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# ADVANCED THERMAL INSULATIONS FOR LAUNCH VEHICLES

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Cryopropellants, including liquid hydrogen, methane and oxygen offer distinct advantages in space propulsion. However, their advantages diminish for vehicle upper stages with mission lifetimes of more than a few hours due to propellant boil off. A need exists for lightweight, very high performance thermal insulations which could reduce boil off and allow cryopropellants to be used on longer missions. We report on innovative composite insulations being developed by Ball Aerospace and Quest Thermal Group, intended to meet the requirements of launch vehicles. Cellular Load Responsive Multilayer Insulation (CLRMLI) is being developed under a NASA Phase II SBIR contract, and Vacuum Cell Multilayer Insulation (VCMLI) is also being developed. These insulations use a rugged, lightweight cellular structure that adsorbs a fill gas to high vacuum (<0.1 Pa) by the cold temperatures of the cryopropellant. The cryosorption action in the cellular structure provides internal vacuum in the system without the need for additional vacuum acquisition and maintenance equipment. The cellular structure also provides a natural damage tolerance by compartmentalizing small evacuated cells. We report on the predicted and measured performance of these insulations and compare it to other insulation approaches. We also report on status of the development. The requirements have been defined with input from launch vehicle providers. Conceptual and detailed design has been performed. Design trades have been made on the materials, cell and LRMLI geometry. Thermal and structural modeling have been done. Prototype insulation panels have been built and preliminary tests have been performed.

## INTRODUCTION

Cryopropellants, including liquid hydrogen, methane and oxygen have several advantages over non-cryogenic propellants in space propulsion. They provide high exhaust velocity ($I_{sp}$) when used in chemical and nuclear engines, are non-toxic and can potentially be created by in situ processes on the moon or Mars. However, their advantages diminish for vehicle upper stages with mission lifetimes of more than a few hours due to propellant boil off. The typical time for the transfer orbit from low earth orbit (LEO) to geostationary orbit (GEO) is seven hours, during which significant cryopropellant boil off can occur. Missions beyond GEO have even longer timelines.

Advanced space propulsion systems are a critical need for future NASA deep space missions. High thrust or high specific impulse ($I_{sp}$) engines could enable more effective or even revolutionize space exploration, such as a human mission to Mars. Future propulsion systems might include technologies that accelerate reaction mass electrostatically (Electric Propulsion) or energize a monopropellant thermally (Nuclear Thermal Propulsion, "NTP"). NTP is a high thrust/high $I_{sp}$ propulsion technology, with the NERVA engine demonstrating in ground testing specific impulses of ~850 seconds and thought able to reach 1000 seconds (chemical rockets have a maximum $I_{sp}$ of 450 seconds). NTP would use nuclear

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energy to heat hydrogen propellant to high temperature and expand it through a nozzle to create thrust. There are advancements in technology needed to increase the TRL for NTP, but long term zero boil off of $LH_2$ for long duration missions is still a critical NTP need [1,2].

Insulation on cryopropellant tanks must protect the propellant from boil off, and must provide protection from condensation of air components during the ground and ascent phases. This means the outer surface of the insulation must be > 0 °C to prevent condensation of air components onto the tank surface, or to prevent frost build up from atmospheric water vapor. Preventing this cryo-condensation has been accomplished by adding Spray on Foam Insulation (SOFI) to the tank or using purge gases within a launch fairing. SOFI is problematic however, it is a relatively poor insulation (thermal conductivity is 13 to 25 mW/m-K) which allows high heat leak into the $LH_2$ tank during ground operations and launch ascent, and perhaps more importantly, SOFI is not a structural element. It is not capable of supporting any loads, such as multilayer insulation or thermal shield. SOFI is a relatively poor vacuum insulation compared to multilayer insulation.

One method suggested to prevent condensation on the ground is to pass a purge gas through conventional MLI insulation. The purge gas might be helium for $LH_2$ tanks at 20K, but the highly conductive helium causes high heat leaks during ground hold, and helium is a scarce and expensive resource. Tank insulation layers with an outer temperature above 77K can use a dry nitrogen purge. Heat load into the cryopropellant tank during the ground and ascent phases while purging can be quite substantial. A study done by Ball Aerospace under NASA contract measured the heat added during purge gas venting. The study showed conventional MLI filled with $GN_2$ had a thermal conductivity of 11.5 mW/m-K (220X higher than the baseline MLI heat leak of the 40 layer MLI); and when filled with GHe the conductivity was 27.2 mW/m-K (520X higher than baseline). With a $GN_2$ purge, 214,000 joules/$m^2$ tank surface were added during venting; for a 30 day mission this was 36% of the entire mission MLI heat load. With a GHe purge, 457,000 joules/$m^2$ were added during purge and venting, 78% of the modeled 30 day mission.[3] A need exists for lightweight, very high performance thermal insulations which could reduce boiloff and allow cryopropellants to be used on longer missions.

