Liquefaction and Storage of In-Situ Oxygen on the Surface of Mars

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The In-Situ production of propellants for Martian and Lunar missions has been heavily discussed since the mid 1990's. One portion of the production of the propellants is the liquefaction, storage, and delivery of the propellants to the stage tanks. Two key technology development efforts are required: large refrigeration systems (cryocoolers) to perform the liquefaction and high performance insulation within a soft vacuum environment. Several different concepts of operation may be employed to liquefy the propellants based on how and where these two technologies are implemented. The concepts that were investigated include: using an accumulator tank to store the propellant until it is needed, liquefying in the flow stream going into the tank, and liquefying in the flight propellant tank itself. The different concept of operations were studied to assess the mass and power impacts of each concept. Additionally, the trade between insulation performance and cryocooler mass was performed to give performance targets for soft vacuum insulation development. It was found that liquefying within the flight propellant tank itself adds the least mass and power requirements to the mission.

Nomenclature

Q'_{Rad} = radiator heat flux, W/m^2
Q'_{Convec} = convection heat flux, W/m^2
$Q'_{Insolation}$ = solar insolation heat flux, W/m^2
Q'_{load} = radiator heat load, W/m^2
$H_{Nat,conv}$ = natural convection heat transfer coefficient, $W/m^2/K$
$T_{Radiator}$ = radiator temperature, K
T_{sky} = Mars sky temperature, K
T_{atm} = Mars atmosphere temperature, K
α = solar absorptivity
ε = infrared emissivity
σ = Stefan-Boltzman constant, 5.67×10^{-8} $W/m^2/K^4$

I. Introduction

Proposed human missions to Mars have always relied on launching multiple large cryogenic upper stages into Low Earth Orbit and having them assembled prior to firing into a Martian transfer orbit.[1,2] In order to minimize the number of launches required, the payload mass delivered to the Martian surface must be minimized. On the return stage (often called the ascent stage) between 75% to 80% of the mass is propellant, of which approximately 80% of

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that is liquid oxygen. In order to reduce the initial mass needed for a human trip to Mars, oxygen can be produced on the surface of Mars.

Oxygen production through the electrolysis of carbon dioxide using a high temperature solid oxide electrolyzer is currently being investigated by NASA and appears to be the leading technology for in-situ production of oxygen[3]. However, for it to be available for propulsion and other end users, the oxygen needs to be cooled, liquefied, and stored for durations up to two years, either in the actual Mars Ascent Vehicle propulsion tanks or in a separate tank. Recent investigations have demonstrated the feasibility of not only zero-boil-off but also pressure control of oxygen using high efficiency cryocoolers to prevent boil-off and control pressure within a tank using a cryocooler with excess lift capacity[4]. By using the excess lift capacity to liquefy an oxygen flow rate, a similar system could be used for in-situ liquefaction. Initial analytical investigations into scaling the reverse turbo-Brayton cycle cryocoolers to the sizes needed for this liquefaction effort has also yielded promising results[5].

While the oxygen production has been developed over the last several years, the cryogenic systems for operation on Mars have not been heavily invested in or studied[3,6,7]. Many of the cryogenic fluid storage technologies needed are similar to in-space technologies developed over the years[8], but the liquefaction systems such as demonstrated for zero boil-off, would operate under different parameters. Top level investigations into the surface architecture have not fully been vetted, but differences even of whether the liquefaction system would be on the lander or the In Situ Resource Utilization (ISRU) system drive wildly differing cryogenic system requirements. These trades are affected by both the duration of stay and whether or not assets left on the surface can be used for future missions. Understanding the mass impacts of these trades are needed to allow proper system design for Martian missions using ISRU to move forward.

