, it is possible to equalize imbalances between opposing oxidizer tanks and/or fuel tanks. The equalization process can take a relatively long time due to very small differential heads. Because of this, predictions of equalization times were generated to ensure compatibility with other orbital insertion activities.

The analysis of the transient balancing characteristics starts with the assumption that the interconnect valve acts as a flow control line orifice. The flow can be approximated as:

$$Q = CA_0\sqrt{2g_eH} \tag{1}$$

The flow coefficient, C, is a function of Reynolds number. Experiments on the latching valve operating under low pressure heads yielded the following:

$$C = 0.04\ (Re)^{0.37} \qquad Re < 10^3$$

$$C = 0.49 \qquad Re \geq 10^3$$

with Reynolds number defined as follows:

$$Re = \frac{D_0\rho}{\mu}\sqrt{2g_eH} \tag{2}$$

The volumetric flow rate, Q, is then related to the rate of change of the fluid level in one tank with time, dh/dt. By nondimensionalizing the resulting differential equation, the following term can be defined.

$$\tau = \frac{A_s}{CA_o}\sqrt{\frac{h_i}{4g_e}} \tag{3}$$

where τ is the characteristic time constant for the transient flow between tanks.

During the LEASAT F-1 mission, variations in flow rates between the thrusters resulted in an estimated 20-lb (9-kg) mass imbalance, which was equalized by opening the interconnecting valve. Since there is no measure of propellant level, the rate of equalization cannot be directly measured. However, spacecraft sun angle is a measure of orientation and is altered by mass shifts between the propellant tanks. The transient sun-angle change subsequent to the opening of the interconnecting valve was correlated with the response, as predicted by Eq. (3). The observed time-dependence of the sun angle is compared with the calculated time-dependence of the mass imbalance as shown in Fig. 6 and shows excellent correlation.

Propellant Usage and Budgeting

It is imperative for mission success that accurate propellant-budgeting techniques be employed. During the propulsion design phase, the expected component performance characteristics are linked with the spacecraft velocity change requirements to estimate propellant usage. To ensure mission objectives, uncertainties in component characteristics are compensated for by adding an analytically derived amount of propellant. This amount, an unusable residual, is added to the nominal propellant requirement to determine the amount of propellant that will be loaded.

During the mission, the propellant usage is calculated based upon performance measured during component tests. Should parameters such as temperature and pressure deviate from the expected nominal range, mixture-ratio changes may result. These changes can cause an increased residual with a corresponding decrease in spacecraft life, propellant unbalance, and associated wobble. The accurate modeling of these errors allows for compensation by transferring propellant between the tanks through the interconnecting latch valves and by preferential firing of one thruster.

Monitoring inflight propellant usage is not a straightforward process, since a direct propellant-measurement system is not incorporated. To estimate the amount of propellant remaining, it is necessary to approximate the propellant used during each maneuver and subtract this from the known loading. The accuracy of the propellant usage predictions is increased by using actual flight data and analytical modeling, based on acceptance test data for the individual components.

The analytical model used to estimate thrust chamber propellant flowrates involves curve fitting of vendor-supplied test data. This curve fitting is based on the assumption that flow losses in the propellant passages downstream of the thruster inlets are composed of both laminar and turbulent components.

This assumption allows for the best fit of the data and results in the following expression for the difference between the thruster inlet pressure, P_{in}, and the combustion chamber pressure, P_c, as a function of either the oxidizer or fuel flowrate.

$$P_{\text{in}} - P_c = \frac{\text{K1}}{\rho}(\dot{w}) + \frac{\text{K2}}{\rho}(\dot{w})^2 \tag{4}$$

where K1 and K2 are flow coefficients, ρ is the propellant density, and $\dot{w}$ is the propellant flowrate.

Also, as a first order approximation, the thrust-chamber pressure is written as a linear function of the total propellant flowrate, $\dot{w}_t$. A nonzero intercept is used to help account for small nonlinear effects.

$$P_c = \text{K3}(\dot{w}_t) + \text{K4} \tag{5}$$

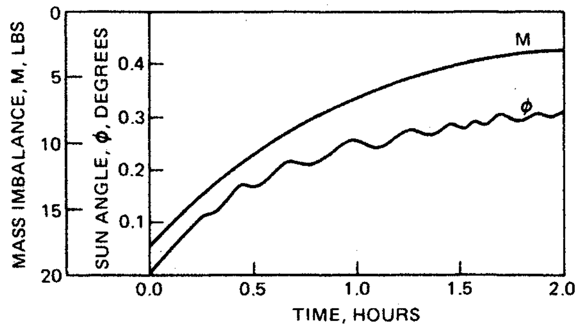


Fig. 6 Mass imbalance and sun-angle transient response.

