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Heat Transfer OF A 1N THRUSTER IN A 2.5U CUBESAT MODULE

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Dedication

To the people that have played a role in my academic development this work is dedicated to you. From the people that took time to show me concepts to those who made fun of me when I didn't understand what they were saying – Gerardo, you will never read this, but I owe you a bachelor's degree and I kept one of your thermal books. To my family for assisting me in wherever they could and for allowing me to leach off their pantries for all these years, your efforts might be in vain but they don't go unnoticed. Finally, thank GOD for public transportation, the money it has allowed me to save is proportional to the money I have wasted on diet soda – I'm trying to quit cold turkey.

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by

JAVIER MADRID

THESIS

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Acknowledgements

Starting from the macro and progressing into the micro is how I will describe the order which people will be thanked, hopefully they will not understand this as a slight on them. To the university that allowed me to make it my alma mater without it none of this work would be possible; regardless of what I say behind its back. To the Aerospace Center who allowed me to work on the project and taught me everything from engineering and networking. I've never been one to think that luck is on my side but working in the center did feel like finding a pot of gold at the end of a rainbow. To the GMPE team which gave me a team to work with, nothing I can say express how fortunate I felt when working amongst you everyone was a pleasure to work with and obviously moved the project to a point where I could butcher the concepts you will see in this thesis. Special thanks to Raul Cuevas and Daniel Belio, you two have made a work environment transform into an unholy matrimony of inside jokes and bitter realities which are taken with a tablespoon of sugar; without you two I would have not finished this work. To the committee that has agreed to give me their time and knowledge to review my work nothing but gratitude is expressed to you all. Finally, to my mentors Dr. Amelia Greig and Dr. Joel Quintana, I'm not pretending that I'm an easy student to deal with, thank you for your patience.

Abstract

With the aerospace industry expanding and moving forward with more ambitious missions, small satellites such as CubeSats have proven to be at the forefront of innovation for research satellites. This innovation have introduced complications which if gone unattended can lead to the loss of the satellite; one such complication coming in the form of propulsion heat transfer. This paper presents an investigation into the propagation of heat transfer within a 2.5 CubeSat module when a 1N thruster is fired inside it. This is done by leveraging the equation of heat transfer and comparing results to those simulated by ANSYS. Not being satisfied with simply analysis this work will also occupy itself with the mitigation of high temperatures reaching critical components of the satellite by different methods those either be insulation or changing the proposed design of the satellite.

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Nomenclature

Q	Heat Transfer	W
T	Temperature	
ε	Emissivity	
σ	Steffan Boltzmann Constant	
F	View Factor	
A	Area of Module	
α	Absorptivity	
IR_{flux}	Infrared flux from celestial body	
$\dot{m}$	Mass Flow Rate	
C_p	Specific Heat	
r	radius	
R	Thermal Resistance	
k	Conduction coefficient	
B	Surface radiosities	
$e_b(y)$	Total black-body emissive power	
$\in$	Monochromatic hemispherical emittance of a surface	
e	Emissive power	
$E(\tau)$	Exponential integral function	
$G(t)$	Incident energy per unit area	
t	time	
τ	Optical thickness	

Subscripts

s	solar
albedo	Albedo
IR	Infra Red
flux	Flux
surf	Surface
rad	Radiation
h	Hot
c	Cold
a	Front side of MLI reflective layer
b	Back side of MLI reflective layer

Introduction

Technological advancements have always served to open doors for organizations and individuals to explore their chosen avenues of interest. This has been no different in the area of aerospace; especially for those seeking to become more involved in the level of academia. Universities take advantage of new technologies and standards that the CubeSat has been able to open for these institutions; with universities taking advantage of the CubeSat in two primary ways, first to send payloads into orbit and second to educate their pupils on the topic of satellite development.

As the small satellites have become more accessible/common the technology that is onboarded onto them becomes increasingly more expansive and serves to push the envelope to exciting new horizons. Exciting as these opportunities are, the complexities associated with these, forcing the designers to answer new questions about their satellite design and capabilities. Propulsion has been one of the subsystems that have become necessary to further push the envelope of space exploration in CubeSats. Other subsystems are then forced to be copresent with that of propulsion - some of those including: ADCS, firing algorithms, trajectory analysis, and thermal management.

The University of Texas at El Paso has been facing a similar challenge to that which was described above, in the development of the “Green Mono-Propellant Engine.” - a 2.5U CubeSat made with the intention of serving as a propulsion module for other CubeSats. It is this work’s goal to investigate the connatural link between heat transferred within the module when the thruster is firing. Furthermore, the mitigation of heat transfer into critical components of the module is also developed within this work.

Analysis of the module will be carried out in – what is best described as – steps; these steps consist of a simplified form of heat transfer which is expected to be experienced by the module. These simplified forms of heat transfer were those identified as most dangerous to the structural integrity and mission completion of the GMPE module.

Making Acquaintances with the GMPE

Due to the importance that the CubeSat structure and specifications play in this work – along with its previous introduction without proper explanation – it is the first concept that will be expanded on. “CubeSat” is the name given to the small satellite whose size, mass, and shape are specified by Cal Poly’s “CubeSat Design Specifications”; similarly, these satellites are characterized by their purpose of research. Size specification given by Cal-Poly is in a standard of measurement denoted by “U”, which means a volume of 10cm x 10cm x 10cm; similarly, mass is specified to not exceed a value of 2kg per U; as seen in [1]. The figure below demonstrates the shape which a CubeSat can take.

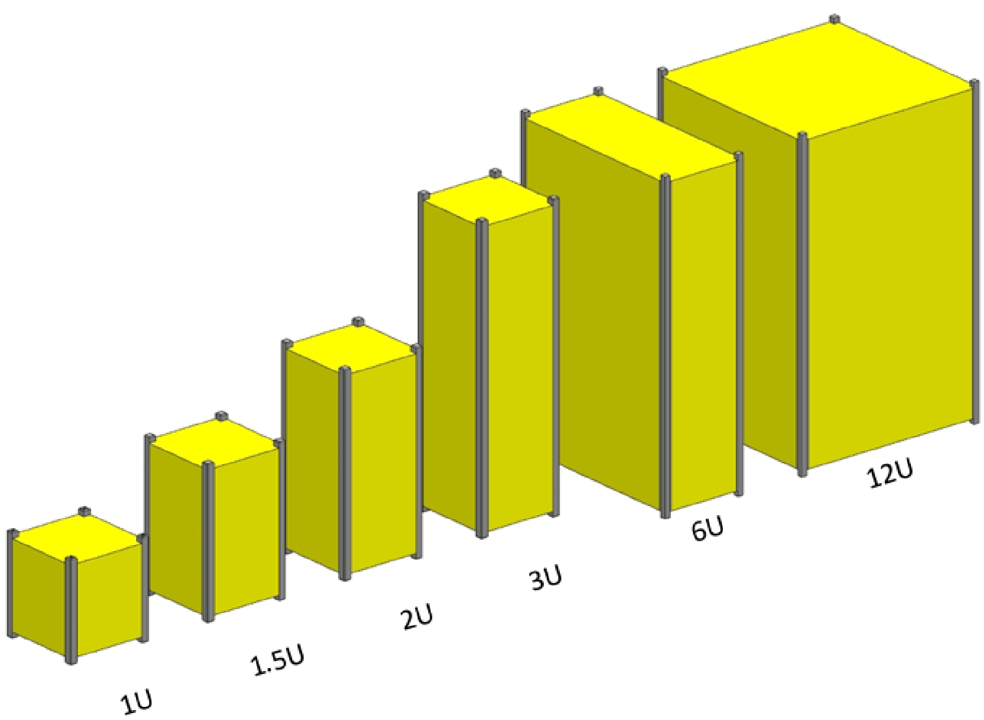


Figure 1:Typical sizes of CubeSats [2]

Now, leaning off both the explanation given above and the Spanish saying “de la vista nace el amor” – roughly translated to “from sight love is born” – the GMPE is introduced below:

Figure 2: GMPE configuration

The second topic of interest can be located by questioning the name which was given to the module, why is it called the Green Monopropellant Engine? Advanced Spacecraft Energetic Non-Toxic (ASCENT) is the fuel that is leveraged in the satellite; its major characteristic being its development by the Air Force Research Lab (AFRL) to replace fuel hydrazine. Hydrazine has many dangerous properties to it, some notorious among them are: hypergolic, high vapor pressure, corrosive, and extensive regulation for handling. ASCENT, in comparison, has the opposite characteristics to those described above: less toxic, low vapor pressure, higher Isp, and safe to handle with minimum PPE. All those factors were what granted the moniker of “green” to the monopropellant ASCENT – and the ‘G’ in GMPE. Monopropellant is the name given to fuel that does not need an oxidizer to ignite, instead using a catalyzer to decompose. As a reference, bipropellants need a combination of fuel and oxidizer to produce ignition, which is then used to produce force. Finally, “engine” is just arbitrary, it has been changed to “effort” from time to time– in other words, it would be a lie to say that the ‘E’ in the acronym is set in stone.

With all these concepts now more clearly defined – hopefully to a degree that is useful to the reader – it is time to advance to the realm of heat transfer.

Theory of Heat Transfer

There is a set of words in this work that are to become something of a watchword, it is then best to give it some thought and a brief explanation. Heat transfer is a term - which has already been introduced and will be repeated ad nauseam – that can be simply defined as seen in [3]:

“Heat transfer (or heat) is thermal energy in transit due to a spatial temperature difference”

To insert the above words into the current analysis would then lead to the following, an investigation will be carried out of the temperature difference which will occur when the thruster fires; or “transforms” chemical energy into kinetic energy. There are three methods that allow for the temperature gradients to develop – all of these occurring from a high energy source to a lower energy source – conduction, convection, and radiation. Conduction occurs through a solid medium; imagine a metal bar that is heated on one side and results in the whole specimen being hot. Convection, on the other hand, occurs when one body is in physical contact with another body of a different state of matter; an example is seen in heat sinks, as they increase the area for a hot body to be in contact with a cool medium. Finally, radiation is the heat transfer that occurs when two bodies, which are not in direct contact, exchange energy in the form of IR wavelength; an example, a person that is exposed to the sun’s rays and thinks to themselves “It is hot today!”.

Applying knowledge to satellite

By cogitating on the methods of heat transfer and now focusing on the GMPE, a clear explanation of what form of heat transfer occurs at which point is possible. When the thruster fires, an expected temperature difference between itself and the rest of the module will develop. Here the crux of the problem is reached, described by a simple question, is the thruster's heat a danger to the module? To explore the above question, look at the module below, which is now presented with its fuel added.

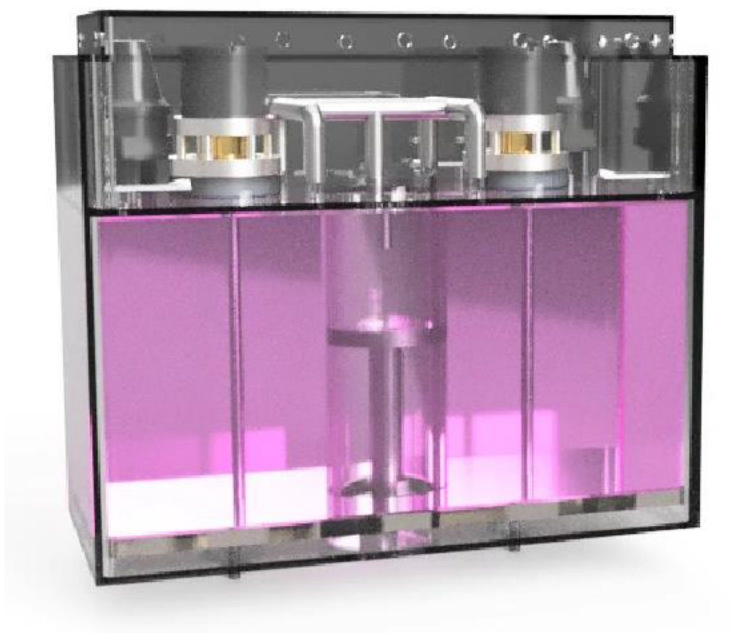


Figure 3: GMPE with propellant

To answer the above-posed question, only two things – if the simplest view is taken – are of any danger to the module: either the propellant reaching ignition temperature or the electronics failing. The propellant has a temperature of ignition of 140°C but a temperature above 80°C is not recommended for long periods of time – as specified by the AFRL. The temperature the electronics should be at – which depends on each component individually but will be taken as a bulk here – will be taken to be around 50°C. Now with those two dangers identified, the simplification of a module can be made to determine where heat transfer is most likely to propagate into these two components. Allow the following to be a semblable of the module:

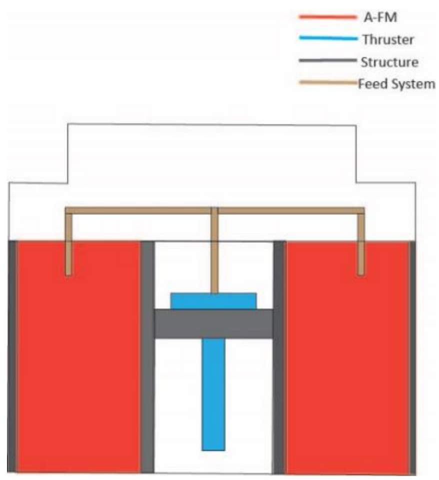


Figure 4: Simplified Semblable

To make sure that the jump from the module and its semblable is not too jarring a brief explanation is in order. The semblable above should be thought of as a "slice" of the module, a small section from which the thruster, structure, ASCENT, and feedlines can be seen in a simplified manner. With the above semblable, the representation of where heat transfer will occur can be illustrated far easier; as demonstrated below:

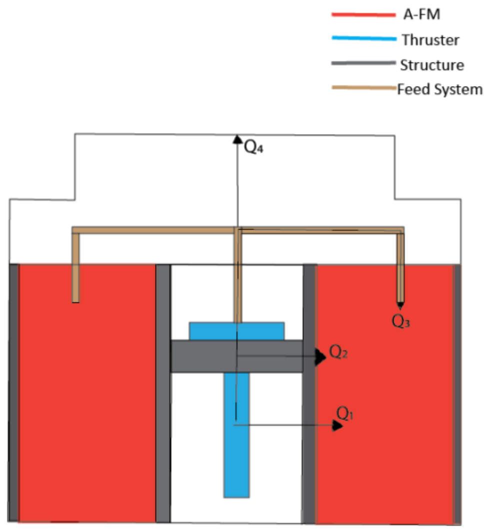


Figure 5: Predicted Heat Transfer in Module

By noticing the arrows in the above illustration – which is the only difference from the previous – then the heat transfer which will occur inside the module can be articulated. To navigate an explanation of each arrow the tried-and-true method of numerical order will be used -it is asked that the reader keeps in mind that all these thermal gradients develop due to the firing of the thruster.

1. Q_1 is the heat transfer that is expected to occur by a combination of radiation and conduction. Radiation – which occurs in a vacuum - is expected to occur in between the thruster and the wall surrounding it. When the heat reaches the wall, the temperature gradient which develops will be due to conduction within the wall. The heat then "comes home to roost" at the propellant tank, which is where the danger for the module is present.
2. Q_2 is heat transfer that will occur entirely by conduction. Occurring as the thruster is in direct contact with the structure, allowing for energy transfer by conduction

from the get-go. As was the case in the previous scenario, from the thruster-structure interface the heat can progress into the propellant.

3. Similarly, Q_3 also occurs by conduction but this time through the feedline which delivers the propellant into the thruster. A quick analysis of the structure would suggest that the small area in which heat is forced to travel will lead to an isolation of the high temperature produced in the thruster. Regardless, an analysis should be conducted to determine if the feedline will carry heat that will affect either the propellant inside it or the structure further down the module.
4. Q_4 is once again expected to contain two modes of heat transfer radiation and conduction – not exactly in that order, but close enough. In this instance, Q_4 will build off the initial process occurring in Q_3 and branch off into the module by radiation into the back of the module.

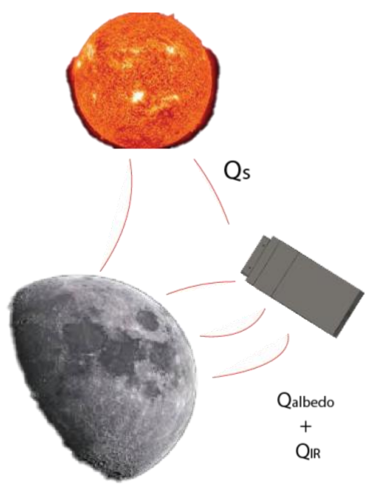
With this top-level analysis now in the rearview, another question can be formulated, which of the above-described scenarios poses a bigger threat to the module? The question is answered uninterestingly by admitting that all of them carry their dangers, but as much as that is the truth another answer - which is much more beneficial - can be given.

The question can also be answered simply with: whatever is closest to the thruster is in the most danger. This answer, like most simple answers, has one caveat; conduction being a better medium for heat transfer than radiation. This above point is made to emphasize the increased difficulty in which Q_1 is arriving at the structure than Q_2 which is immediately introduced into the structure. Other than that, the biggest threat to the module proceeds in the numeric order that was attributed to them in the figure above; final order of threat then being Q_2, Q_1, Q_3, Q_4.

Environment Analysis

In the above analysis, no mention was made of the environment in which the satellite is to operate, but it plays an incredibly significant role. Environment plays two roles in the analysis, the first as a background to the activities the module will perform in its mission cycle, second as a participant in heat rejection as the module is done with its ‘active’ period. To get a better understanding of the environment’s effect on the module, the analysis proposed in the work [4] will be performed, which considers taking “snapshots” of the satellite at its coldest and hottest predicted points. For the module and the environment where it will traverse, two cases are presented below:

Hot Case:



Cold Case:

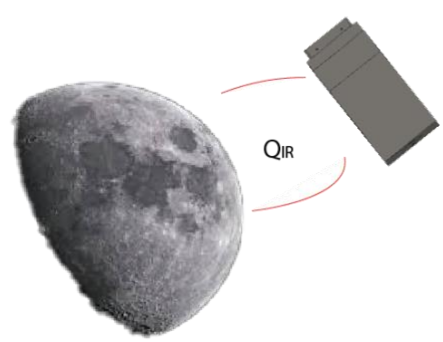


Figure 6: Module orbiting light side of the moon

Figure 7: Module orbiting the dark side of the moon

Two distinct explanations have to be constructed for the scenarios that are presented above, they are as follows:

1. Hot Case: When orbiting the moon, there will be a section at which the satellite will orbit around the lit side of the moon. This is where the satellite will be exposed to sun rays, the albedo of the moon, and infrared radiation from the sun which is reflected by the moon. In addition to this, the thruster might fire during these times, which will be discussed later.
2. Cold Case: As the satellite keeps orbiting, it will traverse into the dark side of the moon. When the satellite travels in these conditions the heat source into the module is reduced to two, the albedo and infrared radiation from the sun reflected by the moon. In addition on occasion, the thruster will fire at this section of the orbit.

Determining the temperatures at the two scenarios described previously is then a matter of using the equation of heat transfer by radiation and was demonstrated by [4]:

$$T = \left(\frac{Q}{\varepsilon\sigma F A}\right)^{\frac{1}{4}}$$

The heat transfer term – which is composed of the radiations described above – can then be expanded on in two distinct forms:

$$T_H = \left(\frac{Q_s + Q_{albedo} + Q_{IR}}{\varepsilon * \sigma * F * A}\right)^{\frac{1}{4}}$$

$$T_C = \left(\frac{Q_{albedo} + Q_{IR}}{\varepsilon * \sigma * F * A}\right)^{\frac{1}{4}}$$

The values for heat transfers above can then be calculated as shown below:

$$Q_s = \alpha_s * S * A$$

$$Q_{albedo} = Alb * \alpha_s * S * A$$

$$Q_{IR} = \varepsilon_s * IR_{flux} * A * F$$

The values presented below are those calculated for both the hot and cold cases:

Table 1: Environment Calculation

Term	Hot Case	Cold Case
Q_s (W)	2.531	0
Q_{albedo} (W)	0.354	0
Q_{IR} (W)	1.172	0.403
Q_{total} (W)	4.057	0
T (K/°C)	251.101/11.499	238.258/- 119.021

The importance of the above table – if everything else is to be ignored – lies in the temperatures that were calculated. As a reminder to the more forgetful, these temperatures represent the equilibrium temperature for the hot and cold cases that the module will encounter. Now these values can be used as a starting point when calculating the temperature produced when the thruster is firing.

