Transient Thermal Analysis of the 12.5 kW HERMeS Hall Thruster

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I. Abstract

NASA's Dawn spacecraft entered orbit around the dwarf planet Ceres on March 6, 2015 thereby becoming the first mission to orbit two celestial bodies outside of the Earth-moon system. This feat was made possible due to the use of electric propulsion (EP) to provide the 11 km/s of delta-V that was necessary. This type of propulsion is also enabling for many of NASA's future missions, such as the proposed Asteroid Redirect Robotic Mission (ARRM). Specifically, the Hall Effect Rocket with Magnetic Shielding (HERMeS) is the propulsive design baseline due to its long operational lifetime and high specific impulse. Extensive modeling is required to verify the thruster's predicted thermal performance and survival during the bounding mission phases. Since the thruster would constrain plasma on the order of tens of thousands of hours, components in the thruster must be able to accommodate much higher sustained temperatures than typical spacecraft hardware. Furthermore, the high temperatures necessitate care in analyzing thermal gradients within the thruster. In this paper, we describe a thermal model of the HERMeS thruster and focus upon the thruster's transient performance under different initial firing conditions. The thruster's transient thermal response is critical to verify margin in the thermal stresses inherent in the thruster and to determine safe start-up procedures to protect the thruster's critical components. This model is used to perform transient thermal analysis of the thruster during startup and shutdown, provide temperature maps for structural analysis, and plan for thermal cycle testing conducted at JPL. The thermal cycle testing includes thruster starts at the cold temperature limits and operation at maximum expected external heat flux. This work will support laboratory environmental testing of the HERMeS thruster, which is being conducted at JPL in 2016.

II. Introduction

The Hall Effect Rocket with Magnetic Shielding (HERMeS) is an electric thruster being jointly developed by the Jet Propulsion Laboratory (JPL) and Glenn Research Center (GRC) [1]. The HERMeS thruster is intended to be the primary method of propulsion for the proposed Asteroid Redirect Robotic Mission (ARRM). ARRM seeks to launch a robotic probe to a near earth asteroid, land, and retrieve a roughly two ton boulder from the surface of the moon and return it to a lunar orbit. Once in lunar orbit, the asteroid would be visited by astronauts who would obtain samples for return to Earth and demonstrate other aspects of deep space operations with the Orion Capsule. More detail on this planned mission can be found in work by Brophy [2].

Electric propulsion (EP) is appealing for this application as it has a very high specific impulse, which can generally be understood as to require less fuel mass to achieve the necessary change in spacecraft velocity to accomplish a mission. NASA's Dawn mission was the first human-made spacecraft to visit two celestial bodies on the same mission, which was made possible due to its use of the NSTAR grid ion thruster.

EP uses electric and magnetic fields to ionize and accelerate charged particles (ions), transferring momentum to a spacecraft. These charged particles exist as a plasma that is contained in the thruster. Neutral particles are injected from the anode, ionized by electrons sourced from the cathode, and accelerated downstream by electric fields. The gridded ion thruster on Dawn used a pair of high-voltage grids in order to accelereate ions into space. While this was proven successful, the lifetime of ion engines of this type is limited by the frequent high energy ion collisions with the grids. NASA conducted extensive testing of ion engines, prior to the Dawn mission, including NASA's Deep Space 1 mission and the development of NASA's Evolutionary Xenon Thruster (NEXT) [3,4,5]. Hall thrusters, like HERMeS, differ from grided ion thrusters in electrons are constrained by magnetic fields in an open circular channel. The ions do not pass through a pair of high-voltage grids but still suffer lifetime issues associated with erosion of the

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thrusters surfaces due to ion impact. In a Hall thruster, magnetic fields are used to constrain the electrons necessary to ionize the propellant [6]. A cross section of a typical Hall thruster can be seen in Figure 1.

