Design of Thermal Control System for the Spacecraft MIST

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“On a visit to the space program, President Kennedy asked me about the satellite. I told him that it would be more important than sending a man into space. “Why?” he asked. “Because,” I said, “this satellite will send ideas into space, and ideas last longer than men.”

Newton N. Minnow
KTH ROYAL INSTITUTE OF TECHNOLOGY

Abstract

Department of Space and Plasma Physics
School of Electrical Engineering

Master of Science

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by Andreas Berggren

In 2014 KTH Royal Institute of Technology initiated a space technology and research center, KTH Space Centre. MIST (MIniature Student saTellite) is the first student project conducted at KTH Space Centre and also the first student satellite from KTH with a predicted launch in 2017. This report includes the thermal analysis and control of the spacecraft MIST.

One of the main systems in a spacecraft is the thermal control system. In order for the payloads and subsystems to withstand the harsh thermal environment in space a thorough thermal analysis is needed. In this project the thermal model has been built and thermal control design of the spacecraft has been started. As a start a preliminary thermal analysis was performed where the spacecraft was approximated as a sphere in order to get some estimates on the temperature in orbit due to the space environment. Furthermore the temperature decrease in eclipse was studied. Since most of the spacecraft will consist of Printed Circuit Boards (PCB) the thermal behavior of PCB has been investigated and as a part of this investigation a thermal vacuum chamber test was performed where the conductance from a PCB through the mounting interface to a metal plate was measured. This report will also guide the reader through the model built and assumptions made. As a part of the thermal control, Multi Layer Insulation (MLI) has been studied and modeled in two different ways which have been compared with each other in order to know the level of detail needed for the MLI model.

Last but not least the design of the thermal control system has been started where some payloads have been wrapped in MLI and thermal contact conductance coefficient has been changed in order to meet the thermal requirements of the payloads and subsystems.
Acknowledgements

The author wish to express his sincere gratitude to Mr. Lars Bylander of the Space and Plasma Physics Department of KTH for his supervision, feedback and constant participation in assistance throughout the project together with his everlasting enthusiasm and encouragements. The author is also grateful to Mr. Sven Grahn, Project Manager of the MIST project for his guidance, expertise and inputs to the project regarding the different subsystems of the spacecraft and particularly for his guidance of how to run a professional space mission. A special thank you also goes to Mr. Tobias Kuremyr of the Space and Plasma Physics Department of KTH for his inputs and advice regarding Printed Circuit Boards. Furthermore, the author would like to thank Mr. Vincent Haugdahl, Student Team Leader of the MIST project for his ability to lead the MIST team and take on the leadership such an project needs. The author would also like to take this opportunity to forward his appreciativeness to the remaining part of the MIST student team, Ms. Joanna Alexander, Mr. Gábor Felcsuti, Mr. Sharan Ganesan, Ms. Agnes Gårdeböck, Ms. Rebecca Ilehag, Mr. Davide Menzio, Mr. Manish Sonal and Mr. Daniel Sjöström. The author also place on record, his sense of gratitude to one and all, who directly or indirectly, have lent their hand in this venture.
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**Abbreviations**

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<th>Abbreviation</th>
<th>Description</th>
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<td>CUBES</td>
<td>CUbesat x-ray Background Explorer using Scintillators</td>
</tr>
<tr>
<td>ESA</td>
<td>European Space Agency</td>
</tr>
<tr>
<td>FR</td>
<td>Flame Retardant</td>
</tr>
<tr>
<td>IR</td>
<td>Infra Red</td>
</tr>
<tr>
<td>ISIS</td>
<td>Innovative Solutions In Space</td>
</tr>
<tr>
<td>JUICE</td>
<td>JUpiter ICy moon Explorer</td>
</tr>
<tr>
<td>KTH</td>
<td>Kungliga Tekniska Högskolan</td>
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<tr>
<td>MIST</td>
<td>MIIniature Student saTellite</td>
</tr>
<tr>
<td>MLI</td>
<td>Multi Layer Insulation</td>
</tr>
<tr>
<td>MOREBAC</td>
<td>Microfluidic Orbital RESuscitation of BACteria</td>
</tr>
<tr>
<td>NASA</td>
<td>National Aeronautics and Space Administration</td>
</tr>
<tr>
<td>NTC</td>
<td>Negative Temperature Coefficient</td>
</tr>
<tr>
<td>OBC</td>
<td>On Board Computer</td>
</tr>
<tr>
<td>PCB</td>
<td>Printed Circuit Board</td>
</tr>
<tr>
<td>PET</td>
<td>PolyEtylenTereftalat</td>
</tr>
<tr>
<td>RATEX-J</td>
<td>RAdiation Test EXperiment for JUICE</td>
</tr>
<tr>
<td>SAA</td>
<td>South Atlantic Anomaly</td>
</tr>
<tr>
<td>SEAM</td>
<td>Small Explorer for Advanced Missions</td>
</tr>
<tr>
<td>SEU</td>
<td>Single Event Upset</td>
</tr>
<tr>
<td>SiC</td>
<td>Silicon Carbide</td>
</tr>
</tbody>
</table>
Physical Constants

Stefan-Boltzmann constant \( \sigma = 5.670 \, 373 \, 21 \times 10^{-8} \, \text{Wm}^{-2}\text{K}^{-4} \)

Solar Flux \( G_s = 1377 \, \text{Wm}^{-2} \)

Radius of Earth \( R = 6378 \times 10^3 \, \text{m} \)
## Symbols

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
<th>Unit</th>
</tr>
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<tbody>
<tr>
<td>A</td>
<td>Area</td>
<td>(m^2)</td>
</tr>
<tr>
<td>a</td>
<td>Albedo</td>
<td></td>
</tr>
<tr>
<td>C</td>
<td>Specific Heat</td>
<td>(J/kgK)</td>
</tr>
<tr>
<td>F</td>
<td>View Factor</td>
<td></td>
</tr>
<tr>
<td>G</td>
<td>Thermal Conductance</td>
<td>(W/K)</td>
</tr>
<tr>
<td>H</td>
<td>Altitude above Earth Surface</td>
<td>(m)</td>
</tr>
<tr>
<td>(h_c)</td>
<td>Thermal Contact Conductance</td>
<td>(W/m^2K)</td>
</tr>
<tr>
<td>I</td>
<td>Current</td>
<td>(V)</td>
</tr>
<tr>
<td>(I_i)</td>
<td>Intensity</td>
<td>(W/m^2)</td>
</tr>
<tr>
<td>(J_i)</td>
<td>Radiation emitted and reflected by surface i</td>
<td>(W/m^2)</td>
</tr>
<tr>
<td>k</td>
<td>Thermal Conductivity</td>
<td>(W/m)</td>
</tr>
<tr>
<td>m</td>
<td>Mass</td>
<td>(kg)</td>
</tr>
<tr>
<td>Q</td>
<td>Heat</td>
<td>(W)</td>
</tr>
<tr>
<td>q</td>
<td>Heat Flow</td>
<td>(W/m)</td>
</tr>
<tr>
<td>(R_i)</td>
<td>Thermal Resistance</td>
<td>(m^2K/W)</td>
</tr>
<tr>
<td>r</td>
<td>Radius</td>
<td>(m)</td>
</tr>
<tr>
<td>T</td>
<td>Temperature</td>
<td>(K)</td>
</tr>
<tr>
<td>t</td>
<td>Thickness</td>
<td>(m)</td>
</tr>
<tr>
<td>x</td>
<td>Distance</td>
<td>(m)</td>
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</table>

- \(\alpha\) Absorptivity
- \(\gamma\) Angle \(\text{rad}\)
- \(\epsilon\) Emissivity
- \(\theta\) Angle \(\text{rad}\)
- \(\rho\) Density \(kg/m^3\)
\( \omega \)  Angle  \( \text{rad} \)
Chapter 1

Introduction

During a two year period students at KTH Royal Institute of Technology will build a satellite called MIST (MIniature Student saTellite) under the supervision of Mr. Sven Grahn with a predicted launch in 2017. The satellite is a 3U CubeSat with the dimensions $10 \times 10 \times 30 \text{cm}^3$. Each semester a new team of approximately ten students will be selected to continue the work on the satellite. The satellite will host eight different payloads which is, to the best of the authors knowledge, the highest number of payloads on a 3U CubeSat ever launched before. The high number of payloads has and will continue to create great challenges within several areas such thermal, power consumption, data storage, data rate etc. The payloads comes from both industry and academia and are called Nanospace, Ratex-J, Morebac, SiC, Cubes LEGS, SEU and the last payload is a camera.

The Nanospace payload is a propulsion module suitable for cubesats and the overall goal is to gain flight heritage of this module. Ratex-J stands for RAdiation Test EXperiment for JUICE and is a prototype of an solid state detector based anti-coincidence system for measurements with ceramic channel electron multipliers and multichannel plate to be implemented in the JDC for the ESA (European Space Agency) JUICE (JUpiter ICy moon Explorer) spacecraft. Morebac, which stands for Microfluidic Orbital Resuscitation of Bacteria is an experiment proposed by the Division of Proteomics and Nanobiotechnology at KTH. The purpose of the experiment is to resuscitate freeze dried bacteria in orbit and study how they grow in microgravity. The SiC experiment is a transistor made of Silicon Carbide with the goal of being tested under the harsh space
environment, applications for this experiment has been suggested for electronics for a Venus lander. Cubes stands for CUbesat x-ray Background Explorer using Scintillators with the goal of study the in-orbit radiation environment using a detector comprising a silicon photomultiplier coupled to scintillator material. LEGS is a payload provided by PiezoMotor. The LEGS motor is based upon piezoelectric elements that can stretch and bend in a certain way to create a walking principle. The purpose of the experiment on MIST is to gain experience of using this motor in space application together with getting better numbers of how the motor really work in the space environment. SEU stands for Single-Event Upset detection and the purpose of the payload is to test their in-house concept for self-healing/fault-tolerant computer systems in a hostile environment such as the space environment to see if it is able to heal itself by correcting faults during runtime. The second purpose of the payload is to measure the expected SEU frequency in near Earth orbit. Lastly the camera will take pictures of Earth that will be displayed at Tekniska Museet in Stockholm, Sweden and will be a part of their exhibition MegaMind that will open in the fall of 2015.

The space environment is an extremely harsh environment when it comes to radiation, vacuum, temperature etc. The temperature in deep space is about 2.7K while when an object in space is subjected to direct Sun light the temperature can be several hundred degrees Celsius. Due to this extreme fluctuation in temperature in space it is important that each spacecraft has a thermal control system. Different parts of a spacecraft have different thermal requirements in which they can operate and survive in. If the temperature exceed the thermal requirements for a specific subsystem or payload the entire mission can be compromised. This is also the reason why it is important for the MIST spacecraft to have a reliable thermal analysis and thermal control system in order to verify and guarantee that the thermal requirements of each payload and subsystem is fulfilled.

This report contains the thermal analysis and thermal control philosophy of the spacecraft MIST. In Chapter 2 the theory behind thermal heat transfer is explained. Chapter 3 guides the reader through the preliminary thermal analysis where the temperature contribution from the Sun, albedo and Earth IR are studied. Since most of the spacecraft will consist of Printed Circuit Boards (PCB) effort has been put into investigating the thermal behavior of PCBs. A thermal vacuum chamber experiment has also been performed in order to measure the conductance from a PCB to a metal plate through
the mounting interface between them. PCBs has been dedicated an own chapter which is \textit{Chapter 4}. \textit{Chapter 5} explains the detailed modeling of the thermal analysis of the spacecraft. In this chapter each subsystem and payload is explained individually and how they have thermally been modeled. \textit{Chapter 6} covers the thermal controlling of the spacecraft i.e. what measures that has been taken in order to control the temperature of the different subsystems and payloads. Here deeper effort has been taken into investigating the thermal modeling of Multi Layer Insulation (MLI) and one detailed model has been compared with a simpler model suggested by ESA.
Chapter 2

Theory

Conventional heat transfer consist of three modes: conduction, radiation and convection. Convection, heat transfer due to the interaction of a fluid over a surface, can be neglected in space due to the vacuum that exist there. However, conduction and radiation account for most of the heat exchange in vacuum among the spacecraft components.

2.1 Conduction

Thermal conduction is the transfer of internal energy by collisions of particles or microscopical diffusion within a body. Every material conducts heat but the thermal conductivity, the ability to conduct heat given in $W/mK$, can vary severely between different materials. In a spacecraft most of the heat transferred is through conduction. By choosing the materials in a spacecraft the thermal engineer can direct and insulate the heat transfer within the spacecraft. When two materials are in physical contact with each other heat is conducted between the two interfaces, this is called thermal contact conductance and is explained below.

2.1.1 Thermal Contact Conductance

Thermal contact conductance, $h_c$ is the ability to conduct heat between two bodies. Consider two bodies in contact with each other where heat flows from the hotter body to the colder body. According to Fourier’s law heat flows as:
Figure 2.1: Thermal contact conductance [1]

\[ q = -kA \frac{dT}{dx} \]  

(2.1)

where \( q \) is the heat flow, \( k \) the thermal conductivity, \( A \) the cross sectional area and \( dT/dx \) the temperature gradient. From the energy conservation the heat flow between two bodies \( A \) and \( B \) is given by:

\[ q = \frac{T_1 - T_3}{\frac{\Delta X_A}{k_A} + \frac{1}{h_c} + \frac{\Delta X_B}{k_B}} \]  

(2.2)

where \( T_1 \) and \( T_3 \) is the temperature at each end of the two bodies in contact and \( \Delta X_A \) and \( \Delta X_B \) is the distance the heat has conducted through material \( A \) and \( B \) respectively given by Figure 2.1.

We can also define the thermal conductance as:

\[ G = \frac{kA}{dx} \]  

(2.3)

or

\[ G = h_c A \]  

(2.4)
which both have the unit $W/K$. Furthermore, the sum rule for two conductances, $i$ and $j$, is given by:

$$G_{tot} = \frac{1}{1/G_i + 1/G_j}$$  \hspace{1cm} (2.5)

There are mainly five phenomena that can effect the thermal contact conductance:

- **Contact pressure** is the most important factor and the contact conductance will increase as the pressure increases. This is due to the fact that the contact surface between the two bodies will increase as applied pressure will decrease the microscopical distance between the interfaces.

- **Interstitial materials**: No surface is truly smooth, they all have some imperfections which means that contact interface between two surfaces is in contact only in a finite number of points separated by gaps which are relatively large. The gas or fluids that fill these gaps may influence the heat flow across the contact interface. For the spacecraft in space it is the thermal conductivity and pressure of the interstitial material that influences the contact conductance because of the lack of gas and fluid in vaccum.

- **Surface deformations**: When two surfaces come in contact there may occur a surface deformation which can be either elastic or plastic depending on material. However, when a surface undergoes plastic deformation the contact area will increase and hence also the contact conductivity.

- **Surface cleanliness, roughness and flatness** will also all effect the contact conductance.

### 2.2 Radiation

Radiation is the emission of energy in the form of particles or waves, including particle radiation such as $\alpha$, $\beta$ and neutron radiation, electromagnetic radiation in the form of visible light, radio waves and x-rays and also acoustic radiation. A spacecraft in orbit will encounter energy from three sources of radiation, solar radiation, Earth albedo and Earth infrared [7] which is illustrated in Figure 2.2.
Radiation from a black body (a perfect emitter) is given by:

$$Q = A_r \sigma T^4$$  \hspace{1cm} (2.6)

where $A_r$ is the radiating area, $\sigma$ is the Stefan-Boltzmann constant and $T$ is the temperature of the body. Radiation from a non black (gray body) is:

$$Q = \epsilon \sigma A_r T^4$$  \hspace{1cm} (2.7)

where $\epsilon$ is the emissivity which is the ratio between the energy that the gray body emits to the energy emission it would have if it were a black body.

### 2.2.1 Solar Radiation

The Sun contribute the most to the heating of a spacecraft in orbit around Earth and is considered to be a black body radiating at a temperature of 5780K [8]. This results in an incoming solar radiation to Earth, or to a spacecraft in the near vicinity, of a mean flux of 1377W/m$^2$. The solar radiation to Earth varies about 6.9% due to the varying distance between the Sun and the Earth with a maximum flux of 1414W/m$^2$ and a minimum flux of 1322W/m$^2$ [9].
2.2.2 Albedo - Reflected Solar Radiation

When a material is struck by energy in the form of light it can either be absorbed, transmitted or reflected. Since the Earth is opaque the light will be reflected off the surface of the Earth, this is called albedo. A portion of this energy will hit the spacecraft orbiting the Earth. However it is very hard to precisely say how much of the incoming light to the Earth is reflected since each material reflects different portion of incoming light. That means that the albedo is dependent on several aspects, for example on the weather where the formation and density of the clouds play a crucial roll. Snow is another factor as its albedo can vary between 0.9 for new fallen snow and down to 0.4 for melting snow [10], just to mention a few examples. Forests and water are considered to have very low albedo in spite of the high reflectivity of water at high angles of incident light. On average the Earth and its atmosphere has a combined albedo of about 30% [3].

![Albedo of the Earth's terrestrial surface as measured by the TERRA satellite. Data collected from the period April 7-22, 2002. (Source: NASA Earth Observatory) [3]](image)

2.2.3 Earth Infrared Radiation

For a body to be in thermal equilibrium the body needs to re-radiate the same amount of energy it absorbed. As seen in **Equation 2.7** the energy radiated from a body is dependent on the temperature it has. Earth re-emits energy which is in the infrared spectrum and the energy changes for different locations on Earth due to the different temperatures around the globe. Also a portion of this energy will hit the spacecraft and affect the temperature of it. However, by considering Earth as a black body radiator at
$-20^\circ C$ the Earth flux is given by equation 2.6 to be $236 W/m^2$ [7] which is an average value that will differ by $\pm 21 W/m^2$ [11].

2.2.4 Re-radiation to Space

In the same way that Earth needs to maintain a thermal equilibrium so will the spacecraft MIST. The spacecraft will radiate heat to the surrounding space according to the equation:

$$Q = \varepsilon \sigma A (T_{surface}^4 - T_{space}^4)$$  \hspace{1cm} (2.8)

where $A$ is the radiating area of the spacecraft, $T_{surface}$ is the surface temperature of the spacecraft and $T_{space}$ is the ambient temperature of space which is about $-270^\circ C$ [9].

2.3 View Factor

Radiation heat exchange is dependent on the orientation of the surfaces relative to each other which is accounted for by the view factor.

Consider the two differential surfaces $dA_1$ and $dA_2$ in Figure 2.4. The distances between them is $r$ and the angles between the surface normal and the line $r$ are $\theta_1$ respectively $\theta_2$, $d\omega_{21}$ is the angle subtended by $dA_2$ when viewed by $dA_1$. The portion of radiation

\begin{figure}[h]
\centering
\includegraphics[width=0.5\textwidth]{view_factor.png}
\caption{Geometry for determination of the view factor between two surfaces [4].}
\end{figure}
that leaves $dA_1$ in the direction of $\theta_1$ is $I_1 \cos\theta_1 dA_1$, where $I_1$ is the intensity of what surface 1 emits and reflects, is given by:

$$\dot{Q}_{dA_1 \rightarrow dA_2} = I_1 \cos\theta_1 dA_1 d\omega_{21} = I_1 \cos\theta_1 dA_1 \frac{dA_2 \cos\theta_2}{r^2} \quad (2.9)$$

The differential view factor $dF_{dA_1 \leftarrow dA_2}$ is given by:

$$dF_{dA_1 \leftarrow dA_2} = \frac{\dot{Q}_{dA_1 \leftarrow dA_2}}{\dot{Q}_{dA_1}} = \frac{\cos\theta_1 \cos\theta_2}{\pi r^2} dA_2 \quad (2.10)$$

The view factor from a differential area $dA_1$ to a finite area $A_2$ can be determined from the fact that the fraction of radiation leaving the differential area that strikes the finite area is the sum of the fractions of radiation hitting the differential areas $dA_2$. Therefore the view factor $F_{dA_1 \leftarrow dA_2}$ is determined by:

$$F_{dA_1 \leftarrow dA_2} = \int_{A_1} I_1 \cos\theta_1 \cos\theta_2 dA_2 \quad (2.11)$$

To determine the portion of radiation that leaves the entire area $A_2$ and hit the differential area $dA_2$ is given by:

$$\dot{Q}_{A_1 \rightarrow dA_2} = \int_{A_2} \int_{A_1} I_1 \cos\theta_1 \cos\theta_2 dA_2 dA_1 \quad (2.12)$$

Integrating this over the area $A_2$ will give the radiation that hit area $A_2$.

$$\dot{Q}_{A_1 \rightarrow A_2} = \int_{A_2} \int_{A_1} I_1 \cos\theta_1 \cos\theta_2 dA_2 dA_1 dA_2 \quad (2.13)$$

The view factor $F_{A_1 \rightarrow A_2}$ or $F_{12}$ is given by dividing EQUATION 2.13 by the total radiation leaving area $A_1$ that strikes area $A_2$ which is $\dot{Q}_{A_1} = \pi I_1 A_1$. The expression for the view factor becomes:

$$F_{12} = \frac{1}{A_1} \int_{A_2} \int_{A_1} \frac{\cos\theta_1 \cos\theta_2}{\pi r^2} dA_1 dA_2 \quad (2.14)$$
Chapter 3

Preliminary Thermal Analysis

A preliminary thermal analysis has been performed in order to get an estimate of the temperatures that the spacecraft will encounter due to the environmental effects. For this preliminary thermal analysis the three main thermal contributions, which are solar radiation, albedo and Earth infrared radiation are considered. Heat dissipation within the spacecraft is not considered in this preliminary thermal analysis. For the thermal model the suggested "reference orbit one" from the MIST team is chosen. Reference orbit one is an orbit at 640 km altitude with an eccentricity of 0.001, orbital inclination of 97.943°, argument of perigee 0°, orbital period of 5851 sec and a local time at the ascending node of 10:45:00.