Analyzing Beginnings

There is but one more phenomenon left to consider before the module can be analyzed, that being the temperature which the thruster reaches when it is fired. Luckily, the GMPE team has produced hot fire test to characterize the temperature which is reached by the thruster when firing. Below are the results of one of those tests, conducted by allowing the thruster to fire for thirty seconds from the range of ten to forty seconds:

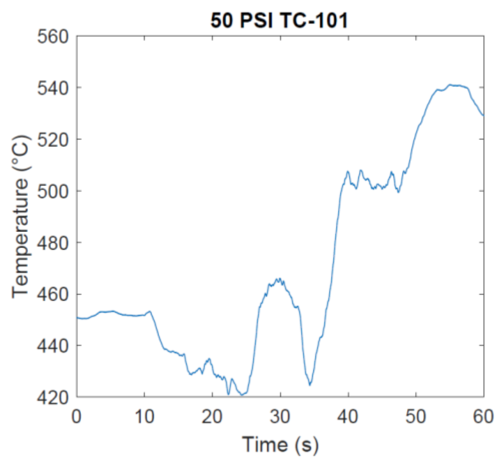


Figure 8: Thruster Firing Temperatures

For anybody following along, there should be a peak in interest when observing the above graph – this peak of interest should occur exactly at the first peak occurs in the graph. Question that comes to mind is, why is the temperature not rising continuously as the thruster fires? There is a theory that has been workshopped; due to the fluid being flown into the catalyst bed, the temperature decreases as the catalyst and propellant reach a point in which the chemical reaction can occur continuously; unfortunately it does not explain the second dip in temperature which occurs at thirty five seconds or why temperature keeps increasing after the thruster stops firing between fifty and sixty seconds. More test have taken place after this one, with the same firing schema as the one above, with the results being shown below:

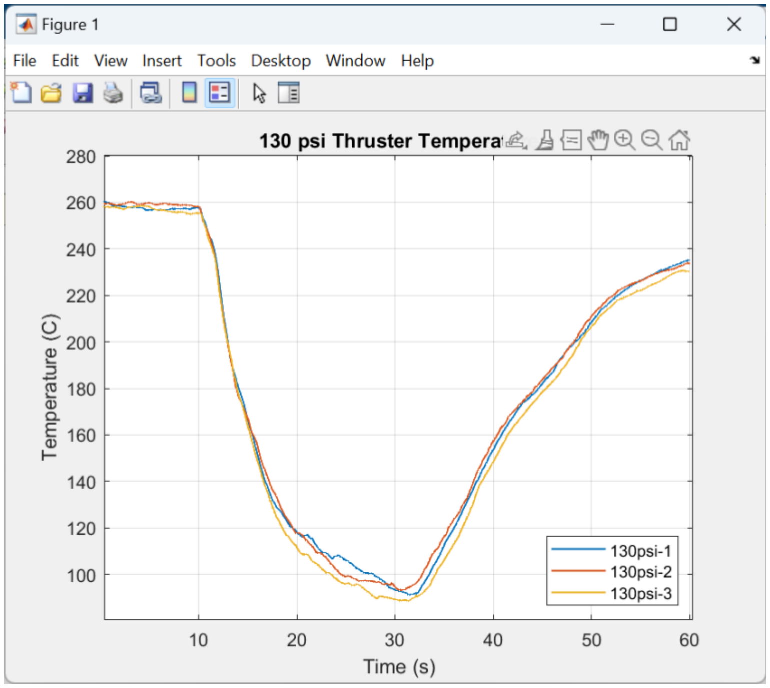


Figure 9: Second Thruster Firing Temperatures

A similar trend is observed above– presumably for the same reason as was thought of before – but in this instance, the temperature is much lower than the first test; the temperature recorded not being (theoretically) able to produce the reaction needed for the propellant to decompose. Now if the temperature seen above is to be expected in the module, then there is reasons to worry, but the above temperatures are not incredibly high – as that which are usually expected from these systems.

There was a previous study done at UTEP that showed that the temperature of the decomposition of ASCENT can reach a temperature of as high as 1200°C; unfortunately, this was a word-of-mouth study, no results were recorded (bad science? Maybe, interesting storyline? Absolutely!) – results can be seen in the reference [5]. If this is true then the temperatures that are seen above do not tell the whole story – as is almost always the case, no story is complete without a bit of twist, or without some misunderstanding– and those claims should be taken into consideration. No studies can be found from outside sources on the temperature of decomposition of the monopropellant, finding the picture of a thruster glowing red is not uncommon, but reporting the temperature it reaches seems to be taboo. To take the claims of 1200°C into consideration, a simulation was carried out to see how high a temperature the module can face if the above is true. Factors that were considered when producing the simulation then are as follows:

1. A 10W heater is used when heating the catalyst bed to a temperature of 400°C
2. Considering that we have a fluid flowing that is subject to temperature change, a constant mass flow rate, and a known specific heat, the following equation can be used to calculate the energy equation for a moving fluid:

$$Q = \dot{m}C_p\Delta T = 160.56 \text{ W}$$

3. Taking into account that this firing will occur in space, which will only allow for radiation, then the temperature of the environment that the thruster will be firing in must be calculated – which as you must recall has been done. The temperature that will be studied in the below simulation will be that of -22°C.

With all these things considered the simulation below was produced:

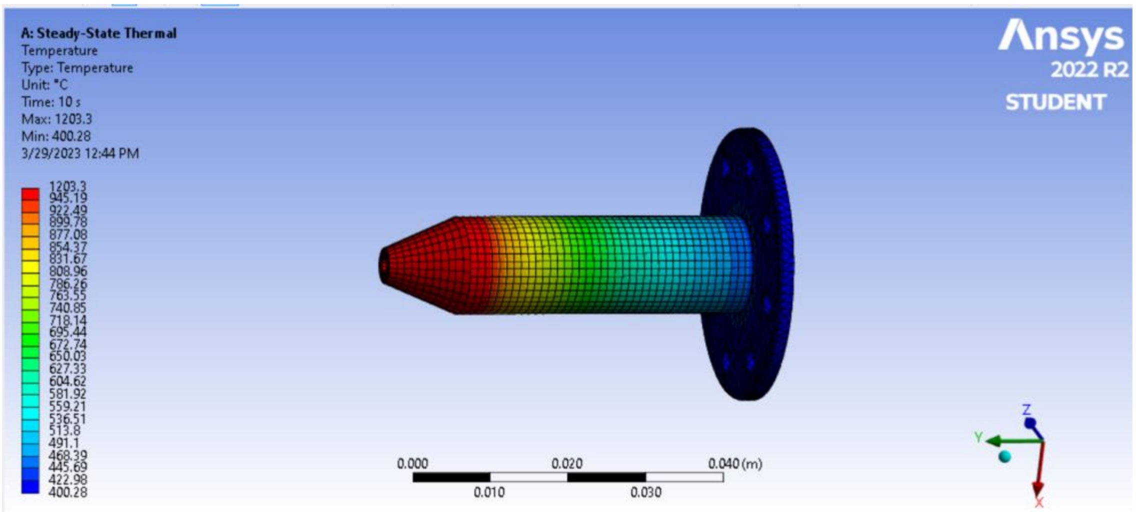


Figure 10: Simulated Thruster Firing Temperatures

The above shows two interesting characteristics:

1. Max temperature which is much higher than what has been seen in the recorded hot fire test – those which were shown above.
2. Rather large gradient of temperature, this can be explained by recalling that the above thruster is being fired in an environment of -22.05°C; making it easy to dissipate its heat into the blackbody surrounding it. The temperature gradient seen above does show the temperatures that were recorded in the first hot-fire test, but this being seen as anything other than a coincidence would be a stretch.

From the case seen above this paper has now presented two cases of real data from a physical thruster tested in ambient conditions, and one produced being purely imaginary – if the words is allowed – but informed by calculations and previously seen data. From these three options one must be analyzed, and it would be prudent to pick the worst case scenario, as if it is later proven to be wrong the thermal mitigation can be reduced.

Explosive Starting Point

A small (but critical) aspect of the heat transfer analysis is the geometry. A closer look at the GMPE module leads us to notice on important implication for the analysis:

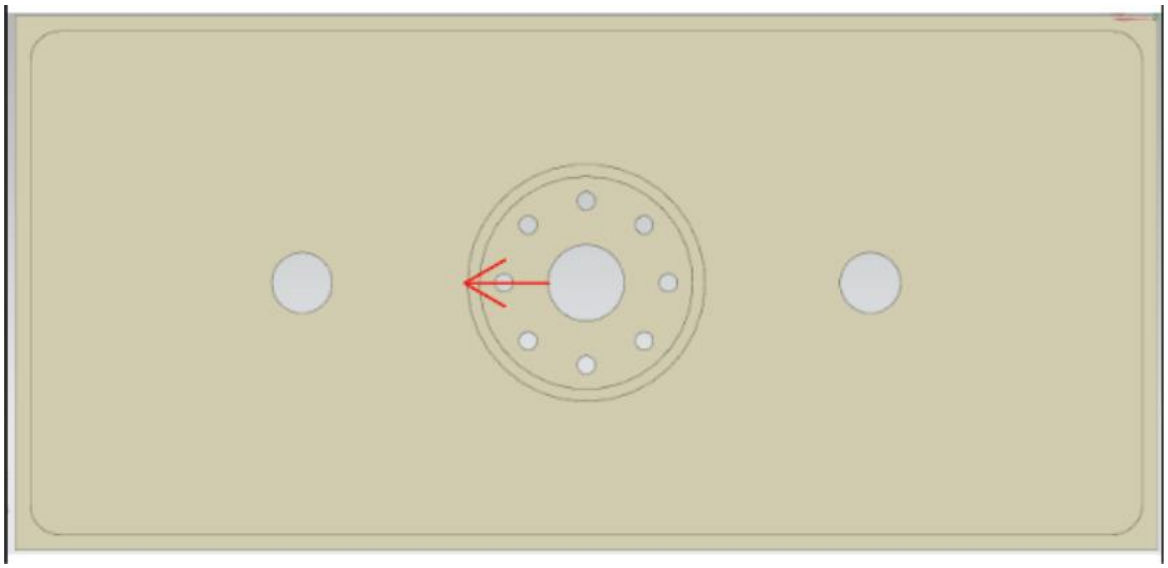


Figure 11: Wall Holding Thruster

The figure above shows the structure which is set to mount the thruster in the GMPE, the circle at the center being where the thruster is to rest inside the module. Walls might not be the most interesting aspect of a satellite, but they are important for thermal analysis, and the one above demonstrates something critical for this work to proceed. The wall above is circular, meaning that a radial coordinate system will be used to analyze the heat transfer seen in this section of the module. The following equation base will then be leveraged, as seen in [6] and [7]:

$$T(r) = C * \ln(r) + K$$

The above being a general equation of heat transfer in a radial geometry without heat generation within the wall. The 'C' and 'K' are both integration constants, to solve for these, boundary conditions have to be determined, those expressed below:

$r_1 \leq r \leq r_2$ $\qquad\qquad$ $r_2 \leq r \leq r_3$

$T_1(r_1) = T_1$ $\qquad\qquad$ $T_2(r_2) = T_2$

$T_1(r_2) = T_2$ $\qquad\qquad$ $T_2(r_3) = T_3$

Verbally expressing what is shown by the above boundary conditions will only help the understanding of the phenomena which is trying to be described, so the following presents just that:

1. Ranging from a radius of r_1 to r_2 there exist a function which describes the temperature within the wall, the function is denoted as $T_1(r)$; the exact same thing can be said from r_2 to r_3 with a function $T_2(r)$ being used to describe this section of the wall. There exist two points at which the temperature can be defined, one being at distance r_1 and the other at distance r_2; same goes between the ranges from r_2 and r_3. Here, it is important to note that the word "defined" is used very liberally, as in the analysis only temperature T_1 is known. Below is a visualization of what was explained above:

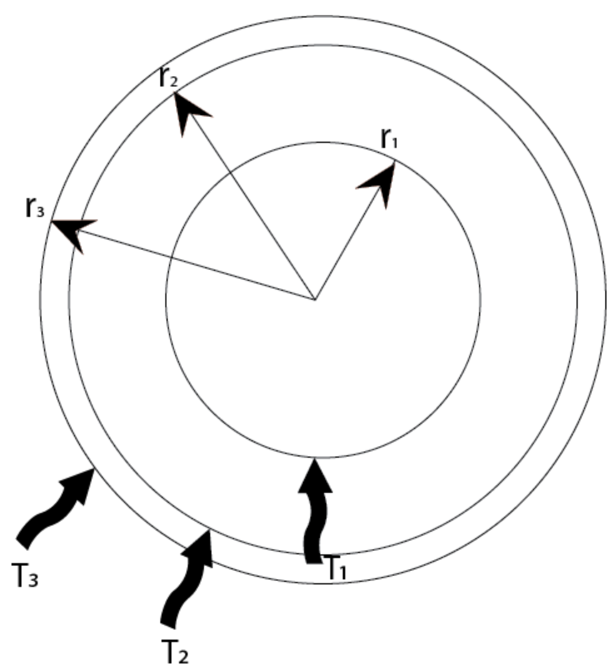


Figure 12: Configuration of Temperature Gradient in Titanium Wall

2. If insulation is added – as might be needed – then the explanation in the first case must be expanded by introducing one extra function, $T_3(r)$. $T_1(r)$, ranging from $r_1$to r_2, describes the temperature gradient inside the insulation. $T_2(r)$ and $T_3(r)$, ranging from r_2 to r_3 and r_3 and r_4 respectively, described the temperature gradient across the titanium wall. Temperature can be described at the edges of the intersections as seen below:

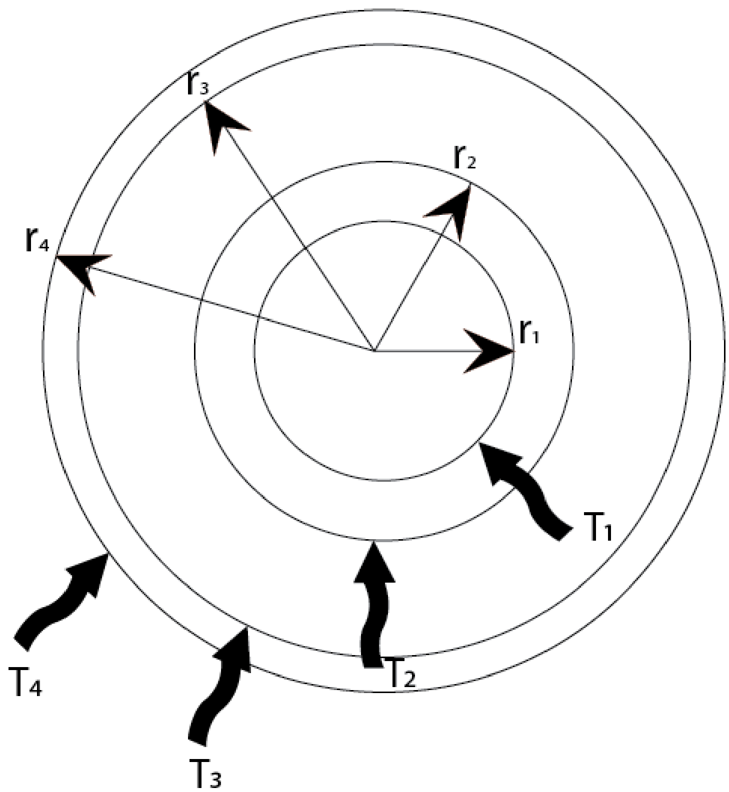


Figure 13: Configuration of Temperature Gradient in Titanium Wall with Insulation

To properly start the analysis, another question must be presented and answered – this is the final time, promise – what temperature is the thruster-wall interface going to reach? This question was investigated – to some success – in the previous section, were the temperature of around 400°C - 500°C is seen. This value must be seen with a degree of skepticism, as it was outside of the structure. A small assumption will be made – which will be justified later in the work – which will take this temperature to be around 1000°C.

Gauging the starting temperature gradient which the thruster-wall interface will develop is the first step of understanding where the module is at danger. In this scenario the only thing preventing heat transfer is the titanium body which, as is the case with metals, is a rather good conductor of heat. Understanding the concept of how much resistance the heat transfer will face inside of a structure is done through what is known as thermal resistance. Thermal resistance is the resistance that a material/geometry has to resist temperature change; equation for thermal resistance in this case is then:

$$R_1 = \frac{\ln\left(\frac{r_2}{r_1}\right)}{k*2*\pi*L} \qquad R_2 = \frac{\ln\left(\frac{r_3}{r_2}\right)}{k*2*\pi*L}$$

Where:

k – is the thermal conductivity of the module wall; this value is dictated by the material.

L – is the length of the wall; in this analysis think of it as the depth of the wall

r_1,r_2,r_3 – are all the radius (as shown in the previous figure)

Calculating the resistance of the individual sections leads us to the following values:

Table 2: Resistance Values of Titanium Structure

R_1	R_2	R_{tot}
1.63 K/W	0.17 K/W	1.8 K/W

Total thermal resistance seen above is on the lower side, this can be further seen when the heat transfer in the module is calculated with these values:

$$Q = \frac{\Delta T}{R} = \frac{1273.15 - 353.15}{1.8} = 511.11W$$

From the above value the following physical implication is extrapolated, if the desired temperature of 80°C is to be reached then 511.11W need to be dissipated inside the wall – which in the case currently analyzed is unlikely to happen. A clearer idea of this can be seen if a simulation is carried out:

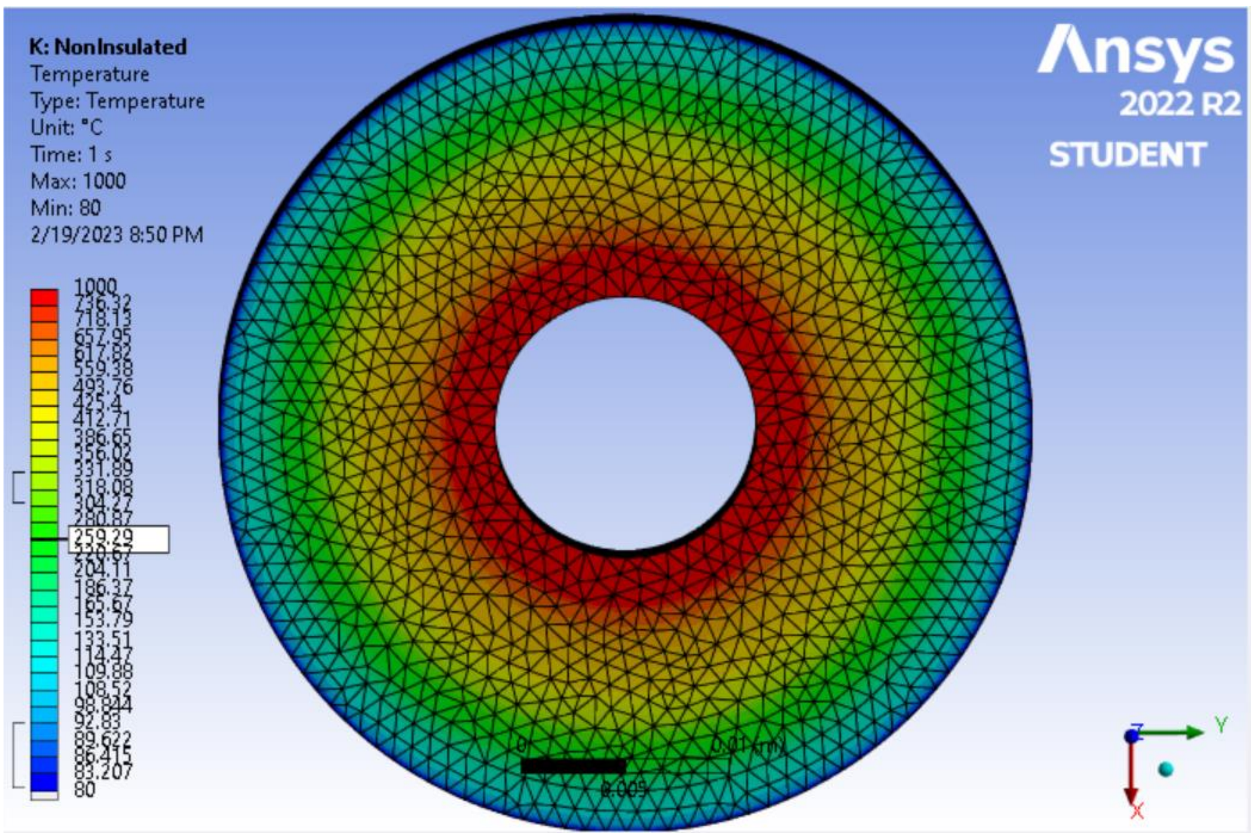


Figure 14: Temperature gradient in Titanium Wall

Simulation above demonstrates the temperature gradient which would be needed inside the module wall to reach the desired temperature inside the module. Observe the drastic temperature gradient inside the module which is almost unnatural – this is even more suspicious in a material which is considered a good conductor. Reconsidering the problem faced above, insulation will be added to the wall and see how results change.

