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CubeSat Active Thermal Control via Microvascular Carbon Fiber Channel Radiator

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ABSTRACT

Small spacecraft rarely have volume for thermal control subsystems and often must perform operations in "burst" mode as a result. The few spacecraft who do have control rely on low-complexity thermal control systems which conduct heat to the bus structure and then radiate the heat away. These simplistic techniques are sufficient for low power missions in Low Earth Orbit (LEO) but are not capable of dumping the heat produced in new mission profiles that are in development. This is due to small spacecraft incorporating increasingly advanced subsystems which have difficult thermal control requirements such as propulsion systems or high-power antennas. The University of Illinois at Urbana-Champaign, in partnership with NASA Ames Research Center, is developing an active thermal control system for small spacecraft. This control system uses a deployable radiator panel made from carbon fiber with micro-vascular circulatory system for coolant. This paper is a follow-up on the previous year's SmallSat conference. A bench prototype of the thermal control subsystem was designed and built. The prototype has been tested and debugged under vacuum. Test results and lessons learned are presented. Moving forward, test conclusions will require some design parameters to be changed and the subsystem will reach TRL 6 by the end of the two-year program.

INTRODUCTION

Spacecraft with small size constraints such as CubeSats are progressing beyond low-power missions in Low Earth Orbit (LEO). Several subsystems such as power generation, power storage, propulsion, telecommunications, onboard computing, cryogenic systems, and biology experiments continue to grow with increasingly complex capabilities. Small spacecraft are also embarking on deep space missions where the spacecraft will be exposed to even harsher thermal environments, require more advanced propulsion, and need high power communications to relay back to Earth. These high-power subsystems generate a significant amount of heat which can only be radiated away and cannot easily escape electronically dense small spacecraft bus.

Current methods for mitigating high thermal loads involve operating in "burst" mode or using passive thermal cooling. Burst mode involves drawing significant amounts of power and thus producing high thermal loads in a short period of time. Afterward, the spacecraft must wait until it cools off before it can operate again. The downside of this method is that the spacecraft cannot be operating continuously and therefore is limited in the data it can collect. Passive thermal cooling uses heat pipes to connect the heat source to the bus structure so that the spacecraft can radiate the heat through the structure and exterior of the spacecraft [1]. The main problem with this system is that it can only conduct the heat out which results in cooling of the satellite in an uncontrolled fashion. Additionally, there are mission profiles in which the spacecraft should not be cooled because of its location in remote parts of space. Therefore, an active thermal control system is necessary to properly adapt to the various applications small spacecraft are being fabricated for.

The University of Illinois at Urbana-Champaign, in partnership with NASA Ames Research Center, is developing a thermal control support module for small spacecraft. This support module will utilize a deployable radiator as a means of dumping heat to the environment. The support module will also have a pump with throttle control thus allowing the bus to toggle the amount of heat dumped for survivability in a wider environment temperature range than traditional passive systems. Additionally, while the subsystem is being designed for a CubeSat form factor, the radiator is scalable for larger spacecraft which might have greater thermal loads.

To date the University of Illinois has published on preliminary leak tests and coolant selection [3] as well as optimization of the channels in the radiator panel for heat exchange and pressure drop [4]. This paper outlines the design of a bench prototype and testing in a vacuum chamber as well as the design challenges faced.

DESIGN

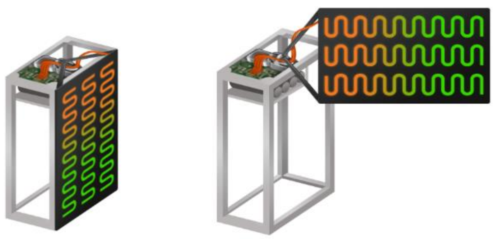

Figure 1 Left: Stowed Radiator system,

Right: Deployed system

Figure 1 shows an active cooling system concept for a 6U CubeSat. It consists of the deployable radiator and a 0.5 U support module in the bus of the spacecraft. The panel has a microvascular channel network by utilizing VaSC technology.

