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Preheating Cold Gas Thruster Flow through a Thermal Energy Storage Conversion System

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A thermal energy storage system capable of receiving, absorbing, and collecting solar energy, and storing it within a phase change material, has been designed as part of a power and propulsion system for use in low Earth orbit. The design includes thermophotovoltaic cells for the conversion of stored heat to electrical energy for various satellite systems, as well as a heat exchanger imbedded in the phase change material for propellant heating during thrust maneuvers. Through computational analysis, the likely thrust was evaluated as well as the impact of the propellant flow on the thermal to electric conversion. For the scenario analyzed, a propellant exit temperature of 1500 K, translating to an Isp of 300 seconds, was achieved. Passing the propellant through the phase change material decreases the temperature of the emitter used to power the thermophotovoltaics and resulted in a 20% decrease in electric power output.

I. Introduction

SOLAR thermal propulsion systems are known to offer significant advantages over chemical and electric propulsion for some mission scenarios [1]. In a typical solar thermal propulsion system, the sun's energy is

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concentrated and used to directly heat a propellant. The heated propellant then enters the nozzle of the system at an elevated temperature resulting in significantly enhanced thrust and an increased specific impulse relative to a cold gas flow.

Electric propulsion systems, the primary competitor to solar thermal propulsion systems, rely on the rather inefficient process of converting the sun's energy to electrical energy via photovoltaic panels, storing that energy chemically within a battery, and finally using it electrically to power a thruster. While a solar thermal rocket can directly use light and heat from the sun without requiring inefficient energy conversions, the primary drawback of the technology is that solar thermal propulsion typically relies on direct solar illumination at the time of propulsive maneuvers; when illumination is not available (e.g., during eclipse), the thruster cannot operate effectively. It has been proposed that combining solar thermal propulsion with high-temperature thermal energy storage can provide propulsive energy at all points in the orbit without the conversion inefficiencies of a solar panel battery system. Further, by utilizing a phase-change material (PCM) to store that energy within the latent heat of fusion of a phase transition (i.e.: solid to liquid), the energy can be stored and drawn out at a relatively constant temperature (e.g.: the melting temperature), allowing for predictable performance. A final enhancement to this design comes through the addition of a means to convert stored thermal energy to an electrical output to power various satellite systems. With a system designed in this way, a highly efficient power and propulsion system could be assembled without the use of traditional, relatively heavy, and inefficient photovoltaic and battery systems; previous work has shown that the proposed system can reduce launch mass or increase satellite performance relative to traditional systems, especially in the case of a mission in low Earth orbit requiring large velocity changes [1].

As the core of the proposed design, a conceptual system capable of receiving, absorbing, and converting solar energy was designed for use in low Earth orbit. Silicon was selected as the phase change material, and thermophotovoltaics (TPVs) were selected as a relatively efficient means to convert the high-temperature radiation emitted from the silicon to electricity. Silicon was identified as the phase change material due to its melting point temperature being suitable for propulsion and its large energy density (e.g.: heat of fusion). Additionally, the radiation wavelengths emitted by molten silicon are nearly ideal for use with typical thermophotovoltaics [1, 2]. Propellant preheating was added to the system to make direct use of the thermal energy, utilizing ammonia as the propellant and passing it through a heat exchanger imbedded in the phase change material before accelerating it through a nozzle. The purpose of this investigation was to determine if coupling the propellant heating to the latent

heat system would provide the desired nozzle inlet temperature and at what cost to the thermal energy storage and electric conversion. Figure 1 provides a general overview of system composition and the associated energy flow. The heat added to the system or utilized by the satellite is depicted with solid arrows, while losses from the system are shown with cross-hatched, light gray, arrows.

Data from resistojet propulsion systems were used as a starting point for the propellant heating portion of the design [3, 4]. The specific impulse of ammonia gas as a function of temperature entering the nozzle of these systems, as well as for several proposed solar thermal propulsion systems, has been investigated by a number of sources [3, 5, 6, 7]. If ammonia is heated to between 1500 and 1800 K, the specific impulse of a thruster can be raised from approximately 60 seconds to 300—350 seconds, as shown in Figure 2. From the published data, it was determined that for the proposed propellant heating method to be advantageous compared to more conventional systems, the ammonia must leave the PCM heat exchanger and enter the nozzle at no less than 1500 K. This exit temperature was assigned as the primary design goal for the propellant heating system.