Additionally, the insulation system would have to work on the Martian surface at a pressure of 1 kPa[9]. Insulation performance (especially multilayer insulation, which is typically used for spacecraft) is highly dependent on the residual gas pressure of the system as shown in Figure 1. Increasing the pressure from 10^{-5} torr to 5 torr can cause a 1 – 2 order of magnitude loss in thermal performance for high performing systems. Fesmire and Augustinowicz[10] investigated multiple insulation types for these pressures, and resulting from their work, some progress has been made on insulation systems such as aerogel based systems[11]. Similarly, a vacuum jacket can be used to maintain a space like vacuum by adding an outer shell. By increasing the performance of the insulation, the cryocooler and liquefaction requirements decrease. Understanding the mass based trades between the insulation and the liquefaction systems is needed to set performance targets for both systems.

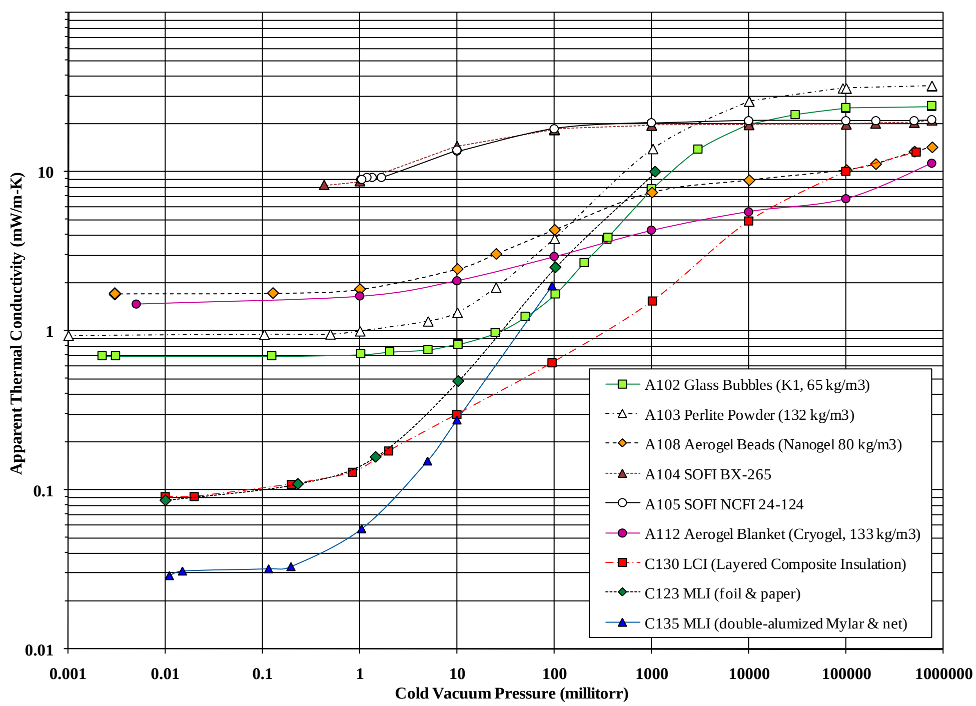

Figure 1. Insulation Thermal Conductivity as a Function of Vacuum Pressure

This paper investigates the various liquefaction processes and storage options that could be used on the surface of Mars. System mass and power estimates for both a demonstration system and a full scale system will be investigated. The components within the system are generally a cryocooler, a radiator, power production unit, and a tank with integrated heat exchanger for liquefaction and storage. Additionally, with the thermal balance being performed, there is a trade between insulation system performance (including mass) and the refrigeration system mass.

II. Liquefaction System Analysis

Based on the most recent system analysis done for the current Mars Ascent Vehicle (MAV), the MAV will arrive on the surface of Mars years in advance to liquefy enough liquid oxygen to allow for a return: (23,000 kg)[12]. In order to produce enough liquid oxygen, a liquefaction rate of 2.2 kg/hr is required assuming a duration of 15 months.