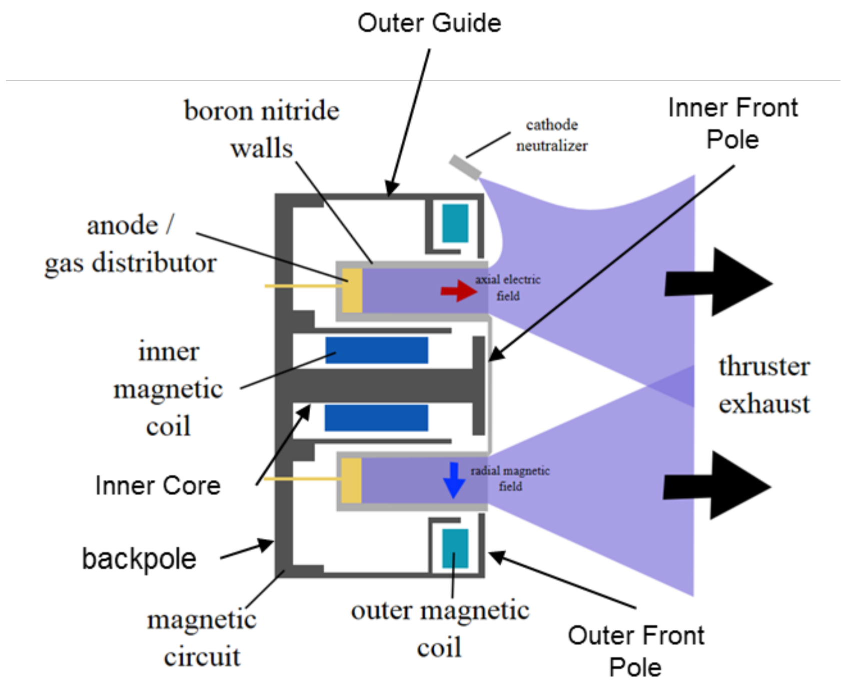

Figure 1: Cross section of a typical Hall thruster. Note the example above has a cathode mounted outside of the magnetic circuit, whereas the cathode for HERMeS is mounted along thruster centerline through the inner core.

Where HERMeS differs from traditional Hall thrusters is that the magnetic fields are manipulated in such a way as to minimize high energy ion collision with thruster surfaces in order to minimize erosion. This shielding enables magnetically shielded Hall thrusters to work for longer durations than had previously been considered for electric propulsion. A schematic and photograph of the HERMeS thruster can be seen in Figure 2 and Figure 3, respectively.

