Thermal Management for High Power Cubesats

Boris Yendler
YSPM
19190 Dagmar Dr, Saratoga, CA 95070
(408) 6663141
byendler@yspm.net

Ashton Meginnis, Adam Reif
Pumpkin Space Systems
750 Naples St, San Francisco, CA 94112
949-514-1106
ashton@pumpkininc.com

ABSTRACT

Thermal management systems for small satellites have traditionally been neglected entirely or only considered as an afterthought. This approach to small satellite systems design is no longer acceptable as technology has matured over the past decade and payload operational power has increased. Higher power leads to an increase in waste heat generated on-orbit. Trends in industry indicate that power demand for small satellite class (10-100 kg) can reach up to kilowatt range in the near future. A scalable Thermal Management System (TMS) has been developed which is applicable to small satellites ranging from CubeSats to ESPA class spacecraft. The TMS can handle up to 1 kW of waste heat. The TMS solution leverages breakthroughs in additive manufacturing, flexible heat pipes, and material science to dissipate extremely large quantities of waste heat in a small SWaP system. The system consists of:

- A rollout deployable radiator maximizing radiation of waste heat into space.
- Structurally integrated heat pipes providing an efficient heat transport;
- Energy storage based on a Phase Change Material (PCM) for mitigation of extreme temperature excursions during a peak power

The TMS is a modular system, flexible to be customized to particular mission requirements and spacecraft form factor. The paper discusses the TMS concept, components, and challenges. Results of performance evaluation are demonstrated.

The TMS development has been sponsored by the United States Air Force Research Laboratory.

1 TRENDS IN THE SMALL SATELLITE INDUSTRY

As small satellite technology has matured over the past 20 years, the applications for CubeSats and microsats have extended from university missions to government and commercial applications delivering real values for commercial, science and military missions. Several areas of smallsat applications are rapidly growing:

- On-board computing. Using graphics processing units (GPU) to process payload data onboard reduces the required amount of data needed to be downlinked through the comms systems.
- Synthetic aperture radar and other imaging techniques primarily used for Earth imaging.
- Laser based communications and high frequency/high speed radio frequency (RF) communications are increasing the bandwidth available to small satellites.
- Electric propulsion systems a major propulsion system from low Earth orbit (LEO) to deep space missions

All of the above trends areas are necessitating larger and larger power systems (in the 100's of watts for CubeSats, multiple kW for microsats). With an average of 40%-60% efficiency for electronics, there is a profound increases in waste heat generated on-orbit as a byproduct of spacecraft functions. The small satellite industry is trending towards higher power applications. A developed thermal management system which is discussed in this paper will manage the waste heat of these high-power applications.

1.1 Problem Statement: Small Satellite Thermal Control

Thermal management systems (TMS) for small satellites have traditionally been neglected entirely or only considered as an afterthought. This has been acceptable as smallsats have played the role of technology demonstrations with higher risk tolerance than traditional space missions. This approach to small satellite systems design is no longer adequate as small satellite technology has matured over the past decade and missions' transition from tech demonstrations to military, scientific, and commercial operations. As spacecraft power increases, so does heat waste generated on-board. Therefore, small satellite (smallsat) thermal control must be seriously addressed to ensure the success of future smallsat missions.

The TMS has been designed for small satellites ranging from CubeSats (10kg) to ESPA (EELV Secondary Payload Adapter, 200kg+) class spacecraft under the current effort. The TMS solution leverages breakthroughs in additive manufacturing, flexible heat pipe design, and material science to dissipate large quantities of waste heat in small satellite form factors.

1.2 Radiators

In the 10-200kg satellite class (those which classify as smallsats), radiators are typically integrated into the spacecraft structure, i.e. rigid and body mounted. Oftentimes the spacecraft chassis itself id the body mounted radiator. Body mounted radiators are a simple solution for thermal dissipation but are limited in their size (and therefore heat rejection capability) by the form factor of their host spacecraft, while also consuming the mass budget of the system. However, deployable radiators are currently being explored (though no small satellites have flown with deployable radiators to date, to the knowledge of the authors) to increase surface area for heat rejection. A key problem with deployable radiators is the heat transfer from the spacecraft body into a deployable radiator with minimum temperature drop, as hinges act as thermal choke points, limiting the efficacy of deployable radiators.

Consider the following example. Using the typical heat rejection capability of a radiator as 350 W/m^2 at 300K surface temperature one can calculate that the radiator must be at least 3.3m^2 to dissipate 1kW (the upper limit specified in the SBIR solicitation). Such surface area greatly exceeds the available surface area of most small satellites which makes a deployable radiator critical for dissipating large amounts of heat on high power small spacecraft.

1.2.1 Types of Deployable Radiators

A deployable radiator provides additional surface area to body-mounted radiators for heat rejection. A radiator's heat rejection capability is determined by its surface area, optical treatment of the radiator surface, temperature distribution across the radiator, and absolute temperature. In order to increase heat rejection from a radiator and to reduce radiator size, the radiator should have uniform temperature distribution and maximum operational temperature. The best way to achieve temperature uniformity (being isothermal) is to use heat pipes to spread heat throughout the radiator. Heat pipes have significantly higher heat conductance than any other material due to the power of heat convection through the heat pipe.

Radiator temperature considerably affects the radiator's heat rejection capability which is proportional to the absolute temperature of the body to the fourth power, that is, ~T^4 per Stefan Boltzmann law. For example, if the radiator temperature increases by 10°C say from 20°C to 30°C, the heat rejection into space will increase by 14% (293K→303K = 3% increase in absolute temperature). However, the radiator temperature can't rise significantly because it will lead to rise of temperature of components which are connected to the radiator through the thermal/mechanical interface. In general, component temperatures have maximum limits defined by the manufacturer for reliability and performance purposes. This combined with an estimated temperature drop between the component and radiator defines a maximum achievable radiator temperature. The radiator should then be sized according to the power of the payload, its power conversion efficiency, and its maximum temperature.

1.2.2 State-of-the-art Large Spacecraft Deployable Radiators

Deployable radiators have been developed for high power, large spacecraft because such radiators reduce the total mass and volume of the system. Large spacecraft employing deployable radiators include commercial communication satellites, military, and scientific satellites. Lockheed Martin, Orbital, etc. have also built satellites with deployable radiators for DoD, NASA and NOAA. To the authors' knowledge, all commercial uses of deployable radiators employ advanced heat pipes like Loop Heat Pipes (LHP) or Capillary Pumped Loops (CPL) to transfer heat from the spacecraft main body to the deployable radiators.

1.2.3 State-of-the-art Small Spacecraft Deployable Radiators

Small satellites have the same problem of transferring heat from the spacecraft main body to a deployable radiator across the hinge with minimal temperature drop. However, this problem is more difficult compared to large spacecraft due to much higher density of heat flux across the hinges for smallsats. For example, heat rejection density for a 12kW GEO communication satellite is about 26.8 W/m (across hinges), while heat rejection density for 500W 12U cubesat is above 193 W/m across hinges. This high heat rejection density requires efficient thermal paths from components to radiators and efficient radiator heat rejection. Otherwise, the temperature drop across the path becomes very large, reducing the efficacy of the radiator.

High-power small satellites will benefit from deployable radiators which significantly increase the rejection capacity as shown in work from AFRL, Northrop Grumman, and Loadpath. NASA reports on efforts to develop deployable radiators[1]. One of the designs is a deployable radiator developed by Thermal Management Technology (TMT) capable of rejecting 100 watts. Unfortunately, the product datasheet does not provide enough information to evaluate the suitability of the TMT radiator for our purpose. Another flexible radiator is built by JAXA, using graphite composite material developed by Kaneka Corp[1] .Evaluation of this design indicates that 44 sheets of this graphite material are required to transfer of 1 watt per Kelvin from the chassis into deployable radiator. For required heat transfer of several hundred watts into radiator across the hinge line, the design would require use more than 5000 sheets of the graphite material. A design relying on graphite sheets to transfer feat across a hinge becomes unwieldy as power increases up to levels specified in the AF191-070 requirements,

A deployable radiator should fit within the cubesat envelope to be applicable in the widest range of smallsat missions. This means that the dimensions of a single radiator panel cannot exceed the dimensions of the spacecraft chassis. The proposed Rollout Deployable Radiator (RDR) does not have this limitation since it can be rolled and inflated. The RDR will be able to be stowed within the spacecraft during launch for CubeSat applications; however, in a larger spacecraft that will not be launched via a canister dispenser, the RDR may be mounted externally.