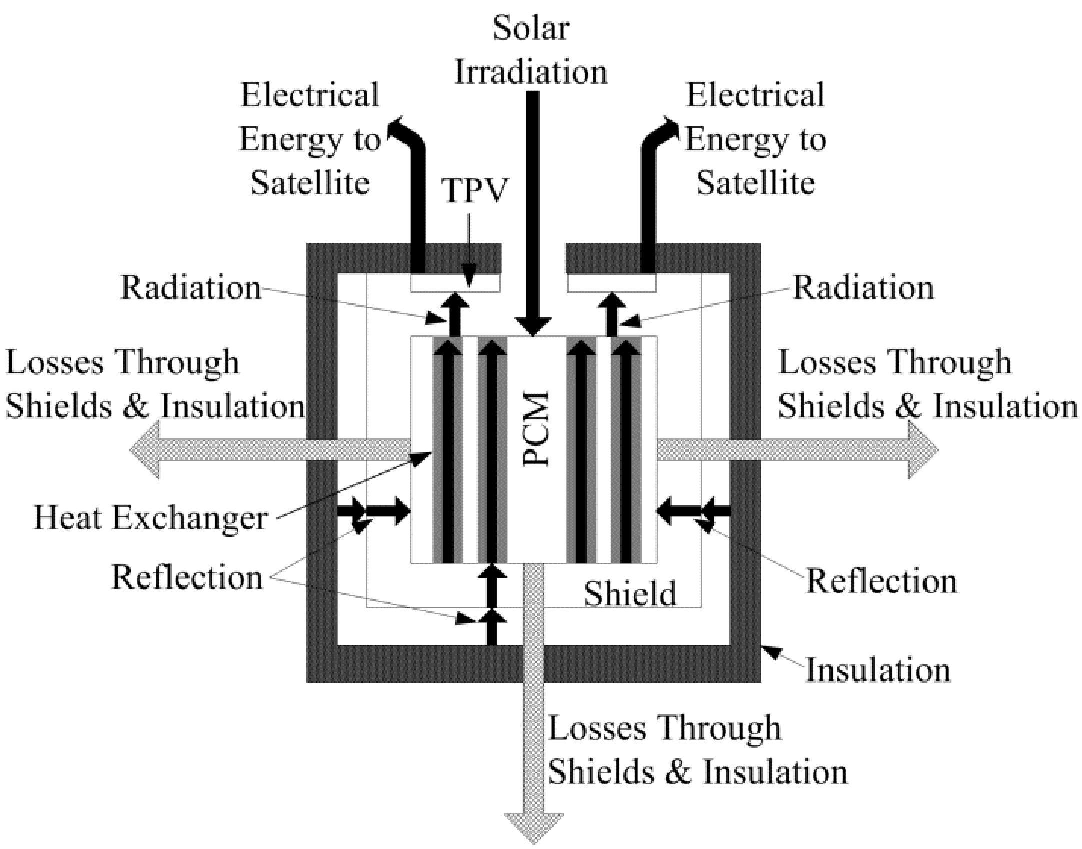

Figure 1: Thermal energy storage and conversion system overview with energy paths indicated.

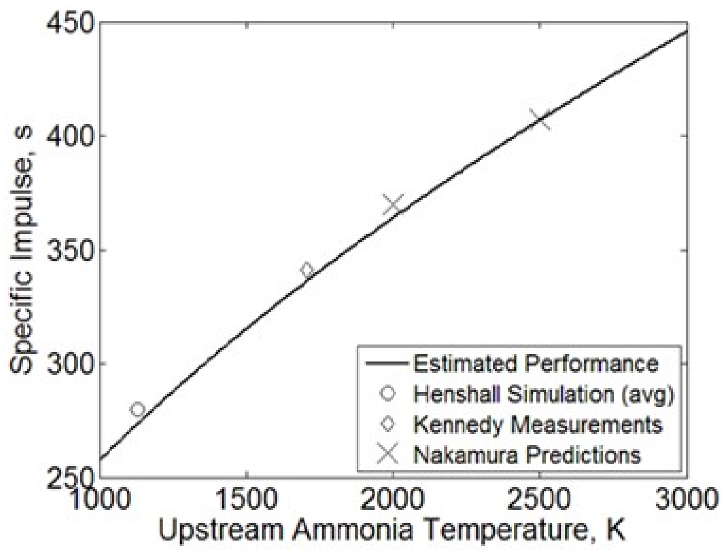

Figure 2: Temperature dependence of specific impulse for ammonia propellant [1].

