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# Compensating for Cryogenic Propellant Boiloff for a Cargo Mission to Mars

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**A previous work by the authors (*Architecture Study for a Fuel Depot Supplied from Lunar Resources* – AIAA 2016-5306) examined architectures for a fuel depot supplied from lunar ice deposits. The study recommended a fuel depot be located at Earth-Moon L1. The study used three design reference missions in the formulation of candidate architectures – among them a Mars Cargo Vehicle (MCV). Each vehicle used 60 layers of multi-layer insulation (MLI) to minimize the boiloff of cryogenic propellants. Boiloff was shown to not be a driver in architecture selection, but it was noted that reducing the number of layers of MLI from 60 to 30 reduced the MLI mass by almost 4,000 kg, but the predicted boiloff only increased about 2,000 kg. This suggested that further investigation was needed to determine the optimum balance between MLI mass, predicted boiloff, and the mass of propellant needed to compensate for that boiloff. While holding the payload mass constant, this paper uses the Modified Lockheed Model to calculate predicted boiloff, and uses multiple iterations of the classic rocket equation to determine how much propellant will be needed to compensate for boiloff over a 288-day mission to Mars.**

## Nomenclature

### A. Calculating outside temperature of the spacecraft

| | | |
|---|---|---|
| $T$ | = | outside temperature of the spacecraft (K) |
| $\Sigma$ | = | Boltzmann's constant = 5.67051 x $10^{-8}$ W/m$^2$ K$^{-4}$ |
| $\alpha$ | = | absorptivity |
| $\varepsilon$ | = | emissivity |
| $S$ | = | solar flux |
| $A_p$ | = | projected area of the propellant tank |
| $A$ | = | total surface area of the propellant tank |

### B. Modified Lockheed Model

| | | |
|---|---|---|
| $q$ | = | heat transfer rate in W/m$^2$ |
| $\varepsilon$ | = | emissivity of the inner layers of MLI |
| $T_h$ | = | temperature on outside tank surface (K) |
| $T_c$ | = | propellant temperature |
| $T$ | = | $(T_h+T_c)/2$ |
| $N^*$ | = | number of layers/cm of MLI |
| $N_s$ | = | number of layers of MLI |
| $P$ | = | pressure between the layers of MLI (Torr) |

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# I. Introduction

A previous work by the authors (*Architecture Study for a Fuel Depot Supplied from Lunar Resources* – AIAA 2016-5306) examined potential architectures for a fuel depot supplied from lunar ice deposits.[1] Among other recommendations, the study recommended that a fuel depot be located at Earth-Moon L1. The study used three design reference missions (DRM) – a Commercial Satellite Servicing Vehicle (CSSV), a Mars Cargo Vehicle (MCV), and a Lunar Tanker Vehicle (LTV) in the formulation of candidate architectures. Each vehicle used 60 layers of multi-layer insulation (MLI) to minimize the boiloff of cryogenic propellants.

While boiloff was shown to not be a driver in architecture selection, it was noted that reducing the number of layers of MLI from 60 layers to 30 layers reduced the MLI mass by almost 4,000 kg, while the predicted boiloff only increased approximately 2,000 kg. This suggested that further investigation was needed to determine the optimum balance between MLI mass, predicted boiloff, and the mass of propellant needed to compensate for that boiloff. This paper examines this issue by focusing on the hypothetical Mars Cargo Vehicle (MCV) and its journey from the depot location at Earth-Moon L1 to Mars – comprising Trans-Mars Injection (TMI), a 288-day trajectory, and insertion into Martian orbit.

# II. Mars Mission Specifics

## A. Mars Cargo Vehicle (MCV)

The Mars Cargo Vehicle in this study is adapted from NASA's 2005 Exploration Systems Architecture Study (ESAS).[2] The ESAS described NASA's plans for going back to the Moon and on to the planet Mars. For the Mars mission, NASA planned to send four cargo vehicles to Mars. These cargo rockets would arrive at Mars in advance of the astronaut crew, which would be transported in a crew vehicle. The cargo vehicles would carry supplies, a Mars habitat, rovers, and anything else needed. Once the cargo vehicles had arrived safely, the astronauts would then follow.