VaSC Technology

Vaporization of Sacrificial Components (VaSC) is a process developed at the University of Illinois to embed composites with three-dimensional vasculature. The radiator panel is made from a carbon fiber composite. Fabricating a carbon fiber panel to have microchannels requires the sacrificial material to be woven into the carbon fiber layers. Once epoxy is added and the panel cures, the sacrificial fibers are removed by heating the panel to 200 C under vacuum. The fibers evaporate and are removed from the panel by vacuum. This technology was motivated by developing cooling cases for electric car batteries in terrestrial applications. Figure 2 shows what a VaSC microchannel cross section looks like in the layers of carbon fiber. Previous work focused on the integrity of the panel post vaporization and looking for sources of leaks.

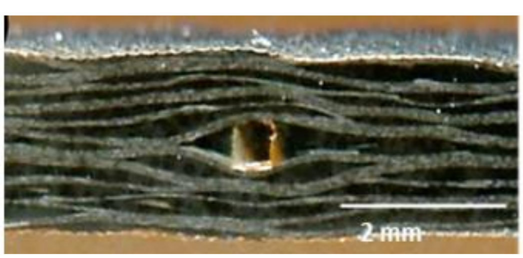

Figure 2 VaSC channel between carbon fiber layers.

The benefit to this design is its scalability. For more heat dissipation, a larger panel is required. For the analysis presented a 2U panel is used which will be tested to dump 25 Watts in the form of heat. The justification for this approximation is detailed in the following section.

THEORETICAL THERMAL MODEL

Previous analysis showed that for a LEO mission, coolants with large constant pressure specific heat coefficients (C_p) and low freezing points were ideal for efficient heat transfer and survivability in colder thermal environments [4]. The coolants selected were Dynalene HC-50 and Water/Ethylene Glycol 50/50 (vol.) mixture. These fluids were tested using NX Space Systems Thermal.

A method of optimizing channel configuration for given thermal and pressure constraints was developed [3]. The resulting tool can optimize channels for many different mission and pump requirements. Further analysis needs to be done on a prototype of the optimized channel to verify if the flow behaves like the theoretical model in a low gravity environment. For the purposes of this paper a simple bifurcating channel is used.

The optimization code requires an environment temperature. For LEO, there is no accepted ambient temperature value. The software assumes the panel is within 95% of a perfect conductor when circulating fluid. Therefore, for steady state analysis an assumption is made that the panel becomes the temperature balance of heat produced and radiation exchanged with the environment.

$$\boldsymbol{Q_{EA} + Q_{int} = Q_{s,E} + Q_{s,sp}}$$
Equation 1

Where Q_{EA} is the heat from the Earth's Albedo, Q_{int} is the internally generated heat from active subsystems, $Q_{s,E}$ is the heat radiated away in the direction of the Earth, and $Q_{s,sp}$ is the heat radiated away toward deep space. Each individual expression is shown in equations 2 – 4:

$$\boldsymbol{Q_{EA} = \alpha_R a_E I_s A_R}$$
Equation 2

Where a_E is the calculated average reflection coefficient of Earth from 50 to -50 latitude [1], I_s is the solar flux of the sun at 1 AU approximated as 1400 W/m^2, α_R is the thermal-optical absorbance coefficient of the radiator panel, and A_R is the area of panel facing the Earth.

$$\boldsymbol{Q_{s,E} = \mathcal{E}_{R,E} \sigma A_R (T_s^4 - T_E^4)} \qquad \textbf{Equation 3}$$

Where $\mathcal{E}_{R,E}$ is the thermal-optical emissivity coefficient on the side of the panel facing the Earth, σ is the Stephan-Boltzmann coefficient, T_s is the temperature of the surface of the panel, and T_E is the average temperature of the Earth taken to be 290 K.

$$\boldsymbol{Q_{s,sp} = \varepsilon_{R,sp}\sigma A_R(T_s^4 - T_{sp}^4)} \qquad \textbf{Equation 4}$$

Where $\varepsilon_{R,sp}$ is the emissivity coefficient for the side facing deep space and T_{sp} is the temperature of deep space approximated as 4 K. Since the panel is very thin, about 1 mm, it is assumed that the panel will be just one temperature and there will be no thermal gradient across the thickness. Combining equations 2 through 4 into equation 1 and rearranging for T_s yields the following expression for the temperature of the panel:

$$\boldsymbol{T_s = \left(\left(\frac{1}{\sigma A_R(\varepsilon_{R,sp} + \varepsilon_{R,E})}\right)(\alpha_R a_E I_s A_R + Q_{int} + \varepsilon_{R,sp}\sigma A_R T_{sp}^4 + \varepsilon_{R,E}\sigma T_E^4)\right)^{\frac{1}{4}}} \qquad \textbf{Equation 5}$$