A process schematic of the liquefaction system can be seen below in Figure 2. The power and heat rejection requirements are based off of GO_2 supplied by the electrolyzer at 800℃ and 101.3 kPa and a liquefaction rate of 2.2 kg/hr for the full scale system. The radiators and refrigeration system will be sized using the predicted worst case (maximum) Mars sky and atmospheric temperatures[9]. However, just because this assumption is made in the present study doesn't mean it is appropriate. It is highly probable that when a full mission is actually designed, the system can be undersized from worst case conditions so that a cyclical tank pressure profile can be achieved using the extra power available to the system at non-worst case conditions.

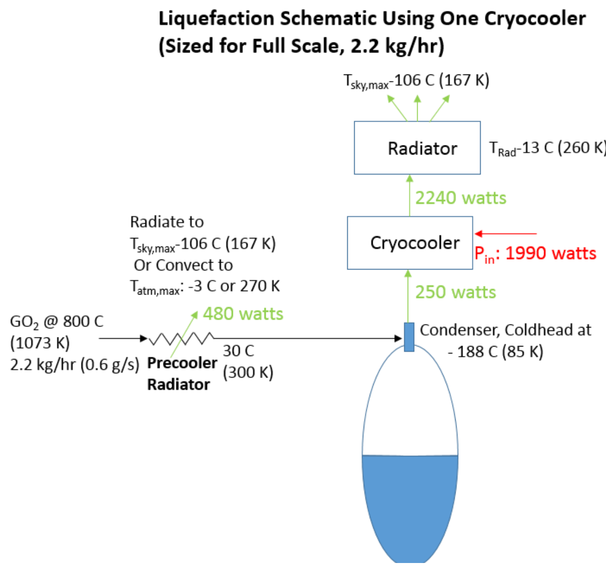

Figure 2. Schematic of ISRU Liquefaction Plant

Mass estimates of the main components of a liquefaction system shown in Figure 2 were estimated for a liquefaction rate of 2.2 kg/hr. The mass estimates can be seen in Table 1 below. Estimates for the cryocooler power and mass are based on cryocooler performance and sensible and latent heat required to liquefy oxygen at 1 atm and a gas temperature of 30℃ [13]. The estimated cryocooler lift is 250 watts, with an input power of 1990 watts.

Table 1. Liquefaction System Mass Estimates

Component	Mass, kg
Cryocooler	65
Radiator	63
Precooler Radiator	6
Condenser	2.5
Total	**136.5**

The radiator surface area required was estimated by performing an energy balance on the radiator surface accounting for the various heat transfer mechanisms shown in Equations 1-4. Equation 1 is the thermal balance, Equation 2 is the radiation heat transfer from the radiator to the atmosphere, Equation 3 is the natural convection interactions with the Martian atmosphere, and Equation 4 accounts for the residual solar radiation. The radiation and insolation heat transfer was estimated by assuming the radiator was assumed to be made out of 5 mil silver Teflon with an emissivity and absorptivity of 0.78 and 0.07. The convective heat transfer was estimated by assuming natural convection across a 2 meter tall vertical plate using Mars atmospheric conditions[15]. The sky (167 K), atmospheric (267 K), and radiator temperatures (260 K) as well as the solar insolation heat flux (400 watts/m^2) on the radiator were used for a worst case hot day[9].

$$Q'_{Rad} + Q'_{Convec} + Q'_{Insolation} = Q'_{load} \tag{1}$$

$$Q'_{Rad} = -\sigma\varepsilon\left(T^4_{Radiator} - T^4_{sky}\right) \quad (2)$$

$$Q'_{Convec} = H_{nat.conv} \cdot (T_{Radiator} - T_{atm}) \quad (3)$$

$$Q'_{Insolation} = Q'_{Total\ Insolation} \cdot \alpha \quad (4)$$

The radiator size, heat load, mass and the corresponding modes of heat transfer rates on the surface of the radiator are shown below in Table 2. The radiator surface area was estimated by dividing the cryocooler heat rejection power by the radiator load after accounting for solar insolation and natural convection. The mass of the radiator was estimated using an aerial density of 4 kg/m² [16]. The radiation heat transfer rate was found to be significantly larger than natural convection due to the reduced pressure and gravity.