The Quest Thermal/Ball Aerospace advanced insulation development team has been working on two new insulation concepts that could reduce the heat load into large LH2 tanks during ground and launch ascent phases and greatly reduce the heat load on-orbit. One concept is Cellular Load Reponsive Multilayer Insulation (CLRMLI) and the other concept is Vacuum Cell Multilayer Insulation (VCMLI). These two concepts are similar; CLRMLI offers the advantages of lower in-air heat flux, and higher structural strength, while VCMLI has lower on-orbit heat flux and is lighter.

## CLRMLI CONCEPT

Figure 1 shows the CLRMLI insulation system. Note that the radiant barriers are both separated and structurally supported by the Quest/Ball proprietary Load Responsive discrete spacers. These spacers reduce heat leak from layer to layer, and also support the outer face sheet which has atmospheric pressure acting on it. The structural load bearing capability of the load responsive spacers and the core is able to withstand the pressure difference.

Heat leak through the structure is a function of spacers and core material/cell size/cell wall. Thermal modeling of an 8-layer CLR system estimates the heat fluxes and temperatures at the outer surface shown below in Table 1. A 0.7 inch thick SOFI layer has a 249 W/$m^2$ heat leak, while CLR is predicted to have 27.9 W/$m^2$ heat flux in air, and 8.0 W/$m^2$ in-space.

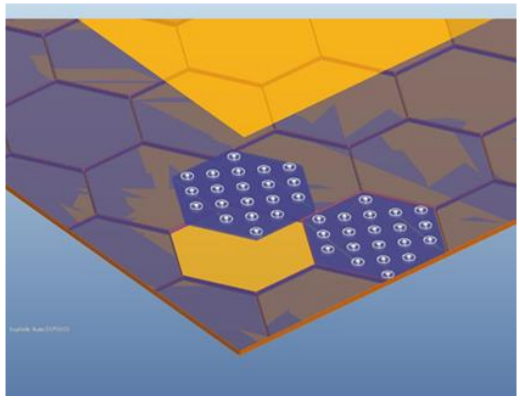

**Figure 1. The CLRMLI insulation system**.

**Table 1. Modeled CLR system performance for in-air and in-space.**

| CLR Updated Calculations | | | | | In Air | | | | In Space | |
|---|---|---|---|---|---|---|---|---|---|---|
| 6" Cell Size | | | | | LN2 Tank | | LH2 Tank | | LN2 Tank | LH2 Tank |
| # of Radiant Lyrs | Cell Area Ratio($m^2$) | Cell wall thickness(mm) | Cell height(m) | Cell A/L | Heat Flux W/$m^2$ | Outer Temp (° C) | Heat Flux W/$m^2$ | Outer Temp(° C) | Heat Flux W/$m^2$ | Heat Flux W/$m^2$ |
| 4 | | | 0.005715 | 1.088364 | 51.6 | -5 | 59.6 | -9 | 10.5 | 11.5 |
| 6 | 0.00622 | .25mm | 0.009525 | 0.653018 | 33.1 | 5 | 38 | 2 | 8.6 | 9.6 |
| 8 | | | 0.013335 | 0.466442 | 24.4 | 9 | 27.9 | 7 | 7.1 | 8.05 |

