Overview of NASA's Thermal Control System Development for Exploration Project

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The now-cancelled Constellation Program included the Orion, Altair, and Lunar Surface Systems project offices. The first two elements, Orion and Altair, were planned to be manned space vehicles while the third element was much more diverse and included several sub-elements. Among other things, these sub-elements were Rovers and a Lunar Habitat. The planned missions involving these systems and vehicles included several risks and design challenges. Due to the unique thermal operating environment, many of these risks and challenges were associated with the vehicles' thermal control system. NASA's Exploration Technology Development Program (ETDP) consisted of various technology development projects. The project chartered with mitigating the aforementioned thermal risks and design challenges was the Thermal Control System Development for Exploration Project. These risks and design challenges were being addressed through a rigorous technology development process that was planned to culminate with an integrated thermal control system test. Although the technologies being developed were originally aimed towards mitigating specific Constellation risks, the technology development process is being continued within a new program. This continued effort is justified by the fact that many of the technologies are generically applicable to future spacecraft thermal control systems. The current paper summarizes the development efforts being performed by the technology development project. The development efforts involve heat acquisition and heat rejection hardware including radiators, heat exchangers, and evaporators. The project has also been developing advanced phase change material heat sinks and performing a material compatibility assessment for a promising thermal control system working fluid. The to-date progress and lessons-learned from these development efforts will be discussed throughout the paper.

Nomenclature

Q_{max}/Q_{min} = heat rejection turndown ratio
m = mass of phase change material (PCM) (kg)
E = PCM energy storage requirement (J)
h_f = PCM heat of fusion (J/kg)

I. Introduction

IN early 2004, President Bush announced a bold new vision for space exploration. One of the goals included in this vision was a human return to the moon by 2020. In response to this vision, NASA established the now-defunct Constellation Program (CxP), which included several project offices. In addition to this program office, NASA established a separate office chartered with advancing technologies to support these future exploration missions. The technology development program was the Exploration Technology Development Program (ETDP) and its primary customer was the aforementioned Constellation Program. ETDP was responsible for developing technologies capable of mitigating key CxPO risks. Specifically, ETDP personnel were required to advance technologies to a Technology Readiness Level (TRL) of six prior to the customer's Preliminary Design Review (PDR).

ETDP consisted of 24 separate projects ranging from software development to entry, descent, and landing. One of the projects included in the program's portfolio is the Thermal Control System Development for Exploration. This project, herein referred to as the Advanced Thermal Project, was chartered with mitigating thermal risks and design challenges for various element within the Constellation Program.

For several years, the Advanced Thermal Project was focused on developing technologies for three of the different Constellation elements. These elements included Orion, Altair, and Lunar Surface Systems (LSS). Orion was planned to be the manned capsule capable of transporting crew to the International Space Station (ISS) and Lunar orbit. The second Advanced Thermal Project customer was Altair. Altair was being designed to be the Lunar lander element that was responsible for transporting crewmembers to the Lunar surface and supporting them while living on the Lunar surface for short mission durations. The third, and final, customer was the Lunar Surface Systems (LSS) project. The Advanced Thermal Project was developing technologies for two elements within the LSS project. The first element was the Lunar Habitat which was planned to serve as the long-term habitat for astronauts while

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living on the Lunar surface. The second element was the Lunar Electric Rover (LER), which was a pressurized rover designed to provide life support and transport crewmembers on the Lunar surface.

It is readily apparent that the elements/projects being supported by the Advanced Thermal Project are quite different than one another. They range in function from a crew capsule, to a lander, to a habitat and a roving element. In many instances, the thermal technologies being developed for one of these elements were equally applicable for another element/vehicle. Even though CxP has been cancelled, the thermal control technology development effort is being continued. The continued development of these technologies is easily justified by the breadth of vehicles to which they are relevant. For instance, the radiator technologies required for Orion are similar to those needed for both Altair and the Lunar Habitat. Therefore, it is not a stretch to say that these technologies will be required for future vehicles (both crewed and unscrewed). Thermal is the ultimate cross-cutting discipline and some form of thermal control is required for every spacecraft and vehicle designed to operate in the extreme thermal environment of space. As a result of these observations, the thermal technology development efforts have continued despite the cancellation of the Constellation Program.