The computational work described here characterizes the heat transfer to the propellant flow through a heat exchanger imbedded in the phase change material. The temperature at which the propellant exited the heat exchanger and entered the nozzle was predicted. In addition, the thermal to electric conversion capability of the system was quantified and the impact of propellant flow through the phase change material on power output determined.

II. Model

A system capable of Receiving, Absorbing, and Converting solar energy as well as heating ammonia propellant, referred to as the RAC, was evaluated computationally. The system was designed with silicon phase change material to store the thermal energy and thermophotovoltaics to convert the stored energy to electricity. A schematic cross-section of the axisymmetric system is shown in Figure 3. Starting from the center and describing the system radially outward, the core is a rod of silicon PCM housed in a cylindrical inner container. A vacuum gap separates the outer walls of the inner container from three radiation shields, a layer of carbon-bonded carbon fiber (CBCF) insulation, and an outer container. The TPV, composed of GaSb cells, is located above the top surface of the inner container. The top of the inner container is the emitter used to radiate power to TPVs. A channel passing through the insulation, standoff, and inner container wall feeds propellant into a manifold located at the bottom of the inner container. From the manifold, the propellant enters a heat exchanger composed of two concentric rings located in the

PCM. Certain aspects of the design were neglected by this study. Specifically, the aperture required for solar input and an exit for the heated propellant flow were not included. Obviously these are crucial components of the real design, but their exclusion does not affect the model and therefore were neglected for simplicity. The system dimensions and materials are detailed in Table 1.

Table 1: System component dimensions and materials

Component	Inner Height, cm	Inner Radius, cm	Outer Height, cm	Outer Radius, cm	Material
Core	-	-	8.4	6.2	Silicon
Inner container	8.4	6.2	10.4	6.7	Silicon carbide
Casing	29.5	19.6	31.5	20.6	Graphite
Insulation	20.5	11.4	29.5	19.6	Carbon-bonded carbon fiber
Shield 1	17.3	7.7	17.5	7.9	Tantalum
Shield 2	18.3	8.9	18.5	9	Tantalum
Shield 3	19.3	10	19.5	10.2	Tantalum

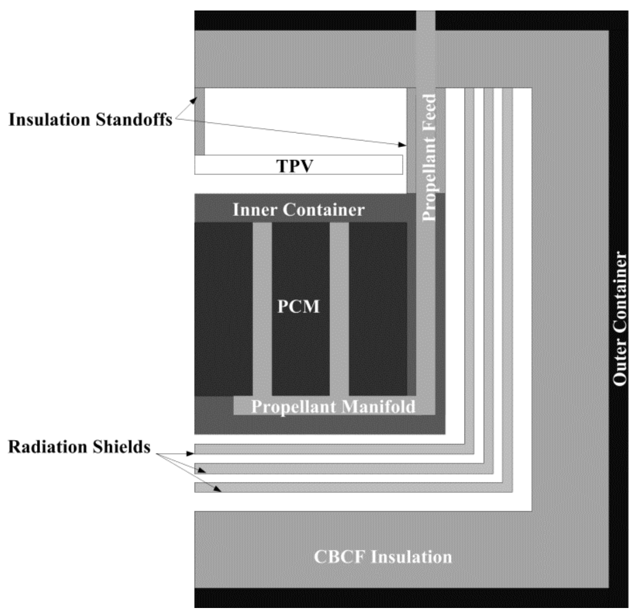

Figure 3: One-half of a cross-sectional view of the full RAC model; the center of the model is on the left, and the axis of symmetry is vertical.