## CLRMLI DEVELOPMENT PROCESS

A set of 36 x 36 cm panels were planned to be tested. As a 4 layer system is ineffective for the desired outermost surface temperature, six and eight layer prototypes are being built. While the honeycomb cell structure selected for this work has internal cell walls of 0.25 mm, the samples will have large cells cut out from the inside of a standard honeycomb with small 0.6 cm (0.25 inch) cells. The prototypes will effectively have more wall contact area than future, specialized cellular structures that will offer reduced solid conduction. With some assumptions about average cutout area and remaining non-cutout honeycomb area, an average wall to cell area ratio of 0.0236 was estimated. This implies the prototype thermal performance will be near that for a system with 1 mm wall thickness. Taking into consideration the test and equipment requirements, detailed design and fabrication of small scale test articles has been completed to perform the first tests and evaluate the results and determine if the fabrication and test methods are sound. Early test articles were designed to demonstrate the fundamentals of the test approach. The cell structure was machined from polymer in both single cell and multiple cell configurations. The intent was to determine if cryocondensation occurred and were reasonable internal vacuum levels achieved, and to evaluate if vacuum level in the cells were affected by a leak in adjacent cells. The systems will be tested both with and without LRMLI insulation blankets to determine any changes to vacuum level or outer boundary temperature. Preliminary structural analysis was performed to assure that the cell walls would be thick enough to withstand atmospheric loading without collapse. A single cell was used to monitor cryocondensation initially.

After reviewing the test results, it was decided to construct a prototype panel with a single cell. A quad cell with an increased core cell wall area had a higher predicted heat flux. The single cell would give the team the opportunity to produce a higher performing test article. This entailed reducing the honeycomb area to an estimated cell area ratio of 0.016 or a 36% reduction from the original design and a 58% reduction from the second quad cell panel previously tested. The highest risk would be possible delamination of the laminate to honeycomb cell wall. No pressure sensor was installed on this panel to eliminate that possible leak. Based on manufacturing process and visual comparison to other panels, it was believed that the panel was in the 4000 Pa range when built. The single cell panel was mounted to the side of the test tank, foam guards placed around panel, and thermal testing performed. The test sequence for this series of tests was to measure boil off at atmospheric chamber pressure, at 5600 Pa, and at high vacuum (less than 0.1 Pa.)

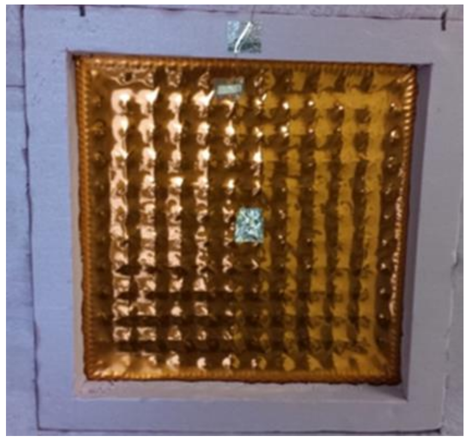

**Figure 2. The 8 layer single cell test installed on tank and ready for test**

With testing completed, the data was reviewed and thermal performance of the CLR system calculated. The approach of determining the CLR heat leak was based on the baseline boiloff data of the foam, subtracting the heat leak associated with the foam face being replaced, and comparing measured heat leak of the CLR panel system to the foam baseline data.

**Table 2. CLR single cell results with liquid nitrogen**

| | **635 torr** | **42 torr** | **$<10^{-3}$ torr** |
|---|---|---|---|
| Foam tank | 115.9 W | 29.6 W | 18.1 W |
| CLR/foam tank | 102.7 W | 28.6 W | 16.5 W |
| CLR heat flux | 46.1 W/m$^2$ | 29.4 W/m$^2$ | 11.4 W/m$^2$ |

Note these heat flux results were for a Phase I CLR test article built with off the shelf, non-optimized components and materials. Nonetheless, these are quite favorable results and achieved two goals of the Phase I program, to demonstrate that CLRMLI works and has substantially better thermal performance than SOFI.

**Table 3. Modeled and measured results**

| | **1 atm** | **vacuum** |
|---|---|---|
| Modeled: CLR prototype with 1 mm core walls (LN2) | 43.5 W/m$^2$ | 11.6 W/m$^2$ |
| Measured: CLR Phase I prototype (LN2) | 46.1 W/m$^2$ | 11.4 W/m$^2$ |
| Effective thermal conductivity (k) | 4.0 mW/m-K | 0.9 mW/m-K |
| Modeled: Possible prototype with 0.25mm core walls on LH2 tank | 27.9 W/m$^2$ | 8.0 W/m$^2$ |

The measured heat flux through the CLR prototype was in excellent agreement with the thermal model predictions. The Quest/Ball team typically finds with discrete spacer based insulation systems like IMLI the SINDA-like thermal models are in good agreement with real world measured results. This predictability is because IMLI and discrete spacer insulation systems are structurally well defined with precisely controlled layer spacing and geometries.

