MASSACHUSETTS INSTITUTE OF TECHNOLOGY

2025 HUMAN LANDER CHALLENGE

THERMOS

Translunar **HE**at **R**ejection and **M**ixing for **O**rbital **S**ustainability

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Student co-leads:
Beverly MA[1]
Daniel ROJAS[1]
Team members:
Jonathan BEAS[1]
Nicole DING[1]
Stone SMITH[1]
Pranav BALA[1]

Lead Faculty Advisor:
George LORDOS[2]

Co-advisors:
Olivier DE WECK[3]
Jeffrey HOFFMAN[4]

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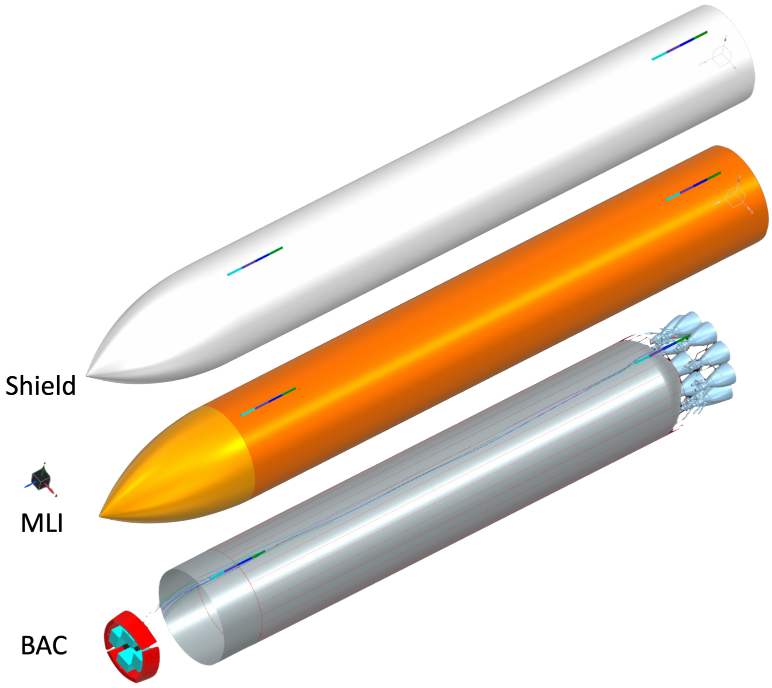

[1] Undergraduate Student, Aeronautics and Astronautics
[2] Research Scientist, Aeronautics and Astronautics & System Design and Management
[3] Apollo Program Professor of Astronautics and Professor of Engineering Systems
[4] Professor of the Practice of Aerospace Engineering

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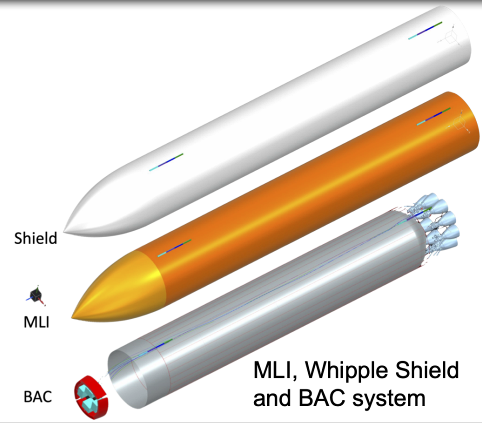
Massachusetts Institute of Technology

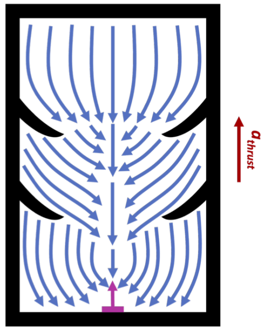
Translunar Heat Rejection and Mixing for Orbital Sustainability (THERMOS)

Theme Category, Major Objectives & Technical Approach

- **Theme Categories:** (1) Large Surface Area Radiative Insulation and (2) Propellant Mixing Devices
- **Major Objective:** A true systems solution for a healthy Lunar Ascent Propellant Margin of Starship HLS
- **Technical concept:** intercept and reject heat loads, and also mix the propellant for thermal homogeneity (HLS & Depot)
- **Systems approach:** evaluate & stress- test conceptual designs for alternative Boil-off Control Systems, for their impact on Ascent Propellant Margin, number of tanker launches required, *and also* for resilience to unexpectedly high MLI degradation



MLI, Whipple Shield and BAC system



Modified Baffle Geometry + Pump = Propellant Mixing Device

Key Design Details & Innovations of the Concept

- **Model-based engineering:** simultaneous decisions for mission ConOps, technology infusion and system sizing, for CFM technologies which could reduce boil-off *but* also add mass
- **Technical innovation 1**: **modified baffle geometry** plus recirculation pump, leveraging regular flight operations (i.e. spin-stabilizing, settling thrusts) to mix & destratify propellants.
- **Technical innovation 2:** a **combined MLI / Whipple Shield system mounted on tank-welded BAC cryocooler tubes**, saving structural mass and intercepting heat leaks at source
- **A key finding:** contingency operational modes compensate for high MLI Degradation Factor (DF) up to **DF of 9X** if V3 Starship + Cryocoolers are used, vs. just 5X if passive-only V2 Starship

Summary of Schedule & Costs: Design, Build, Fly, Fix & Fly Again!

2025 *H1:* Present at HuLC; *H2:* Detailed Design and Project Planning

2026 *H1:* Build tanks with new MLI, Cryocoolers & Baffles;
H2: Flight test, aboard one of the regular Starship tests

2027 *H1*: Incorporate flight learnings; uncrewed HLS flight to LEO / HEO;
H2: Uncrewed HLS to HEO, NRHO & LSP, uncrewed Depot to LEO

2028 *H1*: Incorporate changes to HLS and Depot;
H2: repeat all uncrewed flight tests

2029 *Q1*: Conclude human-rating certification of THERMOS;
Q2-Q4: THERMOS available for use in crewed Artemis III mission

Lifecycle development cost (internal to SpaceX): from **$200M - $500M**
Annual Operating cost during Artemis III: from **$40M - $100M**

1 Executive Summary

THERMOS is an innovative, systems-level solution to the problem of significant boiloff during long-term storage of cryogenic liquids in the space environment, carefully designed for implementation on Starship HLS and Depot vehicles within 3-5 years, in time for Artemis III. Our scope is limited to long-term mitigation or elimination of cryogenic propellant boil-off in LEO, HEO, NRHO, and on the surface at the lunar South Pole, as well as some aspects of propellant mixing and destratification, but only to the extent that these are required for the purpose of safe and efficient operation of our boil-off control system.

The long-term storage challenge implies **selecting** and **optimizing** technologies to reduce propellant boil-off, traded against the added dry mass, complexity, and various forms of risk, including schedule and programmatic risk. However, technology selection can change entirely depending on when, where, and how the propellant mass is conserved by the proposed technologies. Therefore, we used the model-based systems engineering paradigm to evaluate a wide range of passive and active thermal control systems in terms of their predicted impact to two key mission-level figures of merit: the Lunar Ascent Propellant Margin (LAPM), and the number of tanker flights required for the mission.

Using this approach, we evaluated 26 alternative system architectures and down-selected to 3, two involving a combination of MLI integrated with BAC (one on V3 and one on V2 Starship HLS), and one with just MLI on a V2 Starship HLS. All three also include a novel propellant mixing system, and a novel solar array backpanel radiator design. We then stress-tested these three configurations for MLI Degradation Factors (DF) from 4X to 12X, and for failure of up to all 7 cryocoolers, and found that the V3 Starship with MLI, cryocoolers and propellant mixing system (architecture 023-MCB) can withstand MLI DF up to 9X, and the failure of up to all but one cryocoolers, and still complete the Artemis III mission with a LAPM of 10%. We then proceeded to conceptual and detailed design of the proposed storage and mixing technologies, and also considered system development, testing and integration.

2 Mission Analysis and Design

We started by understanding the needs of our key stakeholder. We found that the primary stakeholder is NASA, and their primary need is for the Artemis III mission to bring the crew back to Earth safely, with the mission taking place within the well-publicized target time frames. This understanding drove our choices for the metrics we would use to evaluate alternative system designs, and also led us to first fully analyze and quantitatively simulate the HLS Concept of Operations of the Artemis III mission, including all burns and orbital propellant transfers.

The first result from this analysis was our decision to adopt two top-level metrics aligned with NASA's needs for Artemis III. For a **mission benefit metric** best aligned with the impact of reducing boil-off rates on crew safety, we selected the fraction Lunar Ascent Propellant Margin (LAPM), defined as the propellant remaining in Starship HLS tanks returning to Gateway / Orion, divided by the propellant in the tanks just before lunar ascent. For a **proxy for mission cost metric**, we selected the number of tanker flights that will be required to refuel the Starship HLS, so that it can complete the Artemis III mission with the level of LAPM already calculated for that simulated mission.

Next, we developed a mission architecture model simulating the following events for Starship HLS, Depot and Tanker: all orbits and locations used by the Artemis III HLS ConOps, all propulsive burns, all loitering events and their duration in seconds, and all propellant transfer events. Submodels were added to simulate boil-off as a function of environmental and internal heat fluxes, the mitigation of boil-off as a function of the technologies selected for infusion, the calculation of dry mass as a function of the technologies selected, and various constraints including propellant load limits, which depend on the version of Starship used (Appendix G), as well as power limits for technologies that require power.

Using this model, we generated 26 system designs, analyzed them within our parametric mission context, and scored each system according to its impacts of benefit and cost *to the overall* *mission*, using the metrics LAPM and the number of tanker launches required.

From our analysis, we learned three key things that would influence our design decisions. The first key finding was that **the last refueling of Starship HLS using a Depot should take place not in LEO, but at a highly elliptical orbit (HEO)**, of the order of TLI - 1600 to TLI - 1000 m/s. At the cost of a few additional tanker launches, this would generate viable alternatives to use *either* Starship V2 (1500 tons propellant) *or* V3 (2300 tons) for the HLS program. The former would have more mission risk due to lower LAPM than the larger V3, but the latter would carry significantly more programmatic risk, as it is still a paper rocket. So, we felt it was essential for NASA to have the option to apply our long-term storage technology in either V2 or V3. The second key finding was that **NRHO (90 days stay) would be a very benign thermal environment relative to the Lunar South Pole (LSP) (30 days stay)**, mainly because of the solar insolation angles. This would inform decisions for different forms of passive insulation at the front and side of the vehicle. The third key finding was that **HLS could produce twice as much electrical power in NRHO than at LSP**, so we assumed that the excess power at NRHO could be used by an active cryogenic storage system. This finding reduced the burden of including an active system to augment the passive system.

3 Conceptual Design

3.1 Concept Generation and Selection

The preceding mission analysis, together with a technology search and model-building, resulted in the generation of 26 alternative mission and system concepts summarized in the tradespace plot shown in Figure 1:

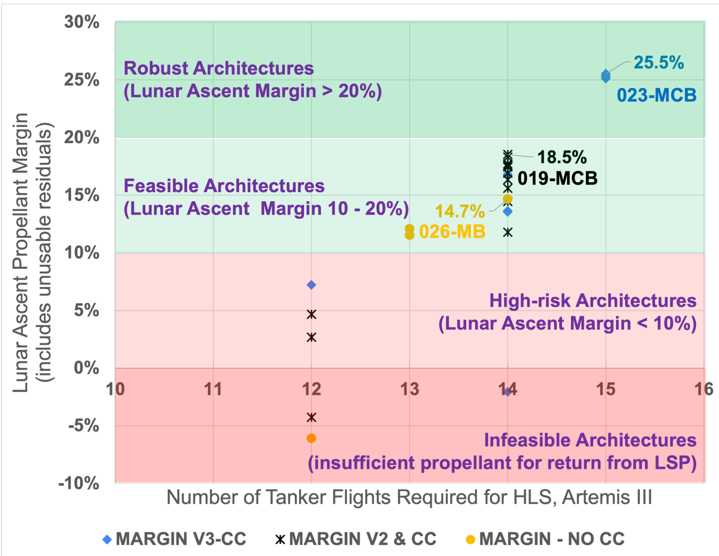


Figure 1: Tradespace of 26 alternative system concepts, trading LAPM for number of tanker launches. Three high-performing concepts with different degrees of mission and programmatic risk, named 023-MCB, 019-MCB and 026-MB were down-selected for further design development.

