Thermal Design and Analysis Methodologies
Applied to the DAUNTLESS Bus and GHGSat-C Microsatellite

by

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A thesis submitted in conformity with the requirements
for the degree of Master of Applied Science
Institute for Aerospace Studies
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Abstract
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Throughout a mission’s preliminary design through to final acceptance, various thermal analysis and control techniques are implemented to verify feasibility through worst case hot and cold conditions. A satellite developed using the DAUNTLESS bus, the latest platform developed at SFL, faced many thermal challenges due to the large bus with an emphasis on methodologies to reduce risk. This resulted in a detailed thermal model leading up to the launch of the spacecraft, capturing details around the large antenna dish and the internal propulsion tank. GHGSat-C is a greenhouse gas monitoring satellite with high-resolution IR imaging capabilities. The satellite is a successor to the pre-existing GHGSat-D, which demonstrated the mission and its future constellation. The satellite features updates in almost every subsystem and introduces an optical downlink which drives aspects of the design. These topics are expanded on further in this thesis and any major milestones and results are presented accordingly.
Acknowledgments

Dr. Robert E. Zee, thank you for offering me the opportunity to be part of the best Canadian space team. SFL is a special place where we are all free to celebrate our passion and come together for developing and launching exciting space missions. You are somehow simultaneously firm and understanding towards students like myself and I will always appreciate your interest in providing me the resources I needed to explore my abilities.

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To my fellow classmates and roommates, thank you for making Toronto feel like another home and supporting me when I was down. I wish for a future where our paths will cross again. For my friends from Montreal, you have no idea how much it meant to me to have your support when you came to visit for the weekends. We had so much fun and I know that we will always be there for each other when it truly counts.

To my wonderful Valentina, you simply bring out the best in me. We managed to constantly encourage and support each other, especially since moving away. I look forwards to all the wondering things our future has to offer together.

For my mother and father, I cannot express how my life is rich because of you two. You supported me for 26 years and I could not have asked for a more loving family. You made sure everything was perfect, even for the latest chapter of my life, and I will cherish those moments forever. Unfortunately, it was short lived as my father was diagnosed with cancer. We did everything we could to try and convince ourselves things would be okay. I visited most weekends and our moments together felt timeless, until it was suddenly taken from us. I will never get the chance to share my life with my father again but I am so grateful our last time together was charming and joyful. He showed me the world and I promise to do what I can to leave this world in a better place than I found it too.
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<th>Definition</th>
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<tr>
<td>ACS</td>
<td>Attitude Control System</td>
</tr>
<tr>
<td>ADCS</td>
<td>Attitude Determination and Control System</td>
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<tr>
<td>AU</td>
<td>Astronomical Unit</td>
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<td>BC</td>
<td>Boundary Condition</td>
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<tr>
<td>BIM</td>
<td>Battery Interface Module</td>
</tr>
<tr>
<td>C&amp;DH</td>
<td>Command and Data Handling</td>
</tr>
<tr>
<td>CAD</td>
<td>Computer-Aided Design</td>
</tr>
<tr>
<td>CFC</td>
<td>ChloroFluorocarbon</td>
</tr>
<tr>
<td>DAUNTLESS</td>
<td>Daringly Uncommon Technical Leadership in Smaller Satellites</td>
</tr>
<tr>
<td>ECI</td>
<td>Earth-Centered Inertial</td>
</tr>
<tr>
<td>EM</td>
<td>Electromagnetic</td>
</tr>
<tr>
<td>EMI</td>
<td>Electromagnetic Interference</td>
</tr>
<tr>
<td>EOL</td>
<td>End of Life</td>
</tr>
<tr>
<td>FEA</td>
<td>Finite Element Analysis</td>
</tr>
<tr>
<td>FEM</td>
<td>Finite Element Method</td>
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<tr>
<td>GEO</td>
<td>Geostationary Orbit</td>
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<tr>
<td>GHGSat</td>
<td>Greenhouse Gas Satellite</td>
</tr>
<tr>
<td>GPS</td>
<td>Global Positioning System</td>
</tr>
<tr>
<td>HEO</td>
<td>Highly Elliptical Orbit</td>
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<tr>
<td>IR</td>
<td>Infrared</td>
</tr>
<tr>
<td>ISS</td>
<td>International Space Station</td>
</tr>
<tr>
<td>JGM</td>
<td>Joint Earth Gravity Model</td>
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<tr>
<td>LEO</td>
<td>Low Earth Orbit</td>
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<tr>
<td>LTAN</td>
<td>Local Time Ascension Node</td>
</tr>
<tr>
<td>LTDN</td>
<td>Local Time Descension Node</td>
</tr>
<tr>
<td>MLI</td>
<td>Multi-Layered Insulation</td>
</tr>
<tr>
<td>MPS</td>
<td>Modular Power System</td>
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<tr>
<td>OBC</td>
<td>On-Board Computer</td>
</tr>
<tr>
<td>PCB</td>
<td>Printed Circuit Board</td>
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<tr>
<td>PL</td>
<td>Payload</td>
</tr>
<tr>
<td>RAAN</td>
<td>Right Ascension of Ascending Node</td>
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<tr>
<td>SIB</td>
<td>Serial Interface Board</td>
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<tr>
<td>SFL</td>
<td>Space Flight Laboratory</td>
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<tr>
<td>SORCE</td>
<td>Solar Radiation and Climate Experiment</td>
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<tr>
<td>SSO</td>
<td>Sun-Synchronous Orbit</td>
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<tr>
<td>STK</td>
<td>Systems Tool Kit</td>
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<tr>
<td>TLE</td>
<td>Two-Line Element Set</td>
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<tr>
<td>TVAC</td>
<td>Thermal Vacuum</td>
</tr>
<tr>
<td>UHF</td>
<td>Ultra High Frequency</td>
</tr>
<tr>
<td>VHF</td>
<td>Very High Frequency</td>
</tr>
<tr>
<td>WCH</td>
<td>Worst Case Hot</td>
</tr>
<tr>
<td>WCC</td>
<td>Worst Case Cold</td>
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Chapter 1

Introduction

The satellite market is rising more than ever thanks to the decreasing size of electronics and various instrumentation that can be made to work in orbit. This shift in technology has allowed for new avenues to be explored, such as the field of small satellite technology. These satellites are appealing due to their relatively inexpensive and short development cycle, as seen with the nanosatellite market producing plenty of cubesats, a similar truth is apparent for micro-satellites, which range from 10 to 100kg. With satellites at these scales, the microspace philosophy is the working mindset that produces and iterate on these satellites very successfully [1].

The technical challenges begin when designing these instruments to function in the space environment. Any spacecraft orbiting in low Earth orbit (LEO) experiences no atmospheric pressure, larger dosages of radiation and extreme temperature fluctuations. Temperature fluctuations are amplified in a vacuum and with intermitted direct sunlight. The vacuum limits the transfer of heat through the structure and less effective thermal radiation, and due to unfiltered sun being the primary heat source, only when in view. In LEO this ultimately results in large thermal cycles as a satellite orbits through direct sunlight and being eclipsed by the earth. This is controlled by the design of the spacecraft by deliberately taking advantage of the material properties of the structure to isolate the internal bus from the large external temperature fluctuations, but simultaneously ensure that any electronics are also well connected to deposit their heat loads. The design process if conceptualized based on simplified calculations and models that can determine average temperatures, and later developed into fully detailed models to ensure if a given design will yield the desired outcome.

Thermal control is a critical aspect of spacecraft design that can often drive the design and the system. This is especially true when avoiding adding additional risks with complicated systems, thus leaving the number of parts in the satellite as low as possible. This is possible because there is inherit conservatism and large margins in most decisions and design aspects, which carries over to the thermal subsystem and ultimately avoids potential problems altogether.
1.1 Scope

This thesis covers the thermal design, analysis, integration and validation methodologies for various microsatellites. The topics are explored through example from existing spacecraft that went through development at Space Flight Laboratory (SFL) and form into a discussion about procedures and reasoning for concluding a design. The subjects presented are primarily summaries of the underlying physics, with an emphasis on what are the relevant connections and influences from each subsystem into the system.

1.2 Thermal Environment

The environment a spacecraft is exposed to once outside of the protection of Earth’s atmosphere can vary substantially. Satellites needs to meet many requirements to successfully function once in orbit, which exist in different forms. Each of these environmental impacts can change the way a thermal problem is approached and solved.

The exchanges of thermal energy with the satellite system are in the form thermal radiation. The primary energy source is from solar irradiance directly emitted by the sun in predominantly the visible spectrum, the reflected sunlight from the Earth and Moon, along with the primarily infrared radiation (IR) exchanged by the satellite, local celestial bodies, and deep space [2]. Modelling these interactions takes a practical approach described in Section 1.5 which focuses on the interactions with the sun and Earth. Due to the sun being the dominant energy source of the spacecraft, transitioning from eclipse to direct sunlight, the exposed surfaces of the spacecraft can experience huge temperature swings, on the order of ±100°C [3]. The specifics of these sources are further explored in Section 1.2.4. The heat transfer mechanisms are driven by temperature differences where within a satellite, conduction is the primary mode of heat transfer where physical connections exchange energy linearly, and radiation is a secondary mechanism where surfaces in view of each other exchange energy exponentially. Depending on the possible temperature difference within the spacecraft and the conductivity of materials, internal radiation can be significant and otherwise a primary source of heat transfer for isolated components.
There are also structural loads due to the launch itself, which impose a very high acceleration in the form of static loads and random vibrations, along with mechanical shock. When it comes to the design of the system, this is where the mission specific balance with thermal control often overlap, as the most structurally rigid bodies tend to be the most thermally conductive [4]. This also benefits the power subsystem as well by taking advantage of the well coupled bus and use it as a heat sink during high power modes. These methodologies overlap with the microspace philosophy as the system must work as passively as possible to avoid adding risk with any active systems [1].

Once the satellite is in orbit, it is effectively in vacuum due to the lack of an atmospheric pressure which can cause an outgassing effect on materials transitioning from a high to low pressure environment. This is due to trapped gasses from processing the material on Earth, that start to escape the material once in space. This could potentially change material properties or impose an effect on other systems, especially any exposed electrical boards. The design approach is to avoid such materials which can limit certain design possibilities. For thermal control, this is an important factor as surface coatings effectively dictate the energy flow to and from the system.

1.2.1 Low Earth Orbit

Orbits around the Earth can vary in many ways. They can range at different altitudes where LEO is considered below 2000 km from the surface. For comparison, the Earth’s radius is 6371 km, with the International Space Station (ISS) averaging about 300 km while geostationary orbits (GEO) are close to 36,000 km [5]. The altitude of the satellite is important to consider for a few key factors beyond thermal control, such as the Van Allen radiation belt which ranges from 500 to 58,000 km [6] which can cause effects on the spacecraft, especially in the form of bit flips in the electronics [4]. Additionally, there is also the end of life (EOL) of the satellite that needs to be considered. Due to the relatively new international initiative to reduce space debris for the future, spacecraft must be designed to de-orbit within 25 years after EOL is reached [7]. These can all
factor into the design for a satellite in LEO, and consequently impact the thermal environment that is experienced.

1.2.2 Sun-Synchronous Orbits

The most common orbits used for LEO are sun-synchronous orbits (SSO), which are an elegant type of orbit that takes advantage of the geophysical properties of Earth to achieve a relatively constant orbit relative to the Sun. Normally, anything spinning will resist changing its plane of rotation, and remains gyroscopically stable unless an external force is acted upon the system. This is most obvious with the spin of the earth, and why our axial tilt relative to the Sun oscillates throughout the year. This oscillation can be measured as the solar declination and is what causes our seasons [8]. This can be seen in Figure 1 and the oscillations are further emphasized in Figure 2. Note the angle of the earth relative to the sun during the solstices and equinoxes.

![Figure 1: The phases and elliptic orbit of the earth throughout the year [8]](image1)

![Figure 2: Solar declination relative to Earth [9]](image2)
When a satellite is in orbit, the angular momentum vector naturally stays constant without any external torques, however due to the non-uniformity of the earth’s gravitational influence, the acceleration experienced by the satellite can vary, thus resulting in small external torques and changing its trajectory slightly. Due to the spin of the earth, and the fact that it is not a rigid body, the earth bulges out at the equators, which effectively increases the mass compared to the poles. The resultant force this imposes on an orbiting object can be seen in Figure 3 where the force vector due to gravity is off centered resulting in a torque on orbit. The gravitational field of Earth can be modelled by spherical harmonics where the bulge of the earth is described by the second harmonic $J_2$ term. The resulting effect on the system is experienced in the form of gyroscopic precession relative to the earth.

![Figure 3: Forces from the earth’s bulge](image)

![Figure 4: Inclination visualization](image)

The effect on the spacecraft is minor, and the effect on the orbit is more inconsequential the lower the inclination [5], however it is possible to purposely take advantage and induce the SSO. Inclination is defined from the equator to the orbital plane, as seen in Figure 4. Synchronizing with the sun throughout the year is a precise trajectory because the orbital plane of the satellite needs to do a $360^\circ$ rotation every year, which is only possible at a narrow range around $98^\circ$ [5]. It is a question of the accuracy of the launch vehicle or active re-adjustments for long-term missions.

1.2.3 Beta Angle

For thermal analysis, every orbital variable can be used to compose the Beta angle $\beta$ which is defined as the angle of the orbital plane relative to the sun [3]. It is easy to assume the Beta angle must be constant throughout the year because a SSO follows the sun through the year, however
the Beta angle also oscillate in phase with Earth’s declination. This mechanism is the result of three-dimensional interactions, with the need for a constant inclination, which due to Earth’s axial tilt cause the changing solar declination throughout the year. This can be visualized with Figure 5 where it shows Earth from the perspective of the sun through the different phase peaks of the year. The local time ascending node (LTAN) is one of many ways of defining an orbit, which means the local time of the satellite above the earth as it crosses the equator from the southern to northern hemisphere and vice-versa for the descending node (LTDN). For any SSO, this is a useful term because the local time will remain constant as long as the ideal orbit is achieved where the consequences of orbital drift are minimized [3].

Figure 5: Various orbits throughout the year from the sun's perspective with Beta angles

Thermally, any relevant LTAN can be defined from 6:00 which is almost always in sunlight and referred to as dawn-dusk, up to 12:00 which spends nearly the same amount of time in Earth’s eclipse as it does in sunlight, also known as noon-midnight. The beta angles are mirror on either side of the 12:00 LTAN, as seen in Figure 5 with 9:00 and 15:00. The overall reason why Beta
angles are significant to thermal analysis is due to the time spent in sunlight where the satellite is exposed longer for higher beta angles as the orbit transfer over the earth’s curvature [3].

The Beta angle can be calculated using the orbit’s LTAN, altitude, and the time of year. Figure 5 might imply that the Beta angles always oscillate exactly on solstices or equinoxes, however due to the nature of a specific LTAN, the actual peak could vary days or months from the solstice or equinox. For example, the 12:00 Beta angle oscillates from equinox to solstice exactly, while the 9:00 Beta angle oscillates from the exact winter solstice to a couple months before the summer solstice ($\beta = 45.5^\circ$). For thermal analysis, it is best to calculate the Beta angle throughout the year to determine the absolute maximum and minimum angle experienced by the satellite.

The first orbital parameter is to determine the inclination needed to achieve an SSO which considers the resulting torque from the bulge of the earth with a changing altitude. The inclination needed to maintain a SSO is expressed in Eq. (1) in terms of radius [11] where $r$ is the radius of the satellite’s orbit [km], $\omega$ is Earth’s orbital rate around the sun [$\text{rad s}^{-1}$], $\mu$ is Earth’s gravitational parameter [$\text{km}^3 \text{s}^{-2}$], and $J_2$ is the coefficient for the second zonal term [$\text{km}^5 \text{s}^{-2}$] that relates to the oblateness of the earth using the geopotential model (JGM-3) [12].

$$i = \cos^{-1} \left[ \frac{-2\omega \sqrt{\mu}}{J_2} r^{7/2} \right]$$

Following this, the right ascension of the ascending node (RAAN) ($\Omega$) is calculated as in Eq. (2) [13] where $\gamma$ is the angle from the Autumnal equinox (seen in Figure 1) to Earth’s position during the year and the LTAN is expressed as an angle where $0^\circ$ is at 12:00 and $90^\circ$ is at 18:00. The RAAN is used as a reference for the axial tilt of the earth, centered on the equinox in the fall. The vector that is normal to the orbit of the satellite is expressed in Eq. (3) where it is relative to the earth-centered inertial frame of reference (ECI) [13].

$$\Omega = \gamma + LTAN$$

$$\hat{n}_E = \begin{bmatrix} \sin(i) \sin(\Omega) \\ - \sin(i) \cos(\Omega) \\ \cos(i) \end{bmatrix}$$
The solar vector is the expression in which the sun is relative to the ECI frame of reference and can be determined using Eq. (4) [13] where $\lambda$ is the solar declination defined with Figure 2.

$$\hat{s}_E = \begin{bmatrix} \cos(\gamma) \\ \sin(\gamma) \cos(\lambda) \\ \sin(\gamma) \sin(\lambda) \end{bmatrix}$$

(4)

Finally, the Beta angle ($\beta$) can be calculated using inclination, declination and Autumnal angle and combined in forms seen in Eq. (5) [13]. The end result will only capture the angle of the orbital plane relative to the sun for the specific time or day the calculation is made. Note that this model assumes a circular Earth orbit, where for more accuracy, the Systems Tool Kit (STK) [14] using orbit determination is recommended. Calculating for the entire year for each orbit is best for determining the Beta angles closest and furthest from 0°. It is the driving boundary condition that will shape the entire thermal design and feasibility of a spacecraft.

$$\cos(90^\circ - \beta) = \hat{n} \cdot \hat{s}$$

(5)

$$\beta = \sin^{-1}(\cos(\lambda) \sin(i) \sin(\Omega) - \sin(\lambda) \cos(\gamma) \sin(i) \cos(\Omega) - \sin(\lambda) \sin(\gamma) \cos(i))$$


### 1.2.4 Thermal Sources

The sources of thermal energy to and from the spacecraft are most significant in the forms of solar irradiance, the reflected sunlight from Earth, and the Earth IR [3]. The solar irradiance is described as a flux, and fluctuates primarily due to the elliptical orbit of the Earth around the sun, as seen in Figure 1 where the aphelion and perihelion are shown. The total power from the sun varies from 1322 \(\frac{W}{m^2}\) during Earth’s aphelion, and 1414 \(\frac{W}{m^2}\) during Earth’s perihelion [3] and changes specifically according to the inverse square law. This is used in Eq. (6) where $S_o$ is the solar constant, $S_E$ is the solar flux at the current Earth distance, $R_o$ is distance at 1 Astronomical Unit (AU) and $R_E$ is the earth’s radius from the sun [m]. There are also variations from the 11-year solar magnetic activity cycle that the sun experiences, although the variations are less than 0.1% of the mean irradiance [15].

$$S_E = S_o \left(\frac{R_o}{R_E}\right)^2$$

(6)
The seasonal change does have an effect on orbiting satellites as well. One way is known as the albedo and effectively means the more snow or clouds cover the Earth, the more sunlight will reflect and act as an additional source of thermal loads to the spacecraft. Albedo is expressed as a percentage, and in reality, is different throughout the planet at any given time. Knowing which one to analyze with is a question of probability and how sensitive the spacecraft is to sunlight [3]. More information on how these values are selected in Section 1.3.2.