The design requirements for the propellant heat exchanger were determined for an orbital station keeping scenario. A 200 kg satellite at 600 km and a 28.5° inclination was evaluated. A 1 m^2 drag area and coefficient of drag of 2.2 were used to determine the orbital degradation. The thrust force, length of thrust, and time between burns necessary to correct for an orbit decay of 1 km per day were calculated. Given a one minute burn, 1.9 N of thrust would be required every 24 hours. With these specifications and a targeted Isp of 300 seconds, the required mass

flow rate of ammonia through the heat exchanger was determined to be 0.64 g/s at a temperature of 1500 K. These requirements guided the heat exchanger design.

The conjugate heat transfer module in COMSOL 4.2a was employed to thermally characterize the heat exchanger in the PCM. To decrease the complexity associated with simultaneously modeling radiation, conduction, phase change, and fluid flow, a simplified model was generated and the modeling process was broken into steps. In an earlier study of the overall system, before the heat exchanger was added, a two-dimensional, axisymmetric model, generated in COMSOL Multiphysics 4.2a, was used to evaluate the melting and solidification fronts in the phase change material as well as characterize the system heat transfer [8]. The boundary conditions used for this complete model were chosen to accurately simulate low Earth orbit. The results of this full model with its realistic boundary conditions were then used to create the simplified, two-dimensional, axisymmetric geometry with equivalent boundary conditions that were used for the current work and are shown in Figure 4. Equivalent surface-to-ambient boundary conditions were applied to each of the outer boundaries of the container and insulation standoff. The top of the insulation standoff was set to a constant temperature because the temperature at that point fluctuated little throughout the thermal cycle. These conditions resulted in the same heat transfer rate out of the core as resulted for the full geometry exposed to an ambient temperature condition of 300 K. The equivalent model thermal boundary conditions are shown in Table 2.

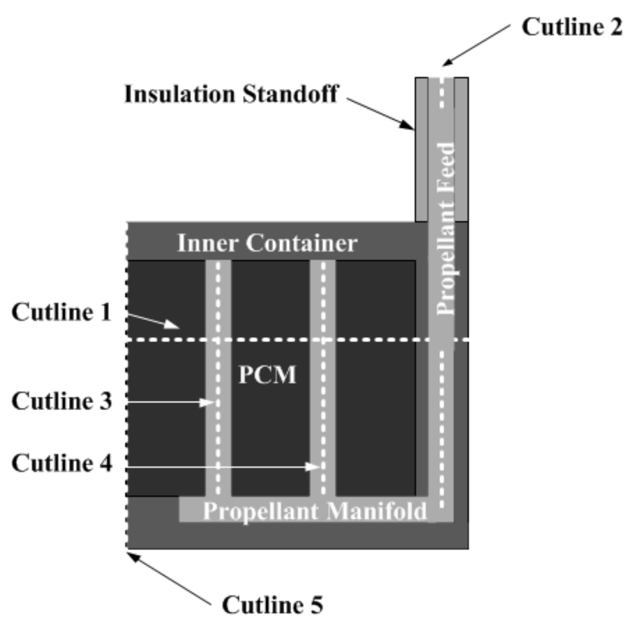

Figure 4: One-half of a cross-sectional view of the equivalent RAC model

Table 2: Equivalent boundary conditions

Surface	Emissivity	Ambient Temperature, K
Container top	0.4	591.1
Container sides	0.024	1050
Container bottom	0.024	1050
Standoff in. surface	0.2	1000
Standoff out. surface	0.006	1050
Standoff top	-	1300

Although the use of the equivalent boundary conditions reduced computational time, the addition of fluid flow to the remaining radiation, conduction, and phase change analysis still proved to be computationally intensive. To help mitigate this issue, the fluid flow associated with the heat exchanger was added in increments. In all flow studies, the ammonia was treated as an ideal gas and the flow was considered to be single phase and laminar. In Study 1, the flow in just the propellant feed channel was analyzed. At the entrance of the propellant feed a temperature of 300 K and a mass flow rate of 0.64 g/s were set. At the propellant feed exit, an exit pressure of 0 Pa was set using the pressure, no viscous stress setting. In addition, constant temperature conditions were applied to the propellant feed channel wall. The constant temperature conditions were determined from the full study of the overall system [8]. The propellant feed outer radius was varied and the manifold inlet temperatures (temperatures at the exit of the propellant feed) that could be expected were determined.