Table 2. Radiator Heat Rates and Properties

Q'_{rad}	**168 watts/m²**
$Q'_{insolation}$	-28 watts/m²
$Q'_{convection}$	-1 watts/m²
Q'_{load}	139 watts/m²
$Q_{required}$	**2,250 watts**
Area required	**16 m²**
Mass	**65 kg**

III. Liquefaction Location

Three different liquefaction schemes were analyzed and their mass and electrical power requirements were estimated. The baseline liquefaction scheme currently being carried by the MAV team is to liquefy in a separate tank in a batch process and transfer the LOX to the MAV tank once the tank is full. The second option is to liquefy continuously inline upstream of the MAV tank. The third option investigated liquefies directly inside the MAV tank.

The cryocooler lift was calculated for the three options including the tank heat leak and liquefaction cooling load required to liquefy 2.2 kg/hr. A 25% margin was added on top of the combined heat leak to account for thermal uncertainty. The cryocooler lift for the three options can be seen in Table 3. The estimated mass for the three options can be seen in Table 4. The power consumption for the three options can be seen in Table 5. It is assumed for all cases that the nested bulkhead MAV tanks are already full of methane and maintaining zero-boil-off which cools the oxygen tank as well, so no extra power, mass, or time would be used to cool down the oxygen tank.

Table 3 Crycooler Cooling Lift

	Baseline Option A	**Option B**	**Option C**
Cryocooler Lift, watts	410	375	310

Table 4. Mass Estimates For Options 1-3

Component	Baseline Option A	Option B	Option C
Cryocooler	100	104	74
Radiator	110	112	80
Tank	450	-	-
Tank Insulation	19	-	-
Vacuum Jacket	115	-	-
Support Structure	225	-	-
Plumbing & Insulation	27	27	3
Condenser	-	3	-
Pump	12	12	-
Valves	10	10	10
Total, kg	**1,068**	**268**	**167**

Table 5. Liquefaction Power Consumption

Component	**Baseline Option A**	**Option B**	**Option C**
Cryocooler	3,355	3,530	2,500
Valves	100	100	100
Total, watts	**3,625**	**3,630**	**2,600**

A. Baseline Option A: Liquefy in Separate Tank

The baseline option, liquefying in a separate tank, requires that a separate insulated tank as well as an insulated line with pump and valves to transfer the fluid to the MAV tanks. A schematic of the process can be seen in Figure 3.

Some assumptions that were used in the sizing analysis are listed below.

- Liquefaction Tank

   - Diameter: 3 meter
   - Length: 3.75 meter
   - Dome Shape: √2:2
   - Thickness: 4.0 mm
   - Design Pressure: 200 kPa (30 psia)
   - ASME Code Tank
   - Aluminum 6061 material
- Liquefaction Tank Insulation
   - Vacuum Jacketed Tank
   - 40 Layer MLI
   - 15 layer/cm
   - Heat Leak of 82 watts
- Transfer Frequency: one
- Transfer Line Dimension: 1” diameter, 50’ long.
- Pump Flow Requirements:
   - 20 GPM at 12 PSID (based on transfer of 23,000 kg in 4 hours)
   - Pump efficiency of 60%.