3.1 The Model

For the preliminary thermal analysis it is convenient to model the spacecraft as a sphere. Because of the great distance between the Sun and the spacecraft the illuminated area on the spacecraft can then be approximated to be a circular disc with the same radius as the sphere, i.e. one fourth of the sphere’s area. The same approximation can be made for the effect of the Earth’s infrared radiation due to the great difference between the size of the spacecraft and the Earth.
3.2 Analytical Analysis

For this analytical analysis a spherical spacecraft is considered.

3.2.1 Method

As mentioned earlier the spacecraft is modeled as a sphere for the preliminary thermal analysis, however the total area of the sphere should be the same as the area of the real MIST spacecraft which has the form of a rectangular cube with the dimensions 10x10x30cm$^3$. Since the total area of the spacecraft is:

$$A_{s/c} = 0.1 \times 0.3 \times 4 + 2 \times 0.1 \times 0.1 = 0.14m^2$$  \hspace{1cm} (3.1)

and the area of a sphere is given by:

$$A_{sphere} = 4\pi r^2_{sphere}$$  \hspace{1cm} (3.2)

By equating these areas the radius of the sphere is given by:

$$r_{sphere} = \sqrt{\frac{A_{s/c}}{4\pi}} = 0.1056m$$  \hspace{1cm} (3.3)

The energy absorbed by the sphere due to solar radiation is given by [7]:

$$Q_{solar} = \alpha G_s A_{absorbed}$$  \hspace{1cm} (3.4)

where $\alpha$ is the absorption coefficient, $G_s$ is the solar flux and $A_{absorbed}$ is the absorbing area of the sphere (approximated to be the one of a circular disc with the same radius as the sphere, i.e. $\frac{1}{4}$ of the total area of the sphere). The heat contribution due to Earth infrared radiation is given by:

$$Q_{IR} = \epsilon A_{absorbed} \frac{q_{IR}}{(R+H)^2}$$  \hspace{1cm} (3.5)
where \( q_{IR} \) (W/m\(^2\)) is the Earth flux, \( R \) is the radius of Earth and \( H \) is the altitude of the spacecraft above Earth. Note that the Earth flux decreases as one over the distance squared. The heat contribution from the Earth albedo is given by [7]:

\[
Q_{\text{albedo}} = \alpha A_g \frac{G_s}{F} \tag{3.6}
\]

where \( \alpha \) is the percentage of the solar flux that is reflected off the surface of Earth and \( F \) is the View factor which for a large sphere to a small hemisphere is given by [12]:

\[
F = \frac{1}{4} - \frac{\left(2 \frac{H}{R} + \left(\frac{H}{R}\right)^2\right)^{1/2}}{4(1 + \frac{H}{R})} + \frac{\cos(\gamma)}{8} \left(\frac{1}{1 + \frac{H}{R}}\right)^2 \tag{3.7}
\]

where \( \gamma \) is given by:

\[
\gamma = \arcsin\left(\frac{R}{R + H}\right) \tag{3.8}
\]

The hot case, i.e. the case when the highest temperature on the spacecraft will occur is when the spacecraft is exposed to sunlight and albedo effect. By equating the absorbed and emitted heat subjected to the spacecraft one can solve for the maximum temperature:

\[
T_{\text{max}} = \left(\frac{Q_{\text{solar}} + q_{IR} + Q_{\text{albedo}}}{\epsilon \sigma A} \right)^{1/4} \tag{3.9}
\]

For the cold case, i.e. the case when the lowest temperature on the spacecraft will occur is when the spacecraft is in eclipse (when the Earth is between the Sun and the spacecraft). The only heat the spacecraft is subjected to in eclipse is the Earth’s infrared radiation. By equating the absorbed and emitted heat one can solve for the spacecraft temperature:

\[
T_{\text{min}} = \left(\frac{\epsilon A_{\text{absorbed}} \frac{q_{IR}}{(R + H)^2} \sigma}{\epsilon \sigma \frac{3A}{4}} \right)^{1/4} \tag{3.10}
\]
Note that the factor \( \frac{3}{4} \) in front of the area, \( A \), comes from that the spacecraft radiates heat to space where it "sees deep space" i.e. where the spacecraft is not subjected to the infrared radiation from the Earth (the Earth radiation is approximated to hit the spacecraft with an area as a circular disc with the same radius as the spherical spacecraft which is \( \frac{1}{4} \) of the area).

### 3.2.2 Results

In order to know a suitable absorption coefficient, \( \alpha \), and emissivity coefficient, \( \epsilon \), to use for the analysis the maximum and minimum temperature was calculated for a set of \( \alpha \) and \( \epsilon \) resulting in maximum temperatures ranging between 0 – 40\(^\circ\)C, this interval was chosen because it is the interval where the spacecraft electrical components often can operate within.

![Graph showing set of \( \alpha \) and \( \epsilon \) that will give a maximum temperature interval on the spacecraft between 0 – 40\(^\circ\)C](image)

**Figure 3.1:** Set of \( \alpha \) and \( \epsilon \) that will give a maximum temperature interval on the spacecraft between 0 – 40\(^\circ\)C

From Figure 3.1, \( \alpha \) and \( \epsilon \) can be chosen that will generate a good operating temperature. In this analysis \( \alpha = 0.6 \) and \( \epsilon = 0.7 \) was chosen.

**Figure 3.2** shows the same thing as Figure 3.1 but here the temperature interval has not been limited to be within the interval of 0 – 40\(^\circ\)C. With the method presented above and with \( \alpha = 0.6 \) and \( \epsilon = 0.7 \) at an orbital altitude of 640km the maximum and minimum temperature for the spacecraft for the steady state case was found. The
maximum temperature the spacecraft would encounter was $T_{\text{max}} = 18^\circ C$. The minimum temperature for a steady state case was $T_{\text{min}} = -89^\circ C$.

### 3.3 Numerical Analysis Using Siemens NX™

For this numerical analysis a spherical spacecraft is considered.

#### 3.3.1 Method

In order to verify the analytical results the spacecraft (modeled as a sphere) was modeled in Siemens NX™ as a primitive with absorption coefficient, $\alpha = 0.6$ and emissivity coefficient $\epsilon = 0.7$. The material of the sphere was chosen to Aluminum2014 with a density of $\rho = 2794 kg/m^3$ [13]. Furthermore, radiation was defined on the outward surface of the sphere (for this preliminary analysis only the outer radiation effects are considered). Orbital heating was also defined on the sphere with an orbit of 640 km altitude with an eccentricity of 0.001, orbital inclination of 97.943°, argument of perigee 0° and an orbital period of 5851 sec. In order to verify the analytical solution the thickness of the sphere was chosen to be very thick (10cm), almost completely solid. However the thickness was then changed to 5cm, 1cm and 1mm to see the effect that the thickness of the sphere has on the temperature of it. Lastly the thickness was calculated to correspond to a
spacecraft with a mass of $4kg$ which is the approximated mass the MIST spacecraft will have. The thickness of the sphere to correspond to a mass of $4kg$ is given by:

$$t_{sphere} = r_{sphere} - r_{innersphere} = 1.14cm$$  \hspace{1cm} (3.11)

where $r_{sphere} = 10.56cm$ is the outer radius of the sphere and $r_{innersphere}$ is the inner radius of the sphere and is given by:

$$r_{innersphere} = \left( r_{sphere}^3 - \frac{3m}{4\pi \rho} \right)^{1/3}$$  \hspace{1cm} (3.12)

where $m = 4kg$ is the mass of the sphere and $\rho = 2794kg/m^3$ is the density of Aluminum 2014 which is the second most popular aluminum alloy in the 2000 series and is often used in the aerospace industry. Furthermore, not only the steady state cases were studied with Siemens NX™ but also the transient behavior.

### 3.3.2 Results

Note that, in the following figures in this section, nadir is defined in the XC-direction and the velocity vector in the YC-direction.

#### 3.3.2.1 Steady State Analysis

**Figure 3.3** shows the hot steady state temperature, i.e. the maximum temperature, for the spherical spacecraft with thickness 10cm, which can be compared with the analytical result given in Section 3.2.2. One can here conclude that the analytical and numerical results using Siemens NX™ is only differing 0.5°C. **Figure 3.4** show the cold steady state temperature for the spacecraft with thickness 10 cm i.e. the lowest temperature which also can be compared with the analytical result in Section 3.2.2 with the same conclusion that the analytical and numerical results are very similar, in this case the difference is about 1.5°C. When changing the thickness of the sphere to 5 cm the hot case steady state temperature only differs about 0.1°C in comparison with a thickness of 10 cm, the same goes for the cold steady state temperature. When investigating the sphere with thickness of 1 cm it is noted that the temperature distribution over the
Figure 3.3: Temperature distribution over the spherical spacecraft with a thickness of 10cm for the hot steady state case

sphere for the hot steady state case is between 22.4 – 24.7°C which is about 0.6 – 1.5°C difference from the solid (thickness of 10 cm) sphere. However the temperature for the cold steady state case is still −86°C. When the thickness of the sphere is scaled down to 1 mm the temperature starts to vary more. For the hot steady state case the temperature distribution over the sphere is between 14.6 – 35.6°C. For the cold steady state the temperature differs between −84°C and −88°C. For the spherical spacecraft with thickness of 1.14 cm, corresponding to a spacecraft with mass of 4 kg which is the approximate mass of the real MIST spacecraft, the temperature distribution for the hot steady state case varies between 22.4 – 24.6°C. This shows that the thickness of the material has an impact on how fast the heat is transferred within the material, the thicker material the slower the rate of heat transfer is. MATLAB code for the calculations and plots can be found in Appendix A.

Table 3.1: Maximum and minimum temperature for different thickness of the sphere, thickness of 1.14 cm corresponds to a sphere with mass of 4 kg which is the approximate mass the real MIST spacecraft will have

<table>
<thead>
<tr>
<th>Thickness</th>
<th>Hot case [°C]</th>
<th>Cold Case [°C]</th>
</tr>
</thead>
<tbody>
<tr>
<td>10 cm</td>
<td>23.5</td>
<td>-86.6</td>
</tr>
<tr>
<td>5 cm</td>
<td>23.6</td>
<td>-86.6</td>
</tr>
<tr>
<td>1.14 cm</td>
<td>24.6</td>
<td>-86.8</td>
</tr>
<tr>
<td>1 cm</td>
<td>24.8</td>
<td>-86.8</td>
</tr>
<tr>
<td>1 mm</td>
<td>14.6 → 35.7</td>
<td>-84.1 → -88.7</td>
</tr>
</tbody>
</table>
3.3.2.2 Transient Analysis

In order to investigate the temperature distribution over the sphere for an entire orbit a transient solution was calculated. For the solid sphere with thickness of 10 cm the transient solution gives that the temperature over the sphere will vary between 11.2°C and 23.5°C. For the sphere with thickness of 1.14 cm the temperature will vary between
−9.3°C and 24.6°C. It can be observed in Figure 3.5 that the temperature is stable after approximately six orbits.

3.4 Temperature Distribution for Different Orientations

For this analysis the real geometry of the MIST spacecraft has been used, i.e. the structure of a rectangular cube with the dimensions 10x10x30cm³. Siemens NX™ was used for the analysis. Two different orientations were analyzed, one where the side with the greatest area is always pointing towards the Sun and the other where the side with the smallest area is always pointing towards the Sun. This analysis will show the two extreme cases for the hot and cold temperatures the spacecraft will encounter due to the orientation of the spacecraft, i.e. if the spacecraft is illuminated by the Sun on the larger or smaller side of the spacecraft.

3.4.1 Largest Area Towards the Sun

Table 3.2: Average temperature for different thickness of the spacecraft when the Sun is illuminating the side with the largest area. Thickness of 1.21cm corresponds to a spacecraft with mass of 4kg which is the approximate mass the real MIST spacecraft will have. The hot and cold case corresponds to the steady state solution. The temperature variation in the transient column is dependent on where in the orbit the spacecraft is located.

<table>
<thead>
<tr>
<th>Thickness</th>
<th>Hot case [°C]</th>
<th>Cold Case [°C]</th>
<th>Transient [°C]</th>
</tr>
</thead>
<tbody>
<tr>
<td>5 cm</td>
<td>15.0</td>
<td>-88.6</td>
<td>-5.5 → 15.0</td>
</tr>
<tr>
<td>1.21 cm</td>
<td><strong>15.7</strong></td>
<td><strong>-88.7</strong></td>
<td><strong>-16.0 → 15.7</strong></td>
</tr>
<tr>
<td>1 cm</td>
<td>15.8</td>
<td>-88.7</td>
<td>-17.0 → 15.8</td>
</tr>
<tr>
<td>1 mm</td>
<td>14.8</td>
<td>-88.6</td>
<td>-58.8 → 23.9</td>
</tr>
</tbody>
</table>

As in Section 3.3 the analysis has been performed for different thicknesses of the spacecraft. In this case the thicknesses analyzed were 5 cm, 1.21 cm, 1 cm and 1 mm where the thickness 1.21cm corresponds to a spacecraft with mass 4kg. For the spacecraft with thickness of 5 cm, which corresponds to a solid spacecraft the hot steady state case the temperature distribution over the spacecraft is between 14.7°C and 15.0°C. For the cold steady state the temperature is quite stable around −88.6°C. For the transient case the temperature varies between −5.5°C and 15.0°C. As can be seen in Table 3.2 the temperature does not change much when scaling the thickness down to about 1 cm. However, for a thickness of 1 mm the temperature starts to vary drastically. The
Chapter 3. Preliminary Thermal Analysis

3.4.2 Smallest Area Towards the Sun

Table 3.3 displays the temperature distribution over the spacecraft for different thicknesses for the hot, cold and transient cases. Comparing Table 3.2 with Table 3.3 it is obvious that the spacecraft will have a higher temperature when the Sun is illuminating the side with the larger area. It is noted that the temperature difference between the two cases is as much as 38.9°C for the hot case for the spacecraft with thickness 1.21 cm. One can also conclude that the temperature variations are greater when the spacecraft has a smaller thickness. The less thickness the spacecraft has the less heat will transfer.
Chapter 3. *Preliminary Thermal Analysis*

Figure 3.7: Temperature variation for the spacecraft with thickness 1.21 cm while being illuminated by the Sun towards the side with the greatest area during ten (10) orbit periods.

Table 3.3: Average temperature for different thickness of the spacecraft when the Sun is illuminating the side with the smallest area. Thickness of 1.21 cm corresponds to a spacecraft with mass of 4 kg which is the approximated mass the real MIST spacecraft will have. The hot and cold case corresponds to the steady state solution. The temperature variation in the transient column is dependent on where in the orbit the spacecraft is located.

<table>
<thead>
<tr>
<th>Thickness</th>
<th>Hot case [°C]</th>
<th>Cold Case [°C]</th>
<th>Transient [°C]</th>
</tr>
</thead>
<tbody>
<tr>
<td>5 cm</td>
<td>-22.8</td>
<td>-87.4</td>
<td>-36.2 → -22.3</td>
</tr>
<tr>
<td>1.21 cm</td>
<td>-22.6</td>
<td>-87.5</td>
<td>-46.8 → -21.5</td>
</tr>
<tr>
<td>1 cm</td>
<td>-22.6</td>
<td>-87.5</td>
<td>-47.6 → -21.3</td>
</tr>
<tr>
<td>1 mm</td>
<td>-22.8</td>
<td>-87.4</td>
<td>-68.7 → -10.7</td>
</tr>
</tbody>
</table>

through the spacecraft due to conduction and the temperature will not be distributed as much. If the thickness is greater there is a higher conduction and the heat is more equally distributed throughout the spacecraft. Figure 3.8 shows the temperature distribution over the spacecraft with a thickness of 1.21 cm for the hot steady state case with the Sun illuminating the side with the smallest area while Figure 3.9 shows the transient temperature variation during ten orbits around the Earth. In comparison with Figure 3.7 it takes about eight orbits for the temperature variation to be stable and the total temperature is lowered to about $-40^\circ C$ to $-46^\circ C$ which is intuitive since the area illuminated by the Sun is three times less than for the case in Section 3.4.1.
Chapter 3. Preliminary Thermal Analysis

Figure 3.8: Temperature distribution over the spacecraft with a thickness of 1.21 cm for the hot steady state case with the Sun illuminating the side with the smallest area, i.e. the XC - YC plane in the ZC direction.

Figure 3.9: Temperature variation for the spacecraft with thickness 1.21 cm while being illuminated by the Sun towards the side with the smallest area during ten (10) orbit periods.
3.5 Decrease in Temperature During Eclipse

The decrease in temperature during eclipse has also been studied in order to investigate how fast the temperature decreases as a function of time. Considering the spacecraft in eclipse, the energy balance reads: Rate at which heat is gained - rate at which heat is lost = rate at which heat is stored, which is given by:

\[ mC \frac{dT}{dt} = Q_{IR} - A_{sc} \epsilon \sigma (T^4 - T_a^4) \] (3.13)

where \( Q_{IR} \) is given by \textbf{EQUATION 3.5}, \( m \) is the mass of the spacecraft, \( C \) the specific heat capacity, \( A_{sc} \) is the area of the spacecraft, \( T_a \) is the ambient temperature of the environment around the spacecraft, \( T \) is temperature and \( t \) is time. By defining:

\[ A = \frac{A_{sc} \epsilon \sigma}{mC} \] (3.14)

\[ B = \frac{Q_{IR} + A_{sc} \epsilon \sigma T_a^4}{mC} \] (3.15)

By rearranging \textbf{EQUATION 3.13} and taking the integral of it:

\[ \int_{T_0}^{T} \frac{dT}{B - AT^4} = \int_{0}^{t} dt \] (3.16)

where \( T_0 \) is the temperature the spacecraft has when entering eclipse. The primitive function, \( I \), to the left integral in \textbf{EQUATION 3.16} is given by:

\[ I = \frac{- \ln \left( \sqrt{B} - \sqrt{AT} \right) + \ln \left( \sqrt{AT} + \sqrt{B} \right) + 2 \arctan \left( \frac{\sqrt{AT}}{\sqrt{B}} \right)}{4 \sqrt{AB^{3/4}}} \] (3.17)

With this, \textbf{EQUATION 3.16} can be solved numerically which has been done for ten different values on the emissivity, \( \epsilon \) which can be studied in \textbf{FIGURE 3.10}. As can be seen the decrease in temperature is less for lower values of \( \epsilon \) which is intuitive since the spacecraft then radiates less heat to the surrounding environment. The case where \( \epsilon = 0.7 \) has been specifically studied in order to be able to compare with previous analysis where \( \epsilon = 0.7 \)
Figure 3.10: Decrease in temperature for ten different values on emissivity, $\epsilon$

Figure 3.11: Decrease in temperature for the spacecraft as a function of time where $\epsilon = 0.7$

has been used. Figure 3.11 shows the decrease in temperature for this case. It can be noted that after two hours in eclipse the temperature has decreased to about $-25^\circ C$. With the orbit presently defined, the spacecraft will be in eclipse for a period of approximately 37 minutes which means that the temperature will have decreased to $5^\circ C$. However, the approximation made here is that the entire spacecraft sees deep space and
hence the temperature $T_a = 2.7K$. In reality this is not true since approximately one quarter of the spacecraft will see Earth. The spacecraft will be heated by the Earth’s infrared radiation and average temperature in orbit due to Earth’s IR is $248K$. If we approximate that half the spacecraft will see this IR and the other half the temperature of deep space of $2.7K$ the average ambient temperature of the spacecraft is $T_a = 64K$. This change in ambient temperature leads to that it will take less than one minute longer for the spacecraft to reach a temperature of $-30^\circ C$. The MATLAB script for calculating and generating the graphs can be found in Appendix A.
Chapter 4

Printed Circuit Boards

Since the spacecraft will consist mostly of Printed Circuit Boards (PCB) a large part of the project has been put into investigating and understanding the thermal behavior of PCBs. The radiation exchange between two PCBs and the "wall" of the spacecraft has been investigated both analytically and numerically. Furthermore, a PCB Thermal Conductivity Calculator has been developed where the user can input the number of copper layers, the ratio of copper in those layers and the total thickness of the PCB and get the average thermal conductivity through the PCB. To be able to model the thermal contact conductance between the PCBs and the main structure of the spacecraft a thermal vacuum chamber experiment has been conducted, all of this is explained in this chapter.

