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Robert C. O'Brien
Andrew C. Klein
William T. Taitano
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Thermal-hydraulics analysis of a radioisotope-powered Mars Hopper propulsion system

Robert C. O'Brien[1], Andrew C. Klein[2], William T. Taitano[1], Justice Gibson[2], Brian Myers[2] and Steven D. Howe[1]

[1]*Center for Space Nuclear Research, Idaho National Laboratory, Idaho Falls, ID 83415*
[2]*Department of Nuclear Engineering and Radiation Health Physics, Oregon State University, Corvallis, OR 97331*

Abstract. *Thermal-hydraulics analyses results produced using a combined suite of computational design and analysis codes are presented for the preliminary design of a concept Radioisotope Thermal Rocket (RTR) propulsion system. Modeling of the transient heating and steady state temperatures of the system is presented. Simulation results for propellant blow down during impulsive operation are also presented. The results from this study validate the feasibility of a practical thermally capacitive RTR propulsion system.*

Keywords: Radioisotope, Power, Mars, Hopper, Nuclear, RTG, Cermet

INTRODUCTION

A radioisotope-powered hopper concept has previously been identified by the Center for Space Nuclear Research as an advanced technology for the collection of ground-truth data over a significant portion of the Martian surface [*1, 2*].The hopper platform concept utilizes CO_2 from the Martian atmosphere as a propellant that is heated by a thermally capacitive mass of beryllium used to accumulate and store energy released from the decay of a ^{238}Pu heat source. During heat up of the thermal capacitor, the platform would simultaneously facilitate the collection of scientific data while compressing CO_2. When a storage tank is filled with compressed CO_2 and sufficient science data is collected, the CO_2 would be released through channels in the beryllium matrix and expelled through a nozzle to provide impulsive thrust. The thrust produced by this Radioisotope Thermal Rocket (RTR) would be used to provide a controlled vertical ascent and descent of the platform to another location. This process would be repeated numerous times, thus allowing the hopper to travel across the Martian surface over a period of months or years.

FIGURE 1. ***(Left)*** 3D render of a concept RTR engine ***(Right)*** Artist's impression of a Mars hopper in flight.

Thermal analyses have been conducted on a heat source and thermal capacitor concept design using the Solidworks, STAR-CCM+ and GAMBIT commercial software codes to identify the thermal design features of the propulsion system and to verify its feasibility. The baseline concept design is capable of storing a minimum of 10 MJ of usable thermal energy that is used to heat a total of 22 kg of CO_2 propellant. The thermal design of the system ensures that the beryllium matrix may achieve a peak temperature that is in excess of 1200 K. A 1 kW_{th} $^{238}PuO_2$ heat source encapsulated within a tungsten matrix was selected for volumetrically efficient integration within the beryllium matrix while providing resistance to impact and off-nominal operational conditions including Earth launch abort without the need for a separate aeroshell [*3-5*]. Initial thermal analyses indicated a suitable configuration of the thermal capacitor that will reach the design temperature after approximately 22 hours and will successfully transfer its energy to the CO_2 propellant in approximately 42 seconds resulting in a down range flight of 10 – 15 km to be achieved per cycle. The results from this study have been used to verify the technical feasibility of the thermally capacitive Radioisotope Thermal Rocket (RTR) concept.

STEADY STATE AND TRANSIENT HEATING ANALYSIS

Physical systems model

A steady state thermal analysis of an axially symmetric model of a preliminary RTR heat source design, geometrically meshed via the GAMBIT software code and illustrated in Figure 1, was performed using the STAR-CCM+ multiphysics code to determine both the maximum steady state temperature that may be achieved by the thermal design and the time required to reach the peak operational temperature. This study included conductive, convective and radiation heat transfer mechanisms to determine both the peak steady state temperatures of the idle RTR system and the transient time scales required to establish steady state conditions with respect to 3 varied carbon aerogel [*6*] insulation thicknesses of interest (1 cm, 1.25 cm and 1.5 cm). In order to maximize computational efficiency and minimize processing time, the flow channels containing stagnant CO_2 at Mars atmospheric pressure and the beryllium thermally capacitive matrix was treated as a homogenized mass, thus volume and mass averaged thermo-physical properties were used.

