Thermal Design and Analysis of Green Propulsion Dual Method CubeSat

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Green Propulsion Dual Method (GPDM) is a collaboration between NASA Marshall Space Flight Center (MSFC) and Georgia Tech to create a 6U CubeSat demonstrating the use of the green propellant Advanced Spacecraft Energetic Non-Toxic (ASCENT) in two propulsion systems: using a traditional chemical thruster and an electrospray thruster system. The driving thermal considerations includes a planned orbit always in Sun view without any time in Earth's eclipse, and a chemical thruster reaching 1300°C during firing. Thermal design and analysis on CubeSats are not always emphasized, particularly in experimental or academic environments. This paper will describe the thermal design and analysis performed on GPDM and the mitigations implemented for this mission, which will highlight some of the thermal challenges unique to experimental small satellites.

Acronyms and Nomenclature

ADCS	=	Attitude Determination and Control
ASCENT	=	Advanced Spacecraft Energetic Non-Toxic
°*C*	=	degrees Celsius
COTS	=	Commercial off-the-shelf
cm	=	Centimeter
CPRS	=	Compact Pressure Reducing System
CubeSat	=	Nanosatellite using a standard size and form factor
EPS	=	Electronic Power System
FGM	=	Foxglove Model
GEVS	=	General Environmental Verification Standards
GPDM	=	Green Propulsion Dual Method
GSFC	=	Goddard Space Flight Center
IR	=	Infrared
m	=	Meters
MSFC	=	Marshall Space Flight Center
MIT	=	Massachusetts Institute of Technology
NASA	=	National Aeronautics and Space Agency
NPR	=	NASA Procedural Requirements
PCB	=	Printed Circuit Board
PPU	=	Power Processing Unit
R&T	=	Research and Technology
RSS	=	Rubicon Space Systems
TD	=	Thermal Desktop
TVAC	=	Thermal vacuum
U	=	Unit, CubeSat standard of measurement (1U = 10cm x 10cm x 10cm)
W	=	Watts
XREF	=	External reference within Thermal Desktop

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I. Introduction

GPDM is a collaboration between NASA Marshall Space Flight Center (MSFC) and Georgia Tech to create a 6U CubeSat which will demonstrate the use of the green propellant Advanced Spacecraft Energetic Non-Toxic (ASCENT) in two propulsion systems. ASCENT will be used to power a traditional chemical thruster and an electrospray thruster system. ASCENT offers propulsion in both modes using a single propellant and feed system, which will give both high thrust with the chemical mode and high propellant efficiency with the electrospray mode, as well as a lighter integrated flight system due to sharing a common tank and feed system. See Figure 6 in Section III for a visual of the propulsion sub-system configuration. The propulsion sub-system is integrated into a spacecraft bus that supplies the radio communication components, the power system components, the avionics boards, and the attitude determination and control system (ADCS).

Full success criteria of this mission is classified as performing a successful orbit change using a combination of chemical and electrospray thrusters, as well as performing attitude control maneuvers with the electrospray thrusters. Minimum success criteria is defined as the performance of one successful chemical thruster burn and one successful electrospray thruster burn in space. This is a Research and Technology (R&T) mission as per the categorization in NPR 7120.8 [1]. This analysis covers an orbit with Beta angle from 0° to 90°, the nominal orbit is subject to change, due to launch vehicle uncertainty. Such uncertainties are common with lower-cost missions such as R&T and other designations discussed later in Section IV with the associated impacts.

This project maintains several partnerships. The main partner, Georgia Tech, is providing the spacecraft bus and avionics system, performing environmental testing, and conducting mission operations support. Additional partnerships with MIT/E-Space to provide the electrospray system and powered controllers, and Rubicon Space Systems (RSS) to provide the Sprite propulsion unit, including the propellant tank.

The thermal model for GPDM includes both the propulsion system and the spacecraft bus. The initial scope of the project for MSFC was onlythe propulsion system, but changes in responsibility led MSFC to increase the scope to include the design and thermal analysis of the spacecraft bus.

II. GPDM Spacecraft Bus Thermal Model and Analysis

The purpose of the GPDM spacecraft bus thermal analysis is to determine the thermal behavior of the bus system and in developing any mitigation strategies, if necessary. The thermal analysis is completed using a Thermal Desktop (TD) model, version 23.2 Patch 4.

The bus system is modeled using TD primitive solids, with the propulsion system incorporated using an external reference to the propulsion model. The bus components include battery arrays, an ADCS, a radio transmitter, a receiver, and several avionics circuit boards. The avionics stack includes the flight computer, EPS board, and an interface board. A diagram of the bus model with the referenced propulsion model is shown in Figure 1.