The current paper will describe the preferred technology development process being followed to advance the various technologies to a TRL six. The current document will also briefly introduce the various technology development efforts. More detail on each of these efforts is included in other papers[a, b, c, d, e, f, g, h, i, j, k] at the current conference. Interested parties should reference these papers for a more exhaustive description of a particular development task.

II.   Technology Development Process

The project's technology development process typically begins with the hardware component possessing a technology readiness level of two or three. The first step in the process is often the completion of coupon-level bench top tests. The objective of these tests is to better understand the basic physics and the critical development challenges associated with the technology. This phase is often followed by a design and analysis cycle focusing on addressing the previously identified development challenge(s). The design and analysis cycle finishes with detailed drawings for an Engineering Development Unit (EDU), which is a scaled-down model addressing the key technology issues. The EDU goes through a rigorous test program and new, correlated, thermal models are developed based on the previous test results. After completing the initial EDU tests and the subsequent thermal models, another design and analysis cycle is performed. During this second design and analysis cycle, the EDU performance data is assessed, performance improvements are recommended, and detailed requirements are defined. The result of the second design and analysis cycle is the generation of drawings and the fabrication of prototype hardware. The prototype hardware is then tested as a stand-alone entity to verify its performance.

The technology development process finally culminates with an integrated thermal test. The integrated test is performed in an environment relevant to the supporting vehicle (i.e. Altair) and simultaneously includes all of the previously developed hardware prototypes. The primary difference between the integrated test and the previous prototype tests is that the integrated test involves all of the developed hardware as an integrated Thermal Control System (TCS). At the conclusion of the integrated test, the developed technologies possess a TRL six, which is defined as system/subsystem model or prototype demonstration in a relevant environment (ground or space).

III.   Thermal Control Overview and Project Content

In its most basic form, thermal control is the maintenance of all vehicle surfaces and components within an appropriate temperature range throughout the many mission phases despite changing heat loads and thermal environments. For satellites this requires that the thermal control system must maintain all of the equipment within its operating and/or storage temperature range. Similar to the system for satellites, the thermal control system for human-rated vehicles must also maintain all of the equipment within the appropriate temperature ranges. In addition to component-level temperature maintenance, the crewed spacecraft's thermal control system must also safely maintain the internal cabin temperature within the proper temperature range to ensure both crew survivability and comfort.

An effective thermal control system must provide three basic functions to the vehicle/system design: heat acquisition, heat transport, and heat rejection. Heat acquisition is the process of acquiring excess thermal energy from various heat dissipating components including electronics, avionics, computers, and metabolic loads from the crewmembers for a manned mission. Heat acquisition is typically accomplished using a myriad of hardware components including, but certainly not limited to, coldplates, air/liquid heat exchangers, and liquid/liquid heat exchangers. As the title implies, heat transport is the process of moving the now acquired energy to another location within the vehicle/system. Heat transport can be accomplished using active means such as a pumped fluid loop, but can also be performed using more passive methods such as a simple conductive path and/or heat pipes. Some combination of the active and passive methods is typically employed for a manned-vehicle, while many small satellites use either conductive paths or heat pipes. The third, and final, thermal control system function is heat rejection which is the process of rejecting the thermal energy to the surrounding environment. Heat rejection is performed using radiators, evaporators, and/or sublimators. The heat rejection function can also be aided by the use of phase change material heat sinks. The technology development project currently includes several technical tasks spanning the three critical functions of an effective thermal control system.

IV.   Heat Transport

The overwhelming majority of United States' manned space vehicles have used a pumped fluid loop as the primary means of thermal control during the mission. The fluids in these systems have ranged from an ethylene glycol/water mixture on Apollo to the Orbiter with water (internal loop) and Freon[®] 21 (external loop). The Orion TCS design included a 50/50 (by mass) mixture of DOWFROST[TM] HD and water. This particular fluid formulation had never been used for a manned vehicle so there was very little long duration stability data for this fluid in a flight-like TCS.