The cargo vehicles themselves were the upper stage (Earth Departure Stage, or EDS) of the heavy lift Ares V vehicle of the [now-cancelled] Constellation Program. Four Ares V rockets with the EDS upper stage were to be launched over a period of 26 months.[2] In the ESAS, the EDS stages were assumed to be powered by nuclear-thermal propulsion (NTP). Nuclear thermal propulsion has two advantages over chemical propulsion. It has a specific impulse ($I_{sp}$) roughly double that of the best chemical engines – as much as 925 seconds -- yet much less mass overall. For the purposes of this study, however, the EDS configured for lunar missions is used instead. This EDS, what is called the Mars Cargo Vehicle here, is powered by a single LH2/LO2 J-2X engine with an $I_{sp}$ (vacuum) of 449 seconds.[3] Characteristics of the MCV are summarized in Fig. 1 below.

Mars Cargo Vehicle (MCV)

- Heavy lift vehicle to pre-position equipment and supplies in Mars orbit prior to crew arrival; based on Ares V Earth Departure Stage
- The MCV launches into a 200 km orbit. It docks with its cargo payload, rendezvous with a depot at Earth-Moon L1 and refuels, and performs TMI. Upon arrival at Mars the MCV performs an orbit insertion burn into a 200 km orbit.
- Dry mass (less MLI): 24,000 kg; Payload Mass (for this study): 44,000 kg
- Powered by a single J2-X engine; $I_{sp}$ = 449 seconds

**Figure 1. Characteristics of the Mars Cargo Vehicle (MCV)**

## B. Mission Design

In the ESAS study, the heavy lift vehicle places the EDS and its payload into a 200 km/ 28.5 degree orbit.[4] The EDS docks with a lunar lander, and performs a trans-lunar injection from LEO. For the Mars Cargo Vehicle DRM, the MCV is delivered to the same orbit as the EDS. The MCV docks with its cargo, then maneuvers to the depot and refuels. Refueling at the depot enables the MCV to perform the TMI maneuver and the Mars Orbit Insertion upon arrival. Like the EDS, the MCV launches with 250,000 kg of propellant. After achieving LEO, the MCV has 103,500 kg of propellant remaining, which limits the mass of the payload which can be taken to the depot at L1.[3]

## III. Objective Function

Initially the authors sought to identify the point at which reductions in MLI mass equaled the increases in propellant boiloff. Calculations showed that the reduction in MLI mass (from 60 layers) equals the increase in propellant boiloff (from that predicted using 60 layers) when 11 layers of MLI are used. However, this information by itself is incomplete and tells the wrong story. What is really necessary is to account for the mass of the MLI, the mass of the propellant lost to boiloff, and the mass of the propellant necessary to compensate for that which is lost.

Accordingly, this effort proposes to investigate the relationship between predicted boiloff and MLI mass, and will seek to determine an optimum number of layers of MLI that will minimize overall spacecraft mass according to the following objective function Eq. (1):

$$\text{Minimize Mass}_{S/C} = \text{Mass}_{VEH} + \text{Mass}_{MLI} + \text{Mass}_{PROP} + \text{Mass}_{P/L} \tag{1}$$

where $\text{Mass}_{S/C}$ = overall mass of the spacecraft
$\text{Mass}_{VEH}$ = dry mass of the vehicle, less the MLI = 24,000 kg
$\text{Mass}_{MLI}$ = mass of the MLI
$\text{Mass}_{PROP}$ = mass of the propellant
$\text{Mass}_{P/L}$ = mass of the payload = 44,000 kg

As stated earlier, the dry mass of the vehicle is 24,000 kg. For this study, the mass of the payload is held constant at 44,000 kg. It is necessary to hold the payload constant in order to illuminate the relationships between the MLI, the propellant lost to boiloff, and the propellant needed to send the vehicle to Mars. The value of 44,000 kg represents (to the nearest 1,000 kg) the maximum payload the MCV could send to the depot using the 103,350 kg of propellant remaining after its initial launch into LEO.