This equation is used for a large design space where panel area, thermal optical properties, internal heat load, and environment temperatures are varied to model different missions. Additionally, T_s is solved with and without albedo to get the total range of steady state temperatures throughout an orbit. The desired configuration must have an area that can be constructed in the lab, must ensure that the coolant does not freeze at its coldest (without albedo and minimum internal heat generation), and T_s must be less than 80 C at its hottest (with albedo and maximum heat generation). 80 C is picked as a standard temperature where most electronics risk being damaged.

The results of this first principles model show that the minimum temperature for the maximum Q_{int} case occurs when absorption is minimized and emissivity for both sides is maximized. Additionally, the facilities at the University of Illinois are limited to making a panel no larger than 2.5U long. For these thermo-optical properties as well as temperature and size constraints, a theoretical upper bound for maximum allowable internal heat generation was determined to be 25 Watts.

BENCH PROTOTYPE

A bench test prototype was developed to evaluate all the individual components that would be necessary for the support module. This bench test would be a system level evaluation of all the components working together. Figure 3 shows a simplified block diagram form the support module.

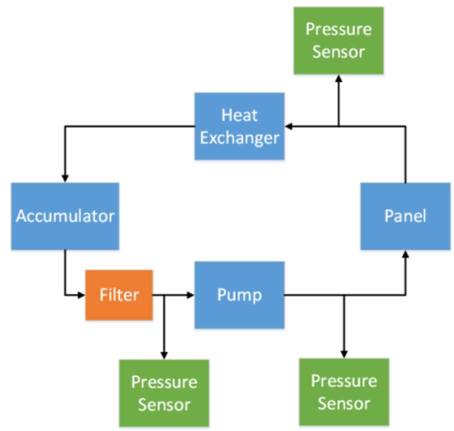

Figure 3 Block diagram of support module.

There are five main components to the support module: the pump that drives the fluid, the panel which displaces the internal heat, the heat exchanger which grabs waste heat from other subsystems, and a tank in the form of an accumulator which accounts for micro leaks in the system while maintaining system pressure. However, the accumulator has proven particularly challenging to manufacture for the given size constraints, so for the purposes of this paper a simple reservoir is used. Unlabeled are various connectors needed to complete the circuit. The last component, not seen in the block diagram, is the micro controller which drives the pump based off of pressure and temperature feedback from various sensors throughout the satellite.

Design Challenges

The main design challenges were form factor, thermal environment constraints, and maintaining leak tightness. All the components, except for the heat exchanger and panel, need to fit in a 10 cm x 10 cm x 5 cm or 0.5U form factor. It is assumed that the heat exchanger will be a custom piece the user will design to fit against their respective subsystems. This volume constraint was especially difficult for the micro pump. Additionally, the working fluid could experience a wide range of temperatures (from -40 C to 80 C) but most pumps had plastic components which could not operate less than -5 C. Thus, the pump was placed after the heat exchanger to make sure that the components were never too cold.

While previous work [4] showed the leak integrity of permeation through the panels, a means of connecting to and from the panel in a way that met the volume and leak constraints is detailed in this paper. From the University of Illinois CubeSat bus a maximum of 6 mm was given to objects extending beyond the frame. Therefore, in a stowed state the panel and connector combined could not protrude beyond 6 mm. Additionally, the diameter of the microchannels in the panel are between 250 and 500 microns and any connector designed would need to connect to an orifice of this size on the panel.

Components

The connector to the microchannels in the panel took several iterations before a solution was found. Each successive connector concept improved upon the previous in terms of leak rate, but three major iterations were necessary until an acceptable design was finalized. The first design concept involved embedding needles into the microchannel and sealing the two with epoxy. The epoxy did not bond well to the metal needle and often slid into the orifice and clogged the microchannel. Additionally, the needles were fragile and prone to bending shut. Multiple iterations of that concept were attempted with varying materials, epoxies and tube diameters before the next major concept was attempted. This second design concept involved embedding metal tubing between the carbon fiber layers. This is distinct from the needles as the tubing could be longer, had a much larger diameter, and further eliminated a leak point in the tube-to-needle interface which was previously required. This idea failed due to the differences in thermal expansion coefficients between the metal and the carbon fiber during the VaSCing process. Since thermal expansion was the challenge, the next iteration replaced the metal tubing with thermal resistant Teflon, which has a better thermal property match to the carbon fiber (Figure 4). VaSC was placed through both panel and tube to ensure a clear channel through the system. However, a good seal could not be formed between Teflon (even with rough, acid etched walls) and the carbon fiber layers.