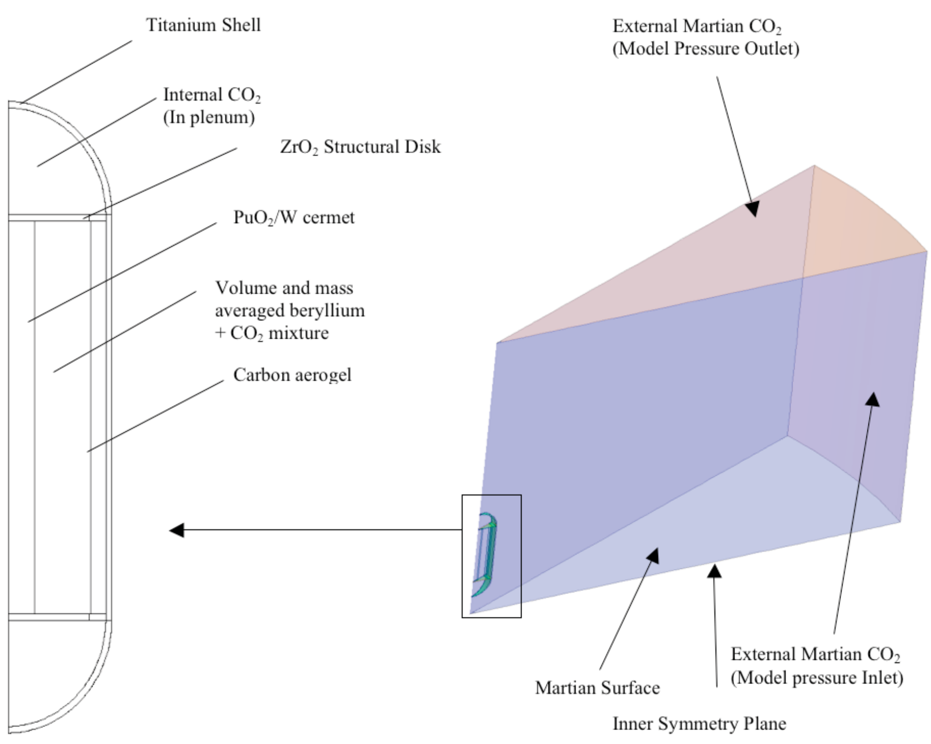

FIGURE 2. Schematic of the preliminary model for RTR steady state heat transfer

As indicated in Figure 2, a pressure inlet boundary condition was established at the periphery of the computational model such that the natural convective behavior of the system would draw Martian atmosphere into the system. Similarly a pressure outlet boundary condition was established on the upper surface of the model so as to provide a pressure sink to convected gases. Table 1 below indicates some of the key parameters and boundary conditions used within this model.

TABLE 1. Selected design parameters and boundary conditions used for thermal-hydraulics analyses.