It will be seen that MLI mass influences the mass of the predicted propellant boiloff. More MLI reduces boiloff, but the additional mass of the MLI adds to the overall vehicle mass and thus requires additional propellant. If less MLI is used, less propellant is required. But less MLI means increased propellant boiloff. Taking on additional propellant to offset the predicted boiloff itself requires additional propellant. This is the “tyranny of the rocket equation.” So overall, the challenge is to select the number of layers of MLI that provides the least mass penalty to the vehicle, but also minimizes the predicted boiloff and thus the mass of propellant needed to compensate for the predicted losses.

## IV. Required Calculations

### A. Calculating MLI Mass

The MLI mass is computed assuming a 1 mm separation between layers. The MCV LH2 and LO2 tanks are assumed to be cylinders, with the dimensions taken from the earlier study. (The LH2 tank is 6.36 x 10 m, while the LO2 tank is 2.29 x 10 m.) The densities of the MLI in grams/m$^2$ will be taken from the Sheldahl Red Book[5], a commercial specification handbook. A formula for the surface area of a cylinder will be used to calculate the surface area of the MLI, with the dimensions incremented to account for the 1mm separation between layers.

It is common practice for the outer layer of MLI to be chosen such that it has low absorptivity but high emissivity. Thus, the outer layer of MLI reflects as much of the incoming energy as possible, but the high value for emissivity means the MLI allows as much heat to escape away from the propellant tank as possible. The inner layers, on the other hand, use MLI with low emissivity. The low emissivity limits the amount of infrared radiation transmitted from layer to layer, and limits the amount of infrared reaching the propellant tank. The characteristics of the MLI chosen for the study are shown in Table 1 below.

Table 1. Characteristics of the MLI used in the study.

| Layer/source | MLI Chosen | Absorptivity | Emissivity | Thickness | Density |
|---|---|---|---|---|---|
| Outer layer/ Sheldahl p.53 | Aluminum coated fluoro ethylene propylene (FEP) | 0.14 | 0.60 | 2 mils | 109 g/m$^2$ |
| Inner layers/ Sheldahl p.19 | Aluminum coated polyethylene terephthalate (PET) | 0.14 | 0.035 | 2 mils | 71 g/m$^2$ |

Although the MLI has little mass, using it in a significant number of layers adds up. Table 2 below gives the mass for the MLI to insulate the liquid hydrogen and liquid oxygen tanks of the MCV. It can be seen that as the number of layers of MLI increases, the mass of the blankets becomes significant.

Table 2: Calculated mass for selected layers of MLI.

| Layers of MLI | MLI Mass LH2 tank (kg) | MLI Mass LO2 tank (kg) | Combined Mass (kg) |
|---|---|---|---|
| 10 | 772 | 579 | 1,351 |
| 20 | 1,508 | 1,130 | 2,638 |
| 30 | 2,245 | 1,683 | 3,928 |
| 40 | 2,985 | 2,238 | 5,223 |
| 50 | 3,726 | 2,795 | 6,521 |
| 60 | 4,470 | 3,324 | 7,794 |

### B. Calculating Propellant Tank Surface Temperature

To determine propellant losses, the thermal environment surrounding the spacecraft must be characterized. This permits the calculation of the temperature of the external surface of the spacecraft. The surface temperature, along with the propellant tank size and shape and other factors, allows estimation of the boiloff rate.

For spacecraft in Earth orbit, the thermal environment consists of three external sources of heat – energy from the Sun (solar flux), Earth-reflected heating (albedo times the incident solar flux), and Earth-emitted radiation, also called Earth infrared radiation, or simply Earth-IR. At Earth-Moon L1, the values for Earth reflected heating and Earth-IR are almost non-existent, and can be ignored.

From Thornton, environmental heating rates depend on the altitude and orientation of the spacecraft with respect to sources of heat.[6] The solar heat received by the spacecraft surface ($q_s$) is given by Eq. (2)

$$q_s = 1{,}367\ a_s \cos \psi \tag{2}$$

where $a_s$ is the surface absorptivity, and $\psi$ is the angle between the solar flux vector and the surface normal.[6] The solar constant is 1,367 W/m$^2$ at 1 AU. Surface absorptivity is set conservatively as $a_s = 1$. Conservatively assuming the spacecraft normal to the Sun and the Earth and Mars as coplanar with the Sun such that $\psi = 0$ degrees, leaves cosine of $\psi = 1$. Thus the solar flux received by the spacecraft at L1 is 1,367 W/m$^2$.