Albedo itself tends to have an inverse relationship with IR emitted from the object since an object reflecting more sunlight is absorbing less. In the case of Earth, this inverse relationship is due to the atmosphere causing a seasonal lag where more fresh ice forming increases the albedo of the planet, and thus reducing the amount of absorbed energy [16]. The delay is mostly tied to the northern hemisphere because of its larger land mass relative to the oceans. The land dictates the average temperature of Earth’s surface because it can absorb and dissipate heat on the order of days compared to the ocean which is on the order of months. The result of this is the hottest global average temperature peaks towards July and the coolest towards January. It is important to note that the global average temperature is highest around the aphelion and lowest near the perihelion, which is inversed to the variations in solar irradiance Earth is receiving. The local temperature is driven by the local seasons of each hemisphere where the amount of energy absorbed by the northern hemisphere drives the global average temperature [17] as seen in Figure 6 despite occurring near the aphelion. The amount of heat that is emitted as IR is a function of temperature which is further elaborated on in Section 1.3.2. With the earth being an average of 15°C [3], the average temperature of satellites can vary well beyond that, so the direction of energy flow will be relative from the coldest object to the hottest. How this is modelled is explained in Section 1.5.2.
1.3 Heat Transfer

Thermal energy is specifically microscopic kinetic energy within a material, which is known as heat. The kinetic energy that transfers through each atomic interaction and ultimately the material, is known as conduction which occurs within both solids and liquids, however liquids can also displace its material to transfer heat which is known as convection. This thermal energy also excites electrons which are emitted as electromagnetic (EM) radiation, where the frequency and intensity is a function of temperature. How well these interactions transfer energy and influences the resulting net energy transfer is what defines any thermal system.

1.3.1 Conduction

The rate of energy transferred is relative to the difference in temperature, and in the case of conduction, it is a linear relationship. The expression for conductive heat transferring [W] through a homogeneous material according to Fourier’s Law is seen in Eq. (7) [19] where $k$ is the thermal conductivity of a material [W/ km], $A$ is the cross-sectional area [m$^2$], $T$ is the temperature difference of two ends [K] and $x$ is the material thickness being considered [m]. Conductance itself can also be related as a thermal resistance $R$ [K/W], which is simply the inverse of conductance [W/K].

$$\dot{Q} = -kA \frac{dT}{dx} = \frac{\Delta T}{R} \quad (7)$$
A common source of thermal resistance is found internally in a material among the molecular bonds that make up the atomic structures, although it is not the only source. There can be also be thermal resistance from non-bonded connections such as when two materials are in physical contact. This contact resistance exists because every independent solid will not be in perfect contact with another solid as seen in Figure 7 where heat flow favors solid connections over the normally air-filled gaps. This is further emphasized for spacecraft design as these microscopic gaps effectively become vacuum pockets where extremely little heat transfer occurs and limiting the total transfer to the point contacts only.

![Figure 7: Distribution of heat flow through uneven surface contact][19]

Additionally, the interface is also greatly influenced by the pressure of the connection in question. A greater pressure will effectively compress the material enough on the molecular level, cumulatively deforming the material to create a larger contact area and improving the stiffness of the atomic interactions, increasing the efficiency of the transfer of kinetic energy [19]. This effect can be seen in Figure 8 where for all materials and surface finishes, the resistance decreases universally with an increase in pressure. Pressure can also increase due to thermal expansion because of the effects of internal kinetic energy increasing the volume, however keeping the same molecular bonds.
Internal and contact resistance can also be mixed together if a material has a non-homogenous grain structure, which often happens for composites since they are made of multiple materials and layered such that a structural property is achieved. The resulting effect on energy flowing through the system will be unevenly distributed heat paths biased towards the direction of the composite layup, which are commonly varied for each layer.

1.3.2 Radiation

Every atom above absolute zero emits light at varying wavelengths in the form of EM radiation. The radiation intensity depends on the temperature above absolute zero and is on the fourth order. The conversion to EM radiation is due to the microscopic kinetic energy exciting electrons into higher energy states. This phenomenon can be studied as blackbody radiation and can be categorized with respect to temperature and wavelength by looking at Figure 9. Notice how the sun is approximately 5777 K and emits predominantly in the visible spectrum, and how most objects on Earth are around 300 K and emit in the IR range. Black body radiation can be expressed as a power density function $j^* \left[ \frac{W}{m^2} \right]$ using the Stephan-Boltzmann law as seen in Eq. (8) [2] where $\sigma$ is the Stefan-Boltzmann constant for radiation $\left[ \frac{W}{m^2 \cdot K^4} \right]$ and $T$ is temperature [K]. The solar irradiance and Earth IR emitted are calculated this way.
The term blackbody is referring to an object being perfectly opaque and non-reflective, therefore absorbs and emits all EM radiation. It is used as a way of studying these phenomena fundamentally with uniformity [2]. After being emitted and interacting with another object, light can be reflected, absorbed, or pass through the material entirely, which changes how the object is observed and measured. Materials are measured and catalogued based on their thermo-optical properties. The fraction of the energy interactions and conservations when interacting with an object can be seen in Eq. (9) [2] where $\rho$ is reflectivity, $\tau$ is transmittance and $\alpha$ is absorptance.

$$\rho + \tau + \alpha = 1$$  \hspace{1cm} (9)

When discussing the energy absorbed by the sun [W], it is most commonly expressed as in Eq. (10) to determine the absorbed energy flow due to solar power [2] where $S$ is the solar irradiance $[\text{W/m}^2]$ derived from its black body power output and distance from Earth, $A$ is the area in direct sunlight $[\text{m}^2]$, and $\alpha$ is the absorptance factor in percentage.

$$\dot{Q} = \alpha SA$$  \hspace{1cm} (10)

Fundamentally, the expression for all thermal radiative heat transfer Eq. (11) is more representative since it includes a relative temperature difference because there is always an exchange of energy among objects in view of each other, and results in net power flowing in one direction [2]. Black bodies by definition have their absorptance and emittance equal to 100%, where $\varepsilon$ is the emissivity.

$$j^* = \sigma T^4$$  \hspace{1cm} (8)
of the surface as a percentage. When compared to the background temperature of space [2.7 K], it is often appropriate to exclude its temperature term due to its insignificance from the exchange. Similarly, the sun and the earth are such massive bodies the effect of thermal radiation from the spacecraft are negligible. This means the temperatures of celestial bodies can be treated as constants when solving the resulting temperatures of the spacecraft system.

\[
\dot{Q} = \varepsilon \sigma A (T_2^4 - T_1^4)
\]  

(11)

The names that denote absorptance and emittance may be misleading in the sense that both of these terms are representing the same physical property of emitting and absorbing together. The difference is the sensitivities were averaged for specific wavelengths when referring to the measurements. This is known as Kirchhoff’s law of thermal radiation as denoted in Eq. (12) [2] which states at the same wavelength [\(\lambda\)] of light, the amount of radiation an object can absorb and emit is equal within that spectrum. Note that practically, \(\alpha\) and \(\varepsilon\) are often used without specifying \(\lambda\), where the specific wavelengths are associated to \(\alpha\) is for visible light and \(\varepsilon\) for IR.

\[
\alpha_\lambda = \varepsilon_\lambda
\]

(12)

For the emissivity between two surfaces, it may be convenient to express an equivalent emissivity, as Eq. (10) and Eq. (11) assume one of the objects are a blackbody. The sun and moon themselves are almost perfect black bodies, however the earth and artificial satellites are not, which are known as graybodies. The effective emissivity must be considered between two arbitrary graybodies as seen in Eq. (13) [19] where \(\varepsilon_{eq}\) is the equivalent emissivity and \(\varepsilon_1, \varepsilon_2\) are the emissivities for each graybody.

\[
1/\varepsilon_{eq} = \left(\frac{1}{\varepsilon_1} + \frac{1}{\varepsilon_2} - 1\right)
\]

(13)

Realistically, objects where radiation is significant are in view of multiple surfaces and temperatures at once. This is defined by the surface’s view factor, which is a percentage of each area of a certain temperature and thermo-optical property. This is visualized in Figure 10 where each surface is at a uniform temperature and emissivity and an interference would decrease the view factor of the original interaction but add a new interaction with the object itself. In the context of LEO, the earth is a significant portion of the view factor of the satellite, being approximately 25% varying slightly with altitude with the rest being the background of space, the sun, moon and
other distant objects. Realistically the sun and moon are typically not calculated in LEO due to their small view factors but may be significant when taken beyond LEO [3].

![View factor diagram shown with and without interference](image)

**Figure 10:** View factor diagram shown with and without interference [2]

### 1.3.3 Heat Capacity

Any mass in the universe can store kinetic energy in the form of thermal capacitance, which is also known as thermal mass or heat capacity \([J/\text{K}]\). This is described in Eq. (14) [19] where \(m\) is mass [kg], \(c\) is specific heat \([J/(\text{kg.K})]\), typically for constant pressure with solids, \(Q\) is energy [J]. Capacitance can be described as the amount of energy needed to increase the temperature of an object by a degree. Thermal mass is only considered for modelling transient systems as capacitance itself does not add or remove heat to the system through a steady-state but instead stores energy as it flows through [19]. The influence capacitance has on a thermal system is that it introduces a thermal inertia where changes in temperature causes heat to flow at a higher rate, and if through an object with high capacity, the temperature change on the other end will take much longer to observe. All transient models will eventually approach a steady-state if the conditions are constant.

\[
C = mc = \frac{dq}{dT} \quad (14)
\]
1.4 Control Methods Overview

Most components that are within a microsatellite are designed to work around room temperature. There exist many practical ways of controlling the thermal energy within a spacecraft which can be separated into passive and active systems. Passive methods are desirable because they operate without inputting any power or electrical control system, thus reducing complexity and risk of failure [1]. Active systems however, use electrical energy and rely on feedback, which consumes power and introduces the possibility of a failure mode where the spacecraft can no longer function due to a critical component misbehaving. Failure modes are a result of a death mode existing within the system’s design which allow for the possibility of these components risking failure or otherwise going wrong. Active systems can make up for this shortcoming by offering a significantly larger range of temperature control [3] as there are powerful mechanisms for heat transfer. For microsatellites, it is desired to achieve thermal control with primarily passive methods, as the following describes.

1.4.1 Isolating Spacers

The difference in temperature from the external panels of the bus and the internals can vary significantly and is most often well beyond the temperature range of the internal components. The most common control method within a satellite bus is to deliberately choose the physical connection with each component and vary the materials and mounting types. This can be achieved by effectively isolating the components, such as printed circuit boards (PCB) from the bus so that the reduced conductance will result in a large enough delay from overheating or overcooling and its net result is closer to the orbital average temperature. The spacers physically do this by reducing the total surface area of the component in contact and can be further isolated by changing the material to low conductive materials such as stainless-steel, Teflon or Delrin if needed. Conversely, if more conduction is needed, additional or shorter spacers made from aluminum are an effective choice.

1.4.2 Torqued Fasteners

Each fastener on-board a spacecraft absolutely needs to be torqued to a specific value based on the fastener and materials. The primary reason for torqueing is related to the random vibrations a spacecraft will undergo on its journey beyond Earth [4], however this is greatly beneficial for the
thermal subsystem as well. Since the pressure of materials in contact is critical to conductance as seen previously from Figure 8, it is important to ensure each fastener and surface contact is reliable and predictable. Since pressure is a function of area, during the design phase it is usually possible to adjust the amount of contact a component has without changing the number of fasteners or torque values. When it comes to insulating components, there is a balancing of thermal isolation to structural stiffness [4].

1.4.3 Coatings

Every surface on the spacecraft is associated with their respective thermo-optical properties. To remain within the temperature limits for all components, it is most critical to consider the external surfaces first when trying to achieve this balance since it is exposed to the high power of solar radiation and nearly absolute zero background of space. To tune and control the material surface to achieve the desired properties, there are a few techniques available to optimize the coatings of the available surface of the spacecraft. The effective properties can be summarized based on their average qualities which is further elaborated in Section 2.6.2. This includes specific components, most commonly are solar cells, which are high emissivity and absorptivity, along with the surface finish from structural material. A very effective way is to use specially designed tapes or paints on free surfaces to achieve the desired thermo-optical properties without changing the structural properties. These tapes and paints do outgas to a certain extent but are degassed as much as possible before the tapes are applied, and in the case of paints, they are degassed afterwards, with careful consideration for the components in question.

Surfaces over long periods in orbit are exposed to solar radiation which may degrade the tape and change their effective properties. This needs to be accounted for as well depending on the mission length. This is not a problem for internal tapes, which are often used for high emissivity to reduce temperature gradients in the bus to avoid additional thermal expansion and stress. Note that for small microsatellites, this is often not needed as the bus is small enough to conduct the heat without causing problematic gradients. Ultimately, the need for internal coatings depends on the expected thermal gradients and how significant they are to the function of the spacecraft.
1.4.4 Heaters

The most common active thermal controller is the use of heaters with a thermostat. These heaters are most often flexible Kapton heaters since they are simple enough to avoid risk of failure, and the thermostat can be as simple as an independent PCB that controls the logic to determine if the heater should be on or off when crossing the threshold. The component that would need heaters are ones that cannot operate at colder temperatures, or if the thermal system needs certain components to be biased cold when balanced, and heaters could solve most cases if power is abundant [3]. This is most often seen for lithium-ion batteries as their efficiency significantly drops below 0°C.

1.5 Modelled Representations

Each of the physical properties and technological solutions described in the previous sections ultimately must be modelled and solved computationally to conclude any design. This is done by considering the primary physical drivers and excluding the insignificant factors to minimize the solve time [20]. A wise approach for developing any kind of complicated system using any finite element analysis (FEA) is to check the feasibility of the problems that are attempted to be solved. This is done by simplifying the satellite down to only a single or few nodes and calculated the resultant average temperatures from the orbit. A similar approach can be achieved by building a simplified computer model that integrates more details but is ultimately easy to consider all the parameters at once for checking. The practice is to reduce any potential sources of error in the modelling process and to verify relative changes with design choices as the project evolves.

1.5.1 Finite Element Method

All the physical mechanisms described previously can me modelled using FEA tools. The process involves representing the same physical traits of the system, and mathematically simplifying the system so that it is solvable within a practical time frame. This can be accomplished in many ways as it depends on the specific problems at hand.
There are various types of elements used for modelling in thermal analysis. Elements are used to capture the heat flow and capacitance of a system by often simplifying the materials and geometry to the most influential components and connecting them appropriately. A common element used, especially for early feasibility studies are in the form of zero-dimensional nodes which act as point masses to represent a physical capacitance. These are used when the expected temperature gradient is either negligible or unneeded, so the resulting element does not include internal conduction or any radiative connections to the rest of the system. One level above these elements are one-dimensional elements which connect two nodes together, and are often used to capture straight conductive connection. These elements can be defined with capacitance as their radius, length and material are also defined. Two dimensional elements are the primary build of the thermal model for microsatellites, which are often representing thin metal panels or shells. These elements are an array of nodes connected to each other where the thickness and material are specified to capture the total capacitance, the internal conduction and the radiative view factors relative to the other geometric elements in view. The build of a 2D mesh and its interconnected nodes can be seen in Figure 11 where the geometric volume around each node is computed and defined in the volume element. A similar case for three-dimensional elements exists, which are useful for complicated geometry where capturing the internal conductive paths are relevant. These 3D elements also produce 2D elements on the surface for computing the radiative connections with other 2D elements.
Calculating radiation and view factors is usually the most taxing aspect of the simulations. The radiation is calculated as a type of conductances based on the geometry and special differences of each element with respect to each other. Radiative conductances for each element are complex because they are based on the thermo-optical properties, area, position, and are all a function of the absolute temperatures of each element, unlike the relative temperatures for normal conduction. These radiative conductances are further explained in Section 1.5.4. When it comes to spatial resolutions, each node and its representative volume is describing an average of that volume, where high resolutions might be necessary to capture details near thermal junctions. This is especially true when modelling highly resistive materials where a larger temperature gradient is expected, although with increased resolution also increases the solve time of the solution. This fact is exponential for increasing the number of elements for calculating the radiative conductances. The entire art of creating a finite element model of the satellite is to maximize accuracy while minimizing computation time. This can only be done by understanding the mathematical representations of each of the physical interactions discussed.
1.5.2 Boundary Conditions

For building thermal models, the boundary conditions (BC) need to be defined before any model can be built. These are the external factors that are imposed on the spacecraft, such as the solar irradiance, the IR from Earth and its albedo, the background temperature of space and the Beta angle with an altitude. The foundation for thermal analysis is to separate the cases into extremes of hot and cold, known as worst-case hot (WCH) and worst-case cold (WCC). The idea is any design that closes with these parameters would envelope over the real on-orbit conditions that the spacecraft could possibly experience, ideally avoiding any problems entirely.

While solar irradiance is simply the respective maximum and minimum flux measured from existing space missions, Earth IR and albedo are not as simple. Their complexity comes from the nature of the earth being incredibly diverse on its surface, composed of land that can be any combination of dry, wet, icy, forested, along with oceans and clouds, which all change with the seasons, creating a dynamic set of possibilities for deriving the BCs. A practical and realistic approach is to treat these variables as probabilities, which are tied to known correlations such as seasons. Following this is approximating the thermal time constant $\tau$ [s] as seen in Eq. (15) [3] where $M$ is total mass [kg], $c_p$ is the specific heat $[\frac{1}{kg \ K}]$, $T_o$ is the orbital mean temperature [K] and $Q_o$ is the total average heat load [W] including all radiation and internal heat loads [3] [21]. The time constant is how long it takes the satellite to go from its orbital average temperature to 63% of its steady-state temperature.

$$\tau = \frac{M c_p T_o}{4 Q_o}$$ (15)

The use of the time constant with the approximate average emissivity and absorptivity, is used to determine which is the most sensitive to the spacecraft from the albedo, Earth IR, or both. One can conclude which set of BCs are appropriate for each WCH and WCC given the time constant, the most sensitive flux with the probable worst case, and nearly inverse relationship of albedo and Earth IR. These values were put together for thermal engineers by the Aerospace Corporation based on measurements and probability studies in the nineties to have practical realism [3].

Furthermore, how heat loads are applied can be simplified as well, yet still represent the environment accurately. As seen previously from Eq. (11), radiative heat transfer between the earth
and satellite is a relative interaction, however for LEO, it is best to compute the spacecraft as emitting to space in all directions (for a view factor of 1) and add the incoming Earth IR as an additional separate heat load [3]. This effectively simplifies the transient solving process as view factors for space are always idealized regardless of the satellite’s attitude.

There are also sources of heat from all the electrical components that are operating within the spacecraft. Every electrical component will dissipate all its power as heat, except for devices that cause mechanical motion, or transmitting antennas that by design emit a certain amount of their energy out of the system as EM radiation. Although all these components are typically and effectively solar powered, the total energy into the system is not changed by much (aside from the exceptions mentioned). The difference is when the batteries are charging, that energy hitting the solar cells are instead transferred to the batteries and follow through to the desired components internally rather than converting to heat on the cells directly. This is referred to as negative heat where the mechanism changes where the heat loads are applied, despite the total energy being identical.