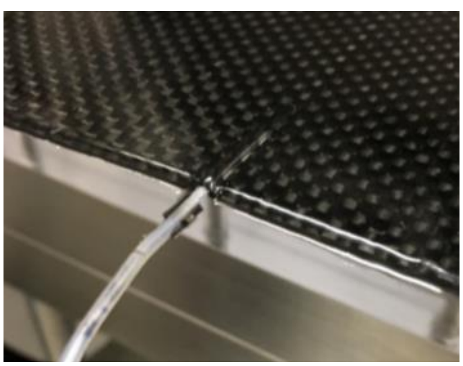

Figure 4 Embedded Teflon tubing.

Given all the problems with accessing the microchannels through the edge of a panel (epoxy going into the tubes, thermal contact issues, flexibility in the joint leading to leaks), a new approach was necessary. A face seal on the top of the panel using a conventional O-ring would address many of the challenges exhibited by the previous iterations. Such a face seal posed new design challenges and limitations, such as how to secure it to the panel, and staying within the standard CubeSat deployer volume. Although many face seals exist commercially, since the connector and panel combined could not extend beyond the CubeSat bus more than 6 mm (such that the bus and panel would fit inside a standard P-Pod deployer) a custom face seal connector was designed (Figure 5). Two variants of the design were fabricated, one that utilizes a barb fitting to connect to a flexible tube and one which uses a compression fitting. The leak test comparing the two variants showed that the barb fitting maintained a tighter seal than the compression fitting.

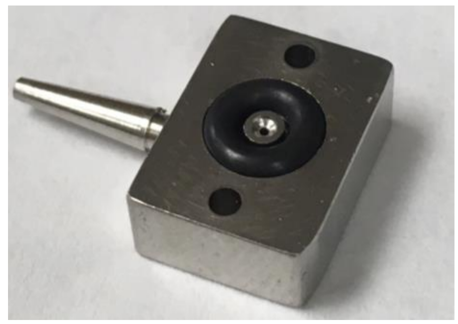

Figure 5 Custom microchannel face seal connector.

Initially, Teflon tubing proved too rigid for the barb fitting. However, Viton tubing proved flexible enough to make a good seal with the barb fitting. Figure 6 shows a panel coupon connected to fluid lines via the face seal connector. Viton degrades quickly in LEO and the section of a final, space rated tubing is the subject of future efforts.

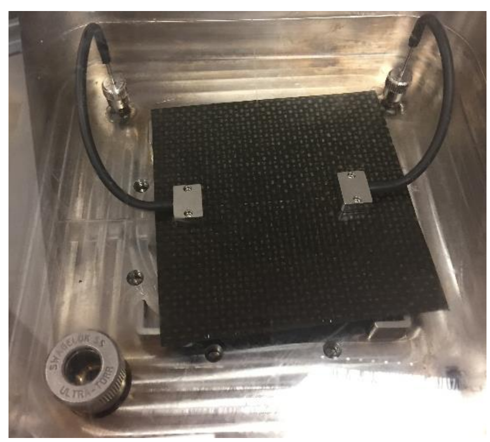

Figure 6 Panel with edge connectors in a custom vacuum testing chamber.