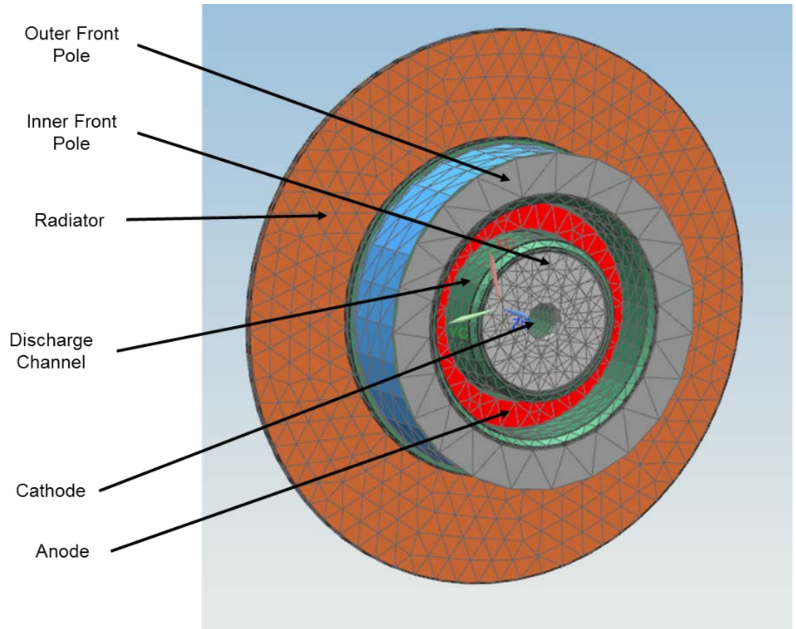

Figure 2: Schematic of HERMeS layout

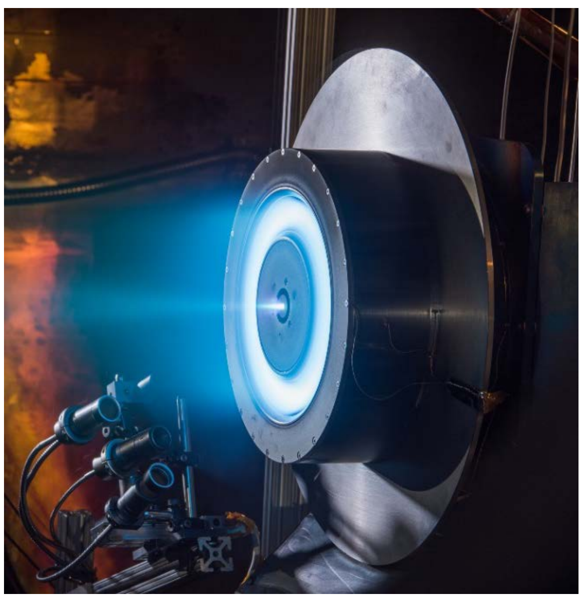

Figure 3: Operation of HERMeS TDU-1.

While the magnetic shielding protects the thruster from high energy ion bombardment, it is still expected to endure high heat loads for the extreme operational duration assocatied with these devices. These thrusters must be able to absorb and safely emit heat loads in the kilowatt range passively, without significant heat bleeding back to the spacecraft. Furthermore, given the nature of their mission, they will be cycled on and off hundreds to thousands of times, from both hot and cold states in hot and cold thermal environments. To help ensure the success of these thrusters through the life of the mission, modeling and extensive testing is required.

A detailed study of steady state Hall thruster thermal behavior done exclusively by IR was performed by Mazouffre [7]. This study was primarily looking at plasma-surface interactions but was useful in understanding some of the features of steady state Hall thruster operation. NASA has conducted research into Hall effect thrusters, such as the work by Jacobson [8]. While this paper is primarily a review of Hall thruster testing in 2004, it does provide some insight on the transient heat up behavior of a few Hall thrusters from a room temperature like condition. The thrusters mentioned in this work typically heated to steady state within approximately 4 hours. Thermal and structural modeling of Hall thrusters was done by Yim [9] to help understand temperature response of the thruster. However, in this test, an analytical method was used to estimate the steady state behavior since the collection of temperature data was secondary as an experimental goal. Still their work showed good agreement with the temperature data collected and was useful as reference for this work.

Thermal development testing carried out by Snyder et. al. [10] was particularly relevant to the present work. During the thermal development test of the NEXT thruster, the primary objective was to subject the thruster to expected maximum external heat loads and a heat load by which they would reach the temperatue limits of the hardware. Since the spacecraft NEXT could be mounted to was not prescribed at the time, conservative numbers were used.

The first version of HERMeS was the Thruster Development Unit (TDU)-1, built and operated at GRC. The second version, TDU-2, will be operated at JPL for the environmental test. The modeling effort discussed in this work was created to support the environmental testing of TDU-2, to take place in May of 2016. The environmental test is designed to stress the thruster, in both hot and cold extremes, similar to the testing performed on NEXT. The worst case hot environment is approximated by the thruster operationg at full power with an exhaust vector pointed at the Sun at 0.8 AU. The worst case cold is igniting the thruster from a -110 °C cold soak. For the environmental test, these conditions are intended to be replicated as close as possible to assess the robustness of the thruster. Limitations of HERMeS testing required the use of the models to determine how the test facility could most closely simulate the flight heat loads. The results of this testing and a more detailed description of the test setup will be discussed in a later work.

A more detailed primer on the HERMeS design as well as other modeling efforts can be found in the work of Hofer [1] and Kamhawi [11]. Thermal modeling has previously been conducted at GRC for TDU-1 under steady-state conditions, which is not the focus of the present work. This work discusses the use of a thermal model to study transient thruster behavior for the upcoming environmental test campaign to be conducted at JPL.

In this paper, the capabilities of a transient thermal model of the HERMeS TDU-2 thruster are demonstrated. The development of the thermal model is described from its beginnings as a model of TDU-1 through its current use as a tool to inform test design. This includes the addition of several components of the model which were included expressly to accommodate transient predictions. The results of transient modeling of the TDU-2 environmental test will be shown and analyzed with some rough comparison to TDU-1 data. Hall thrusters are unique amongst spacecraft components in that they maintain relatively high temperatures for extreme durations, compared to other typical spacecraft components. This work is intended to highlight some of these issues for use in future work.