1.5.3 Effective Surfaces

Given how the boundary conditions drive the satellite system, it is important to represent the thermo-optical properties to an accurate level. There will always be discrepancies between real structures and the CAD model, since the geometry needs to be simplified, a technique is used to ensure the effective optical properties are represented on the spacecraft. This is done by comparing the measured external surface areas, such as solar panels, structural panels and any distinguishable features of the spacecraft, and comparing each to their modelled surface areas. This results in the effective emissivity or absorptivity applied to the modeling surfaces as shown in Eq. (16) [13], where $\varepsilon$ and $\alpha$ are solved the same way. This ensures that the correct amount of energy is absorbed or dissipated regardless of small inaccuracies in the geometric size. For thermal coatings, there are tolerances to their thermo-optical properties, which must be considered to capture the worst-case conditions. This is done by artificially selecting the lowest $a/\varepsilon$ ratio for WCC and the highest for WCH.

\[
\varepsilon_{\text{effective}} = \frac{A_{\text{real}}}{A_{\text{model}}} \quad \alpha_{\text{effective}} = \frac{A_{\text{real}}}{\alpha_{\text{model}}} \quad (16)
\]
When modelling each of these worst cases, the attitude of the spacecraft relative to the sun will determine the edge cases that define the thermal envelope. No matter how unlikely, it is possible for any spacecraft to become inertially locked in such a way that the same faces are always oriented towards the sun. This is crucial for choosing the appropriate worst-cases, where for WCC it will be the side with the least area and lowest $\alpha/\varepsilon$ ratio, and for WCH, it will be the corners of the satellite with the highest area-absorptivity product. The exact angle of each corner will vary as each face of the spacecraft likely have a different absorptivity and is especially true if the satellite is any uneven shape. This can be calculated by first finding the average thermo-optical properties by using Eq. (17) [13] where $\sum A_n\alpha_n$ and $\sum A_n\varepsilon_n$ are the sum-multiplication of each visible component of a spacecraft’s face and $A_{Total}$ is the total area of the faces exposed to sunlight.

$$\varepsilon_{Total} = \frac{\sum A_n\varepsilon_n}{A_{Total}} \quad \alpha_{Total} = \frac{\sum A_n\alpha_n}{A_{Total}} \quad (17)$$

This is followed up by then computing the largest area-absorptivity product for each corner to determine the angle that receives the most sunlight. This is expressed in Eq. (18) [13] to compute the vector parts of each axis. Using the X axis as an example, $\hat{s}_i$ is the partial sun vector in the X axis, $A_x\alpha_x$ being the unit vector in the X axis (normal to the face), and $\|A_{xyz}\alpha_{xyz}\|$ is the norm of the vector from every exposed surface to the sun. Only three axes can be sunlit at a time, so adjust for negative faces however appropriate. The result will be a unit vector fraction that defines each axis, thus producing the WCH vector when calculated for each corner of the spacecraft.

$$\hat{s}_i = \frac{A_x\alpha_x}{\|A_{xyz}\alpha_{xyz}\|} \quad \hat{s}_j = \frac{A_y\alpha_y}{\|A_{xyz}\alpha_{xyz}\|} \quad \hat{s}_k = \frac{A_z\alpha_z}{\|A_{xyz}\alpha_{xyz}\|} \quad (18)$$

1.5.4 Resistances

Every material and interconnections of components have their own thermal resistances. These can be modelled and solved analogously as electrical circuits, where voltage is temperature, current is energy flow, with electrical resistance and capacitance being thermal resistance and capacitance [19]. Determining the resistance based on the physical conduction and connecting those resistances between elements is how conduction in the model is built for physical connections.
Resistances in series are added together to calculate the equivalent resistance $\frac{K}{W}$, as seen in Eq. (19) [19] and Figure 12 for two surfaces with a compressed material in between, such as a spacer or thermal gap filler. Here the connections are in series, and are ordered from the first surface resistance, to the internal resistance of the spacer, and to the second surface resistance in contact with the second object, where this example assumes the two objects are the same material. This is where $l$ is the thickness [m], $k$ is the internal conductivity of the spacer $\frac{W}{Km}$, $\sigma_c$ is the contact conductance $\frac{W}{Km^2}$, and $A$ is the cross-sectional area of the connection [m$^2$].

$$R_{Path} = R_{internal} + 2R_{contact} = \frac{l}{kA} + \frac{2}{\sigma_c A}$$

(19)

**Figure 12: Cross section of filler in between two objects**

As discussed in Section 1.3.1, the pressure of materials in contact will have an effect on the conductance, and therefore its resistance. This difference is captured by the contact conductance where measured values with common materials and their connections are used for high or low-pressure connections. Also mentioned in Section 1.5.1, nodes take up a volume within elements, so when connecting elements together and defining their total resistances, the individual resistance through the half-element volume and the contact resistance must be calculated and modelled in a similar way [13].

Beyond a series of resistances, there are parallel resistances where two nodes can have multiple different pathways between the two points. This can be visualized in Figure 13 where there are two thermal connections with one through surface contact and the other through the fastener itself. This example involves a PCB that is fastened directly into the structure’s elevated boss. Notice the pathways visualized are through the high-pressure connections only, neglecting
the sides of the PCB’s clearance holes. To determine an equivalent resistance for parallel paths is the same process as with electrical circuits, as described in Eq. (20) [19] where all serial connections are merged together prior where $R_{Path1}$ are the series of resistances that connect the hypothetical PCB to the boss on the tray and $R_{Path2}$ is the series of resistances that connect the PCB through the fastener and into the threaded connection of the structure.

![Diagram](image)

**Figure 13:** The thermal layout for a fastened PCB, with the resulting thermal network [22]

$$\frac{1}{R_{eq}} = \frac{1}{R_{Path1}} + \frac{1}{R_{Path2}}$$  \quad (20)

For internal radiative connections, they cannot be directly converted into a constant resistance and solved, however they can be approached as a constant radiative resistance during small enough time steps. An approximation can be seen in Eq. (21) [19] where if the temperatures of the radiative surfaces are assumed constant, a resistance can be inferred for modelling which is computed at the start of every iteration for transient simulations. This can also be useful for a feasibility study where internal radiation is significant. This approach effectively acts as a variable resistance that changes with temperature, so if the time steps for the transient solution are small enough, the resistance is constant for that time step and the estimate is adequate.

$$R_{rad} = 1/[\varepsilon\sigma A(T_1^2 + T_2^2)(T_1 + T_2)]$$  \quad (21)
1.5.5 Transient Solutions

Transient solutions are absolutely necessary for solving any complicated thermal system. Steady-state solutions may have their place for specific cases, such as with a sun-synchronized dawn-dusk nadir tracking orbit, the model must capture the constant changing shifts from eclipse to sunlight as a spacecraft orbits about every 90 minutes in LEO to represent reality. The necessity for a transient solution over a steady-state one is further apparent with the number of nodes that may be present in a system, where many hundreds are used for microsatellites, but can extend into the many thousands for larger buses where temperature gradients are much more prevailing and desired to measure.

Transient solutions are powerful as they can make a few assumptions to compute a vast amount of data. Let’s consider Figure 14 where there are two capacitive nodes representing the surface of the satellite connected to the internal core, each with thermal mass, and an internal resistance, with all the typical BCs are in effect.

![Figure 14: Two node system with all parameters](image)

Following the conservation of energy, each node is represented with the system of equations in Eq. (22) where each of the terms have been defined in detail in Section 1.3 and are expressed in [W]. Each equation describes the net energy flow to and from the node where \( \dot{Q}_{gen} \) is the heat generated within the satellite from powered systems, \( \dot{Q}_{emit} \) is the energy emitted as IR into space, \( \dot{Q}_{IR} \) is the energy absorbed by Earth IR, \( \dot{Q}_{solar} \) and \( \dot{Q}_{albedo} \) is the energy absorbed from sunlight, and \( \dot{Q}_{cond} \) is the energy flow from one node to the other through conduction. Note that the energy absorbed by the thermal mass is \( \dot{Q}_{mass} \) for each node where the energy flow is relative to time rather than space by the use of specific heat \( c \) or thermal capacitance \( C \) from Eq. (14).

\[
\text{Surface: } \dot{Q}_{mass\text{surface}} + \dot{Q}_{emit} = \dot{Q}_{cond} + \dot{Q}_{solar} + \dot{Q}_{albedo} + \dot{Q}_{IR} \\
\text{Core: } \dot{Q}_{gen} = \dot{Q}_{cond} + \dot{Q}_{mass\text{core}} \quad (22)
\]
The process for transient analysis is to define a time step where the assumption can be made that all heat transfer rates during that time are constant [19] which means an initial temperature remains constant, and the resulting temperature at the end of the time step is calculated as part of the delta thermal mass. The smaller the time steps, the more accurate the solution, as realistically, temperature changes continuously with any heat transfer, however this can be taxing on the processor to compute. The same iterative approach is used for solving steady-state systems, which can be impossible to solve analytically due to the number of nodes involving radiation [23]. Another taxing process are the calculation of view factors, as each element needs to calculate which fraction of every other element are in its view, which can become extremely bloated with finer elements. A standard approach is to separate the view factor calculations into their own larger time steps, while the radiative heat transfer calculations are tied to the smaller increments described previously. This will have no effect for internal radiation but is a safe assumption for external radiation when calculating view factors of each element to the earth and smaller time steps for a rotating satellite.
Chapter 2

DAUNTLESS

The Daringly Uncommon Technical Leadership in Smaller Satellites (DAUNTLESS) platform is the latest bus from SFL and is capable of supporting up to 500kg for a wide array of missions in LEO, GEO and highly elliptical orbits (HEO). This is also the largest microsatellite platform developed at SFL, which due to its large size is capable of housing a large propulsion tank inside the bus. The system also includes the modular power system (MPS) and target tracking capabilities common with other SFL platforms.

2.1 Mission Overview

A mission was fully developed using the DAUNTLESS platform that will be used as a demonstration for the thermal design and methodology of the spacecraft. The design of the spacecraft is for a LEO SSO, involving several antennas, the largest of which is a 54 cm diameter dish, which required pointing through nadir tracking to function properly. The 1000 km altitude being much higher than most LEO satellites includes a CO₂ propulsion system to de-orbit the satellite within 25 years after its year-long mission is complete. It can be seen in Figure 15 where it is approximately 59 dm³ and 72 kg.

![Completed satellite based on the DAUNTLESS platform](image)

Figure 15: Completed satellite based on the DAUNTLESS platform
2.1.1 Mission Requirements

The driving mission thermal requirements can be summarized in Table 1 where many of the rationale have been discussed in Section 1.4. The primary requirement is to ensure that all components in the spacecraft stay within their respective temperature limits, which have been specified by either the manufacturers, or derated internally, with margins added on top of the existing specifications. Derating components are a process adopted from ESCC guidelines [24] where all electrical components are verified to assure their reliability to function in vacuum. Additionally, unit level testing is performed on every component where the survivable temperature limits are verified, along with a narrower operational limit.

<table>
<thead>
<tr>
<th>No.</th>
<th>Description</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>General Requirements</td>
<td></td>
</tr>
<tr>
<td>THM-R001</td>
<td>The thermal control subsystem shall ensure that all units and payloads are kept within their specified non-operational temperature limits at all times.</td>
<td>To ensure that all units on the spacecraft are inherently safe at all times, even at initial power on when temperature of the unit may not be known.</td>
</tr>
<tr>
<td>THM-R002</td>
<td>The thermal control subsystem shall ensure that all units and payloads are kept within their specified operational temperature limits during nominal operations.</td>
<td>To ensure that all units are operational during nominal operations when all units on the spacecraft are required to perform nominally to support payload operations.</td>
</tr>
<tr>
<td>THM-R003</td>
<td>Thermal control measures shall satisfy requirements for a minimum duration of 1.5 years.</td>
<td>Thermal tapes degrade with time. This requirement specifies EOL conditions assumed in thermal analysis. Duration derived from 1-year service life plus commissioning, plus deorbit maneuvers.</td>
</tr>
<tr>
<td>THM-R004</td>
<td>On the hot end of a thermal range, thermal control margins shall be at least 10°C for the battery and at least 5°C for everything else.</td>
<td>Standard SFL practice.</td>
</tr>
<tr>
<td>THM-R005</td>
<td>On the cold end of a thermal range, thermal control margins shall be at least 5°C.</td>
<td>Standard SFL practice, battery margin relaxed to help bias spacecraft and payload toward cool end of range.</td>
</tr>
<tr>
<td>THM-R006</td>
<td>Thermal control measures should be passive.</td>
<td>To conserve mass and power and to limit design complexity. Requirement is 'should' because battery heaters are often required.</td>
</tr>
<tr>
<td></td>
<td>Specific Control Requirements</td>
<td></td>
</tr>
<tr>
<td>THM-R101</td>
<td>The thermal control subsystem shall ensure that each cell in the battery pack is within 2°C of every other cell in the pack.</td>
<td>To ensure even charging, discharging and aging of cells.</td>
</tr>
<tr>
<td>THM-R102</td>
<td>The thermal control system shall maintain the solar cells under 80°C during nominal nadir observing operations.</td>
<td>To ensure power generation consistent with what is assumed in the power budget.</td>
</tr>
</tbody>
</table>

The difference with operational and survivable temperatures are components can operate safely within the operational limits. If units are powered on between their operating and surviving range, they may not perform as desired, with anything beyond survivable can cause permanent damage.
or degradation regardless of the power state. The non-operating temperatures are a range between operation and survival where the component may be turned on, although not function as desired. In most cases the non-operational temperatures and survival temperatures are defined as the same temperature and so will be referred to as the survival temperature.

Temperature limits for every thermal sensitive component are listed in Table 2 where the components that impact the thermal design the most are the batteries and payload for the cold cases, and the propulsion tank for the hot cases. As mentioned, many of these requirements come from their respective unit testing, where with new technology, there must be a qualification test and acceptance testing before the final values are determined. The payload antenna itself went through this process independently with the developers of the unit, in addition to the entire spacecraft for environmental acceptance testing once the flight model was assembled. Note that the tank fuel average has non-applicable survival temperatures due to the requirement being derived from the resulting pressure from the given mass and volume of the fluid, therefore is always in an operating state.

Table 2: Temperature requirements for the DAUNTLESS satellite [25]

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Components</th>
<th>Overall Requirements [°C]</th>
</tr>
</thead>
<tbody>
<tr>
<td>C&amp;DH</td>
<td>OBCs</td>
<td>-30</td>
</tr>
<tr>
<td></td>
<td>SIB</td>
<td>-30</td>
</tr>
<tr>
<td>Power</td>
<td>Solar Cells on Panels</td>
<td>-40</td>
</tr>
<tr>
<td></td>
<td>Battery</td>
<td>0</td>
</tr>
<tr>
<td></td>
<td>MPS</td>
<td>-30</td>
</tr>
<tr>
<td>ADCS</td>
<td>Star Tracker</td>
<td>-40</td>
</tr>
<tr>
<td></td>
<td>Sensors</td>
<td>-30</td>
</tr>
<tr>
<td></td>
<td>Reaction Wheels</td>
<td>-30</td>
</tr>
<tr>
<td></td>
<td>Smart Torquers</td>
<td>-40</td>
</tr>
<tr>
<td></td>
<td>GPS Receiver</td>
<td>-30</td>
</tr>
<tr>
<td></td>
<td>GPS Patch Antenna</td>
<td>-55</td>
</tr>
<tr>
<td>Communications</td>
<td>S-Band/K-Band Components</td>
<td>-30</td>
</tr>
<tr>
<td></td>
<td>S-band Patch Antenna</td>
<td>-55</td>
</tr>
<tr>
<td></td>
<td>K-band Antennas</td>
<td>-40</td>
</tr>
<tr>
<td>Propulsion</td>
<td>Tank Composite</td>
<td>-20</td>
</tr>
<tr>
<td></td>
<td>Tank Fuel Average</td>
<td>N/A</td>
</tr>
<tr>
<td></td>
<td>Electronics Board</td>
<td>-30</td>
</tr>
<tr>
<td>Payload</td>
<td>PL Antenna - Reflector</td>
<td>-100</td>
</tr>
<tr>
<td></td>
<td>PL Antenna - Sub-Reflector</td>
<td>-100</td>
</tr>
<tr>
<td></td>
<td>PL Components</td>
<td>-20</td>
</tr>
</tbody>
</table>
2.2 Boundary Conditions

The thermal environment for the DAUNTLESS satellite can be simulated with the proper parameters inputted into the system. These boundary conditions (BC) are used to envelope the entire satellite’s worst-case conditions, and converge on a thermal design, primarily through the use of thermal tapes that will change the effective thermo-optical properties of the spacecraft. For reference, the axes are defined in Figure 16 where the +Z axis is in the direction of the payload antenna dish used for nadir tracking, and the -X axis is the direction of the thruster.

![Figure 16: DAUNTLESS satellite render with reference axes](image)

All the boundary conditions for the final spacecraft are represented in Table 3 where each BC is with an LTAN of 15:05 and an altitude of 1000 km. The cases are separated into the enveloping WCH and WCC which represent the hottest and coldest conditions the spacecraft can physically experience, with additional margins. For WCH, a combination of the highest Beta angle and solar flux are chosen to represent the largest heat load to the spacecraft, with the addition of the largest projected area to the sun for maximum absorptance, and with the highest operating power mode that further adds heat to the system. For the WCC case, the Beta angle and solar flux are chosen to represent the smallest heat loads to the system throughout the orbit, with a single face inertially locked to the sun for the smallest area and the spacecraft is in its lowest power mode of Safehold mode. The nominal modes are also determined in a similar way, however include the realistic attitudes and power modes expected for the mission. Hot and cold nominal are tracking a target on the surface of the earth where the power modes include the payload operating for the hot case and
most systems running except for the payload for the cold case. The thrusting case is similar to the nominal cases, but includes the propulsion system degrading the orbit since the thruster is mounted on the X axis. The power modes are further elaborated in Section 2.2.2 where the powered units dissipate heat into the system.

Table 3: Final BC for the DAUNTLESS satellite

<table>
<thead>
<tr>
<th>Case</th>
<th>LTAN</th>
<th>Altitude [km]</th>
<th>Beta Angle</th>
<th>Solar Flux ([\text{W} \text{m}^{-2}])</th>
<th>Albedo</th>
<th>Earth IR ([\text{W} \text{m}^{-2}])</th>
<th>Mode</th>
<th>Attitude</th>
</tr>
</thead>
<tbody>
<tr>
<td>WCH</td>
<td>15:05</td>
<td>1000</td>
<td>50.8</td>
<td>1392.74</td>
<td>0.31</td>
<td>232</td>
<td>S-Band Non-Operational</td>
<td>Max projected area absorptivity</td>
</tr>
<tr>
<td>Hot Nominal</td>
<td>15:05</td>
<td>1000</td>
<td>50.8</td>
<td>1392.74</td>
<td>0.31</td>
<td>232</td>
<td>Nominal Operational</td>
<td>+Z Nadir Tracking</td>
</tr>
<tr>
<td>Hot Thrusting</td>
<td>15:05</td>
<td>1000</td>
<td>50.8</td>
<td>1392.74</td>
<td>0.31</td>
<td>232</td>
<td>Thrusting</td>
<td>+Z Nadir, X axis orbit vector</td>
</tr>
<tr>
<td>Cold Nominal</td>
<td>15:05</td>
<td>1000</td>
<td>35.5</td>
<td>1322.56</td>
<td>0.25</td>
<td>230</td>
<td>Nominal Non-Operational</td>
<td>+Z Nadir Tracking</td>
</tr>
<tr>
<td>WCC</td>
<td>15:05</td>
<td>1000</td>
<td>35.5</td>
<td>1322.56</td>
<td>0.25</td>
<td>230</td>
<td>Safehold</td>
<td>Min projected area absorptivity</td>
</tr>
</tbody>
</table>

Throughout development, due to the satellite’s large size, the parameters started with a broader envelope of operation cases and Beta angles, which eventually narrowed to capture more accurate thermal cases to close the design. The final results and any deviations from the operating modes are discussed in Section 2.3 and the BCs are discussed in Section 2.6 where the design was able to close after countless adjustments.
2.2.1 Orbits

Throughout the development of the spacecraft, there were two potential orbits that were considered. The second orbit was defined with an 11:30 LTAN, which effectively meant the beta angles were closer to 0° for both worst-cases by about 35°. The differences in Beta angles for the 15:05 orbit affects the system such that the hot cases become much hotter compared to the shift for the cold cases, resulting in larger gradients experienced by every subsystem. Because of the small temperature margins for the components, this warranted two different thermal tape scheme designs to close the spacecraft.