Bendable radiators were investigated by NASA in the mid 1990's as an innovative means to get rid of large quantities of energy. A concept of a rollout radiator was discussed at the end of 1970's and beginning of the 1980's[2]. A flexible, variable-surface-area radiator was first introduced in 1978[Error! Bookmark not defined.3]. A pressurized gas was used to roll out the radiator. Retraction was done mechanically. Heat transfer along the radiator was done by conduction only. A retractable radiator for high power missions has been also discussed[4]. Material selection was recognized as an issue which will be difficult to resolve. Recent published research details breakthroughs in manufacturing of thermal spreaders, heat pipes, and other thermal systems. For example, flexible films for heat spreading have been developed by Airbus and ESA for use in geostationary satellites[18]. High thermal conductivity sheets also have been developed by other companies[16, 17].

Typically, in a traditional rigid panel deployable radiator the problem of transferring heat from the spacecraft body into the radiator has been solved by using Loop Heat Pipes (LHP). However, recent failure of the LHP on GOES-17[10] indicates that LHPs are inherently complex. They are also quite expensive which makes them less attractive options for low cost smallsats. Our solution seeks to integrate simplicity with effectiveness to deliver a product which outperforms all other commercial systems.

1.3 Heat Pipes

Waste heat generation from electronics and external heat loads results in large temperature gradients throughout the spacecraft body, with hot spots near high power payloads and on sun facing surfaces. Temperature can be reduced either by distributing heat throughout the spacecraft or by imposing duty cycles on payload/bus operations which may impose additional constraints on the CONOPS. Integrated Heat Pipes (IHPs) address this by distributing heat loads throughout the spacecraft, consequently reducing the effective temperature of fixed radiators. Also, IHPs are supposed to transfer external heat load from sunny side to the cold side of the spacecraft. Typically, the following heat pipes are used in space applications:

- Constant Conductance Heat Pipes (CCHP)
- Variable Conductance Heat Pipes (VCHP)
- Advanced Heat Pipe - Capillary Pump Loops Heat Pipes (CPLHP)
- Advanced Heat Pipe- Loop Heat Pipes (LHP)

The CCHP is the most popular heat pipe for space applications. Advanced heat pipes like CPLHP or LHP are typically used for deployable radiators. However, a high cost and lack of flight heritage for smallsats make use advanced heat pipes questionable.

1.3.1 State-of-the-art Small Spacecraft Heat Pipes

Several prototype small satellite heat pipes exist. Traditional grooved heat pipes with ammonia have been used on BIRD (Bispectral InfraRed Detection, launched in 2001)[2]. A thermal system was designed to handle on average power 35W with peak power of 200W. A flat-plate heat pipe on-orbit experiment was conducted on Small Demonstration Satellite-4[13] in 2011. The flat–plate heat pipe was made of rectangular stainless-steel tubing sandwiched between two aluminum plates and charged with refrigerant HFC-134a. . The flat heat pipe was actually an Oscillated Heat Pipe with check valves. The purpose of the experiment was to confirm developed model of a flat pipe in microgravity.

For a smaller spacecraft, ISIS has published work on a CubeSat copper/water heat pipe[6]. However, this unit consumes a large internal volume (1U), has minimal thermal transport capability (10W); and, to the knowledge of the authors, this unit has neither flown nor been qualified for spaceflight. The proposed solution integrates heat pipes into the structure to remove the volume penalty of using the system. IHPs will also transport much more than 10W throughout a spacecraft. IHPs will facilitate a significant step forward in using heat pipes for thermal management on small spacecraft. IHPs will increase thermal transport capability and reduce thermal management size, weight and power (SWaP) compared to existing systems.

1.4 Phase Change Material

Consider a spacecraft with peak payload power of 1kWe, of which 40% is waste heat, and a 5% on-orbit duty cycle. If there is no heat accumulation within the spacecraft, the waste heat of 400 Watts must be rejected into space instantly. That is, the spacecraft radiator must be sized to reject 400 W of waste heat to keep the payload temperature within operational ranges. On the other hand, if the 400 Watts, or some fraction of it, can be absorbed, stored, and rejected into space gradually over time, the radiator size may be reduced. Using the previous example, if the 400 W gets absorbed during peak payload (low duty cycle, 5% of orbit) and rejected into space during an entire orbit including the time of peak load, the radiator needs to only be sized for only 20 Watts. Therefore, having a thermal accumulator on board enables a significant reduction in radiator size.

1.4.1 PCM Applications

In the case of high duty cycle payload operations, a small radiator does not have ample time to dissipate the heat generated by a high-powered payload. In such applications, a thermal accumulator would be used to mitigate excursions into extremely high temperatures.

Another useful application of a thermal accumulator is to absorb external environmental load on the spacecraft panels, store and reject into space later. For example, if one side of a spacecraft is exposed to sun, the panel absorbs some of the sun load which can lead to an increase of spacecraft temperature. However, the thermal accumulator will absorb the external heat and store it. This heat is later rejected into space when the spacecraft face is in eclipse.

As mentioned earlier, PCM can be also used to moderate effect of external environment load on spacecraft temperature.

One of the challenges of using PCM is very poor thermal conductivity. Heat which is applied to the PCM boundary distributes through PCM bulk very slowly. It leads to creation of a very high temperature on heat application location while the rest of PCM is cold and solid. Therefore, enhancement of PCM thermal conductivity is required.

In our case, we need a PCM with melting temperature below or equal of 50°C, high latent heat and thermally stable. Review of the literature reveals that organic paraffin PCMs are the most desirable for our application. Organic paraffin brings many advantages such as cost, ease of access, toxicity, thermal stability, etc.

The most studied PCMs are paraffin waxes with melting temperature and latent heat of fusion are in the range of 40–60°C and 150–225 kJ/kg, respectively. Paraffin waxes are stable, non-toxic, and compatible with an aluminum container. However, they have low thermal conductivity.

1.4.2 Enhancement of PCM Thermal Conductivity

Latent heat of fusion and thermal conductivity are key factors to facilitate the absorption and release of energy in the PCM accumulator system. If the thermal conductivity is low, the heat flux cannot effectively diffuse and be stored within PCM volume. Essentially, only the layer of PCM closest to the heat source melts[8]. Enhancement of heat transfer in PCMs leads to an improved thermal accumulator performance. Most organic PCMs have an undesirably low thermal conductivity, in the region of 0.15-0.22 W/m/K[8]. Therefore, heat transfer enhancement is required for applications of PCM as thermal accumulator, in particular, for smallsats which might have a low duty cycle. Enhancement in heat capacity increases the amount of energy that can be stored as sensible heat. This is important particularly when the PCM works beyond its phase transition range.

A literature review revealed several methods to increase thermal conductivity throughout a PCM volume[12], common solutions are fins, metallic foams, embedded particles, etc. All solutions consume additional volume that would otherwise be occupied by the PCM, reducing the thermal storage capacity of the PCM Thermal Accumulator (PCM-TA) for a given volume. However, PCM that never melts is wasted volume and mass, so a solution to increase thermal conductivity to guarantee uniform PCM melting is merited. Published experimental and simulation data[12] indicates that use of the fins as the best method of increasing heat conduction through a PCM. However, no relation exists for optimal number of fins or fin spacing[7]. In general, the optimal number of fins is influenced by heat sink width, fin thickness, heat sink height (H), and heat flux[7].

Also, it is worth noting that the majority of the tests studied during this review have been conducted under standard terrestrial conditions so natural convection and gravity could influence obtained data. However, as estimations indicate an impact of gravity on heat propagation through horizontal PCM layer should be taken into account only for layers thicker than 3 cm. Typically, a Grashof number (Gr) is used to determine strength of natural convection vs viscous force acting on a fluid. If Gr>> 1 (like 1000 or more), natural convection overcomes viscous force and liquid PCM starts to move during heating at terrestrial conditions. For horizontal paraffin layer of 3 cm thick, Gr is around 100. It means that the natural convection is not strong enough to move liquid paraffin. The natural convection does not take place and terrestrial results are valid for space applications

1.4.3 State-of-the-art Small Spacecraft PCM

The authors are aware of companies which manufacture PCM based products for small satellites, for example Thermal Management Technologies, Advanced Cooling Technologies, Loadpath etc. To the knowledge of the authors, none of the PCM products have flown to date. Further, no known product uses a PCM thermal accumulator as a thermal switch between the spacecraft bus and a deployable radiator (as in the case of the TMS system). Finally, the PCM-TA is but one component of the TMS and only needed in a subset of TMS applications. It facilitates the integration of multiple scalable thermal solutions.