Design Parameter	Data
Total usable energy stored	> 8 MJ
Radioisotope Heat Source Composition	$^{238}PuO_2$ encapsulated within a tungsten cermet
Heat source thermal inventory	1 kW_{th}
Insulation	Carbon Aerogel [*6*]
Insulation thicknesses examined	1.00 cm, 1.25 cm, 1.50 cm
Insulation thermal conductivity as a function of temperature (T)	$2.215\text{x}10^{-11}T^3 – 1.026\text{x}10^{-7}T^2 + 1.673\text{x}10^{-4}T – 8.599\text{x}10^{-3}$ (W/m/K) [*6*]
Heat capacitor outer radius	6.88 cm
Heat source & flow channel length	30 cm
RTR engine outer casing materials	Titanium
RTR engine casing thickness	0.5 cm
ZrO_2 support ring thickness	0.5 cm
Lumped Be & CO_2 Specific Heat at 1.01 kPa ($C_{p\ Be\ \&\ CO2}$)	$-1.058\text{x}10^{-8}T^4 + 3.436\text{x}10^{-5}T^3 – 4.045\text{x}10^{-2}T^2 + 21.51T – 1796.0$ (J/kg/K) [*7*]
Lumped Be & CO_2 thermal conductivity (k_{Be})	$1.096\text{x}10^{-4}T^{-0.6970}$ (W/m/K) [*7*]
$^{238}PuO_2$-W cermet Specific Heat at 1.01 kPa ($C_{p\ cermet}$) as a function of temperature (T)	$-1.073\text{x}10^{-10}T^4 + 3.961\text{x}10^{-7}T^3 – 5.172\text{x}10^{-4}T^2 + 0.3361T + 110.5$ (J/kg/K) [*7, 8*]
$^{238}PuO_2$-W cermet thermal conductivity (k_{cermet}) as a function of temperature (T)	$[0.66][-3.69\text{x}10^{-8}T^3 + 1.51\text{x}10^{-4}T^2 – 0.223T + 226.8]+[0.34][-1.813\ln\|T\|+14.71]$ (W/m/K) [*7, 8*]
Mars atmospheric pressure with respect to axial height, z (model inlet)	$1013.25 + [8.9453\text{x}10^{-2}]\times[1.389 – z]$ (Pa)
Propellant Inlet temperature	270 K
Propellant inlet stagnation pressure	2.85×10^6 Pa
Propellant mass flow rate	1 kg/s
Model outlet pressure	1013.25 Pa
Computational time steps used (temporal resolution)	1 Second, 5 Seconds
RTR thermal capacitor system mass	13.4 – 13.7 kg (insulation thickness dependant)

Transient heating and steady state results

Initial transient heating results were obtained using the system geometry defined in Figure 2 using radiative heat transfer only. These results for this can be seen in Figures 3 and 4 below. Finally natural convection physics was added to the model to include interaction with the surrounding environment. The results for the 1 cm thick insulation model are presented in Figure 4