Because of the limited data, the technology development project designed and completed a fluids life test. The fluid loop was designed to include all of the wetted materials included in the Orion TCS design. This test was originally designed to last for a period of ten years to provide meaningful data for the Lunar Habitat which was planned to operate for this duration. The test was terminated after only two months because the system filters become clogged and the system was rendered inoperable. Although oversized for the fluid loop design, the pump was not able to maintain the specified flowrate with the increased system pressure drop. Post-test analysis of the fluid showed that the coolant pH increased from 10.1 at the beginning of the test to approximately 12.2 at the end. Orion personnel were tasked with identifying the root cause of the failure and recommending another fluid for the vehicle design. After several months of analysis and bench-top tests, the root-cause was determined to be the fluid's incompatibility with the large ratio of wetted aluminum surface area to fluid volume. The specifics of the fluid failure can be found in another paper at this conference[a].

In parallel with this failure analysis, a second generation fluids life test cart was designed and fabricated so that the alternative fluid could be tested. Although based on the first generation fluids life test, the updated test cart included several improvements. These improvements included a redesign of the surface area module to accurately represent the updated wetted surface area to fluid volume ratio. An addition surface area module was also fabricated and included in the test cart. This second surface area module included nickel fins rather than aluminum. Finally, the test design incorporated gas vents at various locations around the test loop. The vents would be used to sample any gas that is generated within the test loop.
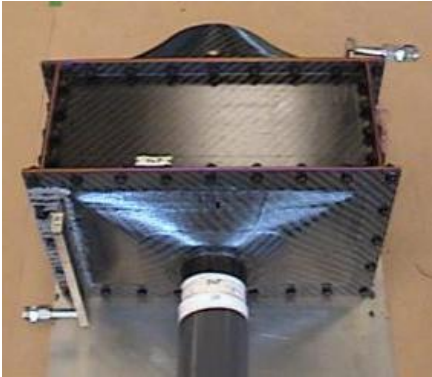
The aforementioned failure analysis and subsequent fluid selection effort resulted in the selection of Amsoil ANT as the ideal working fluid Orion and the fluids life test. A 50/50 mixture (by mass) of the selected fluid and water was created and placed into the test cart. The second generation life test was officially started on 03 March 2011. The baseline pH for this test was between 9.1 and 9.3. Thus far, the system has shown no evidence of an increased pressure drop or fluid incompatibility.

V.   Heat Acquisition

As mentioned above, an effective thermal control system must accomplish three basic functions: heat acquisition, heat transport, and heat rejection. One of these functions is the acquisition of excess thermal energy from the cabin air and other heat-generating devices. There are two technologies being developed within this category and each of them seeks to advance the state of the art by reducing hardware mass and volume.

A. Composite Heat Exchangers

Heat exchangers are traditionally used to transfer energy from one fluid loop to a second fluid loop. The use of composites for these types of hardware is an attractive alternative to traditional metallic heat exchangers. Carbon-based composites have very high thermal conductivities, making them more effective heat transfer devices. In addition, composites also have a high strength- to- mass

Figure 1.   Composite air/liquid heat exchanger development unit.

ratio, which has the potential of reducing the hardware mass. The project has previously explored the benefit of using these advanced materials to fabricate radiators. However, heat exchanger construction is significantly different and requires a more detailed investigation.

As reported in last year's project overview paper[a], a composite heat exchanger engineering development unit was successfully designed and fabricated. A second heat exchanger was then developed incorporating several design improvements based on lessons-learned from the engineering development unit. The second heat exchanger was included in a thorough test program. The primary objective of this test program was to generate empirical data to be used in a subsequent model correlation effort. The test program was successfully performed throughout the winter of 2010 as described in another paper at this conference.[q]

The correlated thermal model was used to predict the mass of a composite air/liquid heat exchanger designed to meet the performance requirements of an existing mass-optimized, metallic heat exchanger. This type of heat exchanger design represents the state-of-the-art in heat exchanger technology. The metallic heat exchanger was designed to transfer approximately 3.4 kW for a prescribed set of inlet conditions (mass flowrate and inlet temperature). The results from the correlated thermal model suggest that an air/liquid composite heat exchanger designed to meet the same performance specifications would have a mass of 29 pounds. This mass estimate is approximately 38% lower than the comparable state-of-the-art heat exchanger alluded to above. In the coming months, project personnel will use the now-correlated thermal model to design a composite heat exchanger capable of meeting the same performance specifications imposed on the metallic heat exchanger.

