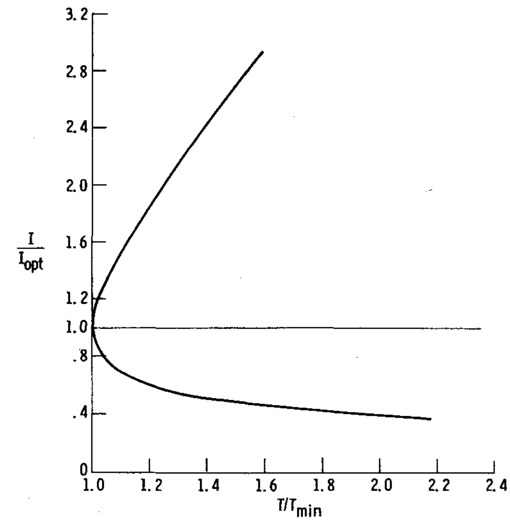

Fig. 1 Propulsion time as function of specific impulse [from Eq. (11)].

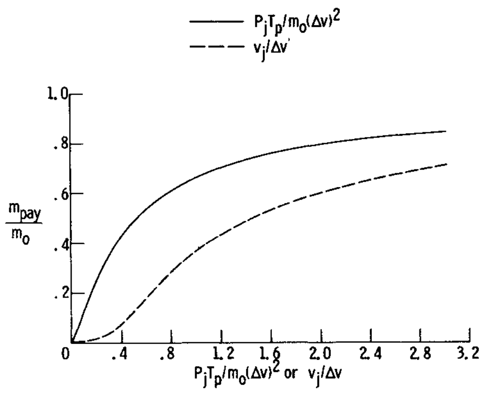

Fig. 2 Relationships for estimating mission capability for laser-driven rockets.

Minimum propulsion time is in the range of a few hours for $\alpha \approx 10^{-4}$ kg/w (a typical value for this mission).

Shown in Fig. 1 is a plot of propulsion time vs specific impulse ($I = g_0 v_j$) [from Eq. (7)]:

$$\frac{T_p}{T_{p,\min}} = \frac{I^2}{I^2_{\text{opt}}} \frac{e^{1.59\, I_{\text{opt}}/I} - 1}{e^{1.59} - 1} \tag{11}$$

This variation of T_p with I agrees qualitatively with numerical results.

The time-minimization shown in Fig. 1 results, as usual, from competing factors: For constant power, as v_j is reduced from high values the thrust increases [Eq. (5)] and thus the acceleration tends to increase and propulsion time goes down. However, the propellant mass, and hence total vehicle mass, increases as v_j is reduced, which tends to reduce acceleration and increase trip time. These competing effects due to reducing v_j produce the observed optima.

Inserting the optimum value of $\Delta v/v_j$ into Eq. (1) yields the result that, for minimum propulsion time on any mission with laser-driven rockets, about 80% of the initial mass is propellant mass and less than 20% is payload. Although this discussion was restricted to minimizing propulsion time, note that the parameter really minimized in Eq. (7) is $T_p/\alpha(\Delta v)^2$. Hence, for a given mission, the $v_{p,\text{opt}}$ from Eq. (9) also maximizes the ratio m_{pay}/P_j for given T_p.

Even more generally, the quantity minimized by Eq. (9) is the exhaust-jet energy ($E_j = P_j T_p$) required to achieve a given payload energy ($E_{\text{pay}} = (½) m_{\text{pay}} (\Delta v)^2$). This accounts for the fact that Dipprey[6] recently obtained the same optimum $v_j/\Delta v$ (Eq. 9) by minimizing the amount of antimatter needed to achieve a given Δv for a given payload mass.

A similar derivation with m_0/P_j as a parameter instead of m_{pay}/P_j [using Eqs. (1, 3, and 4)] leads to

$$P_j T_p / m_0 = ½ v_j^2 \, (1 - e^{-\Delta v/v_j}) \tag{12}$$

which increases monotonically as v_j is decreased. Thus, for fixed initial mass, no minimum trip time occurs.