The value of the solar flux at the surface of Mars is 593 W/m$^2$.[7] For simplicity, the two values are averaged: (1,367 + 593)/2 = 980 W/m$^2$. This average value of 980 W/m$^2$ will be used to calculate surface temperature of the MCV propellant tanks.

To calculate the surface temperature of the propellant tanks, the surface area of the tanks and the projected (2-dimensional) area of the tanks is required. For the MCV propellant tanks, the surface area is simply the surface area of a cylinder, and is equal to $2\pi r^2 h + 2\pi rh$. Since the tanks are assumed to be at right angles to and coplanar with the Sun, the projected area is simply a rectangle with the length being the length of the tank and the width being the diameter of the tank. MCV propellant tank dimensions, surface area, and projected area are shown in Table 3 below.

Table 3. MCV propellant tank dimensions

| Tank | Dimensions (l x d) | Surface area (A) (m$^2$) | Projected area ($A_p$) (m$^2$) |
|---|---|---|---|
| LH2 | 6.36m x 10m | 357.3 | 63.6 |
| LO2 | 2.29m x 10m | 229.0 | 22.9 |

Given the thermal environment and the illuminated area of the tank, the surface temperature can be estimated using the expression Eq. (3) from Wertz and Larson.[8] This equation approximates the outside temperature of a spacecraft in Kelvin, and uses values for the solar flux, absorptivity and emissivity of the outer layer of MLI, and the ratio of the projected area to the surface area being considered.

$$\sigma T^4 = (\alpha/\varepsilon)(S) \times (A_p/A) \quad (3)$$

where $T$ = outside temperature of the spacecraft (K)
$\sigma$ = Boltzmann's constant = $5.67051 \times 10^{-8}$ W/m$^2$ K$^{-4}$
$\alpha$ = absorptivity (= 0.14 for outer layer of MLI)
$\varepsilon$ = emissivity (= 0.60 for outer layer of MLI)
$S$ = solar flux (980 W/m$^2$) (average)
$A_p$ = projected area of the propellant tank
$A$ = total surface area of the propellant tank

The calculated values for the average surface temperature for the MCV propellant tanks during the journey from L1 to Mars orbit are given in Table 4 below:

Table 4. Average surface temperature for MCV propellant tanks

| Propellant Tank | Absorptivity $\alpha$ | Emissivity $\varepsilon$ | Average solar flux (W/m$^2$) | $A_p/A$ | Average surface temperature (K) |
|---|---|---|---|---|---|
| LH2 | 0.14 | 0.60 | 980 | 0.1781 | 163.7 |
| LO2 | 0.14 | 0.60 | 980 | 0.1000 | 141.7 |

### C. Calculating Predicted Propellant Boiloff

The Modified Lockheed Model was used to calculate the predicted propellant boiloff.[9] The Modified Lockheed Model, Eq. (4), considers three heat transfer mechanisms i.e., solid conduction, radiation between blanket layers, and gas conduction, and yields the rate (q) of heat transfer through the layers of insulation into the fuel tank in W/m$^2$.

$$q = 0.00024*(0.017+7E\text{-}6(800-T) +0.0228*\ln(T))*(N^*)^{2.63}(T_h-T_c)/N_s + 4.944E\text{-}10*\varepsilon*(T_h^{4.67}-T_c^{4.67})/N_s + 1.46E4*P*(T_h^{0.52}-T_c^{0.52})/N_s \quad (4)$$

where $q$ = heat transfer rate in W/m$^2$
$\varepsilon$ = emissivity of the inner layers of MLI (here = 0.035)
$T_h$ = temperature on outside tank surface (K)
$T_c$ = propellant temperature (20 K for LH2, 80 K for LO2)
$T$ = $(T_h+T_c)/2$
$N^*$ = number of layers/cm of MLI
$N_s$ = number of layers of MLI, and
$P$ = pressure between the layers of MLI (Torr)

A density (thickness) of the MLI blankets of 10 layers per centimeter is assumed. As stated earlier, an outer layer of low absorptivity-high emissivity aluminum coated fluoroethylene propylene (FEP) is used, along with inner layers of low emissivity aluminum-coated polyethylene terephthalate (PET, commonly known as Mylar). The outer layer and the inner layers work together to minimize the transfer of heat into the propellant tanks.