4.1 Three-Surface Radiation Enclosure

4.1.1 Analytical Analysis

An analytical calculation of the radiation between two PCBs and the wall has been performed using a three-surface enclosure approach. Consider Figure 4.1 and 4.2 where surface 1 and 2 represent one PCB each (it does not matter which is the top or the bottom PCB since we approximate this as a symmetrical problem) and surface 3 represents the wall (in reality there are four walls but again because of the symmetry it can be approximated as one wall with the area of the four real walls, i.e. the area that the
Figure 4.1: Schematic of a three-surface enclosure and the radiation network associated with it [4]

Figure 4.2: Drawing of the radiation exchange between the two PCBs and the wall.

PCB can see). In Figure 4.1 $\dot{Q}_i$ is the net rate of radiation heat transfer from surface $i$ and $\dot{Q}_{ij}$ is the net rate of radiation heat transfer from surface $i$ to surface $j$. $E_{bi} = \sigma T_i^4$ and $J_i$ is the radiation emitted by surface $i$ plus radiation reflected by surface $i$. Recall from Section 2.3 that $F_{ij}$ is the view factor from surface $i$ to surface $j$. At each node the sum of the currents, which is the net radiation heat transfer, must be equal to zero. This gives the following system of equations:
Chapter 4. Printed Circuit Boards

(4.1)

\[ \frac{E_{b1} - J_1}{R_1} + \frac{J_2 - J_1}{R_{12}} + \frac{J_3 - J_1}{R_{13}} = 0 \]
\[ \frac{J_1 - J_2}{R_{12}} + \frac{E_{b2} - J_2}{R_2} + \frac{J_3 - J_2}{R_{23}} = 0 \]
\[ \frac{J_1 - J_3}{R_{13}} + \frac{J_2 - J_3}{R_{23}} + \frac{E_{b3} - J_3}{R_3} = 0 \]

Furthermore, the view factors are given by [14]:

\[ F_{12} = \frac{2}{\pi XY} \left( \ln \left[ \frac{(1 + X^2)(1 + Y^2)}{1 + X^2 + Y^2} \right]^{1/2} + X(1 + Y^2)^{1/2} \arctan \frac{X}{(1 + Y^2)^{1/2}} + Y(1 + X^2)^{1/2} \arctan \frac{Y}{(1 + X^2)^{1/2}} - X \arctan X - Y \arctan Y \right) \]  

(4.2)

and

\[ F_{13} = 1 - F_{12} \]
\[ F_{21} = \frac{A_1}{A_2} F_{12} \]
\[ F_{23} = 1 - F_{21} \]  

(4.3)

where

\[ X = \frac{x}{L} \]
\[ Y = \frac{y}{L} \]  

(4.4)

where \( x \) and \( y \) are the length and width of the PCBs and \( L \) is the spacing between them. Solving EQUATIONS 4.1 for \( J_1, J_2 \) and \( J_3 \) one gets the net rate of radiation heat transfer to and from a surface, given by:

\[ \dot{Q}_i = \frac{E_{b_i} - J_i}{R_i} \]  

(4.5)
and the net radiation heat transfer between two surfaces as:

\[ \dot{Q}_{ij} = \frac{E_{bi} - E_{bj}}{R_i + R_{ij} + R_j} \]  

(4.6)

As observed from Figure 4.1 and Equations 4.1 - 4.6 the net radiation heat transfer is dependent on the emissivity of the PCBs and the emissivity of the wall.

![Diagram of net radiation heat transfer](image)

**Figure 4.3:** The net radiation heat transfer from PCB to the wall in the case where the temperature on the PCBs are 25°C and the temperature of the wall is 35°C. Negative value indicates reverse direction of heat flow.

By solving the equations and plotting the results one can study the dependence of the emissivities closer. Figure 4.3 displays the net radiation heat transfer to/from one PCB and the wall respectively as a function of the emissivity of the PCBs and the emissivity...
of the wall. It can be noticed that as the emissivity of the PCB and the emissivity of the wall increase the wall radiates and the PCB radiates more heat respectively. In Figure 4.4 it can be observed that the radiation heat exchange between the two PCBs \(Q_{12}\) is zero, this has to do with the mentioned symmetry. For the radiation heat exchange between the PCBs and the wall on the other hand it has the same behavior as for the top graph in Figure 4.3. That is, the PCB absorbs the heat from the wall better as the emissivity of the PCB and the wall increase.
4.1.2 Analysis in Siemens NX™

In order to verify the analytical results in Section 4.1.1 a model was built in Siemens NX™ and solved for five different values on the emissivity. Figure B.1 in Appendix B displays the net radiation flux of one node of a PCB for the different values of the emissivity. It can be observed that for higher emissivity the greater net radiation flux there is. Positive flux means that heat is leaving the PCB while negative flux means that heat is added to the PCB. It is important to understand that this result only shows radiation flux from one node of the PCB and not the total radiation exchange between surfaces. Also worth noting here compared with the analytical calculations is that for the analytical analysis the temperatures of the surfaces was fixed while in the numerical steady state solution, the solution is run until all temperatures have reached equilibrium. The temperature fluctuation is exactly what can be seen in Figure B.1 and is the reason why the radiation flux varies. Furthermore, the relation between the different values of emissivity can be observed where it is clear that the higher emissivity the surface has, the greater radiation flux there is.

4.2 PCB Thermal Conductivity Calculator

Since a large part of the spacecraft will consist of PCBs the thermal conditions of the spacecraft will be dependent on the thermal properties of the PCB. Therefore it was decided to investigate the structure and thermal properties of the PCB further. A PCB is made by alternating copper layers with insulating layers of FR4. The outer copper layers often have a thickness of 35µm while the inner layers have a thickness of 17.5µm. Two of the inner layers are often used for grounding and power which means that these layers are consisting of as good as 100% copper. The thermal conductivity of a PCB is dependent of how many copper layers the stack consists of, how much copper each layer consists of and the total thickness of the PCB. A MATLAB script, see Appendix A, has been developed where the thermal conductivity of a PCB can be calculated and where the user can input the number of layers wanted and how many percent copper each layer consists of. The insulating material is FR4. In the script the following assumption is made: Each copper layer will have parallel traces (conductors) where all traces on one layer is all in the same direction. The layer next to another layer will have its
traces orthogonal to the traces of the first layer. This means that two copper layers are approximated to be one layer and that the thermal conductivity is isotropic throughout the plane. This gives how much copper the entire PCB consist of and from there the thermal conductivity can easily be calculated by taking the fraction of copper times the thermal conductivity of copper plus the fraction of FR4 times the thermal conductivity of FR4.

By assuming eight copper layers with 50% copper on the outer layers, 40% copper on the inner layers and a total thickness of the PCB of 1.6\,mm the average thermal conductivity through the PCB is calculated to 20.5\,W/mK. Using the same approach for calculating the average density and specific heat of the PCB resulted in an average density of $\rho = 2223\,kg/m^3$ and an average specific heat of $C_p = 589\,J/kgK$. These values have been used from now on in the thermal model of the spacecraft MIST where an average material with these properties has been defined and applied to model the PCBs.

### 4.3 Measurement of PCB Mounting Conductance

Each PCB in the satellite will be mechanically connected with the main structure of the satellite (railings in the thermal model). The thermal conduction interface between the main structure and the PCBs will greatly determine the temperature on a PCB. Therefore the conductance between a Printed Circuit Board (PCB) and the mounting screw was measured in order to more accurately model the thermal coupling between a PCB and the mounting point in the thermal model. The experiment has taken place in a vacuum chamber in order to not have any convection. A MATLAB script of the calculations performed and measurement values taken can be found in Appendix A.

#### 4.3.1 Set-Up

A heating wire was wrapped around a PCB which was mounted to a metal plate which in turn was connected to the inside wall of the vacuum chamber and used as a heat sink in order to get a heat flow from the heating wire through the PCB, the screw and the metal plate to the wall of the vacuum chamber. The screw goes through a sleeve which physically separates the PCB from the metal plate. Two NTC thermistors [15]
were placed on the PCB and two on the metal plate. The NTC thermistors measure the resistance which can be translated to temperature with the following formula:

\[
T = \frac{1}{\log\left(\frac{R}{R_{25}}\right) + \frac{1}{T_{25}}} 
\]

where \( R_{25} = 5k\Omega \) is the resistance at the standard temperature of \( 25^\circ C \), \( B_{25/100} = 3497K \) represent the temperature coefficient, \( T_{25} = 298.15K \) and \( R \) is the measured resistance in \( \Omega \).

Multi Layer Insulation (MLI) was wrapped around the PCB in order to avoid radiation exchange between the PCB and the metal plate.

4.3.1.1 Theoretical Calculations

\[ \text{Figure 4.5: The left hand sketch shows the conductance from the PCB - screw - metal plate while the right hand sketch shows the conductance PCB - sleeve - metal plate} \]

In order to know how much current that should be sent through the heating element a theoretical calculation and estimation of the conductance was performed. The current is given by:
where \( R_h = 26.6 \Omega \) is the resistance of the heating element and \( Q \) is the power given by:

\[
Q = G_{tot} dT
\]

(4.9)

where \( dT \) is the temperature difference between the PCB and the metal plate (here chosen to 50\(^\circ\)K) and \( G_{tot} \) is the total conductance from the PCB, through the screw to the metal plate and is the sum of the conductance through the screw and the conductance that are going through the sleeve that physically separates the PCB and the metal plate, this can be summarized as:

\[
G_{tot} = G_{Screw} + G_{Sleeve}
\]

(4.10)

where

\[
G_{sleeve} = \frac{1}{\frac{1}{G_{slc}} + \frac{1}{G_{slb}}}
\]

(4.11)

where \( G_{slc} \) is the conductance given by the connecting interface between the edge of the sleeve and the PCB respectively the metal plate and \( G_{slb} \) is the conductance going through the sleeve. In the same way the conductance through the screw is given by:

\[
G_{screw} = \frac{1}{\frac{2}{G_{sch}} + \frac{1}{G_{scb}} + \frac{1}{G_{scc}}}
\]

(4.12)

where \( G_{sch} \) is the conductance through the interface between the screw head, \( G_{scb} \) the conductance through the screw body and \( G_{scc} \) the conductance through the interface between the screw and the metal plate which is all visualized in Figure 4.5.

The conductance is given by:
\[ G_i = A_i h_c \]

or

\[ G_i = A_i K \frac{dx}{dx} \]  

(4.13)

where \( h_c \) is the thermal contact conductance coefficient \([W/\text{m}^2\text{K}]\), \( K \) is the thermal conductivity coefficient \([W/\text{mK}]\), \( A_i \) the cross sectional area or the contact area between two interfaces and \( dx \) is the height of the object. In these calculations the thermal contact conductance coefficient has been taken to be equal to 450. This value will vary depending on the contact pressure which is the factor of most influence on contact conductance. As contact pressure grows, contact conductance grows. The second of the two equations in Equation 4.13 has been used for the conductance calculations through the screw and through the sleeve, in this case the thermal conductivity coefficient for Aluminum has been used \((K_{Al} = 170 W/mK)\). These calculations resulted in total conductance between the PCB and the metal plate of \( G_{tot} = 0.0167 W/K \) and the current that should be used is calculated to be \( I = 0.177 A \).

4.3.2 Calculations

Since the thermistors have some small errors relative to each other they where calibrated against each other where the thermistor on the PCB was used as the reference thermistor. The thermistors will follow a resistance/temperature curve (not presented here) where each thermistors curve will have a deviation from each other. The calibration between them was managed in the sense that each thermistor curve was moved (up or down) with the factor seperating them from the reference thermistor curve. This gives the following equation for the temperature:

\[ T_i = \log\left(\frac{R_{i}}{R_{25}}\right) \frac{1}{R_{25/100}} + \frac{1}{T_{25}} \pm (T_{0i} - T_{0ref}) \]  

(4.14)

where \( T_0 \) is the temperature of the thermistors in room temperature. Furthermore, the conductance is given by:
\[ G = \frac{Q}{dT} \]  

(4.15)

where

\[ Q = R_h I^2 \]  

(4.16)

and \( dT \) is the temperature difference between the PCB and the metal plate, \( R_h = 26.6 \Omega \) is the resistance of the heating element and \( I \) is the current sent through the heating element. Since the conductance between the PCB and the metal plate is dependent on how tightly the screw has been screwed, i.e. the pressure and the moment of the screw, several experiments with different pressure have been conducted. In order to be able to measure the torque in some way the angle the screw had been tightened was measured. The maximum angle was when the screw had been tightened as much as possible. The minimum angle was defined where the PCB is still firmly connected to the metal plate but not tightened to the maximum.

### 4.3.3 Results

In Figure 4.6 the contact conductance between the PCB and the metal plate, connected through a screw with associated sleeve, has been plotted. Also the least mean square fit is shown. As can be observed there are some variations in the conductance as the screw is being tightened. However, the result seems to be somehow ambiguous, theoretically, as the screw is tightened and the contact pressure between the PCB and the metal plate is increased, the conductance should increase. As seen in Figure 4.6 for the angles 10 and 20 degrees the corresponding result is questionable. The mean value of the contact conductance is \( G_{mean} = 0.0160 \text{W/K} \) which is within 5\% of the theoretically calculated value.

### 4.3.4 Behavior of Thermistors

As mentioned, two thermistors were placed on the PCB and two on the metal plate. Nevertheless, it was noticed that the temperature difference for the two thermistors on
Chapter 4. Printed Circuit Boards

Figure 4.6: The contact conductance between a PCB and a metal plate through a screw with an associated sleeve as a function of the angle the screw has been tightened, symbolizing the torque.

The PCB differed about 5° C from each other, while the two thermistors on the metal plate differed only about 0.5° C. It order to verify the relationship between resistance and temperature the resistance of the thermistors was measured for known values of the temperature.

Figure 4.7 shows the measure resistances at the know temperatures. Also note that the theoretical behavior/curve is also shown in the figure. It can be seen in the figure that the thermistors are following the theoretical behavior quite well which means that their behavior is as expected.

4.3.5 Error Analysis

The propagation of error for a function \( y = f(x_1, x_2, \ldots) \) is given by:

\[
\Delta y^2 = \left( \frac{\delta f}{\delta X_1} \Delta x_1 \right)^2 + \left( \frac{\delta f}{\delta X_2} \Delta x_2 \right)^2 + \ldots
\]  

(4.17)
Figure 4.7: The relation between the resistance of the thermistors and temperature and the propagation of error for a function $z = x + y + \ldots$ is given by:

$$\Delta z = \sqrt{\Delta x^2 + \Delta y^2 + \ldots} \quad (4.18)$$

The conductance is given by the substitution of Equation 4.16 into Equation 4.15 giving:

$$G = \frac{R_hI^2}{dT} \quad (4.19)$$

Using Equation 4.17 with $z = G$ and $x_1 = R_h, x_2 = I, x_3 = dT$ and combining this with Equation 4.18 the following equation is obtained:

$$\Delta G = \sqrt{\left(2R_hI\frac{\Delta I}{dT}\right)^2 + \left(R_hI^2\frac{\Delta dT}{dT^2}\right)^2 + \left(\frac{I^2\Delta R_h}{dT}\right)^2 + \Delta x^2} = 2.002\% \quad (4.20)$$
where $\Delta I$ is the error of the current reading which is about 0.5%, $\Delta dT$ is the error of the thermistors (2%), $\Delta R_h$ the error in the resistance of the heating element (4%) and $\Delta x$ is the approximated reading error of 2%. This gives an error in the conductance of about 2%.

### 4.3.6 Conclusion

The conductance measured as a function of the angle of how much the screw had been tightened (the torque) seems to contain some errors or ambiguities for low values on the angle. As the torque is increased the conductance should increase. The reason for the fluctuation in the result can have to do with that the screw was not tightened hard enough to start with and that the PCB plate moved when wrapping around the MLI or connecting it to the vacuum tank etc. It was very hard to have a measurable interval of the tightening of the screw (from loose to hard) since there was a very small interval from where the screw was tightened firmly to where it was impossible to tighten the screw any more. Another reason can be that the pressure load was not uniform. Other sources of errors includes that the MLI was not covering the entire PCB so that some radiation exchange took place between the PCB and the metal plate. However, averaging the experimental result shows that it is in good accordance with the theoretical result.
Chapter 5

Thermal Analysis

This chapter will go more into details of how the detailed thermal analysis has been performed, i.e. how the spacecraft thermal model has been built. For the thermal analysis an orbit of 640 km altitude with an eccentricity of 0.001, orbital inclination of $97.943^\circ$, argument of perigee $0^\circ$ and an orbital period of 5851 sec has been used which is the reference orbit one which is one of the two proposed orbits for the spacecraft MIST and can be seen in Figure 5.1. The MATLAB codes for the calculations of the thermal couplings mentioned in this chapter can be found in Appendix A under “Thermal Coupling Conductances Calculations” and the material properties can be found in Appendix C.

Figure 5.1: Spacecraft in orbit viewed from the Sun.
5.1 Building the Thermal Model

All thermal couplings from a PCB to the satellite structure (rails) are taken from the results from Section 4.3. The thermal conductivity through a PCB is modeled according to the result in Section 4.2. The optical properties mentioned in this chapter and used for the meshes can be found in Table 5.1. A complete overview of the payloads and subsystems location within the spacecraft and a comparison of the CAD model and the meshed model can be seen in Figure B.2 in Appendix B.

Table 5.1: Optical properties where $\alpha$ is absorptivity, $\epsilon$ is emissivity.

<table>
<thead>
<tr>
<th>Optical property name</th>
<th>$\alpha$</th>
<th>$\epsilon$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aluminum</td>
<td>0.15</td>
<td>0.05</td>
</tr>
<tr>
<td>Copper oxidized</td>
<td>0.9</td>
<td>0.8</td>
</tr>
<tr>
<td>Glass</td>
<td>0.75</td>
<td>0.83</td>
</tr>
<tr>
<td>Titanium</td>
<td>0.52</td>
<td>0.1</td>
</tr>
<tr>
<td>Silicon</td>
<td>0.6</td>
<td>0.9</td>
</tr>
<tr>
<td>Steel</td>
<td>0.5</td>
<td>0.25</td>
</tr>
<tr>
<td>Brass</td>
<td>0.096</td>
<td>0.03</td>
</tr>
<tr>
<td>Polypropylene</td>
<td>-</td>
<td>0.97</td>
</tr>
</tbody>
</table>

5.1.1 Rails

The rails, which is the main supporting structure of the spacecraft, shall be of anodized hard aluminum according to the CubeSat Design Specifications [16]. For the thermal model these rails have been modeled as Aluminum 6061 which has a thermal conductivity of 170 $W/mK$. When meshing a surface the thermal engineer has to define the material and also its thermo-optical properties, such as emissivity and absorptivity, $\epsilon$ and $\alpha$, together with thickness (for a 2D mesh, which is used in this model). The thermo-optical properties for the rails are $\epsilon = 0.05$, $\alpha = 0.15$.