Using Eqs. (1, 2, and 12) (again with $k \ll 1$), the following relationship is derived between a propulsion time parameter and the payload ratio:

$$\frac{2P_j T_p}{m_0 (\Delta v)^2} = \frac{1 - m_{\text{pay}}/m_0}{\ln^2 (m_0/m_{\text{pay}})} \tag{13}$$

This relationship (shown in Fig. 2, together with the parameter $v_j/\Delta v$) is useful for estimating mission capabilities of laser-driven rockets.

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Reusable Multilayer Insulation Development

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Introduction

DEVELOPMENT of multilayer insulation (MLI) and its design is strongly dependent upon the environment in which the system must function. In recent years, much effort has been expended toward developing MLI materials and design concepts characterized by single use and moderate temperature environment requirements. The advent of reusable vehicles with the Space Shuttle development has imposed much more severe environmental requirements upon MLI systems. The MLI must be purged during ground hold conditions to remove condensible gases and repressurized during re-entry to neutralize the crushing atmospheric pressure loads. Provisions must be made to allow MLI venting during atmospheric ascent and to protect the MLI from repeated exposure to moisture and high temperatures during

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Index category: LV/M Fuel and Propellant Systems (including Storage and Transfer).

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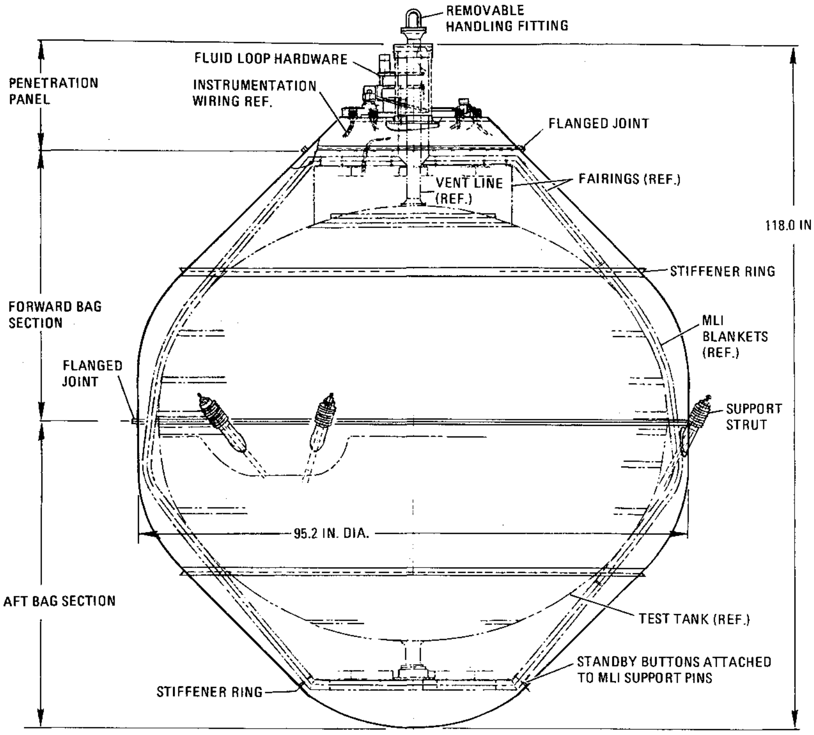

Fig. 1 Purge bag, multilayer insulation, and test tank assembly test configuration.

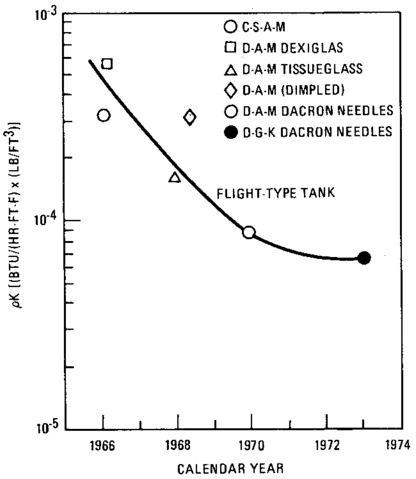

Fig. 2 Multilayer insulation system thermal performance history.