In Study 2, the flow from the propellant manifold through the heat exchanger was evaluated. The temperature in the manifold was set to a constant value dictated by the predicted propellant feed exit temperatures of Study 1. Constant velocity conditions were set at the inlet of the inner channel and outer channel of the heat exchanger, which ensured a constant mass flow rate of 0.64 g/s. At the exit, a pressure, no viscous stress condition was employed. The outer radii of the heat exchanger rings were varied and the temperatures as the ammonia exited the heat exchanger were calculated. The phase change mechanics for the PCM was modeled using the temperature transforming method.

Study 1 used a steady-state, fully coupled, direct PARDISO solver. The channel was meshed with 4192 free triangular elements. Study 2 used a transient, fully coupled, direct PARDISO solver. The system shown in Figure 4 was meshed with 44026 free triangular elements.

III. Validation

The heat transfer and phase change mechanics predicted by the temperature transforming method, employed by this study, were compared to published experimental data as well as an alternative numerical method in [8]. The temperature transforming method matched well with both the experimental data and numerical predictions.

The equivalent boundary conditions and simplified model used in this study were determined in an iterative manner. The progression of the melt front in the full model during orbit was compared to that of the simplified geometry while varying the simplified system's boundary conditions. The temperature profile along a horizontal cutline, cutline 1 in Figure 4, through the PCM and inner container wall for the full model at different instances during the sunlight phase is compared to that of the equivalent model in Figure 5. The trends match very closely with the equivalent model, under-predicting the full model by less than 20 K. This temperature difference was deemed an acceptable trade for the significant decrease in computational time.

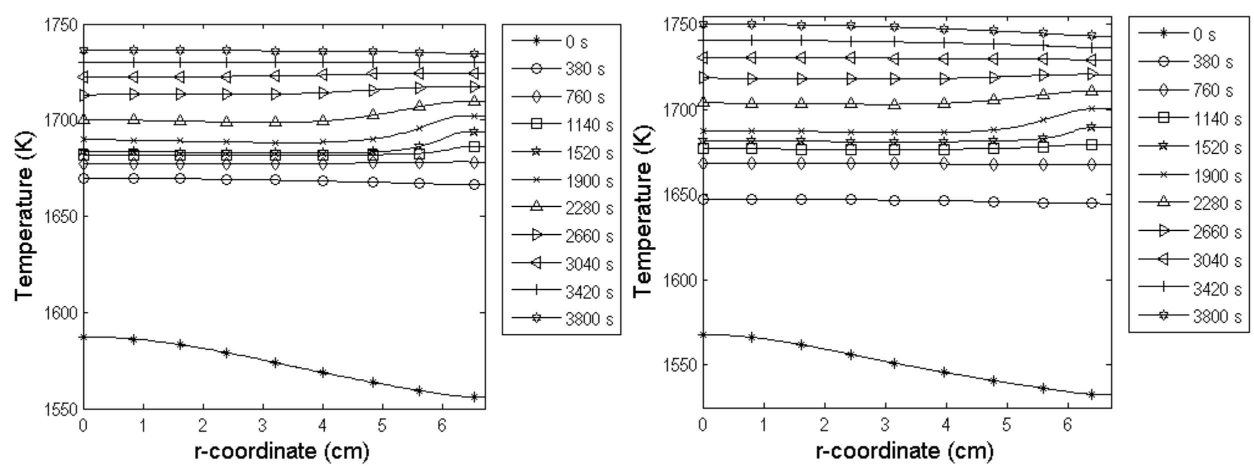

Figure 5: Temperature along cutline 1 in the full geometry (left) and equivalent geometry (right) as a function of time during a sunlight period.

IV. Results

Propellant flow through the molten silicon core was analyzed for a regular, small, station-keeping burn to lift the satellite from a degraded orbit back to its original orbit. Assuming a sixty second duration, a 1.9 N burn was required. For a specific impulse of 300 s, a propellant mass flow rate of 0.64 g/s and exit temperature of 1500 K were necessary.