MaxFire HP was selected for the task of insulation, with a thermal conductivity of 0.085W/m*K and resistance to high temperatures. Determining the amount of insulation needed is of importance, so a range of 1mm - 4mm were investigated, with the following results being seen:

Table 3: Insulation Heat Transfer and Resistance Values

Heat Transfer Rate										
	No insulation		1mm		2mm		3mm		4mm	
	R	Q	R	Q	R	Q	R	Q	R	Q
Layer 1	1.63	365.92	64.73	8.86	143.028	4.007	198.20	4.372	246.28	3.618
Layer 2	0.17	1.88 *10^3	1.63	183.99	1.63	183.99	1.40	20.422	1.19	11.747
Layer 3	N/A	N/A	0.17	273.63	0.17	273.63	0.17	145.55	0.17	87.33

When inspecting the table above, it becomes obvious that the insulation starts playing an immediate role as it is placed. Below are the simulations for the four cases with insulation presented:

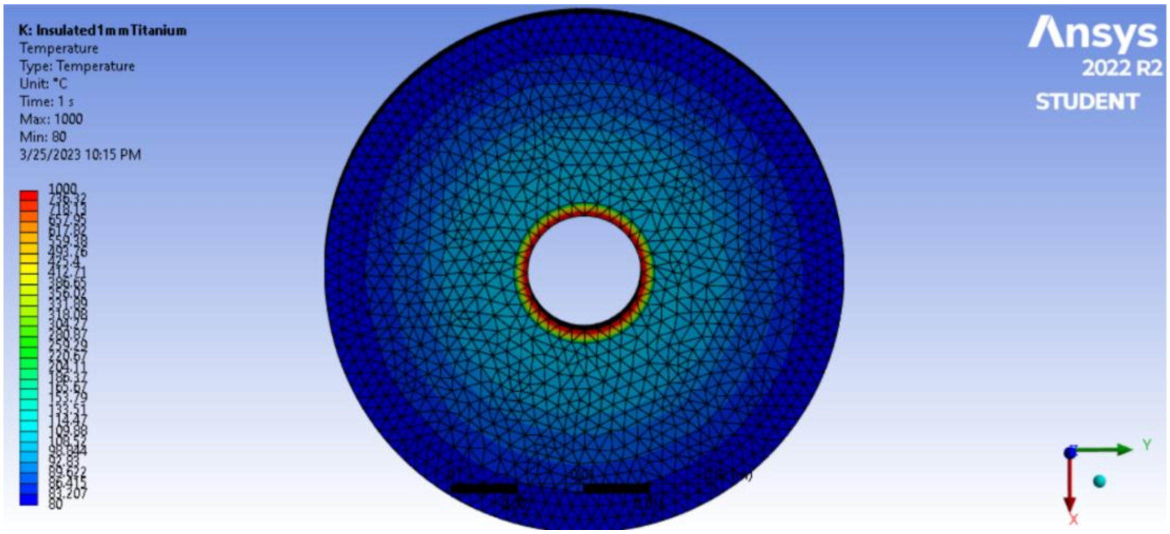


Figure 15: Temperature Gradient in Titanium Wall With 1mm Insulation

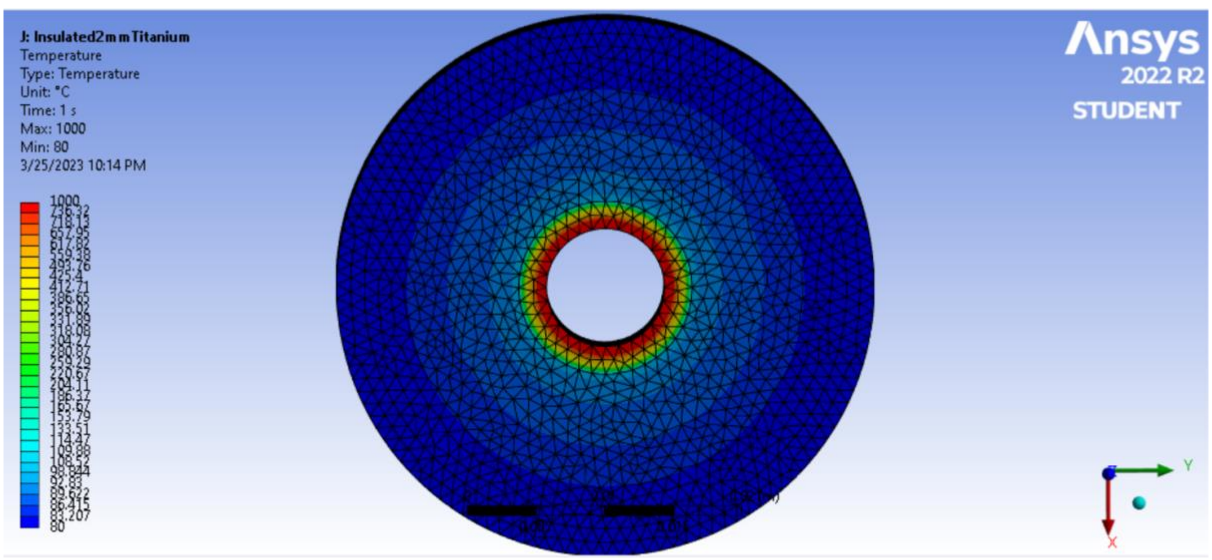


Figure 16: Temperature Gradient in Titanium Wall With 2mm Insulation

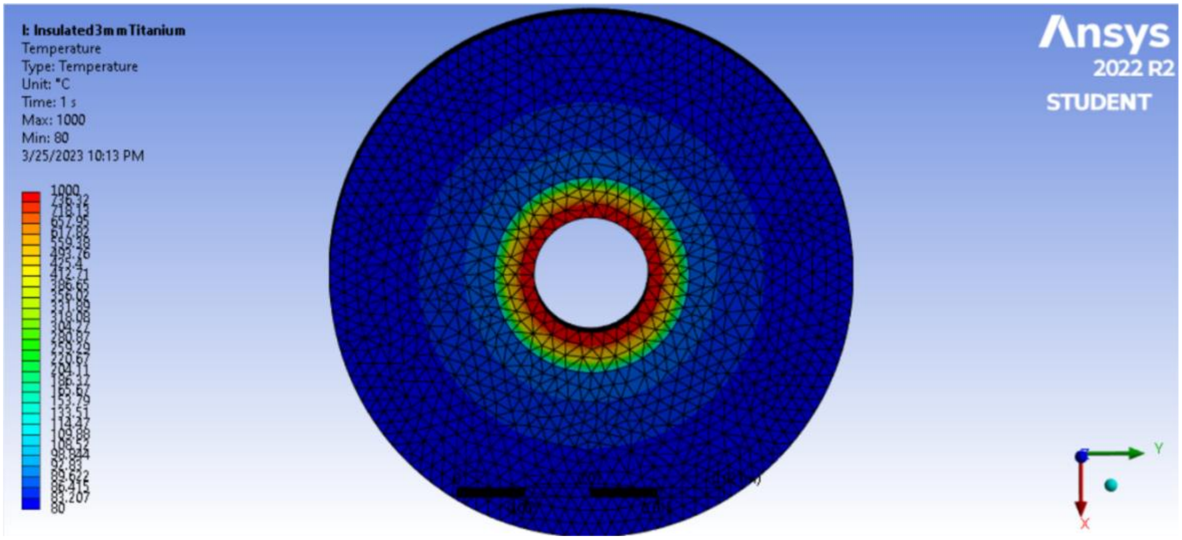


Figure 17 Temperature Gradient in Titanium Wall With 3mm Insulation

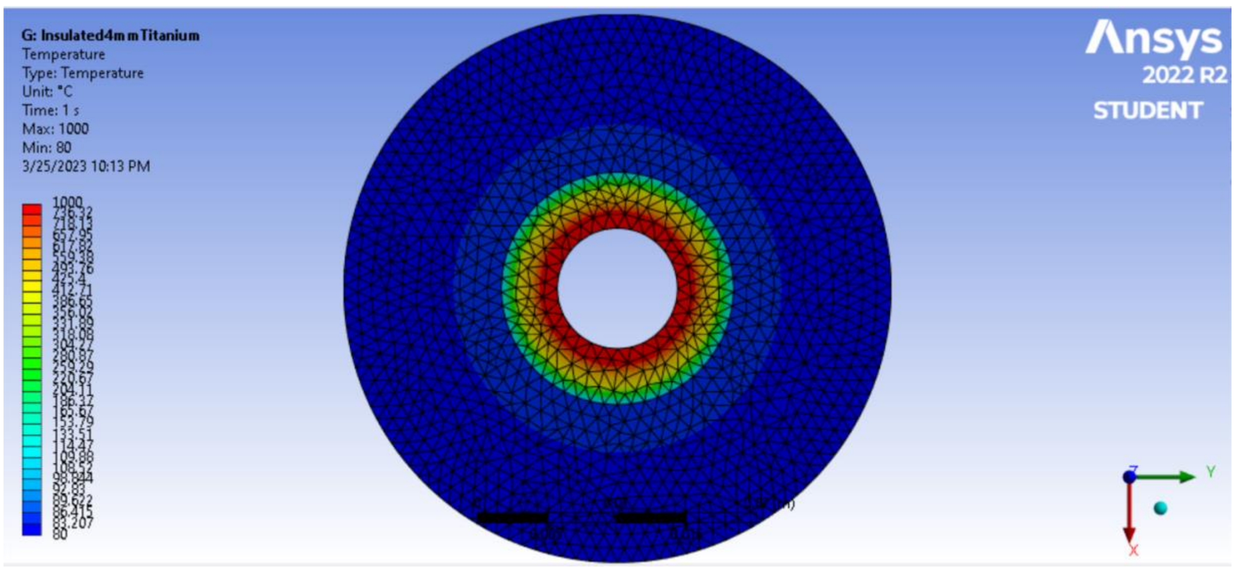


Figure 18: Temperature Gradient in Titanium Wall With 4mm Insulation

Simulations above show the same pattern that were presented in the calculated values in the table presented above – and the pattern that most people could have deduced – as the insulation is increased the temperature gradient in the wall is concentrated in the insulation. Previously it was stated that this type of "unnatural gradient" was not probable, what has changed from then and now? Answer lies in the thermal resistance, with 4mm of insulation provides 246.27K/W versus

1.63 K/W given when the titanium is left alone, hence the gradient is much more understandable. Finally, the outermost temperature calculated around the edge of insulation is around 98°C. Even though this temperature is not the 80°C that is being aimed for, its much safer than the case predicted without insulation. This section then concludes with the values and parameters presented here, physical testing of the module will serve to disprove or assert the values and methods presented above.

Radiating Not Exactly Confidence

The next form of heat transfer to be explored is that of radiation. When the thruster is placed inside the module it should look as follows:

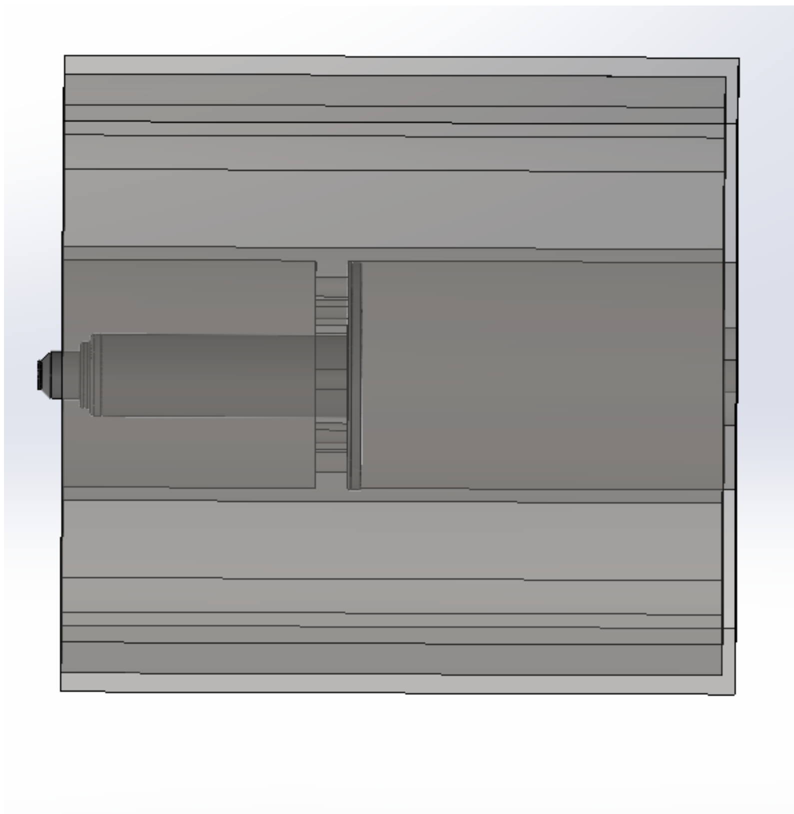


Figure 19: Thruster Inside Module

First step in analyzing the above geometry is to recall the equation for heat transfer by radiation:

$$Q_{rad} = \varepsilon * \sigma * F * A_{surf} * (T_h^4 - T_c^4)$$

From the above equation the interest is in the increase of temperature of the wall due to the thruster firing; simply put the term T_{hot} is to be calculated. Algebraically solving for this variable in the above equation leads to:

$$T_h = \sqrt[4]{\frac{Q_{rad}}{\varepsilon\sigma F A_{surf}} + T_c^4}$$

As expressed in a previous section of the work, Q_{rad} is composed of two different heat sources; heat of decomposition and heat form the heater.This is expressed algebraically as:

$$T = \sqrt[4]{\frac{Q_{decomp} + Q_{in}}{\varepsilon \sigma F A_{surf}} + T_c^4}$$

Next term of interest is T_c, which luckily has previous been calculated – in the environment section – and will be taken to be -22.049°C. With all these values, plugging them into the previous equation results in:

T = 821.59°C

As become customary in this work, a simulation was carried out to verify the results of the equation above:

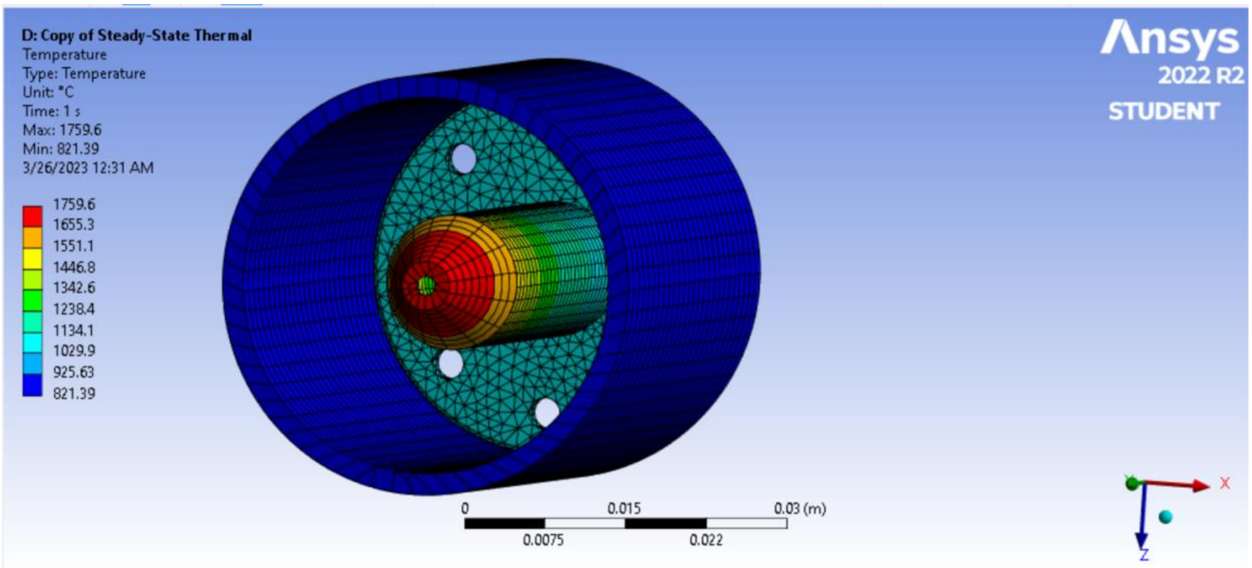


Figure 20: Temperature of Wall and Thruster Firing

Multiple points can be made from the above simulation – with but a brief moment of celebration allowed for the agreement of the equation and simulation. Now delving into the simulation deeper leads to the following discussion:

1. If the maximum temperature of the previous simulation is compared to the one produced in this section, something is immediately discernable. Making sense of why the temperatures of these simulations are different is not an easy task, the best educated guess that has been entertained is that the lack of radiation into the environment is not allowing for heat to dissipate. Without the ability to dissipate the heat the thruster is reflected back some of its radiation back and in turn is making the structure hotter.
2. Notice how the temperature at the base of the thruster is 1029.9°C, which is very close to the temperature that was "assumed" in the past section. This helps to stamp down the temperature explored in the past section – once again allow for a parade.

Before the celebrations get out of hand, realize how detrimental these types of temperatures are for the module. A temperature of 821.59°C that close to the propellant and through a material that can (and by most is) classified as a good heat conductor puts the module in a dire situation.

Mitigating the above described "dire situation" becomes the goal of this section, with MLI giving a reason for hope. The layman's explanation for MLI is as follows, in order to reduce radiation heat transfer, highly reflective materials are 'stacked' on each other with a small distance of separation being left between them. This structure allows for heat to be reflected by one layer and the rest progressing onto the next layer, as was described in [8]. To give a visualization to the concept explained above of MLI the figure below will be of assistance:

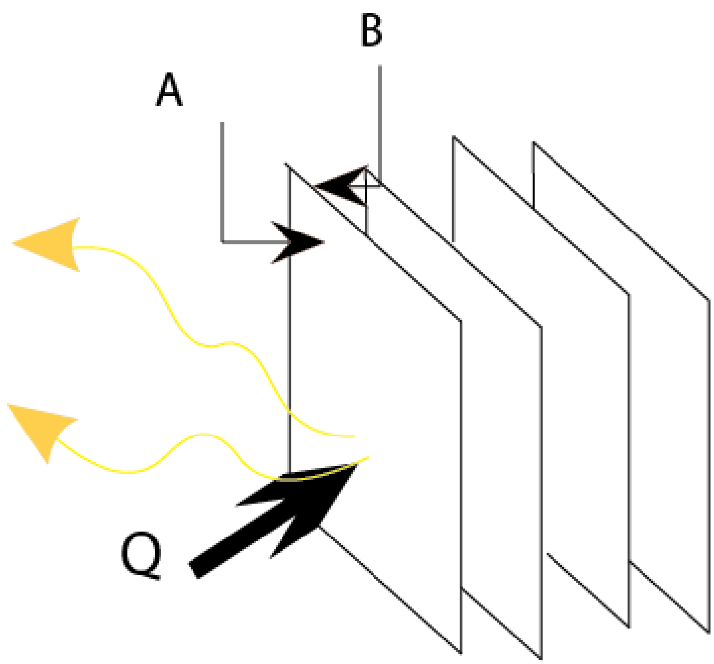


Figure 21: Ideal MLI Simulation

When heat is applied onto the MLI, the layer will reflect some heat back, either into the environment or the previous reflective layer, with some percentage of the heat then progressing into the next layer. As the above illustrates, ideally an MLI blanket is composed of empty space between the layers. Having nothing but empty space between the layers would then only allow for heat transfer by radiation to occur; unfortunately, in practice this cannot be achieved. Before this harsh reality is faced, looking at the ideal case for MLI can lead to a good understanding of where to start – as it will be demonstrated that mathematically speaking MLI is much more complex than the systems explored before.

To start this simplified analysis of MLI equations presented by [4] are once again leveraged. The equation presented in this work is as follows:

$$\varepsilon^* = \frac{1}{(N+1)(\frac{1}{\varepsilon_A} + \frac{1}{\varepsilon_B} - 1)}$$

The above equation takes two simplifications that will be important to keep in mind as the concept of MLI is further explored in this section; those being as follows:

1. Length between the layers that compose the MLI are assumed to be small and equal between all layers.
2. As stated before, the MLI above does not contain the spacers usually placed between the layers. Meaning that the above equation only considers heat transfer by radiation.

Because the equation above also specifies that opposite sides of the reflective layers are made out of different materials, they would then have a different value. If both sides of the reflective layers are made of the same material, which is to say that ε_A and ε_B have the same values, then:

$$\varepsilon^* = \frac{1}{(N+1)(\frac{2}{\varepsilon} - 1)}$$

From the above equation the most important variable for the analysis is that of emissivity (ε). To make a choice about the emissivity value, two criteria have to be constructed:

1. Temperature that the material can withstand. When introducing this section the temperature at the wall was calculated, concluding that (when faced with its harshest conditions) the system will face a temperature of about 821°C. Hence, material selection should be made with the predicted temperature and melting temperature of the– yet to be selected – material in mind.
2. Tautologically, the value of emissivity should also be given thought when considering emissivity value. Recall that emissivity is a value which “measures how efficiently a surface emits energy relative to a blackbody” as expressed in [3]. In the case that is currently being presented and analyzed, a behavior that is opposite to a blackbody is

desired; that being a material that can reflect most of the heat transferred to it. Putting the above two sentences into a mathematical language, if a blackbody has a value of emissivity equal to one then the material which will be selected for the MLI should have an emissivity value as close to zero as possible.