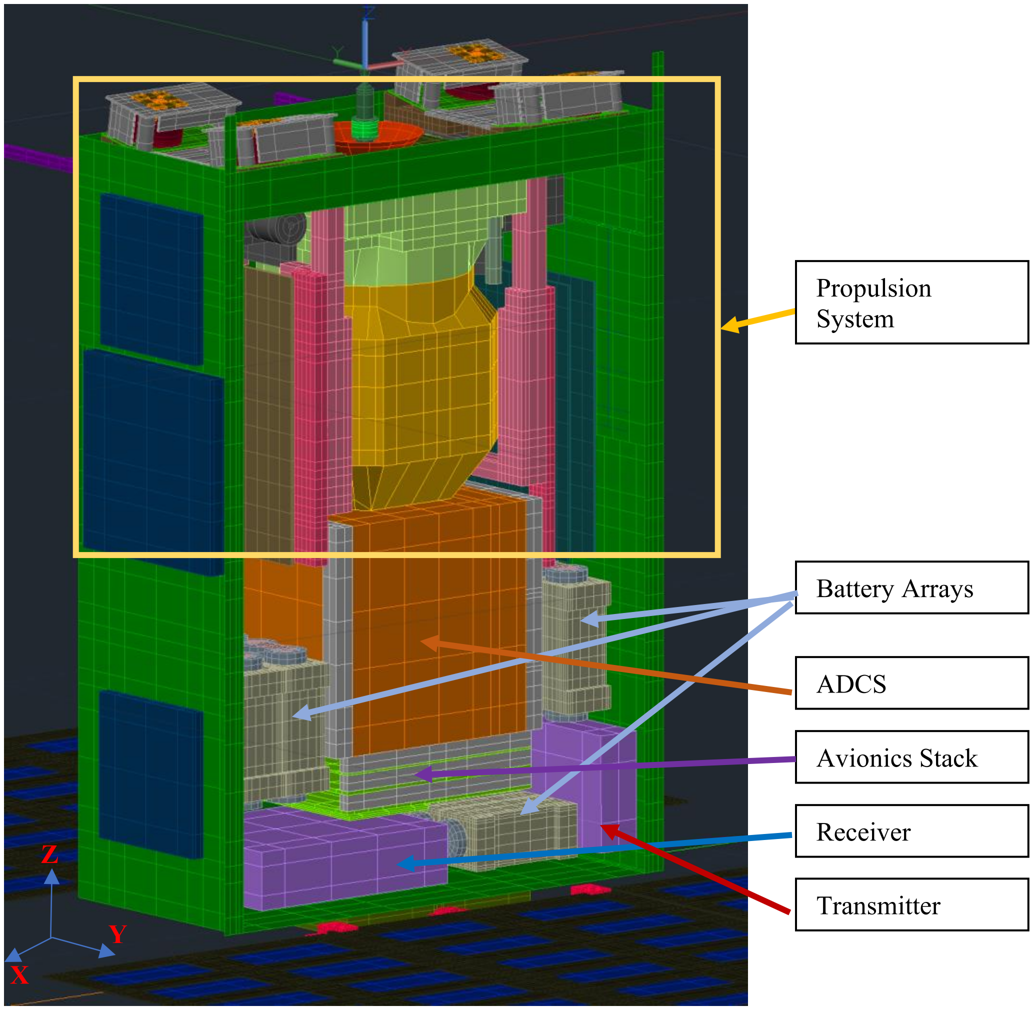

Figure 1. Thermal Desktop model of the spacecraft bus with the propulsion model visible.

The bounding cases for worst-case hot and cold orbital environments are as described in the following table. This table lists the analysis cases, Beta angles for orbit, and orientation of the spacecraft. The nominal orbit for this project is an orbit with Beta angle 75°. To account for the possibility that GPDM will launch in a different orbit, the lower bounding case with maximum time in eclipse and minimum solar flux is used for analysis as well. The solar flux, albedo, and planetshine values are derived from the Spacecraft Thermal Control Handbook [2].

The worst-case operational conditions for the hot case come from the Concept of Operations for GPDM, which states that the transmitter will transmit back to ground operations for a maximum contact time of eight minutes and the thruster will fire for a maximum of 40 seconds. Both operations have a significant effect on the temperature of other components despite being short-duration operations. Therefore, the hot case is a thermal steady state in the given orbit before propulsion operations. The second half of the hot case analysis is a transient case initialized from the steady state case with the propulsion system using a heater wire to preheat for firing until reaching the preheat bed temperature needed for the propellant of 440°C. Subsequently, the thruster is firing for 40 seconds while the transmitter is powered and transmitting. See the GPDM Propulsion System Thermal Analysis Introduction section for more information on propulsion operations and Figure 2 for a timeline of the transient heat loads.

Table 1. Spacecraft Bus Analysis Case List

Case Title	Beta Angle	Solar Flux (W/m^2)	Albedo	Planetshine	Sun-Facing Side
Hot Orbit Positive X Sun Facing Transient Burn	90°	1414	0.35	Blackbody model with value of -23.15°C	+X

Case Title	Beta Angle	Solar Flux (W/m^2)	Albedo	Planetshine	Sun-Facing Side
Cold Orbit Negative Z Low Power	0°	1322	0.11	Blackbody model with value of -23.15°C	-Z

A. Initial Conditions and Constraints

The outside of the spacecraft except for the thruster end is coated with Elimstat paint. This paint has a solar absorptivity value of 0.4 and an infrared (IR) emissivity of 0.9. A low solar absorptivity to IR emissivity ratio coating is necessary to enable the spacecraft to radiate the heat produced while also minimizing solar heating. Unless otherwise stated, components are assumed to have high thermal conductivity to the external structure of the spacecraft, given that the bus components are all metal structures bolted onto the inside of the metal external structure of the spacecraft.

The temperature constraints given by Georgia Tech are as shown in Table 2. These do not include additional thermal margin. The power supplied to each component is given in Table 3.