2 THERMAL MANAGEMENT SYSTEM

The overall goal of any thermal control system is to keep the temperature of payload and bus components within required temperature limits during all phases of the mission, from launch to de-commissioning for given environment conditions and concept of operations. In this regard, a designer of smallsat thermal management systems should be concerned with the following factors:

- Optimization of external surface area and surface treatment for heat rejection into space.
- Efficient heat delivery from electronics to radiator(s) with minimum temperature drop
- The ability to manage extreme temperature excursions (both hot and cold)

To address these issues, a thermal management system was designed as composed of 3 components:

- Rollout Deployable Radiator use if which provides several benefits: increase surface area for waste heat rejection, increase efficiency of the radiator, maximizing stowed efficiency.
- Integrated Heat Pipes have two functions. Firstly, they facilitate waste heat transfer through the structure to radiators with minimum temperature drop. Second, they help to equalize satellite temperature by transferring the environmental heat load between lit and dark sides.
- The Phase Change Material Thermal Accumulator serves several functions. It manages temperature excursions into extremes for low duty cycle, high power operations, which then reduce the required radiator surface area.

Thermal conduction, convection in heat pipes, and radiation all contribute to the TMS's ability to provide conditions required for safe high-power satellite operation.

2.1 Mission of Interest Case Study

To show an impact of the entire TMS and its components on thermal performance of a spacecraft we choose a mission which we come across often in our practice. However, typically such missions have lower power requirements than specified in the solicitation. Also, we've conducted a survey among our customers to get an understanding what power requirements are anticipated by our customers in near future. In order to meet solicitation requirements and to be in tune with expectations of our customers, a mission of interest was created. CONOPS of Mission of interest/use case is similar to typical to our customers but power was increased in order to approximate the solicitation power requirements.

Pumpkin 12U SUPERNOVA was chosen as spacecraft in the mission of interest. Figure 1 shows external and internal views of the spacecraft. The payload consists of 5 components: LCE 1, LCE 2, Radio, Avionic Bus, and Electric Propulsion Thruster (EPT). Duty cycle for each component resembles an actual mission but power generation of each component is significantly increased.

Details of Mission of Interest:

- ***Mission***: Earth observation
- ***Orbit***: Sun-synchronous noon-midnight orbit at 550km (shown in Fig.2 **.**)
- ***Spacecraft***: 12U Pumpkin SUPERNOVA
- ***Payloads***:
  - Science payloads LCE1 and LCE2, (each @ 50W, 50% duty cycle alternating).
  - Avionics BUS (@15W, 100% duty cycle). This is Pumpkin's SUPERNOVA avionics stack.
  - Primary communications Radio (@20W, 20% duty cycle). This is representative of many COTS CubeSat radios. (shown in)
- ***Temperature Requirements***: The spacecraft bus temperature must always remain below 60°C ***Spacecraft Orientation***: Nadir facing along a 6U face
- ***Component Mounting***
  - Payload LCE 1 has 3 connections: (1) bolted to + X panel; (2) connected by heat pipe to +X panel; (3) connected by a heat pipe to the main heat pipe
  - Payload LCE 2 has 3 connections: (1) bolted to + X panel; (2) connected by heat pipe to +X panel; (3) connected by a heat pipe to the main heat pipe
  - EPT is bolted to -Z panel
  - All other components are bolted to the nearest panel

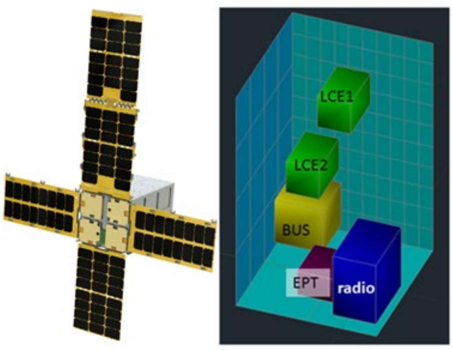


Figure 1 Pumpkin 12U SUPERNOVA: External view – left; Internal view –right

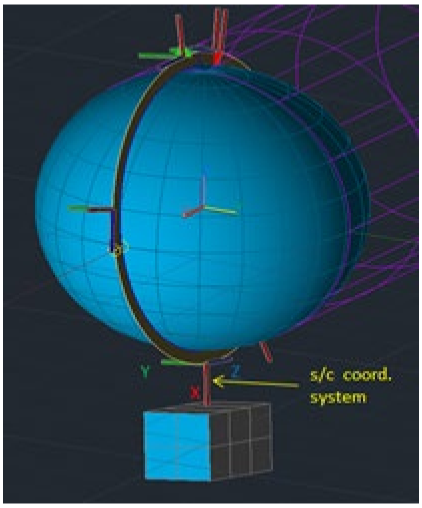


Figure 2 Mission of interest

The TMS provides a universal thermal interface for small satellite components, guaranteeing a maximum temperature at the interface between component and spacecraft. The temperature requirements are based on a mission profile and manufacturer specifications.

2.2 Rollout Deployable Radiator

As discussed previously, a deployable radiator provides surface area supplementary to body-mounted radiators for additional heat rejection capability. A radiator's heat rejection capability is determined by its surface area, optical treatment of the radiator surface, temperature distribution, absolute temperature, and thermal environment exposure. Preferably, the radiator should have uniform temperature distribution and maximum absolute temperature (i.e. minimum temperature drop from heat source to radiator surface) in order to maximize heat rejection from a radiator and to reduce radiator size. Radiator "isothermalization" comes with a hefty price of making the radiator quite heavy. The right approach is to allow some temperature gradient across the radiator while minimizing radiator mass. The best way to achieve an optimal temperature distribution is to use heat pipes to spread heat throughout the radiator. Due to its high conductance heatpipes distribute heat along the radiator with less mass penalty than any material by conduction.

Deployable radiators present a solution to the problem of rejecting large amounts of the waste heat generated by high power payload operations, while changing only slightly the dimensions of the spacecraft. This change is due to the need to accommodate the radiator in its stowed position during launch, which can either be accomplished by reducing the volume available for the payload or by increasing total spacecraft size. Deployable radiators have an inherent challenge of transporting heat from within the spacecraft into the radiator across the hinge while minimizing temperature drop. The proposed RDR design circumvents this problem via connection the rollout radiator to the spacecraft without any hinging mechanisms.

2.2.1 Conceptual Design

We've considered two concepts of deployable radiator depicted in Figure 3 (Rigid Panel) and Figure 4 (Rollout)**.**. A rigid panel deployable radiator has the problem of transferring heat across a hinge with minimal temperature drop. As discussed previously it could be done using LHP. But, for smallsats, LHP is problematic due to high cost, volume, and complexity. Another option would be flexible thermal straps of high thermal conductivity material (copper, for example). However, to scale this approach to 100 Watts of heat transfer the number of the straps must be so large that it becomes bulky and difficult to manage.

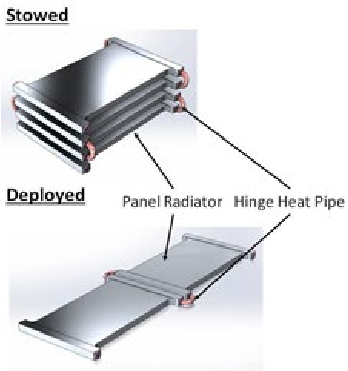


Figure 3 Rigid Panel Deployable Radiator

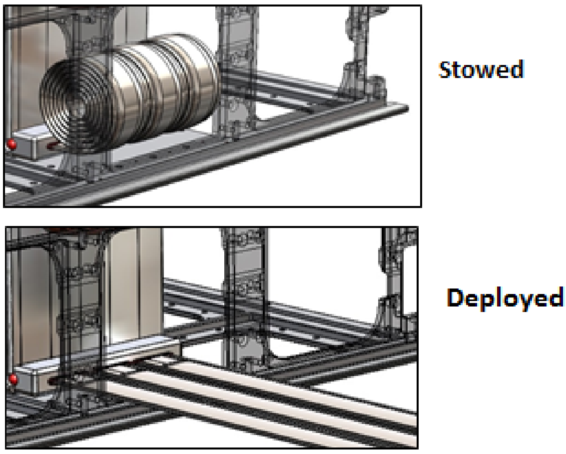


Figure 4 Rollout Deployable Radiator

On other hand, a rollout deployable radiator has two distinct advantages over rigid panel deployable radiators. First, it does not require hinging between the spacecraft and the radiator. It allows heat transfer from the spacecraft to the radiator with very little or no thermal resistance. The radiator itself does not contain hinges, which makes it much more thermally efficient than traditional radiators (no large temperature drops between bus and radiator). Traditional radiators could be even heavier than RDR if hinges are included. Secondly, it requires less space to be stowed. For this reason, the primary radiator design of interest is the Rollout Deployable Radiator (RDR). Figure 5 shows the concept of the RDR.