This technique yields a system of three simultaneous equations, with the oxidizer and fuel flowrates and chamber pressure as unknowns.

$$P_{\text{in,ox}} - P_c = \frac{\text{K1}_{\text{ox}}}{\rho_{\text{ox}}}(\dot{w}_{\text{ox}}) + \frac{\text{K2}_{\text{ox}}}{\rho_{\text{ox}}}(\dot{w}_{\text{ox}})^2 \tag{6}$$

$$P_{\text{in,f}} - P_c = \frac{\text{K1}_{\text{f}}}{\rho_f}(\dot{w}_f) + \frac{\text{K2}_{\text{f}}}{\rho_f}(\dot{w}_f)^2 \tag{7}$$

$$P_c = K3(\dot{w}_{\text{ox}} + \dot{w}_f) + \text{K4} \tag{8}$$

To use this model, the constants K1 through K4 must be calculated using least-squares curve fitting of acceptance test data for each thruster.

A computer model was developed which utilizes the telemetered propellant tank pressure data, the analytical thruster modeling and the pressure loss data for the lines and components (latch valves and filters) between the tank outlets and the thruster inlets. Using this information, the program follows an iterative process to solve for the oxidizer and fuel flowrates as well as the chamber pressure. Once the flowrate history is known during the maneuver, the propellant usage can be determined by means of numerical integration.

The computer program that was developed for this analysis performs the following sequence at each specified time step:

1) The pressures at the inlet to the thrusters are estimated.

2) The thruster performance characteristics dictate flowrates corresponding to the estimated pressures.

3) The flowrates are coupled with the component and piping characteristics to compute the system pressure loss.

4) The pressure loss is added to the estimated thruster inlet pressures to determine the propellant tank pressures.

5) These pressures are then compared with the telemetered tank pressures. An iterative procedure is performed until the calculated tank pressures match the telemetry.

The uncertainty in this method of propellant usage estimation is primarily due to the accuracy of the telemetered pressure data and variations in the thruster performance from the acceptance test data. Temperature-measurement error is also involved but only has second-order effects. As expected, the uncertainty in remaining propellant increases as propellant is used, due to the accuracy of the flow-rate predictions.

When the tanks are loaded, the total propellant mass is approximately 4100 ± 3 lb (1860 ± 1.4 kg). The initial uncertainty is due to the accuracy of the loading equipment. At the end of LAM life, the predictions have an uncertainty dispersion of ± 20 lb (± 9 kg) in each oxidizer tank and ± 13 lb (± 6 kg) in each fuel tank.

This analysis technique was applied to the data obtained on the F-1 and F-2 spacecraft. At the end of LAM life (when the spacecraft reaches its desired geosynchronous orbit), the propellant residuals as listed in Table 2 were estimated. The negative signs indicate that the model predicted propellant depletion prior to the end of the final maneuver. The larger

Table 2 LAM propellant residuals

Spacecraft	Oxidizer tanks, lb(kg) #1	Oxidizer tanks, lb(kg) #2	Fuel tanks, lb(kg) #1	Fuel tanks, lb(kg) #2
F-1	−1.6 (−0.7)	−4.1 (−1.9)	−1.6 (−0.7)	−3.0 (−1.4)
F-2	20.5 (9.3)	16.4 (7.4)	9.6 (4.4)	6.1 (2.8)

residuals realized during the F-2 mission are the result of the more favorable August launch data as opposed to the November F-1 launch.

In general, it is not possible to verify propellant-usage predictions that result in a positive residual, since the remaining propellant is such a small percentage of spacecraft weight. However, during the LEASAT F-1 mission, oxidizer tank number two was taken to depletion. This established a data point at zero remaining propellant for that particular tank. At this point, the propellant usage model was in error by only 4.1 lb (1.9 kg); well within the predicted uncertainty dispersion of ±20 lb (±9 kg). During the LEASAT F-2 mission, it was estimated that the LAM residual would extend the spacecraft life five years beyond the nominal seven-year life.

Conclusion

The liquid apogee motor subsystem has met all design requirements and has had only minor deviations from predicted performance. Hence, the implementation of liquid bipropellant subsystems to perform orbit-transfer functions has been verified as a viable technique from both subsystem design and orbital operation standpoints.

Acknowledgments

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References

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[2]Anon., "R-4D-10, Model Specification for the Marquardt 100 lb Thrust Bipropellant Rocket Engine," Marquardt Co., Van Nuys, CA, Nov. 1977.

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