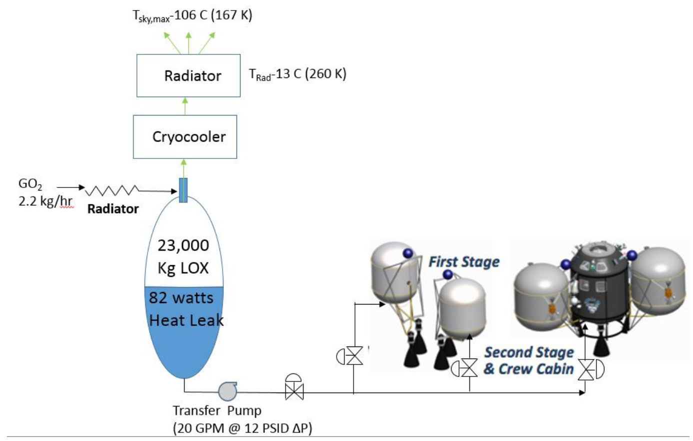

Figure 3. Baseline Option A, Liquefy in Separate Tank

The heat leak into the transfer line was assumed to be negligible considering it will only operate for a period of four hours during the year and a half on the surface of Mars but should be considered for a more detailed analysis. The mass of the tank in the baseline option could be reduced if a transfer occurred every month. Using the same tank material assumptions, the weight of a system where a transfer occurred every month is estimated to be 330 kg.

B. Option B: Liquefy Inline before MAV Tank

A second option investigated is to liquefy inline upstream of the MAV tank (shown in Figure 4). This would include integrating a cryocooler inline to condense the gas before it enters the MAV tank. Because this system would operate continuously the lines and components downstream of the cryocooler would need to be insulated to reduce heat leak. While vacuum jacketed lines could be considered to lower the heat load, rigid lines would be hard to transport and install robotically. Flexible jacketed lines have been shown to have similar heat losses when not in the straight configuration.

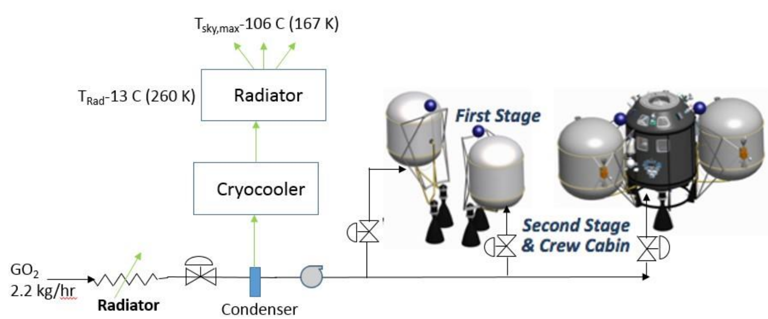

Figure 4. Option B, Liquefy Inline Before MAV Tank

Some additional assumptions in the analysis are:
- Transfer Line Dimension: 1” diameter, 50’ long.
- Transfer Line Insulation: 1” thick Aerogel (approximately 100 watts)

The cryocooler lift would be greater to account for the continuous heat leak entering the lines and components compared to the tank in Option A. However the additional cryocooler and heat exchanger mass is much less than the tank mass that was removed. In total, option B is approximately 800 kg less weight than the baseline option.

C. Option C: Liquefy Directly inside the MAV Tank

A third option is to liquefy directly inside of the MAV tank (shown in Figure 5). This includes relying on condensation and liquefaction to occur inside the MAV LOX tanks. This option is advantageous because it does not require carrying an additional tank and corresponding support hardware. The heat loss and cryocooler load is also minimized because only the MAV tank must be cooled, not any transfer lines or additional tanks.

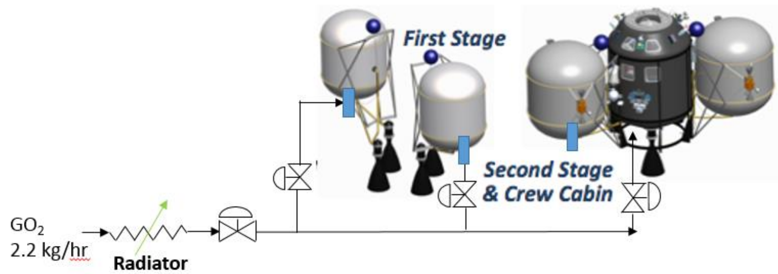

Figure 5. Option C, Liquefy Directly In MAV Tank

A variant of this would use forced convection along the tank wall or within the fluid to enhance the cooling of the fluid. Initial analysis suggest that 0.6 meter of tube in in the bottom of the tank can provide adequate precooling to cut the convection loads by approximately 90%. Further evaluation of this option is ongoing.