**VCMLI**

Vacuum Cell MLI is similar to CLRMLI in that it has ground thermal performance equal to or better than SOFI and much better on-orbit performance than SOFI. It consists of a layer of low conductivity, self-evacuated honeycomb (Vacuum Cell) next to the propellant tank with one or more external layers of ruggedized IMLI (Figure 3). The honeycomb layer is filled with ambient pressure carbon dioxide ($CO_2$) before being sealed during fabrication. The honeycomb cells also contain a small amount of vacuum getter that cryosorbs the cells to high vacuum when the honeycomb layer is cooled by the cryopropellant. The result is the honeycomb (vacuum cell) layer has a lower effective conductivity than SOFI in air, while having much higher strength.

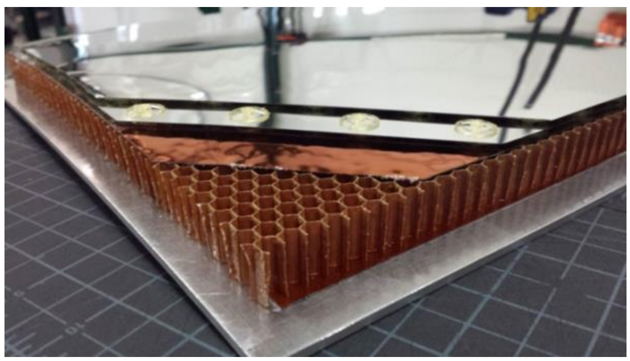

**Figure 3: VCMLI cut-away view**

The external ruggedized IMLI layers provide high performing insulation in space, drastically reducing boiloff during on-orbit loiter, saving significant mass. Table 4 below compares the calculated thermal performance of SOFI and VCMLI designs having the same areal density. The in-space performance of VCMLI provides significant mass savings over SOFI due the reduced propellant boiloff. The in-space performance of VCMLI provides significant mass savings over SOFI for missions as short as 10 hours as shown in figure 4 for a liquid hydrogen propellant tank.

**Table 4: Calculated Thermal performance of SOFI and VCMLI**

| Insulation Type | Insulation Thickness (cm) | Area Density, ($kg/m^2$) | Heat Flux in Air ($W/m^2$), 295 to 20K | Heat Flux in Vacuum (W/m2), 295 to 20K |
|---|---|---|---|---|
| SOFI | 1.9 | 0.79 | 249 | 170 |
| VCMLI | 1.3 | 0.79 | 132 | 7.1 |

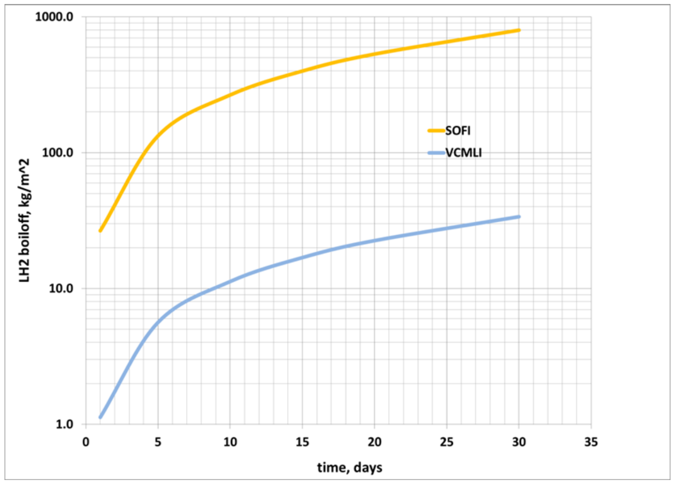


**Figure 4: On-orbit boil off of liquid hydrogen per exposed tank area.**

## VCMLI DEVELOPMENT PROCESS

A phase 1 development of VCMLI was recently performed for a possible application to the liquid hydrogen tank on a launch vehicle upper stage. The honeycomb layer provides the strength to support the IMLI layers during launch, having a compressive strength of 896 KPa (130 PSI) and a shear strength of 379 Pa (55 PSI). The ruggedized IMLI (also known as Launch Vehicle MLI or LVMLI) had already been developed under NASA SBIR contract[4]. A major concern was the structural strength of the LVMLI compared to the aerodynamic loads. Load testing was performed on samples to failure and is compared to the estimated flight loads in Table 5. These strengths, along the honeycomb strength, provide a large margin on the aerodynamic flight stresses seen by the launch vehicle.