The design of the spacecraft had some thermal restrictions that resulted in adjusting the worst-case Beta angles. The driving cases were due to the propulsion tank upper limit during WCH conditions and the payload lower limit during WCC conditions. The changes on the system is that some conservative assumptions about the BCs had to be explored to reduce the mathematical impact it had on these components. From the orbital parameters, the Beta angle, and solar irradiance initially started for using their maximum and minimum for the WCH and WCC respectively, however eventually were chosen to be more accurately representing the actual conditions from the orbit since the initial conservative assumptions take the worst-case irradiances and Beta angles together, which is often unrealistic. As discussed in 1.2.3 about Beta angles, the angle furthest from 0° is the angle where the spacecraft experiences the most sunlight for WCH and vice-versa for WCC. Realistically, the time of the year this occurs does not necessarily coincide with the maximum or minimum solar flux due to the Earth’s elliptical. The resulting BCs were chosen to a specific day of the year where the total energy absorbed of the orbit was taken at the highest for WCH and lowest for WCC.

2.2.2 Heat Dissipation

Included in the boundary conditions are the heat loads applied to every electrical component within the spacecraft. Table 4 shows a list of components with some of the various modes with their average power consumption per orbit of operation. This is not an exhaustive list but captures the main edge cases that will be used for discussion.
Table 4: DAUNTELSS component power consumption [26]

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Components</th>
<th>Power [W]</th>
<th>Nominal Operational</th>
<th>Safehold Mode</th>
<th>Thrusting</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>#</td>
<td>Duty Cycle %</td>
<td>Orbit Average [W]</td>
<td>Duty Cycle %</td>
</tr>
<tr>
<td>C&amp;DH</td>
<td>OBC 1*</td>
<td>0.550</td>
<td>1</td>
<td>100%</td>
<td>0.550</td>
</tr>
<tr>
<td></td>
<td>OBC 2</td>
<td>0.550</td>
<td>1</td>
<td>0%</td>
<td>0.000</td>
</tr>
<tr>
<td>SIB</td>
<td></td>
<td>0.350</td>
<td>1</td>
<td>5%</td>
<td>0.500</td>
</tr>
<tr>
<td>Power</td>
<td>MPS*</td>
<td>0.575</td>
<td>1</td>
<td>100%</td>
<td>0.575</td>
</tr>
<tr>
<td></td>
<td>BIM*</td>
<td>0.150</td>
<td>1</td>
<td>100%</td>
<td>0.150</td>
</tr>
<tr>
<td></td>
<td>Battery Heater</td>
<td>1.480</td>
<td>1</td>
<td>0%</td>
<td>0.000</td>
</tr>
<tr>
<td>ADCS</td>
<td>GPS Receiver</td>
<td>1.100</td>
<td>1</td>
<td>100%</td>
<td>1.100</td>
</tr>
<tr>
<td></td>
<td>Reaction Wheels*</td>
<td>0.300</td>
<td>3</td>
<td>100%</td>
<td>0.900</td>
</tr>
<tr>
<td></td>
<td>Sun Sensors</td>
<td>0.150</td>
<td>5</td>
<td>100%</td>
<td>0.750</td>
</tr>
<tr>
<td></td>
<td>Rate Sensor</td>
<td>0.330</td>
<td>1</td>
<td>100%</td>
<td>0.330</td>
</tr>
<tr>
<td></td>
<td>Magnetorquers*</td>
<td>0.400</td>
<td>6</td>
<td>100%</td>
<td>2.400</td>
</tr>
<tr>
<td></td>
<td>Magnetometer</td>
<td>0.160</td>
<td>1</td>
<td>100%</td>
<td>0.160</td>
</tr>
<tr>
<td></td>
<td>Star Tracker</td>
<td>0.600</td>
<td>1</td>
<td>100%</td>
<td>0.600</td>
</tr>
<tr>
<td>Comms</td>
<td>S-Band Rx</td>
<td>2.000</td>
<td>1</td>
<td>100%</td>
<td>2.000</td>
</tr>
<tr>
<td></td>
<td>S-Band Tx*</td>
<td>5.000</td>
<td>1</td>
<td>5%</td>
<td>0.250</td>
</tr>
<tr>
<td></td>
<td>K-Band Rx</td>
<td>5.000</td>
<td>1</td>
<td>5%</td>
<td>0.250</td>
</tr>
<tr>
<td></td>
<td>K-Band Tx*</td>
<td>45.000</td>
<td>1</td>
<td>5%</td>
<td>2.250</td>
</tr>
<tr>
<td>Propulsion</td>
<td>Thruster*</td>
<td>75.000</td>
<td>1</td>
<td>0%</td>
<td>0.000</td>
</tr>
<tr>
<td></td>
<td>Solenoid Valves</td>
<td>0.520</td>
<td>2</td>
<td>0%</td>
<td>0.000</td>
</tr>
<tr>
<td></td>
<td>Electronics Board</td>
<td>0.685</td>
<td>1</td>
<td>0%</td>
<td>0.000</td>
</tr>
<tr>
<td>Payload</td>
<td>PL Components</td>
<td>28.100</td>
<td>1</td>
<td>5%</td>
<td>1.405</td>
</tr>
</tbody>
</table>

Not every power mode is listed here, as some can be a combination of the modes shown. Specifically, are the non-operational modes where this refers to the payload in an off state. This is from considering the payload’s survivable limits for the worst-case conditions rather than the operational limits across every case. This is due to the main payload components mounted on the +Z panel internally which are well coupled with the space environment the panels are exposed to. With all worst cases considered, the resulting temperatures exceed the ability to close the design passively when the sun is directly on the panel with the components on. This case is safe enough to study as the sensitive components have a thermostat safety which shuts off the components if mis-commanded to turn on outside of a nadir tracking orbit. This is a necessary outcome of the DAUNTELSS design because the payload components are directly mounted to an external spacecraft panel where the problem is further explained in Section 2.4.4. Another sub-mode is the S-Band mode which is similar to nominal modes, except excludes the use of the K-Band antennas since they can only practically function during nadir tracking of a ground station and not while the spacecraft is arbitrarily inertially locked to the sun. This case was specifically to meet the upper
operational limit of the propulsion tank, which also has a thermostat safety function. For more information on the specific cases, see Section 2.4.

It is also worth discussion that Table 4 is a simplification of the actual heat loads which are accurately modelled into the system. The Safehold mode used for the WCC conditions are modelled with conservatism since the receiving antennas are always on and the transmitters are partially on if entered in Safehold mode. The components with * are referring to various conditions that are specific to the mode or system. For example, the MPS changes constantly throughout an orbit as it is charging the batteries from the solar cells and will change depending on which components are on during specific times. The reaction wheels and magnetorquers are also dependent on the attitude of the spacecraft, since they are converting electrical energy into mechanical motion, which only some of the power is converted into heat. Nominal, the system is target tracking and operates in a low power state to remain fixed on the target, which is represented in the table. The other components can have a varying load throughout operations such as the onboard computer (OBC) running in a low power state during safe hold, or the thruster using less power once its target temperature is reached. All of the operational power modes are modelled such that they begin operations once the batteries are fully charged after passing the eclipse period. Although this may not necessarily be actual timing of the on-orbit functionality of the satellite, the assumption is it captures the WCH condition because the primary operations are modelled to occur during the peak heat loads of the orbit in sunlight.

2.3 Design Concepts

The DAUNTLESS spacecraft was modelled using Siemens NX 8.0 [27] where all of the modelling representations from Section 1.5 were used. The assembly layout can be broken down from top to bottom into 3 different stages. At the top is the payload which includes main antennas and its supporting equipment, followed by the internal bay with the propulsion equipment, and finally the avionics bay at the bottom which includes all the electrical components that support power, communications and attitude control. This can be seen in Figure 17 where components are largely grouped together and the main structural tray supporting the tank and avionics is called the internal panel.
Figure 17: The DAUNTLESS satellite internal components with meshes

2.3.1 Internal Balance

Due to the larger bus of this satellite, temperature gradients are much more prevalent to the system and therefore requires finer meshes to properly capture the internal radiation. These gradients occur because the cumulative conductive path from one side of the spacecraft to the other is larger than other microsatellites, so thermal energy takes longer to transfer and influence any opposing end of a sun illuminated surfaces. To help average temperatures throughout the internals, high emissivity tapes were applied to all internal faces across the satellite, as seen in Figure 18 with the orange tapes on every free surface. The tank is made of a carbon fiber composite, which carry more restrictive thermal limits because thermal stresses and gradients in the tank are more likely to cause a rupture compared to a solid metal shell.

Figure 18: The internal bay with the propulsion tank coated with gold tape
The dominant thermal path for the tank, as with most objects in a spacecraft, is through the conduction of the supports. The tank is connected with aluminum legs with the actual contact points limited to the ribs of the internal panel. The actual connecting point from the tank to the supporting legs are connected through a series of stainless-steel rings which further add to the total thermal resistance and decoupling of the tank. Additionally, due to its large surface area, it becomes very sensitive to radiative heat flows and therefore requires as much isolation as possible. Note that the fuel is modelled as a point mass but is not used for the majority of the simulations to add conservatism for capturing the worst cases without fuel. Because of the internal radiation sensitivity on the tank, the tank is coated in very low emissivity gold tape to decouple the exposed surfaces from any of the hot sides of the spacecraft. It is worth noting that due to the high emissivity panels, the effective emissivity on the tank is reduced, as seen from Eq. (13) to determine the equivalent emissivity.

The avionics bay also needed internal radiation modelled due to the same reason where certain inertially locked sun stare cases could result in large thermal gradients where radiation becomes significant. The avionics bay can be seen in Figure 19 where the majority of the units are modelled to capture their radiative influences. In the case of the K-Band unit and the radio enclosures, the internals of those modules did not include radiative connections as they are relatively small enclosures made of aluminum. This effectively means that they are well coupled with the boards inside, and they were modelled as explained in Section 1.5.4.

![Figure 19: The mesh of the avionics bay, showing the radiative components](image-url)
A sensitive component in the avionics bay is the battery pack, which does not work as efficiently when the cells drop below 0°C. The best way to ensure they remain warm enough considering all conditions is to equip them with a heater. The heater is also modelled in the thermal model as a thermostat which only turns on or off within a temperature threshold. The threshold is set much higher than the operational requirement from 15°C to 18°C. The full assembly is seen in Figure 20 where there is one heater mounted on a thin aluminum plate, and thermally coupled to the batteries with conductive gap filler. To ensure the battery pack itself is as isolated as possible, it is held within Delrin, an isolating material with little outgassing properties and high rigidity. The pack is further isolated with spacers to reduce the total area in contact with the internal panel. Similar to the tank, the batteries and Delrin were coated with gold tape to reduce the emissivity in all cases to ensure any IR is reflected from the unit.

![Figure 20: The battery pack assembly and flight pack](image)

2.3.2 External Payload

All the payload components are mounted on the inside of the +Z panel, with the reflectors exposed on the outside of the spacecraft with the electrical components mounting inside, on the same panel. The internal payload components are the only major subsystem to be modelled as 0D nodes as seen in Figure 21. The primary reason for this is all of the components are well coupled with the panel, and do not approach their limit for the hot cases, which are the cases where radiation becomes sensitive and therefore the extra detail is not necessary. These nodes cannot radiate as they have no physical dimensions, however determining the effective emissive properties on the internal side of the panel itself, combined with the components being well coupled, offer an accurate representation of the payload components and how the panel radiates internally.
Figure 21: Payload components and their 0D nodes in the thermal model

The payload reflectors are mounted to the external panel and connected by the feedchain directly below the dishes. The payload reflectors and the feedchain connecting the signal have also been modelled effectively being made primarily of 2D meshes as seen in Figure 21. Since the reflector is large and is fixed with relatively isolating mounted legs, they are decoupled from the bus. Because of this, there are some inertially locked cases where the spacecraft can be partially shadowed, and effectively prevent most of the solar energy from entering the bus, thus dropping the average temperature extensively. The conduction of the legs, and radiation from the reflectors to the bus play an important role determining the effects of the shadowing accurately.

2.4 Studies

Throughout the development of this spacecraft, studies were performed to understand and conclude design choices. The concluded critical components are presented in this section. These components are the payload antenna structural fixture and coatings, the coating and modelling approach for the tank, the effects of the thruster on the system, and a system level keep away attitude. The results from these studies allowed for the development to conclude within schedule.
2.4.1 Payload Antenna

The payload antennas and components were outsourced to function as a single unit of the spacecraft. Throughout its development, the thermal requirements imposed on the structure were eventually solved as solutions were iterated towards the conclusion. The antenna coatings themselves were part of these discussions, where as seen previously in Figure 15, the two reflectors are almost entirely coated in a white paint which acts as a high emissivity and low solar absorptivity. The smaller sub-reflector itself however is not coated on the bottom section facing the main reflector resulting in a low emissivity surface. The overall effect of this inclusions was to keep the antennas cold biased to ensure there is no thermal noise influencing the measurements. There is also further consideration for the overall temperature gradients across the antennas due to thermal expansion imposing stresses, as it was required that the units experience as little gradients as possible to remain accurate during their operations. The study for the antennas are centered around WCH, WCC and worst-case gradients as can be seen in Figure 22 where the relevant temperatures for each reflector, and which case those results are from. These results were used to explore the effects of thermal stress from expansion due to the possibility of large changes from -99°C to 56°C which are beyond what was originally expected during preliminary design.

![Inertial Sun Stare Attitudes](image)

Figure 22: Antenna temperature extremes and gradients
2.4.2 Propulsion Tank

The carbon fiber propulsion tank despite being decoupled from the bus, still drives the thermal design for the spacecraft. As mentioned in Section 2.3.1 with regards to the internal balance, the methods used for decoupling are to limit structural connections, which are the largest source of heat transfer, along with low emissivity tapes on the surface. The tape was a critical part of reducing the possible temperature range the tank could experience. Without any coatings, the tank is a highly emissive black color, which combined with the highly emissive tapes of the internal panels results in the tank being well coupled with the spacecraft. Due to the smaller temperature range for the tank to function, a study was done to see how critical the use of tapes and multi-layered insulation (MLI) could be. From the measured black emissivity of the tank to the gold coating, the simulated variations showed that if the spacecraft were to approach its orbital average temperature, the WCH cases did not affect the temperatures by more than a degree. Although the primary conduction is through the supports, realistically the gold tape did affect the rate of heat transfer as it would result in the same maximum temperature but would take a dozen orbits longer to reach. Drastically different is for the WCC cases, where the gold tape actually prevents the tank from getting over 10°C colder, effectively concluding that the gold tape successfully narrows the temperature range by 15°C with all worst-cases considered. The difference for the cold case is due to the tank being such a large surface area, that if any side of the spacecraft is cold, as with the inertially locked cold cases, the tank will emit its stored energy to that side of the spacecraft, effectively acting as an additional heat sink. Furthermore, being fixed to the internal panel, which is highly influenced by the avionic electrical components, are already kept relatively warm in Safehold mode relative to the coldest faces, and results in a warmer tank during these cases when gold tapes are used.

Another important consideration is the CO₂ fuel itself which is over 10 kg and one of the highest thermal masses of the spacecraft. As stated previously, most of the simulations exclude fuel in the simulations, as properly modelling the fluid introduces the possibility of producing optimistic results due to the complex behavior of fluids, thus adding risk for such a sensitive component. In addition, no fuel adds conservatism such that realistically would result in a cooler WCH and warmer WCC due to the thermal inertial of the high capacitance. A study eventually concluded that the effect of fuel in the thermal simulations are small enough to be considered
without. With the restrictive temperature range of the tank, it is not only limited by the thermal stresses of the material itself, but also due to the fluid pressure it can handle, which referring back to Table 2 is the tank fuel average temperature. This is the temperature of the fuel that results in approaching the pressure limit. It was worth exploring the effects of the fuel on the tank to know how much of an influence it can have. To do this, the thermophysical properties of the fluid must be determined. Knowing the fuel and density for a constant volume isochoric case, it is possible to gain everything needed. Using the NIST Chemistry database [28], and inputting the knowns, we can determine the fluid pressure and quality, with conductivity and capacitance with respect to temperature seen in Figure 23. As the fluid increases in temperature, it approaches its critical point at around 30°C, where the fluid has the highest specific heat, and decreases as the temperature goes higher being a supercritical fluid.

![Constant Volume Specific Heat vs. Temperature](image)

Figure 23: CO₂ fuel tank thermal properties vs. temperature [28]

When the fluid is not in its supercritical state, the vapor and liquid result with two different sets of properties that are related thermodynamically. This can be seen in Figure 23 where they represent two different curves below 30°C. When finding the resulting heat capacity of the system, an average relative to quality is appropriate as long at the change from liquid to vapor is captured with a variation in temperature. Using the vapor quality of the fluid as seen in Eq. (23) [29], the resulting heat capacity can be calculated based on the ratio of mass for each part. It is worth noting that since this is a constant volume fluid treated as an ideal gas, specific heat is defined with $c_v$
where for solids and constant pressure gases, it is defined as $c_p$ [29]. For actually connecting the fuel to the thermal model, rather than model the dynamic natural convection within the tank, a blanket approach was used where the realistic convection will be somewhere between behaving as a solid with conduction and a perfectly conductive point where the conservative assumption is that the convection will be able to transfer the heat instantaneously throughout the volume.

$$X = \frac{m_{\text{vapour}}}{m_{\text{total}}} \quad (23)$$

After exploring the worst-cases with the study, contrary to the initial assumption, including the CO$_2$ fuel in the thermal model did little to narrow the experienced temperatures. This is primarily due to the assumption that the WCH attitudes will remain constant indefinitely, resulting in a converged repeating transient cycle. The largest difference measured was the fuel modelled as a solid where the maximum temperature experiences was less by 1°C which relative to the limits and sensitives of the tank is adequate, however cannot be assumed because of the solid property assumption. Modelled as a perfect ideal fluid, the temperature of the fuel is the same as the average temperature of the tank. The conclusion is the fuel was not included due to its complexity while doing little to improve the margins. The tank average temperature is representative enough of the fuel temperature with minor conservatism.