The powers shown in Table 3 are the highest expected component powers given for worst-case hot and the lowest power for worst-case cold conditions.

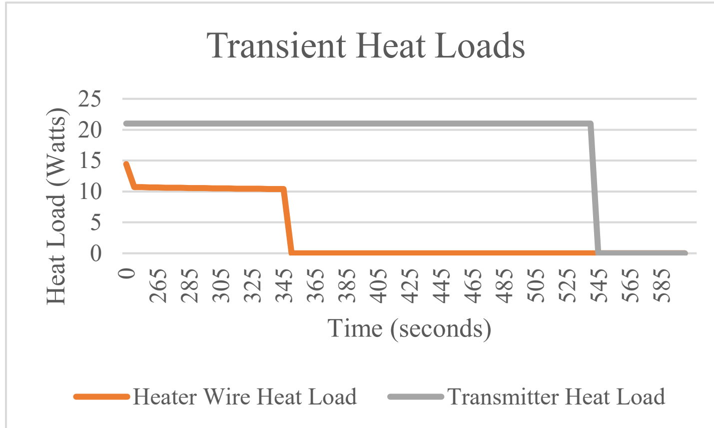

Figure 2. Transient heat loads for worst-case hot operations of GPDM.

Table 2. Spacecraft Bus Temperature Constraints

Component	Operational Minimum Temp (°C)	Operational Maximum Temp (°C)	Non-Operational Minimum Temp (°C)	Non-Operational Maximum Temp (°C)
EPS Board	-5	70	-15	70
Interface Board	-5	70	-15	70
Flight Computer	-20	85	-20	85
XACT-50 (ADCS)	-20	60	-30	70
Battery Pack	-20	75	-30	60
Transmitter	-40	85	-55	100
Receiver	-20	70	-40	85

Table 3. Spacecraft Bus Power

Component	Worst-Case Hot Power (W)	Worst-Case Cold Power (W)
X-ACT 50	9.13	6.8
Battery	1.5	1.5
EPS Board	1.5	1.5
Interface Board	2.45	1.45
Flight Computer	7.28	5
Receiver	16.7	0.5
Transmitter	21 when transmitting, 0 otherwise	0

B. Analysis Results

The two cases described in the introduction are analyzed with these parameters and the temperature results for the cases are shown next to the constraints for each component in Table 4. The following plots are the temperature profiles during the hot case analysis preheat and firing time.

The span of component temperatures for maximum worst-case hot and minimum worst-case cold are all within 10 degrees Celsius of each other for the components except for the flight computer in both the hot and the cold case. This result is reasonable, given that the spacecraft bus components are all well connected thermally to the spacecraft that can efficiently spread the heat.

Table 4. Thermal Desktop Model Temperature Results

Component	Hot Orbit Maximum Temp (°C)	Cold Orbit Minimum Temp (°C)
EPS Board	41	14
Interface Board	41	14
Flight Computer	52	14
XACT-50 (ADCS)	38	13
Battery Pack	45	14
Transmitter	46	13
Receiver	42	16

Limitations on this analysis include the spacecraft bus having no prior TVAC testing at the component level and the spacecraft-level TVAC test has not been completed yet, so no model correlation to test data has been done. The majority of bus components are commercial off-the-shelf (COTS) components with limited insight into the component thermal design.

The Georgia Tech-designed subsystems, the battery packs and the avionics stack, also have not been tested in thermal vacuum. Subsequent subsections provide more in-depth results of each component.

1. Radio Transmitter

Both the radio transmitter and receiver are modeled as Aluminum 6061-T6 blocks with density multipliers to align with the known mass, as more detailed information about the internal systems besides total mass is not known. This adds a degree of uncertainty to the temperature predictions that the spacecraft TVAC test will help to better characterize these component's performance. This density multiplier ensures the model is not overestimating the thermal capacitance of the radio components, while using the known external material of Aluminum 6061-T6 throughout demonstrates worst-case conductivity through the transmitter and receiver.

The transmitter has a strong impact on the temperature of nearby components when transmitting in its high-power state, which dissipates 21 Watts. When transmission is not required, the transmitter will be turned off to a fully unpowered state. The maximum temperature of the transmitter is at the end of its transmission, before returning to being unpowered.

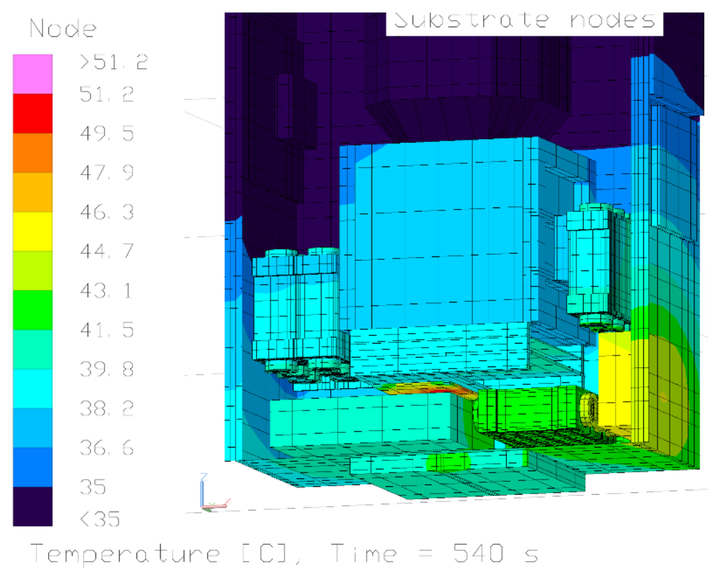

Figure 3. Color contour of maximum nodal temperatures of radio transmitter.