Figure 5.2 shows the four rails of the spacecraft together with two PCBs. In this model the ladders holding these four rails together have not been geometrically modeled. These connections have thermally been defined in Siemens NX™ using thermal couplings. Figure 5.2 displays these thermal couplings at two levels in the satellite for simplicity but there are actually four levels with ladders and thermal couplings between the rails. This means that the rails are thermally coupled and heat can be transferred between them. When defining thermal coupling the thermal engineer needs to input the total
Figure 5.2: Spacecraft rails and PCB seen with thermal coupling between each other

conductance from one rail to the other. The cross sectional area of one ladder (which is symbolized as a thermal coupling between the rails) is taken as a square with sides of 8.5 mm. The distance between the rails is $dx = 97.6$ mm and the thermal contact conductance is taken as $h_c = 400$ W/m²K. Using Equation 2.3, 2.4 and 2.5 the total thermal conductance through one ladder is 0.013 W/K. Please also note that Figure 5.2 displays the thermal couplings between a PCB to the rails, as explained in Chapter 4, and the thermal coupling from one PCB to another which is symbolized by the pin connectors between the PCBs.

5.1.2 Solar Panels

During this thesis project a decision was taken to choose the solar panel configuration shown in Figure 5.3. Since the spacecraft will need as much energy as it can get the spacecraft will have solar panels on every side except for on the top and bottom since the bottom will host a propulsion system and the top a camera and antennas. However there will be one section (10x10cm²) on one of the long sides where there will be no solar panels since the experiment Ratex-J will need a free view out to space. Due to late
Figure 5.3: Spacecraft solar panels

decision making the free view to space has not yet been modeled, the model still has solar panels covering Ratex-J. However, explained further down the Ratex-J experiment has been modeled to see free space when it comes to orbital heating. The solar arrays have a total thickness of about 1.7 mm with a thin (\(\sim 100 \mu m\)) cover glass [17]. The backside of solar panels are often covered with a Copper fill and the underside of the cells is flooded with vias (Latin for path or way, is an electrical connection between layers) for thermal conductivity purposes [18]. The solar panels covering the spacecraft (excluding the deployable solar panels) have therefore been modeled with two meshes, one for the front and one for the backside. The front mesh has the properties of glass and the back mesh (the mesh towards the inside of the spacecraft) the properties of Copper. Furthermore, thermal coupling between the front and the back mesh was defined and total conductance through the front to back side of all the four solar panels was calculated to be 20 W/K. Each solar panel is connected to the fictional ladders which have been modeled with thermal coupling between the mesh elements appearing where the fictional ladders are located and the geometric rails. The cross sectional area of the ladder rails that is in contact with the solar panels are the sides facing out towards the solar panels.

Figure 5.3 shows the meshed model of the rails and solar panels with front side in blue and backside in black. In reality the ladders will not be in direct contact with the thin
copper surface but instead an FR4 material. However there are thermal vias between the copper layer and the interface with the ladder in order to increase the thermal interface between the solar panels and the ladders. In this model this has not been taking into consideration but instead it is modeled as if the thin copper layer was in direct contact with the ladder. This interface can in turn be manipulated by changing the contact conductance between them but here the value was set to $h_c = 200$. Furthermore, the thermal coupling conductance between the backside of the solar panels and the rails was calculated to $0.41 \, W/K$.

The deployable solar panels will have solar cells on both sides and for that reason the deployable solar panels have been modeled as one mesh with an average material consisting of glass, FR4 and Copper. The average material consists of $200 \, \mu m$ glass, $140 \, \mu m$ Copper and $1.6 \, mm$ FR4. This resulted in an average conductivity of $34 \, W/mK$, a density of $2553 \, kg/m^3$ and an average specific heat of $606 \, J/kgK$. The area of the hinges that are in thermal contact with each panel are about $19 \times 7.5 \, mm^2$ large. With a maximum contact conductance of $h = 500 \, W/m^2K$ this will give a thermal coupling between the deployable solar panels and the spacecraft of $0.07 \, W/K$ per hinge.

5.1.3 Radiation

In order to investigate the impact of radiation exchange between the bodies within the spacecraft two cases was studied. One where an inner radiation was defined and one with no radiation exchange between the bodies within the spacecraft. Orbital heating was defined on all surfaces that see deep space. This means that heating from the Sun, albedo and Earth IR are taking into account. Furthermore, radiation was defined on all the outer surfaces so that these can exchange radiation with each other. Also when defining radiation in Siemens NX™ the thermal engineer tells the software to calculate the view factors between those surfaces.

5.1.4 Nanospace

The Nanospace experiments is one of the few experiments with a well defined and required position within the spacecraft. Nanospace is testing a propulsion system with four nozzles and therefore this experiment is placed at the bottom of the spacecraft. In
Chapter 5. Thermal Analysis

Figure 5.4: Nanospace experiment inside the spacecraft.

Table 5.2: List of mesh collectors for the Nanospace payload.

<table>
<thead>
<tr>
<th>Name</th>
<th>Mesh collector</th>
<th>Mtrl</th>
<th>Thickness (mm)</th>
<th>Optical property name</th>
</tr>
</thead>
<tbody>
<tr>
<td>Horizontal PCB</td>
<td>PCB</td>
<td>Average composition</td>
<td>1.6</td>
<td>Copper oxidized</td>
</tr>
<tr>
<td>Metal plate</td>
<td>Aluminum7075</td>
<td>Aluminum Alloy 7075</td>
<td>1</td>
<td>Aluminum</td>
</tr>
<tr>
<td>Vertical PCB</td>
<td>PCB</td>
<td>Average composition</td>
<td>1.6</td>
<td>Copper oxidized</td>
</tr>
<tr>
<td>Cover plate</td>
<td>Aluminum7075</td>
<td>Aluminum Alloy 7075</td>
<td>1</td>
<td>Aluminum</td>
</tr>
<tr>
<td>Tank</td>
<td>Nanospace tank</td>
<td>Titanium Alloy</td>
<td>4</td>
<td>Titanium</td>
</tr>
<tr>
<td>Nozzle</td>
<td>Nanospace Nozzle</td>
<td>Silicon</td>
<td>0.5</td>
<td>silicon</td>
</tr>
</tbody>
</table>

Figure 5.5: Schematics of the thermal couplings of the Nanospace payload.
order to simulate the thermal contact and the thermal heat exchange between the different parts thermal couplings have been defined between the parts that are in mechanical contact, this is between the rails and the cover plate, horizontal PCB, metal plate and to the vertical PCBs through a supporting structure (modeled within the thermal coupling). Furthermore, thermal couplings have been defined between the metal plate and the tank, the vertical PCBs and the nozzles, the horizontal PCB and the vertical PCBs (flexible connectors) and between the top and bottom meshes of each plate (the PCBs, metal plate and cover plate). When it comes to the orbital heating the nozzles will be affected and that is why orbital heating has been defined at the bottom side of them together with the cover plate. See Table 5.3 for the values used for the conductance of the thermal couplings and Figure 5.5 for the schematics of the thermal couplings.

Nanospace will have an idle power consumption of $50 \text{ mW}$ which has been modeled as a uniformed distributed thermal heat load on the entire structure except for the tank. The power consumption when the thrusters are being fired is between 500 $\text{ mW}$ and 3200 $\text{ mW}$ depending on the number of thrusters fired and the control of them [19]. Since neither the time when nor for how long the thrusters will be fired is decided yet these power consumptions and hence thermal heat loads are not taken into consideration of the thermal model. In order to account for the specific heat capacity, which is a measure of the thermal inertia, the correct mass of each object has been specified. This has been done by making the mesh of an object as thick as it needs to be to have the mass it is supposed to have, or by adding non-structural mass to the mesh. In the Nanospace experiment case the most of the mass will be on the titanium tank and that is also why the thickness of the tank has been calculated and defined to be 4 $\text{ mm}$ thick.

Table 5.3: Thermal couplings for the Nanospace experiment

<table>
<thead>
<tr>
<th>Coupling between objects</th>
<th>Conductance $[\text{W/K}]$</th>
<th>Total / per element</th>
</tr>
</thead>
<tbody>
<tr>
<td>PCB vertical and satellite structure</td>
<td>0.03</td>
<td>Total</td>
</tr>
<tr>
<td>PCB vertical and nozzles</td>
<td>0.288</td>
<td>Total</td>
</tr>
<tr>
<td>Metal plate and satellite structure</td>
<td>0.0238</td>
<td>Total</td>
</tr>
<tr>
<td>Metal plate and tank</td>
<td>0.09</td>
<td>Total</td>
</tr>
<tr>
<td>PCB horizontal and satellite structure</td>
<td>0.016</td>
<td>Per element</td>
</tr>
<tr>
<td>Cover plate and satellite structure</td>
<td>0.04</td>
<td>Total</td>
</tr>
<tr>
<td>Flex rigid connectors</td>
<td>0.0001</td>
<td>Total</td>
</tr>
</tbody>
</table>
Table 5.4: List of mesh collectors for the Ratex-J payload.

<table>
<thead>
<tr>
<th>Name</th>
<th>Mesh collector</th>
<th>Mtrl</th>
<th>Thickness (mm)</th>
<th>Optical property name</th>
</tr>
</thead>
<tbody>
<tr>
<td>Electronic boxes</td>
<td>Ratex-J Al7075</td>
<td>Aluminum Alloy 7075</td>
<td>3</td>
<td>Aluminum</td>
</tr>
<tr>
<td>PCB</td>
<td>PCB</td>
<td>Average composition</td>
<td>1.6</td>
<td>Copper oxidized</td>
</tr>
<tr>
<td>Detectors</td>
<td>Ratex-J Al7075</td>
<td>Aluminum Alloy 7075</td>
<td>3</td>
<td>Aluminum</td>
</tr>
</tbody>
</table>

5.1.5 Ratex-J

Ratex-J has been modeled as in Figure 5.6 where the two vertical plates are PCBs and the rest is modeled as Aluminum 7075. Thermal couplings have been defined between the two PCBs, between the cylindrical detectors and the PCBs, between the electronics boxes and between the electronics boxes and the rails with the values in Table 5.5, the schematics of the thermal coupling can be found in Figure 5.7. It is unknown how it looks like inside the electronics boxes but to account for the specific heat capacity they have been given a thickness of 3 mm.
Table 5.5: Thermal couplings for the Ratex-J experiment

<table>
<thead>
<tr>
<th>Coupling between objects</th>
<th>Conductance $[W/K]$</th>
<th>Total / per element</th>
</tr>
</thead>
<tbody>
<tr>
<td>PCBs to satellite structure</td>
<td>0.016</td>
<td>Per element</td>
</tr>
<tr>
<td>Between electronics boxes</td>
<td>0.605</td>
<td>Total</td>
</tr>
<tr>
<td>Cylinder detectors to PCBs</td>
<td>0.10</td>
<td>Total</td>
</tr>
<tr>
<td>PCB to PCB</td>
<td>0.016</td>
<td>Per element</td>
</tr>
<tr>
<td>Electronics box to satellite structure</td>
<td>0.20</td>
<td>Per element</td>
</tr>
</tbody>
</table>

Figure 5.8: Morebac experiment inside the spacecraft.

Table 5.6: List of mesh collectors for the Morebac payload.

<table>
<thead>
<tr>
<th>Name</th>
<th>Mesh collector</th>
<th>Mtrl</th>
<th>Thickness (mm)</th>
<th>Optical property name</th>
</tr>
</thead>
<tbody>
<tr>
<td>PCB detection layer</td>
<td>PCB</td>
<td>Average composition</td>
<td>1.6</td>
<td>Copper oxidized</td>
</tr>
<tr>
<td>Microfluid layer</td>
<td>Polypropylene</td>
<td>Polypropylene</td>
<td>1.6</td>
<td>Polypropylene</td>
</tr>
<tr>
<td>PCB illumination layer</td>
<td>PCB</td>
<td>Average composition</td>
<td>6</td>
<td>Copper oxidized</td>
</tr>
<tr>
<td>Liquid chamber</td>
<td>Polypropylene</td>
<td>Polypropylene</td>
<td>6</td>
<td>Polypropylene</td>
</tr>
</tbody>
</table>

Figure 5.9: Schematics of the thermal couplings of the Morebac payload.
5.1.6 Morebac

The Morebac experiment will consist of (from bottom to top) one PCB illumination layer, microfluid layer and PCB detection layer. The box to the right in Figure 5.8 is the liquid chamber and actuation. At this stage the microfluid layer and the liquid chamber are estimated to be made of polypropylene which is a plastic suggested by the experimenter. Thermal couplings have been defined between the two PCBs (connectors), between the microfluid layer and the PCBs, between the liquid chamber and the PCB illumination layer and between the PCBs and the rails of the satellite (satellite structure). The power consumption of the Morebac experiment is not yet available and therefore there have been no thermal loads presently defined on the experiment. The thermal control of the spacecraft is explained in the next chapter. In order to give the Morebac experiment a total mass of 300 g the polypropylene mesh has been defined with a thickness of 6 mm which is also the estimated thickness of the microfluid layer according to the experimenter. Also the microfluid layer will consist of mostly polypropylene and some small compartments where the bacteria will be.

Table 5.7: Thermal couplings for the Morebac experiment

<table>
<thead>
<tr>
<th>Coupling between objects</th>
<th>Conductance [W/K]</th>
<th>Total / per element</th>
</tr>
</thead>
<tbody>
<tr>
<td>PCB to satellite structure</td>
<td>0.016</td>
<td>Per element</td>
</tr>
<tr>
<td>PCB connectors</td>
<td>0.0257</td>
<td>Total</td>
</tr>
<tr>
<td>PCBs to microfluid layer</td>
<td>1.35</td>
<td>Total</td>
</tr>
<tr>
<td>Liquid chamber to illumination layer</td>
<td>0.675</td>
<td>Total</td>
</tr>
</tbody>
</table>

5.1.7 SEU

SEU is modeled as a PCB with thermal coupling to the rails of 0.016 W/K, according to the result from the measurement of the PCB mounting conductance, for each of the corners of the PCB. SEU will have a continuous power consumption of 1.8 W which has, from a thermal point of view, been modeled as a uniformed distributed thermal heat load on the PCB.

5.1.8 On Board Computer

Also the On Board Computer (OBC) is modeled as a PCB with thermal coupling to the rails of 0.016 W/K. The power consumption of the OBC is very unclear at this stage,
however the consumption has been estimated from the SEAM project where the power consumption was $1.7\, W$ which has been modeled as a uniform distributed thermal heat load on the OBC.

### 5.1.9 Spacelink

![Figure 5.10: Transceiver position inside the spacecraft.](image)

**Table 5.8:** List of mesh collectors for the Spacelink.

<table>
<thead>
<tr>
<th>Name</th>
<th>Mesh collector</th>
<th>Mtrl</th>
<th>Thickness (mm)</th>
<th>Optical property name</th>
</tr>
</thead>
<tbody>
<tr>
<td>Transceiver</td>
<td>Steel</td>
<td>Steel</td>
<td>0.2</td>
<td>steel</td>
</tr>
<tr>
<td>PCB</td>
<td>PCB</td>
<td>Average composition</td>
<td>6</td>
<td>Copper oxidized</td>
</tr>
</tbody>
</table>

![Figure 5.11: Schematics of the thermal couplings of the spacelink.](image)

**Table 5.9:** Thermal couplings for the spacelink

<table>
<thead>
<tr>
<th>Coupling between objects</th>
<th>Conductance [W/K]</th>
<th>Total / per element</th>
</tr>
</thead>
<tbody>
<tr>
<td>PCB to satellite structure</td>
<td>0.016</td>
<td>Per element</td>
</tr>
<tr>
<td>Transceiver to satellite structure</td>
<td>0.0286</td>
<td>Per element (one element)</td>
</tr>
<tr>
<td>Transceiver to PCB</td>
<td>0.75</td>
<td>Total</td>
</tr>
</tbody>
</table>

The spacelink is modeled as a PCB with the transceiver meshed as a $0.2\, mm$ thick Steel layer which the GOMspace spacelink has for thermal reasons. Furthermore, non-structural mass of $5\, kg/m^2$ has been defined and added to the model in order to account
for the specific heat capacity. The power dissipated can be radiated or conducted away. The transmitter has a thermal interface to one of the rails and that is also why one of the thermal couplings from the spacelink PCB to the railing is defined with a higher thermal conductance than the others. Nevertheless, thermal coupling has also been defined between the transceiver and the PCB which is given in Table 5.9. Also for the transceiver the power consumption has been taken from the SEAM project where the consumption was 0.5 W. This was modeled as a uniform distributed thermal heat load at the transceiver itself.

5.1.10 Battery

![Battery position inside the spacecraft.](image1)

**Table 5.10:** List of mesh collectors for the battery.

<table>
<thead>
<tr>
<th>Name</th>
<th>Mesh collector</th>
<th>Mtrl</th>
<th>Thickness (mm)</th>
<th>Optical property name</th>
</tr>
</thead>
<tbody>
<tr>
<td>Battery</td>
<td>Al7075 Batteries</td>
<td>Aluminum Alloy 7075</td>
<td>0.01</td>
<td>Aluminum</td>
</tr>
<tr>
<td>PCB</td>
<td>PCB</td>
<td>Average composition</td>
<td>6</td>
<td>Copper oxidized</td>
</tr>
</tbody>
</table>

![Schematics of the thermal couplings of the battery.](image2)
When it comes to the batteries there is one battery placed on top of a PCB and one battery on the bottom of the same PCB. Thermal couplings have been defined between the PCB and the rails (0.016 W/K per element) and between the batteries and the PCB with a total conductance of 0.85 W/K. The batteries are modeled and meshed as Aluminum 7075, which is what the GOMspace battery cover is made of.

5.1.11 Magnetorquer

![Figure 5.14: Magnetorquer position inside the spacecraft.]

![Table 5.11: List of mesh collectors for the magnetorquer.]

<table>
<thead>
<tr>
<th>Name</th>
<th>Mesh collector</th>
<th>Mtrl</th>
<th>Thickness (mm)</th>
<th>Optical property name</th>
</tr>
</thead>
<tbody>
<tr>
<td>Rod magnetorquer</td>
<td>Copper mgnt</td>
<td>Copper C10100</td>
<td>1.1</td>
<td>Copper oxidized</td>
</tr>
<tr>
<td>PCB</td>
<td>PCB</td>
<td>Average composition</td>
<td>6</td>
<td>Copper oxidized</td>
</tr>
<tr>
<td>Air magnetorquer</td>
<td>Copper mgnt</td>
<td>Copper C10100</td>
<td>1.1</td>
<td>Copper oxidized</td>
</tr>
</tbody>
</table>

![Figure 5.15: Schematics of the thermal couplings of the magnetorquer.]

The ISIS (Innovative Solutions In Space) magnetorquer consist of a PCB with two rod magnetorquers on the top and one air magnetorquer on the bottom of the PCB as can be seen in Figure 5.14. Since the magnetorquers are a toroid of Copper wire both the
rod and the air magnetorquer is defined and meshed as a 1.1 mm thick copper layer.
Non-structural mass of 3 kg/m² has also been defined to account for the specific heat
capacity. Furthermore, thermal couplings are defined between the magnetorquers and
the PCB and between the PCB and the rails with conductances according to Table 5.12. Presently the power consumption of the magnetorquer is unknown and is therefore
not considered at this point.

Table 5.12: Thermal couplings for the magnetorquer

<table>
<thead>
<tr>
<th>Coupling between objects</th>
<th>Conductance [W/K]</th>
<th>Total / per element</th>
</tr>
</thead>
<tbody>
<tr>
<td>PCB to satellite structure</td>
<td>0.016</td>
<td>Per element</td>
</tr>
<tr>
<td>Rods to PCB</td>
<td>0.056</td>
<td>Total</td>
</tr>
<tr>
<td>Air torquer to PCB</td>
<td>0.76</td>
<td>Total</td>
</tr>
</tbody>
</table>

5.1.12 SiC

The SiC experiment is modeled as half a PCB with thermal couplings to the rails of
0.016 W/K between each interface.