### 2.4.3 Thermo-Electric Thruster

The hot gas thruster was developed by the propulsion team, and subsequently performed their own thermal tests resulting in plenty of data. The thruster works such that it heats up the barrel that the CO$_2$ leads out of, thus adding energy to the gas and effectively accelerating it out of the spacecraft increasing the thrust. The thruster physically has its own thermostat controller that regulates the device to operate the most efficiently at 800°C where it can consume 75 W. The device and thermal model can be seen in Figure 24 respectively. On the flight model, the thruster is mounted on a series of ceramic spacers with many stainless-steel washers. A thermal unit test was performed to determine the effective conductance of the unit to the mounting bracket.
The unit was then implemented into the thermal model to see its effect on the system, where two main outcomes needed to be determined. The primary study was to see if the thruster itself would be able to reach its target temperature with the available power in an open system, which seen on the mesh in Figure 24, the unit was able to go over 200°C beyond the target temperature. Note the figure is showing the model with a constant power load, where other simulations included the thermostat for regulation. The advantage of simulating with such a high temperature on the thruster is the supporting electronics-propulsion board and the solenoid valves were also generating much more heat in the simulation than in reality, thus being very conservative for this WCH condition. This case explored the effects on all units that could be affected, such as the tank and avionics bay, which showed to have a minimal influence, despite operating at a much higher temperature. The results for main electronics board can be seen in Figure 25 where the thrusting maneuver is constantly applied, and the effects on its board resulted in two orbits of continuous use to pass the operational range. Since the thruster will never operate for more than 10 minutes at a time, it was deemed acceptable to integrate into the DAUNTLESS satellite.

**Figure 24: Thermo-Electric Hot Gas Thruster**

**Figure 25: Propulsion electronics board with constant thrust**
2.4.4 Cold Platform

Throughout iterating on the tape schemes and closing the design, there were two cold cases that where certain parts of the satellite always exceeded their temperature limits. Specifically, was to balance the external thermo-optical properties with the right amount of absorptivity and emissivity so close the avionics components, the payload and the tank. The limitation of restricting the thermal control to mostly passive systems is the little heat transfer from one side of the bus to the other while only one face is locked to the sun. Although the bus is composed of mostly aluminum with high pressure connections, the larger size of the bus is the reason for the weak heat conduction from one end to the other. The two cold cases that were problematic are both the +Z and -Z sun stare attitudes for the WCC conditions. The problem for closing these cold cases is centered around the system as a whole with the propulsion tank’s upper limit. As seen in Figure 26, where the payload antenna is shadowing the bus, the resulting average temperature gets cold compared to every other case. This is especially true for the avionics bay that is opposite of the main source of solar energy where it may experience more than 10°C below their survivable limits. A similar situation is observed when the -Z panel is locked to the sun and the payload components go below their survival limits as well by 15°C. Regardless of which side, the subsystem opposing the side exposed to the sun goes well below the WCC temperature limits.

Figure 26: WCC temperature gradients across the DAUNTLESS bus
The only practical way of solving these issues are to increase the absorptivity of the exposed faces or reduce the overall emissivity of the bus. This effectively cannot be balanced because either of those solutions cause the propulsion tank to go well beyond its upper limit for the WCH. The fundamental reason is heat needs to transfer from the hot end to the cold, where if balanced to solve the cold cases where the avionics and the payload are warm enough, results in the tank getting too hot in the WCH conditions and vice-versa. The only option was to impose the keep away attitudes where the spacecraft could not actively point itself towards the sun in the +Z and -Z faces. The acceptable angles and slew rates are discussed in Section 2.6.1. In the case where by coincidence either of these faces are facing the sun, due to the natural magnetic dipole of the satellite, it is unstable along that axis and was found to orient into a tumble along the X or Y axes within half the orbit. For the purposes of the mission, this was found to be acceptable.

2.4.5 Preliminary Results

Considering everything discussed prior, it is clear that the DAUNTLESS bus and its mission have had challenges closing the thermal design by using exclusively passive methods. The preliminary results are seen in Figure 27 where the +Z and -Z inertially locked sun stares are excluded and every worst-case condition and nominal conditions are overlapped for the full results. It can be seen that the batteries and payload are extending past their cold temperature limits when considering the 5°C margin for the cold cases, and the propulsion tank extends well past the hot limit during the hot cases. Despite balancing the spacecraft temperatures using thermal tapes and narrowing the possible ranges from previous studies, it was made apparent with this configuration that DAUNTLESS cannot be closed with the same standards applied to other microsatellites at SFL. Comparing the two omitted sun stare cases, the resulting WCC temperatures for every component average -30°C across the entire spacecraft. Moving forwards, more accuracy in the model is needed to determine what options are available to ensure the spacecraft will guarantee to function appropriately.
Figure 27: DAUNTLESS bus worst case preliminary simulated results
2.5 Validation

The DAUNTLESS satellite underwent acceptance testing internally at SFL. This was the opportunity to validate the spacecraft with the thermal model after the flight model went through various tests in the thermal vacuum (TVAC) chamber. The need for validation is due to potential model inaccuracies or previously unforeseen interactions, which can ultimately lead to wrong predictions. During this stage of testing, the final thermo-optical tapes have not been finalized, instead a base layer part of the finalized scheme was applied as it was the primary layer on all surfaces, and the final layers could be added after validation. This layer is a low absorptivity and emissivity which was determined needed for the final flight, with possible optimizations with extra layers on top of the base layer. As long as a simulated case was made to match the tested spacecraft, validation is still accurate. The main components that were studied for validation were the batteries, the payload electronics and the propulsion tank. Up to this point, these components were driving the tape scheme design and the keep away attitude definition for the satellite where the solution for each of these individual components presented problems for the others. The batteries could get too cold with margin, the propulsion tank was too hot for the WCH cases by almost 10°C, and trying to balance for these would change the keep away angle driven by the payload (among other components). Being fundamentally a systems problem, validating the model was an opportunity to steer the thermal design with the intention of finding a tape scheme that balances these opposing components.

2.5.1 Thermal Vacuum Chamber

The TVAC chamber is a hermetically tight chamber that is able to create a high vacuum and has lamps as a source of heat while keeping the chamber walls cold. The vacuum is able to achieve pressures as low as 10^{-6} torr which is effectively a vacuum for the purposes of testing and acceptance. There are 24 lamps inside which can output power that is varied, and depending on the simulated environment, are tuned to match those. The chamber also features a cold background of 80 K (-193°C) by passing liquid nitrogen to the walls to keep a cold background temperature to emulate the cold background of space. The DAUNTLESS bus was in the TVAC chamber for a week and went through a series of different parameters. The spacecraft itself was physically mounted by being suspended from 8 wires near each corner of the bus, which is done to reduce
the conductive paths through high resistive contact points. The setup can be seen in Figure 28 where the chamber, wires, lamps and thermocouples are visible. The thermocouples are mounted on many points of the bus and panels, along with sensor data from the OBC to log from the internal temperature measurements. Note that the payload antennas were not part of the setup due to their size with the TVAC chamber lamp mounts.

![Figure 28: TVAC setup with a physical model and the flight model respectively](image)

After the chamber pumped out the air and reached the desired pressure, the thermal tests were performed for over a week. This included a steady-state initial temperature where the external panels plateaued at 20°C and the lamp voltages were marked as reference. The lamp voltages were estimated beforehand by calculating the lamp distances, effective output while also matching the amount of absorbed power into the bus and determining the effective emissivity of the spacecraft and the walls since it is not a black body and the main structural component obscures some of the total view factors from the bus.

Following this were various hot and cold slews where for the hot slew, the lamps would increase their output power and bring the external panels to 50°C where enough time would pass that the external panels would plateau and be steady. Afterwards the lamps are shut off and the cold slew begins where it is continued until the external panels plateau at -20°C. Because the primary source of heat transfer is through radiation, these slews take hours to achieve, especially for the cold slew. There are also slews that capture a nominal orbit where the lamps cycle such that the lamp power and time match the eclipse and sunlight cycles experienced on-orbit. As testing continues various system tests are performed at room, hot and cold temperatures. Specific for
thermal validation, are the unit tests where the TVAC chamber is reset so all the panels are constant at 20°C and different units are turned on one after the other as they would normally. The two critical systems these are beneficial to are the batteries and the payload components. The batteries have a small lower limit which with the combination of the active heater and thermostat, require careful modelling since this critical component powers the entire spacecraft. The payload components are the mission holders of the satellite and must function despite the limited lower temperature requirement. Extra precaution is given to the fact that these components are mounted to an external panel which is susceptible to the large ranges and stresses of the thermal environment. The more accuracy built into the model is greatly beneficial for not only the DAUNTLESS bus, but all spacecraft currently in development at SFL, and future larger microsatellites alike.

2.5.2 Model Preparation

Leading up to the TVAC tests were the modifications of the thermal model to match the physical assembly that was mounted in the chamber. The reason for this is to ensure all measurements and simulated results correlate as closely as possible so that any changes needed afterwards can be extrapolated for the final build. The primary changes for the flight model (FM) itself was the lack of a payload antenna, which in the model was simply disconnected completely through conduction and was transparent for any view factor calculations. Furthermore, were the effective thermo-optical properties where only one set of base layers were applied to the bus. This only matters in the simulation for the emissivity properties as the heat loads applied from the lamps are applied as a total power sum spread across the surface area of the spacecraft, manually adjusted according to their absorptivity. This was done to easily iterate on the power inputs and match the internal boundary conditions of the satellite. The best way to validate components inside the spacecraft is to match the temperatures of the external panels first to remove potential sources of errors from the TVAC environment and treat them as a new set of BCs. The internal conductances and radiation can therefore be verified based on the matched conditions from the enclosure.

Additional boundary conditions of the environment also needed to be modelled. Along with the manually calculated and iterated input heat sources, the background temperature of the chamber also needed representation, which acts as an 80 K with a near black body emissivity.
Additionally, the different slews needed to be modelled as their own simulations, which would involve starting the simulation at an initial temperature and slew it to the desired temperature while logging the slew rates.

### 2.5.3 Validating Batteries

The batteries onboard are a pack of seven lithium-ion batteries which have their own flexible heater mounted underneath them. The effectiveness of the heater is critical to ensure they can function through the various conditions on-orbit. The batteries were validated during unit testing in the TVAC chamber where the external panels were kept constant at 20°C. The unit test for the batteries involved turning on the heater for a set amount of time while keeping everything else consistent as to not introduce any other variables. Throughout the validation process, various properties were verified due to an inconsistency in the model with the battery temperature measurements. This led to verifying the conduction of the conductive gap filler between the heater plate and the batteries, the emissive properties and interactions of the battery pack along with the avionics bay thermal tapes. In addition, a study was done to explore the possibility of the battery pack expanding, thus increasing conduction to the highly isolating Delrin material of the battery pack. The method for verifying were to first see the potential difference from a simplified hand calculated model and compare that with manual overrides to the model. Extra steps were added by checking extreme values such as perfect conduction with the internal tray or acting as a perfect black body for internal radiation. This proved to be good practice as these considerations and their extreme variations would highlight the dominant heat transfer mechanisms measured and ideally ensure accuracy in the modelled through its thermal connections and capacitances. Ultimately, the specific heat was the source of the inconsistency as the battery’s specification datasheet did not include these properties and the original source was incorrect where the real specific heat was much higher than previously thought. The results for the study can be seen in Figure 29 where the battery temperature with the heater on matches the simulation within 0.5°C. With everything discussed and considered, the results show a near perfect match between the flight model performance and simulated results. The result of this change to the orbital simulations resulted in the batteries keeping warm enough in every cold case, and thus relaxing the tight range of experienced temperatures to ultimately narrow on a specific tape scheme.
2.5.4 Payload Verification

The same unit level of testing was performed on the payload components. All the individual components of the payload and their conductances to the +Z panel have been specified from the manufacturers and so the validation was expected to be a close match. The unit testing conditions are with the lamps kept at a constant power such that the panels remain at a constant temperature, with the units powered on their temperature measured. As seen in Figure 30, the temperature of two payload components are measured and match closely with the simulated case. The small discrepancies are due to the components being modelled as 0D point masses where radiation from the unit is not captured directly. As mentioned in Section 2.3.2 about the payload modelling setup, the emissivity was captured as an average over the entire surface, however becomes more prevalent at higher temperatures. In the results, it is clear that the SSPA for example, experiences a larger temperature change due to it dissipating more power, which the physical unit is radiating some of that energy into the bus, thus showing a slightly cooler temperature. Whereas for the IMUX unit, the same radiative transfer is in effect, although not as substantial due to the lower temperature. It is clear from these results that the model is safe to use considering the added accuracy would not be necessary for solving the cold cases.
2.5.5 Passive Tank

Validating the propulsion tank from the TVAC test results was not similar to the other units due to the lack of any active heat dissipation from the tank itself. This complicates the process because the tank is passively influenced by all other aspects of the spacecraft and therefore also needed to be validated. Validating the tank is a systems problem so the cases explored were the hot slew and cold slew since they are the only test modes that are simple enough to avoid extra variables however dynamic enough to compare a rate of change for its temperature which will reveal much more about the system. Since the tank is tested without fuel, it is sensitive to anything significant in the system, such as the conduction of the panels amongst each other, or especially the internal panel where the tank and avionics are mounted. In these cases, internal radiation is now a driving thermal connection with that tank, along with the connections to its physical structure, and the properties of the tank itself. Since the tank is made and outsourced by another company, its thermal properties are proprietary and without specifications which consequently had to be deduced by our team. In retrospect for future cases with a similarly specialized propulsion tank, it would be best to do unit level thermal test to determine its properties, especially for components with restrictive temperature ranges.

The first part of the process is to match the change in temperature of the panels by comparing the measured values to the simulated values. Because of the rate of change, there were some adjustments that needed to be made to eventually match the data as needed. One of the changes on a system level for the bus was the assumed pressure from the external panels to the
internal support brackets, where because it is spread over a larger area, the overall pressure was in fact lower than every other torqued connection on the spacecraft. This was adjusted once comparing the results to show a difference of about 1°C on the internal panel which matched the measured data as seen in Figure 31.

Figure 31: Compared temperatures of the -X and internal panel during a hot slew

With all the panel measured temperature matching the simulated temperature, the variables left to validate the tank are minimized. With only these changes however, the temperature measured on the tank did not match the slew rates in both hot and cold, however comparing both it was clear the simulation shows the tank heating and cooling too slowly compared to the measured data. After exploring conductive links with the brackets and its interconnections, the change did not correlate with what was being observed. Eventually, the tank was remodeled to capture the thin carbon-fiber shell, where most of the thermal mass is located on the aluminum and steel mounting ends which protrude into the tank a fair bit. This results in the tank with the same total thermal mass, however redistributed so the ends are more capacitive, and the main body surface can gain or lose heat faster. This is the proper and more accurate representation of the tank as it was based on the known aluminum dimensions compared with the total mass and volume of the tank itself. The simulated effects from this change resulted in the tank losing and gaining heat far more quickly than the measured data. Conduction of the mating components was considered again but could not be justified to change so drastically by comparing with exaggerated values. Radiation was also explored since with the change it is more sensitive along its surface, however by varying the bus and tank emissivities to the extremes of what was possible, it was found that the emissive
tolerances were not the reason for the discrepancies because the observed differences were still too large. More research went into the tank itself where the carbon-fiber epoxy used could vary drastically in its internal conductivity depending on how it is made and what it is designed for [30]. This meant the explanation for the large deviations are due to the internal conductivity through the carbon-fiber composite and epoxy from the main body relative to the location of the thermocouple on the body itself. As seen in Figure 32, the thermocouple is placed on the underside of the tank.

The conductivity of the composite had a range of possibilities. With the possibility of a highly resistive composite, it made the position of the thermocouple on the tank much more significant where Figure 32 shows the hot slew takes much longer for the sensor to experience a change and similarly in the cold slew, the entire tank takes a while to lose its heat from the middle section. Note that in the hot slew case, the radiation from the panels add to the top surfaces of the tank which have been captured in the model. With the new information from a journal about carbon fiber composites [30], it was found that using the in-plane conductive properties of a low-conductive composite was able to match the measured results perfectly when accounting for the location of the thermal couple specifically. This is seen in Figure 33 where the hot and cold slews are nearly identical using the conductivity of $15 \frac{W}{K \cdot m}$.

Figure 32: Modelled tank temperature gradients during respective slews
Due to the tank being a passive system, it required solving more unknowns to validate. However, in the process also improved the accuracy of additional parts of the satellite and gained confidence in robustness of the thermal model. The changes made throughout all the validation conclusions were carried over to the final DAUNTLESS bus thermal model for this satellite. For the orbital simulations with all the changes described, the hottest tank temperatures decreased by 4°C where the average fuel temperature was able to stay within the limit without margin. There is still a discussion to be had about the exact corner case which is seen in Section 2.6.3 due to the tank not closing within margin for the inertially lock sun stare case for any of the -Z corners.
2.6 Final Design

The DAUNTLESS satellite was launched in November 2017 with its finalized design features. The spacecraft is balanced using almost exclusively passive systems with one active heater for the batteries and several thermal safety cutoffs for various components. It features a large antenna payload along with a propulsion tank inside for deorbiting after EOL. The system required a keep away definition for its attitude control and has a few temperature safety cutoffs programmed into the OBC. This was to prevent the three-axis attitude control from being commanded, either intentionally or accidentally into those orientations. Extra verifications were taken to ensure any attitudes that cause risk thermally are not stable and will tumble into thermally acceptable attitudes. DAUNTLESS pushed the boundaries of the microspace philosophy at SFL but managed to follow through with a working design.

2.6.1 Keep Away

The system required two keep away regions where the spacecraft should not point towards the sun within an angle defined in its cones. This definition can be seen in Figure 34 where both the +Z and -Z attitudes are specified. This is due to the payload of this specific mission where its large reflectors shadow most of the bus, and its electrical components are mounted on an external surface. The result of this is the shadowing decreases the bus temperature substantially, causing the avionics components on the opposite end to get too cold, and due to the externally mounted components of the payload, they can get cold much faster, not allowing heat from the opposite side to reach it adequately. The angles were defined such that if inertially locked at the specified angles with the smallest projected area, the opposite side of the spacecraft would not drop below the cold temperature limits during the WCC conditions. As explained in Section 2.4.4, the platform gets too cold on the opposite side due to two reasons. One is where the +Z panel facing the sun will cause the main reflector to shadow a majority of the satellite, resulting in the avionics bay getting too cold. The angle is defined at 45° and is large to overcome the shadowing effect where the bus absorbs enough heat in Safehold mode to keep within the operational temperature limits. The second reason is due to the payload components being directly mounted to a panel that is exposed to space where the -Z panel exposed to the sun alone is not enough to keep the payload warm, especially given the larger bus size of the spacecraft. The angle is defined at 25° where
exposing the any side panel will keep the payload within the operational limit. These definitions reduce the amount of keep away angles to only two angles and do not impose on the limits of the other components as the tape scheme design narrowed the expected temperatures as much as possible.

Figure 34: Sun stare keep away attitudes

There were plenty of tumbling cases that were modelled and simulated. From the expected satellite deployment kickoff attitude tumbling rates, the results were very similar to the nominal cases as heat is distributed relatively evenly along most of the surfaces, depending on the type of tumble. Considering the keep away attitudes with a tumble, it was determined that a slew of less than 0.03 deg/s where the keep away faces rotated past the sun vector would potentially cause the lower cold limits to exceed. The case where the satellite is spinning along the Z axis in both cases is the only relevant one to consider due to the magnetic dipole of the spacecraft. This natural dipole is biased and will cause the spinning attitude to be unstable and precession will cause more surface area to become visible and quickly move out of the attitude within one orbit. As necessary precautions, temperature sensors were added to the tank and payload components where if the cold limit was approached, the reaction wheels would immediately turn off and dump their momentum into the bus causing it to safely tumble.
2.6.2 Tape Scheme

The final tape scheme was eventually concluded prior to the launch of the DAUNTLESS bus. The process involved altering combinations of different tapes to narrow and shift the experienced temperature range the bus would experience to close the design. The process involved over four-hundred simulations with sixteen tape schemes being considered. A limitation for considering tape schemes are that thermo-optical properties cannot be chosen independently and must be considered as a combination of real tapes and their combined properties. This means that there are cases where a lower absorptivity might be desired, but not without also influencing the emissivity properties as well. There are clever techniques to combine other tapes with extremely biased properties, but they all have their limits. A finalized summary of each face is seen in Table 5 although note the +Z face does not include the reflectors and are measured relative to the panel and tapes itself. The primary emissivity and absorptivity across all the surfaces are due to the solar cells where all the tapes themselves are chosen to minimize both as much as possible. It is clear for both the +Z and -Z faces, the tape schemes differ the most from the total average.