Figure 5 RDR design

Two aluminum face sheets are welded to create a pocket, open at one end. Inside the pocket is a mesh material which serves as a wick material. The vapor flows through the vapor core and condenses on low temperature surfaces. Condensed liquid is absorbed by wick in the RDR and returns to the vaporization chamber by return liquid flow partially via wick and partially via return channel.

Thermal model

The RDR thermal model has been developed using CR Thermal Desktop. Acetone was chosen as a working liquid due to low vapor pressure at high temperature. Acetone is compatible with aluminum, the metal of choice for the RDR manufacturing. Low acetone Capillary Transport Figure of Merit can be compensated by increasing the cross section area of the tube so that it is comparable to ammonia based heat pipes[5]. Therefore, acetone or some other working liquid with low Capillary Transport Figure of Merit can be used instead of ammonia in the RDR. The capillary limit for the RDR was found using methods for heat pipes with composite wick structure[11].

Simulations indicate that the maximum RDR heat transport capability is about 12kW with less than 1% of the liquid returns via RDR wick. Our results indicate that the RDR length can be increased significantly without degradation of the heat transfer capability.

2.2.2 Manufacturing Techniques

The RDR is a revolutionary application of ultrasonic welding; creating a flexible, inflatable heatpipe for small satellite thermal control. As it was mentioned previously, a flexible, variable-surface-area radiator used a pressurized gas to roll out the radiator. Retraction was done mechanically. Heat transfer along the radiator was done by conduction only[14].

Several prototypes of different design have been made by two manufactures, namely, Branson Ultrasonics and Fabrisonic, both a US-based ultrasonic welding company. All samples have been tested in order to show a feasibility of the RDR manufacturing and to determine the best design.

2.2.3 Prototype Manufacturing

The primary questions to be addressed during the tests were:

- Can the ultrasonic weld seam withstand the internal pressure expected during RDR operation?
- What is the maximum pressure the RDR can sustain?

- Can the assembly be reliably unrolled through application of pressure?
- Does the integrity of the seal weaken with time?

Addressing these questions represents a major reduction in the risk of the RDR design by demonstrating the feasibility of the primary mechanisms in RDR operation. Further, the lessons learned through the several rounds of prototypes will motivate future designs as the RDR matures towards its first flight unit.

Branson Ultrasonic has an extensive experience designing and manufacturing high altitude balloons composed of metal fabrics. Their experience is very useful for RDR manufacturing.

Fabrisonic was chosen due to their pioneering work combining additive and subtractive manufacturing with their CNC ultrasonic welding process (Figure 6) which provides repeatability of manufacturing.

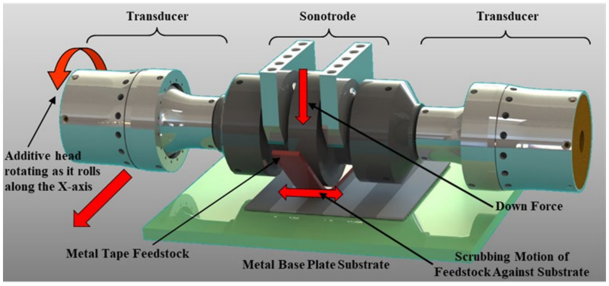


Figure 6: Fabrisonic ultrasonic welding process

2.2.3.1 Test fixture

A fixture has been built (see Figure 7) to test RDR prototypes. The test manifold served as a mounting platform for the RDR samples and provided a hermetic chamber within which pressure is applied and the sample's behavior is observed.

The RDR samples have been mounted to an interface plate with a 3D printed mandrel to be pressurized.

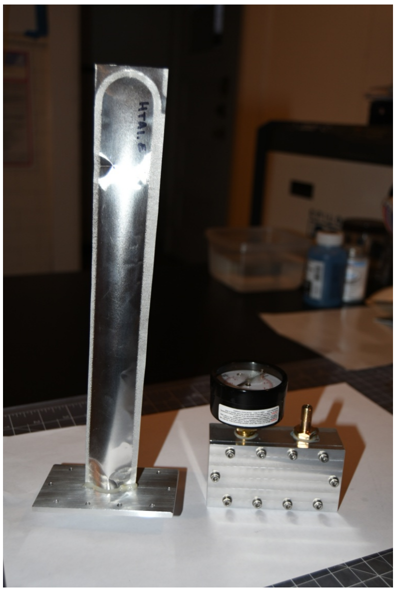


Figure 7: **RDR prototype mounted on the test manifold**

A handheld pump has been used to pressurize the test manifold, the pressure gauge on the manifold reads up to 200 psig and the pump up to 300 psig.

2.2.4 Test Results

The first tests conducted were to demonstrate unrolling of the RDR with the application of pressure. With a sample mounted to the test manifold, the RDR was rolled up to a 1.5" diameter roll. The internal pressure during the roll up was equalized with atmospheric pressure. After application of small pressure the RDR unrolled before any measurable pressure was achieved on the gauge.

The second test was to determine how much pressure a test article can hold. As Fig.8 demonstrates Branson's prototype was able to achieve sustained pressurization of 60 psig, which is above the target pressure. This demonstrates the feasibility of the RDR design and its ability to withstand the expected pressures.

Figure 8: **4 atmosphere hold on RDR sample**

RDR samples from Fibrisonic have been tested as well. After several iterations of the design, the last RDR design could held pressure up to 30 psig,

2.2.5 Comparison of Rollout versus Rigid Radiators

Storage efficiency of a traditional rigid radiator vs. RDR was evaluated using data from[15] for a fixed radiator. Based on this data, one can calculate a volume efficiency of a rigid heat pipe radiator as 0.036.m^3/kW. Scaling the RDR to the same parameters, namely, a two side radiator with 250W/m^2 heat rejection capability, the estimation indicate that RDR mass and volume efficiencies are 3.54 kg/kW and 0.0022 m^3/kW.

This means that the RDR and the rigid panel heat pipe radiator are very close from a mass efficiency point of view. However, the RDR storage efficiency is almost 10 times more than the rigid heat pipe radiator due to the fact that the RDR is flat when it's coiled in storage.

Note, that the analysis for the rigid heat pipe radiator analysis only included the radiator itself. If such an analysis were to be applied to a deployable heat pipe radiator, mass of the hinges must be included into calculations which could significantly increase mass of the deployable heat-pipe radiator. The calculations for the RDR account for all of the mass and volume required.

2.3 Conclusions

A primary focus of our effort is to develop conceptual design of the hardware based on preliminary analysis and to demonstrate by analysis and by test the feasibility of such concepts to meet all requirements. The RDR conceptual design has been developed and shown by analysis that it can meet objectives of the TMS. It was demonstrated that the proposed RDR has much higher mass and volume efficiency than a traditional heat pipe radiator.

The RDR is a high risk development as it is an attempt to build a flexible, rollout radiator. Feasibility was not clear at the beginning of the project due to uniqueness of the RDR design and lack of any similar development in the satellite industry in the past. Manufacturing challenges of the RDR have been a primary focus of the current effort. Several test articles have been manufactured and tested from multiple vendors. Results of early tests indicate that proposed RDR design is feasible to manufacture and can sustain internal pressures expected of acetone heat pipe operation at 50°C. A goal of the current efforts was to gain a confidence in our proposed design that the RDR which can be further developed upon as research continues on the technology.