The materials that best fit the above two criteria are metals, which (in a general case) have a high melting point and low emissivity values. Additionally, such materials have already been investigated for similar purposes in such works as [9], [10], and [11]. With a melting point of 1063°C and emissivity of 0.025, gold fits the criteria presented above.

The number of layers that the MLI will be composed from is another parameter of importance, one that the moment has not been investigated. Due to the lack of investigation in this area, previous work was leaned on by [10] and [11]. In these two papers – which delt with similar temperatures and pressure which are expected in the module – three and four layers were investigated. With the values of emissivity from gold and a number of layers of three and four the two below effective emissivity can be calculated:

N	3	4
E*	0.008	0.006

Table 4: Effective Emissivity Values

With these values, heat transfer through the MLI can be calculated, shown below:

N	3	4
Q_{MLI}	4.345	3.476

Table 5: Heat Transfer Inside the Module

The values that are seen above are reasonable for heat transfer in MLI system – as expressed in the work of [4]. Regardless of the seemingly "positive" results that are seen above, it does not express much about what temperatures will be inside the MLI and satellite once it is all "said-and-done". Along with this, many other unknowns are still unaddressed by the above analysis, below are some of those:

1. This far, ideal MLI has been studied, which is to say, a system where pure radiation occurs. This analysis can lead to erroneous calculations of heat transfer withing the structure- as well as a bad approximation of weight. A better understanding of MLI would then be beneficial for further analysis. Luckily, research has been conducted in this area, but not much in the specific domain the GMPE satellite will be operating in. Two papers – those being [11] and [12]-have focused on the operating conditions that are expected to be encountered in the GMPE's trajectory. In the above work, lower and upper temperature and pressure respectively were set to the following: 300-1300K and $1.33x10^{-5}$-101.32kPa. The first equation presented is the energy equation for radiation/conduction:

$$\rho C_p \frac{\partial T}{\partial t} = \frac{\partial}{\partial x}\left(k \frac{\partial T}{\partial x}\right) - \frac{\partial q_r{}''}{\partial x}$$

Recall that the energy equation given above is for the cartesian coordinate system. This mean that the MLI that is being analyzed in these papers looks something as depicted below:

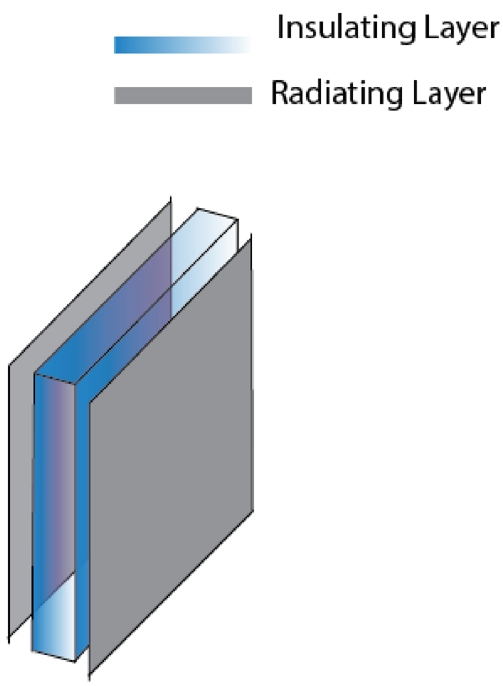


Figure 22: MLI Non-Ideal Configuration

With the equation and figure above, one single thing should be recalled, the module is circular in shape. To better picture what the module's predicted MLI will look like, figure below was made:

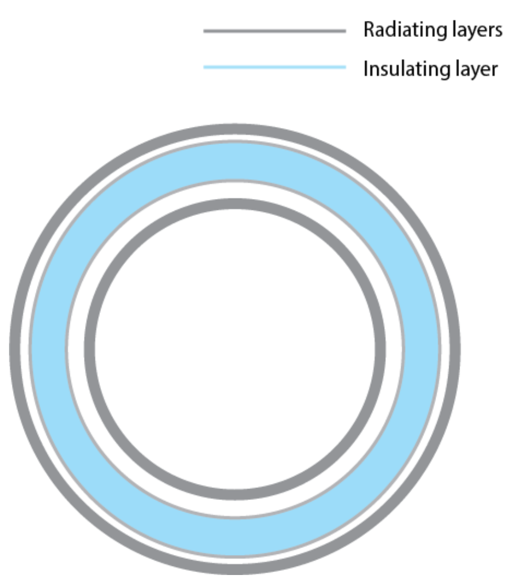


Figure 23: Proposed Circular MLI Insulation

The energy equation that would characterize the above MLI would then be expressed as follows:

$$\rho C_p \frac{\partial T}{\partial t} = -\frac{1}{r}\frac{\partial}{\partial r}\left(-kr\frac{\partial T}{\partial r} + q_R\right)$$

The above equation is now better suited for the heat transfer that the MLI will be subject to inside the module. With that said, there is still a need to expand on the q_R term seen above – if for no other reason as to show how the equation complicates itself. When heat transfer by radiation is done, simplifications of the system are the first to be mentioned; this writing will be no renegade in this aspect. As seen in the work [11]. [12], [13], [14] there are two simplifications that are the most popular: gray-medium and diffuse surface approximations. Gray-medium approximation allows for coefficients of absorption and scattering to be independent of the radiations wavelength; similarly, diffuse surfaces approximation allows for intensity of radiation to be independent of direction. These two approximations, lead to the equation which is shown in [13]:

$$-\frac{dq_R}{dy} = 2\kappa[B_1 + B_2 - 2e_b(y)]$$

$$B_1 = \epsilon_1 e_{b1} + 2\epsilon_1 \left\{B_2 E_3(\tau_0) + \int_0^{\tau_0}\left[\frac{k}{\beta}e_b(t) + \frac{\gamma}{4\beta}G(t)\right]E_2(t)dt\right\}$$

$$B_2 = \epsilon_2 e_{b2} + 2\epsilon_2 \left\{B_1 E_3(\tau_0) + \int_0^{\tau_0}\left[\frac{k}{\beta}e_b(t) + \frac{\gamma}{4\beta}G(t)\right]E_2(\tau_0 - t)dt\right\}$$

Where:

B – is surface radiosities

Subscripts 1 & 2 denoting the surfaces being analyzed

The issues with the above equations should be obvious to anybody that has been reading this thesis, the coordinate system used is in the y-direction. Here is where the author must admit defeat, as the knowledge that is currently held in the topic does not suffice to take this discussion any further. It was attempted to change the above coordinate system to cylindrical, it proved to complex for the usual ignorance that has been displayed in this current work. Out of respect for the subject the analysis of the math will stop here. Papers on the subject are present as in [14]; but any attempt by this paper to touch on the subject would only serve to further confuse. This stop in the analysis is done in the hope that someone with the interest in the subject picks-up the challenge presented here.

2. Weight is another important facet of MLI construction. If a MLI is given too many layers then the risk of reduced performance has been shown, with the added risk of making the satellite too heavy to meet specifications set by Cal-Poly.
3. Final facet to consider in MLI construction is the durability of the material when facing the thermal loads present. A study conducted in [15] did exactly this to explore how the radiation shields, Kapton layers, and adhesives are affected by the temperatures they were subjected to. This study found that at temperatures of 500°C the adhesive started to melt, but the other materials withstood the temperature load. Despite the lower temperature the paper presents a good methods and test set-up to which would serve as a good base when performing the test for the GMPE MLI.

These aspects leave the MLI analysis with much more questions than answers. Regardless of this fact, this section is written to advance the conversation about the thermal protective system. The conclusions that have been made at this point is then the number of layers which will compose the MLI, the materials that will compose the MLI gold (reflective layer), saffil (insulation layer), and quartz thread (as a uniting technique).

Down the Pipe

In the case that the reader has forgotten where the analysis is at present, the semblable and module are shown below:

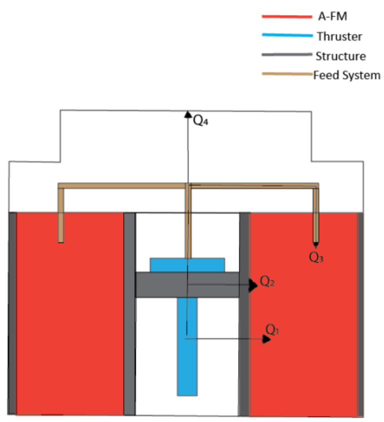


Figure 24: Semblable of Proposed Module Heat Transfer

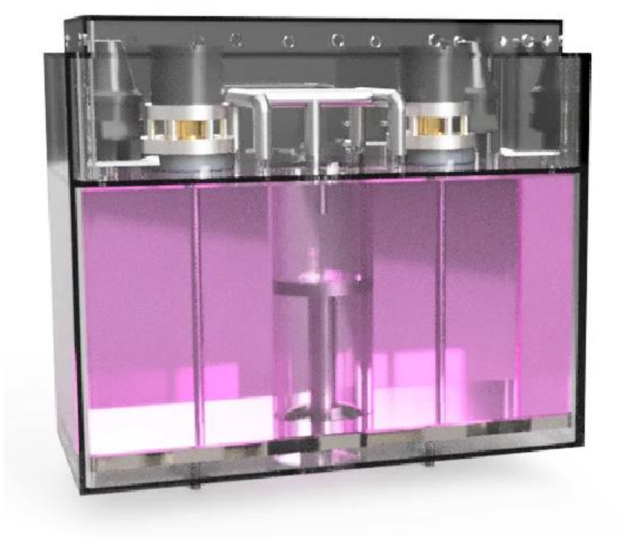


Figure 25: Module

At this point, the analysis is at the section were the feeding-pipe and the thruster meet. This being, where heat transfer – denoted by Q_3 – will occur from the thruster into the pipe further into the module and ultimately into the ASCENT tank. A moment should be dedicated to explaining where the importance of analyzing heat transfer through a stainless steel pipe would be.

Allow for the starting point to be the simple fact that a stainless-steel pipe will not melt under conditions that the thruster presents. Which is to say, structurally the pipe is in no danger; but the heat it conducts into the module & heat it transfers into the propellant that flows inside the pipe might both be problematic. To prove that something must be changed in the module only one of the above-mentioned dangers has to be true. First an analysis of the heat transfer occurring through the pipe and into the module will be done. For this a simple heat transfer by conduction can be done, leading to an answer to the concerns presented above. Recall the equations for heat transfer and resistance in cartesian coordinates:

$$\dot{Q} = \frac{T_H - T_C}{A \times R}$$

$$R = \frac{L}{k * A}$$

L – length of the specimen that heat will be traveling through, in the above analysis this is the length of the pipe from the thruster to the lid;39mm.

In the system being analyzed there is only two things are constant the length of the specimen and the hot temperature – this fact comes a bit prematurely but will be important later. Thermal resistivity can then be calculated:

$$R = \frac{L}{k * A} = 732.60 \frac{K}{W}$$

$$\dot{Q} = \frac{T_H - T_C}{A \times R} = 643.57 \frac{kW}{m^2}$$

This value of thermal resistance is large and would imply a high resistance from the structure (in this case the stainless steel feed-line) to heat transfer; for the above a large thermal gradient can be expected in the structure. However, if the above value of thermal resistance is seen with a degree of skepticism, then a common language has been established! If memory serves, highest value of resistance that was seen in the insulated circular wall being 246.28K/W, the stainless steel feed-line – in contrast being a highly conductive material for temperature – is giving a much higher thermal resistance than this. The heat flux immediately gives a better idea of what is occurring, with a value of $643.57\frac{kW}{m^2}$ of heat that must be dissipated through the specimen which is huge for the thermal gradient speculated above to be achieved. This means that the propellant will enter the feedpipe at a temperature higher than that considered suitable. With this conclusion an introduction to thermal standoffs is merited.

The concept behind the thermal standoff is quite simple, by increasing the area the heat can dissipate through then the temperature at the end of the structure will be much lower. Thermal standoff look something like the one in the picture below:

Figure 26: Thermal Standoff

As can be seen in the above thermal standoff, the structure does not only consist of increasing the area but also of removing pathways from which the heat to travel through. The most technical explanation as to why the holes are present in the design above is expressed in [16] which states “Thermal isolation is produced by staggered holes resulting in torturous path for heat flow”. When searching for other insights on how to construct the standoff not many other wise words were found, with the best advice received being part of [17] where they state “the length of the thermal standoff is determined using a simple thermal analysis”. Hopefully, due to the length of this paper – and the supposed rigor in which it has delved into developing these ideas – the words “simple” in the above sentence is seen as an antiphrasis of what is really necessary to develop these structures. Other such papers were consulted such as [19], [20], and even one from the alma mater [21]. These did not share any mathematical methods for determining the possible results of a standoff, but some design philosophy was given as well as verification by the avenue of simulation. With that said this paper will explore similar avenues by not explaining anything and jumping straight into two design that were constructed. The two below design were CADed when investigating the topic:

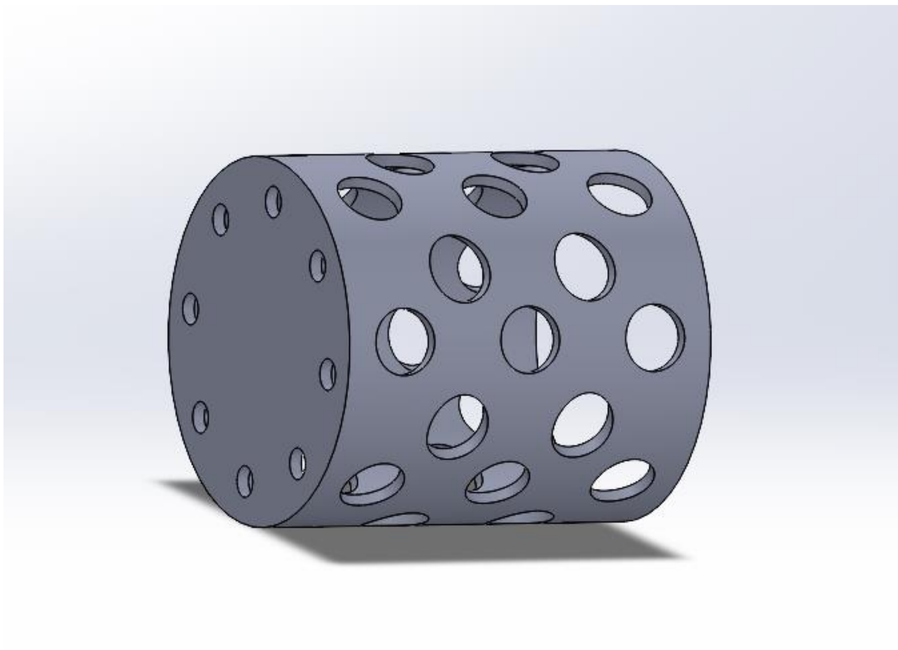


Figure 27: First Proposed Thermal Standoff

The above thermal standoff was created with the length constraint inside the module with mind and the advice of staggered holes. The second design:

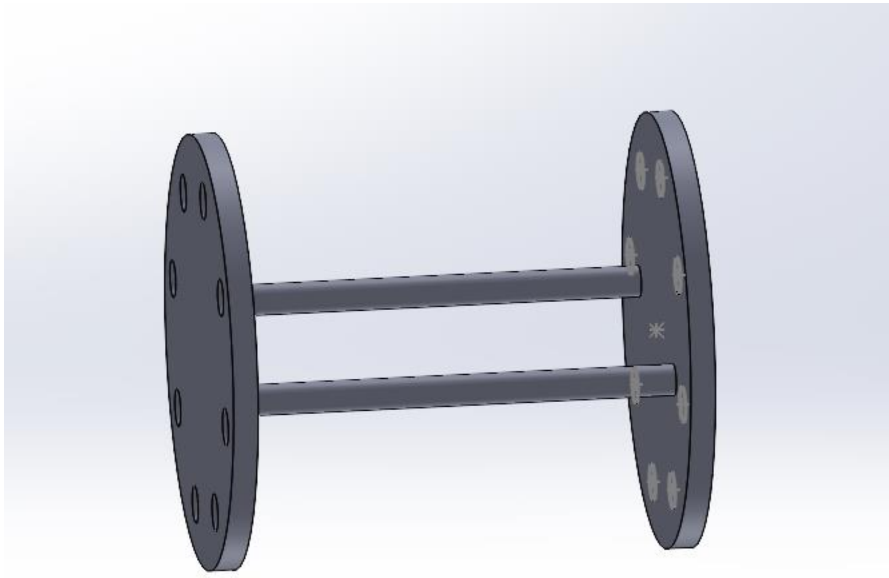


Figure 28: Second Proposed Thermal Standoff

The above followed the design that is currently used as a thermal standoff when hot-fire testing occurs for the GMPE. Both of the above designs interface with the thruster and even though it has not been integrated yet, making the thermal standoff work as an injector would allow for less “wasted space” when interfacing the thruster to thermal standoff interface.

Next step to consider is how to conduct the simulation for these structures; discerning from the first simulated scenario where the thruster is allowed to radiate its heat to the environment vs. when placed inside the module – where the difference is the temperature faced at the end of the thruster. For sake of isolating the analysis to how much heat is dissipated/isolated by the thermal standoff, a simulation with the thermal standoff incased will not be performed. This is not to say that an analysis of heat transfer into the environment is appropriate – this concept will be expanded on further. Temperature at the hot side of the thermal standoff will be set at 1000°C, which is the temperature at the thruster’s end when encased. Ambient temperature that will be taken around the thermal standoff will be that which was calculated in the first section of this paper, that being -22°C. Below is the two cases which were run through the thermal standoffs presented above:

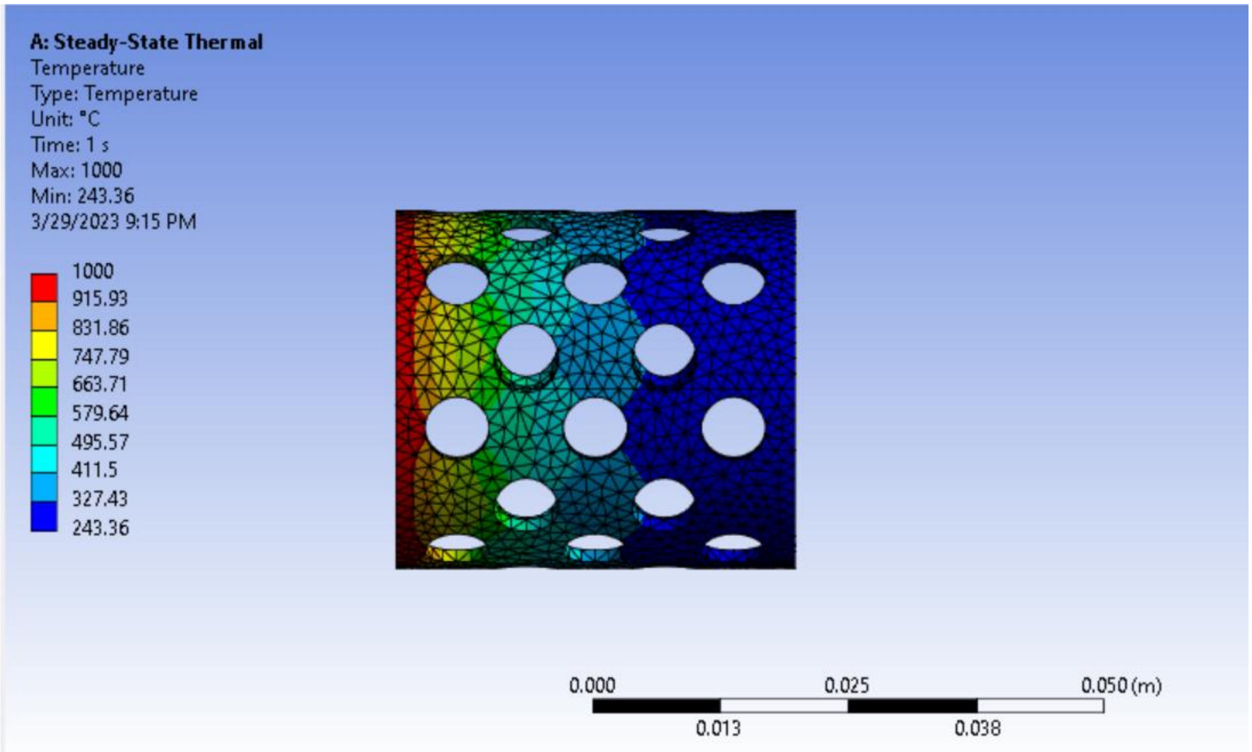


Figure 29: First Proposed Thermal Standoff Temperature Gradient

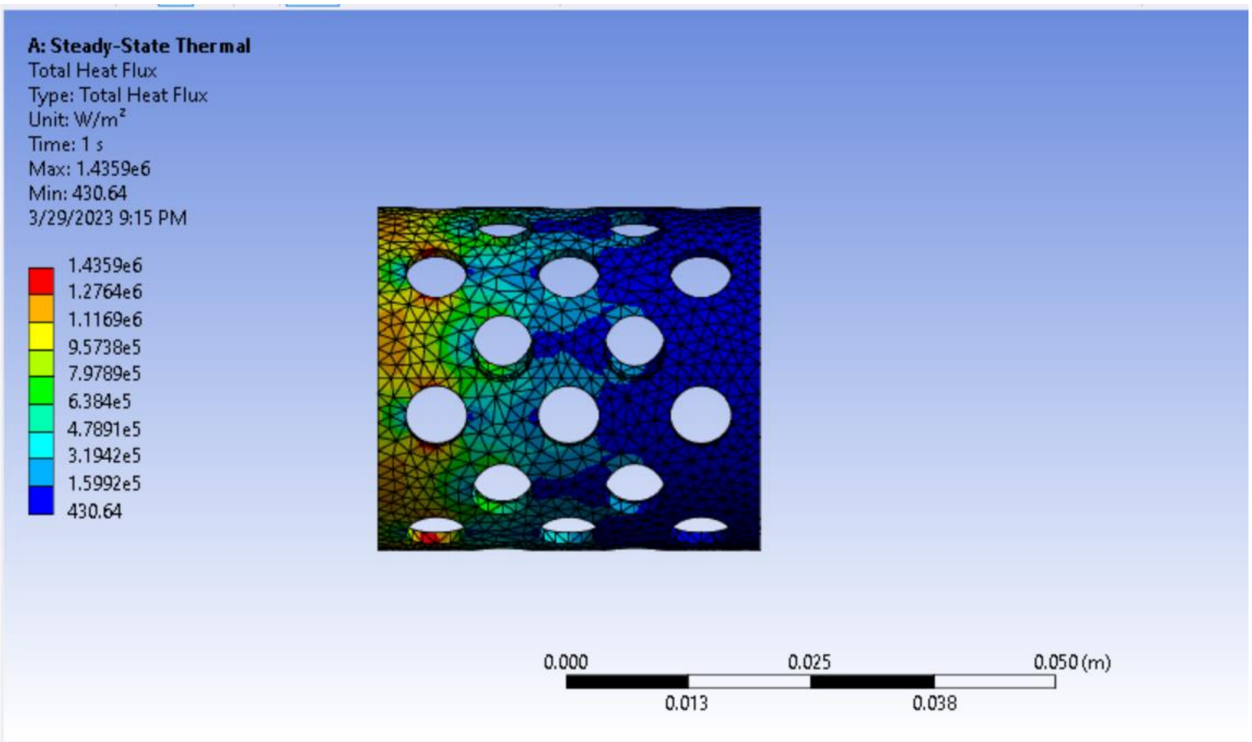


Figure 30: Thermal Standoff Heat Flux

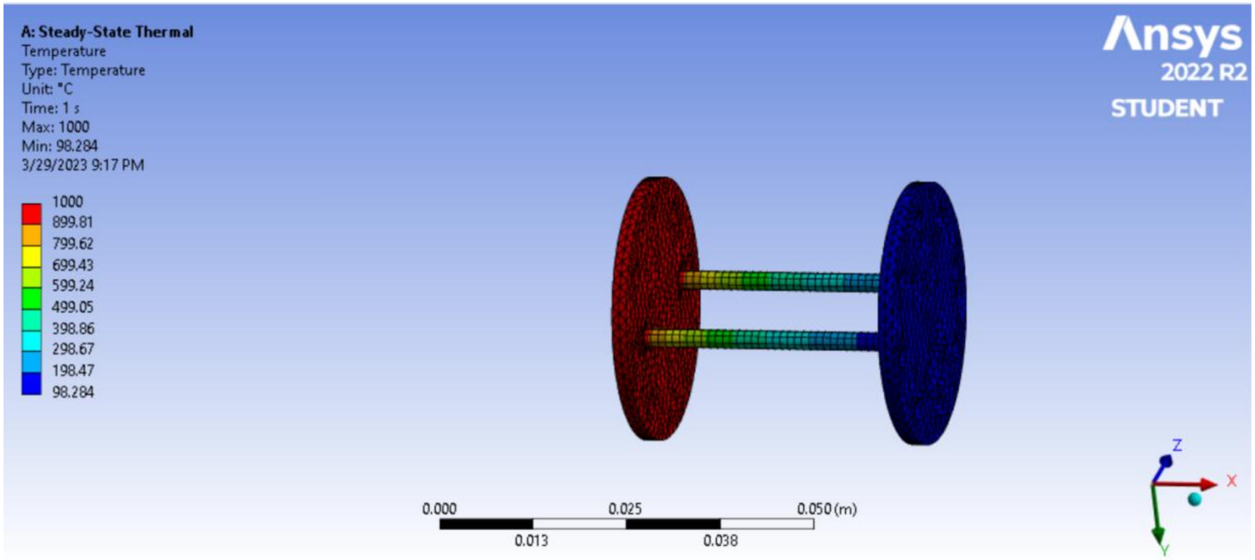


Figure 31: Second Proposed Thermal Standoff Temperature Gradient

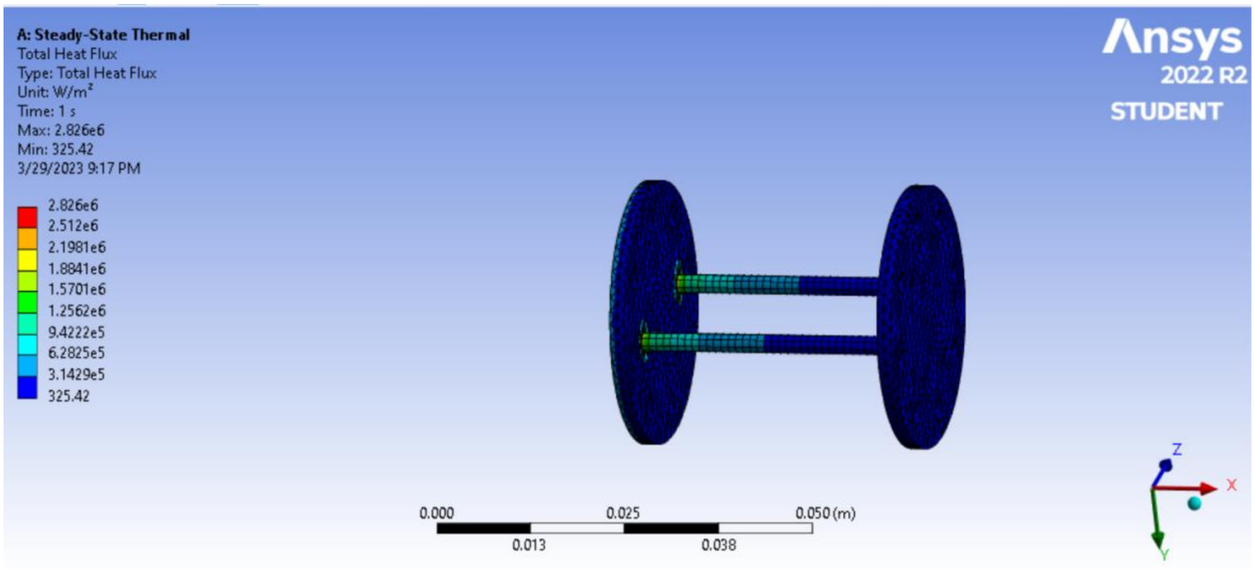


Figure 32: Second Proposed Thermal Standoff Temperature Gradient

As can be seen by the above simulations, both of these thermal standoffs fulfill their task of reducing the temperature which they are being subjected to; with one showing better results than the other. Explaining why the above simulations take the thermal gradient shown in them is a matter of – hopefully this is not an incorrect statement - understanding the allowable area in which the heat transfer is allowed to occur. The first design allows for more radiation to take place – this is said in comparison to the second structure – with the environment that surrounds it; similarly, it allows for conduction through its own body to occur, leading to the temperature at the end of the structure of 243.36°C. In comparison, the second design allows for less radiation to occur due to its reduced area, along with that it allows for much less conduction to occur throughout the body. This reduced heat transfer by conduction allows for a higher temperature difference between one body and the other, as seen in the end temperature of 98°C. Question now becomes, which thermal standoff is better suited for the module? Answering this question needs to be done through the colored lenses that the modules geometry paints, in other words, there will be no place to radiate the heat inside the module; isolating the temperature becomes the better optimal option in comparison to dissipating the temperature. The question might once again be raised, if reducing the area in which heat can be transferred is the answer, then why not simply leave the pipe? This question will be answered in much more detail at the "future work" section of this paper.

The Final Stretch

This section would be discussing the heat flow from the electronics to the back of the module, but due to the fact that there has been no electronics picked yet this section will be skipped. The propellant will be safe as was seen in the above analysis as the temperature will be high, but not enough to bring it to critical conditions. The back of the module will be important to verify but will depend on how much heat is being dissipated by the electronics that are mounted on it.

Possible Errors

Regardless if it is believed that all the analysis done in this work is correct, it is prudent to describe were the above analysis can be seen as either too simple to grasp the complete reality of the module or were simple mistakes could occur. This is done with the hope that future work can use their own discretion when carrying out similar analysis – if they find the work they saw here as worthy of emulating in the first place.

Starting at the first step of the analysis – which is earlier than expected, but exactly were one can expect issues to pop-up – the simplification of the snapshot should be investigated further. Because the satellite's journey can't be fully described in a snapshot, a transient analysis of the module would be much more accurate. The benefit of the transient analysis is rather expansive, with it allowing for the analyst to see the "cooling rate" of the module – cooling rate here is used as a means of saying how quickly it rejects heat to the environment when the thruster stops firing and the module is allowed to travel without the thermal load. Along with this a second point rears its head, a transient analysis would possibly allow for the temperature inside the module to be investigated – allowing for a change in the environmental temperature that was constantly used in this analysis. A transient analysis will allow for the temperature inside the module (empty space between components) to be more thoroughly analyzed; if similar temperatures as seen here are present then not many changes, if the temperature increases substantially the module is still in danger. If the former is the case, then the module should be given an avenue to dissipate excess heat- this is usually done by increasing the area of the module.

Second point where possible errors can occur is in the analysis of the thruster firing (that would make it two-for-two baby!). When discussing the temperature gradient that was presented by the simulation and that this work accepted, it should be seen as perfidy. Primarily, the temperature of the thruster should be centered at the catalyst bed, where the decomposition takes place. Simulating the high temperature that are expected in the catalyst bed proved to be much more complex, to achieve reported temperatures in the simulation – as those which were presented to the author - would require heat transfer rates which would not be justifiable other than to achieve the temperature predicted. Were can this affect the rest of the analysis? It can be suspected that the temperature were the thruster meets the wall might be higher by 200°C. It will be discussed in the next sections why this is seen as not exactly inconsequential but not a cause for further analysis.

In the wall section of the analysis, one fact was ignored which if accounted for with analysis would immediately prove problematic. Below is how the thruster will fit into the module:

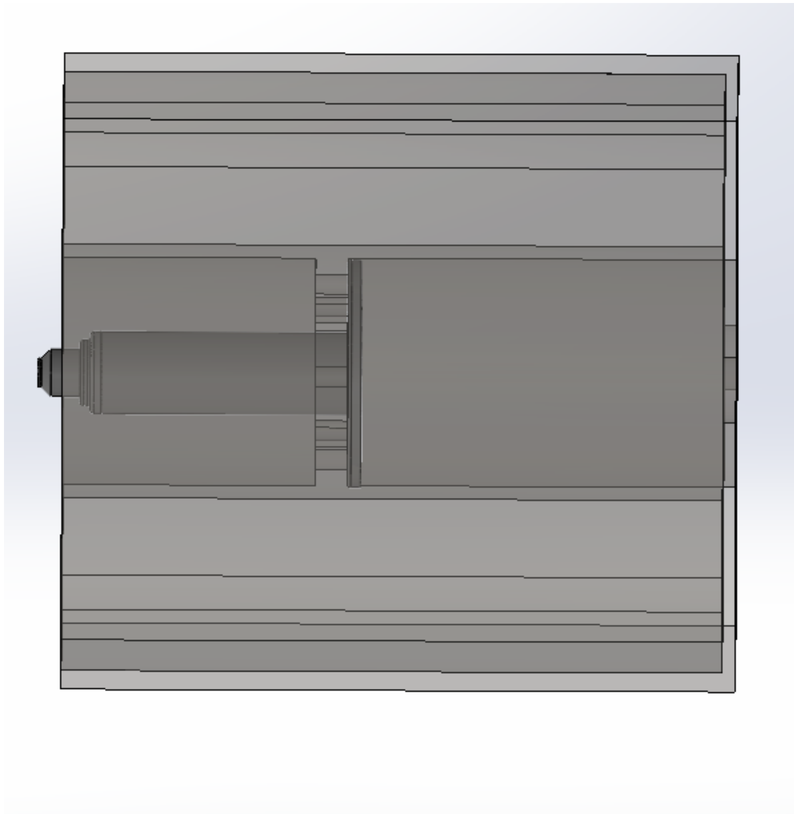


Figure 33: Thruster Inside of Module

The analysis of the wall presented above took the heat from the thruster body being dissipated into the wall in a cylindrical manner but neglected the flange touching the same wall. The flange is touching that wall at all points, and is predicted to be around the temperature of 1000°C, which means all that heat will be dissipated directly into the wall. Due to the design of the module no insulation could be added in between these two structures as the thruster has to mate with the wall at this point. To negate this fact the best option – that can be thought of - would be to eliminate the wall entirely, allowing for the thermal standoff to extend further. As is shown below:

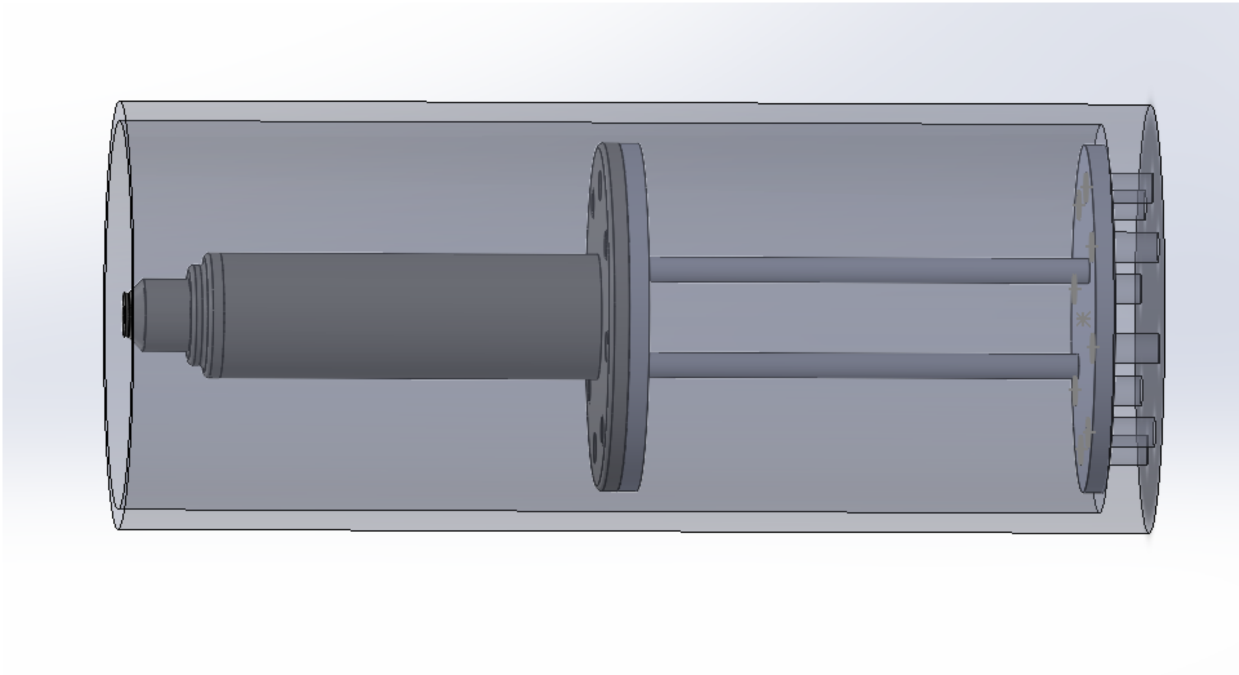


Figure 34: Proposed Configuration in Module Without Wall

Even with this solution there is another issue, the module hugs the flange which once again will be at temperatures too critical for the propellant to be in close contact with. The best recommendation that can be made in this aspect is to redesign the flange to be smaller than what it currently is, or making the wall bigger (as was done in the figure above).

Other than these multiple issues the analysis above has been iron clad- unfortunate as it is to say that.

Future Work

For future work, identifying how to test the temperatures inside the module once a physical version is manufactured is what will progress this work from the theoretical to the empirical. A test proposal is demonstrated below:

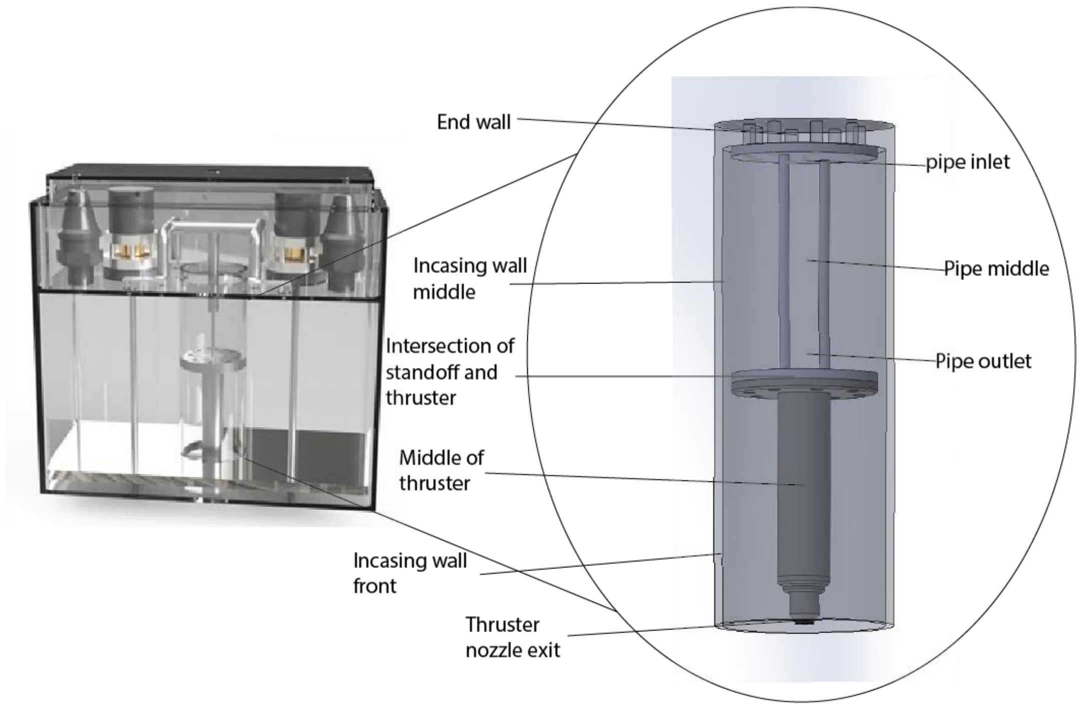


Figure 35: Testing TC Layout

Figure above shows where having measurements of the temperature would be beneficial for further analysis of the module. The importance of each of these is better explained below.

- Temperature gradient of the thruster should be investigated to verify if – as the simulation suggest – the reflection of heat by the structure surrounding it leads to the drastic change in temperatures shown previously. Based on that knowledge the analysis here can be either disproved or accepted, with thermal insulation then being adjusted accordingly around this area. If the thruster never reaches the high temperatures demonstrated here, then a simplification of the suggested thermal mitigation here can be performed.

- Having recorded temperatures at the “encapsulating wall” is of importance because a simulation is only as good as its inputs, and conduction can only be investigated reliably if a recorded temperature is had. Furthermore, there may be a chance that due to the redesign and new area for heat to dissipate into the temperature of the “encapsulating wall” is smaller than previously predicted. This can imply that MLI is needed with different layers at different sections, as only the area close to thruster will experience extreme temperatures.
- Monitoring the temperature inside the feedline is still of importance. In the above figure three points are shown, but depending on the amount of room that is available it can be reduced to two (inlet and outlet) or one (middle of feedline). These measurements should be taken to see if the thermal standoff is accomplishing its task.

- End wall: Due to the nature of the electronics that will be needed at the top portion of the module, having a clear idea of what temperature will present in this section of the module is of clear. It is estimated as of this moment that the temperature could be as low as 60°C; unfortunately, this was with a simplified version of the thermal processes.