IV. Thermal Insulation Trades

A separate trade was conducted to assess the effect of insulation mass and performance on resultant refrigeration system mass and performance requirements on MAV flight tanks. To maintain zero-boil-off within the tanks, insulation mass and performance can be traded for refrigeration system mass and performance. The trades used the thermal models developed for the Mars Ascent Vehicle studies[12,17] using the same ground rules and assumptions for redundancy. No power impacts and availability implications were included in this study, this is a conservative assumption because for all cases the cryocooler power would be reduced. The power assumption is also made more acceptable by the assumption of nuclear power on the Martian surface[12].

The assumptions for the trade study included the warm boundary (environmental) temperature of 267 K, the cold boundary (fluid temperature) of 105 K (corresponding to an oxygen saturation pressure of around 379 kPa (55 psia)). The insulation system was assumed to be composed of two components, a substrate to prevent carbon dioxide solidification on the tank wall and MLI for in-space thermal performance. The thermal conductivity of the MLI system was assumed to vary slightly with mean temperature of the carbon dioxide gas (Mars atmosphere) within the blankets. High performance substrates were modeled that were improvements over the baseline of spray on foam insulation. The thermal conductivity and thickness of the substrate material was then assumed in order to calculate the thermal resistance through the material. The interstitial temperature between the two was then solved so that the heat load through both materials was the same. The interstitial temperature is reported along with the other system parameters in Table 6.

The SOFI was sized to prevent liquefaction and solidification of carbon dioxide from the Martian atmosphere and MLI was then installed on the outside of the SOFI to provide the required thermal performance both in-space during transits and on the Martian surface. Once the insulation system was sized, the energy balance was performed on the tank and the cryocooler size and mass calculated. Once the initial (baseline, number 7 in Table 6) run was completed, the SOFI was replaced with a non-descript insulation systems that had an arbitrary thickness and thermal performance to create the thermal resistance variance desired (as shown in Figure 6). The energy balance and refrigeration system sizing was then recalculated and the change in refrigeration system mass from the baseline was found. This change was used to determine what the allowable non-descript insulation system mass (and areal density) was to maintain a constant system mass. The results are plotted in Figure 6. While this curve is somewhat system dependent, it does give a general baseline for break-even points and design targets for new and novel insulation systems. The points in the curve are shown numerically in Table 7. The MLI was maintained at the design point for the in-space cruise portion of the mission (approximately 0.054 m or 2 inches thick), the minimal mass changes of the MLI due to changing substrate thickness are ignored as they would be on the order of a few kilograms.

Table 6. Insulation System Trade Cases (Number 7 is the baseline)

Case	CBT	k -subst.	x - subst.	Ti	x MLI	WBT	R	Q	Qtot
	K	mW/m-K	m	K	m	K	m^2K/mW	W/m^2	W
1	105	1.0	0.033	250	0.05334	267	3.30E-02	4.39	645
2	105	1.0	0.081	260	0.05334	267	8.13E-02	1.91	273
3	105	0.2	0.008	250	0.05334	267	3.81E-02	3.81	645
4	105	0.2	0.018	260	0.05334	267	8.89E-02	1.74	273
5	105	0.2	0.033	263.5	0.05334	267	1.66E-01	0.95	142.9
6	105	5.0	0.033	203	0.05334	267	6.60E-03	14.84	2225.9
7	105	16.4	0.081	190	0.05334	267	4.96E-03	17.15	2453

The cryocooler was then sized to maintain zero-boil-off of the oxygen tank as it is filled with liquid oxygen by the liquefaction system. The cryocooler was assumed to be a reverse turbo-brayton cryocooler and was sized based off of correlations developed in cooperation with Creare[14].