The best-performing concept overall, delivering LAPM of up to 25.5% and requiring 15 tanker flights, was 023-MCB, which included 60-layer MLI on the sides, 40-layer MLI in the front and seven power-limited cryocoolers. This system also featured a novel sawtooth-shaped radiator on the backside of the solar arrays with lower emissivity on the faces looking at the tanks, and deployable lunar vertical solar arrays to enable running the cryocoolers on the Moon. The next best performing concept was 019-MCB at LAPM = 18.5% and 14 tanker flights, also a hybrid with the same technologies and approach as above, but using a smaller Starship (V2) HLS, which explains the large gap in LAPM performance. In order to include an option with the least programmatic / schedule risk, we also added to the down-selection list the best-performing Starship V2 HLS concept that does not use active cooling: 026-MB with LAPM of 14.7% and 14 tanker flights. All three concepts also included a propellant mixing device developed by modifying existing baffles, as described in subsection 3.3.

3.2 Large Surface Area Hybrid Radiative Insulation Conceptual Design

Based on our findings above, we were interested to explore both passive and hybrid systems, i.e. different kinds of coatings and Multi-Layered Insulation (MLI), as well as systems where the MLI system would be integrated with Broad Area Cooling (BAC), listed in Appendix J. The initial reference design for our cryocoolers was a flight-like 90K, 150 W_{th} concept by Northrop Grumman with specific power of 15 We / Wth and specific mass of 0.8kg / W_{th} [6]. In the second iteration, we switched to a better-performing Creare model, as described below. Cryocooler sizing at the conceptual stage was carried out using simple scaling laws[4]. The cooling tubes are spaced 1m apart and alternate between the seven manifolds, providing surge capacity to remove high spot heat loads. The tubes continue for 7m past the top of the methane tank, so as to capture any heat leaks conducted through the steel wall from the crew compartment above. Insulation for the dome of the methane tank facing the crew compartment remains undeveloped and ought to be revisited during future design development, in particular exploring the use of internal MLI and alternative locations for manifolds and tubing.

In our conceptual design, the MLI-60 on the sides of the Starship (V3) HLS tanks has alternating Dacron nets and foils for an effective emissivity of 0.016, using a conservative MLI Degradation Factor of 4[5]. The blanket segments are structurally mounted on tubes carrying neon gas for the 90K Broad Area Cooling (BAC) system as shown in Figure 3, leveraging the structural and thermal synergies that made the concept attractive. The blanket is also perforated to allow air to escape after launch. In addition, lightweight non-conductive mounts are fixed in between the tubes to create a 10cm grid pattern for secure mounting. An outer aeroshell shield is bonded to the outermost Kevlar layer, and there is also an aluminum layer to address charging[5].

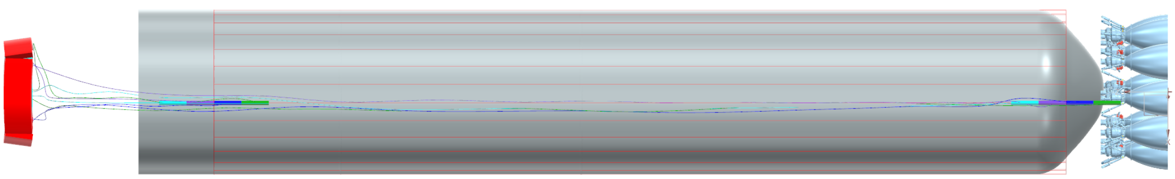


Figure 2: The integrated MLI, BAC and Whipple Shield concept: 28 cooling tubes flat-welded onto the tanks, with 4 connected to each manifold, each of which is served by one of seven 90K cryocoolers.

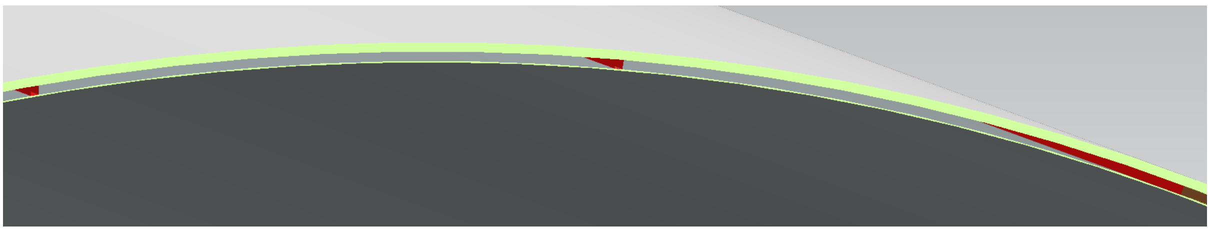


Figure 3: Cross-section of tank wall, cooling tubes, MLI and shield. The 60-layer MLI is mounted at the top of the cooling tubes at a distance of 14mm from the tanks, and is enclosed within Kevlar layers and an external shield. Heat leaks are guided to the cooling tubes where they are intercepted, and the overall arrangement also serves as a double Whipple Shield, protecting the spacecraft from small, high-velocity MMOD impacts.

3.3 Propellant Mixing System Conceptual Design

In case of contingency (e.g. failure of the cryocoolers) we have also designed a propellant management system for HLS, called Equalization of Temperature stratification through Hydrodynamic Exchange and Redistribution (ETHER), which combines innovatively designed tank baffles and axial propellant pumps to reduce temperature stratification in the propellant. The motivation behind this is to take advantage of the spin-stabilization on HLS to create centrifugal forces on the propellant. The existing baffles will be extended and twisted into hydrodynamic spiral-shapes, with a row of baffles at the top and bottom of each tank. These baffles would be optimized to direct propellant to the center of the tank forcing mixing to occur, taking inspiration from boat propellers. Both the depot and the HLS will also incorporate small axial pumps to further ensure mixing and add redundancy. By having passive geometric structures working in tandem with a small axial pump, temperature stratification is mitigated with a relatively low mass penalty. Any boil-off itself is used to provide RCS thrust and the energy to run the thermodynamic vent system.

The conceptual design phase was followed by a more in-depth design phase of the baffles to study their hydrodynamic performance. Initial computational fluid dynamics (CFD) simulations modeled the propeller baffles effectiveness in performing their slosh mitigation function. Remaining future work includes developing the specifications and requirements for the axial pump to reduce temperature stratification. Future detailed design development should strive to find the optimal combination of hydrodynamic baffles and axial pump while also mitigating their added mass to the rocket.

3.4 Design under Uncertainty: Stress-testing the three concepts

Having down-selected to the three concepts above (23-MCB, 19-MCB and 23-MB) and assessed their predicted performance under ideal, nominal conditions, the next step started by recalling that our primary stakeholder, NASA, needs an HLS architecture that will bring the crew back home to safety and which will be ready in time for Artemis III. We interpreted this as a requirement that our HLS should still be able to complete its mission with acceptable LAPM > 10%, *even if the MLI system experiences a significantly higher degradation beyond the conservative DF = 4 already baselined.* So, we decided to simulate the performance of each of the three concepts under uncertainty, by varying DF from 4 to 12, and varying the number of failed cryocoolers from 0 to 7. The resulting scenarios which passed the required LAPM > 10% threshold (or were close to 10%) are shown in Figure 4.

From Figure 4, we observe the following: the LAPM for the Starship V3 hybrid concept 023-MCB is only slightly reduced from 25.5% to about 22.5% if all cryocoolers fail but DF remains at 4. At a DF of 9, 023-MCB has a still-decent LAPM of 14.3% if no cryocoolers fail, and a DF of 9.7% if all but one of the cryocoolers fail. At a DF of 8, 023-MCB exceeds the 10% threshold even if all seven cryocoolers fail. By contrast, the passive Starship V2-based concept 026-MB starts at 14.7% when DF=4 and falls to LAPM of 9.9% with a DF=5. This means that the programmatically more attractive, passive, V2 concept 026-MB is significantly more risky from a mission / technical perspective when compared to 023-MCB. The latter can tolerate an MLI DF of 8 or 9 and still complete its mission. Therefore, to preserve the option of retiring all risks and to complete the design, development and certification of the integrated MLI / BAC / Whipple shield in time for Artemis III, at the end of the stress-testing exercise we made the following design and project planning decisions: NASA and SpaceX could aim for a hybrid passive-active Starship V3-based concept such as 023-MCB, which is capable of sustaining its design performance under significant uncertainty; and they should select high-TRL subsystems and a project plan built around rapid fly, fix, fly again iterations.

3.5 Summary of Down-selected Concepts

In the process of modeling alternative mission modes, we noted that the final refilling for HLS should not take place in LEO, but rather at a highly elliptical orbit approximately 1 kilometer per second of delta V away from a translunar injection. While this decision would increase the number of tankers needed, it would also relax constraints on viable designs. With the last refilling at HEO, it is feasible to configure a HLS that could complete the mission using only passive cooling technologies. All three down-selected concepts have acceptable LAPM's above 10% at our nominal

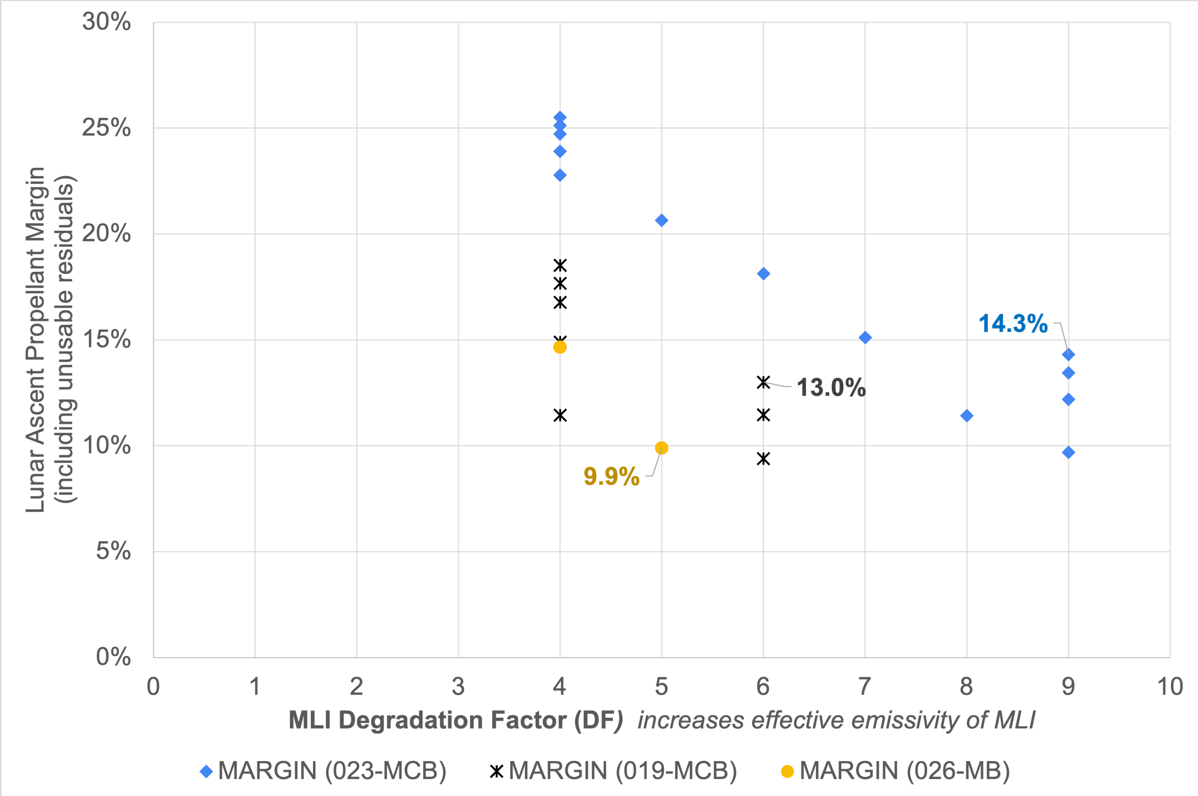


Figure 4: Stress Test Results: the three down-selected concepts were stress-tested for two key technical risks, the possibility of a MLI Degradation Factor (DF) higher than 4, and the possibility of one or more cryocoolers failing. The resulting analysis showed that the V3 hybrid architecture 023-MCB can tolerate a DF of up to 9 if one cryocooler is running or a DF of 8 if all cryocoolers fail, vs. a DF of only up to 5 for the V2 passive architecture 026-MB.