The output of the Modified Lockheed Model is q, the rate of heat transfer through the layers of insulation into the fuel tank in W/m$^2$. The total heat transfer (Watts) is calculated by multiplying the rate of heat transfer times the surface area of the tank. Then, dividing the total heat transfer by the heat of vaporization for the cryogenic fluid in the tank (in Joules/kilogram) yields the rate of boiloff (in kilograms/second). The boiloff rate in kilograms/hour is obtained by multiplying the kg/sec rate x 3,600 seconds/hour.

The time of flight for travel to Mars was based on "conjunction class" trajectories where the Earth at launch and Mars at arrival are nearly in direct opposition. Nine such launch opportunities from the year 2002-2011 are recorded in the NASA's Interplanetary Mission Design Handbook.[10] The time of flight values were averaged. The average time of flight over the nine flights was 288 days. This value (converted to hours) was used in boiloff calculations. Example boiloff rates and masses are shown in Table 5 below. It can be seen that liquid hydrogen, being a smaller molecule than liquid oxygen, boils off much more rapidly than liquid oxygen. Notice, too, the beneficial effect of the MLI. Twenty layers of MLI significantly reduce the overall mass of the predicted boiloff.

Table 5. Sample Boiloff Rates and Boiloff for the MCV

| Number of Layers of MLI | LH2 Boiloff Rate (kg/hr) | LH2 Boiloff (kg) | LO2 Boiloff Rate (kg/hr) | LO2 Boiloff (kg) | Total Boiloff (kg) |
|---|---|---|---|---|---|
| 10 | 0.6357 | 4,394 | 0.3854 | 2,664 | 7,058 |
| 20 | 0.3178 | 2,197 | 0.1927 | 1,332 | 3,529 |
| 30 | 0.2119 | 1,465 | 0.1285 | 888 | 2,353 |
| 40 | 0.1589 | 1,098 | 0.0964 | 666 | 1,764 |
| 50 | 0.1271 | 879 | 0.0771 | 533 | 1,412 |
| 60 | 0.1059 | 732 | 0.0642 | 444 | 1,176 |

**D. Calculating Propellant Consumption**

The classic rocket equation Eq. (5) was used to calculate propellant consumption.

$$\Delta v = I_{sp}\, g_o\, \mathit{ln}\, (m_i/m_f) \tag{5}$$

where $\Delta v$ = Trans-Mars Insertion + orbit insertion around Mars = 4.327 km/s
$I_{sp}$ = specific impulse (seconds) = 449 seconds for J2-X engine
$g_o$ = Earth's surface gravitational acceleration, 9.81 m/s$^2$
$m_i$ = initial vehicle mass (kg)
$m_f$ = final vehicle mass (kg)

Since the MCV is a single stage vehicle, application of the rocket equation is straightforward. Payload mass is held constant at 44,000 kg. Vehicle dry mass is 24,000 kg. MLI mass varies based on the number of layers used. The value for the $\Delta v$, 4.327 km/s, was calculated using the Patched Conic Method, starting at L1, and inserting into a 200 km Martian orbit. Solve for the final mass, $m_f$, then subtract $m_f$ from $m_i$ to determine the propellant consumed.

**E. Calculating Propellant to Compensate for Losses**

Iterations of the rocket equation were used to calculate propellant consumption to the nearest kilogram. Payload mass was held constant at 44,000 kg. The first series of calculations assumed that no propellant was lost to boiloff, and that the propellant was completely consumed (no propellant remaining after Mars orbit insertion; $m_i - m_f = 0$). The second series of calculations assumed that propellant would be lost to boiloff. The mass of fuel at the start of TMI was adjusted until the fuel remaining after Mars orbit insertion ($m_i - m_f$) equaled the mass of the predicted boiloff. Subtracting the propellant mass (with no boiloff) from the propellant mass (with boiloff) then yields the "penalty" – the mass of propellant the vehicle will consume to transport the additional propellant to compensate for the predicted boiloff. Example results are shown in Table 6.