3 INTEGRATED HEAT PIPES

3.1 Methods Assumptions and procedures

Integrated heat pipes are the natural solution of heat transport through and within a spacecraft including chassis. On large spacecraft, where heat is carried many feet, heat pipes are commonly used as a way to transfer heat and increase radiator efficiency. The need for heat pipes is less clear for cubesats. However, the density of the heat transfer in high power cubesats is much higher than in large satellites. It means that heat should be transferred in the most efficient way (using heat pipes) in order to avoid large temperature gradients which are associated with heat conduction through any solid media.

In investigating the feasibility of IHPs for the TMS, the following steps have been undertaken:

- Develop a conceptual design of the IHP which will suit small sat architecture
- Determine Heat Transport Factor in order to evaluate IHP performance
- Customize IHP thermal model which has been developed by CR Tech Thermal Desktop software
- Determine an effect of IHP on temperature behavior of space craft component
- Explored options for IHP manufacturing
- Evaluate IHP performance in concert with other TMS components

The following sections discuss the results of the IHP feasibility study.

3.1.1 *Conceptual Design*

One of the TMS components is Integrate Heat Pipes (IHP) which are embedded into spacecraft chassis walls as shown in Figure 9. In such a case, IHPs occupy no internal volume of the spacecraft. As IHP became a part of a panel, the thermal interface between IHP of different panels will be through flat contact as shown in Figure 9.

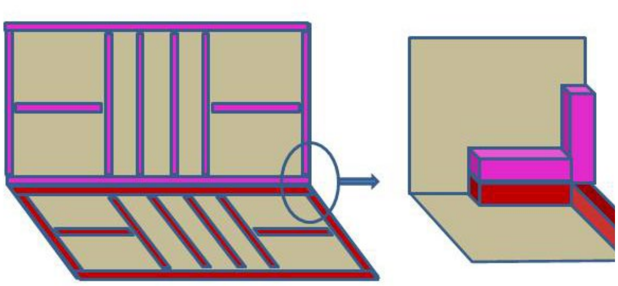


Figure 9: Example of IHP (red/pink) network and connection between IHPs situated on different panels

Thermal conductance through heat pipes interface is estimated at 30 W/m/K per heat pipe length. Increase of the thermal conductance can be achieved by applying a special gasket or conductive paste at the IHP interface. The internal structure of the IHP is shown in Figure 10. It is assumed that the IHP is a longitudinal groove heat pipe.

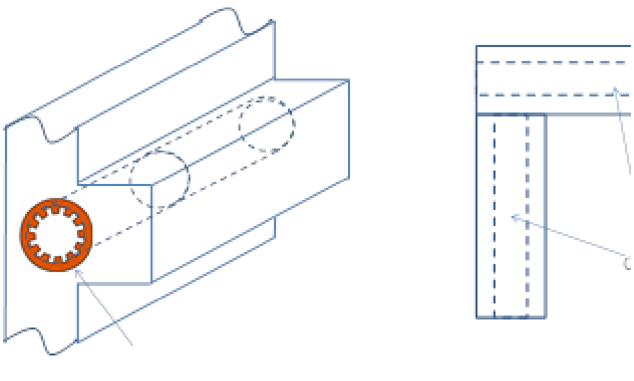


Figure 10: Close up of IHP showing cross section of IHP interface

3.1.2 *Thermal model*

The feasibility study is focusing on the Heat Transport Factor (HTF) and manufacturing feasibility of IHP's on small satellites. The IHP model specifically addresses two major factors of a heat pipe performance: the HTF based on a capillary limit[9] and temperature distribution in the spacecraft. A heat pipe module, which was developed by Thermal Desktop (C&R Technologies), is used to simulate heat transfer by IHP throughout the spacecraft.

3.1.3 *Manufacturing Technique*

As a first step in IHP development and to lower the risk, the most feasible path for IHP implementation in TMS is to use traditional machining techniques (CNC milling) for chassis panels and introduce grooved channels which slot COTS heat pipes sourced from several established. This is a simplified version of the intended IHP application, in this version there are neither inter-panel heat pipes nor are there heat pipes running along the edges of the panels (Figure 11).

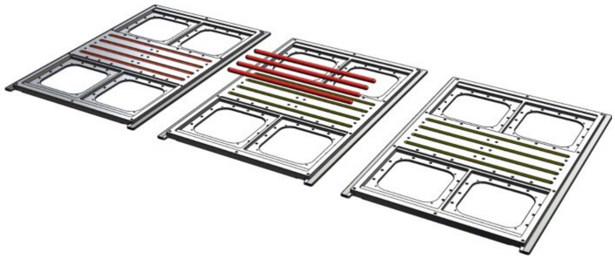


Figure 11: IHP manufacturing process. Chassis walls are machined as usual, and commercial heat pipes are inserted into grooved pockets. The figure shows an empty chassis plate (right), exploded view (mid), and integrated view (left).

3.2 *Results and discussion*

Impact of IHP on space craft thermal performance was determined for the mission of interest/use case (Section 2.1) with some modifications. Only LCE 1payload was turned ON with 50% duty cycle through the orbit. The other payload, LCE2, was turned OFF. The heat dissipation changes in the range 65 W – 375 W during orbit. Two cases, without and with IHP, have been run and temperature behavior of the spacecraft and components has been determined. Simulation results for LCE1 payload and panel are shown in Figure 12 and Figure 13

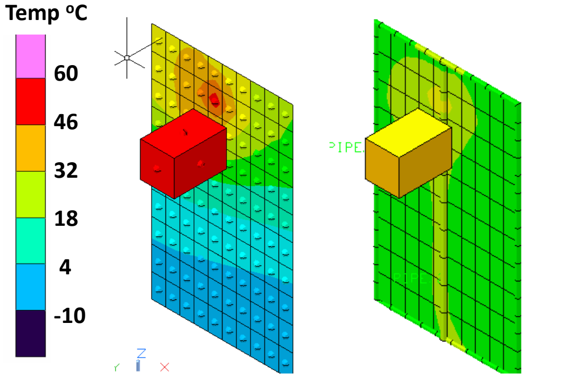


Figure 12: **Temperature distribution without IHP (left) and with IHP (right)**

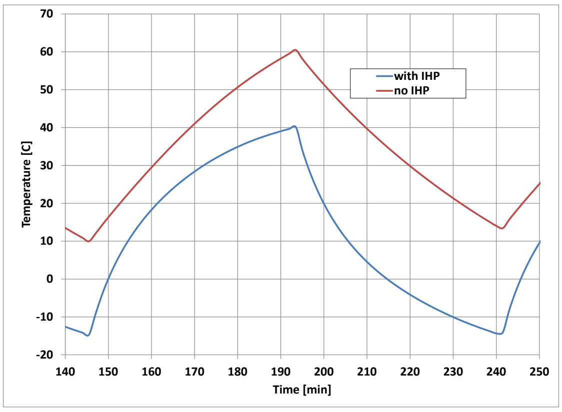


Figure 13: **Temperature without IHP (red) and with IHP (blue)**

As data in Figure 12 shows, IHPs reduce temperature gradients across a panel, increase radiator efficiency and, correspondingly, lowers component temperatures. Typically, a component is bolted to a spacecraft panel. In the case of high power component, such a fastener creates a hot spot on the panel. The cubesat panels are typically quite thin and don't have high thermal conduction. In a typical situation, when component power is low, this does not create a thermal problem. However, as our simulation shows, even with component power of 50 watts, a hot spot temperature of the panel can go up to 50°C (Figure **13**) and component temperatures up to 60°C.

As Figure 13 shows, average LCE1 component temperature drops by almost 20°C when the IHP is employed due to heat distribution throughout the panel by IHP. Essentially, IHP makes panel temperature distribution more even increasing in this way radiator efficiency. It leads to more heat rejection into space while the temperature difference between the component and the panel does not change with IHP. Estimations indicate that the spacecraft payload power can be increased by 50% when IHP is installed while maintaining temperature below the upper operational limit.

3.3 Conclusion

Currently, heat pipes are used in small satellites occasionally but the most of small satellites still do not employ them.

IHPs are a low-cost, effective means to increase the thermal performance of small satellites. Implementation of IHPs within existing small sat structure technology and the TMS is feasible due to well established commercial heat pipe sources which are compatible with current manufacturing techniques. Efficient heat transport through a spacecraft increases heat rejection capability, allowing for higher power payloads, and the presence of IHP significantly improves a spacecraft's thermal efficiency, primarily via heat distribution. Optimization of IHPs and its manufacturing will evaluated during future development.