4x degradation factor. Our passive V2 architecture, 026-MB fails to meet the LAPM requirement if MLI degradation factor exceeds 5. The hybrid V2, 019-MCB, can better withstand MLI degradation due to its cryocoolers but if those fail, the hybrid V2 ends up being less resilient to degradation. We therefore selected the hybrid V3 concept 023-MCB, with a nominal 25.5% LAPM requiring 15 tanker refills, that can withstand both complete cryocooler failure and double the expected MLI degradation and still meet our 10% LAPM target. The ConOps for 023-MCB is shown in App. K.

4 Detailed Design

4.1 Descriptive Goals and Constraints

Table 1 summarizes the major descriptive goals and constraints (i.e., the "Level 1 requirements") driving the design of the hybrid insulation and propellant mixing systems. They were derived from our analysis of stakeholder and mission needs in Section 2, as well as vehicle limitations inherent in the Starship HLS architecture and program.

The most important design goals listed in Table 1 relate to crew safety and schedule readiness for Artemis III; these include targeting a robust Zero-Boil-Off (ZBO) capability for HLS at NRHO, maintaining propellant quality for engine use, and aiming for compatibility and ease of integration of our CFM system with the Starship HLS that will be used for Artemis III.

4.2 Materials and Manufacturing Considerations

All selected materials have been chosen to meet the performance needs of our conceptual design described in Section 3.5 and Table 1, including adhering to the constraints of the Starship HLS vehicle (which extensively uses stainless steel) and the space environment. The material selections are discussed in this section, and Table 3 in Appendix B summarizes all primary materials for each subsystem component together with the selection rationale.

Table 1: Key Design Requirements and Constraints

Requirement or Constraint	Description / Value
Max boil-off at NRHO for 90 days	$\leq$ 0.1% of total propellant (targeting ZBO in NRHO)
Lunar Ascent Propellant Margin (LAPM)	At least 10% of propellant at time of ascent from LSP shall remain upon arrival at NRHO, including any unusable residuals
Propellant thermal stratification	$\Delta T < 2$ K across tank (ensure near-isothermal propellant)
Robustness	System must tolerate severe failures (e.g. multiple cryocooler outages, severe MLI degradation) and still meet > 10% LAPM
Power availability (at NRHO)	Up to 50kW available for active cooling (surplus solar power in orbit where all arrays see the Sun)
Power availability (at LSP)	All on-board power is assumed to be spoken for by ECLSS; cryocooler power & cooling loop during surface stay to be provided by external sources / sinks
Materials compatibility	All materials must be space-rated and compatible with Starship (thermal, structural, and integration constraints)
Launch/landing loads	Systems must survive vibration, acceleration, and depressurization during Earth launch (robust mounting required)
Micro-meteoroid & Charging protection	Thermal insulation system should also support MMOD and Charging protections (dual-purpose shielding)

Our proposed new **stainless steel structures** (cooling loops, baffles) use the same 304L stainless steel and are welded directly on the HLS tanks. This steel alloy has good performance in cryogenic applications. Using the same steel for the external cooling tubes and internal baffles ensures consistent thermal contraction behavior with the tank and reliable weld joints.

For the **Multi-Layer Insulation** system we use alternating layers of aluminized polymer film and spacer meshes to achieve a very low effective emissivity. The blanket also contains specialized structural layers to survive launch and to function as an effective MMOD shield. Since the HLS and Depot are not intended to ever re-enter the Earth's atmosphere, the externally-mounted MLI does not need to survive Earth re-entry. Thermal modeling described in Section 4.3.1 indicated that a total of 40 to 60 layers would be required. Aluminized Kapton provides the reflective layers, with Dacron (polyester) netting as the main separator between layers to minimize solid conduction. The MLI blanket is subdivided into segments for ease of installation and to accommodate tank geometry, with custom segments for required penetrations. Each blanket segment is structurally supported primarily by the network of tank-welded steel cooling tubes (acting as strong attachment rails and aligned with the primary aerodynamic load force vector) and by additional G-10/PEEK/Aerogel stand-offs as needed between tubes.

Finally, for **pump and baffle materials**, the key point is that the mixing pumps and modified baffles reside inside the cryogenic tanks and thus face liquid methane or oxygen at very low temperatures. Stainless steel (304L) is used for baffles and any structural support for the pump, ensuring mechanical integrity and compatibility with the tank. Aluminum pump components can be used in the methane tank to reduce dry mass, but the pump in the oxygen tank may have to be made from heavier 304L steel materials. The pump impellers and internal hardware are stainless or titanium to handle rotation in cryogenic fluid. Bearings and seals in the pump use materials like PTFE (Teflon) coatings and ceramics that can operate with minimal lubrication at low temperatures. Manufacturing of the baffles involves cutting and forming stainless steel plates and welding them into a spiral shape onto existing baffle rings. All welds and assemblies are designed to be inspectable and will be cryo-tested for leaks and strength. The pump units are designed to be compact and bolt into tank access ports, making use of existing tank interface points: for example, a top bulkhead fitting for the pump motor wiring and a bottom sump inlet.

Each material choice has been vetted against NASA's standards for spaceflight (strength, flammability, outgassing, etc.) and against the Starship's known constraints. By using mostly flight-proven materials (steel, aluminum, Kapton, Kevlar), the design minimizes introduction of new material risks and focuses on innovation in geometry and system architecture.

4.3 Engineering Analysis

To validate the feasibility of the proposed design, engineering analyses were conducted in both thermal and fluid domains. These calculations demonstrate that the system can meet its performance targets (near-zero boil-off and reduced stratification) within realistic mass and power limits.

4.3.1 Thermal and Mass Analysis of Passive Insulation

The dominant heat load on the cryogenic tanks in space is radiative transfer from the warm environment (solar radiation, planetary IR and albedo, and soakback from the engines) to the cryogenic propellant tanks. In a worst-case scenario without MLI, a bare stainless steel tank with emissivity of approximately 0.8 would absorb tens of kW of radiative heat for the full tank surface area. With the MLI in place, the effective emissivity can be reduced to approximately 0.01 or less, cutting the radiative heat load by about two orders of magnitude.

A general theoretical model for MLI design to mitigate heat flux has terms for radiative, conductive and convective heat transfer and is driven by material choices, design decisions and environmental factors. Starting from the Lockheed Equation 1, W. B. Johnson in his 2010 thesis [3] rearranged it to search the design space for optimal layer thickness and, in the process, tested various coupons vs. model predictions to conclude that conduction through the MLI is a larger factor than predicted by previous models [3]. We referenced Tables 4, 5 and 6 and Figure 20 from Johnson 2010 [3] to populate our model lookup tables with a few MLI options with realistic mass per unit area and apparent thermal conductivity (k-values), as shown in Appendix H. In our final configuration, based on the best k-value per areal density figure of merit, we selected the A144 Ball MLI 60, Coupon B. This was 60 layers in 15 sub-blankets, 2.6 mm per layer, 0.9 inch total thickness excluding the aeroshell, Kevlar and standoffs. With a warm boundary temperature of 305K and a cold boundary temperature of 78K, the heat leak was 0.369 W/m^2 and the k-value was 0.037 $mW/m-K$.

$$q = \frac{C_s \, \bar{N}^{2.63} \, (T_h - T_c) \, (T_h + T_c)}{2 \, (N+1)} + \frac{C_r \, \varepsilon \, (T_h^{4.67} - T_c^{4.67})}{N} + \frac{C_g \, P \, (T_h^{0.52} - T_c^{0.52})}{N} \tag{1}$$

Where,

q = Heat flux ($\mathrm{W\,m^{-2}}$)

C_s = Coefficient of solid conduction ($\mathrm{W\,m^{-2}\,K^{-2}}$)

C_r = Coefficient of radiation heat transfer ($\mathrm{W\,m^{-2}\,K^{-4.67}}$)

C_g = Coefficient of gas conduction ($\mathrm{W\,m^{-2}\,Pa^{-1}\,K^{-0.52}}$)

T_h = Warm (hot) boundary temperature (K)

T_c = Cold boundary temperature (K)

$\bar{N}$ = Layer density (layers per unit thickness (-)

N = Number of layers (-)

ε = Emissivity of the radiation shields (–)

P = Environmental pressure (Torr), valid for $P < 10^{-4}$ Torr (free molecular flow regime)

Conduction heat leaks, expressed in Eq. 1 by the first C_s term, are addressed in the THERMOS design by carefully exploiting a synergy with the active cooling system. The active cooling tubes are made of steel and welded onto the tank exterior walls for broad area cooling, to structurally support the MLI blankets during launch, and to "pull in" and intercept conductive heat leaks which reach the cold boundary of the MLI blanket. By preferentially guiding heat conduction pathways into the cooled steel tubes, the design efficiently intercepts heat leaks and helps prevent the formation of “hot spots” on the tank. All other supports, mounts and penetrations are made of low-conductivity materials. The last layer at the cold boundary has added thickness for increased thermal conductivity to give rise to lateral thermal gradients towards the cooling tubes and reduce the flow through the low-conductivity supports. Modeling, quantifying, optimizing and experimentally validating the performance of this hybrid design is future work, as per our proposed technology development program described in Section 6.

4.3.2 Lift, Mass and Power Analysis of Active Cooling

To maintain near-zero boil-off (NZBO), the seven Creare cryocoolers must remove as much of the 1,400 W residual heat leak (calculated for NRHO) as possible. Each Creare 150 W @ 90 K Reverse Turbo-Brayton cryocooler provides up to 150 W of lift, for a combined capacity of 1050 W leaving a 350W deficit of heat leak which will result in very slow boil-off in NRHO rather than true ZBO. The electrical power required would be approximately 8.4 kW and the cryocooler system mass, including the seven cryocoolers, the network of broad area cooling tubes and seven manifolds was estimated at approximately 3.4 metric tons (see dry mass calculations in Appendix E). This 8.4 kW power draw is well within the NRHO power and likely heat rejection budgets and, on the lunar surface, would be met by deployable vertical solar arrays, such as the LAMPS or LVSAT concepts under study for NASA.