B. Microchannel Heat Exchanger

In collaboration with Pacific Northwest National Laboratory (PNNL), the technology development project has developed a microchannel liquid/liquid heat exchanger.[w] This heat exchanger was designed to match the performance requirements for a mass optimized heat exchanger fabricated to be used on NASA's crew return vehicle (X-38). Both the microchannel and baseline heat exchangers are shown in Figure 3. The microchannel heat exchanger mass was approximately 24% less than the comparable SOA heat exchanger. In addition to being lighter, the microchannel heat exchanger is noticeably smaller with a core volume of approximately 300 cm^3, which represents a volumetric decrease of 60%.

To achieve these mass and volume savings, the flow passages in the microchannel heat exchanger are significantly smaller than those located within the SOA heat exchanger. Project personnel were concerned that these smaller flow passages would be more susceptible to fouling a clogging thereby affecting the performance of the heat exchanger. Resultantly, the microchannel heat exchanger has been subjected to an ongoing life test. The heat exchanger is constantly exposed to the specified inlet conditions and the resulting performance is evaluated. The data analysis has been focused on overall heat transfer as well as pressure drop on both sides of the heat exchanger. To date, the heat exchanger has been exposed to the required inlet conditions for a total of 4200 hours and there has not been any noticeable degradation in the heat exchanger performance.[e]
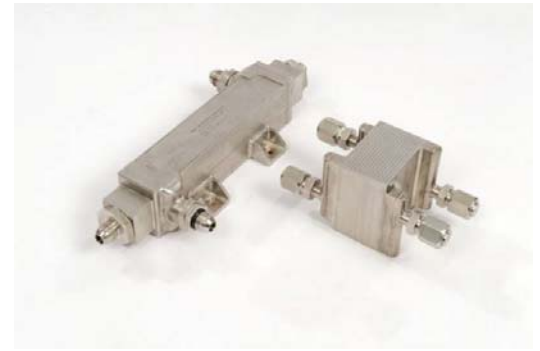

Figure 2. Baseline X-38 liquid/liquid heat exchanger (left) and PNNL's microchannel heat exchanger (right).

VI. Heat Rejection

The third and final critical function for an effective TCS is heat rejection. As the name implies, heat rejection is the process of rejecting a vehicle's waste heat to the local environment. This function is typically accomplished using radiators, but evaporators or PCM heat sinks can also be used to reject energy. All three of these heat rejection devices are being developed as part of the technology development project.

A. Transient Sublimator

A spacecraft orbiting a planetary body may be subjected to a wide range of thermal environments throughout the planetary orbit. This wide range of induced environments can be caused by a varying infrared load or by a constantly changing solar load due to the vehicle entering and exiting the planet's shadow cone. One extreme example where both the infrared and solar load incident upon a vehicle changes significantly throughout the orbit occurs for a vehicle orbiting the Earth's moon. Because the moon does not have an atmosphere, the Lunar surface temperature varies significantly as shown in Figure 3.
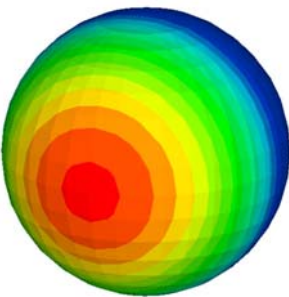
In the preceding Figure, the hottest portion of the lunar surface corresponds to the point directly aligned with the sun (subsolar point). The maximum surface temperature is approximately 400 K while the minimum temperature is less than 100 K on the dark side. This extreme surface variation combined with a spacecraft entering and exiting the shadow cone results in a large swing in radiator sink temperatures while the vehicle is operating in low-lunar orbit (LLO). The large sink temperature variations are problematic because it is impractical (sometimes even impossible) to use a radiator as the sole means of heat rejection during an orbit in which the sink temperature varies significantly over a short period of time. Resultantly, a vehicle subjected to this type of design challenge must include a Supplemental Heat-Rejection Device (SHReD) in the vehicle's TCS design. Figure 4 shows an example of the variability of a vehicle's heat-rejection capability using only radiators for a Lunar orbit beta angle of 0 degrees and an orbital altitude of 100 km.

The orange curve in Figure 4 represents the heat-rejection requirement for the TCS. In this example, the vehicle heat-rejection requirement is constant throughout orbit. The red curve represents vehicle radiator capability assuming a constant average radiator temperature while the green curve shows the SHReD requirement.