Table 6. Example Propellant Consumption Figures for various layers of MLI

| Layers of MLI | Propellant Consumed (no boiloff) (kg) | Propellant Consumed (with boiloff) (kg) | Predicted Boiloff (kg) | "Penalty" to replace lost propellant (kg) |
|---|---|---|---|---|
| 10 | 124,549 | 144,282 | 7,058 | 12,675 |
| 20 | 126,866 | 136,728 | 3,529 | 6,333 |
| 30 | 129,192 | 135,756 | 2,353 | 4,211 |
| 40 | 131,501 | 136,435 | 1,764 | 3,170 |
| 50 | 133,844 | 137,780 | 1,412 | 2,524 |
| 60 | 136,123 | 139,408 | 1,176 | 2,109 |

## V. Results

The relationship between MLI mass and propellant mass is evident in Fig. 2 below. The graph illustrates this relationship for the sample numbers of layers of MLI as used before. At the bottom of each bar in the graph is shown the vehicle dry mass, 24,000 kg, which is constant. Above that is the payload mass, 44,000 kg, which is also constant. Next comes the mass of the MLI blanket. It can be seen that as the number of layers of MLI increase, the mass of the MLI becomes more and more significant. Notice, too, as the mass of the MLI increases, the mass of the propellant needed to for the mission also increases Shown next is the mass of the propellant lost due to boiloff. It can be seen that as few as 20 layers of MLI causes the mass of propellant lost to boiloff to drop sharply. Last, at the top of each bar is shown the "propellant penalty". This is the mass of propellant the vehicle will consume to transport the additional propellant to compensate for the predicted boiloff. As the mass lost to boiloff is reduced by adding layers of MLI, the mass of the propellant penalty is also reduced.

Overall, the graph suggests the "sweet spot" – the point where overall spacecraft mass is the least occurs when approximately 20 layers of MLI are used. Further calculations and a tabular format will put a sharper point on the results.