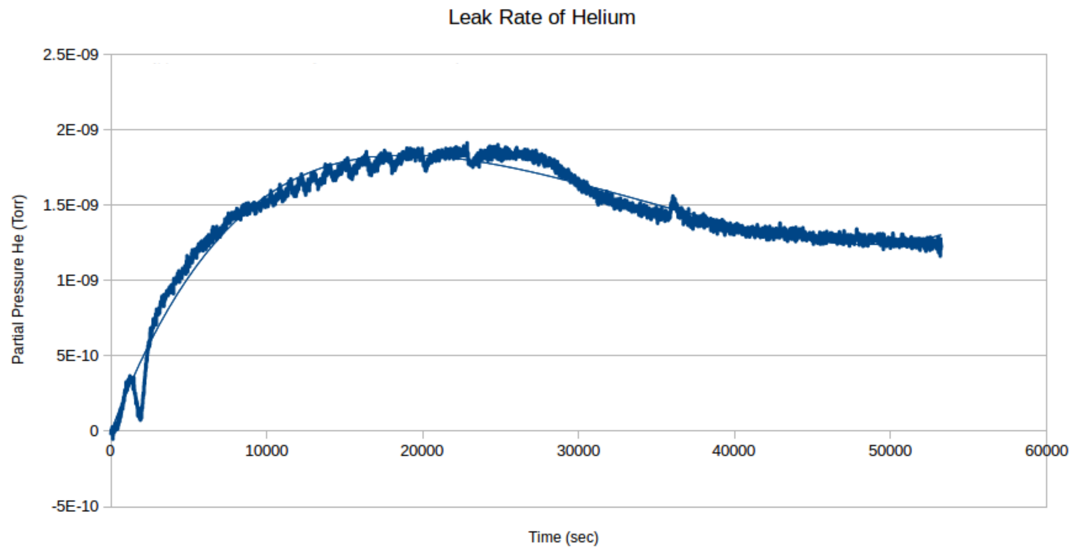

Figure 7 Leak test with helium over an extended period.

shows the leak rate of helium over a extended period of time (roughly 14 hours) to demonstrate the effectiveness of the face seal connectors. This leak rate was determined with a Residual Gas Analyzer (RGA).

An additional challenge in the designing the face connector was bolting the face seal to the panel. Initially, laser cutting was tried to avoid delamination caused from drilling. Unfortunately, due to the focal point of the laser, the cut hole had a taper across the thickness of the panel and had a significant heat-affected area around the hole.

Table 1 Micro pump motor specs.

Gear Pump Motor	Maximum Head Pressure (psi)	Flow rate range (mL/min)	Maximum viscosity (cSt)	Operating temperature (°C)	Power draw (W)	Dimensions (mm^3)	Mass (g)
DC Brushless	94	0 to 630	5000	-80 to 100	3 to 30	65 x 32 x 30	110

Following a recommendation from colleagues at NASA Ames, pinching the panel tightly between two aluminum blocks when drilling the hole proved to be very effective and minimizing the delaminated area around the hole.

The other components such as the pump, tank, and heat sink were more available from suppliers and thus did not require as rigorous of a design process as the face connector. Table 1 shows the different criteria each micro pump was evaluated on as well as the information on the pump that was chosen.

The heat sink was sized to allow up to 30 Watts of heat from a Kapton heater. Each application of the support module would allow for a custom heat sink to fit the user's needs but for the purposes of a bench test a simple rectangular heat sink is sufficient. Also, an accumulator that can regulate the pressure of the fluid in the loop as it changes due to thermal expansion is still in the design process so simple reservoir tank is used.

Lastly, temperature and pressure sensors are placed throughout the bench test to give the micro controller sufficient feedback to adjust the flow rate of the fluid accordingly.

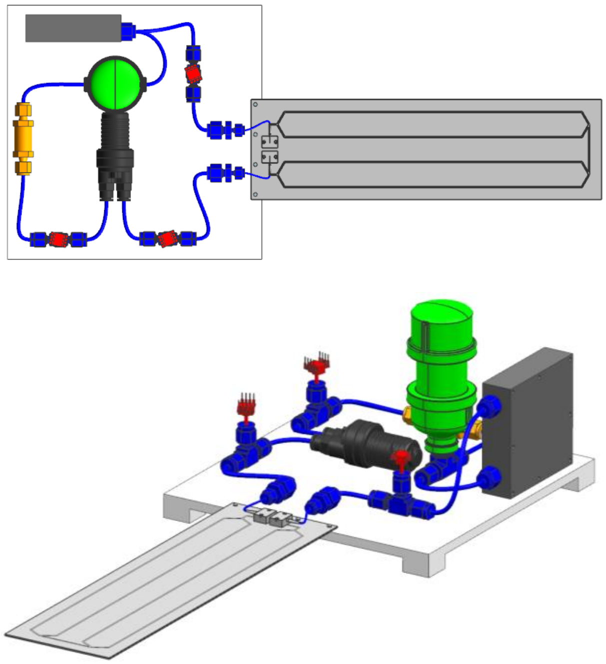

Figure 8 Bench prototype to be mounted in the thermal vacuum chamber (top view and isometric view). Fluid lines and pressure sensors are in blue, the heat exchanger is in black, the radiator panel is in grey, and the reservoir is in green.