Figure 4 clearly shows that the radiator capability varies throughout the lunar orbit. This variability can be explained by studying Figure 3 and understanding that a vehicle in a low beta angle orbit will periodically enter and exit the Moon's shadow. It is apparent that the IR backload incident upon the vehicle will change throughout the orbit due to the changing lunar surface temperature. In addition, the incident solar load

Figure 3.Spatial distribution of the Lunar surface temperature.
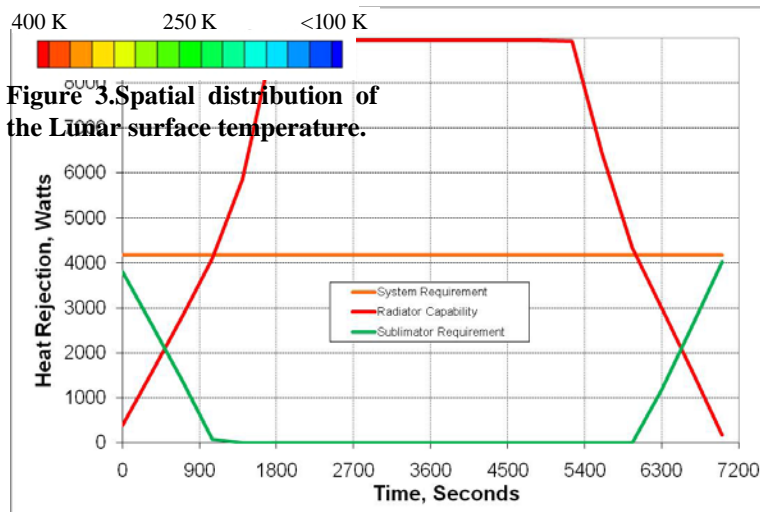
Figure 4. Vehicle heat-rejection requirement (orange), radiator capability (red), and supplemental heat-rejection requirement (green) as a function of mission time in LLO.

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will also vary throughout the orbit. The combination of these effects leads to a wide sink temperature variation in LLO resulting in changes to the radiator heat rejection capability.

In order to maintain the system setpoint temperature, the TCS must be capable of rejecting the full vehicle heat load (4.2 kW in this example) throughout the entire orbit. Therefore, the SHReD must dissipate the difference between the heat-rejection requirement and the radiator capability, which is shown by the green curve in Figure 4.

The selection of proper SHReD depends on the duration of the mission phase during which it will be used. For short-duration missions, an evaporative heat sink would be used as the SHReD. A PCM heat exchanger would likely be selected for longer-duration missions.

Unfortunately, a spacecraft TCS has never been designed to use a sublimator in this cyclical fashion. There are two potential problems associated with using a sublimator as a SHReD. Current sublimators have a minimum heat load requirement that would result in a poor orbit-averaged feedwater efficiency (or utilization). Additionally, there is concern that the hardware may burst during periods when the sink temperature is relatively cold and the heat load on the sublimator is low or possibly nonexistent.

A test program has been developed to determine the most mass efficient method of using a sublimator as a supplemental heat rejection device. This test program was performed using a sublimator developed in-house at NASA Johnson Space Center. The primary focus of this test program was to quantify how feedwater consumption efficiency is affected by the choice of operational scenario.[r] Unfortunately, the empirical results of the previous test program were inconclusive and further investigations are currently being performed. The ongoing test points were based on lessons-learned from the previously completed tests. These current test points will help to determine whether SOA sublimators are sufficiently designed to be used as a SHReD or whether additional technology development efforts must be performed to make this hardware a viable choice for cyclical heat rejection applications.