5.1.13 Camera

The important part to know about the Gomspace NanoCam is that the bottom plate is
a special-made heat sink. Heat will either radiate from the black surface of the anodized
aluminium plate or be transferred to the primary mechanical structure of the camera
via the mounting screws [20]. The heat sink has been given a non-structural mass of
10 kg/m². Thermal couplings have been defined between the PCB and the heat sink
to the railings, between the camera lens house and the PCB and finally between the
Table 5.13: List of mesh collectors for the camera.

<table>
<thead>
<tr>
<th>Name</th>
<th>Mesh collector</th>
<th>Mtrl</th>
<th>Thickness (mm)</th>
<th>Optical property name</th>
</tr>
</thead>
<tbody>
<tr>
<td>Camera lens glass</td>
<td>Camera Glass</td>
<td>Glass</td>
<td>3</td>
<td>Camera lens glass</td>
</tr>
<tr>
<td>Camera lens house</td>
<td>Camera house</td>
<td>Brass</td>
<td>3</td>
<td>Brass</td>
</tr>
<tr>
<td>PCB</td>
<td>PCB</td>
<td>Average composition</td>
<td>6</td>
<td>Copper oxidized</td>
</tr>
<tr>
<td>Heat sink</td>
<td>Camera Heat Sink</td>
<td>Aluminum 6061</td>
<td>1</td>
<td>Black body</td>
</tr>
</tbody>
</table>

Figure 5.17: Schematics of the thermal couplings of the camera.

Lenses glass and the lens house itself. Since most of the heat that the camera dissipates will be radiated from the heat sink a thermal heat load corresponding to the dissipated heat from the camera has been defined on the heat sink. The heat load is defined to be the constant value 600 mW which is the approximated heat produced by the NanoCam electronics [20] during operation. Presently it is not know how often or how much the camera will be in a operational mode but for this thermal model it has been assumed that the camera is always active, either taking a picture or processing it.

Table 5.14: Thermal couplings for the Camera

<table>
<thead>
<tr>
<th>Coupling between objects</th>
<th>Conductance [W/K]</th>
<th>Total / per element</th>
</tr>
</thead>
<tbody>
<tr>
<td>PCB to satellite structure</td>
<td>0.016</td>
<td>Per element</td>
</tr>
<tr>
<td>PCB to heat sink</td>
<td>0.016</td>
<td>Per element</td>
</tr>
<tr>
<td>Camera lens house to PCB</td>
<td>0.045</td>
<td>Total</td>
</tr>
<tr>
<td>Camera lens house to lens glass</td>
<td>0.04</td>
<td>Total element</td>
</tr>
</tbody>
</table>
5.1.14 Cubes

Cubes consists of a PCB with the dimensions of \(5cm \times 5cm\) and have thermal coupling defined between the railings and the PCB with a total conductance of \(0.016 \, \text{W/K}\) per element, in order to simulate the thermal interface between the objects.

5.1.15 LEGS

![Figure 5.18: The LEGS experiment position inside the spacecraft.](image)

<table>
<thead>
<tr>
<th>Name</th>
<th>Mesh collector</th>
<th>Mtrl</th>
<th>Thickness (mm)</th>
<th>Optical property name</th>
</tr>
</thead>
<tbody>
<tr>
<td>Legs motor</td>
<td>Al7075 LEGS</td>
<td>Aluminum Alloy 7075</td>
<td>5</td>
<td>Aluminum</td>
</tr>
<tr>
<td>Supporting plate</td>
<td>Aluminum7075</td>
<td>Aluminum Alloy 7075</td>
<td>1</td>
<td>Aluminum</td>
</tr>
<tr>
<td>Driver card</td>
<td>PCB</td>
<td>Average composition</td>
<td>6</td>
<td>Copper oxidized</td>
</tr>
</tbody>
</table>

![Figure 5.19: Schematics of the thermal couplings of the LEGs payload.](image)

The box seen in Figure 5.18 is representing the Piezomotor which is meshed as Aluminum 7075. Furthermore, the smaller plate seen is the driver card which in principle is a PCB. The larger plate is a Aluminum 7075 supporting plate for mechanical purposes.
Please note that this is not the real design of the LEGS experiment since in the writing moment the experimenters do not know themselves how the mechanical interface within the spacecraft will look. Therefore this is only an estimate of how it might look. The thermal interfaces have been modeled as thermal couplings between the driver card and the supporting plate, between the motor and the supporting plate and then between the supporting plate and the rails. Table 5.16 gives the total conductance for each thermal coupling. The driver card has a continuous power consumption of 0.5 W which has been modeled as a uniformly distributed thermal load. Furthermore, the motor has a power consumption of 0.25 W when active and zero when off. The motor will be periodically active for 2.5 minutes and then off for 2.5 minutes and so on. The behavior is explained by the following function:

\[ \text{Power} = \frac{0.25}{2} \cdot \text{sign} \left( \sin\left(\frac{2\pi t}{150}\right) \right) + \frac{0.25}{2} \]  

(5.1)

where \( t \) is time elapsed in seconds. However, the signum function seems not to be available in Siemens NX™ and the periodic behavior was instead modeled manually in the table constructor. To note here is that in the table constructor the user construct a table with times and power input at the specified time, the table constructor then does a linear interpolation between the input values. By defining the time of the orbital period as the last time, the software will see this as a continuous heat load and adapt it for each orbit.

Table 5.16: Thermal couplings for the LEGs experiment

<table>
<thead>
<tr>
<th>Coupling between objects</th>
<th>Conductance [W/K]</th>
<th>Total / per element</th>
</tr>
</thead>
<tbody>
<tr>
<td>PCB to supporting plate</td>
<td>0.25</td>
<td>Total</td>
</tr>
<tr>
<td>Motor to supporting plate</td>
<td>0.61</td>
<td>Total</td>
</tr>
<tr>
<td>Supporting plate to satellite structure</td>
<td>0.04</td>
<td>Total</td>
</tr>
</tbody>
</table>

5.2 Simplified Model

Figure B.3 in Appendix B shows the equilibrium temperature distribution within the spacecraft at time zero from the model described above, however, these temperatures will change during the orbit.
For this spacecraft as for many other each component and payload have specific thermal requirements. For the thermal engineer it can sometimes be difficult to know how detailed his/her model should be in order to be valid and still be a correct model. Since some experimenters have very specific and narrow temperature intervals they can operate within the thermal model presented in this report is made as thorough as possible within the scope of this master thesis and available information from the experimenters. However, it can be interesting to compare the more detailed model with a simplified model of the spacecraft where only the outer properties and heat dissipation from the interior of the spacecraft is taken into account.

Since the exterior of the spacecraft consists of solar panels (which has a 100 $\mu$m thick glass cover), the Nanospace Aluminum metal plate and the antenna support, the simplified model is a rectangular cube with the same dimensions as the detailed model where a mean material has been defined consisting proportionally of the density, thermal conductivity and specific heat of glass, aluminum and lead (antenna support plate). The simplified model then has a small plate inside the spacecraft where the total interior thermal load is defined. This thermal load is 8.2 $W$ which is the same as the detailed model has in total at the time of analysis. Furthermore, thermal couplings of 0.05 $W/K$ per element are defined from the plate with the thermal load to the main structure of the spacecraft at the same locations where the thermal couplings to the main structure are defined in the detailed model. The sum of the total conductance through the thermal couplings is equal to the sum of all the thermal couplings to the structure of the satellite in the detailed model in order to simulate the same conduction from the interior of the spacecraft to the exterior.

<table>
<thead>
<tr>
<th>Hot/cold case</th>
<th>Simplified, avg. temp. [°C]</th>
<th>Detailed, avg. temp. [°C]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hot case</td>
<td>12.9</td>
<td>9.0</td>
</tr>
<tr>
<td>Cold case</td>
<td>-3.3</td>
<td>-2.6</td>
</tr>
</tbody>
</table>

The average temperatures of the simplified model and the exterior of the detailed model for the hot and the cold case is given in Table 5.17. The temperature difference for the hot case between the simplified and detailed model is 3.9°C and the temperature difference for the cold case is as low as 0.7°C. This shows how valuable and accurate a simple model is if the thermal engineer only is interested in the mean temperature of the spacecraft. It also shows why it is important in the beginning of a project to make simple
preliminary thermal analysis in order to get a good estimate of the overall temperature of the spacecraft. However, in order to meet the payloads thermal requirements it is necessary to have a more detailed model of the spacecraft to know the temperatures at each specific payload. The MATLAB code for calculating the average material can be found in APPENDIX A under "Calculations for Simple Exterior Model of MIST".

# Chapter 6

## Thermal Control

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Req. Operational temp. [°C]</th>
<th>Req. Non-operational temp. [°C]</th>
<th>Temperature before thermal control</th>
<th>Req. reached?</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nanospace</td>
<td>+10 → +40</td>
<td>-10 → +50</td>
<td>-16 → +14</td>
<td>X</td>
</tr>
<tr>
<td>Ratex-J</td>
<td>-20 → +20</td>
<td>-40 → +50</td>
<td>+18 → +32</td>
<td>X</td>
</tr>
<tr>
<td>Morebac</td>
<td>+20 → +30</td>
<td>&gt; +4</td>
<td>+8 → +11</td>
<td>X</td>
</tr>
<tr>
<td>SEU</td>
<td>0 → +85</td>
<td>-65 → +150</td>
<td>+27 → +45</td>
<td>√</td>
</tr>
<tr>
<td>OBC</td>
<td>-40 → +60</td>
<td>-</td>
<td>+27 → +45</td>
<td>√</td>
</tr>
<tr>
<td>Transceiver</td>
<td>-30 → +60</td>
<td>-</td>
<td>+9 → +30</td>
<td>√</td>
</tr>
<tr>
<td>Batteries</td>
<td>-40 → +85</td>
<td>-</td>
<td>-7 → +18</td>
<td>√</td>
</tr>
<tr>
<td>Magnetorquer</td>
<td>-40 → +70</td>
<td>-</td>
<td>-7 → +15</td>
<td>√</td>
</tr>
<tr>
<td>SiC</td>
<td>-40 → +105</td>
<td>-</td>
<td>-2 → +30</td>
<td>√</td>
</tr>
<tr>
<td>Camera</td>
<td>0 → +60</td>
<td>-40 → +85</td>
<td>-1 → +25</td>
<td>X</td>
</tr>
<tr>
<td>Cubes</td>
<td>&lt; +25 ± 5</td>
<td>-20 → +80</td>
<td>+10 → +45</td>
<td>X</td>
</tr>
<tr>
<td>LEGS</td>
<td>+10 → +40</td>
<td>-30 → +70</td>
<td>+33 → +36</td>
<td>√</td>
</tr>
<tr>
<td>Solar Panels</td>
<td>-40 → +85</td>
<td>-</td>
<td>-33 → +67</td>
<td>√</td>
</tr>
<tr>
<td>Deployable Solar Panels</td>
<td>-40 → +85</td>
<td>-</td>
<td>-80 → +35</td>
<td>X</td>
</tr>
</tbody>
</table>

From the thermal analysis it was noted that not all subsystems reached the required temperature interval. In these cases the thermal engineer need to thermally control the spacecraft. There are several ways the thermal engineer can thermally control the spacecraft. He/she can for example use Multi Layer Insulation (MLI) in order to minimize the radiation heat transfer between bodies. He/she can increase or decrease the thermal conductivity between the interfaces of two bodies to increase or decrease the heat transfer between the bodies or he/she can try to change the emissivity and absorptivity.
of materials by for example changing materials or painting them. By adding heaters the thermal engineer can heat areas that are too cold or add radiators where it is too hot. In Table 6.1 the thermal requirements and the temperature the thermal analysis predicts are displayed. It can be observed that seven of the payloads/subsystems already fulfill their thermal requirements. However there are also five payloads/subsystems that do not fulfill the thermal requirement defined and there are payloads/subsystems that need to be thermally investigated further.

### 6.1 Multi Layer Insulation

Multi Layer Insulation (MLI) is a blanket consisting of normally between 10-20 reflective layers of low conducting materials with low emissivity. In general MLI consists of a outer layer, reflector layers and an inner layer together with insulating layers between each layer. The outer cover layer protects the rest of the layers from the space environmental effects and are resistant to flaking, shedding and other forms of particulate generation. The reflective layers typically reflects 90 to 99 percent of the radiation. By adding several layers the coating becomes nearly 100 percent effective. The number of reflective layers used are dependent on the desired thermal effect on the surface to be protected. The reflective layers are often made of Aluminum or other metals. Furthermore, between each reflective layer, between the outer layer and the reflective layers and between the reflective layers and the inner layer there is a separator layer [5]. The separator layers are often made of polyetylenereftalat (PET) or polyimide film [21]. The inner cover often consists of a reinforcement layer which must face the MLI blanket. In order to reduce the chance of an electrical short the inner layer is not metallized.

#### 6.1.1 Modeling Multi Layer Insulation

There are several ways the thermal engineer can model the thermal behavior through MLI. In this report a detailed model modeling ten layers of reflective layers has been compared with a simpler model suggested by the European Space Agency (ESA) in order to investigate how accurate the simpler model is. Here an 100x100x100 mm³ Aluminum box has been modeled with a spherical MLI blanket with radius of 100 mm wrapped around it.
6.1.1.1 MLI Stack with ten Reflective Layers

For the detailed model a stack mesh collector of ten non uniform layers were defined in Siemens NX™ where a one micrometer thick outer layer of Kapton has been defined with an emissivity of 0.52 and absorptivity of 0.41. The density of kapton is 1000 kg/m³ and the thermal properties reads thermal conductivity of 0.12 W/mK and specific heat of 1090 J/kgK. The ten reflective layers have been modeled as a 1000 Å thick Aluminum6061 layers with a heat transfer coefficient of 0.25 W/m²K and an effective emissivity of 0.05. The inner layer has also been modeled as Kapton but here with thickness of 0.025 mm and an emissivity of 0.04. The heat transfer coefficient for thermal coupling to the layer above is defined here as 0.25 W/m²K and the effective emissivity to 0.05.

Thermal coupling of 0.25 W/m²K comes from the more simplified model where a linear conduction of 0.025 W/m²K was suggested throughout the MLI and according to the following equation this gives a thermal coupling between each layer of:

$$0.025 = \lambda_{tot} = \frac{1}{\lambda_1} + \frac{1}{\lambda_{i+1}} + ... = \{\lambda_i = \lambda_{i+1} = ... = \lambda\} = \frac{1}{\frac{10}{X}} \Rightarrow \lambda = 0.25 \quad (6.1)$$

The Aluminum box and MLI was put in an orbit around Ganymede, one of Jupiter’s moons and in orbit around Mercury in order to investigate the MLI model in both a cold and hot environment. The planet, solar and orbital parameters for the Ganymede moon can be found in Table C.2, C.3 and C.4 in Appendix C.
6.1.1.2 Two layer MLI Model

![Diagram of two layer MLI model](image1)

**Figure 6.2:** Schematics of the model and the conduction and radiation between the outer and inner layers of the two layer MLI and the inner layer and the aluminum box.

The simplified model suggested by ESA is to model the MLI as a two layer blanket with a thermal conductivity of 0.025 W/m²K and an effective emittance of 0.03 between the layers. However, some calculations on what the effective emittance should be has also been performed here.

![Diagram of three layer MLI model](image2)

**Figure 6.3:** Schematics of the radiation between three of the MLI layers.

Consider **Figure 6.3** which shows the schematics of the radiation between three layers in the MLI blanket. The schematics gives that:

\[
q_{12} = q_1 = -q_2 = -q_1 = \frac{\sigma (T_1^4 - T_2^4)}{1 - \varepsilon_1 A_1 + \frac{1}{\varepsilon_3 A_3} + \frac{1}{\varepsilon_3 A_2} + \frac{1}{T_3 A_2} + \frac{1}{\varepsilon_2 A_2}} = \varepsilon^* \sigma (T_1^4 - T_2^4) \quad (6.2)
\]
where $\epsilon^*$ is the effective emittance. Simplifying the equation with $A = A_1 = A_2 = A_3$, $\epsilon = \epsilon_1 = \epsilon_2 = \epsilon_{3,1} = \epsilon_{3,2}$, $F_{13} = F_{32} = 1$ and extending the schematics and equation to be valid for $n$ number of layers results in:

\[
\epsilon^* = \frac{1}{n - 1 + (2n - 2)\frac{1 - \epsilon}{\epsilon}} = \frac{\epsilon}{2n - 2 - n\epsilon + \epsilon} = \frac{\epsilon}{(n - 1)(2 - \epsilon)} \quad (6.3)
\]

Please see Appendix D for the derivation of Equation 6.3. Using Equation 6.3 with $\epsilon = 0.05$ and $n = 10$ as in the more detailed model gives $\epsilon^* = 0.003$ which differs a factor 10 from the suggested value from ESA. The result in Equation 6.3 is also in accordance with calculations performed by Che-Shing Kang from the National Space Program Office in Taiwan [22]. The author therefore proceeded with the value of 0.003 for the effective emittance in the modeling of the MLI. A spherical MLI was modeled as a primitive with a stack mesh collector of two non uniform layers. The outer layer was defined as Kapton with emissivity of 0.52 and absorptivity of 0.41 on the outside of the outer MLI layer. The inner side of the outer MLI layer was given an emissivity of 0.003. The inner layer was also defined as Kapton but with an inner (towards the outer MLI layer) emissivity of 0.003 and an outer (towards the spacecraft) emissivity of 0.04 which is a typical value [5]. Furthermore, thermal coupling of 0.025 W/m$^2$K and a radiative thermal coupling of 0.003 was defined between the two layers.

### 6.1.1.3 Comparing two Layer MLI Model with ten Layer Model

It can be observed in Figures 6.4 and 6.5 that the temperature between the simplified two layer model differs some from the ten layer model. However in the case of orbiting Ganymede which is a cold case scenario the temperature difference is only about 0.3°C. In the hot case scenario which is in orbit around Mercury the temperature difference is about 1.3°C. In the more detailed model ten reflective layers were used. Figure 6.6 shows the effective emittance as a function of number of aluminized layers. In practice, after a certain number of layers the performance of the MLI will not improve by increasing the number of layers further. Radiative heat transfer becomes small compared to conductive shorts between layers and other losses as the number of layers increases [23]. According to ESA the value of the effective emittance should be $\epsilon^* = 0.03$ but the analytical calculations gave a value on the effective emittance of $\epsilon^* = 0.003$. In
laboratory tests values of $\epsilon^* = 0.005$ or lower have been achieved. However it has been shown that for spacecraft application the effective emittance of a MLI blanket is more likely to be within the range of 0.015 to 0.03 with current design, installation and manufacturing methods. This is due to heat leaks that results from the influences of edge effects, seams, cable penetrations etc [23]. Therefor a comparison of the Aluminum box temperature in orbit around Ganymede has been performed for the two layer model
Figure 6.6: Effective emittance vs. number of single aluminized layers. MATLAB code for generating the plot can be found in Appendix A.

Figure 6.7: Aluminum box temperature in orbit around Ganymede has been performed for the two layer model with an effective emittance of $\epsilon^* = 0.03$ vs. $\epsilon^* = 0.003$ with an effective emittance of $\epsilon^* = 0.03$ vs. $\epsilon^* = 0.003$ which can be studied in Figure 6.7. The temperature difference between the two cases after 20 orbits are about $2^\circ C$. 
6.1.1.4 Other Ways to Model MLI

A third way of modeling MLI has been tested and that is by using a one layer thin shell mesh as MLI layer. This has been compared with a two layer model which has been modeled in several different ways when it comes to thermal conductance and radiation couplings. The following scenarios have been tested with the following results:

- One layer thin shell of Kapton with thickness 0.1 mm with a thermal conductance coupling between the object within the MLI and the MLI of 0.025 W/K and a thermal radiation coupling of 0.0076. A heat load of 0.3 W was also put on the object. This resulted in an average temperature on the object of $-132^\circ C$ after 20 orbits around Ganymede, however equilibrium was not yet reached at this point.