VIBRATION AND THERMAL TEST RESULTS

To evaluate the bench prototype (Figure 8) it was subject to several vibration and thermal vacuum (TVac) tests at the NASA Ames test facility. The vibration tests followed the standard three single-axis acceleration tests. Figure 9 shows the sample 3U CubeSat being inserted into a PPOD deployer as a vibe test fixture. The radiator panel was connected to the bus by a hinge and tied down against the body by a thin Dyneema wire. Dyneema was chosen because this is the material the University of Illinois uses for thermal knife deployment. During the test the relatively sharp edges of the radiator carbon fiber cut through the Dyneema wire. This did not result in any damage of the panel but this presented a challenge that will have to be fixed by the prototype launch (see Future Work).

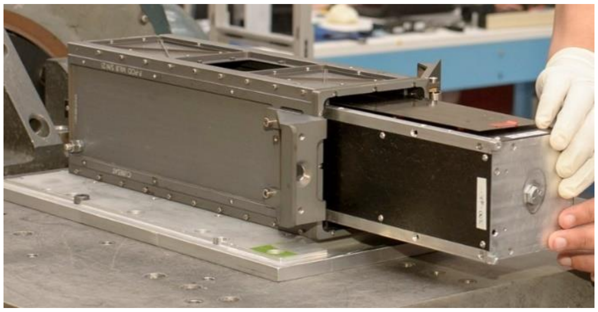

Figure 9 3U CubeSat and radiator deployable in a PPOD fixture for vibe testing.

The thermal tests involve mounting the bench prototype into a thermally controlled vacuum chamber (Figure 10) and measuring transient and steady state temperatures at various points in the system. This allows for proper characterization of the bench prototype and enables comparison to the thermal models. The system is subject to a range of environment temperatures and internal heat loads.

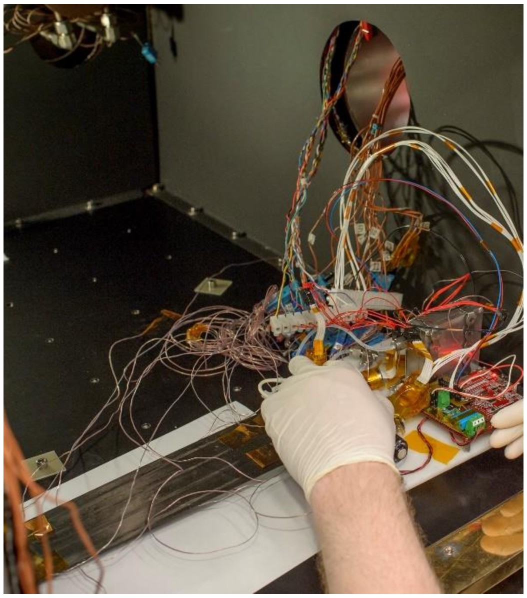

Figure 10 Bench test being fitted with thermocouples in TVac chamber.

The system experienced several technical difficulties during the testing that significantly hindered the range of tests that were able to be performed. The first test evaluated the system dumping 25 W of heat from the heater to the thermal controlled vacuum chamber which was set to -20 C (Figure 11). Over the course of an hour and a half the heater temperature rose roughly 50 C before the power to the heater was cut to observe pure cooling. The test was stopped before reaching steady state because the prototype will be flown in a low-Earth orbit where the period of the orbit will be roughly 90 minutes. Therefore, testing beyond that time would not be useful. The system was able to keep the heater below the upper limits of 80 C during this time. This matched the first principles thermal simulations conducted above. Once the heater was turned off the system was able to return the heater from 45 C to 5 C in an hour showing that when the system is not gaining heat it can quickly return to a cooled state.