With the above points being made in the hope that the individuals that will have to carry out these tests don’t see their execution as just a pointless exercise – regardless of their appearance.

Conclusion

It is difficult to truly distinguish if something has been achieved in this paper due to the nature of what was being analyzed, and the uncertainty that the analysis sparks from. If in future work it is proven that the thruster is achieving its 1N thrust and the temperatures that it produces are constant and reliable, then the analysis presented in this paper will be more concise. Along with this once the verified 1N thruster is put inside the module then the real analysis for heat transfer can begin. In the meantime this work should be seen as setting the basis on how to analyze the module once both of the above steps are finished, and as a warning on what type of temperatures are to be expected once these steps are completed. If anybody finds the work done here to be of any help, the author finds himself compelled to add a couple of points to further assist future engineers. An analysis of the convection between a wall and the propellant would be incredibly useful, it was not done here because the properties of ASCENT have not been investigated to this extent. Admittedly, this work could have used a general fluid to develop the analysis and insert properties later – here it should be admitted that time got ahead and didn't allow for this type of analysis to occur. Along with this, it was brought to the attention of the author that the environment calculations can be skipped and just taken from temperatures of past missions. This point seems obvious in retrospect, but that is what hindsight does. Regardless, hopefully this work helps on expanding the conversation on how a heat transfer analysis should be conducted in a CubeSat and encourages more works of this type.

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Appendices

MATCAD CALCULATIONS

Heat Transfer:

Finalizing Thermal

After recent realizations it is important to revisit the thermal calculations that were made preceding these to verify the results. There will be three major rehauls of the previous work - which if you were not following (and lets be honest who was) is Thermal_8_24_22 - those being as follows:

1. For all the previous works there was a term used to explain the heat giving by the decomposition. In it the hottest temperature that was used for decomposition was 400$^{\circ}C$, this has been discussed with me further and the temperature expected is closer to 1200$^{\circ}C$. Hence a change has to be made
2. Revisiting the past calculations, it was obvious that even though I tried to explain the concepts to the best of my abilities it felt short. A better layout of concepts and how they will be used will be attempted
3. Less jokes, or only when they are truly useful for the concepts that will be presented

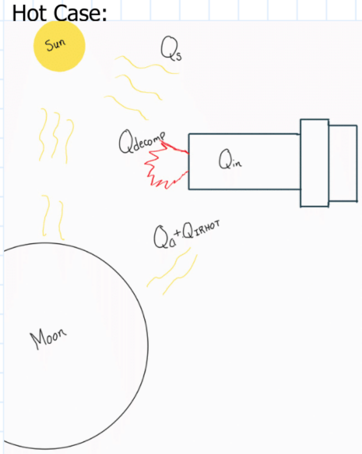
Heat transfer from environment

When I was first asked to make these calculations the effect of the environment on the module was heavily emphasize. It was advised to me, to conduct two separate instances of heat propagation, those being the following:

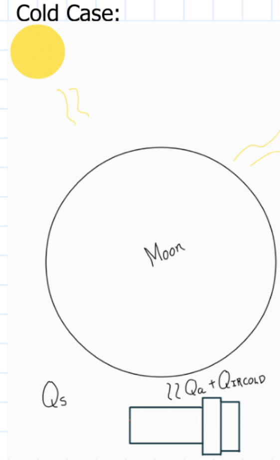
1. The hottest case that the module will experience
2. The coldest case the module will experience

These two cases can happen in the following scenarios; which are depicted below:

Hot Case:



Cold Case:



The hot case can be attributed to the following factors:

1. Q_s - Solar energy
2. Q_a - Albedo

3. Q_{IR} - Infrared radiation
4. Q_{decomp} - decomposition of AF-M
5. Q_{in} - Heater

The pictorial above combined with the labeling of the heat transfer should lead us to understand each individual situation as follows:

1. Hot Case: When the satellite is orbiting the lit side of the moon, the radiation that is being transferred to the system (worst case scenario) is that of the sun, albedo, Infrared, decomposition of AF-M, and the heater
2. Cold Case: When the satellite is orbiting the dark side of the moon and won't be firing, then we will be left with three forms of heat transfer sun, albedo, and infrared.

With all of this discussed I hoped the topic has been made justice in speech and now the calculations will seem much less scary. The equations for each of the heat transfers above can be seen below:

$$Q_s = \alpha_s \cdot S \cdot A_{surf}$$

$$Q_{albedo} = Alb \cdot \alpha_s \cdot S \cdot A_{surf} \cdot F_{surfmoon}$$

$$Q_{IR} = \varepsilon_s \cdot MIR_{flux} \cdot A_{surf} \cdot F_{surfmoon}$$

$$Q_{decomp} = m' \cdot C_p \cdot \left(T_{hot} - T_{cold}\right)$$

Next we will establish the constant which have been identified here.

$\alpha_s := 0.031$	$A_{surf} := 0.06\ m^2$	$MIR_{fluxHot} := 93\ \frac{W}{m^2}$	$MIR_{fluxCold} := 32\ \frac{W}{m^2}$
$S := 1361\ \frac{W}{m^2}$	$F_{surfmoon} := 0.7$	$m' := 0.09 \cdot 10^{-3}\ \frac{kg}{s}$	$C_p := 2230\ \frac{J}{kg \cdot K}$
$\varepsilon_s := 0.3$	$T_{Hot} := 1200\ °C$	$T_{Cold} := 400\ °C$	$Alb := 0.2$

Here we might want to change referance in Albedo

α - http://www.thermalfluidscentral.org/encyclopedia/index.php/Absorptivity_and_emissivity_of_metal_and_deposited_coating_on_metals

ε - *Fundamentals Heat & Mass Transfer*

S - Introduction to Spacecraft Thermal desing

A_{surf} -Area of the surface (calculated on another MathCAD ModuleAreaCalculations)

MIR_{flux} - From INFRARED STUDIES OF THE LUNAR TERRAIN

Alb - https://cneos.jpl.nasa.gov/glossary/albedo.html#:~:text=Our%20Moon%20has%20a%20very,a%20high%20albedo%20(0.60).

m' - mass from past catch and weigh

With all that we can finally calculate the temperature which will be felt in the module at each case:

$$Q_s := \alpha_s \cdot S \cdot A_{surf} = 2.531\ \frac{kg \cdot m^2}{s^3}$$

$$Q_{albedo} := Alb \cdot \alpha_s \cdot S \cdot A_{surf} \cdot F_{surfmoon} = 0.354\ \frac{kg \cdot m^2}{s^3}$$

$$Q_{IRHot} := \varepsilon_s \cdot MIR_{fluxHot} \cdot A_{surf} \cdot F_{surfmoon} = 1.172\ \frac{kg \cdot m^2}{s^3}$$

$$Q_{IRCold} := \varepsilon_s \cdot MIR_{fluxCold} \cdot A_{surf} \cdot F_{surfmoon} = 0.403\ \frac{kg \cdot m^2}{s^3}$$

$$Q_{decomp} := m' \cdot C_p \cdot \left(T_{Hot} - T_{Cold}\right) = 160.56\ \frac{kg \cdot m^2}{s^3}$$

$$Q_{in} := 10\ W$$

Hot case temperature calculation. This is taken at the lit part of the moon (so aliens throwing parties with drugs and alcohol, hangovers all around) and thurster firing:

$$T_{hot} := \left(\frac{Q_s + Q_{albedo} + Q_{IRHot}}{\varepsilon_s \cdot \sigma \cdot A_{surf}}\right)^{\frac{1}{4}} = 251.101\ K$$

$$T_{hot} = -22.049\ °C$$

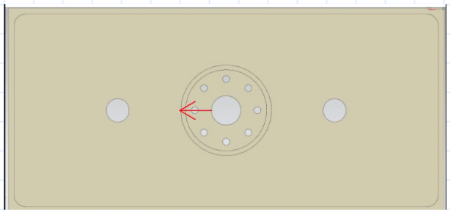
Cold case temperature calculation. This is taken in dark side of the moon (so alien teenagers reading poetry and signing 'Hey There Delilah' the only song they now how to play on guitar.):

$$T_{cold} := \left(\frac{Q_s + Q_{albedo} + Q_{IRCold}}{\varepsilon_s \cdot \sigma \cdot A_{surf}}\right)^{\frac{1}{4}} = 238.258\ K$$

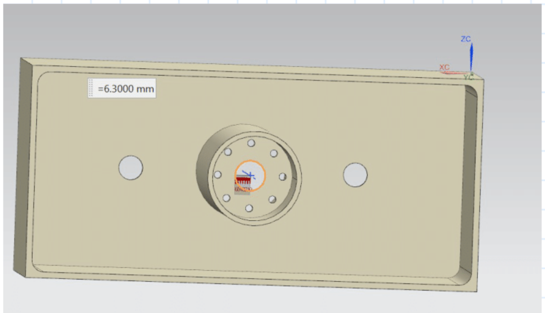
$$T_{cold} = -34.892\ °C$$

Thruster-Wall-AFM

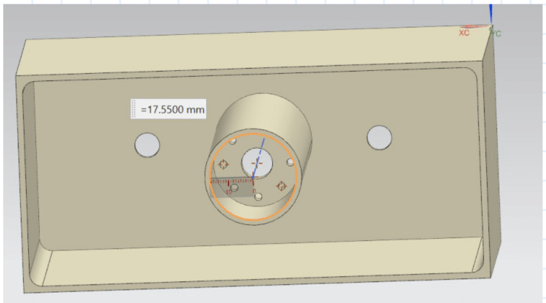
In this section the focus will be that of the heat transfer from the thruster to the AFM by conduction.



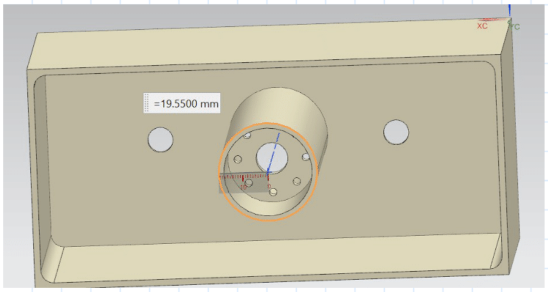
The gap in the middle is where the thruster will rest. As the thruster is firing then we can expect the heat to transfer towards the AFM tanks; how will this affect the module? First lets see the distance from the thruster to the AFM.



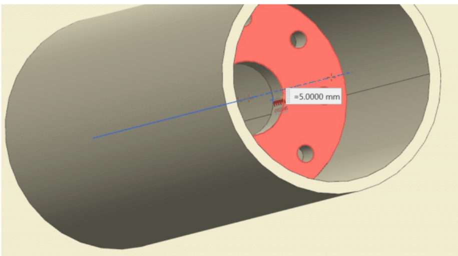
$r_1 := 6.3\ mm$



$r_2 := 17.55\ mm$



$r_3 := 19.55\ mm$



$$L := 5\ mm$$

With all these measurments, it is now possible to determine the resistance that this structure will provide to the heat. This portion of the strucutre will be analyzed as a tube allowing this analysis to be carried out radially. The equation for radial heat transfer is the following:

$$T(r) = c \cdot ln(r) + k$$

This equation is gotten by solving the steady state heat diffusion equation, with the assumption made that no heat generation is occurring between the thruster and the wall. As can be seen in the picture above we would have two surfaces that the heat will be traveling, the analysis produced below will be done as a composite surface. With all that said it can be shown that what will be present in the system can be understood with the following boundary conditions:

$$r_1 \le r \le r_2 \qquad\qquad r_2 \le r \le r_3$$

$$T_1(r_1) = T_1 \qquad\qquad T_2(r_2) = T_2$$

$$T_1(r_2) = T_2 \qquad\qquad T_2(r_3) = T_3$$

After you solve these equations we get the following:

$$T_1(r) = T_1 - \frac{T_1 - T_2}{ln\left(\frac{r_2}{r_1}\right)} \cdot ln\left(\frac{r}{r_1}\right) \qquad\qquad T_2(r) = T_2 - \frac{T_2 - T_3}{ln\left(\frac{r_3}{r_2}\right)} \cdot ln\left(\frac{r}{r_2}\right)$$

By now exploring Fourier's law we can get to the next important realization:

$$q = -k(2\ \pi \cdot r \cdot L)\ \frac{dT}{dr}$$

by deriving the two equations above and substituting $\frac{dT}{dr}$ the following result is achieved:

$$q_1 = k\left(2\ \pi \cdot L\right) \frac{T_1 - T_2}{ln\left(\frac{r_2}{r_1}\right)} \qquad k_{Ti} := 20\ \frac{W}{K \cdot m} \qquad q_2 = k\left(2\ \pi \cdot L\right) \frac{T_2 - T_3}{ln\left(\frac{r_3}{r_2}\right)}$$

k-https://www.hardrok.com.au/thermal-activity-in-stainless-steel-compared-to-other-metals#:~:text=Thermal%20Conductivity%20Of%20Stainless%20Steel,watts%20per%20kelvin%20per%20metre.

It is common practice in heat transfer to simplify the equations above by making the analogy to electric resistance and realizing that in heat transfer resistance would then be:

$$R_1 = \frac{ln\left(\frac{r_2}{r_1}\right)}{k \cdot 2 \cdot \pi \cdot L} \qquad R_2 = \frac{ln\left(\frac{r_3}{r_2}\right)}{k \cdot 2 \cdot \pi \cdot L}$$

Simplifying the Fourier's equation with these new definition leads to a simplified equation of:

$$q_1 = \frac{\left(T_1 - T_2\right)}{R_1} \qquad q_2 = \frac{T_2 - T_3}{R_2}$$

Now the reader can be wondering: why all of this is important? Not for the explanation of theory (that they can be sure of) but to get to the following point. As a group we are not sure what the actual heat transfer will be of the module at the moment of writing these calculations. Right now we have three "know" values of those shown above; resistance of both cases and the initial temperature.

$$R_1 := \frac{\ln\left(\frac{r_2}{r_1}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 1.631\ \frac{s^3 \cdot K}{kg \cdot m^2} \qquad R_2 := \frac{\ln\left(\frac{r_3}{r_2}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 0.172\ \frac{s^3 \cdot K}{kg \cdot m^2}$$

These two values are rather low, the thing to consider here is the temperaute that the propellant (AFM) will be at. In the AF-M datasheet there is experiments that take place from the values of $15\ °C - 80\ °C\ (288\ K - 353\ K)$, for our case we will consider AF-M at $80\ °C$. Now the temperature that is being seen by the thruster is still up in the air, but with a temperature of decomposition as high as $1200\ °C$. So let's do the safe thing and take on average to see were we would have to be at T_2 for this to happen

$$T_{2avg} := \frac{1000 + 353}{2}\ K = 676.5\ K \qquad T_{1first} := 1000\ °C \qquad T_{3ideal} := 353\ K$$

$$q_1 := \frac{\left(T_{1first} - T_{2avg}\right)}{R_1} = 365.92\ \frac{kg \cdot m^2}{s^3} \qquad q_2 := \frac{T_{2avg} - T_{3ideal}}{R_2} = \left(1.883 \cdot 10^3\right)\ \frac{kg \cdot m^2}{s^3}$$

What do these values mean? To answer the question presented by the author to itself it is sufficient to copy what a smarter person than himself said. In the book *Fundamental of Heat and Mass Transfer* the following is said:

"A simple, yet general, definition provides sufficient response to the question: What is heat transfer?

Heat transfer (or heat) is thermal energy in transit due to temperature difference"

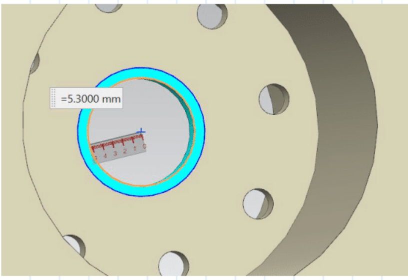
By hanging onto the section that "simply" states "thermal energy in transit" then the following thought can be put together. The values that are seen above for heat transfer rate are then:

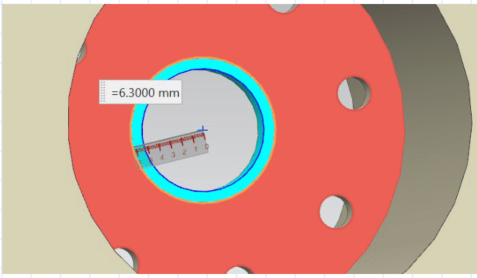
1. Heat transfer rate the wall MUST allow for, in order for the propellant to be a the speculated temperature of $80\ °C$

The real point that the reader should understand is that this is not a probable scenario. Something else will have to be done to get the temperature to something more reasonable, that something is insulation!

Adding Insulation to the Mix

The idea with adding insulation is to make the heat stay at that section of the composite where the insulation is added. In other words - by which the reader should realize that the author of this work is nothing if not an idiot - the insulation should work as insulation! It was concluded in the past that MaxFire hp could be a good candidate for this role with a conductivity of $k_{Mhp} := 0.085 \dfrac{W}{m \cdot K}$. What will now have to be analized, how much insulation will be needed? To answer that question we must go back into our module and imagine it with insulation. Consider the following:





Adding one millimeter of insulation leads to the following

$$T(r) = c \cdot ln(r) + k$$

$$r_1 := 5.3\ mm \quad r_2 := 6.3\ mm \quad r_3 := 17.55\ mm \quad r_4 := 19.55\ mm$$

$r_1 \le r \le r_2$ $\quad$ $r_2 \le r \le r_3$ $\quad$ $r_3 \le r \le r_4$

$T_1(r_1) = T_1$ $\quad$ $T_2(r_2) = T_2$ $\quad$ $T_3(r_3) = T_3$

$T_1(r_2) = T_2$ $\quad$ $T_2(r_3) = T_3$ $\quad$ $T_3(r_4) = T_4$

As the keen eye must have noticed these equations follow a pattern:

$$T_1(r) = T_1 - \frac{T_1 - T_2}{ln\left(\frac{r_2}{r_1}\right)} \cdot ln\left(\frac{r}{r_1}\right) \quad T_2(r) = T_2 - \frac{T_2 - T_3}{ln\left(\frac{r_3}{r_2}\right)} \cdot ln\left(\frac{r}{r_2}\right) \quad T_3(r) = T_3 - \frac{T_3 - T_4}{ln\left(\frac{r_4}{r_3}\right)} \cdot ln\left(\frac{r}{r_3}\right)$$

$$R_1 := \frac{\ln\left(\frac{r_2}{r_1}\right)}{k_{Mhp} \cdot 2 \cdot \pi \cdot L} = 64.727 \ \frac{s^3 \cdot K}{kg \cdot m^2} \qquad R_2 := \frac{\ln\left(\frac{r_3}{r_2}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 1.631 \ \frac{s^3 \cdot K}{kg \cdot m^2}$$

$$R_3 := \frac{\ln\left(\frac{r_4}{r_3}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 0.172 \ \frac{s^3 \cdot K}{kg \cdot m^2}$$

We can then calculate the heat transfer rate for each section

$T_1 := T_{1first} = \left(1.273 \cdot 10^3\right) \ K$ $\quad$ $T_2 := 700 \ K$ $\quad$ $T_3 := 400 \ K$ $\quad$ $T_4 := T_{3ideal} = 353 \ K$

$$q_1 := \frac{\left(T_1 - T_2\right)}{R_1} = 8.855 \ \frac{kg \cdot m^2}{s^3} \quad q_2 := \frac{T_2 - T_3}{R_2} = 183.987 \ \frac{kg \cdot m^2}{s^3} \quad q_3 := \frac{T_3 - T_4}{R_3} = 273.634 \ \frac{kg \cdot m^2}{s^3}$$

Here q_1 looks valid but eveything else still seems to high, up resistance to by adding more insulation

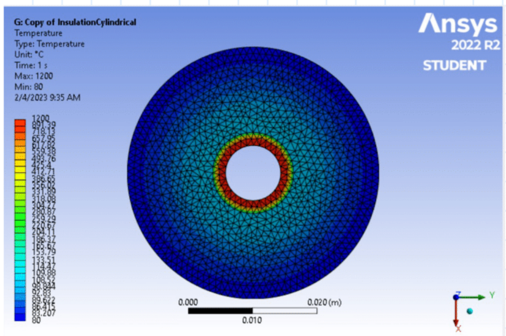
$r_1 := 4.3 \ mm$ $\quad$ $r_2 := 6.3 \ mm$ $\quad$ $r_3 := 17.55 \ mm$ $\quad$ $r_4 := 19.55 \ mm$

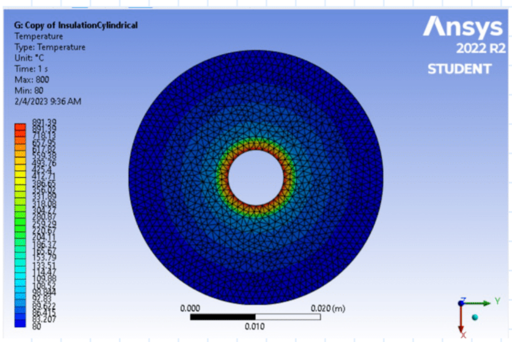
$$R_1 := \frac{\ln\left(\frac{r_2}{r_1}\right)}{k_{Mhp} \cdot 2 \cdot \pi \cdot L} = 143.028 \ \frac{s^3 \cdot K}{kg \cdot m^2} \qquad R_2 := \frac{\ln\left(\frac{r_3}{r_2}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 1.631 \ \frac{s^3 \cdot K}{kg \cdot m^2}$$

$$R_3 := \frac{\ln\left(\frac{r_4}{r_3}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 0.172 \ \frac{s^3 \cdot K}{kg \cdot m^2}$$

$$q_1 := \frac{(T_1 - T_2)}{R_1} = 4.007\ \frac{kg \cdot m^2}{s^3} \quad q_2 := \frac{T_2 - T_3}{R_2} = 183.987\ \frac{kg \cdot m^2}{s^3} \quad q_3 := \frac{T_3 - T_4}{R_3} = 273.634\ \frac{kg \cdot m^2}{s^3}$$

As you can see the following only reduced the heat transfer of the first section. If we don't know how the temperature changes due to the added insulation it will be hard to calculate the rest of the system's heat transfer. To aid the analysis a simulation will be performed and results analyzed from there.