- Two layer MLI with thermal conductance coupling of 0.05 W/K and a radiation coupling of 0.03 through the MLI. Thermal conductance coupling between the object and the inner layer of the MLI was 0.05 W/K and the radiation coupling 0.03 which corresponds to values in the one layer configuration above. A heat load of 0.3 W was also put on the object. This resulted in an average temperature on the object of $-105^\circ C$ after 20 orbits around Ganymede.
The same two configurations as above were also tested for an orbit around Mercury in order to also study a hot case. The temperature of the object in this case was 93°C respectively 110°C. One could also observe that the heat had distributed more throughout the MLI in the two layer case indicating that the in-plane conductance is higher for that case. Siemens NX™ calculate in-plane conduction for the middle layer only in a multi layer stack. In-plane conduction is hence most accurately modeled with a few layers (one layer being the most accurate). When calculating in-plane conduction for a multi layer shell, the solver assigns the whole thickness of the shell to the middle layer. However this procedure is for the cases when an odd number of layers are added. The author could not find any information on how the in-plane conductance is calculated for an even number of layers. However, this tells that the software is "cheating" when it comes to the in-plane conductance in the way that it simplifies the calculations. This could be one reason for why there is a difference in the heat distribution around the MLI in the two different cases, one versus two layer MLI modeling.

In reality the MLI is facing the object, there are holes in it for venting reasons and it is stitched and taped. All of this will contribute to heat losses and a contact to the object the MLI is wrapped around. It has also been tested to have the inner layer of the two layer MLI in perfect contact with the object which resulted in an average temperature of the object around Mercury of 110°C. This gave the same result as the two layer MLI with a thermal conductance coupling of 0.025 W/K which shows that the important parameter in the modeling is not the conductance between inner MLI layer and the object but the parameters within the two layers of the MLI. However when defining perfect contact and running the same case but for orbits around Ganymede the temperature of the object was −112°C, differing 7°C from the case with the thermal coupling of 0.05 W/K between the object and the inner layer of the MLI. When using Siemens NX™ and its perfect contact one should be careful to use different geometries since this will case inaccuracies. In order to eliminate possible inaccuracies due to geometrical reasons the Aluminum box enclosed within the MLI was changed to an Aluminum sphere, the same geometry as the MLI. The following two cases were tested:

- A two layer MLI with perfect contact defined between the spherical object and the inner layer of the MLI. The inner layer of the MLI was connected thermally
with the outer layer with a thermal conductance of $0.025 \, W/K$ and emissivity of $0.0076$.

- A two layer MLI with a thermal conductance between the object and the inner MLI layer of $0.05 \, W/K$ and emissivity of $0.03$ was defined together with a conductance of $0.05 \, W/K$ and emissivity of $0.03$ between the inner and outer layers of the MLI.

The two cases above is theoretically and mathematically the same, however a temperature difference on the object of $19^\circ C$ differs the result of the the cases apart once equilibrium had been reached. Possible reasons for this can have to do with the Siemens NX™ way of calculating the two different cases. Regarding the perfect coupling in NX the radiative coupling is not from the primary to the closest secondary elements but from the primary to the average temperature of the secondary set and from each secondary to the average temperature of the primary set. Furthermore, an idea was that the in-plane conductance had an impact on the result, i.e. that one side of the MLI is illuminated and therefore warmer while the other side is colder and that there therefore would be more radiation exchange between the MLI and the object. Therefore it was also tested to define the MLI as $10 \, mm$ thick Aluminum (still with the same thermo-optical properties as Kapton). This in order to have a higher in-plane conduction and thereby have a uniform temperature distribution around the MLI. This did not have any direct effect on the difference between the two models where the temperature difference still was about $19^\circ C$. In general it was observed that in the case of perfect contact between the object and the inner MLI layer the transient temperature and the starting steady state temperature was lower then for the other case. However after several orbits and once equilibrium had been reached the temperature in the non perfect contact case was lower. It takes much longer time for the non perfect contact case to be cooled downed (there is heat load of $0.3 \, W$ on the object which is warm to start with before the heat is dissipated to and through the MLI) then for the perfect contact case. This has to do with the fact that when defining perfect contact the inner MLI layer takes on the temperature that the object has. In the other case, the heat from the object need to go through the $0.05 \, W/K$ conductance and $0.03$ radiation emissivity before the heat reach the inner MLI layer. The conclusion from this MLI study is that there is a numerous of variables that plays a role when it comes to thermally model MLI and that there is
hard to say which model that is correct without doing a measurement on MLI which is strongly recommended as a next step for future work.

6.2 Thermal Control of the Spacecraft

From the thermal analysis of MIST it was noted that not all thermal requirements were met. The thermal requirements together with the temperature that the different subsystems was subjected to before any thermal control of the spacecraft was performed can be studied in Table 6.1. The next step in the thermal model is now to thermally control the spacecraft, i.e. keeping the different subsystems within the required temperature range specified. In order to have as low power consumption as possible when it comes to the thermal control system, as much as possible will need to be managed with passive control.

6.2.1 Morebac

One way of passively controlling the spacecraft and its payloads thermally is by using MLI. MLI was defined around the Morebac experiment according to the two layer model with perfect contact to the object explained in Section 6.1.1.4. This raised the temperature of the experiment to an equilibrium temperature interval from $+8 \rightarrow +11^\circ C$ to $+11 \rightarrow +15^\circ C$. However, the thermal requirement for the experiment state that it need to be within $20^\circ C$ to $30^\circ C$. Since the temperature of the Morebac experiment still had to be raised, a heater was modeled and added to the experiment. Heaters is an active control in which you heat a specific object which is too cold. Minco [6] is a company providing thermal solutions to several applications and they have developed Flexible Thermfoil™ heaters which is low mass, low volume, flexible heaters that can be glued on to the specific object in question. For the Morebac experiment this has been modeled as a thermal heat load with an active heater controller where a heat load of $1W$ has been put in for a cut-in temperature of $20^\circ C$ and a cut-off temperature of $25^\circ C$. The result of this was that the temperature of the experiment now is varying between approximately $20^\circ C$ and $25^\circ C$ which is within the thermal requirement for Morebac.

However, since the spacecraft has a strict power budget as much passive thermal control as possible is desired. Since the SEU experiment is constantly operating and consuming
1.8 W it will have a temperature between 27°C and 45°C. This heat can be used to heat the Morebac experiment instead of applying additional heaters to the experiment. The heater on Morebac was removed and both Morebac and SEU were placed together within the same MLI. Radiation heat exchange will then be present between Morebac and SEU and according to the thermal model this will give Morebac a temperature between 24°C and 29°C which is within the thermal requirement of the experiment. This means that no active heaters are needed for the Morebac experiment and hence saving 1 W of the power budget. However, since SEU now is confined within the MLI and can not exchange radial heat with any other surrounding, SEU will heat up and have a higher temperature than without the MLI. The temperature of SEU due to the MLI was raised to be between 42°C and 53°C which still is well within the operational thermal requirement of the experiment.

### 6.2.2 Nanospace

Table 6.1 also indicates that the Nanospace propulsion payload needs to be thermally controlled. As a thermal design MLI was also put around the Nanospace payload which increased the temperature of the tank to +8 → +12°C and the nozzles to 0 → +25°C. However, this will still not fulfill the thermal requirements and therefore heaters are most likely needed. Since it is a bit vague what the thermal requirements of each part of the Nanospace payload are heaters will be needed to be implemented by the experimenters on the payload. Since the author does not presently know where the experimenter would place these heaters the thermal control of the Nanospace payload is left for the moment. However, further work and communication with the experimenter is needed in order to model the heaters and their effect more accurately.


6.2.3 Ratex-J

According to this thermal analysis Ratex-J will have 12°C higher temperature in comparison with the thermal requirement provided by the experimenter. However the thermal analysis model needs to be updated before the thermal engineer can start to thermally control this payload. The experimenter has indicated that the payload will be operational during passage over the auroral zone and South Atlantic Anomaly (SAA). In order for the thermal engineer to model the heat load correctly these times has to be provided by the orbital and attitude control engineer. However, in this analysis a heat load with very rough times has been modeled. Also more accurate information regarding the mechanical interface between the payload and the satellite will be required. If the temperature still is too high when the analysis has been updated possible solutions is to either increase the conductance between the payload and the satellite main structure. It was tested to increase the conductance from 0.2 W/K to 0.8 W/K which corresponds to $h = 2000 \, W/m^2K$. This gave practically no change in temperature of the payload since the temperature of the main structure (rails) have a temperature of up to 28°C meaning that the temperature difference is quite small. However, another solution is to either paint the payload and hence change its optical properties or to add a cooling plate/radiator.

6.2.4 Camera

The temperature of the camera is only one degree away from fulfilling its thermal requirements and are therefore left for now since once more reliable and accurate numbers are provided by the experimenters the camera’s temperature might be within the required temperature interval.
Chapter 7

Conclusion

7.1 Conclusion

During this master thesis project a detailed transient thermal model has been developed with the purpose to analyze the thermal behavior of the MIST satellite and to control the temperature so that each payload and subsystem are within their temperature requirement. It was noted from the thermal model and analysis that not all payloads was within their desired temperature interval, this was Nanospace, Ratex-J, Morebac, the camera, Cubes and LEGS. The rest of the payloads were already within their desired temperature interval and do not need any thermal control. A possible thermal design for keeping the Morebac payload within the temperature requirement is to stack Morebac and SEU next to each other and cover them in MLI so that they are in the same enclosure. This way the excess heat from SEU will heat the Morebac payload. Furthermore, Nanospace will need MLI around the payload as well, however, also heaters are needed here but it is still unknown how much power these heaters need.

7.2 Future Work

One of the limiting factors in this analysis has been the information from the different payloads. Since many of the payloads still are under design they have not yet decided on the mechanical interface, what materials the payload will consist of and when in orbit the payloads will be operational. This makes the thermal engineer’s work harder
and sophisticated estimates and assumptions have been needed to be made concerning these factors. Further work will then include to update the thermal model in terms of material selection, mechanical interfaces and heat loads as new and more accurate information is received from the experimenters. This includes to add heat load for when the thrusters on the Nanospace payload is fired (now only idle power consumption is taken into account). Also add heat load on the Morebac payload once information from them is received regarding the LED light they will use and their power consumption. Also the solar panels need to be updated in terms of removing one $10 \times 10 \text{cm}^2$ panel where the Ratex-J detectors will look out. Further work also include to continue to thermally control each payload of the spacecraft i.e. making sure that each payload is within the temperature interval of its thermal requirement. This can be done by encapsulating the payloads in MLI, increase or decrease their thermal conductance to the object they have a mechanical interface with. As to the MLI a measurement of the conductance through MLI should be performed in order to be able to better and more securely model MLI thermally in Siemens NX™. Other ways of thermally controlling the payloads is to change their thermo-optical properties i.e. emissivity and absorptivity or to add active control in the form of heaters, radiators, louvers etc. Finally, once an accurate analysis and thermal control system has been constructed and simulated the next step will be to simulate a thermal test case and perform thermal testing on the spacecraft hardware.
Appendix A

Matlab Codes

Preliminary Thermal Analysis - Sphere

% Preliminary thermal analysis for the spacecraft MIST
% KTH Royal Institute of Technology, Stockholm, Sweden
% Andreas Berggren
% 2015-01-27

% These preliminary calculations is for a
% spacecraft modeled as a spherical satellite
%
clear all
close all
clc

%%%%% Constants %%%%%

H = 640E3; % [m] Orbit altitude
R = 6371E3; % [m] Earths mean radius
G_s = 1377; % [w/m^2] same as NX uses
sigma = 5.670373E-8; % Stefan-Boltzmann constant
q_IR = [237 237]; % [Hot value, cold value]
Q_w = 0; % [w] Electric energy dissipation
eps = 0.7; % [Hot value, cold value]
alpha = 0.6; % [w] Electric energy dissipation
a = 0.306; % Albedo effect, Hot value,
Appendix A. Matlab codes

A = 0.1*0.3*4 + 0.1*0.1*2; % [m^2]
A_absorbed = pi.*0.1056.^2; % [m^2]

%%%%% Calculations %%%%%

gamma = asin(R/(R+H));
F = (1/4 - (((2*(H/R) + (H/R)^2)^0.5)/(4*(1+(H/R)))) ... 
+ (cos(gamma)/8)*(1/(1+(H/R)))^2);
% Source: http://www.thermalradiation.net/calc/sectionc/C-140.html
G_IR = q_IR./(R+H)^2; % Intensity drops as 1/r^2
G_r = a.*G_s.*F;

Q_solar = alpha.*G_s.*A_absorbed; % Heat from the Sun
Q_IR = eps.*G_IR(1).*A_absorbed;
% Heat radiated from Earth seen by the spacecraft
Q_albedo = alpha.*G_r.*A;
% Heat from the Sun reflected on the Earth (Albedo)
Q_absorbed = Q_solar + Q_IR + Q_albedo;
% Total heat from the space environment on the spacecraft

disp('Maximum temperature in Degrees Celcius')
T_max = ((Q_absorbed + Q_w)./(sigma.*eps.*A))^((1/4)) - 273.15 % OBS: Celcius

disp('Minimum temperature in Degrees Celcius')
% In eclipse, only radiation from the Earth and internal heating will contribute
T_min = ((eps.*G_IR(2).*A_absorbed + Q_w)./(sigma.*eps.*A*(3/4)))^((1/4)) ... 
- 273.15
% OBS: Celcius %ONLY (3/4) of the sphere see 'deep space' and radiates to it!

%%%%

%%%%%%% alpha - epsilon - temperature 3D plot %%%%%

clear all
close all
clc

%%%%%%% Constants %%%%%

H = 640E3; % [m] Orbit altitude
Appendix A. Matlab codes

R = 6371E3; % [m] Earths mean radius
G_s = 1377; % [W/m^2] same as NX uses
sigma = 5.670373E-8; % Stefan-Boltzmann constant
q_IR = [237 237]; %[258 216]; % [Hot value, cold value]
Q_w = 0.1; % [W] Electric energy dissipation
eps = linspace(0.1,1);
alpha = linspace(0,1);
a = 0.306; % Albedo effect, Hot value
A = 0.1*0.3*4 + 0.1*0.1*2; % [m^2]
A_absorbed = pi.*0.1056.^2; % [m^2]

%%%%% Calculations %%%%%
gamma = asin(R/(R+H));
F = (1/4 - (((2*(H/R) + (H/R)^2)^(0.5))/(4*(1+(H/R)))) + ...
    (cos(gamma)/8)*(1/(1+(H/R)))^2);
% Source: http://www.thermalradiation.net/calc/sectionc/C-140.html
G_IR = q_IR./((R+H)/R)^2; % Intensity drops as 1/r^2
G_r = a.*G_s.*F;

for i = 1:length(alpha)
    Q_solar = alpha(i).*G_s.*A_absorbed;
    % Heat from the Sun seen by the spacecraft
    Q_IR = eps.*G_IR(1).*A_absorbed;
    % Heat radiated from Earth seen by the spacecraft
    Q_albedo = alpha(i).*G_r.*A;
    % Heat from the Sun reflected on the Earth (Albedo)
    Q_absorbed = Q_solar + Q_IR + Q_albedo;
    % Total heat from the space environment on the spacecraft

    T_max(i,:) = ((Q_absorbed + Q_w)./(sigma.*eps.*A)).^(1/4) - 273.15;
    T_min(i,:) = ((eps.*G_IR(2).*A_absorbed + Q_w)./...
                 (sigma.*eps.*A*(3/4))).^(1/4) - 273.15; % OBS: Celcius

end

for i = 1:length(alpha)
    for j = 1:length(eps)
        if T_max(i,j) > 0 & T_max(i,j) < 40
            T_maxreq(i,j) = T_max(i,j);
        end
    end

end
Appendix A. Matlab codes

```matlab
else
    T_maxreq(i,j) = NaN;
end
end

figure(1)
surf(eps, alpha, T_max)
title('Hot case')
xlabel('Emissivity, \epsilon')
ylabel('Absorptivity, \alpha')
zlabel('Maximum temperature, T_{max} [^\circ C]')

figure(2)
surf(eps, alpha, T_min)
title('Cold case')
xlabel('Emissivity, \epsilon')
ylabel('Absorptivity, \alpha')
zlabel('Minimum temperature, T_{min} [^\circ C]')

figure(3)
surf(eps, alpha, T_maxreq)
title('Hot case with temperature interval 0 - 40^\circ C')
xlabel('Emissivity, \epsilon')
ylabel('Absorptivity, \alpha')
zlabel('Maximum temperature, T_{max} [^\circ C]')

Decrease in Temperature During Eclipse

% Spacecraft temperature decrease during eclipse
% KTH Royal Institute of Technology, Stockholm, Sweden
% Andreas Berggren
% 2015-02-09
%
% %%% Analytical %%%
clear all
close all
clc
H = 640E3; % [m] Orbit altitude
R = 6371E3; % [m] Earth's mean radius
```
\[ G_s = 1377; \quad \text{[W/m}^2\text{]} \text{ same as NX uses} \]
\[ \sigma = 5.670373 \times 10^{-8}; \quad \text{Stefan-Boltzmann constant} \]
\[ q_{IR} = 237; \quad \text{[Hot value, cold value]} \]
\[ \epsilon = 0.1:0.1:1; \quad \text{[m}^2\text{]} \text{ Area of an equivalent} \]
\[ \text{Area} = 0.1 \times 0.3 \times 4 + 0.1 \times 0.1 \times 2; \quad \text{[m}^2\text{]} \text{ sphere with same surface area} \]
\[ A_{absorbed} = \pi \times 0.1056^2; \quad \text{[m}^2\text{]} \]
\[ T_a = 2.7; \quad \text{Ambient temperature of 2.7 Kelvin} \]
\[ m = 4; \quad \text{Mass of spacecraft, 4 kg} \]
\[ C = 897; \]

\textbf{for} i = 1:length(eps)
\[ Q_{IR} = \epsilon(i) \times (q_{IR} / ((R+H)/R)^2) \times A_{absorbed}; \]
\[ A = (\text{Area} \times \epsilon(i) \times \sigma) / (m \times C); \]
\[ B = (Q_{IR} + \text{Area} \times \epsilon(i) \times \sigma \times T_a^4) / (m \times C); \]
\[ T_{start} = 273.15 + 25; \]
\[ T_{end} = 273.15 - 30; \]
\[ T_{anal} = \text{linspace}(T_{start}, T_{end}); \]
\[ \text{I}_{upper} = (-\log(B^{1/4} - A^{1/4} \times T_{anal}) + \log(A^{1/4} \times T_{anal}) \ldots + B^{1/4}) + 2 \times \text{atan}(A^{1/4} \times T_{anal}) \ldots \]  \[ \div (4 \times A^{1/4} \times B^{3/4})]; \]
\[ \text{I}_{lower} = (-\log(B^{1/4} - A^{1/4} \times T_{start}) \ldots + \log(A^{1/4} \times T_{start}) \ldots \]  \[ + B^{1/4}) + 2 \times \text{atan}(A^{1/4} \times T_{start}) \ldots \]  \[ \div (4 \times A^{1/4} \times B^{3/4})]; \]
\[ t_{anal}(i,:) = \text{I}_{upper} - \text{I}_{lower}; \]
\textbf{end}

\[ T_{anal} = T_{anal} - 273.15; \]
\[ \text{figure}(2) \]
\[ \text{plot}(t_{anal} / 60, T_{anal}) \]
\[ \text{title}('\text{Temperature decrease of spacecraft during eclipse as a function of time}') \]
\[ \text{xlabel}('\text{Time [Minutes]}') \]
\[ \text{ylabel}('\text{Temperature [}^\circ\text{C]}') \]
\[ \text{legend}('\epsilon = 0.1', '\epsilon = 0.2', '\epsilon = 0.3', '...', '\epsilon = 0.4', '\epsilon = 0.5', '\epsilon = 0.6', '...', '\epsilon = 0.7', '\epsilon = 0.8', '\epsilon = 0.9', '\epsilon = 1') \]
%%
%%%%% Solve the integral numerically %%%%%

clear all
close all
clc
H = 640E3; % [m] Orbit altitude
R = 6371E3; % [m] Earths mean radius
G_s = 1377; % [W/m^2] same as NX uses.
sigma = 5.670373E-8; % Stefan-Boltzmann constant
q_IR = 237;
eps = 0.7;
Area = 0.1*0.3*4 + 0.1*0.1*2; % [m^2]
A_absorbed = pi.*0.1056.^2; % [m^2]
T_a = 2.7; % Ambient temperature of 2.7 Kelvin
m = 4; % Mass of spacecraft, 4 kg
C = 897;