**Table 5: LVMLI strengths compared to flight loads for spacers on a typical 2.5 cm grid.**

| | Measured Static Strength, kPa | Calculated Flight Loads, kPa | Margin |
|---|---|---|---|
| Tension | 9.1 | 1.2[5] | 8 X |
| Shear | 10.6 | 0.3[6] | 35 X |
| Compression | 621 | 36[6] | 17 X |

During the NASA SBIR, IMLI samples were tested in a dynamic simulation of the aerodynamic load created by an oscillating shock wave on a launch vehicle. The samples were subjected to 16.5 kPa (2.4 PSI) load created by a nozzle oscillating at a rate of 17.2 Hz. No failures due to the pressure loading were observed.

The purpose of the VCMLI phase 1 development was to design, build and thermally test VCMLI samples. Based on the launch vehicle requirements it was determined that the flight configuration to be tests should consist of a 0.76 cm thick sealed honeycomb layer core with aluminum face sheets and three layers of IMLI. Thermal modeling was performed to verify the expected performance of this configuration. Square sample panels, 36 x 36 cm. , were fabricated with three different configurations:

1. Vacuum Cell honeycomb core with aluminum face sheets, getter and vented to vacuum; to determine core heat flux due to solid conduction and radiation.
2. Vacuum Cell honeycomb core with aluminum face sheets, getter and filled with $CO_2$ gas; to determine core heat flux due to solid conduction and radiation with residual $CO_2$ pressure after cryosorption.

3. Vacuum Cell honeycomb core, face sheets, with $CO_2$ fill, getter and 3 layers of IMLI; to determine heat flux with full flight configuration in vacuum.

The following matrix of test boundary temperatures was devised, of estimated heat fluxes with the various warm and cold warm boundaries. The cold boundary temperatures were based on the temperature at the bottom of a liquid hydrogen tank (20.4 K), the top of a typical liquid hydrogen tank (55.4 K) and the bottom of a liquid oxygen tank (89.8 K). The heat fluxes were estimated using a preliminary thermal model.

**Table 5: Test Matrix with Estimated Heat Fluxes in Watts/meter$^2$**

| | 294.8 K Warm Boundary | 333.7 K Warm boundary | 372.6 K Warm boundary |
|---|---|---|---|
| 89.8 K, Cold boundary | 8.87 | 13.35 | 19.75 |
| 55.5 K, Cold boundary | 9.15 | 13.66 | 20.00 |
| 20.4 K, Cold boundary | 9.37 | 13.85 | 20.22 |

**Test Configuration**

The honeycomb was bonded to the bottom close out foil with getter and was vacuum baked out. The top sheet was then bonded on in a $CO_2$ environment. The sample was bonded to aluminum heat spreader and mounted to a Gifford-McMahon cryocooler cold head in a vacuum chamber. All sides were insulated with integrated multilayer insulation (IMLI) to reduce parasitic heat leak into test. IMLI was used because its heat flux had been well characterized in previous programs[7]. Two silicon diode thermometers where mounted the on cold side, warm side and outer VC layer for VCMLI panel. The test boundary temperatures were a goal for the testing and not an exact specification. In order to meet these boundary temperatures exactly would have significantly increased the test time, as it took several hours for the warm and cold boundaries to come to a steady state temperature. A film heater was mounted on the top layer. The measured heat flux was the amount of power required to hold warm boundary temperature, after being adjusted for the heat losses through the IMLI to the chamber environment.