<table>
<thead>
<tr>
<th>Surface</th>
<th>Final Optical Properties</th>
<th>Pre-TVAC Optical Properties</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Absorptivity</td>
<td>Emissivity</td>
</tr>
<tr>
<td>+X</td>
<td>32.9%</td>
<td>30.8%</td>
</tr>
<tr>
<td>-X</td>
<td>32.9%</td>
<td>30.8%</td>
</tr>
<tr>
<td>+Y</td>
<td>33.1%</td>
<td>30.9%</td>
</tr>
<tr>
<td>-Y</td>
<td>34.2%</td>
<td>32.1%</td>
</tr>
<tr>
<td>+Z</td>
<td>23.4%</td>
<td>4.0%</td>
</tr>
<tr>
<td>-Z</td>
<td>37.0%</td>
<td>33.6%</td>
</tr>
</tbody>
</table>

The changes the design underwent post TVAC testing can also be seen in the table where the entire system was tuned after validating the batteries and the propulsion tank in the model. As a result of this, the X, Y and +Z panels had their absorptivity decreased to favor a cooler tank in the WCH conditions. The actual patterns played a role in the case of the +Z panel where the payload antenna is mounted. Seen in Figure 35, the +Z panel has a hexagonal pattern with the gold and silver tape, which is due to the shadowing effects of the reflectors. This was necessary as decreasing the emissivity of the entire surface, where only the gold corners are higher absorptivity to maximize warming the bus during the WCC conditions and minimize overheating when the sun is angled below the reflector from an upper corner during WCH. The balance was optimized with high
absorbing black tapes on the edges so the avionics cold cases would close. Additionally, on the lower -Z panel, a small patch was added to the silver tape in the center to tune the worst-case conditions where the higher emissivity benefited the propulsion tank from overheating. As a result of these changes, the system was adjusted to meet the requirements within the limits of this microsatellite.

![Image](image.jpg)

*Figure 35: Final tape scheme for the DAUNTLESS bus*

### 2.6.3 Results

For nominal operations in a nadir tracking attitude, the satellite is expected to function well within the operational ranges of all the individual components. This can be seen in Figure 36 where all the simulated temperatures that are expected are plotted compared to the operational limits of the components with added 5°C margins to account for possible modelling errors. All the cases simulated here are described in Section 2.2 where the nominal cases are nadir tracking. It is clear all the thermally restrictive components of the spacecraft are within their limits, as seen with the batteries, the propulsion tank and for all the payload components.
Figure 36: DAUNTLESS bus nominal simulated results
Figure 37: DAUNTLESS bus worst case simulated results
When considering all the worst cases as defined in Section 2.2, it was important to balance as much as possible within the limitations of the design. The final thermal results are seen in Figure 37 with respect to the temperature limits and with the thermal uncertainty. For these cases, the survivable limits were used for the payload lower limit due to their restrictive requirements while mounted to an external panel, which is warranted with a thermal cutoff and is only expected to operate during nominal conditions. The payload cannot be powered on due to the logic of the temperature safety controller, where during nominal conditions, the payload is always within its limits. The keep away attitudes are captured in these results as well where the coldest temperatures of the X and Y axes sun stare WCC temperature are approximately the same as the angled +Z and -Z angled cases with the smallest projected area, as defined from 2.6.1. Note that the tank propellant temperature and some payload components extend past the thermal uncertainty margin. This is due to the challenge of closing a bus this size where a solution for one will cause a problem for the other. In this case, a conscious choice was made to not risk the payload cold margin and so a reduced emissivity causes the whole bus to be warmer during the WCH conditions, which is the defining case that pushes the tank beyond the uncertainty margin. The reason these results are adequate is both of these systems have a temperature sensor with a safety procedure. If the tank and payload are measured to be too high or low respectively, the spacecraft is programmed to immediately shut off the reaction wheels causing them to dump their momentum into the structure and force a tumble. All the possible tumbling cases yield similar results to the nominal cases.

2.6.4 Orbital Drift

Due to the nature of the accuracy required for a true SSO, launch providers always offer a tolerance for the orbit needed. In the case of this DAUNTLESS satellite, the tolerance was an inclination of 99.2° to 99.8° and so a precession of the SSO could cause an LTAN change, and as a result a Beta angle change. The resulting Beta angle differences are summarized in Table 6 where assuming the worst inclination from the tolerance and after a year of orbital precession. Even with orbital insertion error, the beta angle changes would not be seen until a year after the mission starts, and throughout its EOL. The resulting differences for WCH are about 5°C increase for all the components and for WCC are a small difference of less than 1°C. The actual approach for handling the spacecraft during the mission is to get an accurate orbit once launched rather than a tolerance range, and then predict the final effects on the mission.
# Table 6: Resulting beta angles after one year of a drifting SSO

<table>
<thead>
<tr>
<th>Case</th>
<th>LTAN</th>
<th>Inclination Angle</th>
<th>Beta Angle</th>
<th>Temperature Change</th>
</tr>
</thead>
<tbody>
<tr>
<td>WCH Drift</td>
<td>15:55</td>
<td>99.80</td>
<td>60.0</td>
<td>All components are hotter by 5°C</td>
</tr>
<tr>
<td>WCH</td>
<td>15:05</td>
<td>99.48</td>
<td>50.8</td>
<td>except the tank which is by 10°C</td>
</tr>
<tr>
<td>WCC</td>
<td>15:05</td>
<td>99.48</td>
<td>35.5</td>
<td>All components are slightly colder</td>
</tr>
<tr>
<td>WCC Drift</td>
<td>14:22</td>
<td>99.20</td>
<td>26.4</td>
<td>by less than 1°C</td>
</tr>
</tbody>
</table>

## 2.6.5 Closing Remarks

The thermal design, analysis and completion of the DAUNTLESS bus was a challenging and satisfying accomplishment. Ideally, the inclusions of keep away attitudes would not have been necessary due to the risk of adding complexity to the mission, however, stems from the inclusion of the specific payload rather than the DAUNTLESS bus directly.

For future missions using this bus, there are some fundamental design oversights that need addressing to ensure the thermal design is robust enough for any situation. For example, the payload being mounted to an external panel directly contributes to the -Z keep away angle due to the WCC conditions. This could potentially be bypassed if the payload was mounted in a similar way to the internal panel, except with a payload tray, possible fixed to the side brackets as to avoid being well coupled to a single panel. This would very effectively increase how long the payload will take to cool down due to the decoupled nature of the internal panels to the external surfaces. The same can be said if the reflector did not shadow the bus, the avionics would likely be able to stay warm enough as was the case with any of the side inertially locked WCC conditions. The tank can also be isolated to a greater extent, by using stainless steel or titanium support legs would bring the maximum temperature down 2°C during the WCH conditions. The accuracy of the tank could be improved by directed unit level tests to determine conductivity and thermal capacitance before the unit is integrated.

With that being expressed, the DAUNTLESS platform is a versatile structure that offers plenty of room and capabilities to support a wide range of missions. Knowing more about the attitude designing for a specific artificial during a random tumble and even magnetic dipole would be beneficial to adding more confidence if keep away zones are to be kept for future iterations.
Chapter 3

GHGSat

GHGSat is a greenhouse gas monitoring satellite from GHGSat Inc. from Montreal. The satellite as a whole is a standard microsatellite size at SFL where its mass and volume are about 13 kg and 23.6 dm$^3$. The plan is to have a constellation of these satellites that offer high precision detection of CO$_2$ and methane in the atmosphere and offer a greenhouse gas measuring service for organizations around the world [31]. Previously in 2016, the first satellite was launched from the company as a demonstration of the technology and it proved the feasibility of the mission. Moving forwards the company is continuing with their plan for a constellation of the same concepts with expanded functionality.

3.1 Mission Overview

The GHGSat mission is to measure point sources of greenhouse gases as a service for various companies. This will be important as many countries adopt a carbon tax to combat climate change and will need more accurate ways of sourcing the emissions. The plan for a constellation is to offer greater areas of coverage as the satellite is limited to the equivalent of an LTAN from about 9:00 to 15:00 due to the sun being directly above the earth’s surface for ideal measuring conditions. The satellite is able to measure with a resolution of 50 m by carefully measuring the refraction caused by the density of gases [22]. This is achieved with an IR camera that uses a telescopic aperture through a series of mirrors to magnify the image and process through an onboard spectrograph for finer details. IR is ideal because the greenhouse effect exists specifically because those gases absorb IR emitted from the earth, so being able to measure variations along the surface gives means to the measurements.

SFL was contracted to develop the supporting bus for this payload, as was the case with the previous satellite. Many of the mission elements and design features have already been determined with the first build, GHGSat-D. It employs the same standard design where the avionics components, including the MPS system are bundled together, along with the attitude
control system (ACS) and OBC are mounted to the same tray, and the payload is separated into its own module. This mission, although the same conceptually is changing and upgrading many components and units onboard. To combat radiation damage to the IR camera, it will undergo annealing to repair its photosensitive component intermittently throughout the mission. This is done by heating the unit to the point where the sensor will begin annealing, although more power is required because the IR camera itself is coupled to a radiator due to the need to isolate the camera from the rest of the bus. The damage caused is known as hot pixels and the same annealing process was successfully performed on the Hubble space telescope as seen in Figure 38 where annealing is performed roughly every month. It is important to note that annealing will not extend the lifetime of the camera indefinitely, but will significantly increase its lifetime, something GHGSat-D is limited by.

![Figure 38: Annealing results onboard the Hubble space telescope [32]](image)

Additionally, there is a second payload onboard that consists of an optical-downlink using a class IV laser. This laser is capable of downlinking up to 1Gbps of data which requires high precision to maintain the contact with the ground station during nadir tracking. Due to the high precision needed, the unit is also a star tracker to benefit from the rigid body of the single device.
3.1.1 Mission Requirements

The driving requirements for the mission are similar to other missions at SFL. They are listed in Table 7 where the primary requirements refer to keeping all components within their operational temperatures, and to absolutely keep everything within non-operational limits where the survival temperatures of the components match the non-operational limits. The non-operational temperature specifies the range where the component will not function fully and avoid permanent damages if powered, where the survival temperatures specifies the storage temperature without powering. An additional note is the mission time of 3 years where any tape degrading effects must be accounted for its EOL state.

<table>
<thead>
<tr>
<th>No.</th>
<th>Description</th>
<th>Rationale</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>General Requirements</td>
<td></td>
</tr>
<tr>
<td>THM-R001</td>
<td>The thermal control subsystem shall ensure that all units and payloads are kept within their specified non-operational temperature limits at all times.</td>
<td>To ensure that all units on the spacecraft are inherently safe at all times, even at initial power on when temperature of the unit may not be known.</td>
</tr>
<tr>
<td>THM-R002</td>
<td>The thermal control subsystem shall ensure that all units and payloads are kept within their specified operational temperature limits during nominal operations.</td>
<td>To ensure that all units are operational during nominal operations when all units on the spacecraft are required to perform nominally to support payload operations.</td>
</tr>
<tr>
<td>THM-R003</td>
<td>Thermal control measures shall satisfy requirements for a minimum duration of 3.5 years.</td>
<td>Thermal control surfaces (tapes or coatings) may degrade over time. This requirement specifies EOL conditions assumed in thermal analysis. Duration derived from 3 year service life plus commissioning.</td>
</tr>
<tr>
<td>THM-R004</td>
<td>On the hot end of a thermal range, thermal control margins shall be at least 10°C for the battery and at least 5°C for everything else.</td>
<td>Standard SFL practice.</td>
</tr>
<tr>
<td>THM-R005</td>
<td>On the cold end of a thermal range, thermal control margins shall be at least 5°C.</td>
<td>Standard SFL practice, battery margin relaxed to help bias spacecraft and payload toward cool end of range.</td>
</tr>
<tr>
<td>THM-R006</td>
<td>Thermal control measures should be passive.</td>
<td>To conserve mass and power and to limit design complexity. Requirement is 'should' because battery heaters are often required.</td>
</tr>
<tr>
<td></td>
<td>Specific Control Requirements</td>
<td></td>
</tr>
<tr>
<td>THM-R101</td>
<td>The thermal control subsystem shall ensure that each cell in the battery pack is within 2°C of every other cell in the pack.</td>
<td>To ensure even charging, discharging and aging of cells.</td>
</tr>
<tr>
<td>THM-R102</td>
<td>The thermal control system shall maintain the solar cells under 80°C during nominal nadir observing operations.</td>
<td>To ensure power generation consistent with what is assumed in the power budget.</td>
</tr>
</tbody>
</table>
The individual unit temperatures are the boundaries for the thermal analysis of the satellite. They can be seen in Table 8 where additional components have been included since GHGSat-D. The primary differences that drive additional resources to optimizing the thermal design are with the star tracker and optical downlink hybrid which has a small upper limit despite being exposed to space. In addition is the recently changed auxiliary camera that has a low temperature limit without a heater unlike the battery which supports its own heater system. These are requirements that will drive solutions to be made throughout the development of this spacecraft.

### Table 8: Temperature requirements for components on GHGSat-C

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>C&amp;DH</td>
<td>OBCs</td>
<td>-30</td>
<td>-20</td>
<td>60</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>SIB</td>
<td>-30</td>
<td>-20</td>
<td>60</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>Firecode Detection</td>
<td>-40</td>
<td>-40</td>
<td>65</td>
<td>85</td>
</tr>
<tr>
<td>Power</td>
<td>Solar Cells on Panels</td>
<td>-50</td>
<td>-50</td>
<td>80</td>
<td>80</td>
</tr>
<tr>
<td></td>
<td>Battery</td>
<td>0</td>
<td>0</td>
<td>60</td>
<td>60</td>
</tr>
<tr>
<td></td>
<td>MPS</td>
<td>-30</td>
<td>-20</td>
<td>60</td>
<td>70</td>
</tr>
<tr>
<td>ADCS</td>
<td>Star Tracker / Optical Downlink</td>
<td>-40</td>
<td>-20</td>
<td>40</td>
<td>50</td>
</tr>
<tr>
<td></td>
<td>Sun Sensors</td>
<td>-30</td>
<td>-25</td>
<td>65</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>Other Sensors</td>
<td>-30</td>
<td>-20</td>
<td>60</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>Reaction Wheels</td>
<td>-30</td>
<td>-30</td>
<td>60</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>Smart Torquers</td>
<td>-40</td>
<td>-30</td>
<td>70</td>
<td>80</td>
</tr>
<tr>
<td></td>
<td>GPS Receiver</td>
<td>-30</td>
<td>-25</td>
<td>60</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>GPS Patch Antenna</td>
<td>-55</td>
<td>-55</td>
<td>85</td>
<td>85</td>
</tr>
<tr>
<td>Communications</td>
<td>UHF Rx</td>
<td>-30</td>
<td>-20</td>
<td>60</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>UHF Antennas</td>
<td>-65</td>
<td>-65</td>
<td>60</td>
<td>60</td>
</tr>
<tr>
<td></td>
<td>S-Band Tx</td>
<td>-30</td>
<td>-20</td>
<td>60</td>
<td>70</td>
</tr>
<tr>
<td></td>
<td>S-Band Antennas</td>
<td>-55</td>
<td>-55</td>
<td>80</td>
<td>80</td>
</tr>
<tr>
<td>Payload</td>
<td>Primary Units</td>
<td>-40</td>
<td>-40</td>
<td>60</td>
<td>80</td>
</tr>
<tr>
<td></td>
<td>Auxiliary Camera*</td>
<td>-25</td>
<td>-5</td>
<td>45</td>
<td>60</td>
</tr>
<tr>
<td></td>
<td>Optical Downlink CPU</td>
<td>-40</td>
<td>-40</td>
<td>60</td>
<td>85</td>
</tr>
</tbody>
</table>

### 3.2 Boundary Conditions

The design of GHGSat-C has not been concluded during the time of this thesis. Presented here are the leading updates captured through the thermal analysis for a previously targeted orbit. The updated spacecraft itself can be seen in Figure 39 with axes for reference. The bus size is approximately 43 cm by 27 cm by 20 cm with the baffle of the telescope in the -Y face which is
used for nadir tracking and the star tracker mounted to the +Z upper tray, which is not coupled with the +Z panel. The current state will be used throughout the discussions about configurations, models and any relevant results.

Figure 39: GHGSat-C CAD with internal components and reference axes

The BCs considered for the spacecraft are shown in Table 9 where it is a non-exhaustive list of the main modes of operations. This shows the envelope that was used to capture the conditions GHGSat-C would experience with a LTDN at 11:30. For the purposes of thermal analyses, LTAN and LTDN result with the same Beta angles, with the only difference being the orbit vector around the earth. Here because the LTDN is close to noon, the resulting beta angles are closer to zero compared with the range of orbits that GHGSat was designed to function with from the perspective of the payload. This results in a thermal design that is cold biased meaning using the same tape scheme for another orbit would result in a warmer bus, likely needing to be optimized for that orbit. The solar fluxes and Beta angles used for these cases are indeed the extremes recommended [3] which act as an extra layer of conservatism that can be trimmed if needed. The solar irradiiances here account for the closest and furthest approaches the earth’s orbit will experience along with the appropriate fluxes from the solar cycle. Since the bus is significantly smaller than the DAUNTLESS bus, and GHGSat-D has already successfully operated, it will likely not be necessary to reduce any conservatisms. More information to follow on the orbits, attitudes and internal heat dissipations in the following sections.
Table 9: BC for a GHGSat-C satellite

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>WCH</td>
<td>11:30</td>
<td>550</td>
<td>10</td>
<td>1414</td>
<td>0.29</td>
<td>232</td>
<td>Main-PL Operational</td>
<td>Max projected area absorptivity</td>
</tr>
<tr>
<td>Hot Nominal</td>
<td>11:30</td>
<td>550</td>
<td>10</td>
<td>1414</td>
<td>0.29</td>
<td>232</td>
<td>Operational</td>
<td>-Y Nadir Tracking</td>
</tr>
<tr>
<td>Cold Nominal</td>
<td>11:30</td>
<td>550</td>
<td>3</td>
<td>1322</td>
<td>0.22</td>
<td>230</td>
<td>Non-Operational</td>
<td>-Y Nadir Tracking</td>
</tr>
<tr>
<td>WCC</td>
<td>11:30</td>
<td>550</td>
<td>3</td>
<td>1322</td>
<td>0.22</td>
<td>230</td>
<td>Safehold</td>
<td>Min projected area absorptivity</td>
</tr>
</tbody>
</table>

3.2.1 Orbits

The orbits that GHGSat are designed for are most optimal with an LTAN or LTDN of 10:00 to 14:00 but can also accept some margin depending on the launches available. For the current status of the project, the thermal analysis was nearly completed for a LTDN of 11:30, but since has changed into two other possibilities of 10:30 at 630 km or 12:00 at 720 km. Due to this change, the presented thermal design is not optimized despite approaching a finalized state and some studies are performed with the orbital range rather than the specific orbit. The choices are limited to the launchers available and requires closing the design for the finalized orbit.