Q_IR = eps.*(q_IR./((R+H)/R).^2).*A_absorbed;

A = (Area.*eps.*sigma)/(m.*C);
B = (Q_IR + Area.*eps.*sigma.*T_a.^4)/(m.*C);
T_start = 273.15 + 25;
T_end = 273.15 - 30;
T = linspace(T_start, T_end); %T_end:0.001:T_start;
%T = T(end:-1:1);
f = 1./(B - A.*T.^4);

for i = 1:length(T)
    T_int = linspace(T_start, T(i));
    f = 1./(B - A.*T_int.^4);
    t(i) = trapz(T_int,f);
end

T = T - 273.15;

figure(3)
plot(t./60,T)
title('Temperature decrease of spacecraft during eclipse as a function of time')
xlabel('Time [Minutes]')
ylabel('Temperature ['{\circ}C']')

Three-Surface Radiation Enclosure - Analytical Analysis

% Radiation Heat Transfer in Three-Surface Enclosures
% One PCB radiating to another PCB and a wall
% KTH Royal Institute of Technology, Stockholm, Sweden
% Andreas Berggren
% 2015-03-12

clear all
close all
clc

% Subscript clearifications 1 = PCB1, 2 = PCB2, 3 = Wall all according to
% the "Three-Surface Radiation Enclosure" section in the paper
% "Design of Thermal Control System for the Spacecraft MIST"

SB = 5.67e-8; % Stefan-Boltzmann constant
e1 = 0.65; % Emissivity
e2 = 0.65;
e3 = 0.90;
e = [e1 e2 e3];

T1 = 273.15 + 25; % Temperature
T2 = 273.15 + 25;
T3 = 273.15 + 35;
T = [T1 T2 T3];

x = 0.09; % PCB width
y = 0.096; % PCB length
L = 0.005; % Distance between the PCBs

X = x/L;
Y = y/L;

A1 = 0.09*0.096; % PCB area
A2 = 0.09*0.096; % PCB area
A3 = 4*0.1*0.1*L; % Surrounding space area
A = [A1 A2 A3];

% View factors

F12 = (2/(pi*X*Y))*(log(((1 + X^2)*(1 + Y^2))/(1 + X^2 + Y^2))^(1/2) ... 
    + X*(1+Y^2)^(1/2)*atan(X/(1+Y^2)^(1/2)) ...) 
    + Y*(1+X^2)^(1/2)*atan(Y/(1+X^2)^(1/2)) - X*atan(X) - Y*atan(Y));
F13 = 1 - F12;
F21 = (A1/A2)*F12;
F23 = 1 - F21;

R = diag((1-e)./(A.*e));
R(1,2) = 1/(A1*F12);
R(1,3) = 1/(A1*F13);
R(2,3) = 1/(A2*F23);

Eb = SB.*T.ˆ4;

Matrix A = [-1/R(1,1) + 1/R(1,2) + 1/R(1,3) 1/R(1,2) 1/R(1,3); 
            1/R(1,2) -1/R(1,2) + 1/R(2,2) + 1/R(2,3) 1/R(2,3); 
            1/R(1,3) 1/R(2,3) -(1/R(1,3) + 1/R(2,3) + 1/R(3,3))];
B = -Eb./diag(R)';
J = Matrix A\B'
Q = (Eb' - J)./diag(R)

% The net rate of heat transfer from the respectively surfaces

% Sweeping emission coefficient

clear all
close all
clc

% 1 = PCB1, 2 = PCB2, 3 = Wall

SB = 5.67e-8; % Stefan-Boltzmann constant

T1 = 273.15 + 25; % Temperature
T2 = 273.15 + 25;
T3 = 273.15 + 35;
T = [T1 T2 T3];

x = 0.09; % PCB width
y = 0.096; % PCB length
L = 0.05; % Distance between the PCBs
X = x/L;
Y = y/L;

A1 = 0.09*0.096; % PCB area
A2 = 0.09*0.096; % PCB area
A3 = 4*0.1*0.1*L; % Surrounding space area
A = [A1 A2 A3];

F12 = (2/(pi*X*Y))*(log(((1 + X^2)*(1 + Y^2))/(1 + X^2 + Y^2))^(1/2)...
    + X*(1+Y^2)^((1/2)*atan(X/(1+Y^2))^(1/2))...
    + Y*(1+X^2)^((1/2)*atan(Y/(1+X^2))^(1/2)) - X*atan(X) - Y*atan(Y));
F13 = 1 - F12;
F21 = (A1/A2)*F12;
F23 = 1 - F21;

e1 = linspace(0,1,10); % Emissivity
e2 = linspace(0,1,10);
e3 = linspace(0,1,10);

Q_final = [];

for i = 1:length(e1)
    for k = 1:length(e3)
        e = [e1(i) e1(i) e3(k)];

        R = diag((1-e)./(A.*e));
        R(1,2) = 1/(A1*F12);
        R(1,3) = 1/(A1*F13);
        R(2,3) = 1/(A2*F23);

        Eb = SB.*T.^4;

        Matrix_A = [-(1/R(1,1) + 1/R(1,2) + 1/R(1,3)) 1/R(1,2) 1/R(1,3);
                     1/R(1,2) -(1/R(1,2) + 1/R(2,2) + 1/R(2,3)) 1/R(2,3);]

    end
end
Appendix A. Matlab codes

\[
\begin{align*}
1/R(1,3) & \quad 1/R(2,3) \quad -(1/R(1,3) + 1/R(2,3) + 1/R(3,3))
\end{align*}
\]

\[
B = -Eb./\text{diag}(R)';
\]

\[
J = \text{Matrix}_A\backslash B';
\]

% The net rate of heat transfer from the respectively surfaces
\[
Q = (Eb' - J)./\text{diag}(R);
\]

\[
Q = [Q; (Eb(1) - Eb(2))/(R(1,1) + R(1,2) + R(2,2)); (Eb(1) -
- Eb(3))/(R(1,1) + R(1,3) + R(3,3))]; \quad \% [Q1 \ Q2 \ Q3 \ Q12 \ Q13]
\]

Q_final = [Q_final Q(1:5)];

end
end

for i = 0:length(e1)-1

Q3(i+1,:) = [Q_final(3,i*length(e1)+1:i*length(e1)+length(e1))];
Q1(i+1,:) = [Q_final(1,i*length(e3)+1:i*length(e3)+length(e3))];
Q12(i+1,:) = [Q_final(4,i*length(e3)+1:i*length(e3)+length(e3))];
Q13(i+1,:) = [Q_final(5,i*length(e3)+1:i*length(e3)+length(e3))];
end

figure(1)

subplot(2,1,1)
surf(((1:length(e1))./length(e1)), ((1:length(e1))./length(e1)), Q1)
xlabel('Emissivity of PCB')
ylabel('Emissivity of Wall')
zlabel('Q1')
title('The net rate of radiation heat transfer to/from one PCB, Q1')

subplot(2,1,2)
surf(((1:length(e1))./length(e1)), ((1:length(e1))./length(e1)), Q3)
xlabel('Emissivity of PCB')
ylabel('Emissivity of Wall')
zlabel('Q3')
title('The net rate of radiation heat transfer to/from the wall, Q3')

figure(2)

subplot(2,1,1)
surf(((1:length(e1))./length(e1)), ((1:length(e1))./length(e1)), Q12)
xlabel('Emissivity of PCB')
ylabel('Emissivity of Wall')
zlabel('Q12')
title('The net rate of radiation heat transfer to/from one PCB to another PCB, Q12')
Appendix A. *Matlab codes*

```matlab
subplot(2,1,2)
surf(((1:length(e1))./length(e1)), ((1:length(e1))./length(e1)), Q13)
xlabel('Emissivity of PCB')
ylabel('Emissivity of Wall')
zlabel('Q13')
title('The net rate of radiation heat transfer to/from a PCB to the wall, Q13')

PCB Calculator

% PCB Thermal Conductivity Calculator
% KTH Royal Institute of Technology
% Andreas Berggren
% 2015-04-14

close all
clear all
clc

% INPUTS BY THE USER

t_pcb = 1.6*10^(-3); % Total thickness of the PCB [m]
nlayers = 8; % Number of layers, minimum is 5 layers!
cu_frac_out = [0.5 0.5]; % Fraction of copper in the outer layers
cu_frac_ground = [1 1]; % Fraction of copper in the power/ground layers
cu_frac_inner = [0.4 0.4 0.4 0.4]; % Fraction of copper in the inner layers

% CONSTANTS

k_cu = 387; % Thermal conductivity of copper
k_fr4 = 0.25; % Thermal conductivity of FR4
rho_cu = 8960; % Density of copper [kg/m^3]
rho_fr4 = 1850; % Density of FR4
cp_cu = 390; % Specific heat of copper [J/kg*K]
cp_fr4 = 600; % Specific heat of FR4
t_cu_out = 35*10^(-6); % Thickness of the outer copper layers
```
t_cu_inner = 17.5*10^(-6); % Thickness of the inner copper layers

% Calculations

cu_frac = [cu_frac_out cu_frac_ground cu_frac_inner];

if length(cu_frac) == nlayers

    cu_tot_frac = (sum(t_cu_out.*cu_frac(1:2)) + ... 
    sum(t_cu_inner.*cu_frac(3:4)) + ... 
    sum((t_cu_inner./2).*cu_frac(5:end)))./t_pcb;
    fr4_tot_frac = 1 - cu_tot_frac;
    disp('Total Cu Thickness [m]')
    cu_tot = cu_tot_frac*t_pcb
    disp('Thermal Conductivity in plane of the PCB')
    k_PCB = cu_tot_frac*k_cu + fr4_tot_frac*k_fr4
    disp('Density of the PCB')
    rho_PCB = cu_tot_frac*rho_cu + fr4_tot_frac*rho_fr4
    disp('Specific heat of the PCB')
    rho_PCB = cu_tot_frac*cp_cu + fr4_tot_frac*cp_fr4
else
    disp('The number of entries in "cu_frac" must match the number of layers chosen')
end

Measurement of PCB Mounting Conductance

% Measurement of PCB Mounting Conductance
% KTH Royal Institute of Technology
% Andreas Berggren
% 2015-03-09

clear all
close all
clc

% Theoretical calculations

dT = 50;
Rh = 26.6; % Resistance of power heater
% Matlab codes

k_A1 = 170;
r_sh = 0.00398;  % Radius of the screw head
r_sb = 0.00191;  % Radius of the screw body
R_dist = 0.00397;
r_dist = 0.00105;

A_dist = pi*(R_dist^2 - r_dist^2);
A_sh = pi*r_sh^2;
A_sc = 2*pi*r_sb*0.006/2;  % Screw connection
A_sb = pi*r_sb^2;

d = 0.01007;
% The distance between the PCB and the metal plate
d_sb = 0.016;

G_crossdist = A_dist*450;
% 300 W/Km^2 is a estimated value which changes with pressure
G_dist = (A_dist*k_A1)/d;

G_sh = A_sh*450;
G_sc = A_sc*450;
G_sb = (A_sb*k_A1)/d_sb;

G_tot_dist = 1/(2*(1/G_crossdist) + (1/G_dist));
G_tot_sb = 1/(2*(1/G_sh) + (1/G_sb) + (1/G_sc));

G_tot = G_tot_dist + G_tot_sb

Q = G_tot*dT;
I = sqrt(Q/Rh)  % Current needed to heat

%% Experimental Calculations

R_25 = 5000;  % [ohm]
B_25 = 3497;  % [K]
T_25 = 298.15;  % [K]
Rh = 26.6;  % Resistance of the heating element
I = [0.1683 0.1683 0.1681 0.1693 0.1694 0.1692 0.1691 0.1692 0.1695... 0.1697 0.1700 0.1701 0.1686 0.1687 0.170 0.170];  % Current
% R034_pcb = 5356; % PCB resistance at ambient temperature
% R045_pcb = 5303; % PCB resistance at ambient temperature
% R067_plate = 5261; % Plate resistance at ambient temperature
% R078_plate = 5217; % Plate resistance at ambient temperature

R034_pcb = 5587; % PCB resistance at ambient temperature
R045_pcb = 5527; % PCB resistance at ambient temperature
R067_plate = 5645; % Plate resistance at ambient temperature
R078_plate = 5359; % Plate resistance at ambient temperature

R0 = [R034_pcb R045_pcb R067_plate R078_plate];

% First measurement means that the screw has been twisted
% maximum

% First measurements 2015-03-11 11:35
R_pcb(:,:,1) = [738.3 779.6];
R_plate(:,:,1) = [4504.5 4525.5];
% Second measurements 2015-03-11 13:10
R_pcb(:,:,2) = [737.8 773];
R_plate(:,:,2) = [4468 4490];

% First measurements 2015-03-11 15:50, have loosen the screw with 45 degrees from
% measurements 2015-03-11 15:50.
R_pcb(:,:,3) = [782 839];
R_plate(:,:,3) = [4510 4489];
% Second measurements 2015-03-11 16:35, have loosen the screw with 45 degrees from
% measurements 2015-03-11 15:50.
R_pcb(:,:,4) = [768 825];
R_plate(:,:,4) = [4483 4467];

% First measurements 2015-03-12 13:28, having the screw as loose as
% possible but still tigthened
R_pcb(:,:,5) = [755.8 845.9];
R_plate(:,:,5) = [4551 4474];
% Second measurements 2015-03-12 14:24, having the screw as loose as
% possible but still tigthened
% First measurements 2015-03-12 17:40, have tightened the screw 10 degrees
R_pcb(:,:,6) = [755.4 843.1];
R_plate(:,:,6) = [4491 4416];

% First measurements 2015-03-12 19:08
R_pcb(:,:,7) = [869 955];
R_plate(:,:,7) = [4371 4314];

% First measurements 2015-03-13 11:50, have tightened the screw 10 degrees
% from previous measurement, total 20 degrees from measurement 2015-03-12 13:28
R_pcb(:,:,9) = [842 1006];
R_plate(:,:,9) = [4573 4540];

% Second measurements 2015-03-13 13:25
R_pcb(:,:,10) = [829 981];
R_plate(:,:,10) = [4505 4475];

% First measurements 2015-03-16 11:18, have tightened the screw 20 degrees
% from previous measurement, total 50 degrees from measurement 2015-03-12 13:28
R_pcb(:,:,13) = [812 933];
R_plate(:,:,13) = [4251 4245];

% Second measurements 2015-03-16 12:05
R_pcb(:,:,14) = [808 929];
R_plate(:,:,14) = [4253 4247];

% First measurements 2015-03-16 16:54, have tightened the screw 20 degrees
% from previous measurement, total 70 degrees from measurement 2015-03-12 13:28
R_pcb(:,:,15) = [846 962];
R_plate(:,:,15) = [4086 4062];

% Second measurements 2015-03-16 same time!!
R_pcb(:,:,16) = [846 962];
R_plate(:,:,16) = [4086 4062];

R = [R_pcb;
    R_plate;
];

T0 = (log(R0./R_25).*(1./B_25) + (1./T_25)).^(-1);

T = (log(R./R_25).*(1./B_25) + (1./T_25)).^(-1);

T(1,2,:) = T(1,2,:) - abs(T0(2) - T0(1));
T(2,1,:) = T(2,1,:) + abs(T0(3) - T0(1));
T(2,2,:) = T(2,2,:) + abs(T0(4) - T0(1));

dT = [abs(abs((T(1,1,:) + T(1,2,:))/2) - abs(T(2,1,:) + T(2,2,:))/2)];

disp('Temperature in degrees Celsius')
T = T - 273.15;

Q = Rh.*I.^2;

for i = 1:length(dT)
    dT_final(i) = dT(:,:,i);
end

disp('Conductance')
G_exp = Q./dT_final

for i = 1:(length(G_exp))/2
    % Works only if we always have two measurement points!!!
    G_avg(i) = (G_exp(2*i-1) + G_exp(2*i))./2;
    % Standard deviation
    S(i) = sqrt((G_exp(2*i-1) - G_avg(i)).^2 + (G_exp(2*i) - G_avg(i)).^2);
    % Mean standard deviation
    S_mean(i) = S(i)./sqrt(length(G_exp));
end

G_avg

disp('Average conductance')
mean(G_avg(3:end))
angle = [0 10 20 30 50 70].*(pi./180);
fitobject = fit(angle'./(pi./180),G_avg(3:end)', 'poly1');

figure(1)
plot(fitobject, angle/(pi/180), G_avg(3:end), '*')
grid on
xlabel('Angle the screw has been tightened in degrees')
ylabel('Conductance [W/K]')
legend('Measured data', 'Least mean square fit')
title('Contact Conductance between PCB and metal plate')

%% Error analysis

I = 0.168;
Err_I = 0.005;
Err_dT = 0.02;
Err_Rh = 0.04;
Err_read = 0.02;
Err_G = sqrt(sqrt(((2*Rh*I*Err_I)./dT).^2 + ((Rh*I^2*Err_dT)./dT).^2).*2 ... + ((I^2*Err_Rh)./dT).^2).*2 + (0.02).^2)

%% Behavior of the thermistors

R_25 = 5000; % [ohm]
B_25 = 3497; % [K]
T_25 = 298.15; % [K]
sensor1 = [14300 6500 2900 1530 820];
sensor2 = [14300 6200 3000 1530 820];
sensor3 = [14300 6300 2900 1510 820];
sensor4 = [14300 6300 2900 1530 820];
sensors = [sensor1; sensor2; sensor3; sensor4];

T_sensor = (log(sensors./R_25).*(1./B_25) + (1./T_25)).^(-1) - 273.15;
T_real = linspace(0,80)+273.15; % Degrees C
R_real = R_25.*exp(B_25.*((1./T_real) - (1./T_25)));
T = [0 20 40 60 80];

figure(2)
plot(T, sensors(1,:), T, sensors(2,:), T, sensors(3,:), T, sensors(4,:),...)
T_real = 273.15, R_real
legend('Sensor PCB1', 'Sensor PCB2', 'Sensor Metal Plate 1', ...
     'Sensor Metal Plate 2', 'Theoretical curve')
xlabel('Temperature [°C]')
ylabel('Resistance [Ω]')
title('Resistance vs temperature relationship for the thermsitors')
grid on

Thermal Coupling Conductances Calculations

% Calculations of thermal conductances
% KTH Royal Institute of Technology
% Andreas Berggren
% 2015-02-19

clear all
close all
clc

%%% Thermal coupling between the rails

k_Al = 170;
A_thrm_coupl_rail = 0.0085^2;
x_between_rails = 0.0976;
h_rails = 400;

G_rails_coupl = (k_Al.*A_thrm_coupl_rail)./x_between_rails;
G_rails_tot = 1/((1/G_rails_coupl) + 2/(h_rails*A_thrm_coupl_rail))
% Defined as per element in NX

%%% Thermal Coupling between front and back of the Solar Panels

k_sp_front = 0.8;
k_sp_back = 387; % Conductivity of copper,
                % the back side of the solar panels
A_sp_large = 0.3275*0.083; % Cross section area of the large % solar panels
Appendix A. Matlab codes

```
x_sp_back = 0.0005; % Thickness (or length heat is conducted)
x_sp_front = 0.0004;
h_sp = 200; % Thermal contact conductance

G_sp_large_back = (k_sp_back*A_sp_large)/x_sp_back;
G_sp_large_front = (k_sp_front*A_sp_large)/x_sp_front;
% Total conductans between the solar panels
G_sp_large_tot = 4/((1/G_sp_large_back) + (1/(h_sp*A_sp_large)) ... 
+ (1/G_sp_large_front))

%% Thermal coupling between railings and the solar panels

A_cross_spRails = 0.0085*0.0976; % Cross section area between
x_rails_ladder = 0.0085; % The thickness of the ladder rails
h_sp.back = 300;
G_sp.rails = 16/(1/(h_sp.back*A_cross_spRails) + 1/(h_rails*A_thrm_coupl_rail))

%% Average Solar Panel material - Deployable solar panels

A dsp = 0.1*0.3*2;
t_glass = 200*10^(-6);
t_cu = 140*10^(-6);
t_pcb = 1.6*10^(-3);
t_fr4 = t_pcb - t_cu - t_glass;

k_cu = 387;
k_fr4 = 0.25;
k_glass = 0.8;

rho_cu = 8960; % Density of copper [kg/m^3]
rho_fr4 = 1850; % Density of FR4
rho_glass = 2500;

cp_cu = 390; % Specific heat of copper [J/kg*K]
cp_fr4 = 600; % Specific heat of FR4
cp_glass = 800;

k_avg = k_cu*(t_cu/t_pcb) + k_fr4*(t_fr4/t_pcb) + k_glass*(t_glass/t_pcb)
rho_avg = rho_cu*(t_cu/t_pcb) + rho_fr4*(t_fr4/t_pcb)...
```


Appendix A. Matlab codes

\[ \text{cp}_{\text{avg}} = \text{cp}_{\text{cu}} \times \left( \frac{t_{\text{cu}}}{t_{\text{pcb}}} \right) + \text{cp}_{\text{fr4}} \times \left( \frac{t_{\text{fr4}}}{t_{\text{pcb}}} \right) + \text{cp}_{\text{glass}} \times \left( \frac{t_{\text{glass}}}{t_{\text{pcb}}} \right) \]

\[ \text{A}_{\text{hinge}} = 0.019 \times 0.0075; \]
\[ \text{h}_{\text{hinge}} = 500; \]
\[ \text{G}_{\text{deploy-sp}} = \text{A}_{\text{hinge}} \times \text{h}_{\text{hinge}} \]

%% Thickness of Nanospace tank

clear all
close all
clc
R = 0.031; \quad \% \text{Radius of the cylindrical tank}
h = 0.037; \quad \% \text{Height of the cylindrical tank}
rho = 4453.9; \quad \% \text{Density of the tank}
m = 0.30; \quad \% \text{Estimated Mass of tank (total for Nanospace is 310g)}
r = linspace(0, 0.035);
func = pi.*h.*(R.^2-r.^2) + 2.*pi.*r.^2.*(R-r);
plot(r,func,r,m/rho)
legend('func','m/rho')
grid on
r = 0.027; \quad \% \text{Read from graph where func intersect with m/rho}
t = R-r

%% Thickness of Nanospace metal plate

A_{mp} = 0.95^2; \quad \% \text{Area of metal plate}
m_{mp} = 0.05; \quad \% \text{Estimated mass of metal plate}
rho_{mp} = 2810; \quad \% \text{Density of the metal plate (if material is AL6061)}
t_{mp-collector} = \frac{m_{mp}}{(A_{mp} \times \rho_{mp})}/2 \quad \% \text{Devided by 2 because we have collector both at top and bottom of the plate}

%% Thermal coupling between Nanospace vertical PCB and the main satellite structure

\% lambda = h, thermal contact conductance!