2. *Radio Receiver*

The receiver dissipates a constant 16.7 Watts. The exterior structure of the spacecraft conducts the heat away from the transmitter and around the structure, which increases the temperature of the other components before being emitted from the exterior of the structure to space. The receiver reaches its maximum temperature at 650 seconds as shown in Figure 4, as the heat from the transmitter is conducted through the external structure over to the receiver.

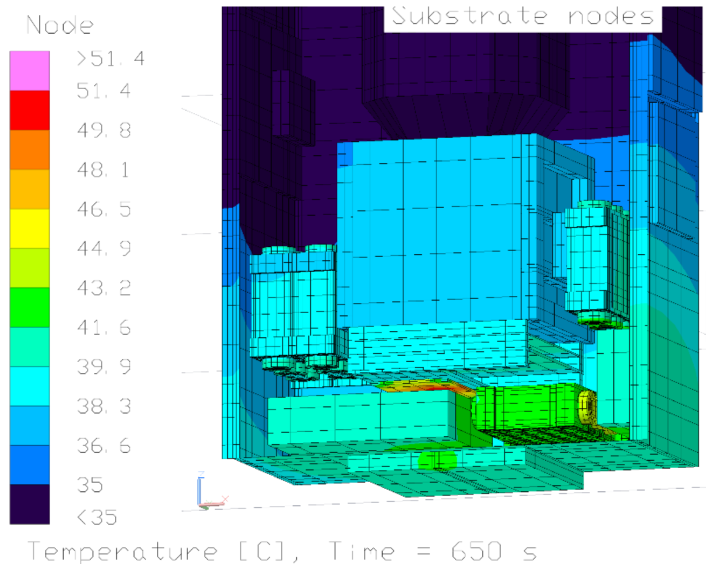

Figure 4. Color contour of maximum nodal temperatures of radio receiver.

3. *Batteries*

The transmitter heat load impact created the conditions for the maximum battery temperature predictions as well. The batteries are modeled as lithium-ion cells encased in Polyethylene Black Plastic, with 3M 2216 adhesive used to adhere the cells inside cylindrical holes in three solid aluminum blocks. The poles of the batteries are visible in the model as cylinders on either side of the block faces. The thermal conductivity through the battery is modeled as a custom material property based on the conclusions in "Thermal Parameters of Cylindrical Power Batteries: Quasi-steady state heat guarding measurement and thermal management strategies" [2]. The maximum battery temperature prediction is on the battery pack closest to the transmitter, after the heat is conducted to the outer block and into the battery cells, at 840 seconds.

The batteries are modeled with a constant 0.5 Watts heat dissipation on each cell, with twelve cells in total in the model.

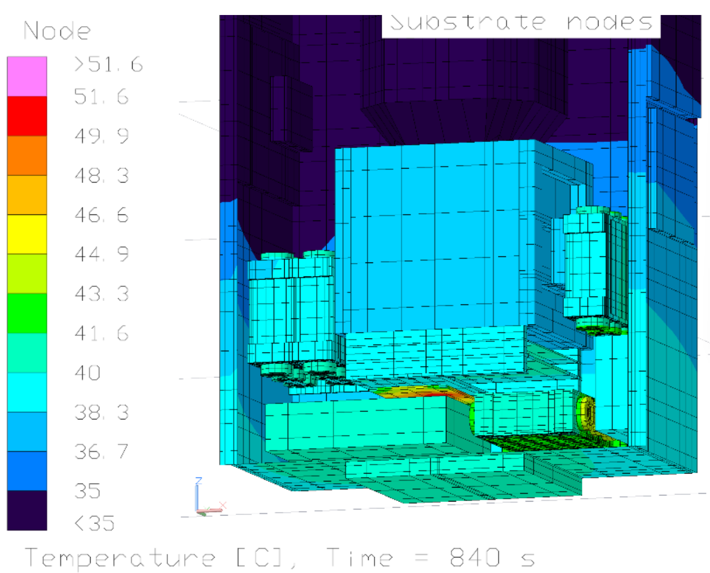

Figure 5. Color contour of maximum nodal temperatures of batteries.

4. *Avionics*

The avionics boards' temperature is relatively stable, as the heat loads are modeled as constants. The EPS board heat load of 1.5 Watts was calculated at Georgia Tech based on the amount of power being supplied to the spacecraft and the efficiency of the board components. The interface board heat load of 2.45 Watts was calculated similarly, and the flight computer heat load is based on benchtop testing at Georgia Tech. The flight computer has a heat load of 5 Watts on the board and a point heat load of 2.28 Watts in the location of a high-powered chip. All the boards are modeled as 13% copper by volume PCB based on supplier information. The flight computer can be seen in the previous three figures as the warmest component visible in the model where the point heat load is applied.