A. Study 1

The simulated propellant feed was conducted via a channel through the insulation, standoff, and inner container wall and into a manifold located at the bottom of the inner container. The change in propellant temperature from feed inlet, where the initial temperature was 300 K, to outlet, for several different feed channel outer radii was determined. Channel radii ranging from 0.1—0.5 cm were evaluated by holding the inner radius of the propellant feed constant and adjusting the position of the outer wall. The temperature along the centerline of these different propellant feed tubes as a function of position is shown in Figure 6. The propellant feed inlet is located at z = 30 cm and the exit is at z = 10 cm. Propellant feed channels with large radii gain very little energy. Significant gains are not seen until the radius is decreased to 0.25 cm. For propellant feed tube radii between 0.15 and 0.25 cm, the flow will leave the feed tube and enter the manifold at temperatures between 650 and 1200 K. These predicted feed channel exit temperatures are used as inputs for Study 2.

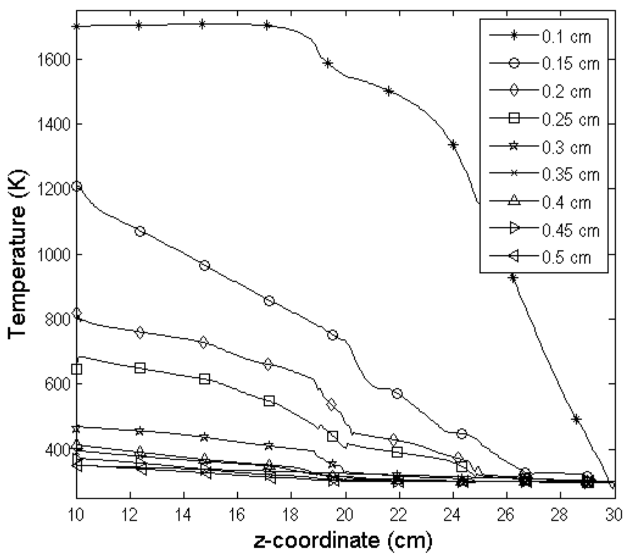

Figure 6: Temperature along cutline 2 for different entry tube diameters; note that the propellant flow moves in the negative z-direction.

B. Study 2

The manifold feeds two concentric rings located in the PCM with propellant flow. Using the temperature results of Study 1, manifold inlet temperatures ranging from 600 K to 1200 K were evaluated for three different PCM core heights: 12 cm, 15 cm, and 18 cm. The temperatures at the exit of the inner and outer heat exchanger rings as a

function of time for the 12 cm tall PCM core are shown in Figure 7. These results show that a manifold temperature of 1000 K is required to maintain the 1500 K exit temperature necessary to achieve a 300 second specific impulse. The outer ring reaches slightly higher temperatures as a greater contact area is combined with a slower flow speed. Exit temperatures drop on average 100 K over the course of the burn. Referring back to Figure 7, this manifold temperature requirement necessitates an inlet tube with a radius between 0. 2 and 0.25 cm. Although lower inlet temperatures were shown to be possible with the longer cores, the 12 cm m tall core is feasible and desirable if issues of mass and volume are of significant concern.

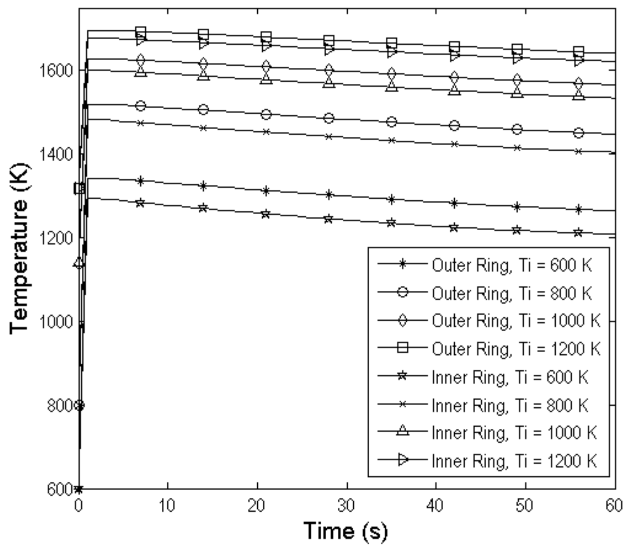

Figure 7: Temperature at the exit of the inner and outer rings of the heat exchanger for a variety of different inlet temperatures.