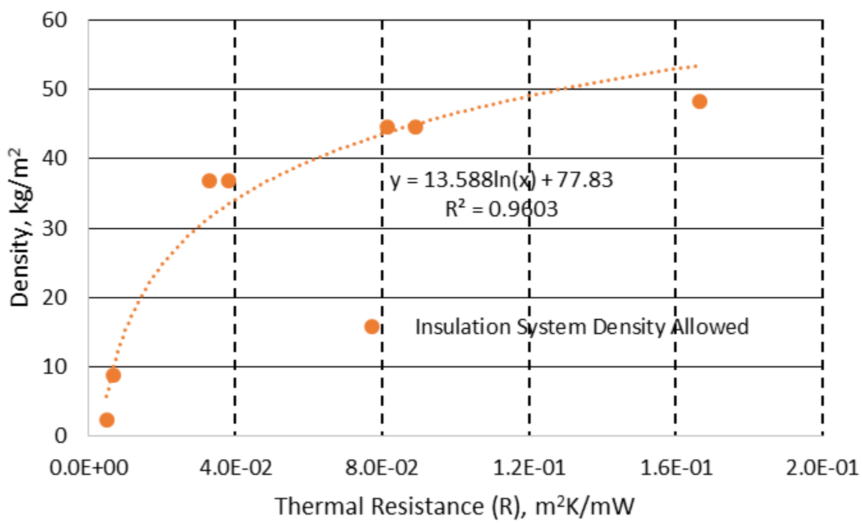

Figure 6. Areal Density Allowances for Improved Insulation Systems

Table 7. Mass Allowances for Improved Insulation Systems

Case	Thermal Resistance m^2K/mW	Maximum Insulation mass allowed kg	Maximum Insulation Areal Density kg/m2
1	3.30E-02	1058	36.91
2	8.13E-02	1276	44.52
3	3.81E-02	1058	36.91
4	8.89E-02	1278	44.52
5	1.66E-01	1384	48.29
6	6.60E-03	254	8.86
7	4.96E-03	68	2.37

A. Vacuum Jacket Analysis

The thermal performance of options 3-5 are representative of what could be achieved with vacuum jacketed systems maintaining a space like vacuum. Further analysis was done to understand mass requirements for vacuum jacketed systems. Vacuum jacketed concepts that require a vacuum jacket to support 101 kPa (15 psid) (i.e. hold a vacuum on the surface of the Earth), came in at several metric tons total. This is much more than the extra mass required for the cryocooler in the soft vacuum based insulations using foam (baseline, option 7). However, using a pressure difference of 689 Pa (0.1 psid) (only holding vacuum on the surface of Mars) could be done for around 500 – 700 kg total for all four tanks (shell thicknesses of around 0.075 inches of Aluminum 6061). This falls within the acceptable mass range and assuming it meets thermal targets could be used. However, the added complexity of a purged system at launch, evacuation during transit, and then somehow isolating the tank from the atmosphere of Mars needs further investigation. Additional work will have to be done on added failure modes and how they may impact mission risk.

V. Conclusions

A total of three different concepts of operation for liquefaction of in-situ produced propellants were studied, liquefaction in a separate tank (the current baseline case), liquefaction in the transfer line to the vehicle from the oxygen production plant, and liquefaction within the ascent vehicle propellant tanks. Option B with inline cooling reduces the mass of the baseline liquefaction system by 75%. Option C, liquefying inside the MAV tank reduces the total mass by 85% over the baseline, saving approximately 900 kg. However, the mass of option A could be reduced by more than 700 kg (by shrinking the storage tank) if the transfer frequency was increased to once a month (the baseline assumption was a single transfer right before launch).