B. Variable Heat Rejection Radiators

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One of the most significant design challenges encountered when developing a vehicle's radiator design is liquid freezing within the coolant lines attached to the radiator surface. Typically, radiators remove energy from the coolant flowing through the radiator and reject the energy to space. Radiator surface area is a key factor contributing to the rate at which energy is rejected into space. Generally speaking, radiators are sized for the maximum heat load in the warmest continuous thermal environment. It is necessary to design a radiator system with a large surface area to dissipate a high heat load in a relatively warm environment. However, when that same large radiator is required to dissipate a much lower heat load in a cold environment, the surface temperature dramatically decreases. If
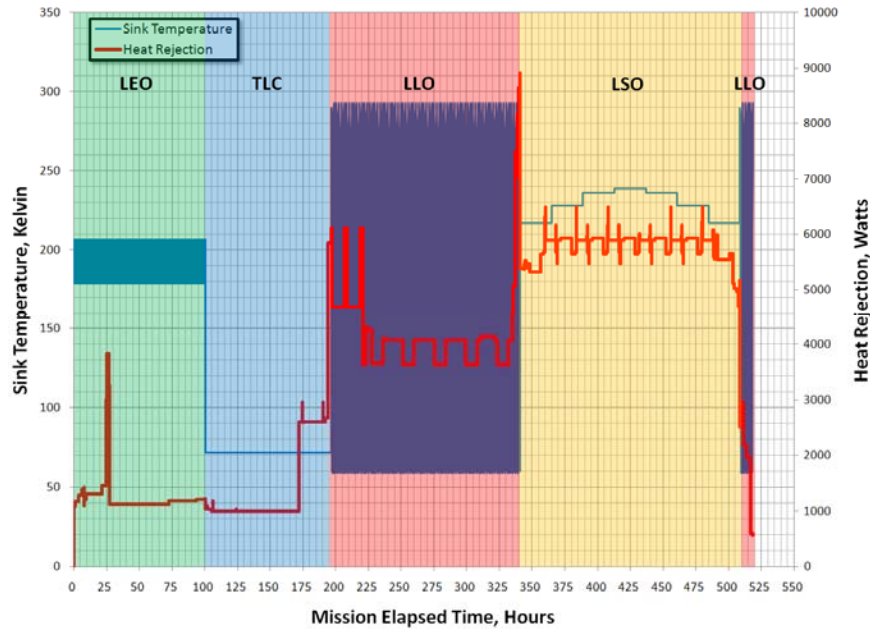
Figure 5.   Sample thermal control system requirements based on latest Altair vehicle design. The vehicle's heat-rejection requirement and radiator sink temperatures plotted as a function of mission elapsed time (MET).

the radiator is not correctly designed, this decreased surface temperature can lead to fluid freeze within the radiator coolant lines which can lead to setpoint control difficulties.he recently designed Altair Lunar Lander provides a good example describing this problem. The vehicle's radiator must be designed to dissipate a high heat load during Lunar surface operations but must also be capable of operations at very low heat loads during Trans-Lunar Coast (TLC). For this particular mission architecture, TLC is extremely cold because the Altair radiatorswill be shadowed from the sun during the entire mission phase. The requirement to operate at both a high and a low load is referred to as the system turndown ratio (Q_{max}/Q_{min}). The expected turndown ratio requirement for future vehicles is an order of magnitude greater than that for the Apollo condition. The challenge associated with the fluid freezing within a radiator can be exacerbated by the fact that the low heat rejection requirement occurs while the vehicle is situated in the cold thermal environment. The previously described design requirements for the latest Altair design are depicted in Figure 5.

The preceding figure shows both the radiator sink temperature and the vehicle's heat-rejection requirement as a function of MET. The radiator sink temperature, which is defined as the temperature that would be achieved for an adiabatic body with similar optical properties, is shown along the left, vertical axis. Due to the resolution of the abscissa scale, it is difficult to discern the transient nature during the planetary orbits. However, the radiator sink temperature repeats during each orbit period. The orbit period during LEO and LLO is approximately 1.5 and two hrs, respectively. The sink temperature varies from approximately 70 K during TLC and is as high as 290 K during LLO. As mentioned above, the radiator system is sized for lunar surface operations, which corresponds to a sink temperature range of 220 K to 240 K. The sink temperature corresponding to the Lunar surface phase of the mission is calculated for a horizontal radiator with the seven day mission phase symmetric about Lunar noon (when the sun is directly overhead).

The vehicle's heat-rejection requirement shown along the right, vertical axis is as low as 1,000 W during TLC. However, there are times during the mission in which the heat load approaches 9,000 W. For the design point, which is Lunar surface operations, the heat-rejection requirement is approximately 6 kW.