III. Model Description

This model was built from a thermal model of the HERMeS TDU-1 unit that was developed at JPL and validated with experimental data. The thermal models for TDU-1 and TDU-2 were both built in Siemens NX9.0- Space Systems Thermal (SST). There is a high degree of commonality between TDU-1 and TDU-2; the main difference being that TDU-2 is designed to survive launch loads while TDU-1 is not.

NX-SST was primarily used due to the ease of incorporating mechanical drawing files into the thermal model. Typically, mechanical drawing files of individual components were imported into NX and their geometries were idealized and simplified to facilitate mesh generation. The meshed idealized parts were then put into an assembly FEM where they were assembled in the configuration of the thruster.

Once the desired geometry was established, the material and optical properties were input into the model as well as contact conductances between the components. Most interfaces are mechanically attached without any thermal interface material and so were generally modeled as bolted interfaces, with respect to contact conductance. This yielded several areas in the model where contact conductances were quite low, specifically with ceramic components that interfaced with metal parts.

While there is some variation in the optical properties of thruster components as a function of temperature, in ground testing, most of the external surfaces tend towards high emissivity due to sputtered carbon being deposited on the exterior surface. When testing a thruster in a vacuum chamber, large carbon targets are used to prevent damage to the chamber and minimize backsputtered material. The sputtered carbon deposition will not occur in flight because the high energy ions will exit into space. Most of the high temperature components did not appear to undergo significant optical or thermophysical property changes, however, the carbon deposition did have a tendancy to alter the external facing surfaces emissivity the longer the thruster was run.

Due to the large magnitude of temperature change of the desired cycles for TDU-2 testing, one item of concern was the resistivity in the inner and outer electromagnetic coils in the thruster. Resistivity per unit length of copper does vary with temperature and with the quantity of copper in the coils, some variation was expected over the full range of temperatures the thruster would observe. The magnitude of these changes can be seen in Figure 4. As a result, we added a FORTRAN subroutine to the model that alters the coil power dissipation as a function of temperature, since the current in the coils generally remains constant for a particular throttle setting.

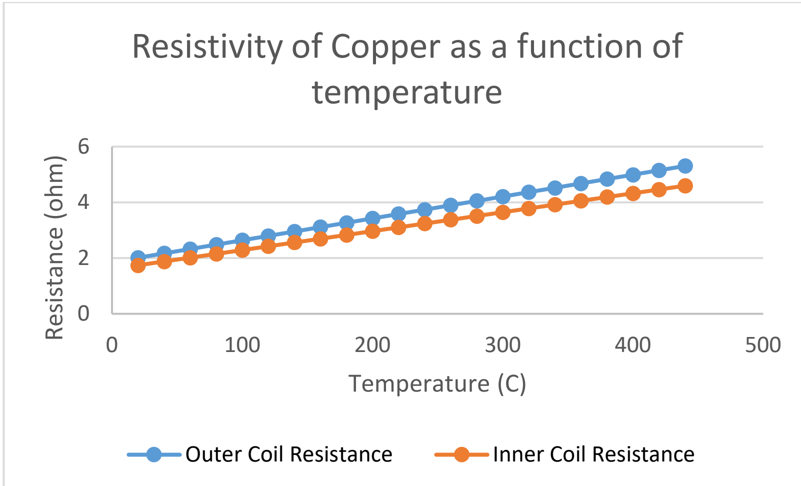

Figure 4: Resistivity of Copper Coils as a function of Temperature

In order to validate the TDU-1 model, the heat loads were adjusted to bring the temperature response of the model to reflect the temperature response observed in testing at GRC. This is because it is possible to estimate the heat loads based on plasma modeling, but there is variability. This variability is what is used to validate the thermal model since the researchers verified the validated thermal loads were reasonable with respect to the plasma modeling. The TDU-1 model was validated to within 20°C, or around 10% relative error in almost all temperature probe locations.

The geometry between TDU-1 and TDU-2 is very similar save for the addition of ribs on the radiator to improve stiffness. Except when necessary, all thermal conductances, material, and optical properties were unchanged when converting the TDU-1 model to TDU-2. This TDU-2 model is the one used in this work to predict transient performance of the thruster in environmental testing.