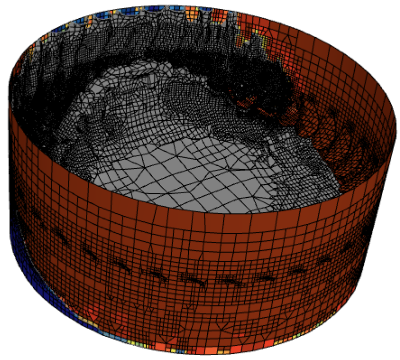
The broad-area cooling (BAC) loop uses neon gas circulating through 28 cooling tubes on the tank walls. Neon remains gaseous at 90 K (boiling point 27 K at 1 atm), avoiding phase-change complications and providing ample thermal capacity. Seven manifolds (one per cryocooler) provide redundancy and segmentation: if one cooler is offline, the remaining six share its load via the interconnected network, preventing any single zone from warming excessively. The tube network also adds thermal inertia (from neon and metal), buffering short heat spikes. Tubes are spaced at 1 m intervals as shown in Figure 2 so that any localized heat leak falls within 0.5 m of a cooled tube—well within the wall’s conduction length. This layout handles both uniform background loads and transients (e.g. sun patches or thruster firings). Any heat conducted through the tank structure from the crew cabin and life-support systems (atop the methane tank) is intercepted by extending the BAC tubing 7 m above the dome, capturing conducted heat from the crew-module interface.

Creare’s mechanical engineering-model hardware has been performance-verified at NASA GRC, achieving TRL 5, while its control-electronics package is at TRL 4. Ongoing vibration, thermal-vacuum, and integrated heat-load tests aim to elevate both to TRL 6, with a suborbital flight demonstration planned under CFMPP in 2025 [1].

4.3.3 Fluid Mixing and Stratification Analysis

To analyze the propellant thermal stratification, we consider the scenario of extended coast or loiter where external cooling is reduced (e.g., cryocoolers off for a period). In microgravity, without active mixing, a thermal gradient can establish: warmer fluid tends to rise to the top of the tank (though in microgravity “rising” is driven by any slight density differences and constrained by lack of convection). In our design, two mechanisms induce mixing: the vehicle’s rotation (if any) creating a centrifugal acceleration, and the internal axial pump circulation.

(a) Design of baffles inspired by ship propellers, with dual functions of slosh mitigation and propellant mixing.



(b) CFD simulation indicates that the new baffle design can fulfill the slosh mitigation functionality.

Figure 5: A modified baffle geometry inspired by ship's propellers will be used to help induce propellant mixing during spin-stabilization.

Even a modest spin rate of the spacecraft (on the order of 2–3 RPM) can generate an effective "gravity" on the order of 0.02–0.05 g at the tank radius, which is enough to cause the heavier (colder, denser) fluid to move outward and the lighter (warmer) fluid to move toward the center. This forms a stable stratification in a rotating frame. Our modified baffles disrupt the flow pattern to deliberately induce radial mixing. The baffles are shaped like a ship's propeller blade so that as the fluid moves due to rotation of the tank. The spinning motion plus the curved baffles set up a secondary flow that drives fluid from the bottom center of the tank upward and pulls fluid from the top periphery down toward the bottom. This convective loop helps equalize any temperature differences.

The axial pump augments this process by actively circulating liquid in a controlled manner. We have conceived a small cryogenic pump (one per tank) that draws fluid from the bottom of the tank and pushes it toward the top. The pump is sized to provide a flow on the order of the tank volume every 24 hours. This turnover rate, combined with the passive mixing from rotation, prevents sustained thermal layers from forming. Any warmer fluid parcel rising to the top is quickly re-mixed with bulk liquid before it can significantly evaporate or increase tank pressure. In the event of excess ullage buildup within the tank, a thermodynamic vent system can be implemented to release small boiloff amounts, which could potentially be utilized as thrust for attitude control as well.

A crucial consideration was ensuring the modified baffles do not impede the original function of suppressing propellant slosh during maneuvers. Early computational fluid dynamics (CFD) analysis results, shown in Figure 5b, suggest that by reducing the size of current Starship baffles together with the addition of a layer of curved baffles, slosh dynamics will remain controlled. Further CFD simulations are planned to iterate on the baffle shape to achieve efficient mixing, while still breaking up slosh waves. We have validated this by simulating linear acceleration cases in CFD. We will continue this validation by simulating rotational acceleration cases as well. Additional CFD analysis will also verify that the pressure drops induced by the baffles and pump are within acceptable limits and that the pump size can indeed deliver the required flow. These simulations will be conducted using StarCCM+, and will directly inform the pump specifications needed. The design goal is an optimal combination of passive (baffles) and active (pump) mixing with minimal added mass or complexity.

In summary, the fluid mixing calculations and simulations indicate that the ETHER system can effectively mitigate thermal stratification with modest rotational spin and a small pump. By doing so, it preserves propellant in a useable state for engine ignition even after long dormant periods. This satisfies the requirement for thermal homogeneity and backup pressure control without imposing large mass or power penalties.

4.3.4 Shield / MLI Design Details to Survive Launch, Orbit, and Descent

The outermost layer of the MLI assembly is a micrometeoroid shield: two Kevlar-reinforced fabric layers and a thin Al_2O_3 aluminum metal oxide aeroshell bonded to it on the hot boundary, with a single Kevlar-reinforced layer near the cold boundary. The outermost grounded aluminum metal oxide shell serves as the first impact surface (Whipple shield bumper) and also dissipates any electrostatic charge acquired while transiting the Earth's radiation belts. The outer, dual Kevlar layer provides impact energy absorption and dissipation and tear resistance, while the inner Kevlar layer supports the blanket curvature and serves as a second point for secondary MMOD fragments impact and energy dissipation. The outer shield additionally functions as an aeroshell to protect the delicate MLI during atmospheric ascent, and is perforated to allow air evacuation so that the blanket segments do not balloon or rip during launch. Once in orbit, ultraviolet degradation and atomic oxygen erosion are mitigated by an aluminized PTFE-impregnated fiberglass fabric (known as Beta cloth) that sits directly on top of the outermost reflective MLI layers. This layer has the added benefits of providing additional ionic charge dissipation and structural support. All MLI materials are rated for vacuum use with low outgassing, and the assembly can withstand temperature extremes without material degradation (Kapton and Dacron remain stable at cryogenic temperatures and through expected temperature cycles). Further research can be done using vibrational and shock modeling to ensure that this structure survives the extreme forces of ascent.

4.3.5 Thermal Performance Summary

With the combined passive and active system boil-off in NRHO is reduced to approximately 260kg of LOX per day, or 23 tons, which is 0.1% of the initial propellant capacity. Therefore, the design meets the initial design goal of $<$ 0.1% prop loss in NRHO, and only a few percent over 30 days on the surface. On the surface, cryocoolers will be used, and will be powered by a carry-along deployable solar power array system. The mass for this solar array system is included in the delta V budget shown in Appendix A. These losses are well within the delta V budget to maintain the required Lunar Ascent Propellant Margin of at least 10%.

5 System Integration

Integrating the required subsystems into the Starship HLS and the Depot Starship requires careful consideration of physical layout, control systems, and operational procedures. The thermal insulation and cooling system and the propellant mixing system must function together seamlessly and interface with other vehicle systems.

5.1 Physical Integration and Layout

The Large Surface Area Hybrid Radiative Insulation system is mounted externally on the HLS propellant tanks, occupying the cylindrical sidewall region of the Starship vehicle. The 28 cooling tubes are arranged vertically along the tank walls, evenly spaced azimuthally and connected at seven manifold rings, dividing tubes into seven groups of four.

These manifolds house the interfaces to the cryocoolers. In our design, we assume the cryocoolers dump heat via Starship's radiator panels, but we have not designed them in detail. If additional heat rejection capacity is needed in orbit, we will use the back side of solar array panels as additional radiators, with low-emissivity coating on the side facing the tanks to prevent unwanted heating. The MLI blanket segments are draped over and attached to the steel tube framework and the 10cm x 10cm grid of G-10 standoffs, forming a multi-panel insulating "jacket" and double Whipple shield around the tanks. Clearances are maintained for all protrusions; for instance, around the attachment points of landing legs or Starship's maneuvering thrusters. Where the MLI must be discontinued (e.g. mounting points or penetrations), careful edge sealing and thermal bridging to a cooling tube are implemented to minimize local heat leaks.

Inside the tanks, the modified baffles are positioned at the top and bottom dome regions. Starship's LOX and CH_4 tanks are separated by a common bulkhead; the methane tank is above the LOX tank. We incorporate a spiral baffle ring near the top of the CH_4 tank and one near the bottom of the LOX tank (since stratification is most likely in low-gravity environments when propellant settles in a layer; the top of an upper tank and bottom of a lower tank are where warmer fluid accumulates in microgravity). The axial pump for each tank is integrated such that it draws fluid via a pipe from the bottom of the tank and discharges near the top. For the CH_4 tank, the pump inlet is at the common bulkhead (which is the bottom of that tank) and the outlet is near the top dome. For the LOX tank, the pump inlet is at its bottom sump and outlet near the top (common bulkhead). The pump can be installed through an access port at the bottom and have a discharge pipe running up the tank wall interior, or it might be built into the central column structure if one exists. In either case, the design avoids impeding the main engine feedlines – the pumps are auxiliary and only operate during coast phases, not during engine feed operation.

The plumbing and wiring integration leverages existing infrastructure as much as possible. Cryocoolers require electrical power and commanding, which ties into the vehicle's power and data systems. They also likely have compressors or moving parts, so mounting them on vibration isolators and outside the crew compartment is prudent. The BAC tubes and cryocooler cold heads contain neon working fluid in a closed loop, so penetrations through the tank wall are needed for the tubing into the cryocooler – these are kept minimal and sealed with high-grade cryogenic fittings. The control of venting is leveraged to assist in mixing when needed (e.g., using vent thrust to spin-up or stir the tank). The axial pumps are electric and draw power from the HLS power system; they will be on switched circuits that activate only when required (to avoid continuous draw). The pump controller receives temperature data from multiple tank thermocouples (sensors distributed vertically in the tank) to determine stratification levels. If a thermal gradient beyond a set threshold is detected, the controller will automatically turn on the pump and, if available, command a gentle vehicle rotation or firing of small RCS jets to help with mixing. All control software is assumed to run on the HLS flight computer with a dedicated cryo subsystem management routine.

6 Technical Management

6.1 Testing, Validation and Technology Maturation

A comprehensive testing and technology maturation plan is in place to bring this combined system from the current conceptual stage (approximately Technology Readiness Level 3) to flight readiness (TRL 6-7 for a full-scale demo, TRL 8-9 for the Artemis III / IV missions). The plan includes progressive component testing, subsystem integration demonstrations, and ultimately a full-scale validation in relevant environments.