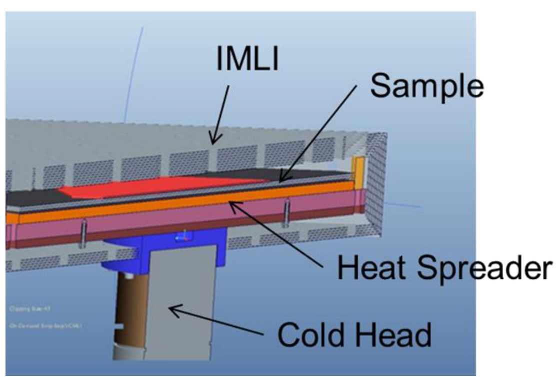


**Figure 5. Cross section rendering of the test configuration.**

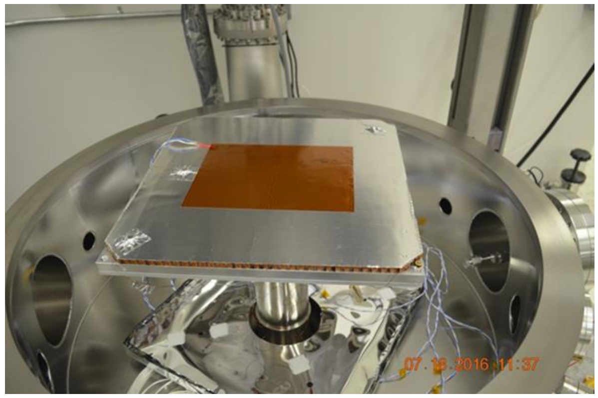


**Figure 6. Sample 2 installed on cryocooler cold head, with temperature sensors and film heater, in vacuum chamber lid removed.**

## Test Results

The results of the test of the first sample approximately matched the thermal model and are shown below. The test data showed the conductivity and emissivity of honeycomb had to be updated and the model adjusted.

**Table 6: Vented Vacuum Cell (Sample 1) Test Results**

| T cold average measured | T warm average measured | Heater power measured | Heater power initial model | Measured vs. initial model | Heater power adjusted model | Measured vs. adjusted model |
|---|---|---|---|---|---|---|
| K | K | Watts | Watts | % | Watts | % |
| 89.17 | 297.91 | 37.28 | 37.86 | -1.54 | 37.11 | 0.48 |
| 55.64 | 294.70 | 38.18 | 37.87 | 0.83 | 38.41 | -0.60 |
| 28.40 | 294.53 | 38.59 | 38.90 | -0.81 | 40.16 | -4.06 |
| 33.31 | 331.89 | 54.59 | 57.63 | -5.57 | 54.65 | -0.11 |
| 57.53 | 333.07 | 52.39 | 58.62 | -11.88 | 53.61 | -2.33 |
| 90.45 | 333.36 | 50.89 | 57.42 | -12.84 | 51.30 | -0.82 |
| 93.34 | 368.13 | 75.39 | 82.83 | -9.86 | 68.99 | 8.50 |
| 57.35 | 372.41 | 79.13 | 87.96 | -11.17 | 74.26 | 6.15 |
| 39.33 | 372.41 | 78.18 | 88.72 | -13.48 | 75.48 | 3.45 |
| | | | average: | -7.37 | | 1.18 |

The test results of the second sample are shown below. The measured effective conductivity of the vacuum cell layer, is approximately half that of the SOFI used on launch vehicles[8]. The thickness of the sample is 46% of the typical SOFI thickness of 1.8 cm so the total flux is 30% more. This test approximates the performance of VCMLI in air. However, the outer LVMLI layers will provide some insulating effect in air, bringing the VCMLI flux close to the SOFI flux. The sample heat fluxes were higher than predicted by the model because of the residual $CO_2$ gas pressure in the cells, estimated to be between 0.65 and 1.7 millitorr depending on the cold boundary temperature.

**Table 7: Sealed Vacuum Cell (sample 2) Test Results**

| T cold measured | T warm measured | Vacuum Cell measured effective conductivity | SOFI average conductivity | Vacuum Cell conductivity/SOFI conductivity | Vacuum Cell measured flux | SOFI calculated flux | Vacuum Cell flux/ SOFI flux |
|---|---|---|---|---|---|---|---|
| **K** | **K** | **watt/m*K** | **watt/m*K** | | **watt/m$^2$** | **watt/m$^2$** | |
| 90.36 | 296.05 | 0.011 | 0.022 | 0.50 | 295 | 249 | 1.18 |
| 55.91 | 296.39 | 0.010 | 0.020 | 0.48 | 302 | 267 | 1.13 |
| 29.73 | 296.68 | 0.010 | 0.019 | 0.55 | 363 | 276 | 1.32 |
| 31.88 | 319.26 | 0.012 | 0.019 | 0.63 | 455 | 308 | 1.48 |
| 57.13 | 327.19 | 0.014 | 0.021 | 0.65 | 482 | 311 | 1.55 |
| 91.57 | 327.75 | 0.011 | 0.022 | 0.52 | 356 | 293 | 1.22 |
| | average: | 0.011 | 0.021 | 0.55 | 376 | 284 | 1.31 |

The test results of the third sample are shown below. The calculated parasitic heat leak through IMLI was subtracted from the raw measured power using the known heat flux of IMLI. The model also included the heat flux due to residual $CO_2$ determined in the sample 2 testing. The average measured flux of the VCMLI in vacuum is 10% of the SOFI flux; this shows the significant performance advantage of VCMLI for on-orbit conditions.