IV. Modeling Results

In order to prepare for the TDU-2 environmental test, the model was first used to predict the temperature response of the thruster as it would be in flight and the Worst Case Hot (WCH) condition. The thruster was assumed to be mounted at the trailing end of a spacecraft bus with an interface temperature of 150°C. For this condition, the thruster was pointed directly at the sun with the thrust vector parallel to the solar vector at 0.8 AU (average solar flux: 2133 W/m^2). The exposed surfaces of the thruster also view deep space at a temperature of 2.7 K. The back side of the radiator surface is assumed not to radiate since it is shadowed by the spacecraft bus in the assumed configuration and is intended to be conservative. This is a conservative estimate and was done to limit the amount of soak back heating from the thruster to the spacecraft bus. A temperature map of the thruster can be seen in Figure 5.

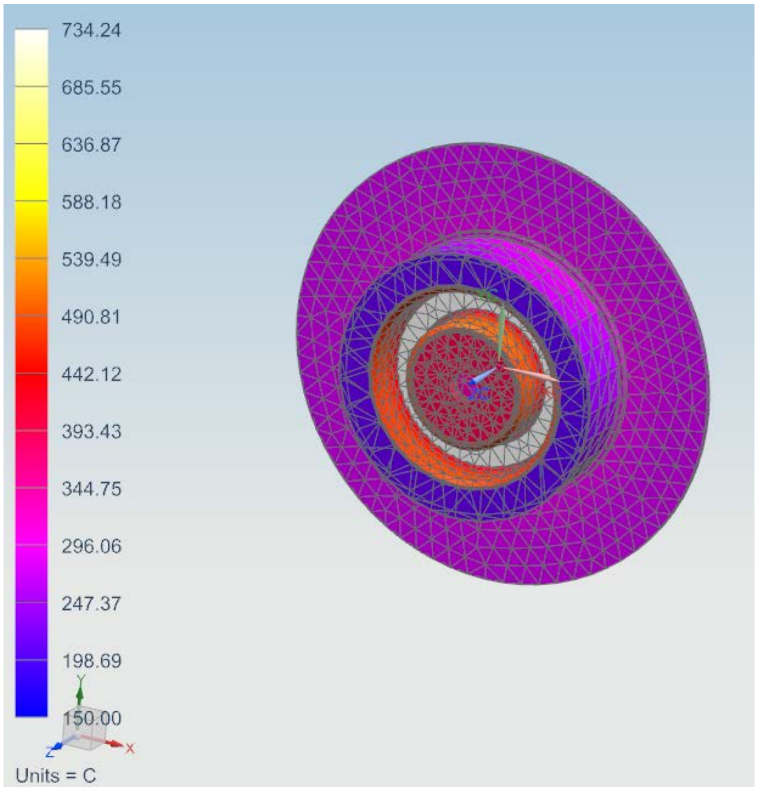

Figure 5: Steady State Temperature map of TDU-2 at 0.8 AU

Note the hottest component is the anode and this is due to its structure and its large exposure to plasma loading in the discharge channel. For the environmental test, however, the researchers were primarily interested in the status of the back pole of the thruster. The backpole control temperature reached roughly 350°C. This temperature would then be the target temperature to reach during environmental testing.

In the environmental test, it is difficult to get an exact representation of the thermal environment predicted on flight due to test equipment limitations. Infrared (IR) heat lamps are not able to be mounted down stream of the thruster to simulate the sun since the high energy ions would eventually destroy them, so lamps must be mounted radially around the thruster. As a result, the IR lamps in test are used more as test heaters in order to heat the operating thruster to the elevated temperatures expected on flight.

Another key difference is that the sink temperature in test is approximately -20°C in the vacuum chamber that will test TDU-2, which is much warmer than a deep space sink of 2.7K. This means that the lamp power required to raise the temperature of the control TC to the desired temperature will be much less than the expected solar flux at 0.8 AU. A schematic of the thruster in the test setup can be seen in Figure 6.