6.1.1 Component-Level Testing

The first step is to validate individual components under laboratory conditions:

- **Cryocooler Performance Test:** Using a development model cryocooler (preferably the Creare 90 K unit or an analogue), we will test its cooling power and efficiency at various loads in a thermal vacuum chamber. This verifies that the unit can achieve the expected 8 We /Wth performance and handle cycling. We will also test the neon circulation in a representative tube loop to ensure heat transfer rates meet predictions.
- **MLI Thermal Vacuum Test:** A 60-layer MLI blanket (with Kevlar and aluminum layers) will be applied to a cold surface in vacuum to measure heat flux and verify 0.016 effective emissivity. Tests will also assess outgassing through perforations and durability under thermal cycling.
- **Baffle and Pump Cryogenic Flow Test:** A subscale cryogenic tank (perhaps a cylinder of 1m diameter) will be outfitted with a prototype spiral baffle and a small cryo pump. Filled with liquid nitrogen or liquid oxygen, this setup will test mixing efficacy. We will intentionally stratify the fluid (by letting it sit and warm at the top) and then activate the pump (and rotation, if possible, on a turntable) to measure temperature equalization over time. Data from thermocouples at different heights will confirm the destratification performance. This subscale tank will be flown in LEO on an Oligo satellite by October.
- **Structural and Vibration Testing:** Sections of cooling tubes and MLI hardware will undergo

launch-like vibration and shock testing to ensure mechanical integrity. Kevlar/aluminum layers will also be tested against MMOD-like impacts to validate Whipple shield performance.

Successful completion of these tests will raise the TRL of each technology (e.g. MLI, pump, etc.) to about 4, demonstrating that they work in a lab environment under relevant conditions.

6.1.2 Subsystem Integration Demonstrations

The next phase involves integrating components into working subsystems and testing them in more mission-like conditions:

- **Thermal Subsystem Demo (TRL 5):** We plan to integrate a large insulated cryogenic tank simulator with one cryocooler and BAC loop in a thermal vacuum chamber. This could be a cylindrical tank of perhaps 2 m diameter equipped with several cooling tubes and wrapped in the MLI blanket. The cryocooler (placed outside the chamber with feed-through connections to the tubing) will actively cool the tank while we simulate orbital heat loads using infrared lamps or heaters on the outside. The objective is to demonstrate near-zero boil-off in a vacuum environment. We will measure boil-off gas production with and without the system active to quantify improvement. This test validates the combined performance of MLI + active cooling as a subsystem.
- **Mixing Subsystem Demo (TRL 5):** In parallel, a larger-scale mixing test will be conducted, potentially in a microgravity simulation. One option is a parabolic flight experiment: a small insulated LN_2 tank with internal baffles and pump can be flown on a parabolic trajectory aircraft. During periods of microgravity (20 seconds), we can trigger a heater at the top to create a stratification and then start the pump to see if mixing initiates. Although the microgravity duration is short, repeated parabola runs with incremental heating can mimic a partial stratification and demonstrate the pump's ability to move fluid. Another option is to use a neutral buoyancy or rotating simulator on ground: for example, spinning the tank to create pseudo-gravity and then stopping it to simulate microgravity while monitoring temperature (this is complex but can provide some data). The main goal is to refine CFD models with experimental data.
- **Integrated System Demonstration (TRL 6):** Once the separate subsystems are validated, we aim for an integrated test of the full system in a relevant environment. A LEO orbital flight on Starship, while short relative to 90 days and with a different thermal environment than NRHO or the surface near the lunar South pole, would demonstrate that the systems operate as intended in zero-g and vacuum together. This could achieve TRL 6 by validating operation of the full-scale integrated system in a relevant environment (space-like conditions).

Throughout these tests, we will use analytical tools (Thermal Desktop for thermal analysis, CFD for fluid, FEA for structural) to correlate predictions with results, increasing confidence in our models. Discrepancies will be used to improve the design (for example, adjusting layer counts, pump sizes, control algorithms).

6.1.3 Prototype Subscale Tank with Baffles

We are collaborating with Oligo Space, a satellite startup, to fly a sub-scale prototype of our novel baffle system in space in early 2026. The experiment requirements and high-level design are complete, and we are now finalizing detailed design, manufacturing, and testing in compliance with Oligo and SpaceX payload integration guidelines. Although the flight unit is based on the model presented here, qualifying a pressurized payload for launch requires extensive validation, all scheduled for completion this summer. The small, agile teams at both Oligo and our group enable rapid development. One of our team members is interning at Oligo during summer 2025, facilitating coordination. Oligo's CEO, Jacob Rodriguez—an MIT alum and former NASA Big Idea Challenge participant—is enthusiastic about this collaboration as a test case for their ability to mature low-TRL hardware. The objective is to advance the baffles from TRL 2 to TRL 6.

6.2 Path-to-Flight Schedule & Milestones

The project requires certain novel technologies which must be prioritized for TRL maturation in order to achieve the 3-5 years schedule for Artemis missions. Following successful subsystem demonstrations, the next steps would involve scaling up to full-size systems and integrating into a prototype lander stage for flight testing, as shown in Figure 6 and Table 2:

- **Full-scale HEO Flight Test for HLS (TRL 7):** taking a prototype Starship HLS, integrating the complete insulation system and mixing system, and performing a long-duration cryogenic storage test in orbital flight in HEO. This would demonstrate the system at full scale in an operational environment that approximates NRHO.
- **Certification Uncrewed Test for HLS (TRL 8):** Perform a multi-month flight test of the full system—including HLS, Depot, and Tanker—in operational orbits (LEO, HEO, NRHO, LSP). A Starship test article outfitted with instrumentation would validate the boil-off control system under crewed mission ConOps, certifying it for Artemis III.
- **Flight Operations (TRL 9):** occurs when the system is proven in an actual mission environment with crew on board (for example, Artemis III or IV HLS using the system to successfully loiter and return). At that point, the design would be fully validated and any last operational tweaks would be incorporated.

In parallel to these hardware tests, the project coordiante closely with SpaceX and NASA to ensure that integration issues are ironed out. For instance, we would pursue a flight qualification of the MLI blanket in terms of flammability (due to Kevlar) and verify that all materials meet NASA's safety standards for crewed flight. We will also develop operational procedures and failure mode contingency plans through simulations and possibly hardware-in-the-loop tests (e.g., simulate a cryocooler failure in a lab and verify that the mixing system and venting can handle it).

This stepwise, build-test-learn approach advances system maturity, reduces risk, and improves design confidence. By crewed mission time, the Large Surface Area Hybrid Radiative Insulation and Propellant Mixing System will be rigorously validated for reliable HLS performance during long-duration loiter and surface phases.

	TRL				
Technology	2025	2026	2027	2028	2029
Petach Cryocoolers	4	6	7	8	8
Double Whipple Shield	9	9	9	9	9
Sawtooth Radiators	4	6	7	8	8
Large MLI Blanket	9	9	9	9	9
LVSAT	6	6	7	8	8
LUNAR SABER	4	5	6	6	8
Mixing Pump	6	7	8	8	8
Twisted Baffles	4	6	8	8	8

Figure 6: TRL maturation plan for technologies used in THERMOS

Year	Milestones
2025 H1	Present at HuLC
2025 H2	Detailed Design and Project Planning
2026 H1	Build tanks with new MLI, cryocoolers & Baffles
2026 H2	Flight test aboard one of the regular Starship tests
2027 H1	Incorporate flight learnings; uncrewed HLS flight to LEO / HEO
2027 H2	Uncrewed HLS to HEO, NRHO & LSP, uncrewed Depot to LEO
2028 H1	Incorporate changes to HLS and Depot
2028 H2	Repeat all uncrewed flight tests
2029 Q1	Conclude human-rating certification of THERMOS
2029 Q2-Q4	THERMOS available for use in crewed Artemis III mission

Table 2: THERMOS Project Timeline

6.3 Risk Analysis and Mitigations

Three critical risks were identified. First, total cryocooler failure would be catastrophic, but the hybrid passive-active system—using MLI, the ETHER mixer, and venting—can maintain safe tank conditions without active cooling. Second, failed propellant mixing (from baffle or pump issues) is a major concern during coast phases, mitigated by CFD-optimized baffle design and redundancy via passive (spin) and active (pump) mixing. Third, excessive boil-off threatening the Lunar Ascent Propellant Margin (LAPM) below 10% is rare but severe; the system has been stress-tested under worst-case scenarios (e.g., triple cryocooler failure, degraded MLI) to ensure propellant retention.

6.4 Cost Estimates & Margin

We assumed that the primary cost driver will be the form of contract used. The subsystems described above are intended to be integrated with vehicles already being developed by SpaceX to fulfill its obligations under the HLS program. So, the natural contractor to carry out this work is SpaceX, and the natural form of contract would be a firm, fixed-price extension or addendum (if one is required) to the current Artemis HLS contract. To estimate the value of this contract, we assumed the following. Firstly, SpaceX has a track record of bringing in systems at 1/10th the cost of traditional space systems development. Edgar Zapata, a now-retired NASA costs expert, estimated that Falcon 9 would have cost $4B instead of the actual amount of $400M to develop, had it been developed using the traditional cost-plus approach [7]. The esteemed space journalist Eric Berger has compared Starship development ($3B to date, and 8 test flights) with SLS development ($30B, 1 test flight) and also concluded that SpaceX and Blue Origin can complete space systems development and operations projects at 1/10th of the traditional cost-plus costs[2]. Secondly, we noted that SpaceX has recently accepted a firm, fixed price contract with NASA to de-orbit the ISS for approximately $850M, to include developing and testing a spacecraft and executing the mission.

Compared to the above, the THERMOS proposal is for only a small subset of a spacecraft's systems. Therefore, SpaceX's costs should, in principle, be significantly less than the contracted cost for the ISS deorbit project. However, we are still in conceptual design; thus, an indicative range for development cost is $200M-$500M. A more accurate development cost estimate will be provided with our final report. Operating costs have been budgeted to not exceed 20% of maximum development costs, which is a reasonable analog for many similar projects, and therefore will not exceed $100M per year for the years of active Artemis III mission operations. The main element of operating cost would be vendor support, training, etc. for the MLI, cryocooler, active pump used by the propellant mixing device, and deployable lunar solar arrays.

7 Technical Innovation Summary

7.1 Novelty & Uniqueness of the Proposed Concept

THERMOS addresses the HuLC challenge area of "long-term cryogenic propellant storage in orbit and on the lunar surface for Starship HLS." We focus on safely and reliably storing cryogenic propellants in liquid form for the Starship HLS vehicle, meeting the quantity and time interval requirements of the Artemis III mission CONOPS. Our approach centers on intercepting and rejecting heat loads that would otherwise cause boil-off and secondarily on promoting cryogenic propellant mixing to maintain thermal homogeneity. We achieve this through an integrated thermal control system comprised of Multi-Layer Insulation (MLI) with integrated broad-area active cooling (BAC) and a novel propellant management system. The propellant management system consists of curved baffles, featuring two-dimensional curvature, integrated with stirring jets. Unlike traditional approaches that treat MLI and cryocoolers as independent subsystems, we propose a tightly integrated design where the MLI is structurally supported by the cryocooler cooling tubes. By placing thermal bridges into the MLI that lead directly to the cryocooler loop, we aim to intercept any heat leaks before they reach the propellant tank walls, maintaining near-ZBO and thermal homogeneity. Our propellant management system incorporates curved baffles featuring two-dimensional curvature and stirring jets. The curvature of the baffles is specifically tailored to induce swirling flows within the tank, increasing the surface area for heat transfer and preventing stratification.

7.2 Potential Impacts & Benefits

The integrated MLI and BAC system minimizes heat leaks into the propellant tanks, significantly reducing boil-off rates during the extended NRHO loiter and lunar surface stay. By achieving near-ZBO in NRHO and controlling boil-off on the lunar surface, our solution maintains healthy propellant margins and enhances mission robustness, ensuring sufficient propellant for ascent and rendezvous with Orion. The core concepts of our integrated thermal control and propellant management system are readily scalable and adaptable to other cryogenic propellant storage applications in cislunar space and on planetary surfaces. This makes our solution a valuable asset for future lunar exploration and development efforts, as well as potential missions to Mars and beyond.