5. *ADCS*

The ADCS is modeled similarly to the radio components, as solid 6061-T6 Aluminum with a density multiplier. Similar to the radio components, the density multiplier ensures the model is not overestimating the thermal capacitance while using the known external material of Aluminum 6061-T6 for worst-case internal conductivity. The ADCS has a GPS slice with a heat load of 2 Watts and a general heat load across the component of 5 Watts. With the connection to the thermally stable propellent tank and relative distance from the radio transmitter, the temperature does not vary more than a maximum of 2°C from the maximum and minimum values in Table 4 for the ADCS as well.

III. GPDM Propulsion Thermal Model and Analysis

The purpose of the GPDM Propulsion System Thermal Analysis is to determine any thermal risks and methods to mitigate them. To accomplish this, a thermal model was created in TD. TD was chosen due to its strong capabilities in system level analysis, as well its strength in orbital environment calculations. In addition, TD has the feature to use external references (XREFs), which allows different AutoCAD drawing files to reference each other. The Propulsion System model is referenced by the Bus model.

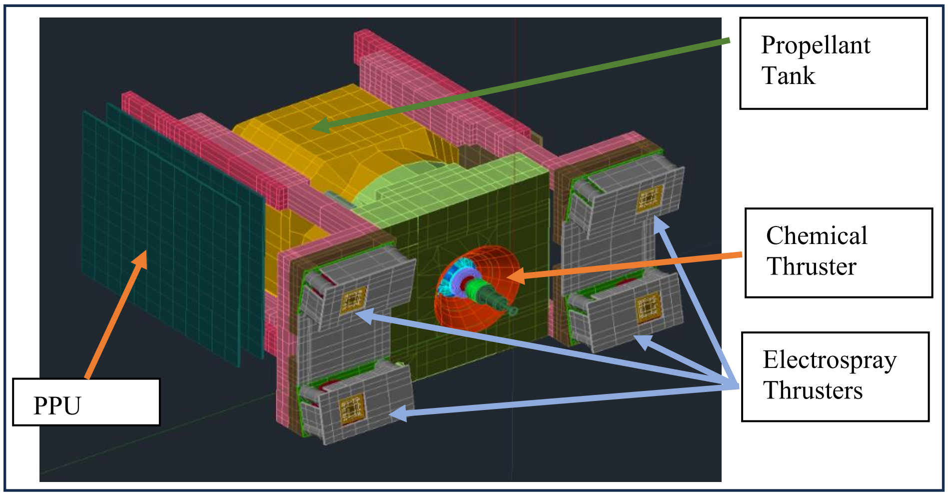

Figure 6. GPDM propulsion system modeled in Thermal Desktop.

Two separate analyses were conducted: a “Hot Case” and a “Cold Case”. These two cases are intended to analyze the worst case hot and worst case cold, respectively, for the Propulsion System. These cases are the same ones as referred to in Table 1.

To assess thermal risks, thermal limits needed to be determined. As many of the components used in GPDM are not COTS, many thermal limits were not readily available and needed to be obtained from the component/subsystem leads.

Increased focus was placed on components believed to contain the highest risks. As such, the fidelity of the Propulsion System thermal model varies, with the chemical thruster modeled at high fidelity and other comoponents such as the valves modeled at a lower fidelity. Any result that is believed to have a high level of uncertainty due to current level of modeling effort will be discussed.

A. GPDM Propulsion System Thermal Analysis Results

A summary of results for this analysis have been provided in Table 5. The analysis completed is an integrated analysis with both the propulsion model and bus model. Colored text represents exceedances in the respective category. Refer to subsequent subsections for more in-depth results of each component.

Table 5. Minimum and Maximum Temperature Results

Component	Minimum Temperature Limit (°C)	Maximum Temperature Limit (°C)	Minimum Predicted Temperature (°C)	Maximum Predicted Temperature (°C)
Sprite Structure	0	250	4	89
Sprite Propellant	-65	90	10	32
Sprite Thruster/Isolation Valves	-5	60	10	34
CPRS Valves	-5	60	9	33
Foxglove Model 1	5	70	9	34
Foxglove Model 2	5	70	8	52
Power Processing Unit (PPU)	-10	50	9	35
Electrospray Thrusters	10	40	4	36

1. Sprite Structure

The Sprite Structure is considered to be any structure that is additively manufactured. In this model, this includes the Sprite Tank and the Sprite Manifold. As seen in Figure 7, the warmest point of the Sprite Structure occurs directly upstream of where the Chemical Thruster mounts. Because the thruster reaches 1300°C while firing, these results do not appear anomalous.

2. Sprite Propellant

The propellant inside the Sprite Tank is currently being modeled as a single node that represents the mass of a roughly 80% full tank. The node is connected thermally to all innter surfaces of the tank by a conductor that assumes natural convection inside of a sphere to model heat transfer. The temperature does not vary more than a maximum of 2°C from the maximum and minimum values in Table 5.