A closer examination of the 12 cm tall core case with an inlet temperature of 1000 K was conducted. The temperature profiles along two vertical cutlines, cutlines 3 and 4 in Figure 4, along the center of the flow path in both rings are shown in Figure 9 at different instances during the burn. The exit temperature (at z = 13 cm) drops 100 K in each ring. The majority of the energy addition occurs, as expected, in the lower portion of the core where the difference between the core temperature and that of the flow is greatest. From a thermal energy storage standpoint, the energy addition to the flow is an energy loss from the PCM, resulting in accelerated solidification. In the case of a design intended to minimize the mass of required silicon, this will need to be accounted for in order to ensure sufficient electrical output can be achieved during eclipse, even if thruster power is also required.

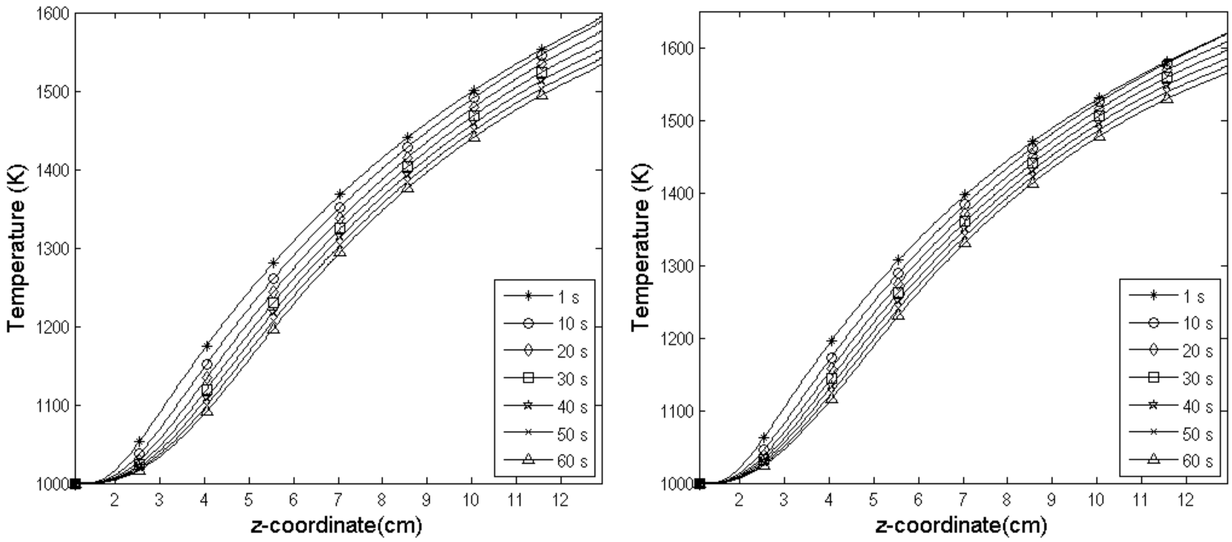

Figure 8: Temperature profiles along cutlines 3 (left) and 4 (right) for a 12 cm tall core at different times during a sixty second burn.

Assuming that the thruster was fired at the beginning of satellite eclipse, these temperature results were then used as the initial condition for the eclipsed portion of an orbit. The fluid was removed from the simulation as if an entry valve had been closed and the entirety of the flow exited through the nozzle. The resulting temperature profiles along the centerline of the PCM, cutline 5 in Figure 4, are shown on the left of Figure 9 for the first minute of thruster flow and on the right of Figure 9 for the remaining 1940 seconds of the eclipse. During the propellant firing, there is significant cooling in the bottom of the core due to the heat exchanger flow. The flow enters the core at a temperature initially 700 K lower than the core temperature at that location; this large temperature gradient causes the silicon in the lower 15% of the core to solidify within the 60 second burn. During the remainder of the eclipse, this cooled, solidified portion acts as a heat sink. This heat sinking, in addition to the general reduction of temperature in the upper core due to the flow, shortens total solidification time and increases sensible heat loss. In this system, the minimum emitter temperature (inner container top) was just above 1350 K. When there is no heat exchanger in the PCM, the average temperature of the emitter surface oscillates between 1450 and 1800 K. The cooler emitter temperatures can be also be attributed to the flow channel structure acting as a fin, thermally connecting the hot top of the core to the cool bottom.