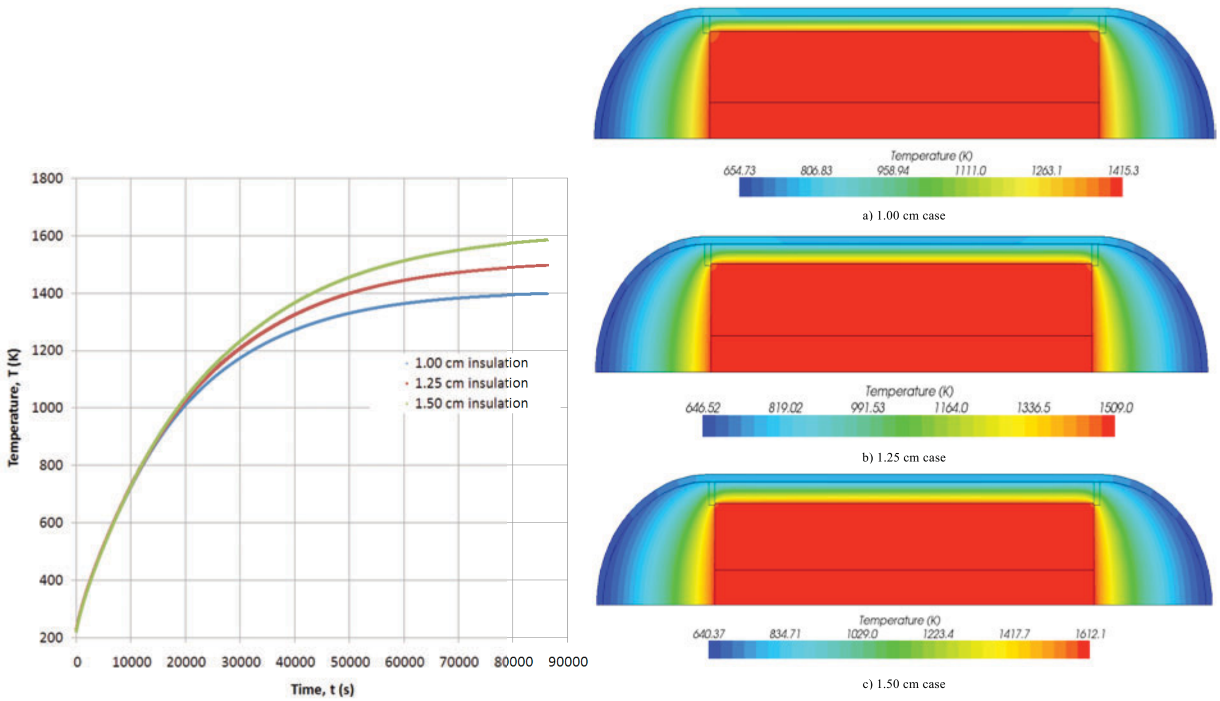

FIGURE 3. ***(Left)*** RTR heat source assembly average temperature plotted as a function of time during transient heat up via ^{238}Pu decay and radiative heat transfer. ***(Right)*** Steady state temperature field results for the RTR heat source assembly with respect to alteration of the carbon aerogel insulator thicknesses of a) 1 cm, b) 1.25 cm and c) 1.5 cm.

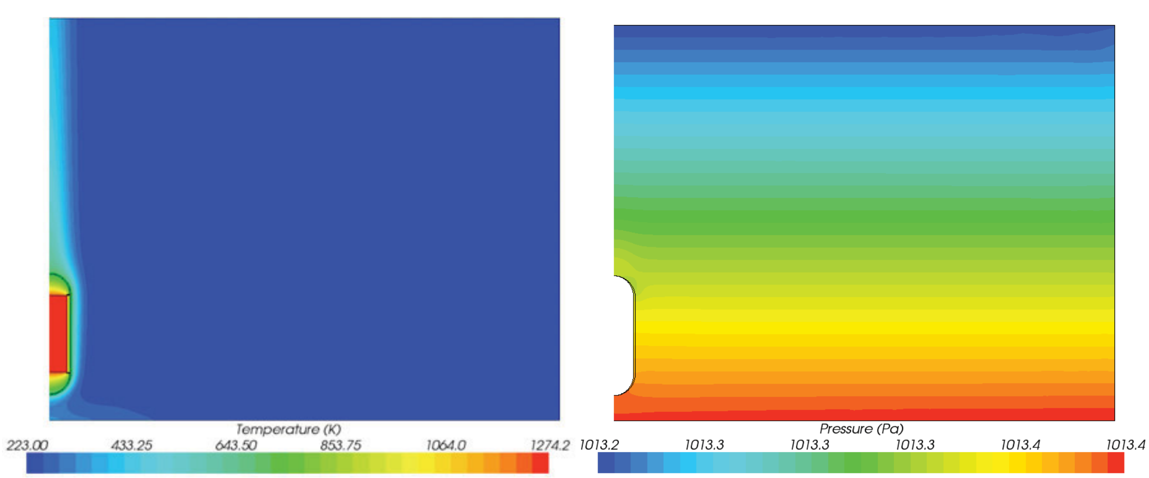

FIGURE 4. Steady state temperature and pressure field results for the RTR heat source model for a carbon aerogel insulator thickness of 1 cm.

THERMAL-HYDRAULICS ANALYSIS UNDER THE CONDITIONS OF CO_2 PROPELLANT BLOWDOWN

Physical systems model

In order to determine the efficiency of heat transfer between the hot thermal capacitor and the propellant within a preliminary design of the Hopper RTR consisting of 1056 coolant channels, a 22.5 degree azimuthally symmetric model was constructed using Solid-Works. The overall length of the RTR engine modeled was 60.88 cm which included a 30 cm long Be thermal capacitor matrix, a 24 cm long converging-diverging nozzle driven by an exit to throat ratio (A_e/A_t) of 50, and a 6.88 cm long upper inlet plenum. Through the steady state analyses described above, it was assumed that a steady state temperature in excess of 1200 K may be achieved for a matrix of the size considered in this model. Thus for the purpose of modeling the transient and heat transfer from the beryllium matrix to the propellant under propulsive operation, an initial matrix temperature of 1200 K was assumed in conjunction with a propellant inlet stagnation pressure of 2.85 MPa and injection temperature of 270 K. An overview of the propellant blowdown model geometry is presented in Figure 5 below.