3.2.2 Attitude

The attitude used to define the WCH are significant for GHGSat-C. This is due to the uneven geometry of the spacecraft. Using the methods described in Section 1.5.3 for determining the worst-case projected area to the sun, it was determined which vectors for each corner of the spacecraft would result in the hottest average temperature. The WCH attitudes tend to face towards the X or -Z faces with the +Y face because they have the most solar cells visible to sunlight. The +Z face is not as prevalent as the radiator on the surface takes up a significant portion of the surface reflecting any potentially absorbed light. Due to the satellite being asymmetrical, the WCC conditions could potentially be much colder than with larger buses due to the smallest surface areas which are the Y faces in this spacecraft. This is normalized by adjusting the thermal optical tapes through the design of the satellite.
### 3.2.3 Heat Dissipation

The heat dissipated within the spacecraft comes from all the components that are active for each respective power mode. For GHGSat-C, the list of units and their duty cycles can be seen in Table 10 with respect to the two primary modes of operations. There is also a non-operation mode and an annealing case not captured in the table. Non-operation is a mode that is used to explore the nominal cases for WCC conditions where most units are active in an idle state, except for both payload units, where there are no IR measurements or optical downlinking in action. There is also the WCH conditions with the worst-case attitudes that the payload is modelled to operate whereas the units are planned to be off entirely. The optical downlink is not described as a main payload and is therefore not considered for these WCH attitudes. This is due to the optical downlink having its own temperature safety cutoff where the operation will cutoff once a temperature threshold is reached. Added safety will be present as a system level shut off will occur with the same condition. This is necessary as for the WCH conditions, the secondary payload is exposed to direct sunlight and is sensitive to heat while operating and is only practically functional when the laser is pointed to a target ground station via nadir tracking.

**Table 10: GHGSat-C component power consumption [33]**

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Components</th>
<th>Power [W]</th>
<th>#</th>
<th>Nominal Operational</th>
<th>Safehold Mode</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td>Duty Cycle %</td>
<td>Orbit Average [W]</td>
</tr>
<tr>
<td>C&amp;DH</td>
<td>OBC 1</td>
<td>0.550</td>
<td>1</td>
<td>100%</td>
<td>0.550</td>
</tr>
<tr>
<td></td>
<td>OBC 2</td>
<td>0.550</td>
<td>1</td>
<td>0%</td>
<td>0.000</td>
</tr>
<tr>
<td></td>
<td>Firecode Detection</td>
<td>0.100</td>
<td>1</td>
<td>100%</td>
<td>0.100</td>
</tr>
<tr>
<td></td>
<td>SIB</td>
<td>0.030</td>
<td>1</td>
<td>100%</td>
<td>0.030</td>
</tr>
<tr>
<td>Power</td>
<td>MPS*</td>
<td>0.440</td>
<td>1</td>
<td>100%</td>
<td>0.440</td>
</tr>
<tr>
<td></td>
<td>BIM*</td>
<td>0.150</td>
<td>1</td>
<td>100%</td>
<td>0.150</td>
</tr>
<tr>
<td></td>
<td>Battery Heater</td>
<td>0.800</td>
<td>1</td>
<td>0%</td>
<td>0.000</td>
</tr>
<tr>
<td>ADCS</td>
<td>GPS Receiver</td>
<td>1.100</td>
<td>1</td>
<td>100%</td>
<td>1.100</td>
</tr>
<tr>
<td></td>
<td>Reaction Wheels*</td>
<td>0.300</td>
<td>4</td>
<td>100%</td>
<td>0.900</td>
</tr>
<tr>
<td></td>
<td>Sun Sensors</td>
<td>0.150</td>
<td>6</td>
<td>10%</td>
<td>0.090</td>
</tr>
<tr>
<td></td>
<td>Rate Sensor</td>
<td>0.330</td>
<td>1</td>
<td>100%</td>
<td>0.330</td>
</tr>
<tr>
<td></td>
<td>Magnetorquers*</td>
<td>0.400</td>
<td>6</td>
<td>100%</td>
<td>1.200</td>
</tr>
<tr>
<td></td>
<td>Magnetometer</td>
<td>0.045</td>
<td>1</td>
<td>100%</td>
<td>0.045</td>
</tr>
<tr>
<td></td>
<td>Star Tracker</td>
<td>0.600</td>
<td>1</td>
<td>100%</td>
<td>0.600</td>
</tr>
<tr>
<td>Comms</td>
<td>UHF Rx</td>
<td>0.170</td>
<td>1</td>
<td>100%</td>
<td>0.170</td>
</tr>
<tr>
<td></td>
<td>S-Band Tx*</td>
<td>5.000</td>
<td>1</td>
<td>5%</td>
<td>0.250</td>
</tr>
<tr>
<td>Primary Payload</td>
<td>Camera Units*</td>
<td>12.000</td>
<td>1</td>
<td>40%</td>
<td>4.864</td>
</tr>
<tr>
<td></td>
<td>Annealing Heater*</td>
<td>6.000</td>
<td>1</td>
<td>0%</td>
<td>0.000</td>
</tr>
<tr>
<td>Secondary Payload</td>
<td>Optical Downlink*</td>
<td>8.000</td>
<td>1</td>
<td>5%</td>
<td>0.404</td>
</tr>
<tr>
<td></td>
<td>Processing Unit</td>
<td>5.000</td>
<td>1</td>
<td>5%</td>
<td>0.251</td>
</tr>
</tbody>
</table>
These modes are effectively captured in the transient analyses by inputting their time of operations throughout the relevant orbit. This includes the time it takes for the batteries to charge after exiting eclipse and transitions among the various modes. There are also differences within the payload as during the sunlit time of the orbit, the payload is expected to function throughout most of this time, where it might have multiple targets and components turning on and off as needed. These are also captured at a specific time internal and implemented as a worst-case situation where the peak of the loads are timed to be when the satellite bus is the hottest average temperature. There is also the annealing case which is effectively a non-operational case with power to the IR camera heater with a thermostat. The annealing temperature for the unit is specified as 80°C which is not a direct measurement of the photosensitive elements themselves, but rather indirect measurements from the camera shell.

### 3.3 Modelled Design

The internal layout of GHGSat-C is separated into a few compartments that are integrated together to make a working system. The layout for the bus can be seen in Figure 40 where the avionics can be referred to on the upper or lower level, and likewise with the payload where the upper section is referred to as the mezzanine. The star tracker and optical downlink are also part of the payload components but functions independent of the greenhouse gas detecting mission. Internal radiation is considered for the upper tray, due to the accuracy needed for the limits imposed by the optical downlink and the auxiliary camera. The system was modelled using NX 11.0 [27] and the approach for the current state of the design are discussed here.

Figure 40: GHGSat-C internal CAD and thermal mesh respectively
3.3.1 Primary Payload

The payload of the spacecraft primarily uses the IR camera mounted with its telescope to measure the density of IR light over the observed surface. This has been largely carried over from the previous satellite as it worked very successfully. The module can be seen in Figure 41 along with the thermal model associated with the design. The system works by allowing IR to pass through the baffle and lenses, then bounce from the mirror and through a series of units to measure the change in refraction at various frequencies centering on a reference frequency. Most of the components in the payload module are modelled as 2D meshes that capture the thermal mass of the main structural components. Both cameras are modelled as 0D nodes with their respective enclosures modelled as 2D meshes. The enclosures also combine various components such as lenses and vanes as point masses, which are conducted to the meshes through their fasteneners. In the case of the main payload lower tray, most of the vanes are not geometrically modelled as internal radiation in the lower section is not critical to compute.

Figure 41: GHGSat-C greenhouse gas detecting payload and its thermal mesh

The payload module also has a requirement to limit the amount of light bleeding into the camera enclosures to ensure the instruments are practically functional for the mission [34]. One of the mitigation tactics used are the vanes that are spaced throughout the subassembly of the main telescope. For the IR camera itself, it is isolated from the rest of the sub-assembly to function as consistently as possible. The feasible way of ensuring the powered camera will not overheat in this situation is to couple it with a radiator that is mounted in the mezzanine and isolated with Delrin and stainless-steel standoffs. Since the IR camera will be annealing at 80°C, the radiator imposes
a problem for this process. Radiators are essentially highly conductive metals that have surface properties to reflect most sunlight but emit (and absorb) IR very effectively. Because the camera will be annealing for days at a time, it is important to capture the heat flow from the unit to the radiator itself. The view factors of the radiator are also calculated and the power needed from the heater are adjusted accordingly. The reason the view factors are important is that the star tracker is now protruding from the surface (unlike GHGSat-D) which can cause some IR to be exchanged from the two units. Ideally, the heater will be enough to keep the IR camera warm enough for annealing, even during the WCC conditions. For the power systems in the bus to support this, no other payload functions will be active during this operation, which still results in a system that remains power positive in sunlight.

Another important addition is the use of the auxiliary camera for detecting clouds and aerosols. Specifically, the current design requires one with a limited lower temperature limit of -5°C. This is the consequence of a design change aiming to improve the functionality of the camera. The barrels used to feed light to the camera have been modelled externally to capture the radiation the system would have with the rest of the mezzanine and the upper tray. The camera and internal subassembly components have been modelled as point masses with the appropriate conductances to the barrel itself.

3.3.2 Secondary Payload

The optical downlink is mounted near the mezzanine of the payload. Although the unit is operating as a low power star tracker most of the time, when the laser is commissioned to data transfers, it consumes up to 8W of power. Most of that energy is dissipated into the structure where about 1W is emitted through the laser diode back to the ground station. There are a few modelling considerations for this unit, mainly due to the fact that it is high power, and partially exposed inside of the bus and to the space environment simultaneously. How much the unit was exposed was a careful systems design choice where a little more than half of it is recessed into the external panel. As noted previously, the unit is not at all coupled with the external panel and is instead mounted to the upper tray of the bus. This results in more stable temperature fluctuations, but also limits the amount of heat dissipation is possible exclusively through conduction. These factors required the unit to be modelled accurately to capture the separate optical downlinking and star tracking
capabilities. The process can be seen in Figure 42 where the high-power amplifier of the laser diode is represented on the right of the unit and applied in the thermal model. The meshes capture the thermal masses separated by each side of the components. The overall assembly is either fastened with direct aluminum contacts of the material, or through electromagnetic interference (EMI) brackets between the various shell components.

For the surface properties, the module can have tapes applied to everything except for the aperture of the star tracking unit. To prevent the high-power unit from overheating, high emissive tapes are ideal, however careful balancing is needed due to the lower limit where that solution could cause the unit to freeze. Similarly, the bracket that couples the unit with the bus also needs to be captured accurately as it will drastically change the unit’s effective temperature when powered off due to the sensitive nature of it being exposed while fixed internally.
3.3.3 Avionics Configuration

The avionics in the upper tray are also part of SFL’s new MPS components that are being tested for future missions. The units use the same mounting holes which is great for integration, and most of the PCBs have been redistributed to convert voltages and route power through different cards. More of an effect are the additional layers of the PCBs which have been updated and included in the model via their thermal masses and internal conductances. All the PCBs on GHGSat-C are modelled to the detail of the layers within the components. This is to capture the in-plane conduction of the copper layers of varying thicknesses, and to distinct it from the through-plane conduction that goes through layers of the copper and FR4 of the PCB materials. With added conservatism to any assumptions, this was done in an effort to capture any potential problems the new system might impose. Currently this system is currently being thermally tested in the TVAC and the temperature measurements will be able to provide more accurate internal conduction.

3.4 Studies

Throughout the development of the satellite, various verifications and design choices needed to be made. The newest mode of operations added to the system is for annealing the main camera to prolong the lifetime of the device. This needed to be verified as the camera is decoupled from the main payload structure and bus but coupled with a radiator. Furthermore, a new star tracker with the high-power optical downlink is added to the spacecraft, which went through a series of design changes that all needed to work in the spacecraft thermally. Most recent is the new auxiliary camera that needs to be qualified to work at colder temperatures than the ones the manufactures have specified. Their design paths are explored in the following sections.

3.4.1 Annealing

The IR camera needs a heater placed somewhere feasible that integrates well with the previous design. The maximum power allocated to the heater itself is 10W with a thermostat on the camera itself to maintain 80°C for the annealing process to be effective. There were a few proposed heater locations that would work, all revolving around various placements on the radiator itself. The most efficient location is the one that is closest to the unit itself as a lot of power will be radiated out to
the environment when in full effect. A case study was performed to verify the existing model with the version built by the payload providers. The ideal was to verify that the IR cameras were modelled correctly with respect to the applied heat source, and if different conditions such as modelling the on-orbit or in a constant environment setting would still yield appropriate results. As seen in Figure 43, the two models were used to verify each other using the 10W with the final heater placement and the properties of the radiator and camera. The primary difference seen is due to the accuracy of the braid of the radiator modelled. For the unit simulation, the full assembly, including the connection from the radiator to the unit, where for the orbital simulation, the heat load is applied directly to the unit. Due to how near the heater is to the camera, the expected difference is as small as seen in the graph. Realistically, the same conditions are verified for WCC where even with a reduced power load, the camera will go well beyond the target temperature without a cutoff limit. Additionally, the time until 80°C is not critical as much as the heaters will be operating for many orbits, long enough to ensure confidence that a difference of less than 60 seconds would not impede the device. Using these comparisons, the heater only needs to operate from 5W to 6W to achieve the desired temperature for the annealing process.

![IR Camera Heating Model Comparison](image)

**Figure 43: Annealing model comparison**
3.4.2 Optical Downlink

The star tracker is the same unit as the optical downlink on-board the satellite. Due to the precision of the laser, the electrical and mechanical components that support the laser are relatively sensitive to temperature fluctuations. The position of the unit within the spacecraft went through many design changes before setting into the current position. Each one of these design changes were modelled thermally to show how effective the temperature can be controlled on the unit. The variations were due to the interaction with the mezzanine where the auxiliary camera was still in a state of constant change. The laser and the star tracker each require a clear line of sight to their target and are angled opposing to each other by 125° which with a star tracking pointing in the +Z face, the laser is pointing in the -Y direction, exactly where the mezzanine is relative to the unit. The various changes include the star tracker being completely recessed into the bus, below the +Z panel, with a window for the laser through the mezzanine. Thermally this was not ideal as the heat dissipated by the optical downlink can only conduct into the bus, resulting in a 10°C above the operational limit. Whereas an elevated exposed unit would be able to radiate using high emissivity tapes rather effectively, enough to adjust the tapes to meet the limits. The final position of the unit is partially recessed into the bus, where about half of it is exposed to space, and the laser passes above the mezzanine rather than through it. The thermal results for this scheme after adjusting the thermo-optical properties on the unit can be seen in Figure 44 where the operational limits are between -20°C and 40°C. The maximum temperature experienced is 35.2°C for a WCH with the +X-Y+Z corner inertially locked to the sun and the minimum is -14.3°C with the +Y face inertially locked to the sun. The WCC condition here is possible due to the unit not in contact with the external panel exposed to space, hence it is able to keep relatively warm from the warmer internal tray despite the high emissivity tapes on the unit.
The nominal case is the only case where the unit is powered and active. This is due to the choice of not including an active unit during the WCH conditions due to the large temperature profile. The unit would reach close to 60°C with the highest emissivity tapes to cold bias the unit, which resulted with the WCC condition experiencing as low as -40°C. Both are well beyond the scope of using passive systems only to keep the unit within the operational and survivable range. The decision is further rationalized by two independent temperature power shut off switches that are linked to two temperature sensors on-board the unit given an accidental commission or likewise.

3.4.3 Auxiliary Camera

The auxiliary camera can detect clouds and aerosols which is part of the subassembly in the mezzanine. The camera is mounted via a threaded connection to the barrels that contain the vanes, a mirror and a lens to properly focus the image. The unit went through a series of design changes and is currently at the stage where the unit is being qualified for use on the mission. The manufacturers have specified the camera to have an operational range of -5°C to 45°C and a survivable range of -25°C to 60°C. The camera unit is currently undergoing qualification testing to ensure it will be able to operate beyond the specified temperatures. The test temperatures are based on the results shown in Figure 45 where the worst-case conditions are expressed relative to the possible orbits the final spacecraft will be launched into. Although the orbits are different, the
same tape schemes were used for each orbit, resulting in a conservative range. From simulating the differences, adjusting the tapes for each orbit would result in a 2°C bias that is within the margin of error for this setup. The study concluded that the expected survivable range would be from -17.8°C to 43.7°C resulting in testing from -25°C to 50°C. Similarly, for the operational range, the tests will be performed from -10°C to 45°C, which will verify if the unit is reliable during these temperatures. Note that in these results, the camera is off during the WCC cases and on during the WCH conditions. This explains the vast differences in the curves for the hot and cold cases where once activated goes through a series of active and idle functions as the series of operations are performed for the entire payload.

![Graph showing auxiliary camera temperature range](image)

**Figure 45: Auxiliary camera worst case temperature ranges experienced within all orbit**

The decision to qualify the unit beyond the specified temperatures are due to the nature of the WCC cold condition. As can be seen in Figure 46, with an inertially locked sun stare on the +Y face, the entire payload side of the spacecraft approaches -18°C. Additional case studies were performed to determine the feasibility of changing the design slightly to help meet the limits of the camera such as further isolated the unit from the chamber with peek standoffs and a Delrin structure, which only yielded a difference of 3°C, not enough to warrant a design change of this magnitude. Another option would be to add an additional active heater with a thermostat, but this option is questionable due to the limited power budget of the spacecraft.
Figure 46: GHGSat-C WCC +Y sun stare gradient with a uniformly cold payload

The final solution will add robustness to the function of the camera with a temperature safety control on the device as part of the bus itself. This will function as the other similar units where power will fail to be supplied to the unit when below a certain threshold that has yet to be defined. Using a temperature control safety cutoff is not an ideal solution for microsatellite missions, although it is the nature of the design when thermally sensitive components are well coupled to the external space environment. This will allow the unit to have a set of requirements for operations, and another for survivable.
3.5 Design State

Leading up to the orbit change, the thermal design was nearly finalized. The simulations were completed using the BCs shown in Table 9 and the thermo-optical properties used are averaged in Table 11. Optimizing the tape schemes is ideally the final step once the final orbit is selected and the other design elements are finalized. This generally involves adjusting the combination of tapes used which results in balancing the bus. The possible effects are narrowing the expected temperature ranges, or biasing them towards the cold or hot, possibly allowing for some flexibility.

Table 11: Current average thermo-optical properties for GHGSat-C

<table>
<thead>
<tr>
<th>Surface</th>
<th>Absorptivity</th>
<th>Emissivity</th>
</tr>
</thead>
<tbody>
<tr>
<td>+X</td>
<td>47.5%</td>
<td>42.7%</td>
</tr>
<tr>
<td>-X</td>
<td>47.5%</td>
<td>42.7%</td>
</tr>
<tr>
<td>+Y</td>
<td>79.7%</td>
<td>55.0%</td>
</tr>
<tr>
<td>-Y</td>
<td>66.9%</td>
<td>29.1%</td>
</tr>
<tr>
<td>+Z</td>
<td>55.9%</td>
<td>56.3%</td>
</tr>
<tr>
<td>-Z</td>
<td>56.7%</td>
<td>46.3%</td>
</tr>
</tbody>
</table>

For the nominal cases, most of the components, including the updated ones seem acceptable. The full plot of results is seen in Figure 47 where the resulting temperatures for each subsystem and component are shown with respect to the unit’s thermal operation limit. The only exception is the auxiliary camera which may get too cold during nominal conditions with 5°C of margin. As discussed previously, the camera is a work in progress and will undergo unit testing to confirm its functionality at -10°C. The simulated results are based on the nominal BCs described in Section 0 where all systems are active during the nominal hot case and idle during the cold case while nadir tracking a ground target.

