DEVELOPMENT AND ANALYSIS OF THE THERMAL DESIGN FOR THE OSIRIS-3U CUBESAT

A Thesis in
Aerospace Engineering

by

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Submitted in Partial Fulfillment of the Requirements for the Degree of

Master of Science

August 2012
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**ABSTRACT**

Proper spacecraft thermal design ensures spacecraft reliability despite the variations in its environment that it will encounter, such as heat inputs from the Sun and Earth, and the influence of full sun-lit and eclipse phases. This project entails the research and development of the thermal controls design for the Orbital Satellite for Investigating the Response of the Ionosphere to Stimulation and Space Weather (OSIRIS-3U), being developed by The Pennsylvania State University’s Student Space Programs Laboratory.

An understanding of the thermal environment expected for OSIRIS-3U is discussed in detail and expected worst case orbital scenarios are described. In addition, analytical calculations of the thermal balance are presented along with FEA thermal models produced in COMSOL Multiphysics for determining the worst case temperature limits that OSIRIS-3U can expect during its mission. The results obtained reveal analytical steady-state temperature limits between +113 °C and −56 °C. Time-dependent computer FEA models show that the CubeSat can reach temperatures between +29 °C and −12 °C for worst-case cold, and +109 °C and −4.4 °C for worst-case hot. These results served as a basis for the development of an Independently Operating Thermal Control for OSIRIS-3U.