**Table 8: VCMLI test results in a flight like configuration (sample 3)**

| T cold measured | T warm measured | Heater power measured | Heater power model | Heater power, Measured vs. model | VCMLI measured flux | 1.8 cm SOFI calculated flux | VCMLI flux/ SOFI flux |
|---|---|---|---|---|---|---|---|
| **°F** | **°F** | **watt** | **watt** | **%** | **watt/m$^2$** | **watt/m$^2$** | |
| 89.41 | 294.39 | 1.78 | 1.82 | -2.57 | 12.2 | 153 | 0.08 |
| 56.66 | 294.31 | 1.86 | 1.92 | -2.99 | 12.8 | 164 | 0.08 |
| 21.53 | 294.25 | 1.89 | 2.00 | -5.68 | 13.0 | 170 | 0.08 |
| 21.70 | 331.62 | 2.63 | 2.81 | -7.00 | 18.1 | 203 | 0.09 |
| 56.74 | 331.64 | 2.60 | 2.74 | -5.28 | 17.9 | 196 | 0.09 |
| 89.49 | 331.72 | 2.53 | 2.64 | -4.42 | 17.4 | 186 | 0.09 |
| 20.79 | 371.32 | 4.08 | 3.98 | 2.51 | 28.1 | 230 | 0.12 |
| 56.88 | 371.19 | 4.16 | 3.89 | 6.44 | 28.7 | 223 | 0.13 |
| 89.67 | 371.43 | 3.98 | 3.80 | 4.52 | 27.5 | 212 | 0.13 |
| | average: | 2.84 | 2.85 | -1.61 | 19.5 | 193 | 0.10 |

## CONCLUSIONS

New insulation concepts Cellular Load Responsive MLI (CLRMLI) and Vacuum Cell MLI (VCMLI) were designed, modeled, prototypes built, tested and proven to be feasible and much superior insulation to SOFI both in-air and in-vacuum. CLRMLI is a novel technology with a self-evacuating cellular core containing Load Responsive MLI layers. This new form of insulation uses cryosorption to self-evacuate when in contact with cryogenic propellant tanks, allowing high thermal performance in-air and in-space. CLRMLI test articles had a measured heat flux of 46.1 $W/m^2$ (1.5 cm thick, 77 K cold, 295 K warm, 1 atm), significantly lower than 1.8 cm SOFI heat flux of 276 $W/m^2$ in air. CLRMLI would be excellent insulation for $LH_2$-powered High Altitude Long Endurance unmanned aerial vehicles with a measured heat flux of 29.4 $W/m^2$ at 65,000'. CLRMLI dramatically outperforms SOFI with a measured heat leak of 11.4 $W/m^2$ in-vacuum compared to over 171 $W/m^2$ for SOFI. The Phase I program successfully demonstrated CLRMLI is a feasible and attractive insulation for new launch vehicle platforms and $LH_2$ or LNG powered aircraft. CLRMLI can reduce cryopropellant boil off losses for SLS cryogenic upper stages, and may help NASA reach Reduced or Zero Boil off goals needed for long duration missions. CLRMLI technology was proven feasible, of very high performance, advanced to TRL 4, and in Phase II CLRMLI will be further technically matured, challenges found in the Phase I prototype resolved, and prepared for integration into new launch vehicles for use by NASA.

VCMLI also uses cryosorption to self-evacuate when in contact with cryogenic propellant tanks, and has outer layers of MLI allowing high thermal performance in-air and in-space. VCMLI measured strength has a large margin on the estimated flight loads. The VCMLI thermal test results were consistent with the modeling done prior to testing and were useful in updating the thermal models. The VCMLI getter was effective in absorbing the $CO_2$ fill gas to a pressure that allowed good thermal performance (approximately 0.1 Pa). The getter performance appeared to be a function of the getter temperature, as expected. The measured Vacuum Cell conductivity is 46 to 43% less than the SOFI conductivity The measured VCMLI heat flux in vacuum is 10% of the calculated 1.8 cm thick SOFI flux. VCMLI is at a TRL 4: an enabling level of performance of a breadboard prototype in a laboratory environment

## ACKNOWLEDGMENTS

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