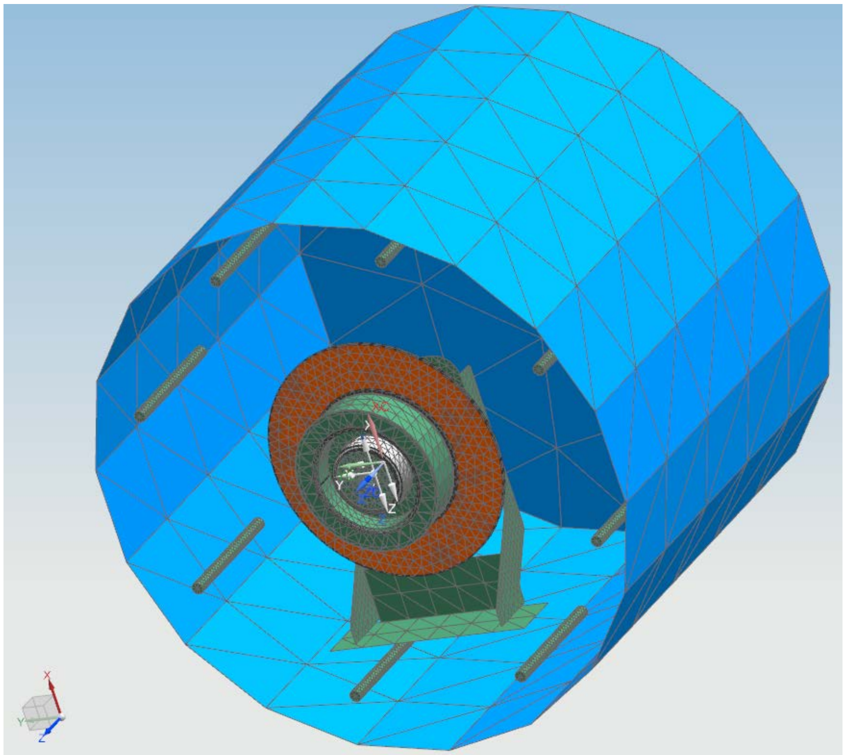

Figure 6: Schematic of HERMeS TDU-2 in Environmental Test Shroud with IR lamps shown

Note that the eight IR lamps are shown in the test setup image, but only the thruster stand is shown. There is a rack mount on which the thruster stand sits that is not shown for clarity. Using a steady state model, it was estimated that the HERMeS TDU-2 control temperature would reach approximately 325°C and that approximately 280 W of emitted power per lamp would be required in order to elevate the control temperature to 350°C.

The planned test thermal cycle is a cold soak at -110°C, facilitated by flowing liquid nitrogen (LN2) through the thermal shroud. The 'cold start' will consist of igniting the thruster at -110°C. After the cold ignition of the thruster, the thruster will be allowed to warm up and the shroud temperature will be allowed to float. During the warm up phase the IR lamps will be activated in order to further warm the thruster to the elevated temperature. Once thermal steady state is reached, the thruster will be deactivated and quickly reignited, demonstrating a 'hot start' of the thruster. This cycle is repeated 2 additional times. Once the lamps and thruster are deactivated, LN2 is flowed through the shroud to begin cooling. The cool down occurs in two stages; the first has the shroud open to the chamber environment as shown in Figure 6 and the second an MLI blanket will be rolled down over the shroud opening, closing the shroud off from the chamber environment. The shroud is open until the thruster cools to roughly room temperature, ~20°C in order to prevent the shroud MLI cover from impeding thruster cooling.

The results in this work will be presented as the transient temperature profile of the control TC on the backpole of TDU-2 for one complete thermal cycle. In Figure 7, the control temperature profile is shown for the heat up from a -110°C to the steady state temperature of 351°C.