8 Conclusion and Next Steps

THERMOS enhances Starship HLS by controlling cryogenic fluid boiloff to extend mission duration, while also respecting the constraints of the original system. This integrated thermal management system combines Multi-Layer Insulation (MLI), broad-area active cooling (BAC), and the Propellant Mixing System (ETHER) to achieve near-zero boil-off in orbit, prevent stratification, and ensure propellant readiness. Carry-along vertical solar arrays such as LAMPS or LVSAT, currently in development, are deployed by the crew to power the cryocoolers on the lunar surface. The design balances passive and active elements, leveraging synergies such as structural cooling tubes doubling as insulation support and vented boil-off aiding the necessary attitude control. Engineering analyses confirm feasibility: thermal modeling indicates minimal boil-off due to MLI and cryocooling; fluid dynamics demonstrate effective mixing, preventing vapor pocket formation; and the systems engineering process, where mission design was carried out concurrently with system design, leads to the CFM system remaining mission-capable even with active cooling failures or severely degraded MLI. A Technology Readiness Level maturation plan will validate the system and its elements through prototyping, testing, and flight demonstrations. Beyond Artemis III these technologies could support propellant depots in cislunar space and extended-duration missions, enabling deep-space human spaceflight to the Moon, Mars and beyond.

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A Calculations Appendix: HLS Delta V and Propellant Budget

Starship HLS Delta V and Prop Boiloff Budget for Artemis III													
Starship version:	V3	Dep. Orbit: TLI -	1040	m/s	PROPNEEDHEO	1,100,363							
Endurance NRHO:	90	Endurance LSP:	30	days									
Front insulation:	MLI40-A139	1-Way Payload:	13,152	kg									
Side insulation:	MLI60-A144	Return Payload:	30,000	kg									
Event - STAGE or Burn #	**From**	**To**	**Purpose**	**Event Start at Mission Elapsed Time (MET) (seconds)**	**Event Ends at Mission Elapsed Time (MET) (seconds)**	**Wet mass before Event (kg)**	**Propellant mass before Event (kg)**	**Delta V (m/s)**	**Total Propellant Received during event (kg)**	**Total Boiloff during Event (kg)**	**Wet mass after Event (kg)**	**Propellant Mass Change**	**Propellant mass after Event (kg)**
1	LEO 250	HEO	Enter HEO	0	300	2,500,226	2,300,000	2,160	0.0	4.3	1,399,838	-1,100,388	1,199,612
REFILL	HEO	HEO	Refill propellant	300	9103	1,399,838	1,199,612	0	1,100,388.1	25.6	2,500,201	1,100,363	2,299,974
2	HEO	TLI	TLI Manuever	9103	268303	2,500,201	2,299,974	1,040	0.0	752.8	1,890,239	-609,962	1,690,013
3	TLI	NRHO	Enter NRHO	268303	268483	1,890,239	1,690,013	450	0.0	0.5	1,675,087	-215,152	1,474,861
LOITER	NRHO	NRHO	Loiter at NRHO	268483	8044483	1,675,087	1,474,861	0	0.0	22,584.4	1,652,503	-22,584	1,452,276
DOCK	NRHO	NRHO	Dock with Orion	8044483	8048083	1,652,503	1,452,276	100	0.0	10.5	1,608,708	-43,794	1,408,482
4	NRHO	LLO 50	Enter LLO	8048083	8062483	1,608,708	1,408,482	750	0.0	41.8	1,315,219	-293,489	1,114,993
5	LLO	LSP	Landing Burn	8062483	8066083	1,315,219	1,114,993	2,050	0.0	10.5	758,438	-556,781	558,212
SURF MISSION	LSP	LSP	Operations during Surface Missio	8066083	10658083	758,438	558,212	0	0.0	134,049	624,389	-134,049	424,163
6	LSP	LLO 50	Enter LLO	10658083	10661683	611,237	424,163	1,860	0.0	10.5	370,923	-240,315	183,848
7	LLO 50	NRHO	Enter NRHO	10661683	10676083	370,923	183,848	750	0.0	41.8	303,220	-67,703	116,146
DOCK	NRHO	NRHO	Dock with Orion	10676083	10679683	303,220	116,146	100	0.0	10.5	295,176	-8,044	108,101
8	NRHO	TEI	TEI Manuever	10679683	10938883	295,176	108,101	450	0.0	753	260,826	-34,350	73,751
9	TEI	HEO	HEO	10938883	10938903	260,826	73,751	1,040	0.0	0	197,272	-63,554	10,197
PROP XFER	HEO	HEO	Prop xfer at HEO	10938903	11025303	197,272	10,197	0	0.0	251	397,021	199,749	209,946
10	HEO	HEO	Phasing at HEO?	11025303	11111703	397,021	209,946	0	0.0	251	396,770	-251	209,695
11	HEO	LEO 250	Enter LEO	11111703	11154903	396,770	209,695	2,160	0.0	125	222,146	-174,624	35,071

Figure 7: HLS Delta V and Propellant Budget accounts for burns, boiloff and propellant transfers and results in the calculation of LAPM

B Calculations Appendix: Material Selection

Table 3: Materials Selection for Major Components

Component	Material	Rationale
Propellant tanks	Stainless Steel (304L)	Baseline Starship tank material; high strength and toughness at cryogenic temperatures
Cooling tubes (BAC loops)	Stainless Steel	Welded directly to tanks for thermal contact; compatible CTE with tank; robust under launch loads
MLI Outer Cover Material	Aluminized Beta Cloth	
MLI reflector layers	Aluminized Kapton film	Low emissivity reflective aluminum layers for radiative insulation; Kapton is space-rated (vacuum & UV). Although Kapton is 50% denser than mylar, it is more resilient to the extreme lunar temperature swings (mylar can become brittle). These layers only make up 25% of the MLI mass, so Kapton proves to be a better option despite the added mass.
MLI spacer material	Dacron netting (polyester)	Low thermal conductivity spacers to separate layers; minimal solid conduction paths
MLI structural layers	Kevlar fiber fabric	Forms outer MLI blankets for tensile support; Kevlar is lightweight, high-strength, and maintains integrity in vacuum
Outer shield (Aeroshell and MMOD protection)	Aluminum alloy thin shell	Mounted external to MLI as Whipple shield, to protect MLI during launch and to prevent charging
MLI blanket mounts	G-10 fiberglass, PEEK and/or Aerogel	Rigid, non-thermally conductive stand-offs and brackets to attach MLI blanket to steel tubes/tank with minimal heat conduction
Internal baffles	Stainless Steel 304L	Only baffle geometry is changed, welding and steel alloy same as current approach
Axial mixing pump	Aluminum housing, SS impeller	Aluminum metal oxide for weight savings in pump body; stainless impeller and hardware for strength in cryogenic fluid
Pump seals/bearings	Teflon-coated or ceramic	Non-reactive, low-temperature-rated seals and bearings for cryogenic fluids (LCH_4, LO_2)

C Calculations Appendix: Model Parameters

Parameter or Intermediate Values	Excel Variable Name	Value	Units
LCH4 to LOX mass ratio	**LCH4LOXRATIO**	0.22	none
Sea level Raptor ISP	**SLRAPISP**	327	s
Vacuum Raptor ISP	**VACRAPISP**	380	s
g	**GEE**	9.80	m/s^2
Duration of landing burn	**LANDBURN**	10.00	s
Delta V needed Starbase to LEO 250 x 250 (incl 2,000 m/s losses)	**DVEARTHTOLEO**	9,500	m/s
Delta V LEO 250 x 250 to TLI	**DVLEOTOTLI**	3,200	m/s
Delta V LEO 250 x 250 to HEO 250 x ????	**DVLEOTOHEO**	2,300	m/s
Delta V HEO 250 X ???? to TLI	**DVHEOTOTLI**	900	m/s
Delta V HEO 250 X ???? to LEO 150 x ? (reentry)	**DVHEOTOLEO**	50	m/s
Delta V TLI to NRHO	**DVTLITONRHO**	450	m/s
Delta V Orion docking burn	**DVORIONDOCK**	100	m/s
Delta V NRHO to LLO 50 x 50	**DVNRHOTOLLO**	750	m/s
Delta V LLO 50 x 50 to lunar South pole	**DVLLOTOLSP**	2,050	m/s
Delta V NRHO to TEI	**DVNRHOTOTEI**	450	m/s
Delta V TEI to HEO 300 x ????	**DVTEITOHEO**	900	m/s
Delta V TEI to LEO 250 x 250	**DVTEITOLEO**	3,000	m/s
Delta V Lunar South Pole to LLO 50 x 50	**DVLSPTOLLO**	1,860	m/s
Delta V deorbit burn from LEO 250 x 250	**DVDEORBITLEO**	100	m/s
Delta V budget for orbital maneuvers of tanker	**DVTNKORBMAN**	50	m/s
Delta V budget for landing burn	**DVLANDBURN**	196	m/s
Super Heavy dry mass (excluding upper stage)	**SHDRYMASS**	250,000	kg
Super Heavy propellant mass (note - not tracking LCH4/LOX for SH)	**SHPROPMASS**	4,050,000	kg
Super Heavy propellant remaining at MECO	**SHPROPREMMECO**	400,000	kg
Total wet mass of upper stage being boosted if boosting tanker	**MASSUPPERSTAGETNK**	2,638,767	kg
Total wet mass of upper stage being boosted, if boosing depot	**MASSUPPERSTAGEDEP**	2,656,544	kg
Total wet mass of upper stage being boosted if boosting lunar cargo s	**MASSUPPERSTAGECARGO**	2,556,484	kg
Delta V imparted on upper stage by SH	**STARSHIPDVATSTAGING**	2,393	m/s
Terminal velocity of tanker (for belly flop maneuver)	**TNKTERMVEL**	80	m/s
Tanker dry mass (structures only, excl. payload)	**TNKDRYMASS**	138,767	kg
Tanker propellant mass (all)	**TNKPROPMASS**	2,500,000	kg
Tanker LOX "payload"	**TNKLOXPAYLOAD**	0	kg
Tanker LCH4 "payload"	**TNKLCH4PAYLOAD**	0	kg
Tanker propellant delivered to HLS at HEO (max)	**MAXPROPDELHEO**	200,000	kg
HLS dry mass (structures only, excl. Artemis crew/eqpmt)	**HLSDRYMASS**	156,484	kg
HLS propellant mass (all)	**HLSPROPMASS**	2,300,000	kg
HLS LOX mass	**HLSLOXMASS**	1,889,286	kg
HLS LCH4 mass	**HLSLCH4MASS**	410,714	kg

Figure 8: Main Model Parameters and Variables (1 of 2)

Tanker propellant transferred to Depot per trip	**TNKPROPDELIVERED**	276,607	kg
Number of tankers required to fully refill tanks of a LEO Depot	**NOTANKERS**	15	number
Depot dry mass	**DEPDRYMASS**	156,544	kg
Depot propellant mass (all)	**DEPPROPMASS**	2,500,000	kg
Depot LOX "payload"	**DEPLOXPAYLOAD**	2,053,571	kg
Depot LCH4 "payload"	**DEPLCH4PAYLOAD**	446,429	kg
Delta V imparted on Depot by SH	**DEPDVATSTAGING**	2,384	m/s
Time for propellant transfer for Depot	**DEPPROPXFERTIME**	20,000	s
Propellant transfer rate	**PROPXFERRATE**	125.0	kg/s
Density for steel 301	**ST301RHO**	7,930	kg/m^3
Area of Starship solar panels		250	m^2
Solar conversion efficiency (MOL)		37%	
Power system efficiency		80%	
Electrical power available in HEO / NRHO orbit (facing Sun)		101,158	W e
Electrical power available on lunar surface		50,579	W e
Excess power available in NRHO to run cryocoolers	**EXCESSPOWERNRHO**	50,579	We
Boiloff Rates			
HLS boiloff rate at LEO	**BORATELEO**	0.01425	kg / s
Depot boiloff rate at LEO	**BORATELEO_DEP**	0.00238	kg / s
Tanker boiloff rate at LEO	**BORATELEO_TNK**	0.17333	kg / s
HLS boiloff rate at NRHO	**BORATENRHO**	0.00273	kg / s
Depot boiloff rate at NRHO / HEO	**BORATENRHO_DEP**	0.00000	kg / s
HLS boiloff rate at LSP	**BORATELSP**	0.05172	kg / s