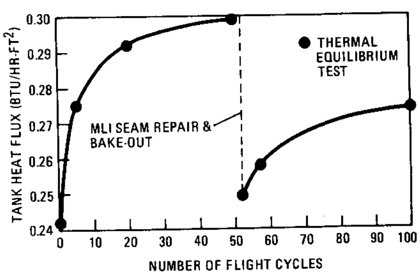

Fig. 3 MLI system thermal performance.

re-entry. The single-use systems previously developed cannot withstand the Space Shuttle environments. It was necessary, therefore, to develop a completely new reusable cryogenic storage system.

Previous studies conducted by Convair[1] indicated that goldized Kapton (a polyimide) would be the ideal reflective shield for a reusable MLI system. The plastic film is compatible with service temperatures to 750°F, and the gold coating is not susceptible to emittance degradation due to moisture exposure.

A new cryogenic propellant storage system was designed, fabricated, and tested to demonstrate the performance and reusability of a flightweight system for Space Shuttle applications. The test system consisted of an insulated hydrogen tank, tank support subsystem, and MLI purge/repressurization hardware.

Reusable Multilayer Insulation System

The MLI assembly is applied over the tank fairing surfaces in gores on the sidewall and flat blanket sections on the top and bottom (Fig. 1). Two blanket layers are used. These blankets are supported from the fairings with pins and interconnected at the seams with rigid "twin pin" fasteners. Individual MLI layers are applied over the vent line and the six tank support struts. The forward flat area of the assembly includes provisions for the vent, purge, and electrical penetrations. A typical gore blanket is a preformed assembly consisting of 22 double-goldized Kapton Superfloc (MLI) shields between reinforced face sheets.

Superfloc, developed by Convair,[1] is a unique, high-performance multilayer insulation system that consists of small tufts of Dacron flock adhesive-bonded in a discrete triangular pattern to one face of a metallized plastic film (radiation shield). When these radiation shields are stacked, the small fibrous tufts prevent adjacent shields from touching and shorting out (thermally) and minimize solid conduction heat transfer through the system.

Purge/Repressurization System

The purge/repressurization system consists of a purge bag surrounding the MLI system and valves and controls for purging, venting, and repressurizing the MLI during ground hold, ascent, and re-entry phases of a vehicle mission. The experimental system operates in three distinct modes: manual purge, manual vent, and automatic pressurization. Helium purge is activated before filling the tank with liquid hydrogen. The purge is continued through the MLI until the condensible gases reach a concentration of less than 1%. Experiments on the full-scale test article have shown that this occurs within 300 seconds with a helium flow of 60 purge volumes per hour.

The automatic mode of operation is used during tank filling and ground hold operations and during re-entry MLI repressurization. Helium is supplied to the purge gas distribution system upon demand. Purge bag pressure is maintained between 0.5 and 1.5 psid by the valve pressure switches, ensuring that crushing pressure loads will not be imposed on the MLI during re-entry.

The vent mode of operation is used during the ascent portion of the mission. The vent valve is locked open and the supply and bleed valves are locked closed, allowing helium purge gas in the MLI to be vented overboard rapidly as ambient pressure decreases with increasing vehicle altitude. Experiments with full-scale test article show that the Superfloc MLI system will vent down to a pressure of less than 10^{-4} torr in less than 180 sec with the ambient pressure at 10^{-5} torr.

Experimental Evaluation

The reusable cryogenic storage system thermal performance evaluation was conducted by evacuating the MLI to a pressure of less than 2.5×10^{-5} torr. A Shulz-Phelps vacuum pressure transducer embedded in the MLI was used to monitor MLI interstitial pressure. The test tank was allowed to equilibrate for three days. Thermal equilibrium was defined as that time when no MLI layer temperature varied by more than ± 1R during a continuous 10-hr period. Tank pressure was maintained within a 0.0056 psi band during thermal equilibrium testing. The propellant boiloff—and, consequently, equilibrium heat leak—was measured with a hot-film anemometer to be 43 Btu/hr, with the ρk at 6.44×10^{-5} Btu-lb/hr-ft^{4}R.