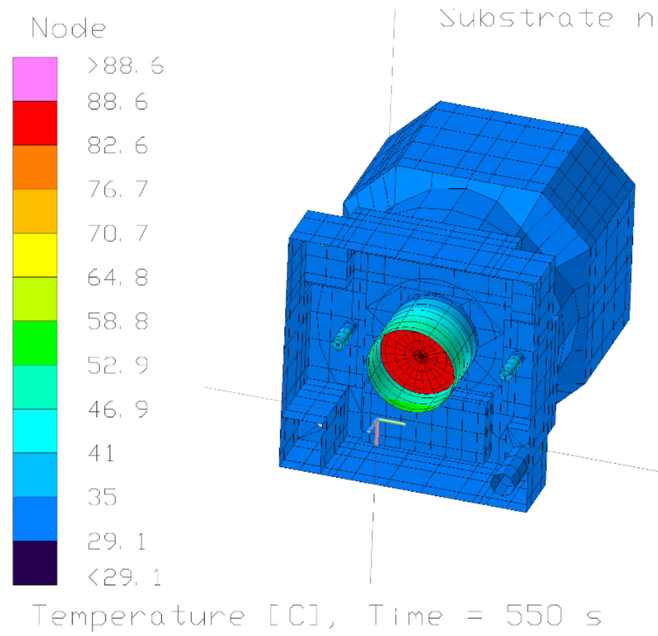

Figure 7. Color contour of maximum nodal temperatures of Sprite structure.

3. Sprite Foxglove

Sprite Foxglove, also called Foxglove Model 2 (FGM2), is an electrical board attached to the Sprite Manifold that is modeled as 13% copper by volume PCB. It is coated in Arathane 5750, with an assumed emissivity of 0.856. There are a total of four heat loads, as seen in Figure 8, placed on the Sprite Foxglove. The most consequential heat load is the Chip U11 heat load, which is the heater driver responsible for preheating the thruster. This heat load is modeled as a function of heater wire temperature, due to the change in electrical resistance with change in temperature. As the heater wire temperature increases (and therefore the electrical resistance increases), the heat generated by Chip U11 decreases. In addition, thermostatic control of the heater wire causes the Chip U11 heat load to decrease to 0 Watts while the heater wire is cooling from 440°C to 420°C.

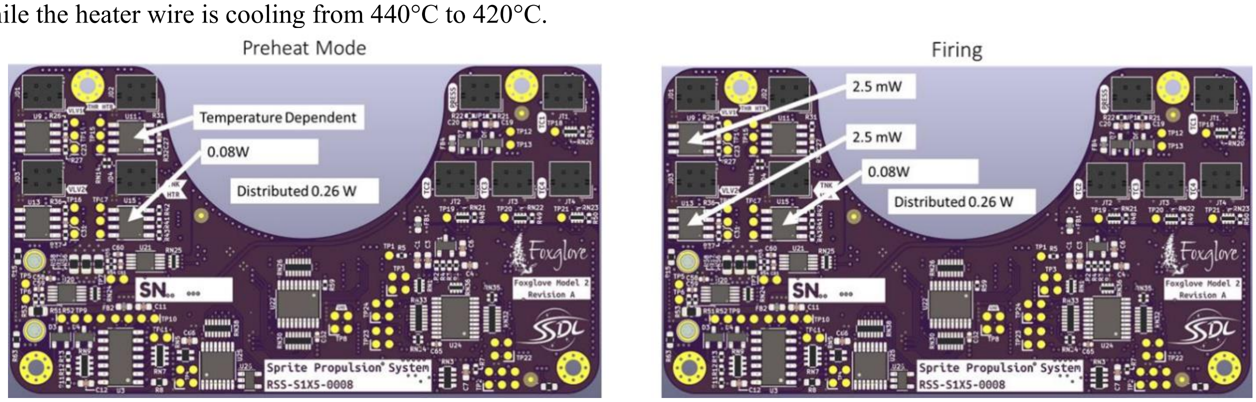

Figure 8. Foxglove Model 2 Heat Load Locations and Values

Figure 9 shows the maximum temperature of the Sprite Foxglove at the end of the preheat phase. This temperature rise is primarily from the board's self-heating, with a small contribution from heat radiating from the Thermal Shield. During the thruster firing phase, this radiative heat transfer is amplified due to the substantial increase in Thermal Shield temperature.

4. Sprite Thruster/Isolation Valves

The maximum temperature on the valves is reached during firing due to the 1.8 Watts operating heat load. While the valves are open, they experience cooling due to the flow of propellant.

5. Compact Pressure Reducing System (CPRS) Valves

The CPRS valves have minimal thermal impact on the other spacecraft components and stay below their maximum operating temperature during firing.

6. Foxglove Model 1

The Foxglove Model 1 remains below its maximum temperature limit but the temperature increases during firing. The primary source of the board's increase in temperature is due to its self-heating from a 3.17 Watts distributed heat load.

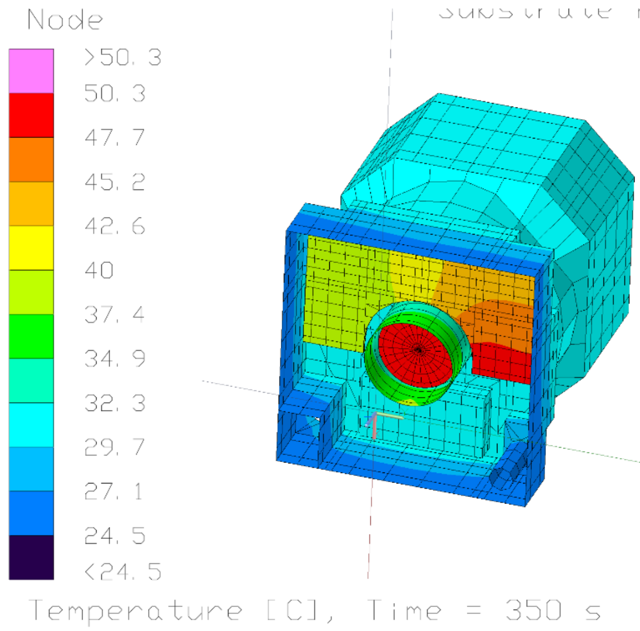

Figure 9. Color contour of maximum nodal temperatures of Sprite Foxglove.