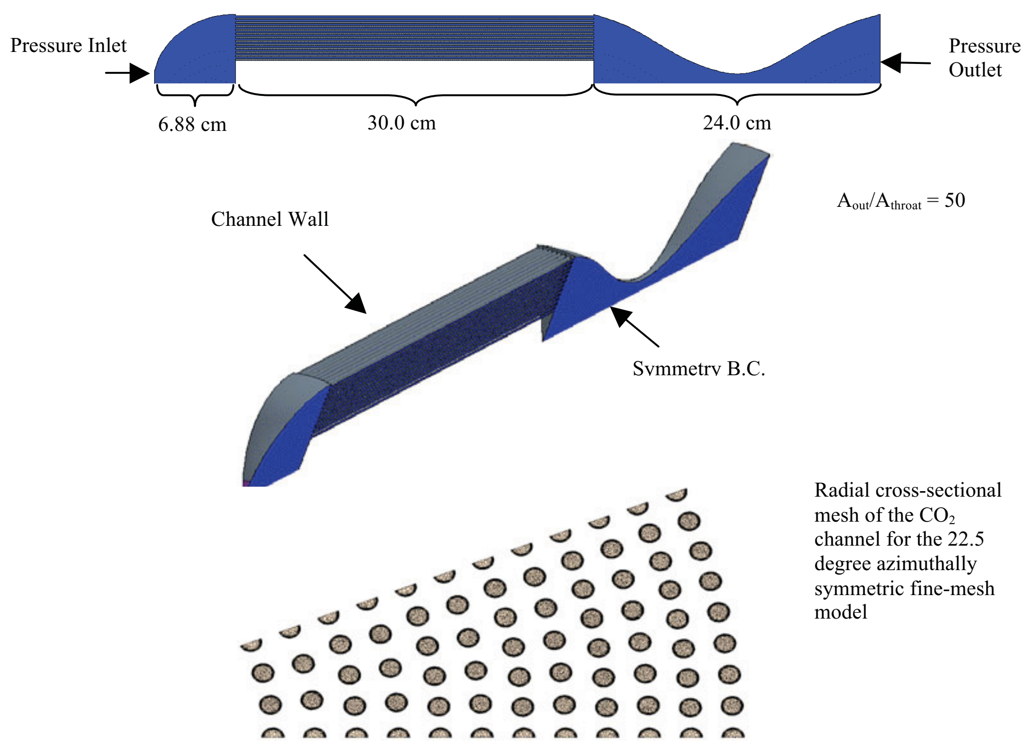

FIGURE 5. Schematic of the preliminary propellant blowdown model. Shaded regions are propellant flow channels.

Propellant blowdown transient results

In order to ensure validity of the model, which was evaluated under turbulent conditions via the "k-ω SST Menter model" within the STAR-CCM+ code, the Reynolds number through the flow channel design as a function of axial length was determined using the code. Laminar flow analysis is valid for $Re_H < 1300$ while the transition regime lies within $1300 < Re_H < 2500$ and a higher Re_H is fully turbulent. The results from a fine and coarse mesh size were both comparable and indicated that the propellant flow through each channel is indeed turbulent. These results are indicated in Figure 6 below.