Above two cases are presented

1. Worst case scenario, were the thruster gets as hot as the propellant that is being decomposed
2. A much more mild scenario, the temperature at the intersection between the thruster and wall - which was picked rather arbitrarily, a fact needed to be admitted - was also chosen.

In both of these scenarios the following fact can be appreciated rather candidlly, the bulk of the heat is retained by the insulation. Leading the mind to ask the question:

Is it resonable to assume that the insulation will bring the temperature down to a value of about $186\ °C$?

Lets look at the math!

$$T_1 = \left(1.273 \cdot 10^3\right)\ K \quad T_2 := 186\ °C$$

$$q_1 := \frac{\left(T_1 - T_2\right)}{R_1} = 5.691\ \frac{kg \cdot m^2}{s^3}$$

These seem like a completely valid number. It is quite different from the high values that these calculations had presented before, but now the second question arises can the stainless steel dissipate the heat? For that we will do the same as above.

$$T_3 := 130\ °C \qquad T_4 := 80\ °C$$

$$q_2 := \frac{T_2 - T_3}{R_2} = 34.344\ \frac{kg \cdot m^2}{s^3} \qquad q_3 := \frac{T_3 - T_4}{R_3} = 291.1\ \frac{kg \cdot m^2}{s^3}$$

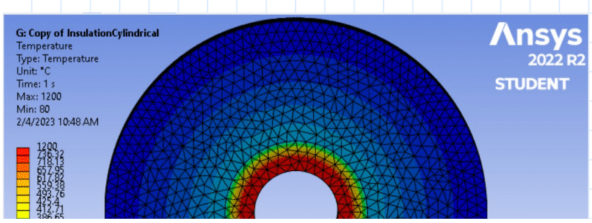
As can be seen we are getting to something more reasonable in the area of heat transfer rate, but these values are still to high for the assumption that it is safe to be made. Another millimeter should be added, to see what effect that would have on the system as a whole:

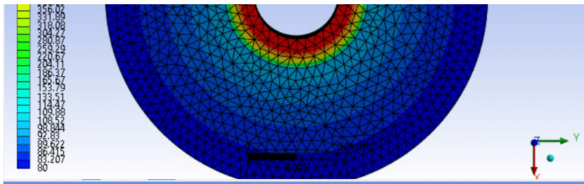
$$r_1 := 4.3\ mm \quad r_2 := 7.3\ mm \quad r_3 := 17.55\ mm \quad r_4 := 19.55\ mm$$

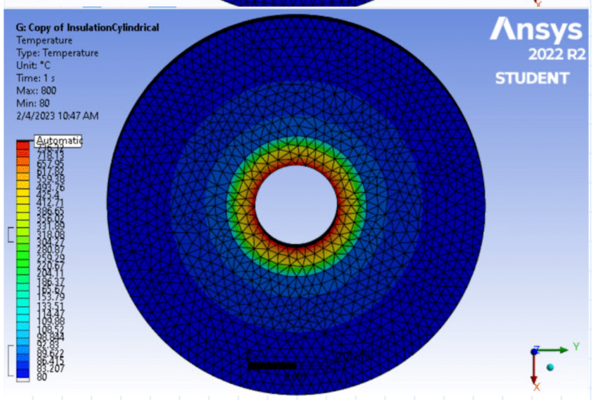
$$R_1 := \frac{\ln\left(\frac{r_2}{r_1}\right)}{k_{Mhp} \cdot 2 \cdot \pi \cdot L} = 198.198\ \frac{s^3 \cdot K}{kg \cdot m^2} \qquad R_2 := \frac{\ln\left(\frac{r_3}{r_2}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 1.396\ \frac{s^3 \cdot K}{kg \cdot m^2}$$

$$R_3 := \frac{\ln\left(\frac{r_4}{r_3}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 0.172\ \frac{s^3 \cdot K}{kg \cdot m^2}$$

Simulations were run with these dimensions and the results are shown below







Above it can be seen that by the end of the insulation at the worst case a temperature of about $165\ °C$ and the not-so bad case we see a temperature of about $133.51\ °C$.

$T_1 = \left(1.273 \cdot 10^3\right)\ K$ $\quad T_2 := 133.51\ °C$ $\quad T_3 := 105\ °C$ $\quad T_4 := 80\ °C$

$$q_1 := \frac{\left(T_1 - T_2\right)}{R_1} = 4.372\ \frac{kg \cdot m^2}{s^3} \quad q_2 := \frac{T_2 - T_3}{R_2} = 20.422\ \frac{kg \cdot m^2}{s^3} \quad q_3 := \frac{T_3 - T_4}{R_3} = 145.55\ \frac{kg \cdot m^2}{s^3}$$

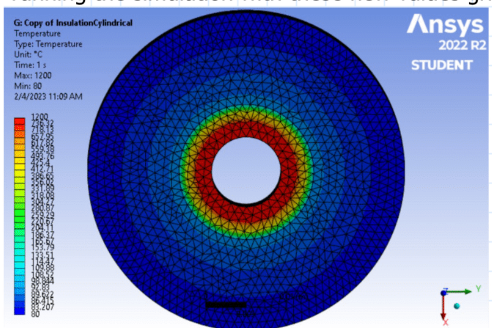
As can be once again appreciated the heat transfer rate is reduced significantly, but still a bit high. The saving grace here is that the temperature for self ignition of AFM is $140\ °C$ and in that case we are seeing something much more safe for the system. If we were to add one more mm

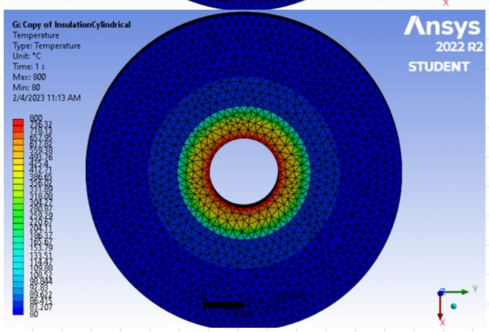
$r_1 := 4.3\ mm$ $\quad r_2 := 8.3\ mm$ $\quad r_3 := 17.55\ mm$ $\quad r_4 := 19.55\ mm$

$$R_1 := \frac{\ln\left(\frac{r_2}{r_1}\right)}{k_{Mhp} \cdot 2 \cdot \pi \cdot L} = 246.275\ \frac{s^3 \cdot K}{kg \cdot m^2} \qquad R_2 := \frac{\ln\left(\frac{r_3}{r_2}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 1.192\ \frac{s^3 \cdot K}{kg \cdot m^2}$$

$$R_3 := \frac{\ln\left(\frac{r_4}{r_3}\right)}{k_{Ti} \cdot 2 \cdot \pi \cdot L} = 0.172\ \frac{s^3 \cdot K}{kg \cdot m^2}$$

running the simulation with these new values gives:





As can be seen now the temperature being present in the end of the insulation is closer to $109\ °C$. By this point the reader should be acquainted with the game that is being played here:

$$T_1 = \left(1.273 \cdot 10^3\right)\ K \qquad T_2 := 109\ °C \qquad T_3 := 95\ °C \qquad T_4 := 80\ °C$$

$$q_1 := \frac{\left(T_1 - T_2\right)}{R_1} = 3.618\ \frac{kg \cdot m^2}{s^3} \qquad q_2 := \frac{T_2 - T_3}{R_2} = 11.747\ \frac{kg \cdot m^2}{s^3} \qquad q_3 := \frac{T_3 - T_4}{R_3} = 87.33\ \frac{kg \cdot m^2}{s^3}$$

Again these are much better values we are seeing and at this point it would be up to physical testing to see if the values calculated and simulated here can be validated. Concluding, an insulation of approximately 3-4mm would be necessary to keep the AFM at safe temperatures from the heat produced by the thruster.

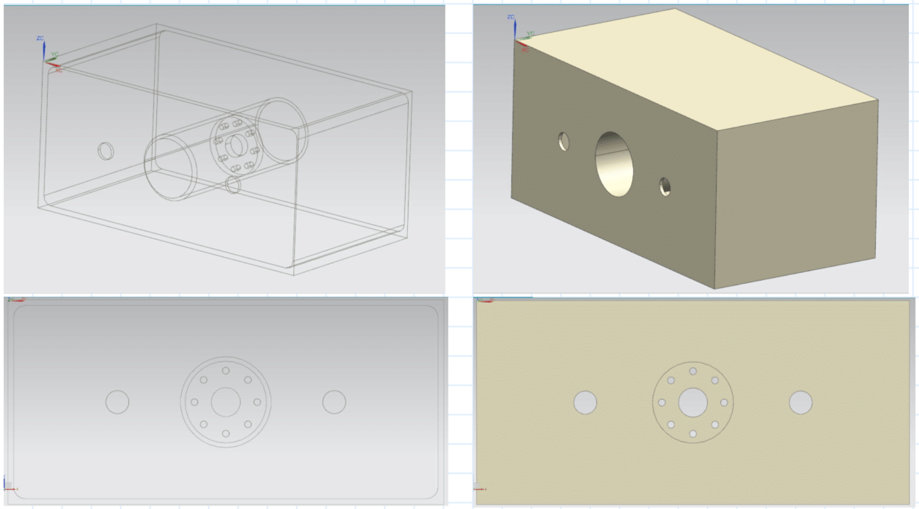
Radiation & Layers After Layers of Issues

As we had seen with the heat transfer from the environment to the module the highest heat transferred to the module is not from the albedo, sun, or IR radiation but from the heat emitted by the decomposition of AF-M and the heat input by the heater. What then should be considered for the next section of the analysis?

1. Only the heat produced inside the module will be considered for this section
2. Following in the steps of the first point, the area considered will be that which

encapsulates the thruster.

At this point a picture would be best to explain - some would say its worth a thousand words, but I assure you that after you see the picture more questions arise! - please see it below:



So what, why is this important if it had been shown before? Well because this empty hole is where the thruster will rest and be fired from, and by that fact the following can be extrapolated: Heat transfer by radiation will occurr here! Then the following equation can describe the heat at the cylindrical wall

$$T_{hot} = \left(\frac{Q_{in} + Q_{decomp}}{\varepsilon_s \cdot \sigma \cdot A_{innerWall}}\right)^{\frac{1}{4}}$$

The only thing to do is find the area of the inner wall, which is done in a document called ModuleAreaCalculations. Emissivity of inconel is $\varepsilon_{in} := 0.02$ gotten from https://www.ncbi.nlm.nih.gov/pmc/articles/PMC8839496/#:~:text=According%20to%20previous%20research%2C%20the,%CE%B5%20%3D%200.2%20%5B18%5D.

$A_{innerWall} := 0.007\ m^2$

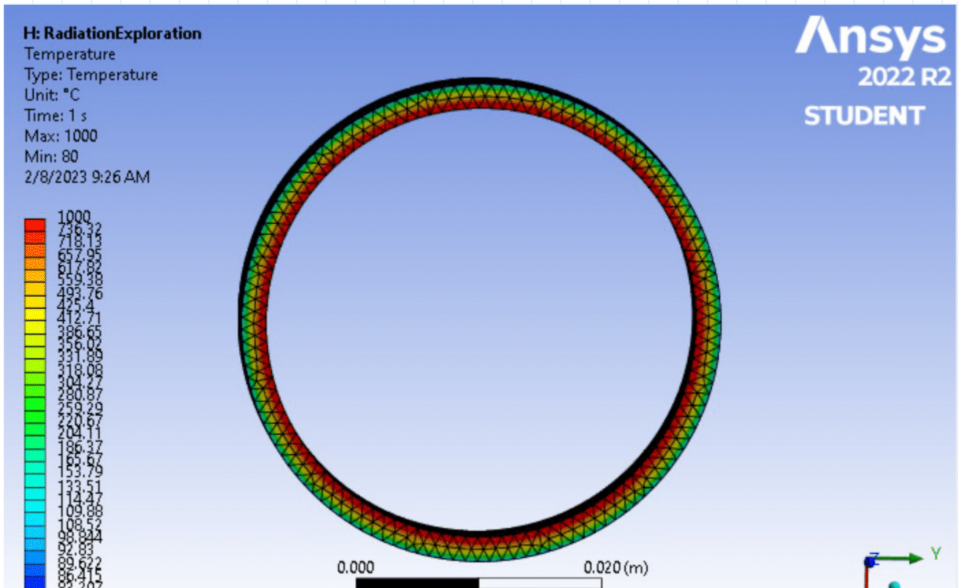
$$T_{innerWall} := \left(\frac{Q_{in} + Q_{decomp}}{\varepsilon_s \cdot \sigma \cdot A_{innerWall}} - (-22\ °C)^4\right)^{\frac{1}{4}} = \left(1.093 \cdot 10^3\right)\ K$$

$$T_{innerWall} = 820.075\ °C$$

As can be seen from the result of the above equation, what can be expected is nothing more and nothing less than what the industry might call a hot wall. If nothing was done what would result? As one can expect the AFM will probably reach the ignition temperature, but just in case the reader needs proof below are some calculations and a simulation that backs up the point:

$$R_3 = 0.172 \; \frac{s^3 \cdot K}{kg \cdot m^2}$$

$$Q := \frac{T_{innerWall} - T_4}{R_3} = \left(4.309 \cdot 10^3\right) \; \frac{kg \cdot m^2}{s^3}$$



The reader might look at the above simulation and think to themselves:

"Well the simulation shows that from 992 $°C$ the structure can reach the 80 $°C$, meaning that it should be possible in the pyshical system."

There is a couple of points that should be pointed out if the above logic is to be trusted:

1. The simulation has to work with the boundaries the person creating the simulation has set. ANSYS doesn't change the parameters the user has set once the simulation is running.
2. As can be seen in the simulation, the red area extends to about the middle of the structure. This means that up to the middle of the structure the temperature will be around 700 $°C$ and from there it reduces rapidly.
3. The same thing can be seen at the end of the structure where temperature goes from green to the blue. This means that it goes from about 200 $°C$ to 80 $°C$.

With all these points hopefully the reader is convinced that the probability of the AF-M remaining safe under these conditions is not only unlikely but erroneous to suspect.

Here is where the MLI comes in.

Reading from the book *Introduction to Spacecraft Thermal Design* the MLI equations for heat transfer can be calculated as follows:

$$Q = \varepsilon_{star} A \sigma \left(T_i^{\,4} - T_o^{\,4}\right)$$

Where ε_{star} is defined as:

$$\varepsilon_{star} = \frac{1}{(N+1)\left(\frac{1}{\varepsilon_A} + \frac{1}{\varepsilon_B} - 1\right)}$$

if the MLI looks as follows:

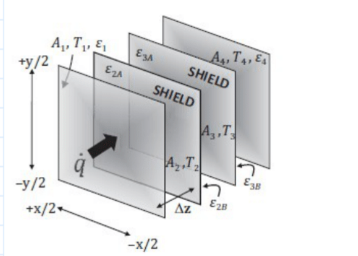


Figure 3.28 Radiation shield between two parallel planes

But if the equation above is to be simpilifed by making the emissivity of the 'A' layer and the 'B' layer the same, then the equation becomes:

$$\varepsilon_{star} = \frac{1}{(N+1)\left(\frac{2}{\varepsilon_A} - 1\right)}$$

From here we can then calculate heat transfer by knowing the emissivity of the surface and the number of layers. But we are faced with one more problem, this one a keystone to what we are discussing here; what material can withstand the temperature that are being approximated here? For that a quick google search demonstrates the following:

Melting Temperatures of Common Metals

These are the melting temperatures of common metal types:

- Aluminum: 660°C (1220°F)
- Brass: 930°C (1710°F)
- Aluminum Bronze*: 1027-1038°C (1881-1900°F)
- Chromium: 1860°C (3380°F)
- Copper: 1084°C (1983°F)
- Gold: 1063°C (1945°F)
- Inconel*: 1390-1425°C (2540-2600°F)
- Cast Iron: 1204°C (2200°F)
- Lead: 328°C (622°F)
- Molybdenum: 2620°C (4748°F)
- Nickel: 1453°C (2647°F)
- Platinum: 1770°C (3218°F)
- Silver: 961°C (1762°F)
- Carbon Steel*: 1425-1540°C (2597-2800°F)
- Stainless Steel*: 1375 – 1530°C (2500-2785°F)
- Titanium: 1670°C (3038°F)
- Tungsten: 3400°C (6152°F)
- Zinc: 420°C (787°F)

From what reaserch the author has seen, gold is used for these high temperature cases due to the emissivity value associated with it and its weight.

For these materials the following emissivity values are listed in the book *Fundamentals Heat & Mass Transfer*

1. $\varepsilon_{Au} := 0.06$

Starting at the assumption of four layers $N := 4$ and a $T_o := 85\ °C$ as it should be assumed that not a lot of heat can be dissapated through a wall with the calculated resistance of the one dealt with here

$$\varepsilon_{star4} := \frac{1}{(N+1)\left(\frac{2}{\varepsilon_{Au}} - 1\right)} = 0.006$$

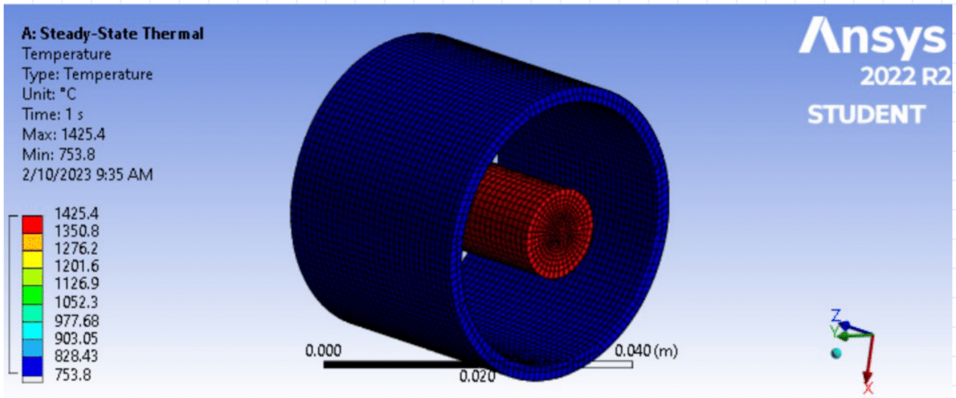
$$\boxed{N} := 3$$

$$\varepsilon_{star3} := \frac{1}{(N+1)\left(\frac{2}{\varepsilon_{Au}}-1\right)} = 0.008$$

$$Q_{MLI} := \varepsilon_{star4} \cdot A_{innerWall} \cdot \sigma \cdot \left(T_{innerWall}^{\ 4} - T_o^{\ 4}\right) = 3.467\ W$$

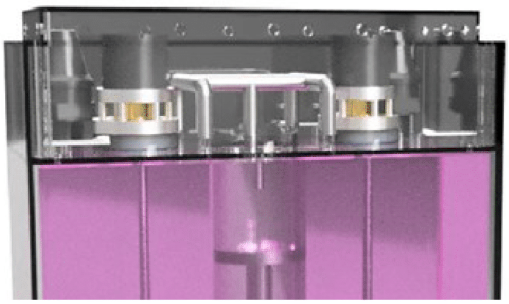
$$\boxed{Q_{MLI}} := \varepsilon_{star3} \cdot A_{innerWall} \cdot \sigma \cdot \left(T_{innerWall}^{\ 4} - T_o^{\ 4}\right) = 4.333\ W$$

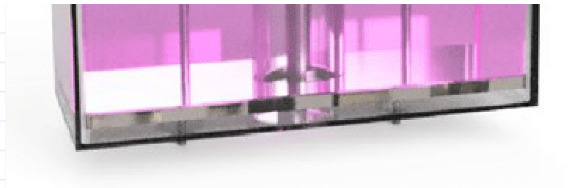
This value seems reasonable, but there has too be more of a backing to this value. Simulations for radiation is something that is foreign to the author but lets see what can be done. First is an analysis of the radiation that is expected inside the module without any type of insulation



The above simulation shows radiation from the thruster to the wall, as can be seen the wall is at a temperature of $T_{simulation} := 753.8\ °C$ - this is about $34\ °C$ difference from the calculations that were done by hand. The ultimate difference between these two methods will then be physical results, with the only difficulty being that these results are still not available. With the small ambiguity present, the worst case scenario choice will then be explored; simulation numbers for radiation are from here on taken into account.