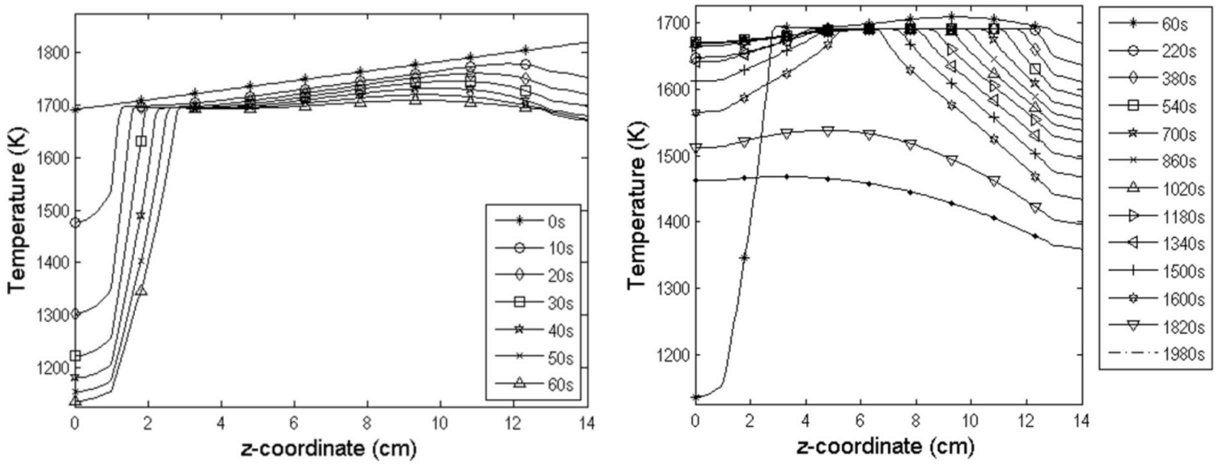

Figure 9: Temperature distribution along the core centerline during the first sixty seconds (left) and the remainder of the 2000 second eclipse (left).

The temperature histories of the top of the container, the emitter, during the eclipse portion of orbit with and without flow passing through the heat exchanger were used as an input to determine TPV power output. Gallium antimony (GaSb) cells were evaluated using the method outlined in [9]. The impact of the thruster flow is shown in Figure 10. The final power output of the TPV drops by approximately 20 W. At this point in the cycle this decrease is approximately 20% of the overall power.

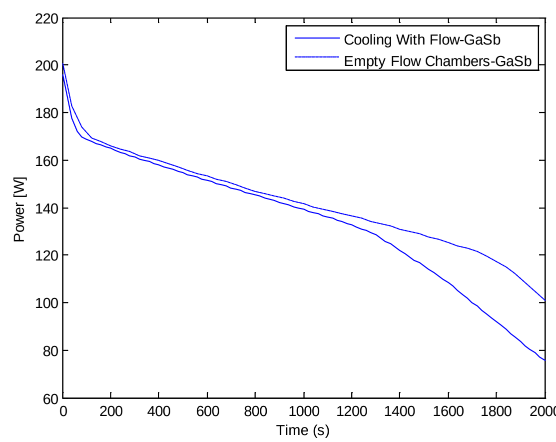

Figure 10: Power production by GaSb cells with and without propellant flow through the PCM.

V. Conclusion

A propellant flow through a heat exchanger imbedded in the phase change material of a thermal energy storage system was thermally characterized. The temperature that the propellant would be when it exited the heat exchanger and entered the nozzle was predicted. It was determined that ammonia could be preheated from 300 K to 1500 K. This translates to a 300 second Isp, and, at the prescribed flow rate, 1.9 N of thrust. For a 200 kg satellite in a 600 km orbit, the mass of propellant required by the thermal energy storage system to make up for drag losses would be 14 kg per year. To achieve the same effective velocity change, a resistojet with 100 second Isp and 100 mN of thrust would require 40 kg of propellant per year. The temperature of the emitter in the thermal energy storage system was used to determine power production. Passing the propellant through the phase change material decreased the temperature of the emitter, resulting in a 20% decrease in electrical power output. The power output may be improved once the system is optimized. Overall, the potential of a thermal energy storage system combined with an imbedded propellant heat exchanger as a power and propulsion system was demonstrated by this initial study.

Acknowledgments

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