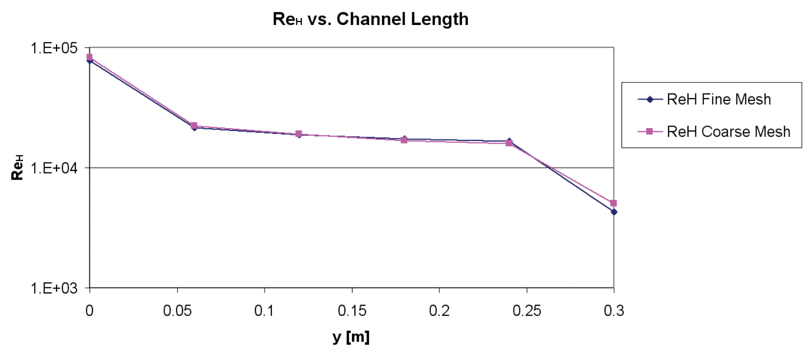

FIGURE 6. Reynolds number determination as a function of flow channel axial length.

A set of complete temperature, pressure and propellant velocity profiles of the 3D model were generated for each operational time step over a period of 20 seconds. Figure 7 below illustrates the system temperature profile at startup where the average propellant matrix-outlet temperature is of the order of 1000 K.

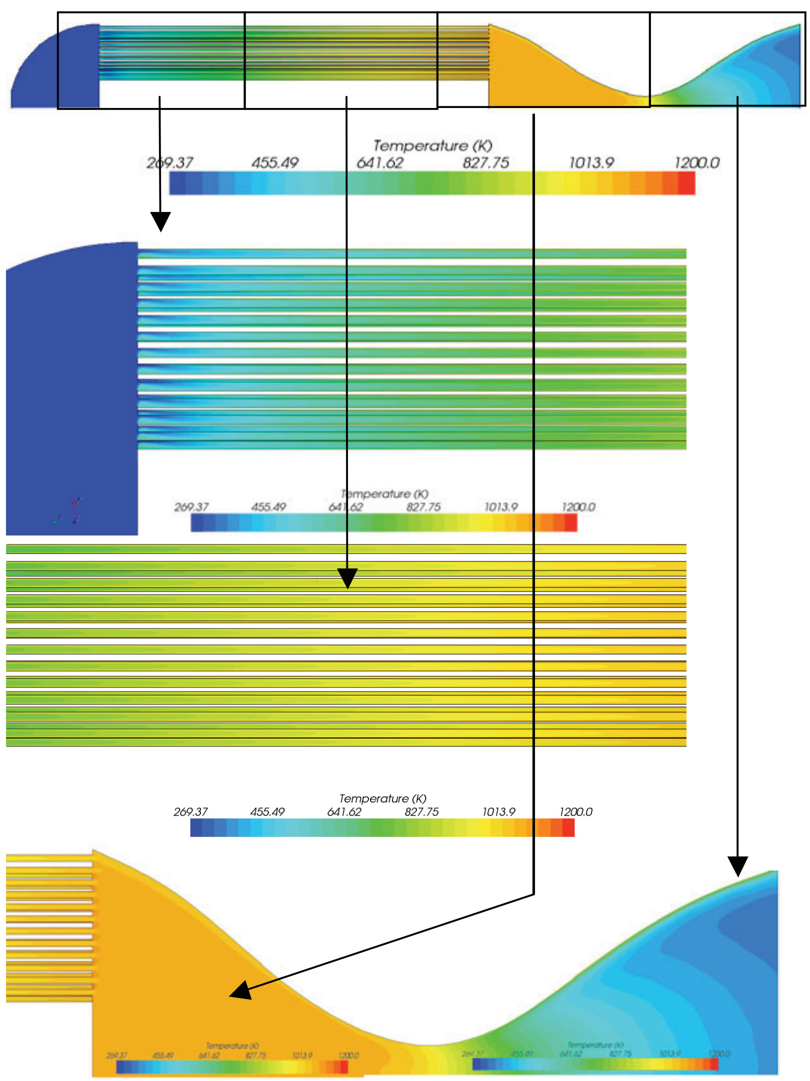

FIGURE 7. Temperature profile of RTR system at startup of propulsive operation.

The average data generated by the temperature profiling with respect to time under propulsive operation for both the beryllium matrix and the propellant as it leaves the matrix is presented in Figure 8 (left).