Down the pipe/Through the pipe





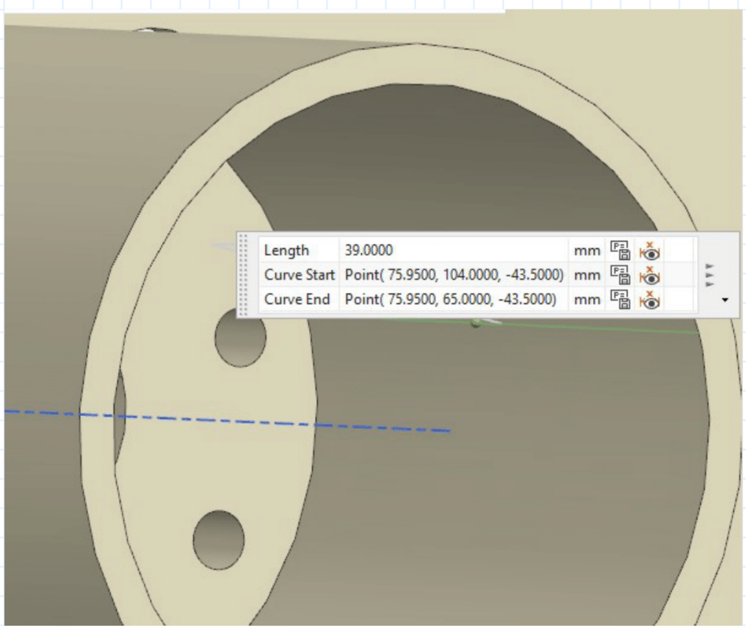
When analyzing the pipe that is connecting the thruster to the back of the module it is important (I belive) to consider two things:

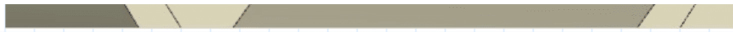
1. A pipe with a small cross section area will work as a thermal stand, meaning that by conduction not a lot of heat will be transferred.
2. Because there is fluid running throught the pipe some respect should be given to the fact that the fluid might reach critical conditions inside the pipe.

To conduct this analysis we first need to know the area of the pipe. For this we know the following:

$$d_{pipe} := \frac{1}{16} \; in \qquad\qquad k_{ss} :=$$

$$A_{pipe} := \pi \cdot \left(\frac{d_{pipe}}{2}\right)^2 = \left(1.979 \cdot 10^{-6}\right) \; m^2 \qquad\qquad L_{pipe} := 39 \; mm$$





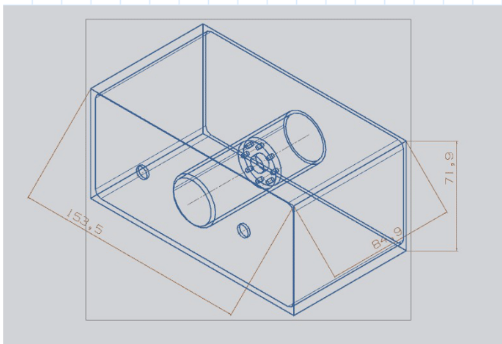
About this will be the length of the pipe from the thruster to the wall. This will give us a good idea of where the temperature will be at in this point of the module and if the propellant is in any type of danger of self igniting.

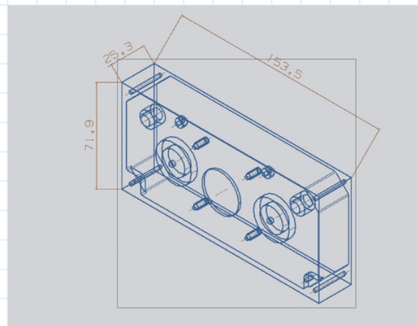
$$R :=$$

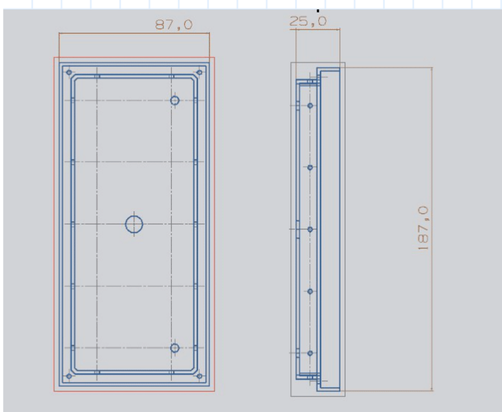
Everything discussed in the above is either discussed within the document or has been discussed in the thesis which precedes the appendix.

Area Calculations:

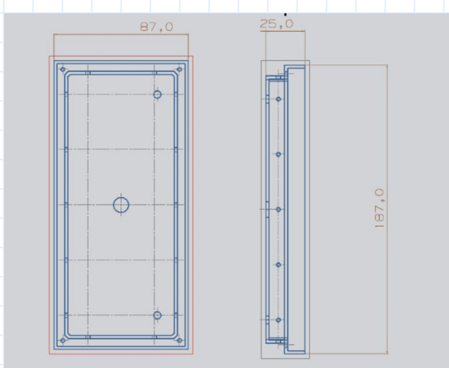
Calculating the area of the module.







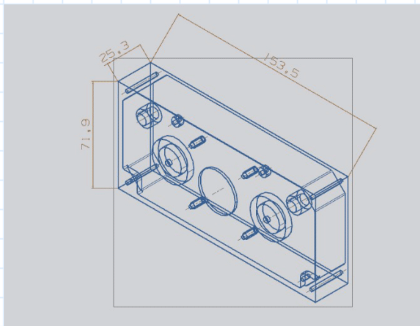
We will start with the smallest shape and move to the next two:



$A_{top} := 187\ mm \cdot 87\ mm = 0.016\ m^2$

$A_{side} := 187\ mm \cdot 25\ mm = 0.005\ m^2$

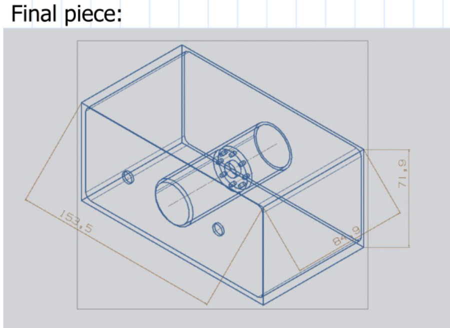
Second smallest shape is up next:



$A_{LongSide} := 153.5\ mm \cdot 25.3\ mm = 0.004\ m^2$

$A_{shortSide} := 71.9\ mm \cdot 25.3\ mm = 0.002\ m^2$

Final piece:



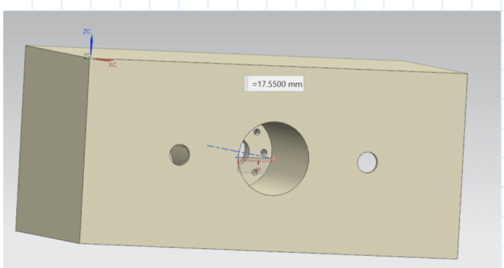
$A_{LongTopSide} := 153.5\ mm \cdot 84.9\ mm = 0.013\ m^2$

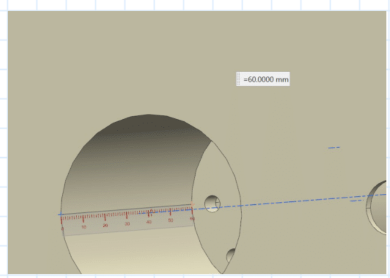
$A_{Side} := 84.9\ mm \cdot 71.9\ mm = 0.006\ m^2$

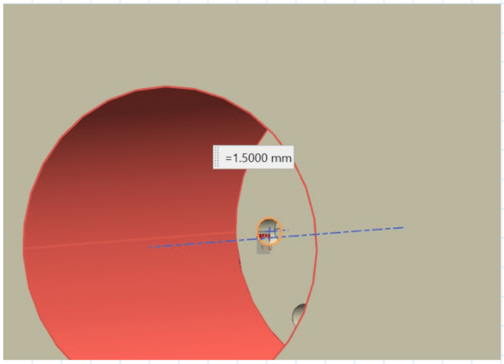
$A_{FrontSide} := 153.5\ mm \cdot 71.9\ mm = 0.011\ m^2$

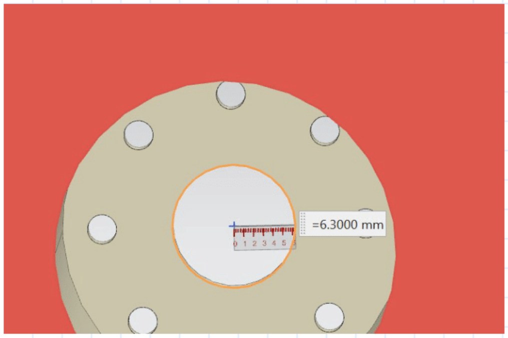
$$A_{tot} := 2 \cdot A_{LongTopSide} + 2 \cdot A_{Side} + 2 \cdot A_{FrontSide} = 0.06\ m^2$$

Now lets calculate the area of the first inner tube, see pictures below:





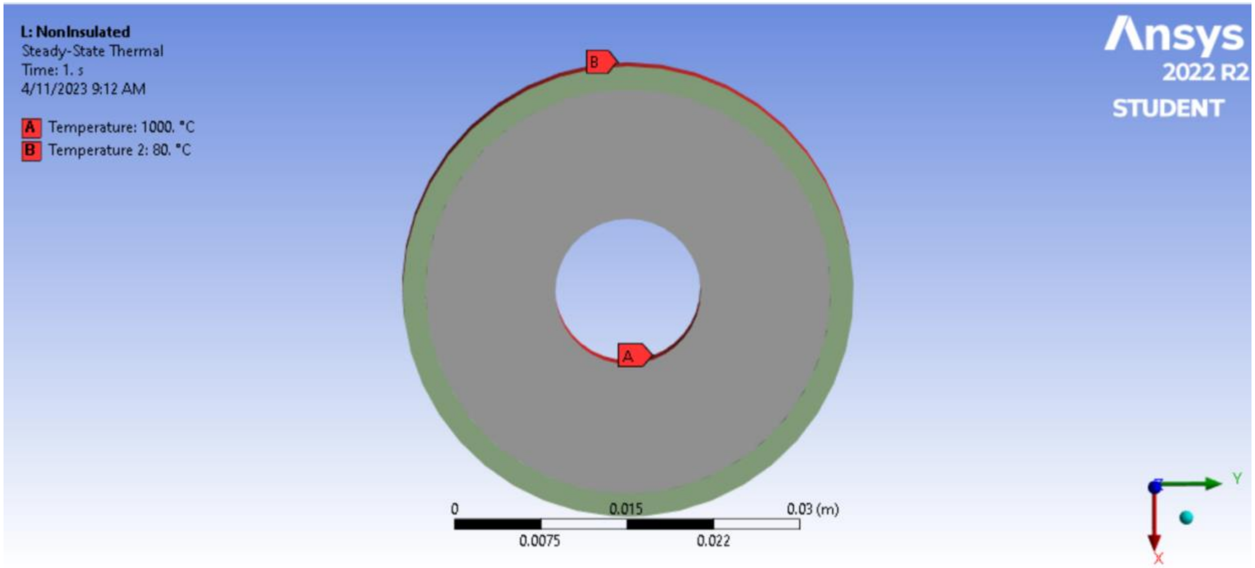


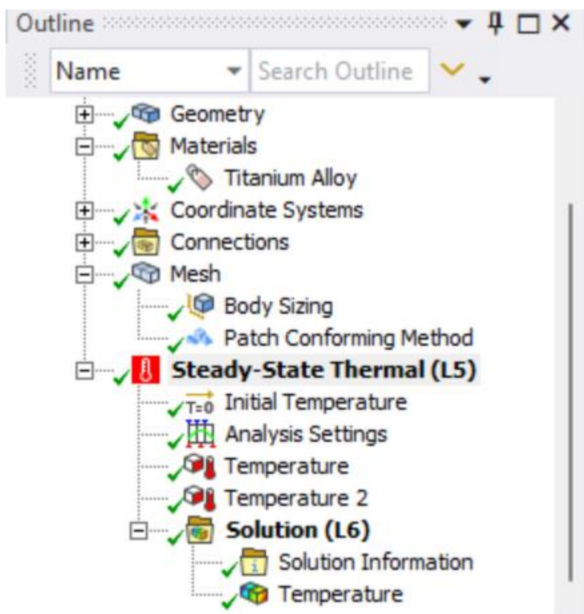


$r := 17.55\ mm \qquad h := 60\ mm \qquad r_{small} := 1.5\ mm \qquad r_{bigger} := 6.3\ mm$

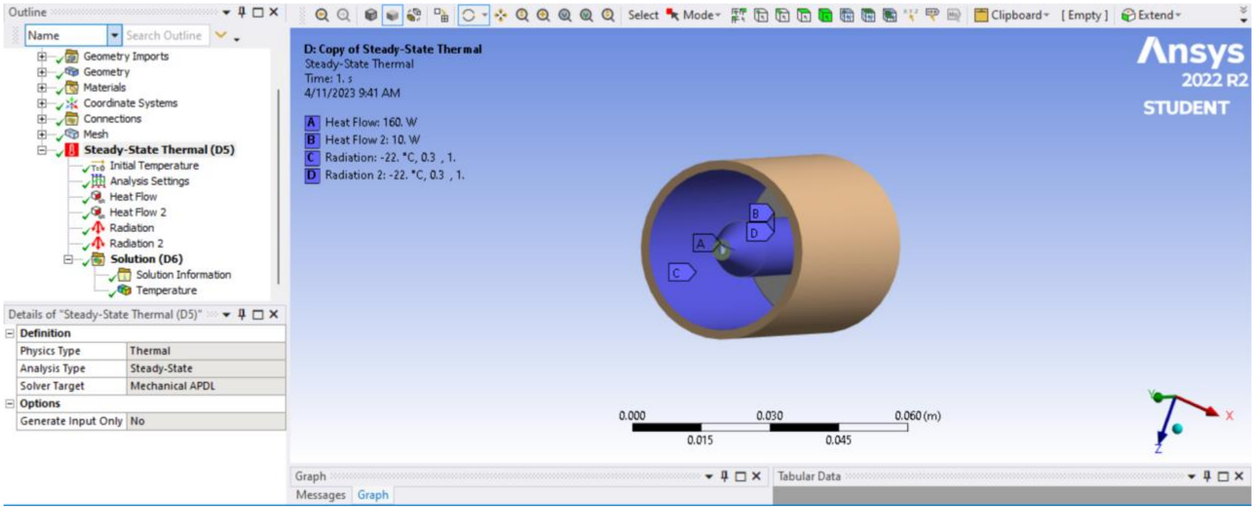
$$A_{inside} := 2 \cdot \pi \cdot r \cdot h + \left(\pi \cdot r^2 - \left(\pi \cdot r_{small}{}^2 + \pi \cdot r_{bigger}{}^2\right)\right) = 0.007\ m^2$$

ANSYS SIMULATIONS SETUP:

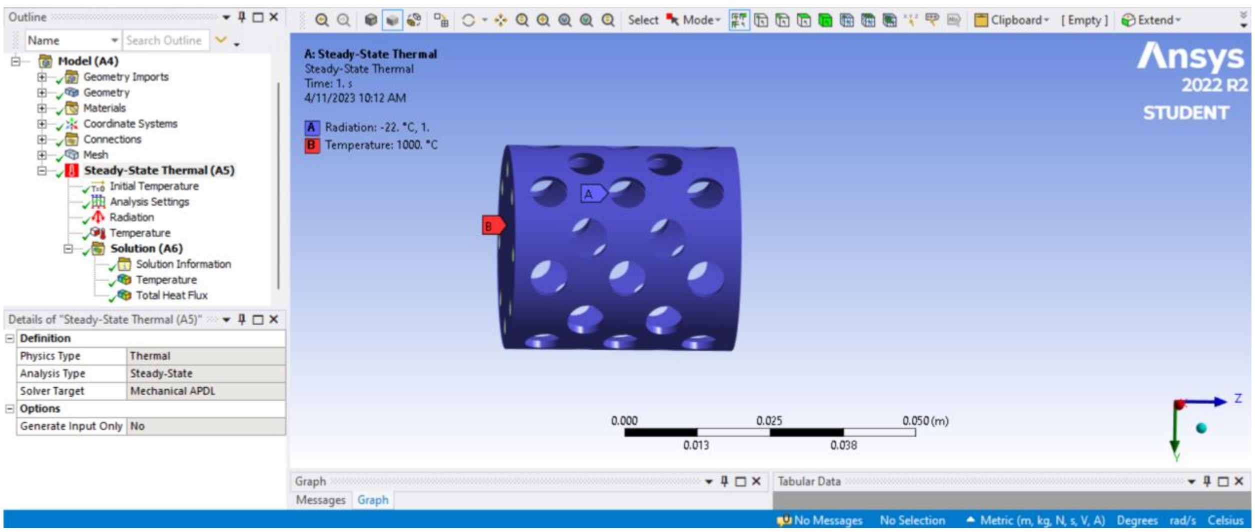


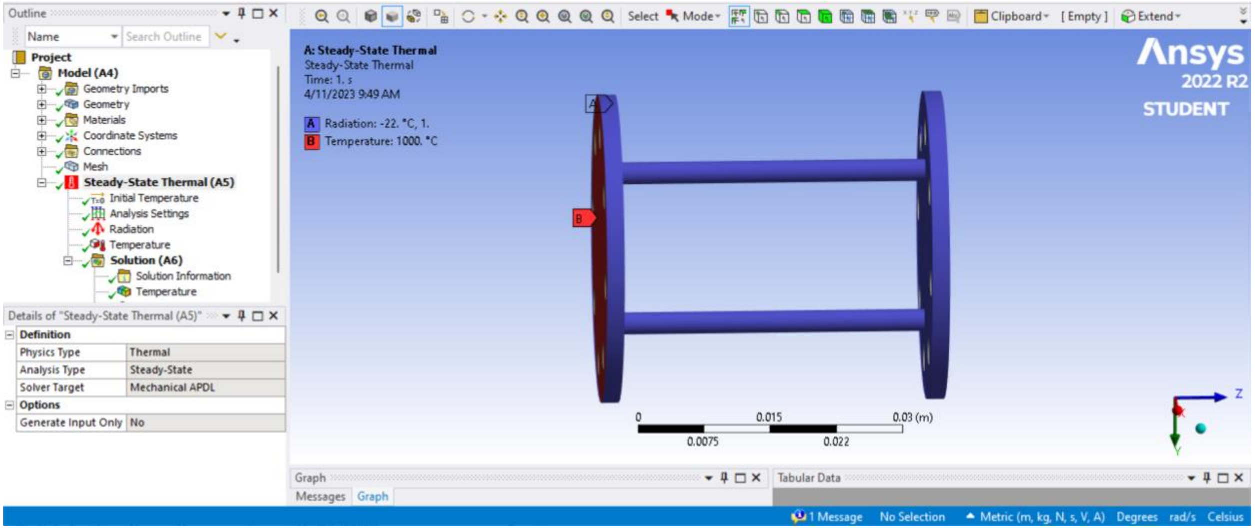


The simulations for the circular wall were all done in a similar fashion as this one shown above. The only difference in insulated simulations is a third body was added but all the inputs and solution information was about the same as seen above.



The radiation simulation inputs are shown above.




Outline
Name
Search Outline
Project
Model (A4)
Geometry Imports
Geometry
Materials
Coordinate Systems
Connections
Mesh
Steady-State Thermal (A5)
Initial Temperature
Analysis Settings
Radiation
Temperature
Solution (A6)
Solution Information
Temperature
Details of "Steady-State Thermal (A5)"
Definition
Physics Type Thermal
Analysis Type Steady-State
Solver Target Mechanical APDL
Options
Generate Input Only No
A: Steady-State Thermal
Steady-State Thermal
Time: 1. s
4/11/2023 9:49 AM
A Radiation: -22. °C, 1.
B Temperature: 1000. °C
Ansys
2022 R2
STUDENT
0
0.015
0.03 (m)
0.0075
0.022
Graph
Tabular Data
Messages
Graph
1 Message
No Selection
Metric (m, kg, N, s, V, A)
Degrees
rad/s
Celsius

Vita

Javier Madrid is very much a person who believes that he is.