4 PHASE CHANGE MATERIAL THERMAL ACCUMULATOR

A thermal accumulator provides a capability of absorbing large quantities of thermal energy during bursts of high energy activity and slow releasing energy afterwards. In small spacecraft, thermal accumulators provide a means to control spacecraft bus temperature for high energy payloads and low to moderate duty cycle. Section 1.4 discusses the fundamental principle of a PCM volume acting as a heat sink in a small satellite. The TMS PCM-TA can be mounted directly to a high-power unit for a direct thermal access to the accumulator. Another option is to mount PCM-TA directly to body mounted radiators. In this case, PCM-TA controls temperature of mounting interface between component and the spacecraft panel. It also allows absorbing an environmental heat load during spacecraft exposure to sun and dissipating it into space later reducing in this way radiator size. Accumulated energy will keep spacecraft temperature up during traveling through eclipse. PCM-TA operates passively and independently of the spacecraft bus.

The PCM-TA could be integrated with payloads and/or bus components or mounted on the chassis. Mission requirements determine PCM-TA dimensions and location. Example of 3U PCM-TA is shown in Figure 14.

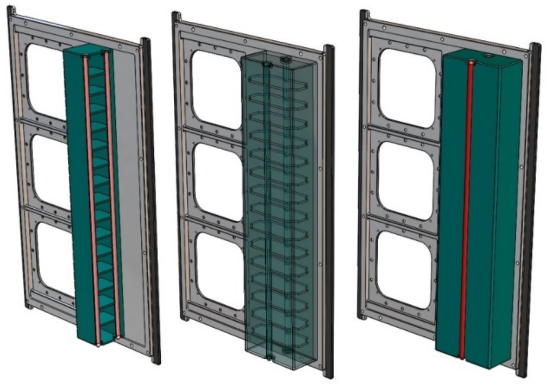


Figure 14: **3U length PCM accumulator, full view (right), transparent view (center), and cut out view (left)**

4.1 Methods Assumptions and procedures

Among several types of Phase Change Materials which have been discussed in Section 1.4 organic PCM (wax type) presents the PCM of choice for space applications. The features which make this PCM attractive:

- Relatively high latent heat, around 200-250 KJ/kg. It allows storing heat in relatively small volume.
- Melting point in the wide range of temperatures, from -25°C up to 70°C, It allows flexible and proper control of component temperature.
- Stability over thousands cycles freezing/thawing Non-corrosive and non-toxic. It is important for container design
- Reliable and stable solid ⇔ liquid transition
- High flash point
- Low coefficient of expansion, this reduces void volume while in solid state
- Low vapor pressure at room temperature

The following tasks have been accomplished for the feasibility analysis:

- Development of a conceptual design of the PCM-TA which is compatible with small sat architecture
- Modular and scalable design
- Evaluation if PCM-TA can work in concert with other TMS components
- Determination of the PCM-TA parameters like mass, volume, etc.
- evaluation of PCM-TA performance
- Determination of an effect of PCM-TA on temperature behavior of space craft component
- Exploration of options for PCM-TA manufacturing

4.1.1 Conceptual Design

In order to meet general thermal management needs in the small satellite industry, the PCM-TA is designed as an integrated unit with standardized mechanical interfaces. Volume of a standard PCM-TA unit is determined by mission requirements. For example, volume of PCM-TA is about 0.5U for the mission of interest. PCM-TA dimensions can be easily scaled to any size. Two heat pipes run along the length of the PCM volume connected to uniformly distributed fins. Internal fins enhance thermal transfer from the heat pipes into the PCM material to ensure evenly distributed melting of the material, thus maximizing the efficiency of the thermal accumulator (see Fig. **15**).

The conceptual design presented (see Fig. 15) here is the result of several iterations based on the results of the thermal simulations. Impact of the PCM-TA on temperature behavior of the satellite and its components is shown in Figure 18.

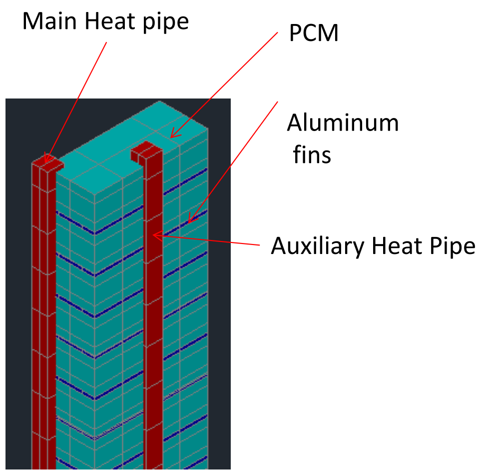


Fig. 15: **PCM-TA schematic**

4.1.2 Thermal model

The PCM-TA modulates temperature swings caused by the heat loads (either payload or environmental). That is, if due to heat load, the radiator temperature reaches PCM melting point, PCM-TA started to absorb heat preventing spacecraft temperature rise. The PCM-TA will release heat through the RDR when RDR temperature drops. To some degree, the PCM-TA plays role of a thermal switch denying external heat to get into the spacecraft and to damage electronics.

The PCM-TA performance was evaluated for the mission of interest described in Section 2.1. A "generic" PCM (close to paraffin wax RT44HC) was used in simulations.

4.1.3 Manufacturing Technique

The PCM-TA consists of a sealed enclosure with fins intersecting the volume along the length. The scale of the features is well within modern 3D metal printing techniques. Among many companies which are engaged into 3D mental printing, a company, Rapida, shows great promise for PCM manufacturing. 'Slices' of PCM-TA can be printed in arbitrary numbers and create a PCM enclosure of whatever length is required for the mission of interest.

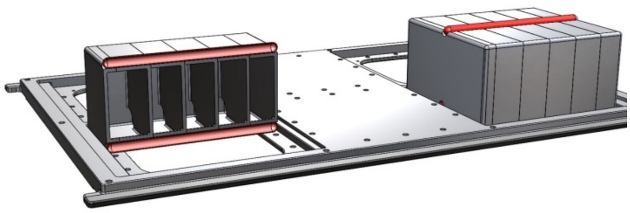


Figure 16: PCM-TA interior view, showing a stack up of five sections on a 6U base plate.

4.2 Results & Discussion

PCM-TA went through several iterations to find an optimal construction design and parameters. Only the final design is discussed in this section. Effect of fin spacing on energy absorption was determined and taken into account for PCM-TA design optimization.

Fins are connected to both, main and auxiliary, heat pipes, providing a path for heat into PCM volume. The auxiliary heat pipe equalizes temperature in PCM stack and provides a thermal interface between the PCM-TA and a spacecraft component, if any.

The current design of the PCM-TA facilitates the following functions depending on TMS configuration:

I. PMC-TA and RDR used together
- I.1. If incoming heat from the spacecraft panel into the main heat pipe is low, then the temperature of the main heat pipe is below the PCM melting temperature and heat will travel from main heat pipe to RDR without melting PCM. The heat is rejected into space via RDR and fixed radiators.
- I.2. If the incoming heat is high enough, the main heat pipe temperature will rise above the melting point, some portion of the heat will be stored as latent heat of fusion inside of the PCM stack. The rest of the heat will be rejected into space via the RDR and fixed radiators.
- I.3. The stored heat will be rejected into space when the RDR temperatures drop below the melting point
- I.4. If incoming heat from external loads onto the RDR or fixed radiators (sun, albedo, etc.) is high enough, the spacecraft temperature will rise. If the temperature exceeds the PCM melting point, the PCM starts to melt absorbing the external heat.

II. TMC-TA along or connected to a component via auxiliary heat pipe
- II.1. If incoming heat from the spacecraft panel into the main heat pipe is low, then the temperature of the main heat pipe is below the PCM melting temperature. PCM does not melt
- II.2. If the incoming heat is high enough, the main heat pipe temperature will rise above the melting point, the heat will be stored as latent heat of fusion inside of the PCM-TA. If incoming heat exceeds the PCM-TA capacity, PCM temperature will exceed melting point.