Figure 9: Appendix: Main Model Parameters and Variables (2 of 2)

D Calculations Appendix: Model Dashboard

Dashboard knobs- Performance goals / decisions	Value			Units	Notes	
Required Endurance at NRHO	90			days	Artemis III requirement, capability to wait at NRHO for Orion; relax it to compare how a purely-commercial mission might perform	
Required Endurance at LSP	30			days	Artemis III requirement, cannot be less than this due to choice of NRHO rendezvous & safety requirements if ascent launch windows missed	
Starship HLS Payloads staying on Moon	13,152			kg	*This can be traded with HLS ascent propellant margins. Includes all specialized exploration equipment, consumables, rovers, deployable solar arrays - anything that will stay on Moon*	
Starship HLS Payloads returning to NRHO	30,000			kg	*This can be traded with HLS ascent propellant margins. It includes all specialized habitation equipment installed on Starship HLS, the crew and any equipment they have to bring back.*	
How much delta V from HEO to TLI?	900			m/s	Set it to 3200 for LEO departure orbit. The higher the HEO apogee, the lower this should be. Default is 700 m/s for v3 depot in the v5 model, 850 m/s for the v2 depot. Lower it further to raise HEO apogee in order to make a mission feasible and/or to increase HLS prop margins, at the cost of having to send maybe more tankers to refuel the depot that will travel to HEO. (Raising HEO apogee will improve HLS starship propellant margins at the expense of a few more tankers required. It is expected that for supporting refilling for lunar missions, the apogee of the (HEO) highly elliptical orbit will end up being in MEO (Medium Earth Orbit))	
Dashboard knobs - Technology Choices	**Value**	**Value**	**Value**	**Propellant Margin LSP - NRHO**	**18.32%**	**LSP BO RATE per day**
	For HLS	For Depot	For Tanker	**Total Number of Tankers**	**16**	**3.2** tons
Starship Version	V2	V2	V2		*Changing this allows us to run the model with V1, V2 or V3 Starship and see how mission feasibility is affected. For Starship specs, see LOOKUP TABLES tab.*	**NRHO BO RATE per day**
Passive Insulation solution - SS front	MLI60-A144	MLI60-A144	301STEEL		This defines the passive insulation strategy for the front of starship (e.g. area above the solar panels) used for calculations.	**0.2** tons
Passive Insulation solution - SS sides	MLI60-A144	MLI60-A144	301STEEL		This defines the passive insulation strategy for the side of starship (e.g. area under the solar panels). If both MLI and cryocoolers are used, the mass of the MLI structural support system is subtracted	
Radiator design - backside of solar panels	Sawtooth	Sawtooth	Regular b…		"Sawtooth" is a design with a small mass penalty where the radiator panel's cross section is like a sawtooth profile, and the sides facing the tank have low emissivity, whereas the sides looking at space have high emissivity	
Cryocooler technology	CREARE	CREARE	NONE		this defines whether active cryocoolers are used to achieve ZBO or not. selecting cryocoolers adds mass to the system, increase power needs == in case of HLS, we have 50kWe spare in NRHO as the design loads for when crew is on board cannot exceed 50 of the 100kWe. If any cryocoolers are used, a fixed mass is added for the network of pipes and manifolds	
Cryo Sizing: Max Thermal Power of Cryocoolers used, per Cryocooler Unit	150	150	0	W(th)	watts of heat removed by cryocooler (coooling capacity)	
Number of Cryocoolers to use	7	7	0		the number of cryocoolers used. Together with the type of cryocooler, this determines total cooling capacity, total power demand and total additional mass	
Take deployable solar arrays as 1-way payload to power the cryocoolers while landed?	Yes			**Relative Schedule Risk**	**29.25**	
Baffles Customized for Prop Mixing	2D Curved	2D Curved	2D Curved	out of V3 Hybrid max	**39.25**	
Dashboard knobs used for stress-testing the HLS mission						
MLI Degradation factor	4				This factor is multiplied with effective emissivity. It is modeled as an average across the entire passive insulation of the HLS system.	
Number of failed cryocoolers	0				Any failures are assumed to occur upon the first startup, and then they remain inactive for the entirety of the HLS mission.	

Figure 10: Appendix: Model Dashboard, showing the Design Vector variables available to change. Each design vector is potentially a different architecture.

E Calculations Appendix: Dry Mass Calculator

			HLS	Depot	Tanker	
Starship Version			V3	V3	V3	
Passive Insulation solution - SS front			MLI60-A144	MLI60-A144	301STEEL	
Passive Insulation solution - SS sides			MLI60-A144	MLI60-A144	301STEEL	
Cryocooler technology			CREARE	CREARE	NONE	
Number of Cryocoolers to use			7	8	0	
Custom Baffles for Prop Mixing			Regular	Regular	Regular	
Baseline Dry Mass			138767	138767	138767	kg
+ Mass of passive insulation (front)	**FRONTAREACONE**	**PASSINSULFRONT**	1501	1501	0	kg
+ Mass of passive insulation (sides)	**SIDEAREA**	**PASSINSULSIDE**	11305	11305	0	kg
+ Delta mass of custom baffles			0	0	0	kg
+ Mass of cryocoolers			420	480	0	kg
+ Delta mass of solar panels (if sawtooth radiators used)			1500	1500	0	kg
+ Mass of cooling tubes & manifolds			4969	4969	0	kg
- Mass of MLI supports, if both Cryo and MLI used	FRONTAREACONE	SIDEAREA	-1979	-1979	0	kg
Total Dry Mass of Configured Starship	**SSDRYMASS**		**156484**	**156544**	**138767**	
Mass of Payloads Staying on the Moon						
Mass of deployable solar arrays			752			kg
Consumables			2400			kg
Rovers			6000			kg
Experiments			4000			kg
Total	**PAYLOADS1WAY**		**13152**			kg
MLI Calculations Section						
Custom Baffles Calculations Section						
Cryocooler Tubes Mass Calculations Section						
	Variable name	**Value**	HLS Result	Depot Result	Tanker Result	**Unit**
Outer diameter of cooling tube	**TUBEOUTERD**	14.00				mm
Inner diameter of cooling tube	**TUBEINNERD**	8.00				mm
Mass of cooling tubes per m of length (incl work fluid)	**LINMASSTUBES**	1.09	1.09	1.09	1.09	kg / m
Lateral spacing of tubes on tank	**TUBESPACING**	1	1	1	1	m
Length of tank	SHIPHEIGHT		69.8	69.8	69.8	m
Number of tubes on tank			28	28	28	#
Total length of tubes			3909	3909	3909	m
Total mass of tubes			4269	4269	4269	kg
Number of manifolds (=2X number of crycoolers)		14	14	14	14	#
Mass per manifold	MANIFOLDMASS	50	50	50	50	kg
Total mass of cryocooler manifolds and tubes	**MASSTUBEMAN**		**4969**	**4969**	**4969**	kg
Mass of lightweight MLI supports per unit area	**MASSMLISUPP**	2				
Deployable Solar Panels Calculation						
Design power Wth of HLS Cryocoolers (total)			1050			Wth
Electrical power demand of cryocoolers, +20%	**LUNARARRAYSPWR**		10080			We
Area of solar panels required			28.9			m^2
Specific mass of solar arrays per unit area	SPECMASSPANELS	20				kg / m^2
Total mass of deployable solar panels required			578			kg
Plus overhead mass for tracking and structure		30%	174			kg
Total mass of deployable arrays	**MASSDEPLARRAYS**		**752**			kg

Figure 11: Appendix: Dry Mass Calculator. The technology choices made result in dry mass changes which update the vehicle and feed through to the delta V budget and the calculation of LAPM

F Calculations Appendix: Boiloff Calculator

			HLS	DEP	TNK	
Starship Version			V3	V3	V3	
Passive Insulation solution - SS front			MLI60-A144	MLI60-A144	301STEEL	
Passive Insulation solution - SS sides			MLI60-A144	MLI60-A144	301STEEL	
Cryocooler technology			CREARE	CREARE	NONE	
Max thermal lift per cryocooler			150	150	0	
Number of Cryocoolers to use			7	8	0	
Custom Baffles for Prop Mixing			Regular	Regular	Regular	
Net heat flux into propellant tanks	**NETHEATFLUX**	LEO	6805	1139	82778	W
Net heat flux into propellant tanks	**NETHEATFLUX**	NRHO / HEO	1304	0		W
Net heat flux into propellant tanks	**NETHEATFLUX**	LSP	24698			W
	HVAPLOX	214000.00				W*s/kg
Boiloff rate LOX tank		LEO	0.01590	0.00266	0.19341	kg / s
Boiloff rate LOX tank		NRHO / HEO	0.00305	0.00000		kg / s
Boiloff rate LOX tank		LSP	0.05771			kg / s
	HVAPMETH	511000.00				W*s/kg
Boiloff rate LCH4 tank		LEO	0.00666	0.00111	0.08100	kg / s
Boiloff rate LCH4 tank		NRHO / HEO	0.00128	0.00000		kg / s
Boiloff rate LCH4 tank		LSP	0.02417			kg / s
Net Heat Flux Calculations						
Projected area front	FRONTAREAPROJ		63.6	63.6	63.6	m2
Area side	SIDEAREA		1,747.4	1,747.4	1,747.4	m2
Orbit-average front area looking at Sun	OAFRONTSUN	LEO	59%	59%	59%	
Orbit-average side area looking at Sun	OASIDESUN	LEO	0%	0%	0%	
Orbit-average front area looking at Earth	OAFRONTEARTH	LEO	41%	41%	41%	
Orbit-average side area looking at Earth	OASIDEEARTH	LEO	29.5%	29.5%	29.5%	
Orbit-average front area looking at Sun	OAFRONTSUN	NRHO	100%	100%		
Orbit-average front area looking at Sun		LSP	0%	0%	0%	
Orbit-average side area looking at Sun		LSP	50%	50%	50%	
Orbit-average side area looking at Moon		LSP	50%	50%	50%	
Front absorptivity			0.0185	0.0046	0.25	
Front effective emissivity			0.0185	0.0046	0.58	
Side absorptivity			0.0185	0.0046	0.25	
Side effective emissivity			0.0185	0.0046	0.58	
Environmental heat flux reaching exterior tank	ENVHEATFLUX	LEO	7211	1803	77771	W th
Environmental heat flux reaching exterior tank	ENVHEATFLUX	NRHO	1610	403		W th
Environmental heat flux reaching exterior tank	ENVHEATFLUX	LSP	24029			W th
Environmental heat flux reaching tanks	ENVHEATFLUX	LEO	7211	1803	77771	W th
Environmental heat flux reaching tanks	ENVHEATFLUX	NRHO	1610	403		W th
Environmental heat flux reaching tanks	ENVHEATFLUX	LSP	24029			W th
Internal heat flux conducted to tanks	INTHEATFLUX		500	500	500	W th
Solar panel heat flux onto side of tanks	BACKPANELFLUX	LEO	144	36	4507	W th
Solar panel heat flux onto side of tanks	BACKPANELFLUX	NRHO	244	61		W th
Solar panel heat flux onto side of tanks	BACKPANELFLUX	LSP	1219			W th
Power-limited cooling capacity (in orbit)			6322	6322		
Power-limited cooling capacity (on lunar surface)			1260			
Heat flux removed by CC (in orbit)			-1050	-1200	0	W th
Heat flux removed by CC (on lunar surface)			-1050			W th

Figure 12: Appendix: Boiloff Calculator. This sheet performs all the calculations to estimate the net heat fluxes into the tanks.