For most vehicles, a high turndown ratio requirement usually results in a two-loop TCS architecture. A two-loop architecture is advantageous because the external loop can include a fluid with a very low freezing temperature. Unfortunately, fluids with low freezing points (e.g., Freon®, ammonia, etc.) are typically toxic and cannot be located inside the pressurized volume due to crew safety concerns. The biggest drawback to a dual-loop system is the increased mass associated with both loops. The addition of a second loop requires several extra hardware components and a slight increase in required radiator area due to inefficiencies associated with an interchange heat exchanger. Orion originally baselined a single-loop system, but quickly switched to a two-loop system. The addition of the second loop increased the TCS mass by approximately 18%.[7]

The Advanced Thermal Project is seeking to use technology development to overcome this extremely difficult design challenge. The project is pursuing development of three separate variable heat-rejection radiator technologies. These technologies include variable

emissivity electrochromics,[8] a digital radiator,[9] and a freezable radiator[10] design. In the preceding years, the project has designed and tested thermal vacuum test samples or bench-top apparatuses to evaluate these technologies.

The basic premise behind the concept of an electrochromic radiator is to actively vary the infrared emissivity of the radiator panel to prevent the fluid within the radiator from freezing. For example, during periods of low heat rejection requirement and/or cold thermal environments, the surface emissivity of the radiator would be decreased. Conversely, when the heat rejection requirement is increased and/or the thermal environment is relatively warm, the surface emissivity would be decreased. In the past, project personnel have completed several thermal analyses to quantify the required emissivity turndown ratio (e_{max}/e_{min}) for various operational scenarios. In addition, a couple of thermal vacuum tests were performed to assess the current state-of-the-art in electrochromic technology. The project continues to work with industry to improve the performance of electrochromic devices.

Fundamentally, the digital radiator attempts to mitigate the risk of fluid freezing by selectively evacuating fluid from the radiator tubes during low load (and/or cold environments) periods and refilling the tubes when the heat rejection requirement is increased. The draining and refilling of the tubes is accomplished using a series of latching valves and a gear pump. Presently, the project is designing and preparing to execute a digital radiator bench-top test focused on preventing fluid coalescence in a radiator tube after the tube has been drained and prior to refilling the tube. Previous bench-top tests have shown that it is likely necessary to rely on surface tension to prevent fluid coalescence from blocking a tube during the refilling process.

The concept of a freezable radiator is rather innovative in its simplicity. In this application, the radiator is designed in such a manner to preferentially and predictably allow the fluid within the radiator panel to freeze. Previous experience with freezable (or stagnating) radiators has highlighted the difficulties associated with accurately predicting the behavior of the fluid during periods of freezing/stagnation and the subsequent recovery. Resultantly, project personnel are preparing to execute a thermal vacuum test for the freezable radiator task. The goal of the thermal vacuum test will be to generate empirical data which will be used to correlate thermal math models. These correlated models will then be used to design a freezable radiator capable of meeting requirements similar to those described in Figure 5.

C. Phase Change Material Heat Exchangers

A typical PCM heat exchanger is used to store excess thermal energy during periods of high heat loads (or hot thermal environments) by melting a material and rejecting the stored energy at a later time when the environment is more favorable or the heat rejection requirement is lower. During the rejection period, the material is frozen again, preparing it for the next heat load period.

The mass of a given PCM mass required to provide thermal control for a PCM application is inversely related to the material's heat of fusion as shown in Equation 1.

$$m = \frac{E}{h_f} \qquad\qquad (1)$$

where:

      m = mass of PCM (kg)
      E = PCM energy storage requirement (J)
      h_f = PCM heat of fusion (J/kg)

Water has a heat of fusion almost 70% higher than a typical PCM with the appropriate control (melt) temperature. Therefore, the use of water as the PCM has the potential to significantly reduce the required heat exchanger mass. Of course, some unique challenges are associated with the use of water as the PCM. Unlike most fluids, water expands when it freezes, which results in unique structural design challenges.