The worst-case results show that this design continues to be inherently feasible regardless of the new components onboard the bus. The results can be seen plotted in Figure 48 where all the limits are shown as their operational limit with the exception of the auxiliary camera. As discussed previously, the auxiliary camera will have a temperature sensor that detects when the unit is past a threshold for when it gets too cold and will prevent the unit from powering on. This justifies the use of the survivable limits for the component for the WCC and WCH conditions. The simulated results are also based on the worst-case BCs described in Section 0 where all systems are active except for the optical payload during the WCH condition and the satellite is in Safehold mode during the WCC conditions.
Figure 47: GHGSat-C nominal thermal results for the LTDN 11:30 orbit
Figure 48: GHGSat-C worst case thermal results for the LTDN 11:30 orbit
Additionally, is the star tracker that goes 0.2°C too warm with margin, which is something that would likely be solved if this were the final orbit and the tapes would be optimized further. Throughout all the simulated cases and adjusting the tapes on the optical downlink itself, the results here just emphasize that the unit will feasibly meet the requirements as the project continues development. Some other components such as the new MPS configuration, the S-Band Tx and the secondary payload processor are all beyond their operational hot limit when including margin. For many of these cases, less than 1°C beyond the hot or cold limit is justifiable, although since the design is not finalized, and the tapes are not optimized, these components can still be made more accurate to actual meet their requirements if the updated changes do not solve these differences indirectly. In the case of the MPS components specifically, since they are newly updated with conservatism, their unit testing will be valuable to reduce the predicted temperature ranges. Another note is GHGSat-D used thermal straps on the MPS boards to dissipate heat more effectively which are not part of the analysis and results. From the ongoing TVAC testing, conductive thermal connections were deemed necessary and will be included in the final version.

3.5.1 Future Work

Stated throughout these sections, GHGSat-C is not yet finalized and these results are a snapshot of the continuous progress made with the thermal model results of the continuously updated structure and payloads. These results are convincing enough that the problems faced are feasible to solve, although still in development to implement. The thermal work needed going forwards involve validating the MPS units with the thermal results, implementing a safety temperature sensor on the auxiliary camera, and optimize the tapes for each of the two potential orbits (LTDN 10:30, 12:00). This design will likely be the first of the upcoming constellation so it is important to document the thermal components that are sensitive to changes between orbits, and the effective tapes that best optimize for each one.
3.6 Model Validation

Throughout developing the thermal model for GHGSat-C, the previous satellite GHGSat-D has been in orbit since its launch in June 2016. Thermal modelling has objective goals to attain the most accurate model while using the least amount of resources, however the details on how to manage the data and build the model vary extensively from person to person and mission to mission. It is useful to explore the modelling techniques used on the previous thermal model and compare the kind of accuracy that can be attained with any of the techniques that were adopted in recent year.

The satellite was previously validated as a first pass at SFL. The thermal model was indeed accurate enough to continue with future missions as all components were within 5°C of error from the measured and simulated results. When exploring the previous model in further details certain aspects were not entirely convincing, such as the conductance of the gap filler, the active internal radiative elements, and especially the capacitance of the same batteries used for the DAUNTLESS bus, GHGSat-D and soon to be GHGSat-C. This may have been validated through the satellite’s TVAC testing for acceptance however, the reasoning for the differences are not justified. The intention was to perform an on-orbit validation to conclude as much accuracy as possible and adjust the properties for the current mission.

The process for validating was also an exercise in understanding the boundary conditions with more details for variables in LEO. Ideally for modelling, an attitude must be chosen that remains constant for several orbits to reach a periodic thermal cycle, in addition to having as much data as possible about the time and orbit to match any external heat loads on the spacecraft. The process and conclusions will be discussed in this section.

3.6.1 Attitude

For choosing the right time to verify the on-orbit parameters, a few conditions needed to be met to avoid adding unnecessary variables to the system. One of those conditions is to have a relatively consistent attitude so that both the satellite and the thermal model achieve a steady thermal cycle that would lead to matched temperature results. The chosen attitude occurred on July 16th 2016 where the spacecraft was consistently performing a nadir tracking with the -Y axis towards the
earth. The onboard sun sensor data is plotted in Figure 49 where each axis of the spacecraft is shown as a vector relative to the sun. During nadir tracking, the Z axis is consistently negative to avoid the star tracker from being exposed to the sun, and the X and Y axes oscillate out of phase and eclipsed exactly as what is expected from nadir tracking.

Figure 49: GHGSat-D onboard sun sensor data and resulting vectors

The orbit and Beta angle were determined using the two-line element set (TLE) data after the spacecraft’s first month in orbit. There is a small difference from the measured sun angle relative to the spacecraft if it were truly nadir and the beta angle is determinable from the date of the orbit. Comparing the beta angle with the sun sensor data, an offset of 1.73° from nadir is inferred. This difference is enough to account for almost a 1°C temperature difference on the -Z external panel due to the total projected area of the spacecraft.

3.6.2 Boundary Conditions

Determining the BCs for this date is a significant part of the validation process as all the conditions here will drastically change the outcome of the external and internal temperatures of the thermal model. The best way to approach this is to match the external panels of the thermal model with the satellite’s temperature sensor, accounting for the location of the sensor itself. Once all the known BCs are determined, any other variables can be iterated on until the experienced temperatures match, and validation can continue relative to the internals of the spacecraft exclusively.
The solar irradiance is a flux that changes for a variety of reasons. Relative to missions in LEO, this is primarily due to Earth’s elliptical orbit around the sun being maximum solar irradiance at the perihelion and the weakest irradiance during the aphelion. To capture the solar flux from July 16\textsuperscript{th} 2016, data was used from the Solar Radiation and Climate Experiment (SORCE) missions. This data comes from a NASA funded mission that began in 2003 to collect accurate solar data using photosensitive sensors in the full spectrum of the sun, centered on visible light [35]. The data relevant for the date in question can be seen in Figure 50 where the solar irradiance is seen changing throughout the month. The date of July 16\textsuperscript{th} is near the aphelion of Earth’s orbit relative to the sun at July 4\textsuperscript{th} where the solar irradiance is nearly weakest throughout of the year. The solar irradiance for that date specifically is $1317.32 \text{ W/m}^2$. Something intriguing is the flux is less than the minimum fluxes described in Section 1.2.4 and recommended by the thermal control handbook [3]. Overall differences measured by the probe are about $7 \text{ W/m}^2$ less for both the WCH and WCC, which could be differences due to the time passing since the original data was used and the accuracy of the modern instruments. For the purposes of validation, this data is treated as accurate as it was used for matching the external panel temperatures within the thermal model since anything else is too hot. For current missions, this difference needs to be verified as it effectively means colder WCC conditions, which could be problematic for balancing the tape scheme on the surfaces.

**Figure 50: SORCE study measured solar irradiance from July 2016 [35]**

Once the simulated external panel temperatures of the spacecraft were all within a close range by using the updated solar irradiance, the Earth IR and albedo were the remaining external BCs to adjust in the simulations. Due to the need of a specific date for these values and the inaccuracies
associated with Earth IR and the albedo, they were iterated upon to match the external panels of the spacecraft. Knowing that the earth shares an inverse relationship with IR and albedo due to the seasons and IR flux is the strongest during the northern summer, it is possible to estimate the final values based on a table from the thermal control handbook as summarized in Table 12. Here the table is assembled to be used for worst-case conditions based on the probability that certain conditions could occur, and how sensitive the spacecraft is to albedo and Earth IR. This table is assembled such that based on the time constant of the satellite and different combinations of solar irradiance, albedo and Earth IR can experience conditions where each heat load will contribute to a possible worst-case condition. The time constant is described in Section 1.5.2 where it is an expression of thermal mass and the heat loads on orbit where for WCC and WCH describe different time constants such that the difference in solar irradiance would increase the amount of time it takes to reach 63% of steady-state, hence why two time constants are ordinarily used for these calculations. Furthermore, the surface sensitivity is determined by computing which loads results in the spacecraft changing in temperature more drastically, where based on the absorptivity and emissivity of the spacecraft could result in cases where the either the albedo, Earth IR or both contribute in a significant way to the change of the system.

<table>
<thead>
<tr>
<th>Surface Sensitivity</th>
<th>Time Constant [s]</th>
<th>WCC Albedo [W/m²]</th>
<th>Earth IR [W/m²]</th>
<th>WCH Albedo [W/m²]</th>
<th>Earth IR [W/m²]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Albedo</td>
<td>5400</td>
<td>0.18</td>
<td>230</td>
<td>0.24</td>
<td>219</td>
</tr>
<tr>
<td></td>
<td>21600</td>
<td>0.19</td>
<td>230</td>
<td>0.23</td>
<td>224</td>
</tr>
<tr>
<td>Earth IR</td>
<td>5400</td>
<td>0.24</td>
<td>202</td>
<td>0.21</td>
<td>242</td>
</tr>
<tr>
<td></td>
<td>21600</td>
<td>0.23</td>
<td>205</td>
<td>0.21</td>
<td>216</td>
</tr>
<tr>
<td>Both</td>
<td>5400</td>
<td>0.21</td>
<td>224</td>
<td>0.23</td>
<td>232</td>
</tr>
<tr>
<td></td>
<td>21600</td>
<td>0.21</td>
<td>226</td>
<td>0.22</td>
<td>230</td>
</tr>
</tbody>
</table>

The iteration process went as taking a WCH Earth IR and associating it to a WCC albedo as these tables are constructed to define the worst-case conditions, and not the simultaneous conditions that are desired for validation [3]. A note is the albedos require a correction factor based on the beta angle that are not displayed here which results in higher values than shown. The final values chosen were based on an albedo sensitive WCH Earth IR of $219 \frac{W}{m^2}$ and with a dual sensitive WCC albedo of 0.247. A combination of these values was closely able to match the external panel temperatures on all faces within $0.5^\circ C$. The final BCs used for the validation model can be seen in Table 13 where all the conditions relevant to GHGSat-D are present. Here variables such as the rocket
orbital insertion have been accounted for by using the TLE provided for the spacecraft and the tolerances in the thermal tapes have been adjusted to be nominal instead of EOL due to the chosen data being the early stages of the mission.

<table>
<thead>
<tr>
<th>Case</th>
<th>LTDN</th>
<th>Altitude [km]</th>
<th>Beta Angle</th>
<th>Solar Flux $[\text{W/m}^2]$</th>
<th>Albedo</th>
<th>Earth IR $[\text{W/m}^2]$</th>
<th>Mode</th>
<th>Attitude</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nominal</td>
<td>9:30</td>
<td>500</td>
<td>37.37</td>
<td>1317.32</td>
<td>0.247</td>
<td>219</td>
<td>S-Band</td>
<td>-Y Nadir Tracking</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>Operational</td>
<td>+Z Offset 1.73°</td>
</tr>
</tbody>
</table>

### 3.6.3 Modelling Variations

The process of validating the model was to explore how accurate certain aspects of the existing thermal model are. After verifying the material properties and conductances to the bus, and simulated with the discussed BCs, the modelled external panel temperature profiles were able to match almost perfectly with the measured on-orbit data. A sample of these comparisons are seen in Figure 51 where the panels vary as little as 1°C in some cases with the majority being within 0.2°C. The difference and tolerances left are within the error of the Earth IR and albedo variations and the small differences in nadir tracking from each orbit. With the external panels matching the measured data, the internal conditions can be verified regardless of the external thermal mechanisms.

![Figure 51: External panel simulated and on-orbit temperature comparisons](image-url)
The motivation to validate the thermal model for the upcoming GHGSat-C originally began to explore the accuracy of the battery model. As mentioned previous, a general validation of the model was performed and yielded results within 5°C differences from the measured results. The accuracy desired here was to explore the battery as the modelled properties did not match the specifications of the cells. This modelled capacitance was not documented and is assumed to be a result of the TVAC tests performed during acceptance of the spacecraft. Going forwards with the current iteration of the spacecraft and thermal model, accuracy is desired for such a critical component.

The batteries when charging or supplying power to any of the units on the spacecraft dissipates heat internally. This is due to the process of the internal charges moving between the electrodes through ionic movement, and thus the particles do not flow entirely freely, causing internal friction. This internal friction is modelled as a heat load and is determined by knowing the internal resistance of each cell with the current flow based on what state the spacecraft is in. The internal resistance of the battery increases with temperature [36] and this is accounted for as well. The chosen orbits are during an S-Band nominal operations mode where there are no payload components active, and the S-Band Tx turns on immediately after entering sunlight for ten minutes. This means the batteries start charging at the same time the S-Band is consuming power, although the satellite is still power positive with the net current establishing that difference. The modelled heat loads are seen in Figure 52 where it shows a common step function that represents the total heat dissipated by the batteries via the curve. The curved heat dissipated from the measured current is due to the battery voltage increasing as it charges. Modelling with NX uses step functions as are normally done with all of the microsatellite projects, where as long as the total energy over the same time is captured, the system is accurate enough. This is especially true for high capacity objects such as the lithium-ion battery cells.

![Battery Heat Dissipation Comparison](image_url)
Upon investigating the battery temperatures and comparing them, there were some relevant thermal connections that were recalculated for the sake of accuracy. This includes the small thermally conductive gap filler to couple the batteries with the heater that was too optimistic. The calculations used were the ones described in the modelling resistances Section 1.5.4 where Eq. (19) is derived from the compression ratio of the gap-filler. Additionally, internal radiation was included on the battery pack and enclosure with the entire upper tray processing the radiative elements. The Delrin enclosure in particular is a highly emissive black coating that couples the batteries to the bus through radiation. These differences are effectively minor for a thermal model to determine the envelope, but for the sake of validation, are welcomed inclusions.

The final results for the batteries resulted in a temperature profile that is within an acceptable accuracy from the measured results. However, the rate of change observed, although within less than 1°C at all times, is not clear where the cause of the difference is. The battery temperature comparison results can be seen in Figure 53 where there are two versions of the same system and boundaries. From an engineering and design perspective, the differences shown are minor and may be used for further analysis.

![Figure 53: Battery temperature comparisons with real and adjusted capacitances](image_url)
The difference seen in the actual battery comparison is still curious. This effectively means the actual battery is dissipating and gaining heat at a faster rate than what the simulation predicts. These are the same battery cells used on the previously discussed DAUNTLESS bus in Section 2.5.3 where they validated nearly perfectly. Figure 53 shows the difference of the batteries with a thermal capacitance that is adjusted on the right to store less heat, and thus slew much faster. This artificial change results in a perfect match, whereas the previous thermal model used a specific heat property of 800 \( \frac{J}{kgK} \). This could have been a compromised value that partially represented the real capacitance, while approaching a previously measured experiment.

The physical difference is still a wonder. The model was already adjusted with more accurate higher conductive connections from the battery, through its subassembly and into the bus. The bus itself is within the predicted temperature range, so increasing the conduction to an arbitrary value does not change the curve enough. This includes reasoning of the battery expanding when warming up to increase its contact pressure with the enclosure. The temperature sensor location is modelled in the correct location as the actual unit. The only possible explanation for this is the temperature sensor itself is cooling faster than the battery due to the Kapton layers on the battery between the coupled connection and the sensor has a non-insignificant conductive path through its wire itself. For the remaining development of GHGSat-C, the actual battery heat capacity will be used throughout the development, with careful attention to the temperature sensor mounting to be coupled with the battery cell directly. This will be confirmed during the unit tested of the spacecraft during the acceptance testing in the TVAC chamber.
Chapter 4

Conclusion

Thermal design and analysis are often an iterative process to balance sensitive components onboard the spacecraft with the structural limitations of what is needed. Satellites in LEO experience wild temperature fluctuations every orbit that must always be verified with the specialized components that keep the satellite alive. The thermal balance is achieved via the internal structural connections due to most components only operating in a small temperature range before either degrading, performing sub-optimally, or breaking altogether. The structural connections are also what ensures a spacecraft will survive the violent vibrations from launch, while acting as a heat sink thermally. During cold peaks, units that are well coupled become as cold as their structural connection, while a weak coupling will increase the thermal inertia keeping the component warmer for longer. A similar case is true for hot cases where heat will always flow towards the cooler parts of the spacecraft.

For the specific cases discussed, there are some recommendations towards a first level system design. Specifically, is to have all sensitive electric components mounted to an internal tray that is not directly coupled with an external panel. This will allow thermal inertia to keep components warm during the cold orbits, while also allowing for control of the internal tray acting as a heat sink for high power components and hot cases. While something like this might not always be possible depending on the mission, it is important to recognize these limitations before the preliminary design review and determine if certain restrictions or designed risks are acceptable. The use of thermal safety switches to cut power based on temperature sensors were used commonly in both missions due to certain components risking operations for unfavorable worst-case attitudes. These measures are simple to implement and therefore one of the most reliable options but increases the overall risk of failure. These tools are available and should be used sparingly for added safety rather than common occurrences. Of course, it may not be obvious until after the design has gained a lot of momentum, where the thermal power cutoff is attractive as it is a relatively easy solution most for most cases.
During the design phase, considering the critical components and producing a simplified model is what allows a mission to conceptualize. This is used to verify orbital parameters and internal thermal connections to determine the feasibility of relying on certain electrical or mechanical systems. The iterative process is also part of the inclusion of more detail as a model increases its representation of the real structure. Throughout the phases of a thermal design, through the microspace philosophy, the importance of adding margins and reducing risk is further emphasized by its simplicity. If a design can close and be proven to work, despite simulating unrealistic conditions that envelopes the real ones, then the design is robust and can be fully developed. There is conservatism starting with the boundary conditions, that assume the highest heat loads for the WCH conditions, and the smallest heat loads for the WCC conditions. These conservatisms are further used for internal heat loads where the spacecraft may enter a power state that dissipates the most or least of its power through its modes, even if the mode would not be desired realistically. This is done to ensure no operator error would introduce a damaging thermal state among other cases.

Furthermore, units are all tested to ensure they work at targeted temperature ranges or beyond. When considered for simulations, there are margins added to the cold and hot limits to account for potential modelling errors. TVAC testing and validation are practices used to minimize these, however where simply adding margin is a safe practice. There are also cases that are studied where the satellite is inertially locked with a certain area always facing the sun. This realistically is hard to maintain unless the spacecraft is commanded either intentionally or through software errors. For larger spacecraft with higher heat capacities, it could take a dozen of orbits or more for the maximum temperatures to be experienced by the system in this condition. Although these attitudes are rare, verifying that the thermal system can work regardless will add to the robustness of the design. All these systems work together to form a practical procedure for isolating problems and converging on practical solutions that appease all subsystems.
Through the experience I attained by working on the thermal analysis and design for microsatellites at SFL, it is clear that there are often flexible solutions to very real engineering problems. Although designing such a complicated system from the ground up is a large team effort with many subsystems providing their expert design decisions, thermal control is often one of the last considerations for design decisions. For smaller spacecraft this can be a safe assumption as the system has undergone countless years of experience and previous design decisions that have panned out, but for feasibility studies, major changes must be modelled appropriately and discovered before other subsystems are dedicated to their design. The process for building a thermal model can be tedious and time consuming, where making sure human error is accounted for and troubleshooting software errors can be a trying experience. Only through this experience can these thermal systems seem obvious in retrospect and offer a sense for where design choices can lead the project. These projects were exercises in verifying thermal design choices and learning model building, for in the end, physics never lies, but can always offer something new to learn.
References