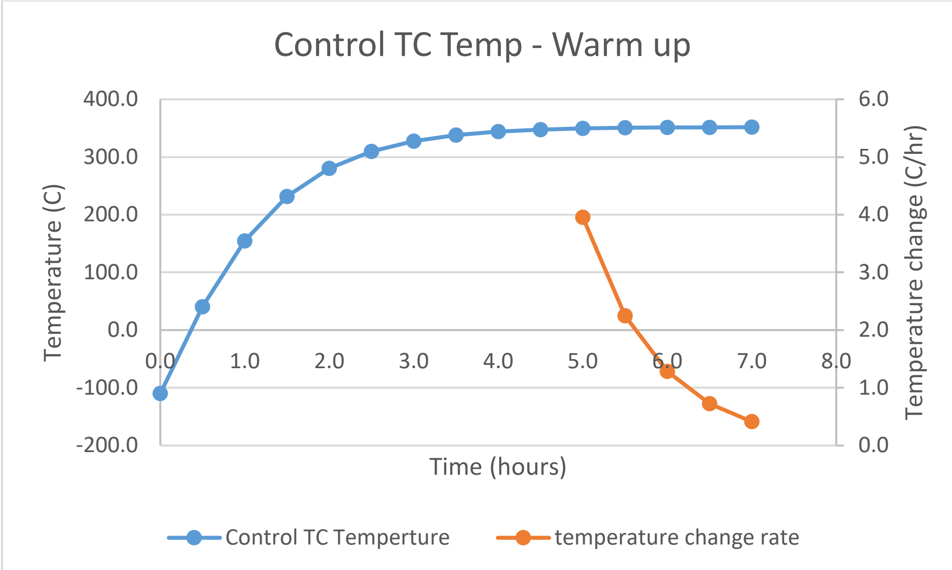

Figure 7: Backpole Control TC Temperature history, transient warm up

The control TC reaches the desired steady state condition of <2°C/hr in roughly 5.5 hours of operation. Note that the lamps are on in this configuration, at roughly 280 W per lamp. Also, in Jacobson's work [8], warmup for two different Hall thrusters of roughly the same size to similar steady state conditions from room temperature took approximately 3.5 hours. In HERMeS' case, HERMeS is starting from a much lower starting temperature (-110°C) but is the same order of magnitude.

Figure 8 shows the transient cool down to room temperature and on to -110°C. The cooldown to room temperature with the shroud door open takes approximately 4.2 hours. Note that the lamps are off in this configuration and the shroud has LN2 flowing through it.

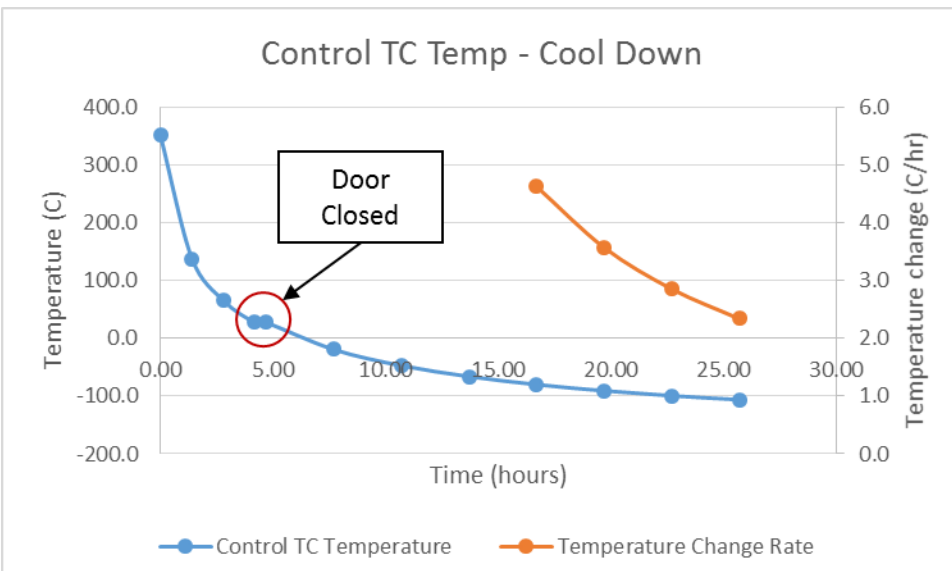

Figure 8: Control TC cooldown to -110°C

Once the door is closed the thruster continues to cool. It is important to note that it does not quite reach steady state by the time it reaches -110°C so it is predicted to be slightly over driven to this temperature. The lamps will be used to hold the thruster at -110°C once this temperature is reached. The cooldown from 20°C to -110°C once the door is closed takes approximately 21 hours. Note the configuration in this case is with the shroud door closed, lamps off, and thruster deactivated.