A_{PCB-support} = 0.005 \times (2 \times 0.28 + 2 \times 0.8);
A_{support-sat} = 2 \times 0.005^2;
\lambda_{PCB-support} = 300;
G_{PCB\_support} = 2 \times A_{PCB\_support} \times \lambda_{PCB\_support};
G_{support\_sat} = 2 \times A_{support\_sat} \times \lambda_{PCB\_support};

% Total conductance for both sides
G_{tot\_PCB\_sat} = 1/(1/G_{PCB\_support} + 1/G_{support\_sat})

%% Thermal coupling between Nanospace nozzles and vertical PCB
A_{nozzle} = 0.018 \times 0.008;
\lambda_{nozzle\_PCB} = 500;
G_{tot\_nozzle\_PCB} = 4 \times A_{nozzle} \times \lambda_{nozzle\_PCB} \quad \% 4 nozzles

%% Thermal coupling between Nanospace metal plate and satellite
A_{mp\_sat} = 0.005^2;
\lambda_{mp\_sat} = 300;
G_{mp\_sat} = A_{mp\_sat} \times \lambda_{mp\_sat}; \quad \% 4 connecting points

A_{thrm\_coupl\_rail} = 0.0085^2;
\h_{rails} = 400;
G_{rail} = A_{thrm\_coupl\_rail} \times \h_{rails};

G_{tot\_mp\_sat} = 4/(1/G_{mp\_sat} + 1/G_{rail}) \quad \% 4 connecting points

%% Thermal coupling between Nanospace tank and metal plate
% The tank is screwed to the metal plate with a contact area of about 7.5E-5 m^2
% which is an estimate taken from the drawings
A_{tank\_screw} = 7.5E-5;
\lambda_{tank\_mp} = 300;
G_{tank\_mp} = 4 \times A_{tank\_screw} \times \lambda_{tank\_mp} \quad \% 4 connecting points

%% Thermal coupling between Nanospace cover plate and the satellite (rails)
A_{cp\_sat} = 4 \times 0.005^2;
\lambda_{cp\_sat} = 400;
G_{cp\_sat} = A_{cp\_sat} \times \lambda_{cp\_sat}

%% Thermal coupling between the vertical and horizontal PCBs
% with flex-rigid connectors
k_{Cu} = 387;
A_{flex.rig} = 0.00007*0.01;
x_{flex.rig} = 0.05;
h_{flex} = 300;

G_{flex.rig} = 4*(k_{Cu}*A_{flex.rig})/x_{flex.rig};
G_{flex.rig.contact} = A_{flex.rig}*h_{flex};

G_{tot.flex.rig} = 1/(1/G_{flex.rig} + 2/G_{flex.rig.contact})

%% Thermal coupling Morebac - PCB connector to PCB
A_{pin} = 0.0005^2;
x_{pin} = 0.01;
h_{pin} = 500;
k_{pb} = 50.0259; % Conductivity of Phosphor bronze

G_{pin} = 104*(k_{Cu}*A_{pin})/x_{pin};
G_{contact} = 104*A_{pin}*h_{pin};
G_{tot.pin} = 2/((1/G_{pin}) + (1/G_{contact}));
% Times 2 if 2 connectors in the thermal coupling and then checkbox
% 'per element' not crossed in!

%% Thermal coupling Morebac - PCB to sample cassetts
A_{samp.cass} = 0.090*0.060;
h_{samp.cass} = 250;

G_{samp.cass} = A_{samp.cass}*h_{samp.cass}

%% Thermal coupling Morebac - Liquid chamber to illumination layer
A_{liq} = 0.03*0.09;
h_{liq} = 250;
G_{liq} = A_{liq}*h_{liq};

%% Thermal coupling Spacelink - PCB/EMI shield (heat sink) to satellite
A_{sh} = 0.1^2; % Area of the metal contact to the screw/PCB
h_{screw} = 300;
G_{\text{EMI}} = A_{\text{sh}} \cdot h_{\text{screw}};

A_{\text{thrm\_coupl\_rail}} = 0.0085^2;
\text{h\_rails} = 400;
G_{\text{rail}} = A_{\text{thrm\_coupl\_rail}} \cdot h_{\text{rails}};

G_{\text{EMI\_sat}} = \frac{1}{1/G_{\text{EMI}} + 1/G_{\text{rail}}}

%% Thermal coupling Spacelink - EMI shield to PCB

A_{\text{EMI}} = 0.05^2;
\text{h\_EMI\_PCB} = 300;

G_{\text{EMI\_PCB}} = A_{\text{EMI}} \cdot h_{\text{EMI\_PCB}}

%% Thermal coupling Battery - Batteries to PCB

A_{\text{bat}} = 0.037 \cdot 0.058;
\text{h\_bat} = 200;

G_{\text{bat}} = 2 \cdot A_{\text{bat}} \cdot h_{\text{bat}}

%% Thermal coupling Magnetorquer - Rod to PCB

A_{\text{interface\_rod}} = 0.07 \cdot 0.002;
\text{h\_rod} = 200;

G_{\text{rod\_PCB}} = 2 \cdot A_{\text{interface\_rod}} \cdot h_{\text{rod}}

%% Thermal coupling Magnetorquer - Air torquer to PCB

A_{\text{air}} = 4 \cdot 0.013 \cdot 0.049 + \pi \cdot 0.02^2;
\text{h\_air} = 200;
G_{\text{mgntq\_air}} = A_{\text{air}} \cdot h_{\text{air}}

%% Thermal coupling Ratex-J - Between electronic boxes

A_{\text{box}} = 0.02 \cdot 0.042 + 0.042 \cdot 0.052;
\text{h\_box} = 200;
G_box = A_box*h_box

%% Thermal coupling Ratex-J - between cylindrical detectors & PCB

A_cyl = 2*pi*0.0127^2;
h_cyl = 100;

G_cyl_PCB = A_cyl*h_cyl

%% Thermal coupling Ratex-J - Between electrical boxes to satellite
% Modeled as a thermal coupling with four screws with a screw head radius of
% 0.00398 m (Numbers taken from the screws from the PCB conductance experiment)

% r_sh = 0.00398; % Radius of the screw head
% A_sh = pi*r_sh^2;
A_sh = 0.02^2
h_box_sat = 500;
G_box_sat = A_sh*h_box_sat

%% Thermal coupling - Camera Lens to PCB

h_lens = 200;
A_lens = pi*0.0085^2;

G_lens_pcb = A_lens*h_lens

%% Thermal coupling - Camera lens glass to camera lens housing

h_lensglass = 250;
A_lensglass = 2*0.0085*pi*0.003;

G_lensglass = h_lensglass*A_lensglass

%% Thermal coupling - Legs PCB to metal plate

A_pcb_legs = 4934*10^-6/2;
h_legsplate = 100;

G_tot_pcb_legs = h_legsplate*A_pcb_legs
%% Thermal coupling - Legs motor to metal plate

h_legsmotor = 500;
A_legsmotor = 0.0218*0.056;

G_tot_legsmotor = h_legsmotor*A_legsmotor

Calculations for Simple Exterior Model of MIST

% MIST - MIniature STudent satellite
% Simplified thermal model
% KTH Ryal Institute of Technology
% Andreas Berggren
% 2015-04-13

clear all
close all
clc

A Rails = (4655/2)*10^(-6); % Areas from NX
A_sp = 0.1715; % Inclusive deployable solar panels
A_top = (9795/2)*10^(-6);
A_bottom = (20089/2)*10^(-6);
A_tot = A_rails + A_sp + A_top + A_bottom;

lambda_rails = 154; % Thermal conductivity of Al6061
lambda_sp = 387; % copper
lambda_top = 35.3; % Lead
lambda_bottom = 130; % Aluminum alloy 7075

lambda_avg = lambda_rails*(A_rails/A_tot) + lambda_sp*(A_sp/A_tot) + ...
lambda_top*(A_top/A_tot) + lambda_bottom*(A_bottom/A_tot)

eps_rails = 0.05;
eps_sp = 0.83;
eps_top = 0.08;
eps_bottom = 0.05;
eps_avg = eps_rails*(A_rails/A_tot) + eps_sp*(A_sp/A_tot) + ...
Appendix A. Matlab codes

\[ eps_{top} \times (A_{top} / A_{tot}) + eps_{bottom} \times (A_{bottom} / A_{tot})\]

\[\alpha_{rails} = 0.15;\]
\[\alpha_{sp} = 0.75;\]
\[\alpha_{top} = 0.15;\]
\[\alpha_{bottom} = 0.15;\]
\[\alpha_{avg} = \alpha_{rails} \times (A_{rails} / A_{tot}) + \alpha_{sp} \times (A_{sp} / A_{tot}) + \ldots\]
\[\alpha_{top} \times (A_{top} / A_{tot}) + \alpha_{bottom} \times (A_{bottom} / A_{tot})\]

%% Thermal conductivity through hinges

\[ A_{hinge} = 0.02 \times (0.02 / 2); \] % Contact area of one hinge with the
\[ h_{hinge} = 400; \] % solar panels
\[ k_{hinge} = A_{hinge} \times h_{hinge} \]

\[ k_{tot_{hinge}} = 1/(1/k_{hinge} + 1/k_{hinge}) \]

---

Generating plot for effective emittance vs. number of aluminized layers

% Plot of effective emittance
% KTH Royal Institute of Technology
% Andreas Berggren
% 2015-05-27

clear all
close all
clc

\[ n = \text{linspace}(2,100); \]
\[ \text{eps} = 0.05; \]

\[ \text{eps}_{star} = \text{eps} / ((n-1) \times (2-\text{eps})); \]

\[ \text{plot}(n,\text{eps}_{star}) \]
\[ \text{title('Effective emittance vs. number of single aluminized layers'}) \]
\[ \text{xlabel('Number of aluminized layers')} \]
\[ \text{ylabel('Effective emittance, \epsilon\star')} \]
Appendix B

Figures
Figure B.1: The net radiation flux of one node of a PCB for values on emissivity of 0.2, 0.4, ..., 1.0 as a function of time.
Figure B.2 show a complete overview of the payload and subsystems location within the spacecraft and a comparison between the CAD model (right image) and the meshed model (left image).

Figure B.2 show a complete overview of the payload and subsystems location within the spacecraft and a comparison between the CAD model and the meshed model.

Figure B.3 show the equilibrium temperature distribution within the spacecraft at time zero, these temperatures will change as the spacecraft during the orbit.
Figure B.3: The equilibrium temperature distribution of the spacecraft at time zero.
# Appendix C

## Tables

Table C.1: Material Properties where $\alpha$ is absorptivity, $\epsilon$ is emissivity, $k$ is conductivity and $C$ is specific heat

<table>
<thead>
<tr>
<th>Material</th>
<th>$\alpha$</th>
<th>$\epsilon$</th>
<th>$k$ [W/mK]</th>
<th>$C$ [J/kgK]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aluminum 6061</td>
<td>0.15</td>
<td>0.05</td>
<td>180</td>
<td>896</td>
</tr>
<tr>
<td>Aluminum Alloy 7075</td>
<td>0.15</td>
<td>0.05</td>
<td>130</td>
<td>960</td>
</tr>
<tr>
<td>Copper C10100</td>
<td>0.9</td>
<td>0.8</td>
<td>387</td>
<td>385</td>
</tr>
<tr>
<td>Glass</td>
<td>0.75</td>
<td>0.83</td>
<td>0.8</td>
<td>800</td>
</tr>
<tr>
<td>Titanium Alloy</td>
<td>0.52</td>
<td>0.1</td>
<td>12.2</td>
<td>525</td>
</tr>
<tr>
<td>Silicon</td>
<td>0.6</td>
<td>0.9</td>
<td>130</td>
<td>700</td>
</tr>
<tr>
<td>Steel</td>
<td>0.5</td>
<td>0.25</td>
<td>45</td>
<td>434</td>
</tr>
<tr>
<td>Lead</td>
<td>0.15</td>
<td>0.08</td>
<td>35.3</td>
<td>160</td>
</tr>
<tr>
<td>Brass</td>
<td>0.096</td>
<td>0.03</td>
<td>116</td>
<td>380</td>
</tr>
<tr>
<td>Polypropylene</td>
<td>-</td>
<td>0.97</td>
<td>0.12</td>
<td>2000</td>
</tr>
<tr>
<td>Kapton</td>
<td>0.41</td>
<td>0.52</td>
<td>0.12</td>
<td>1090</td>
</tr>
<tr>
<td>FR4</td>
<td>-</td>
<td>-</td>
<td>0.25</td>
<td>600</td>
</tr>
</tbody>
</table>
### Table C.2: Ganymede Planet Data

<table>
<thead>
<tr>
<th>Planet data</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Planet Period</td>
<td>615600 sec</td>
</tr>
<tr>
<td>Mean Radius</td>
<td>2631.2 km</td>
</tr>
<tr>
<td>Gravitational Constant</td>
<td>$1.428 \text{m/sec}^2$</td>
</tr>
<tr>
<td>Sun-Planet Distance</td>
<td>7.78547e+008 km</td>
</tr>
<tr>
<td>Account for Planet Flux and Albedo</td>
<td>true</td>
</tr>
<tr>
<td>Albedo vs Latitude</td>
<td>0.44</td>
</tr>
<tr>
<td>Sunlight Side Flux</td>
<td>Black Body Temperature</td>
</tr>
<tr>
<td>Black Body Temperature</td>
<td>160 K</td>
</tr>
<tr>
<td>Dark Side Flux</td>
<td>Black Body Temperature</td>
</tr>
<tr>
<td>Sunlight Side Flux</td>
<td>Black Body Temperature</td>
</tr>
<tr>
<td>Black Body Temperature</td>
<td>90 K</td>
</tr>
<tr>
<td>Radial Mesh Density</td>
<td>8</td>
</tr>
<tr>
<td>Circumferential Mesh Density</td>
<td>16</td>
</tr>
</tbody>
</table>

### Table C.3: Ganymede Solar Data

<table>
<thead>
<tr>
<th>Solar data</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Solar Declination</td>
<td>0 degrees</td>
</tr>
<tr>
<td>Sun Right Ascension</td>
<td>0 degrees</td>
</tr>
<tr>
<td>Solar Flux</td>
<td>46W/m²</td>
</tr>
</tbody>
</table>

### Table C.4: Ganymede Orbital Parameters

<table>
<thead>
<tr>
<th>Planet data</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Local Time at Ascending Node</td>
<td>03/24/2015 0:00:00</td>
</tr>
<tr>
<td>Other Classical Specify Menu1</td>
<td>Minimum Altitude</td>
</tr>
<tr>
<td>Minimum Altitude</td>
<td>500 km</td>
</tr>
<tr>
<td>Other Classical Specify Menu2</td>
<td>Eccentricity</td>
</tr>
<tr>
<td>Eccentricity</td>
<td>0</td>
</tr>
<tr>
<td>Orbit Inclination</td>
<td>90 degrees</td>
</tr>
<tr>
<td>Argument of Periapsis</td>
<td>0 degrees</td>
</tr>
<tr>
<td>Other Classical Position Method</td>
<td>Local Noon to Ascending Node Angle</td>
</tr>
<tr>
<td>Local Noon to Ascending Node Angle</td>
<td>0 degrees</td>
</tr>
</tbody>
</table>
Appendix D

Derivations

Derivation of Effective Emittance

Consider Figure D.1 which shows the schematics of the radiation between three layers in the MLI blanket. The schematics gives that:

\[ q_{12} = q_1 = -q_2 = \frac{1 - \epsilon_1}{\epsilon_1 A_1} + \frac{1}{A_1 F_{13}} + \frac{1 - \epsilon_{3,1}}{\epsilon_{3,1} A_3} + \frac{1 - \epsilon_{3,2}}{\epsilon_{3,2} A_3} + \frac{1}{A_3 F_{32}} + \frac{1 - \epsilon_2}{\epsilon_2 A_2} \]  \hspace{1cm} (D.1)

where \( \epsilon^* \) is the effective emittance. Simplifying the equation with \( A = A_1 = A_2 = A_3 \), \( \epsilon = \epsilon_1 = \epsilon_2 = \epsilon_{3,1} = \epsilon_{3,2}, F_{13} = F_{32} = 1 \), this gives:

\[ q_{12} = q_1 = -q_2 = \frac{\sigma(T_1^4 - T_2^4)}{1 - \epsilon + \frac{1 - \epsilon_{3,1}}{\epsilon_{3,1} A_3} + \frac{1 - \epsilon_{3,2}}{\epsilon_{3,2} A_3} + \frac{1}{A_3 F_{32}} + \frac{1 - \epsilon_2}{\epsilon_2 A_2}} = \epsilon^* \frac{\sigma(T_1^4 - T_2^4)}{1 - \epsilon} \]  \hspace{1cm} (D.2)

Expanding this to yield for \( n \) number of layers one get \( n - 1 \) number of ones and \( 2n - 2 \) of the term \( \frac{1 - \epsilon}{\epsilon} \) which gives:

\[ \epsilon^* = \frac{1}{n - 1 + (2n - 2) \frac{1 - \epsilon}{\epsilon}} = \frac{\epsilon}{2n - 2 - n\epsilon + \epsilon} = \frac{\epsilon}{(n - 1)(2 - \epsilon)} \]  \hspace{1cm} (D.3)
Bibliography