7. PPU

The Power Processing Unit, also called the E-Space Controller, consists of two boards (BITS-HV-X and BITS-SW-7) attached to one another, which is then attached to the Sprite Cage. The primary source of the boards' increase in temperature is due to their self-heating of a 1.2 Watts distributed heat load on the BITS-HV-X, and a 0.5 Watts distributed heat load on the BITS-SW-7.

8. Electrospray Thrusters

The propellant inside of the Electrospray Thrusters are modeled with single nodes that represent a full reservoir. The initial model results were the rationale for a flight rule to limit the direction the electrospray thrusters can face so they are not facing the sun, as the model predicts temperatures in exceedance of the maximum temperature limit in that case.

Table 5 shows the Electrospray Thruster Propellant exceeding the lower temperature limit. However, this concern is minimal for multiple reasons. The first reason is because this lower limit is based on performance, and not some sort of component failure. The second reason is because of potential mitigation options. The spacecraft could be oriented to have the thrusters face the sun and due to the optical properties of the thrusters, this alone would likely increase their temperatures to within performance limits. The spacecraft could then be slewed back to nominal orientation before initiating the burns, at which point the flow of non-cold propellant would help maintain electrospray tank propellant temperature within bounds. Another option is to preheat the chemical thruster, which would impart some heat onto the electrospray thrusters. The risk to the mission, which is already very low, can be easily mitigated.

IV. Overview of Low-Cost, High-Risk Tolerance Missions

NPR 8705.4B, "Risk Classification for NASA Payloads" [3], sorts payloads into risk tolerance categories from lowest tolerance to highest, designated by Class A, B, C, and D. These are flexible designations, meant to assist in defining the programmatic rigor necessary for the mission. An example of the guidance in this document is classes A and B are expected to monitor and downlink to the ground station and record telemetry coverage during potential failure events. Classes C and D are only expected to record the telemetry coverage during potential failure events.

This allows low-cost, low-human risk programs to reduce their overhead expenses on rigorous design and testing with the acceptance of higher risk to the mission. For R&T projects, NPR 7120.8 governs the program requirements and they are not subject to NPR 8705.4B. This and other science missions not governed by NPR 8705.4B are also colloquially call "sub-class D" or "do no harm" missions, as R&T and other projects often use class D as a guideline.

Additionally, CubeSats are typically designed to "CubeSat Design Specification" [4] written and maintained by Cal Poly. This document maintains a standard set of design requirements for all known launch providers. Technically, the only governing document for a CubeSat is the specific launch provider's interface document to ensure compatibility with the deployer. There is no overarching risk management document for CubeSats.

A. Thermal Design and Analysis Approach for Low-Risk Missions

With this relaxation of technical rigor, there is a wide range of design and analysis effort seen in class D and "sub-class D" missions. This tailoring is a net benefit to these missions, allowing resources to focus on particular components or areas crucial to mission success, but does come with the trade-off that other areas are not as heavily scrutinized.

For environmental testing, all projects are required to establish a test program, but the contents are highly tailorable to the project needs. An excellent summary of additional guidance is found in "Thermal Test Tailoring Guidelines for Class C and D Space Programs" [5], which summarizes historical industry standards and has test recommendations for class C and D. For class D, it is stated that physical testing is not required but can be performed at acceptance levels and a thermal model may be constructed without model verification required. The final recommendations in the aforementioned paper are to perform acceptance testing and simple thermal model development with or without model correlation for "high success expectation" class D programs, e.g. with a program development cost over one million dollars. At MSFC, "General Environmental Verification Standards (GEVS) for GSFC Flight Programs and Project" [6] is used as the starting point for ideal thermal test campaigns before beginning tailoring to a project's specific needs.

For CubeSats, there is less thermal environment test guidance. "CubeSat Design Specification" [4], NASA's "CubeSat 101" [7], and assorted launch provider interface documents only mention thermal testing in the context of performing bakeout of components for offgassing purposes.

V. GPDM Lessons Learned

A. Component Thermal Design and Testing

The focus for the design of GPDM was on the propulsion system since its demonstration was the objective of the mission. The two propulsion systems were also produced by industry partners, so the subsystem-level testing for the electrospray thruster and the chemical thruster took place at their respective facilities. Additional NASA insight testing was performed on the electrospray subsystem in vacuum. The chemical thruster is the same as the unit previously employed on the Lunar Flashlight mission. A detailed thermal model had been developed for that mission, with model results validated against partner test data. These earlier analyses demonstrated adequate thermal margins and identified no significant risks. For the GPDM mission, the validated model parameters were transferred to the system model. However, the GPDM propulsion system introduces a distinct difference: the configuration of its electronics, specifically the Sprite Foxglove hardware, has a design such that thermal management is more critical compared to the setup used in Lunar Flashlight. The electrospray subsystem does not generate measurable heat during operation.