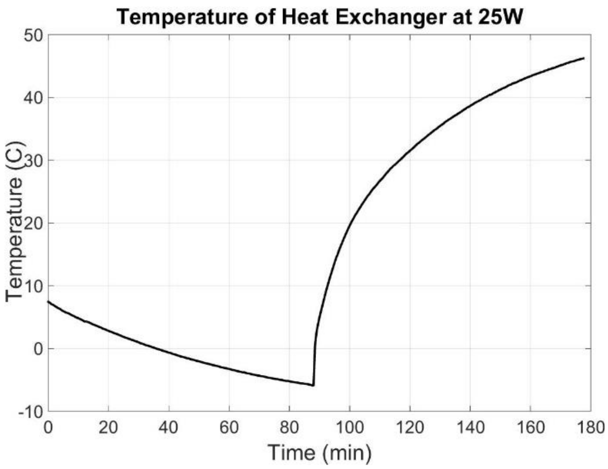

Figure 11 Transient heat transfer of 25W in -20C environment.

A second heating test was run at a -50C environment where the heater gradually increased the amount of waste heat produced and observed the heater reach steady state at each discrete heat load change (Figure 12). By adjusting the heat load the system was able to reach a steady state of 50 C while producing 15 W of heat. This is less than the thermal model predicted but it was observed that the heater was producing a significant amount of heat that may have affected the steady state limits.

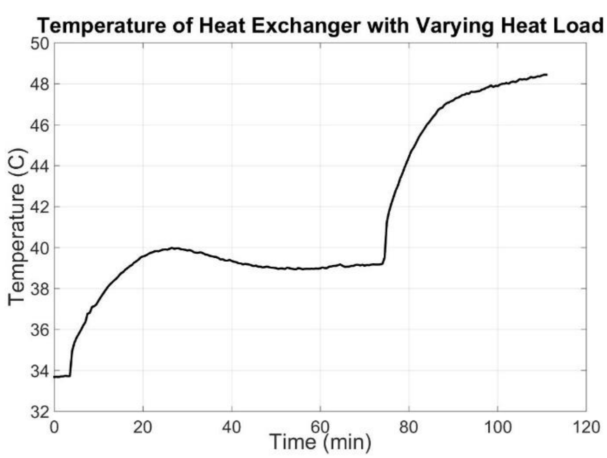

Figure 12 Transient heat transfer of varying heat load in -55C environment.

After these tests the system ran into significant issues that will lead to design decisions moving forward. The fluid used for these tests was Dynalene HC-50 or potassium formate. When the system was brought out of the vacuum chamber there were potassium deposits at the face connectors to the panel and by the system fill valve. This indicated that the valve was unable to prevent leaks to chamber and that the O-rings on the face connectors were not making a good seal. The system was refilled and the edge connectors were refastened. The system was tested in the lab before going back into the vacuum chamber and it was observed that the pump was running hotter than it had in the past. This suggests that the thermal transfer in the previous experiments was likely underperforming due to low pressure, but also that the currently selected pump is incorrectly sized for the actual pressure-head. Observation on the line leading into the pump saw cavitation bubbles indicating the low pressure side of the pump was getting low enough to cause a phase change. This indicated an issue with the fill procedure. Additionally, one of the test panels that had channels relatively close to the edge of the panels delaminated along the edge leaking fluid. This was the first time a panel delaminated under pressure and merits future study.

CONCLUSIONS AND FUTURE WORK

The tests at NASA Ames showed areas of the design that need to be tested further and undergo more rigorous validation. The pump was observed to generate a significant amount of heat on its own as it fought against the large pressure loss of the micro channels. Therefore, future work will have to evaluate pumps for ones that consume less power but also finding ways to reduce the head loss of the system. A new fill procedure will need to be developed such that the system is pressurized enough that the low pressure end of the pump does not experience cavitation. The edge connectors will need to undergo rigorous burst testing with varied torques applied to the bolts to reduce the risk of leaking. The panels need to be tested with varying distance between the channels and the panel edge to reduce the risk of delamination. Finally, variant channel designs need to be tested to reduce the panel back pressure while also increasing the fluid heat transfer into the panel.

Once the mentioned tests are conducted and the necessary design modifications are implemented the system will undergo more thermal vacuum tests and continue to characterize the system. While thermal vacuum tests are ongoing, a separate effort will begin to test the system under vibration to ensure there are no complications with individual components of the system as well as the storage and deployment mechanisms used.

This system will be repackaged into a flight prototype 0.5 U subsystem that will be a part of a 3 U CubeSat funded by NASA's Undergraduate Student Innovation Program (USIP). This launch is scheduled for August 2018 and will give valuable in-flight data for the radiator support system.

ACKNOWLEDGMENTS

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