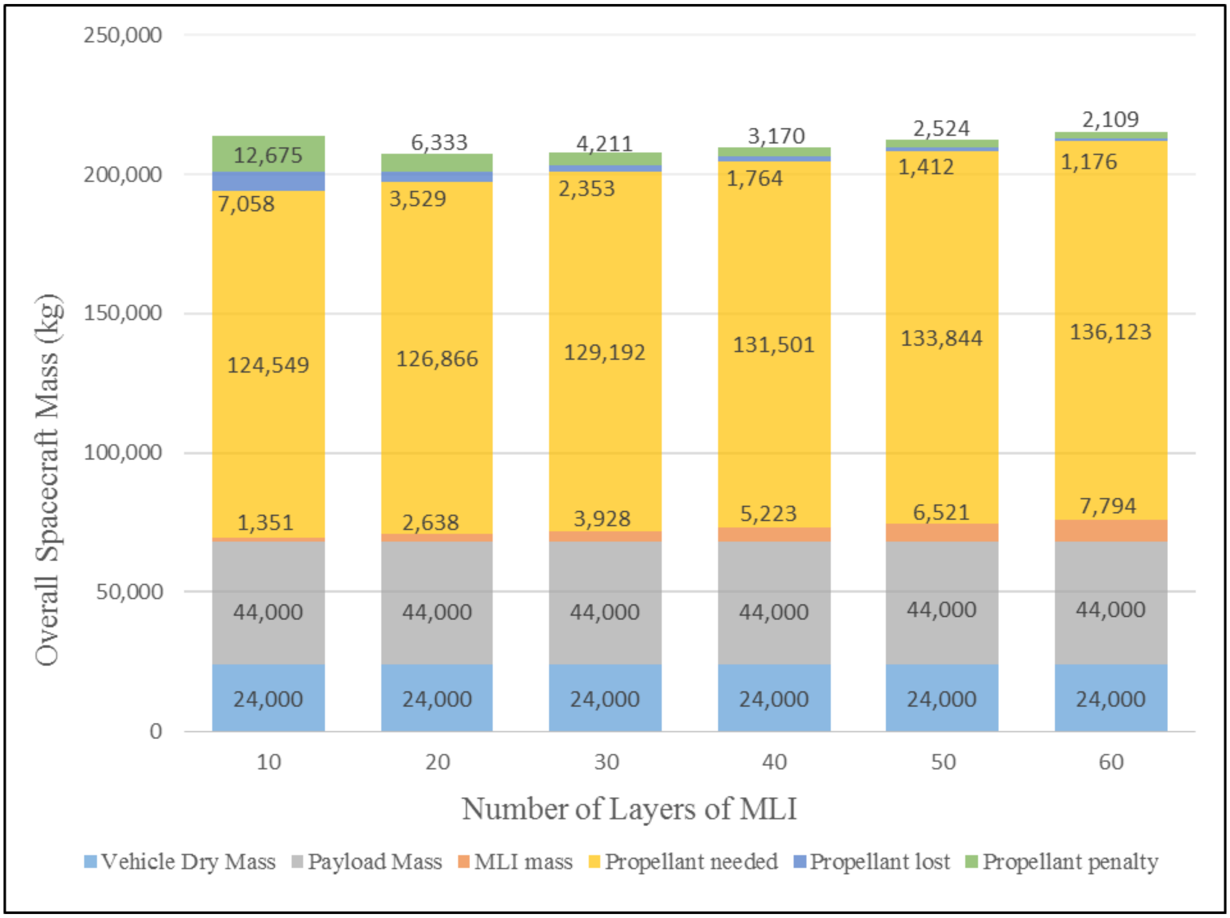


**Figure 2. Relationship between MLI mass and propellant mass**

Table 7 below shows the results of the computations. In the table, the "propellant needed" is that which the MCV would consume if there was no boiloff. The "propellant lost" is that lost to boiloff, which must be replaced. The "propellant penalty" is the additional propellant the vehicle will consume to transport the additional propellant to replace the propellant lost. It can be seen the minimum spacecraft mass is achieved when 23 layers of MLI are used. With fewer layers of MLI, propellant lost to boiloff increases, and the penalty incurred in replacing the lost propellant becomes more dominant, and spacecraft mass increases. With more layers of MLI, spacecraft mass increases directly and the mass of propellant required also increases. The added MLI mass is dominant over propellant lost to boiloff.

Table 7. Results of computations to minimize spacecraft mass

| Number of MLI layers | Veh dry mass (kg) ($Mass_{VEH}$) | MLI mass (kg) ($Mass_{MLI}$) | Payload mass (kg) ($Mass_{P/L}$) | Propellant ($Mass_{PROP}$) | | | Spacecraft mass (kg) ($Mass_{S/C}$) |
|---|---|---|---|---|---|---|---|
| | | | | Propellant needed (kg) | Propellant lost (kg) | Propellant penalty (kg) | |
| 10 | 24,000 | 1,351 | 44,000 | 124,549 | 7,058 | 12,675 | 213,633 |
| 20 | 24,000 | 2,638 | 44,000 | 126,866 | 3,529 | 6,333 | 207,366 |
| 21 | 24,000 | 2,767 | 44,000 | 127,097 | 3,361 | 6,032 | 207,257 |
| 22 | 24,000 | 2,896 | 44,000 | 127,308 | 3,208 | 5,776 | 207,188 |
| **23** | **24,000** | **3,024** | **44,000** | **127,583** | **3,068** | **5,479** | **207,154** |
| 24 | 24,000 | 3,153 | 44,000 | 127,795 | 2,941 | 5,272 | 207,161 |
| 25 | 24,000 | 3,283 | 44,000 | 128,071 | 2,824 | 5,018 | 207,196 |
| 30 | 24,000 | 3,928 | 44,000 | 129,192 | 2,353 | 4,211 | 207,684 |
| 40 | 24,000 | 5,223 | 44,000 | 131,501 | 1,764 | 3,170 | 209,658 |
| 50 | 24,000 | 6,521 | 44,000 | 133,844 | 1,412 | 2,524 | 212,301 |
| 60 | 24,000 | 7,794 | 44,000 | 136,123 | 1,176 | 2,109 | 215,202 |

It is instructive to compare the mass of the predicted boiloff with the mass of propellant required to compensate for that boiloff – that is, the mass of the propellant to replace that which will be lost, plus the mass of propellant the vehicle will consume to transport it. See Table 8 below. It can be seen that regardless of the number of layers of