The two key technologies needed for in-situ production of liquid oxygen are shown to be insulation systems for soft vacuum (the surface of Mars is approximately 5 – 7 Torr (0.67 kPa)) and high heat rejection, high efficiency cryocoolers at approximately 85 K. Insulation systems for space flight have long been assumed to be multilayer insulation, however, the performance of MLI is severely degraded by the presence of any gas within it. As such, the soft vacuum pressure encountered on the surface of Mars decreases the thermal performance of the MLI by over an order of magnitude. While vacuum jacketed tanks drastically improve the system thermal performance, they come with added mass and complexity. Additionally, similarly to on Earth, the temperature of the cryogens is below the boiling point of the atmosphere (in this case CO2), so care needs to be taken to prevent liquefaction and solidification of the atmosphere onto the propellant tank, both from a heat load and insulation performance perspective. The current state of the art cryocoolers for space-flight are on the order of 20 W of refrigeration power at 90 K. In order to meet the requirements of this mission, the lift will need to be increased at least an order of magnitude if not further.

Acknowledgments

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References

1. Lockheed Missiles & Space Company, “High Performance Thermal Protection Systems: Final Report”. LMSC-A96947, Dec 31, 1969.
2. Hofstetter, W.K., Guest, A., McLinko, R., and Crawley, E.F., “Analysis of Launch and Earth Departure Architectures for Near-Term Human Mars Missions”, AIAA 2008-7777, 2008.
3. Linne, D.L., Gaier, J.R., et.al., Demonstration of Critical Systems for Propellant Production on Mars for Science and Exploration Missions”, AIAA 2013,0587, 2013.
4. Plachta, D.W., Johnson, W.L., and Feller, J.R., “Zero Boil-off System Testing”, Cryogenics, doi:10.1016/j.cryogenics.2015.10.009, Accepted for Publication, October 15, 2015.
5. Plachta, D.W., Guzik, M.C., “Cryogenic Boil-Off Reduction System”, Cryogenics, Volume 60, March–April 2014, Pages 62-67
6. Linne, D.L., Barsi, S.J., et. al.”Mars Surface Mobility Leading to Sustanable Exploration” AIAA 2012-0806, 2012.
7. Sanders, G. B., Paz, A, et. al. “ Mars ISRU for Production of Mission Critical Consumables -Options, Recent Studies, and Current State of the Art”, AIAA 2015-4458,2015

8. Meyer, M.L., Chato, D.J., et. al., “Mastering Cryogenic Propellants”, J. Aerosp. Eng., vol 26, pg 334-351, 2013.
9. Novak K., Phillips, C. et. al. “ Development of a Thermal Control Architecture for the Mars Exploration Rovers”, presented at the Space Technology Applications International Forum 2003, Institute for Space and Nuclear Power StudiesAlbuquerque, NM 87131, 2003
10. Augustynowicz, S.D., and Fesmire, J.E., Cryogenic Insulation System for Soft Vacuum”, Advances in Cryogenic Engineering, Vol. 45, Plenum Publishers, NY, pg. 1691 – 1698.
11. White, S., Begag, R., et.al. “Multilayer Aerogel Insulation for Cryogenic Applications”, presented at the 2015 Cryogenic Engineering Conference, Tucson, AZ.
12. Polsgrove, T., Thomas, Dan, et. al., “Mars Ascent Vehicle Design for Human Exploration”, AIAA 2015-4416,2015.
13. Ladner, D. R, “Performance and Mass vs Operating Temperature for Pulse Tube and Sterling Cryocoolers” N-Science Corporation, Golden, CO 80401
14. Guzik, M.C., Plachta, D.W., and Feller, J.R., “Reduced Boil-off System Sizing”, presented at the 2015 Space Cryogenics Workshop, Phoenix, AZ, 2015.
15. Incropera, F., and Dewitt, D. P., Introduction to Heat Transfer, 4th ed, John Wiley and Sons, New York, 2002, pp 507
16. Juhasz, A, “High Conductivity Carbon-Carbon Heat Pipes for Light Weight Space Power System Radiators” NASA TM 2008-215420, 2008.
17. Johnson, W.L, Sutherlin, S.G., and Tucker, S.P., “Cryogenic Propellant Insulation System Design Tools for Mass Optimization of Space Vehicles”. AIAA 2008-7765, 2008.