G Calculations Appendix: Starship Lookup Tables

Lookup table name			*Excel data range name*		*Rows lookup range:*	
Starship Version Lookup Table			**STARSHIPLOOKUP**		**STARSHIPLOOKUP_ROWS**	
Starship Version	**Model var name**	**Row No.**	**V1**	**V2**	**V3**	**Units**
Max payload mass to destination		2	0	100000	200000	kg
Tanker propellant load	**TNKPROPMASS**	3	1200000	1600000	2500000	kg
Depot propellant load	**DEPPROPMASS**	4	1200000	1600000	2500000	kg
HLS propellant load	**HLSPROPMASS**	5	1200000	1500000	2300000	kg
Ship height	**SHIPHEIGHT**	6	50.3	52.1	69.8	m
Tanker dry mass baseline, before tech	**TNKDRYMASS**	7	100000	103578	138767	kg
Depot dry mass baseline, before tech	**DEPDRYMASS**	8	100000	103578	138767	kg
HLS dry mass baseline, before tech	**HLSDRYMASS**	9	100000	103578	138767	kg
Booster prop load	**SHPROPMASS**	10	3300000	3650000	4050000	kg
External surface area of Starship	**SURFACEAREA**	11	1428	1479	1979	m2
Front surface area of Starship, projected	**FRONTAREAPROJ**	12	64	64	64	m2
Front surface area of Starship, conic	**FRONTAREACONE**	13	232	232	232	m2
Side surface area of Starship	**SIDEAREA**	14	1196	1247	1747	m2
Baffles Subsystem Mass	**STDBAFFLES**	15				kg

Figure 13: Appendix: Starship Lookup Tables. This sheet reconfigures the calculation parameters such as Starship dry mass, propellant capacity and surface area, depending on which Starship block (V1, V2, V3) was selected for each element (HLS, Depot or Tanker).

H Calculations Appendix: Coating Properties Lookup Table

Surface Material Description	Lookup name	mass in kg / m2	Abs	Emm
	don't use this			
	BLACK	1	0.99	0.98
No coating, 301 steel	**301STEEL**	0	0.25	0.58
White Paint	**WHITE**	1	0.12	0.9
Magnesium Oxide in White paint	**MGOWHITE**	1	0.09	0.9
AZ Tech Inorganic Low Alpha White Paint	**AZTWHITE**	1	0.09	0.91
silver backed, used on skylab, space shuttle, and hubble	**QIOPTIQ**	1	0.06	0.83
SOLEC LO/MIT	**SOLEC**	1	0.23	0.17
MLI, 40 layers	**MLI40**	12	0.005	0.005
MLI, 20 layers	**MLI20**	8	0.006	0.006
MLI, 60 layers	**MLI60**	16	0.004	0.004
MLI60-A138, Johnson (2010)	**MLI60-A138**			
MLI40-A139, Johnson (2010)	**MLI40-A139**	6.43	0.0049	0.0049
MLI60-A140, Johnson (2010)	**MLI60-A140**	6.78	0.0042	0.0042
MLI60-A141, Johnson (2010)	**MLI60-A141**	6.67	0.0059	0.0059
MLI20-A142Q, Johnson (2010)	**MLI20-A142Q**	5.79	0.0071	0.0071
MLI60-A143, Johnson (2010)	**MLI60-A143**	6.66	0.0050	0.0050
MLI60-A144, Johnson (2010)	**MLI60-A144**	6.47	0.0046	0.0046

Figure 14: Appendix: Coating Lookup Tables. This sheet reconfigures the assumed parameters for absorptivity and emissivity for the front and sides of each element (HLS, Depot or Tanker), depending on what passive insulation strategy is chosen for that element and for its front and side.

I Calculations Appendix: Cryocooler Version Lookup Table

Type of Cryocooler	Model var name	Row No.	SPEC MASS	PWR RATIO
units			kg/W(th)	none
no cryocooler used	**NONE**	2	0	0
generic efficient cryocooler	**GENERIC**	3	2	20
heavy, inefficient cryocooler	**HEAVYGEN**	4	6	60
Petach pulse-tube cryocooler	**PETACH**	5	0.7	15
Creare 150W 90K	**CREARE**	6	0.4	8

Figure 15: Appendix: Cryocooler Lookup Tables. This sheet reconfigures the assumed parameters for the specific mass and power ratio of cryocoolers for each element (HLS, Depot or Tanker), depending on what active cooling strategy (if any) is selected for that element.

J Calculations Appendix: MLI Outer Cover and Reflector Lookup Table

MLI Outer Cover Material Lookup Table			MLIOUTERLOOKUP		MLIOUTERLOOKUP_ROWS	
Type of MLI Outer Layer	**Model var name**	**Row No.**	**SPEC MASS**	**Abs**	**Em**	**thickness**
units			kg/m2	none	none	m
beta cloth	**BETACLOTH**	2	0.237	0.45	0.8	0.00002
aluminized beta cloth	**ALBETACLOTH**	3	0.271	0.37	0.3	0.00002
thin backed teflon	**THINBTEFLON**	4	0.028	0.1	0.4	0.0000013
thick backed teflon	**THICKBTEFLON**	5	0.55	0.1	0.85	0.0000254
thin coated and backed teflon	**THINCBTEFLON**	6	0.11	0.14	0.6	0.000011
thick coated and backed teflon	**THICKCBTEFLON**	7	0.27	0.14	0.75	0.0000127
thin coated and backed kapton	**THINKAPTON**	8	0.019	0.41	0.5	0.0000013
thick coated and backed kapton	**THICKKAPTON**	9	0.19	0.54	0.81	0.0000127

Lookup table name			*Excel data range name*		*Rows lookup range:*	
MLI Reflector Material Lookup Table			**MLIREFLLOOKUP**		**MLIREFLLOOKUP_ROWS**	
Type of MLI Reflector Material	**Model var name**	**Row No.**	**SPEC MASS**	**Abs**	**Em**	**thickness**
units			kg/m2	none	none	m
thin aluminized kapton	**THINALKAPTON**	2	0.011	0.14	0.05	0.0000076
thick aluminized kapton	**THICKALKAPTON**	3	0.19	0.14	0.05	0.000127
thin goldized kapton	**GOLDKAPTON**	4	0.011	0.3	0.04	0.0000076
thick goldized kapton	**GOLDKAPTON**	5	0.19	0.3	0.04	0.000127
thin aluminized mylar	**ALMYLAR**	6	0.007	0.14	0.05	0.0000007
thick aluminized mylar	**ALMYLAR**	7	0.175	0.14	0.05	0.0175

Figure 16: Appendix: MLI Outer Cover and Reflector Lookup Table. This sheet reconfigures the assumed parameters for the MLI outer cover and reflector, detailing a list of characteristics for multiple options of potential MLI configurations used in boiloff reduction calculations.

K Appendix: Concept of Operations

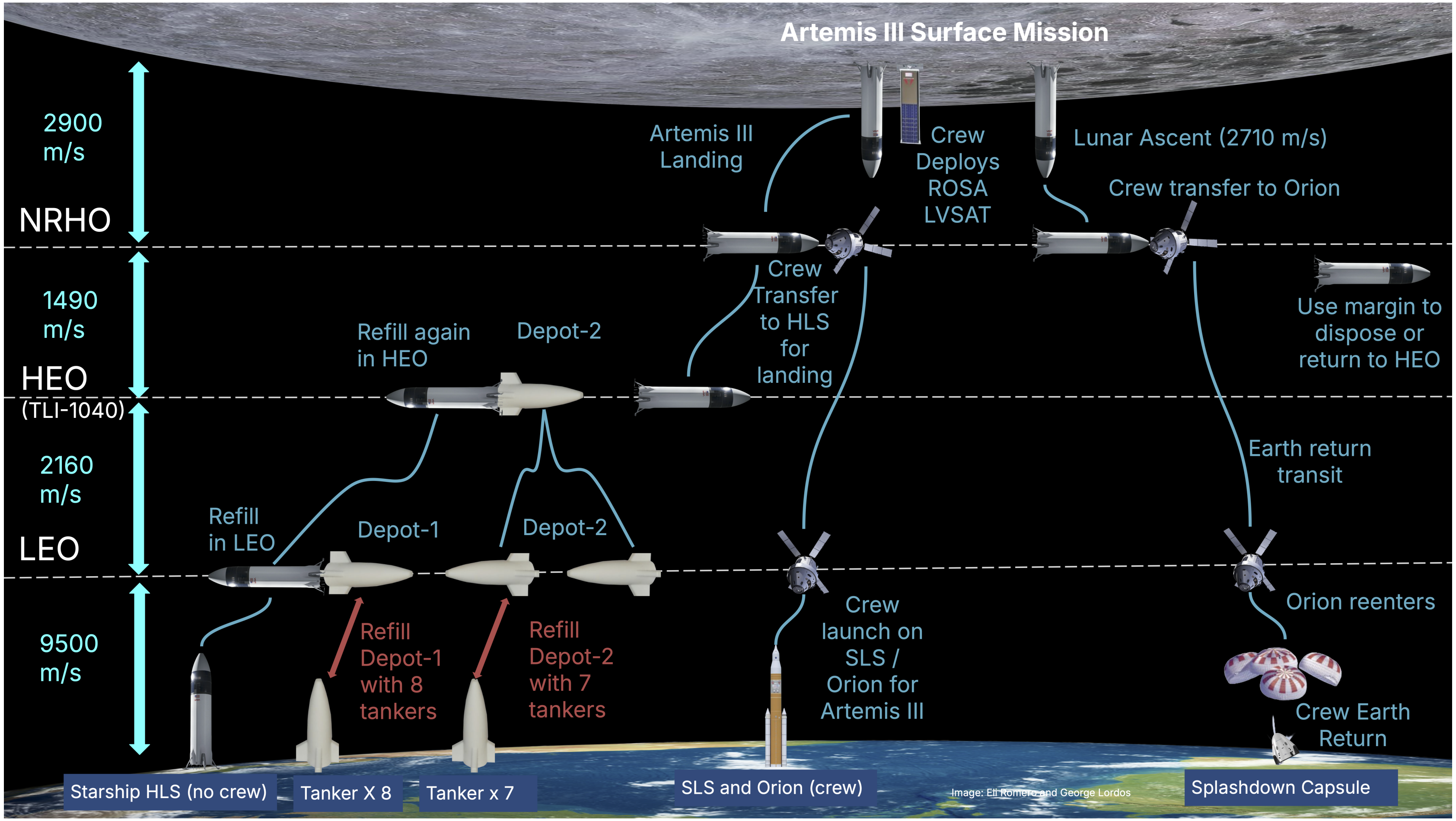

Figure 17: THERMOS Concept of Operations: the key idea is that the HLS is fully refilled in a highly elliptical orbit at approximately TLI minus 1000 m/s, which reduces the required delta V performance for the rest of the mission resulting in higher margin and the opportunity to select and integrate CFM technologies with lower schedule risk.