Table 8: Ratio of Compensating Propellant to the Propellant Lost by Boiloff

| Number of MLI layers | Predicted Propellant Boiloff (kg) | Calculated Propellant to Compensate (kg) | Ratio |
|---|---|---|---|
| 10 | 7,058 | 19,733 | 2.796 |
| 20 | 3,529 | 9,862 | 2.795 |
| **23** | **5,479** | **8,547** | **2.786** |
| 30 | 2,353 | 6,564 | 2.790 |
| 40 | 1,764 | 4,934 | 2.797 |
| 50 | 1,412 | 3,936 | 2.788 |
| 60 | 1,176 | 3,285 | 2.793 |

MLI used, the ratio of mass of propellant needed to compensate for predicted losses is roughly 3-to-1. This appears to be a rule of thumb that mission designers could use in planning such a mission. However, trial calculations show this ratio is tied to the mission delta-v (the delta-v for Trans-Mars-Insertion – departing L1 on a trajectory to Mars -- plus the delta-v required to place the MCV into a 200 km orbit around Mars.) A different mission delta-v to a different planet -- say for a flight to Venus -- would result in a different ratio.

Lastly, it is also instructive to compare the mass of the propellant predicted to be lost to the initial propellant mass (Table 9). At the optimum number of layers of MLI, the percentage of propellant lost to boiloff is only 2.40%. This suggests that expensive zero-boiloff (ZBO) technologies, such as the use of cryocoolers, may not be required for space vehicles using conventional cryogenic propellants. Carrying additional propellant to offset propellant losses is the more simple solution.

Table 9.  Boiloff as a Percentage of Propellant Mass

| Number of MLI layers | Predicted Propellant Boiloff (kg) | Overall Propellant Mass (kg) | Percentage lost |
|---|---|---|---|
| 10 | 7,058 | 124,549 | 5.67% |
| 20 | 3,529 | 126,866 | 2.78% |
| **23** | **5,479** | **127,583** | **2.40%** |
| 30 | 2,353 | 129,192 | 1.82% |
| 40 | 1,764 | 131,501 | 1.34% |
| 50 | 1,412 | 133,844 | 1.05% |
| 60 | 1,176 | 136,123 | 0.86% |

## VI. Conclusions

- The loss of propellant due to boiloff for a mission to Mars can be predicted using the Modified Lockheed Model, based on the time-of-flight, the thermal environment, the size and configuration of the spacecraft's propellant tanks, the number of layers of MLI used, and the absorptivity and emissivity characteristics of that MLI.

- Passive measures such as adding layers of MLI can greatly reduce the mass of propellant lost to boiloff. MLI limits the transfer of thermal energy into the propellant tanks.

- Although MLI is very lightweight – its mass is measured in grams per square meter – MLI "blankets", made up of multiple layers of MLI, can add significant mass to the spacecraft as the number of layers increases. At some point, the savings in propellant boiloff is exceeded by the additional propellant needed to propel this increased MLI mass.

- An optimum number of layers of MLI for a given spacecraft which reduces boiloff yet minimizes overall spacecraft mass can be determined by calculating the MLI mass, the mass of propellant lost to boiloff, and the mass of propellant needed to compensate for the boiloff, and determining which combination of factors minimizes overall spacecraft mass.

- The thermal environment between Earth-Moon L1 and Mars is such that the boiloff of cryogenic propellants theoretically can be limited to an acceptable level.

- The percentage of propellant mass lost to boiloff calculated for this study ranged from less than 1 percent to just under 6 percent, with the optimum value being 2.4%. This suggests that expensive zero-boiloff (ZBO) technologies, such as the use of cryocoolers, may not be required for space vehicles using conventional cryogenic propellants. Not only do cryocoolers reduce the available payload, they require the vehicle to produce additional electrical power, which itself means additional mass which would also reduce the available payload. Simply carrying additional propellant to offset propellant losses appears to be the best solution.

- Regardless of the number of layers of MLI used, the ratio of mass of propellant needed to compensate for predicted losses for this mission is roughly 3-to-1. This appears to be a rule of thumb that mission designers could use in planning a Mars mission.

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