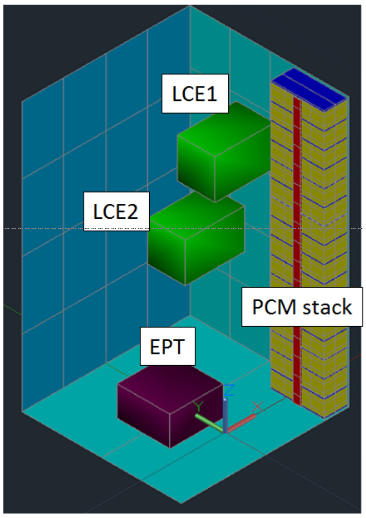
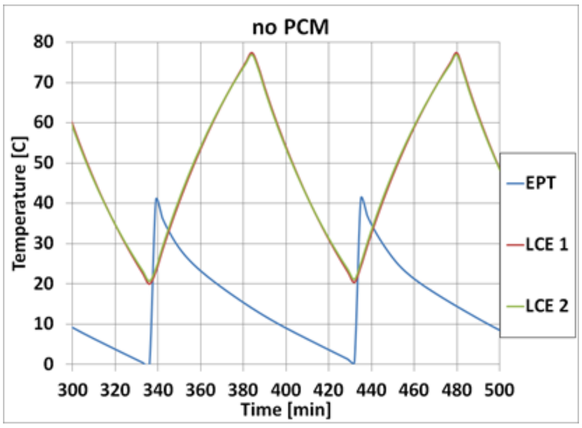


Figure 17: The interior of the spacecraft with payload and EPT components. Bus, Radio modules and the vaporization chamber for the RDR are not shown.

Temperature behavior of the components LCE 1 and LCE 2 when the components are directly connected to PCM-TA is shown in Figure 18 when LCE1 and LCE2 are turned ON simultaneously. EPT component is not connected to PCM-TA. As results of simulation indicate (Figure 18) PCM-TA's presence significantly reduces peak temperature of LCE1 and LCE2 components. The maximum temperature of LCE1 and

LCE2 reduced from 75°C to 55°C due to the PCM-TA's presence.



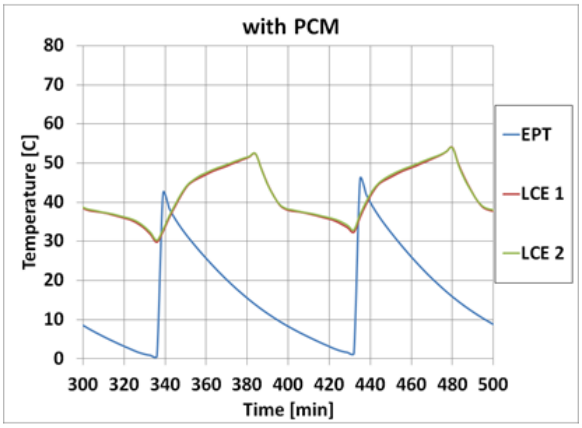


Figure 18: **LCE1, LCE2, and EPT Temperature.**

Figure 19 demonstrates temperature distribution on spacecraft panel when both components, LCE 1 and LCE 2 turned ON. One can see a significant temperature gradient (about 20°C) on this panel. It means that the radiative efficiency of +X panel is low and should be improved. This can be done by installing heat pipes along the panel to equalize the panel temperature because PCM-TA does not reduce temperature gradients along the chassis. This should be addressed by Integrated Heat Pipes.

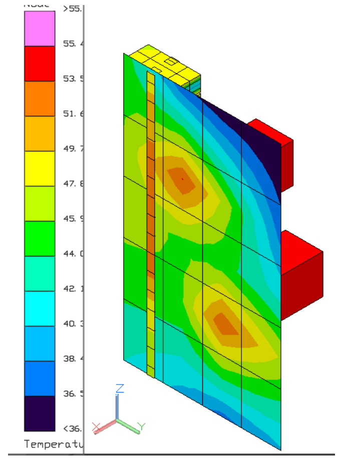


Figure 19: **Temperature distribution on +X panel**

4.3 Conclusion

The results of conducted study clear indicate that a design of PCM-TA is feasible. Manufacturing of such a unit won't be a problem because PCM is known to space industry and used in several missions. COTS phase change materials from a number of commercial suppliers (e.g. Industrial Raw Materials, LLC; RAHA Paraffin Co, etc.) are easily obtainable. It has been shown that PCM-TA is helpful to future users of high power smallsats, although it may only be necessary in a small subset of cases (high power, low duty cycle operations). For electronics with low to moderate duty cycles and environmental heat loads experienced by smallsats, use of PCM-TA will facilitate reduction of the payload temperature which, in turn, will allow use of higher power comparing to spacecrafts without PCM-TA. Paraffin waxes have been selected as the phase change material for it many beneficial properties (low cost, ease of acquisition, low toxicity, etc.).

5 TMS DESIGN MATRIX

Overall, the use of any one of the TMS components results in significant thermal performance improvement. Further, the simulations indicate the merit of integrating the TMS components to leverage the unique advantages of each one on a subset of smallsat missions. The thermal models developed in previous sections namely, RDR, IHP and PCM-TA are used in the development of the TMS Design Matrix which is discussed in this Section. Recall, the IHPs act as efficient thermal paths from high power payloads to other TMS components and serve to equalize the spacecraft chassis temperature. The

PCM-TA can be used as a thermal switch for low duty cycle operations. The RDR is an ultra-efficient radiator capable of rejecting large quantities of heat. All three components in a single spacecraft are shown in Figure 20.

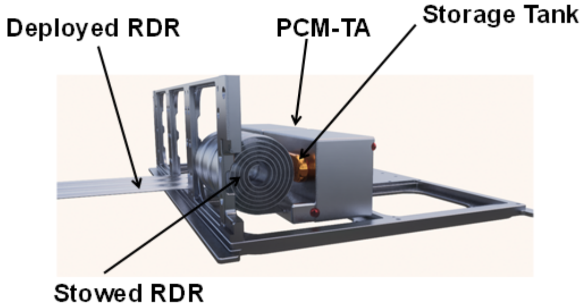


Figure 20 Combined TMS. IHPs are embedded into chassis, hidden from the view

5.1 Methods, Assumptions and procedures

The proposed TMS is very flexible and essentially presents a "tool kit" with a set of different tools which can be used individually or together. This creates a need for guideline how to choose the tool(s) that will be the most effective for a user's mission of interest and will minimize expenditure at the same time. In order to conduct such a study, developed models of RDR, IHP and PCM-TA have been used to determine conditions when an individual component or combination of the components produces maximum and minimum impact.

This study was conducted for a modified mission of interest in the following ways:

a) Only one scientific payload component is in use.
b) the scientific payload duty cycle was set to 100%; 50% and 15%

The following cases have been considered:

- No TMS (control)
- IHP only
- RDR only
- PCM-TA only
- Pair of components
- All components - Entire TMS

The purpose of the study is to show qualitatively the effect of component(s) deployment on ability to use more power by spacecraft components.

5.2 Results and Discussion

Figure 21 demonstrates an effect of different TMS components on ability to use more power for payload. The data is normalized by the power used in the case when no TMS is used and 100% duty cycle. That is, the plot shows a comparison how much more power can be used by payload if one or several TMS components are employed vs. no TMS is employed. The chassis temperature is kept at 55 C.

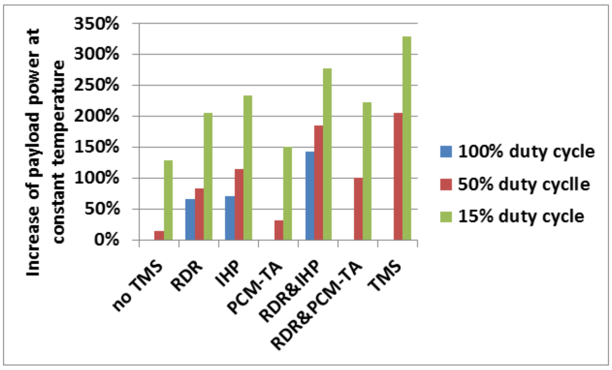


Figure 21. TMX performance matrix

As data indicates (see Figure 21) the payload power can be increased significantly, up to 300%, for 15% duty cycle. With high duty cycle, for example, 100%, the best combination to use is RDR and IHP which enables 150% power increase.