Georgia Tech managed the design and build of the spacecraft bus, as well as the integration of the subsystems into the fully assembled spacecraft. The spacecraft bus components were majority COTS, with the design of the battery packs, avionics boards and software done by Georgia Tech. Relying on space-rated COTS or legacy hardware is common in many programs but can also be a late source of issues if appropriate technical rigor is not applied early in the design. In this case, the spacecraft bus components were chosen and procurement of hardware began before thermal analysis support for the spacecraft bus began. For example, the radio components were chosen for their excellent performance and strength of signal without also trading the thermal implications of a high-power radio system on a small spacecraft.

No component or sub-system thermal vacuum testing was performed on the spacecraft bus prior to spacecraft integration, since no performance issues were anticipated with the space-rated COTS components. This is a reasonable trade to make, as it reduced schedule and cost for the project, but meant that the thermal model would be uncorrelated until the spacecraft-level TVAC test.

All of these factors alone are common and, to some extent, expected, but the combined impact of additional scope increase of spacecraft bus thermal design after bus components were undergoing procurement, initial limited knowledge of the material properties and power efficiency needed for accurate thermal design, and limited TVAC testing created higher than usual uncertainty in the thermal performance of the bus.

B. Thermal Model Fidelity

Thermal model design has an inverse relationship between granularity of components and the difference between temperature prediction and required temperature limits (also called thermal margin). If the components are not well understood, such as in the case of many COTS components where the internal design is proprietary, simple modeling can prove successful when the worst-case assumptions made for component properties and heat dissipation demonstrate positive thermal margin to an acceptable level. However, if a simple model does not show positive thermal margin, the components need to be modeled in more detail. For example, a simple steady-state model assuming constant radio power predicted most components would exceed their temperature limits. Working with the operations team to determine the maximum length of time the radio would be required to be in its highest-powered state and incorporating that into the model is one such granularity improvement made on the spacecraft bus.

There were several such factors on GPDM that necessitated a detailed thermal model. First was the propulsion system, since its performance is the mission objective. If the goal of the model is to provide temperature predictions on the propulsion components itself, then the actual operation of the propulsion system must be modeled rather than the worst-case thermal impacts of propulsion being used as static heat loads or other such impacts on a thermal model component. Second was the need to well define the transient behavior. Simpler thermal models can often use steady-state solutions to bound the worst-case temperature predictions. Since GPDM had the combination of a relatively small thermal mass and short-duration, high heat operations, understanding the transient behavior was vital to defining the worst-case hot temperature predictions.

A major detailed feature of the propulsion model was simulating the chemical thruster operations. While the thruster has thermostatic control on its heater wire with set points of 420°C and 440°C, which theoretically could allow the thruster to preheat indefinitely, the board responsible for controlling thruster preheat cannot operate indefinitely. As discussed previously, Chip U11 on FGM2 generates heat as a function of heater wire voltage and heater wire temperature. Once the thruster reaches its thermostatic control of 440°C, the board begins to cool before beginning to generate heat again at a thruster temperature of 420°C. However, FGM2 does not return to its initial temperature prior to the first preheat. Therefore, each subsequent preheat cycle will raise the board temperature until it reaches its cutoff limit. Model studies were conducted, with information provided from RSS, to show the number of heating iterations before reaching FGM2 temperature cutoff were such that mission success parameters were still able to be met.

VI. Conclusions

The scope of the GPDM Project was initially limited to that of the Propulsion System, and it was intended that it would be modular enough to interface with a variety of spacecraft buses. Once Georgia Tech was chosen as the integrator for the project, which also meant they would be developing a bus system, complexities began to arise. Reasonably so, a university oftentimes lacks the necessary expertise to design and develop a spacecraft bus, and thus, relied on the experience that MSFC could offer. This was true for the thermal engineering aspect as well. However, this growth in scope was not planned for, and there was a lag between Georgia Tech being chosen as the integrator and when a spacecraft thermal engineer was added to the project. This resulted in a design that could have benefitted from thermal insight before being baselined. Future work will be to complete thermal vacuum testing to thermal balance conditions and correlate the thermal models to this data for increased accuracy of temperature predictions.

The challenges faced on GPDM's thermal analysis can be viewed through the lens of the "Swiss cheese theory" of risk management. The combination of not involving thermal expertise during the initial design and procurement of the bus, not having detailed knowledge of the component thermal design for increased accuracy of an uncorrelated model, requiring a detailed transient model to accurately predict the thermal impact of the preheat and Sprite chemical thruster firing and the radio transmitter, and subsystem thermal testing only on the propulsion subsystems, all led to an increased risk of exceeding temperature limits.

Naturally, the greatest chance of success is to have unlimited schedule and funding which would allow for ideal technical insight and procurement. This is never realistic for any mission, and that is particularly true for class D and "sub-class D" missions. However, even a short amount of review by all major technical areas, including thermal analysis, in the initial design of any spacecraft has the potential to mitigate large impacts to the project later in the schedule. Additional guidance or constraints on recommended levels of analysis for these smaller projects could help push these types of discussion to occur more often. A meta-analysis of similar smaller projects could pinpoint specific problem points such as this and the results could be used to define some additional guidance for analysis.

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