Advanced Thermal Project personnel are currently pursuing two parallel efforts in the development of water-based PCM heat exchangers. The first of these efforts is focused on better understanding, and eventually mitigating, the failure mechanisms associated with using water as the PCM.[t] The second effort involves the selection of the proper interstitial material for inclusion in the PCM heat exchanger.[u]

The project has worked with Energy Sciences Laboratory, Inc. (ESLI) to develop several ice PCM heat exchangers for experimentally evaluating the failure mechanisms associated with using water as the phase change material. The first heat exchanger developed by ESLI and test at NASA JSC was called the Replicative Ice PCM (RIP). This heat exchanger was designed to exactly replicate (with no mass optimizations) the energy storage capacity of an existing paraffin-based PCM heat exchanger. Because water has a much higher heat of fusion than the phase change material used in the paraffin-based PCM heat exchanger, it was significantly lighter and smaller than the baseline. The two units are shown side-by-side in Figure 6.

The final RIP mass is approximately 5.4 kg which compares favorably with the baseline heat exchanger mass of 8.4 kg. In addition to being significantly lighter the water-based heat exchanger was much smaller with a volume of 3500 cm^3 versus 6000 cm^3 for the SOA heat exchanger. ESLI also designed and fabricated several other water-based PCM heat exchangers for evaluation at NASA. All of these additional heat exchangers were filled with water to approximately 80% of the void volume. This fill fraction was chosen to allow for the expansion of the ice during freezing.

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Figure 6. Size comparison between paraffin-based PCM (left) and RIP (right).

In all, the test program involved a qualitative analysis of eight different water-based PCM heat exchangers. Seven of the eight test articles were damaged by ice expansion during the test program. Although many of the test articles were damaged, project personnel were able to gain valuable insight into the role that void control plays in the mitigation of PCM heat exchanger damage. The results of this test program are well documented in Reference xx.

As mentioned above, the second effort involving the development of ice PCM heat exchangers involved the proper selection of an interstitial material. There are two competing interests that one must consider when designing a PCM heat exchanger. One objective of the design process is to maximize the energy storage capacity within a given volume. To maximize the energy storage it is desirable to fill the entire volume with the phase change material. However, typical phase change materials tend to have low thermal conductivities. Resultantly, a heat exchanger completely filled with PCM would result in baseplates exceeding their maximum allowable temperature. A heat exchanger completely filled with an interstitial material (aluminum, copper, etc...), while minizing the temperature gradient across the heat exchanger would have a very low energy storage capacity. To address this design challenge, the Advanced Thermal Project developed a bench top test to evaluate various interstitial materials and configurations. Data from this test program was then used to develop correlated thermal math models. These models will be used to design a water-based PCM heat exchanger using a specific set of requirements.

VII.   Conclusion

Although NASA has recently cancelled plans to return humans to the Lunar surface, the technology development efforts that were being pursued to mitigate and address the thermal control system design challenges is being continued. Many of the technologies that were being pursued for these Lunar missions are equally applicable to NASA's future missions. Furthermore, these technologies are relevant to both crewed and unscrewed spacecrafts.

The technology development process begins with technologies possessing a TRL of approximately two or three and advances them to a TRL of six. The TRL six is achieved by completing an integrated thermal test including all of the previously developed hardware components. This test simulates a spacecraft mission including all of the relevant environmental parameters.

The project's portfolio is very broad and includes thermal control system fluids, heat acquisition hardware, and evaporative heat sinks. In addition to these elements, the project is also focused on developing variable heat rejection radiators. The final element within the project's portfolio is the advancement and development of water-based phase change material heat exchangers.

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Acronyms

CxP: Constellation Program

American Institute of Aeronautics and Astronautics

CxPO: Constellation Program Office
EDU: engineering development unit
ESLI: Energy Science Laboratories, Inc.
ETDP: Exploration Technology Development Program
GFY: government fiscal year
 IR: infrared
ISS: International Space Station
JSC: Johnson Space Center
LEM: lunar excursion module
LEO: low-Earth orbit
LER: lunar electric rover
LLO: low-lunar orbit
LSS: Lunar Surface System
MET: mission elapsed time
OAFU: orbit-averaged feedwater utilization
PCM: phase-change material
PDR: Preliminary Design Review
PNNL: Pacific Northwest National Laboratory
RIP: replicative ice PCM
SAM: surface area module
SDC: simulator-driven coldplate
SHReD: supplemental heat-rejection device
SHRIMP: small heat sink of replicative ice material for phase change
TCS: Thermal Control System
TLC: trans-lunar coast
TPP: technology prioritization process
TRL: Technology Readiness Level
WBS: work breakdown structure