Table 1 provides a different presentation of the same data. The cases of the same duty cycle grouped together and normalized to the case with no TMS. Such presentation of the data simplifies analysis of the data. A purpose of the Table 1 is to give a designer of spacecraft thermal system a guideline which component will provide the maximum benefit with the least expenditure.

Table 1: Effect of TMS components on payload power normalized to the control case (no TMS)

WATTS @ 55°C			
Duty Cycle	**100%**	**50%**	**15%**
no TMS	100%	114%	229%
RDR	166%	183%	306%
IHP	171%	214%	334%
PCM-TA	100%	131%	251%
RDR&IHP	243%	286%	377%
RDR&PCM-TA	100%	200%	323%
TMS	100%	306%	429%

As data in Table 1 indicates, TMS facilitates the largest increase, 206%, of power use in the case of 50% duty cycle. For example, the payload power can increase by 66% if RDR is used along at 100% of duty cycle. As data shows, integrated heat pipes are quite effective in facilitating increase the payload power. Such efficiency comes from increase of efficiency of fixed radiators. Combination of RDR and IHP leads to the largest, up to 143% of the control case.

PCM-TA plays role only when payload duty cycle is below 100%.. The PCM-TA along facilitates 15% increase of power at 50% duty cycle. However, employment of PCM-TA allows maximal increase of power, up to 151%, at 15% duty cycle.

5.3 Conclusion on TMS performance

Data in Fig. 21/Table 1 shows effect of TMS configuration on how much power payload LCE can use keeping payload temperature at the interface with the chassis at 55°C relatively to the case when no TMS is employed for each payload duty cycle. For each payload duty cycle, all data is normalized to the case with on TMS and specific duty cycle.

The IHPs is the most beneficial in terms of cost analysis (a 220% increase in power) to spacecraft operator. This is contrary to current opinion which based on experience with low power smallsats. It shows that designers of high power smallsats should pay attention to temperature equalization before going to use deployable radiators which are much more complex in design and deployment.

6 CONCLUSION

The primary goal of the current effort was to prove the feasibility of the TMS design which enables a significant increase payload power for smallsats. The TMS consist of three components: Rollout Deployable Radiator (RDR), Integrated Heat Pipes (IHP) and Phase Change Thermal Accumulator (PCM-TA). A conceptual design of each component was developed and validated.

The performances of each component and TMS as a whole system were evaluated. To accomplish this, a thermal performance of the spacecraft was studied mission of interest which included heat transport through the chassis, thermal switching, heat storage, and heat rejection.

The TMS is modular, meaning each component can be used separately or in combination with other components. The whole system and its components are scalable, that is, the user can increase or decrease the TMS thermal capabilities depending on the mission requirements. The system is versatile and can be mounted on a platform at any spacecraft location.

6.1 Feasibility of TMS and components

A thermal model of each component was developed, and its performance was simulated using CR Tech Thermal Desktop. The simulation results verified the performance of the TMS components' in small satellite thermal control. The results of the simulation show promise for TMS, whole or in part. The results proved that TMS provides a significant improvement of thermal capabilities for spacecrafts under consideration. The TMS Design Matrix discussed in Section 5 could serve as guideline for spacecraft designers.

As a part of the feasibility study, investigation of manufacturing options for all TMS components has been conducted.

The RDR is the primary innovation of TMS and brings with it the greatest benefit for the thermal performance. The RDR was initially considered a high-risk design due to manufacturing uncertainties. Several rounds of RDR test articles have been manufactured and tested. Efforts on improving the design and manufacturing technique resulted in breakthrough and a significant improvement of the RDR design.

The current iteration of the RDR is the fourth revision and achieved sustained pressure above the threshold required for Acetone heat pipe operation at 50°C. The rolled RDR samples were successfully unrolled with the application of pressure. While much work is still required to develop the RDR into a flight ready product, the results of the current effort have demonstrated that the RDR is feasible.

Two other TMS components, namely, PCM-TA and IHP, are well known to the industry and do not present a manufacturing risk. The PCM-TA leverages the latest advances in additive manufacturing to create a modular system which is easily scaled for different missions. In order to expedite TMS implementation, it was suggested (as a first step) to use Commercial-Off-the-Shelf (COTS) heat pipes fitted into machined grooves in chassis components.

In conclusion, the TMS represents a major improvement in small satellite thermal performance with systems that occupy minimum SWaP and transport/reject/store maximum heat. Obtained results have demonstrated both the applicability and feasibility of the TMS and its components. It meets a need for small satellite thermal control.

6.1 Future Steps

All three components have had their applications verified via simulation raising their TRL to 2. Active research has been initiated on RDR manufacturing, with successful preliminary tests, raising the RDR's TRL to 3. Our intention is to bring TRL level for all components and TMS in whole to 5-6. It includes making prototypes, test them and to make a commercial product available to customers for flight applications.

REFERENCES

1. State of the Art Small Spacecraft Technology, Small Spacecraft Systems Virtual Institute Ames Research Center, Moffett Field, California; NASA/TP—2018–220027
2. Juhasz, A.J., Peterson, G.P, " Review of Advanced Radiator Technologies for Spececraft Power Systems and Space Control", NASA Technical Memorandum 4555, June 1994
3. Lura, F, et al," Thermal Performance of BIRD Microsatellite Thermal Control System – 3 Years of Operation in Space. 10.4271/2005-01-2986.
4. Furukawa, Masao. "Preparatory Study for Development of a Hybrid Capillary Pumped Loop with a Retractable Flexible Radiator." *SAE Transactions*, vol. 107, 1998, pp. 715–739
5. Bertoldo, J., et al, "Experimental performance comparison of axially grooved heat pipes charged with acetone and ammonia", International Heat Pipe Conference (16th IHPC) Lyon, France, May 20-24, 2012
6. Brouwer, H.S.B. , "Performance Characterization of Water Heat pipes and their Application in CubeSats Solving the Thermal Challenge of Next Generation CubeSatsH.S.B. Brouwer", Delft University of Technology; https://repository.tudelft.nl/islandora/object/uuid%3A219151e3-aac4-4cc7-8fb4-535e525d0297
7. Kalbasi, Rasool et al," Studies on optimum fins number in PCM-based heat sinks", Energy 171 (2019) 1088e1099
8. Kanimozhi et al, "Review on Heat Transfer Enhancement Techniques in Thermal Energy Storage Systems", Int. Journal of Engineering Research and Applications, ISSN: 2248-9622, Vol. 4, Issue 2(Version 1), February 2014, pp.144-149
9. Karam, R., "Satellite Thermal Control for Systems Engineers", vol.181, progress in Astronautics and Aeronautics, AIAA, 1998, ISBN 1-56347-276-7
10. NOAA. (n.d.). *GOES-17 Loop Heat Pipe Fact Sheet*. Retrieved 08 02, 2019, from NOAA Satellite and Information Service : https://www.nesdis.noaa.gov/sites/default/files/asset/document/NESDIS_factsheet_loopheatpipe.pd
11. Reay, D.A., Kew, P.A. "Heat Pipes, Theory, Design and Application", Fifth Edition 2006, Butterworth-Heinemann is an imprint of Elsevier, ISBN–10: 0-7506-6754-0
12. Singh, Randeep & Sadeghi, Sadeghi & Shabani, Bahman. Thermal Conductivity Enhancement of Phase Change Materials for Low-Temperature Thermal Energy Storage Applications. Energies. 12. 75. 10.3390/en12010075. (2018)
13. https://earth.esa.int/web/eoportal/satellite-missions/s/sds-4
14. Leach, J.W., and Cox, R.L. 1977. Flexible Deployable-Retractable Space Radiators. AIAA Paper 77-764.
15. Gilmore, G, D, "Spacecraft Thermal Control Handbook, Volume I: Fundamental Technologies", The Aerospace Corporation; Published by Aerospace Press, © 2002, 2nd Edition, 836 pages, Hardback; ISBN-10: 1-884989-11-X
16. "Graphinity", http://www.elecdiv.kaneka.co.jp/english/graphite/techno01.html
17. "PGS Graphite Sheet", https://www.mouser.com/pdfdocs/ThermalGraphiteSheets.pdf
18. Benthem, B, Mena, F, "Innovative new High Performance Radiators: Developing heat rejection systems with flexible film technology." ICES-2015-180, 45th International Conference on Environmental Systems ICES-2015-180 12-16 July 2015, Bellevue, Washington