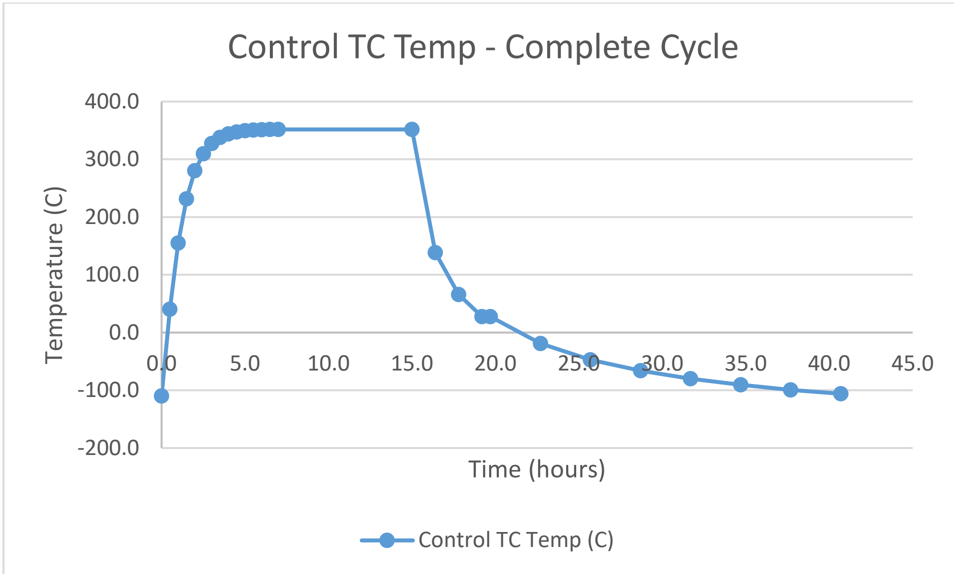

Figure 9: Control TC Transient Temperature history, complete cycle

This means a complete thermal cycle of the thruster is predicted to take approximately 31 hours. However this does not include a dwell time at the WCH at steady state. A more accurate representation of the thermal cycle can be seen predicted in Figure 9, where an 8 hour dwell at the WCH point is included. There is no data from earlier TDU-1 testing that matches a profile of this type that could help the researchers validate the transient data at this time, so this model is relying on the previous validation against TDU-1 steady state values. Data from the forthcoming TDU-2 environmental test will help to validate the thermal model further. Similar thermal cycling for NEXT has yielded a thermal cycle of approximately 30 hours [10] so the authors believe this estimate to be reasonable.

It is important to note the thruster is generally only coupled to its environment via radiation. There is some conduction along the thruster mount to the chamber, but this is idealized to a nominal value of 0.05 W/K due to the many thermal interfaces between the thruster mount and the chamber. The thruster depends almost entirely on passive radiation to the environment in order to maintain its temperature. This puts a strong emphasis on the optical properties of the exterior facing surfaces of the thruster, which are either carbon or black oxide coated in order to maintain high emissivity. For instance, it is known from TDU-1 model validation efforts that more than 55% of the heat load imposed on the discharge channel is radiated away to space, without being conducted into the thruster. A notional energy balance produced using the model while validating can be seen in Table 1.

Table 1: Notional Energy balance on HERMeS thruster

Thruster Heat Loads	Power (W)
Plasma Loading	1732
Inner Coil	83
Outer Coil	71
Heat Dissipation to space	**Power (W)**
Discharge Channel/anode	-751
Inner & outer front pole	-229
Outer Guide	-193
Radiator	-479
Mount Spool	-66
Heat Conducted to Spacecraft	**Power (W)**
Mount Spool	-168

Note that the conduction to the spacecraft is predicted to be high, this is due to the assumptions in the model. The model does not include the spacecraft or spacecraft bus. To be conservative, no radiation is assumed to be emitted from the spacecraft side of the radiator and the spacecraft interface is a fixed temperature (150°C). When the thruster is eventually modeled coupled to the bus, this number should go down.

This reliance on radiation in order to shed heat from the thruster also places a premium on the strong thermal contact conductance of the thruster interior components in order to provide an adequate path for heat to travel from the interior of the thruster to the radiating surfaces. Many of the design changes between TDU-1 and TDU-2 have focused on maintaining strong physical contacts between components.

V. Conclusion

A prediction of the HERMeS TDU-2 transient thermal behavior under environmental testing has been presented, along with analysis of the results. The TDU-2 model discussed in this work is based on a validated TDU-1 model with minor changes to accommodate design changes related to TDU-2. The TDU-2 transient model seeks to provide an accurate representation of the transient temperature response that would be expected to be observed by TDU-2 under its upcoming environmental test. The environmental test is designed to thermally stress the thruster in several worst case cold and worst case hot cycles between -110°C and 350°C. These thermal environments are similar to the NEXT ion thruster testing previously conducted at JPL, with the hot case modified to represent an expected thermal environment at 0.8 AU.

Acknowledgments

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