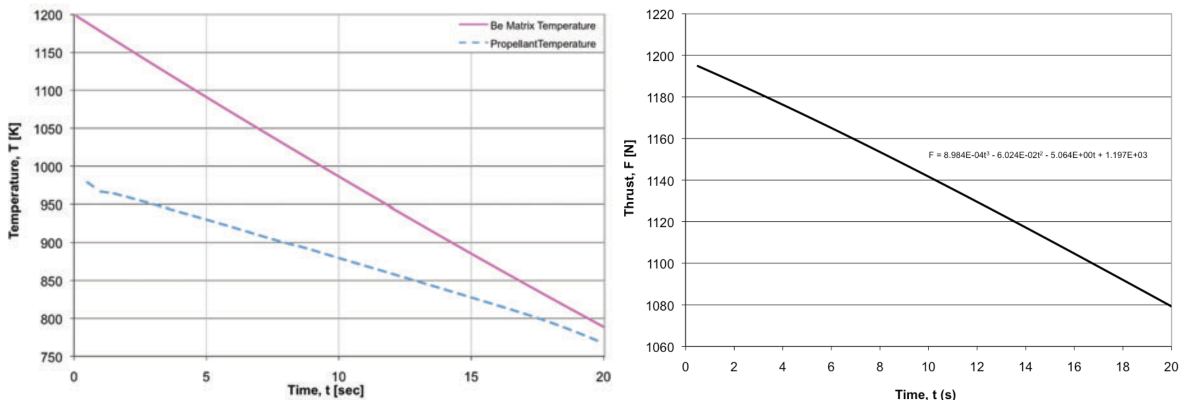

FIGURE 8. ***(Left)*** Average propellant matrix-outlet and beryllium matrix temperatures as a function of time. ***(Right)*** Engine thrust as a function of time.

$$Thrust,\ F\ (N) = \dot{m} v_e \tag{1}$$

The thrust generated by the RTR engine may be determined as a function of time via the use of Equation 1 above in conjunction with the propellant mass flow rate boundary condition, m-dot, of 1 kg/s and the time dependant exhaust velocity v_e.

CONCLUSIONS

As indicated by the transient and steady state thermal analyses presented within this paper, an RTR heat source and thermal capacitor assembly may be designed to achieve an average peak steady state temperature in excess of the 1200 K temperature requirement defined by the CSNR Mars Hopper concept feasibility study baseline model [*1*]. Early concept designs for RTR systems called for the use of vacuum insulation schemes that were in part susceptible to infiltration via CO_2 and thus, introduced a significant perceived risk to insulation failure over prolonged operation. The results from this study indicate that advanced solid state insulation schemes may be used to eliminate the need for vacuum insulation while achieving the required operational temperatures with small radioisotope thermal inventories.

The acceleration due to gravity on Mars is approximately 3.73 m/s^2. Thus, the take-off thrust to weight ratio of a Mars hopper with a wet mass of 100 kg and propelled by the engine modeled in this paper would be in excess of 3.2 and hence exceeds the requirements for a thrust to weight ratio of 3 outlined by the CSNR baseline study [*1*].

Examination of the temperature profiles indicates that the initial design presented has a startup heat transfer efficiency of approximately 83 % under propulsive operation. Due to the increase in thermal conductivity of the beryllium matrix as its temperature is reduced by the passing propellant, the heat transfer efficiency increases to around 96 % at the end of a thermal cycle. The effect of this physical condition results in the linearization of thrust decay with respect to time. Clearly maximization of the initial heat transfer efficiency is vital for optimization of system performance. Current and future work has and will continue to improve on the system design such that the average propellant matrix-outlet temperature is approximately equal to the average matrix temperature.

NOMENCLATURE

k = thermal conductivity
C_p = Specific Heat Capacity
RTR = Radioisotope Thermal Rocket
z = axial height (m)
$\dot{m}$ = m-dot; mass flow rate (kg/s)
t = time (s)
v_e = exhaust velocity

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