Leonhard and Hyde[2] have surveyed the available data on thermal performance of flight-type MLI cryogenic storage systems and obtained the ρk product for the five flight-type single-use MLI systems that had actually been fabricated, installed on flight-type tanks, and thermal-performance tested as of their writing. They obtained flight-type tank installed performance data for each system (Fig. 2). The performance of MLI has been improving with time of development. The measured thermal performance for the reusable goldized Kapton Superfloc system (D-G-K Dacron needles) is significantly better than that of the best single-use system (D-A-M Dacron needles) previously tested in 1969 and provides a major improvement in thermal performance capability for installed flight-type MLI systems.

The reusability of the new goldized-Kapton Superfloc MLI system has been demonstrated by repeatedly simulating the environments of the Space Shuttle flight cycle. A typical flight cycle consists of a ground purge of MLI with helium gas, a venting evacuation of the purge gas in the MLI to space vacuum conditions, and a re-entry repressurization and heating simulation following the Shuttle re-entry flight profile. Re-entry heating raises the system surface temperature during each cycle to 810R. Liquid hydrogen is contained in the tank during each simulated flight cycle.

A total of 100 flight simulation cycles has been run with space equilibrium thermal performance tests conducted intermittently during the testing to evaluate the effects of flight environments on the MLI thermal performance. MLI space equilibrium thermal performance is shown in Fig. 3 as a function of the number of flight simulation cycles performed. A degradation of 23% in thermal performance was found from the "as new" condition to the 50-flight-cycle condition. After 52 cycles, testing was halted and a partial visual examination was made of the MLI system in the vicinity of the purge bag door. Several links joining the blanket gores were found to be broken, causing three seams to separate slightly (about 2 by 0.25 in.). After the seams were repaired, but before resumption of testing, a high-temperature (685R)/vacuum soak was performed on the tank and MLI to remove any absorbed moisture.

A new basepoint thermal equilibrium test showed that, after seam repairs and MLI back-out, the MLI returned to nearly its "as new" condition. Another 48 life cycles were performed on the cryogenic storage system and the thermal performance was found to degrade slightly (approximately 9%) from the new, 52-cycle basepoint value.

Complete disassembly and visual examination were performed following completion of all testing. The only system damage noted was failure of one MLI blanket fiberglass support pin, three twin-pin links, and several goldized tape bond joints used as secondary seam closing support. The effect of these minor failures was to allow blanket gore seams to "gap" locally and was undoubtedly the cause of the degradation in the system performance with increasing system service life.

Conclusions

A new, completely reusable flightweight cryogenic storage system has been developed and demonstrated that represents a significant improvement in thermal performance over previously developed single-use flightweight systems. Simulated Space Shuttle life cycle testing produced some degradation in thermal performance due to the MLI blanket gore seam "gapping." The use of overlapping blanket face sheets and positive seam closure (e.g., Velcro fasteners) should eliminate this problem in future designs. The details of the design and performance data for the recently developed reusable cryogenic storage system are presented in Ref. 3.

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Photographic Pyrometry in an Aeroballistic Range

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IN Ref. 1, a novel method using pyrometric techniques to measure surface temperatures on hyperballistic models has been developed for the AEDC ballistics range. Accurate measurements of surface temperature and its downrange variation would be invaluable for deducing quantitative heating rates. However, a major weakness of the method is the inability to differentiate between radiation (in the sensitive bandwidth of the pyrometer) from the hot model surface and extraneous radiation from the shock layer, boundary layer, or chemilumenescence resulting from gas-surface reactions. Chemiluminescent reactions between the hot boundary layer gases and ablating metallic surfaces are particularly serious and may result in anomalously high "surface" temperature measurements.[2]

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Index categories: Radiatively Coupled Flows and Heat Transfer; LV/M Aerodynamic Heating; LV/M Simulation.

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