Furthermore, a theoretical understanding and analytical study of different types of Thermoelectric Generators (TEGs) for energy harvesting applications in CubeSats was performed. The results indicate that thin-film TEG devices deliver higher power density compared to traditional TEGs, but due to the low temperature gradient expected in a CubeSat, the efficiency of these devices falls below 1%. Hence, with mass and cost constraints on CubeSats, current TEG technology for use in this application does not appear viable.
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<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>1U</td>
<td>1 Unit CubeSat</td>
</tr>
<tr>
<td>2U</td>
<td>2 Unit CubeSat</td>
</tr>
<tr>
<td>3U</td>
<td>3 Unit CubeSat</td>
</tr>
<tr>
<td>CDH</td>
<td>Command and Data Handling Subsystem</td>
</tr>
<tr>
<td>COM</td>
<td>Communication Subsystem</td>
</tr>
<tr>
<td>FEA</td>
<td>Finite Element Analysis</td>
</tr>
<tr>
<td>GMM</td>
<td>Geometric Mathematical Model</td>
</tr>
<tr>
<td>GNC</td>
<td>Guidance, Navigation, and Control Subsystem</td>
</tr>
<tr>
<td>IR</td>
<td>Infrared</td>
</tr>
<tr>
<td>LEO</td>
<td>Low Earth Orbit</td>
</tr>
<tr>
<td>MLI</td>
<td>Multilayer Insulation</td>
</tr>
<tr>
<td>MOM</td>
<td>Multiple Orbit Mission</td>
</tr>
<tr>
<td>OPAL</td>
<td>Orbital Picosat Automated Launcher</td>
</tr>
<tr>
<td>OSIRIS</td>
<td>Orbital Satellite for Investigating the Response of the Ionosphere to Stimulation and Space Weather</td>
</tr>
<tr>
<td>PWR</td>
<td>Power Subsystem</td>
</tr>
<tr>
<td>SAPHIRE</td>
<td>Stanford Audio Phonic Photographic InfraRed Experiment</td>
</tr>
<tr>
<td>SSDL</td>
<td>Student Space Development Laboratory</td>
</tr>
<tr>
<td>TCS</td>
<td>Thermal Control Subsystem</td>
</tr>
<tr>
<td>TE</td>
<td>Thermoelectric</td>
</tr>
<tr>
<td>TEG</td>
<td>Thermoelectric Generator</td>
</tr>
<tr>
<td>WCC</td>
<td>Worst Case Cold</td>
</tr>
<tr>
<td>WCH</td>
<td>Worst Case Hot</td>
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<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
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<tr>
<td>$A$</td>
<td>Area, $m^2$</td>
</tr>
<tr>
<td>$A_i$</td>
<td>Area of surface $i$, $m^2$</td>
</tr>
<tr>
<td>$A_j$</td>
<td>Area of surface $j$, $m^2$</td>
</tr>
<tr>
<td>$dA_i$</td>
<td>Infinitesimal area of surface $i$, $m^2$</td>
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<td>$dA_j$</td>
<td>Infinitesimal area of surface $j$, $m^2$</td>
</tr>
<tr>
<td>$A_{prj}$</td>
<td>Projected area, $m^2$</td>
</tr>
<tr>
<td>$A_{sc}$</td>
<td>Spacecraft area, $m^2$</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>Albedo factor</td>
</tr>
<tr>
<td>$C_1$</td>
<td>$0.59544 \times 10^{-16}$ Wm$^2$</td>
</tr>
<tr>
<td>$C_2$</td>
<td>$14.388$ $\mu$K</td>
</tr>
<tr>
<td>$C_p$</td>
<td>Specific heat capacity, J/K</td>
</tr>
<tr>
<td>$E$</td>
<td>Total emissive power of real body, W</td>
</tr>
<tr>
<td>$E_\lambda$</td>
<td>Energy emitted by real body, W/m$^2$</td>
</tr>
<tr>
<td>$E_B$</td>
<td>Energy emitted by black body, W/m$^2$</td>
</tr>
<tr>
<td>$F_{A_iA_j}$</td>
<td>View factor between surface $i$ and $j$</td>
</tr>
<tr>
<td>$F_\parallel$</td>
<td>Surface normal parallel view factor</td>
</tr>
<tr>
<td>$F_\perp$</td>
<td>Surface normal perpendicular view factor</td>
</tr>
<tr>
<td>$H$</td>
<td>Irradiance, W</td>
</tr>
<tr>
<td>$h$</td>
<td>Convective heat transfer coefficient, W/m$^2$K</td>
</tr>
<tr>
<td>$I$</td>
<td>Current, A</td>
</tr>
<tr>
<td>$i$</td>
<td>Orbital Inclination, degrees</td>
</tr>
<tr>
<td>$J$</td>
<td>Radiosity, W</td>
</tr>
<tr>
<td>$k$</td>
<td>Thermal conductivity, W/mK</td>
</tr>
<tr>
<td>Symbol</td>
<td>Description</td>
</tr>
<tr>
<td>--------</td>
<td>--------------------------------------------</td>
</tr>
<tr>
<td>N</td>
<td>Number of semiconductors</td>
</tr>
<tr>
<td>P</td>
<td>Peltier coefficient, $V$</td>
</tr>
<tr>
<td>$Q_{\text{albedo}}$</td>
<td>Earth albedo heat flux, $W/m^2$</td>
</tr>
<tr>
<td>$Q_{\text{in}}$</td>
<td>Incoming heat energy, $W$</td>
</tr>
<tr>
<td>$Q_{\text{int}}$</td>
<td>Total internal heat energy, $W$</td>
</tr>
<tr>
<td>$Q_{\text{IR}}$</td>
<td>Earth infrared flux, $W/m^2$</td>
</tr>
<tr>
<td>$Q_{\text{net}}$</td>
<td>Net flow of thermal energy, $W$</td>
</tr>
<tr>
<td>$Q_{\text{out}}$</td>
<td>Outgoing heat energy, $W$</td>
</tr>
<tr>
<td>$Q_{\text{planet}}$</td>
<td>Total Earth heat flux, $W$</td>
</tr>
<tr>
<td>$Q_{\text{rad}}$</td>
<td>Radiated heat flux, $W$</td>
</tr>
<tr>
<td>$Q_s$</td>
<td>Solar heat flux, $W/m^2$</td>
</tr>
<tr>
<td>$Q_{\text{solar}}$</td>
<td>Total solar heat flux, $W$</td>
</tr>
<tr>
<td>$q$</td>
<td>Heat flux, $W/m^2$</td>
</tr>
<tr>
<td>$q_{\text{cond}}$</td>
<td>Conductive heat flow, $W$</td>
</tr>
<tr>
<td>$q_{\text{conv}}$</td>
<td>Convective heat flow, $W$</td>
</tr>
<tr>
<td>$R$</td>
<td>Distance between surfaces $i$ and $j$, m</td>
</tr>
<tr>
<td>$R_e$</td>
<td>Radius of Earth, km</td>
</tr>
<tr>
<td>$R_{\text{in}}$</td>
<td>Internal resistance, $\Omega$</td>
</tr>
<tr>
<td>$R_{\text{load}}$</td>
<td>Load resistance, $\Omega$</td>
</tr>
<tr>
<td>$S$</td>
<td>Seebeck coefficient, $V/K$</td>
</tr>
<tr>
<td>$\Delta T_{\text{avg}}$</td>
<td>Average temperature gradient, $K$</td>
</tr>
<tr>
<td>$T_c$</td>
<td>Cold temperature, $K$</td>
</tr>
<tr>
<td>$T_h$</td>
<td>Hot temperature, $K$</td>
</tr>
<tr>
<td>$T_{\text{max}}$</td>
<td>Maximum temperature limit, °C</td>
</tr>
<tr>
<td>$\Delta T_{\text{max}}$</td>
<td>Maximum temperature gradient, $K$</td>
</tr>
<tr>
<td>$T_{\text{min}}$</td>
<td>Minimum temperature limit, °C</td>
</tr>
<tr>
<td>$\Delta T_{\text{min}}$</td>
<td>Minimum temperature gradient, $K$</td>
</tr>
<tr>
<td>$dU/dt$</td>
<td>Change in internal energy, $J/s$</td>
</tr>
<tr>
<td>$V_{\text{oc}}$</td>
<td>Open circuit voltage, $V$</td>
</tr>
<tr>
<td>$W$</td>
<td>Work done by the system to its surroundings, $J$</td>
</tr>
<tr>
<td>$\Delta x$</td>
<td>Change in length, $m$</td>
</tr>
<tr>
<td>$+X$</td>
<td>Port facing surface</td>
</tr>
<tr>
<td>$-X$</td>
<td>Starboard facing surface</td>
</tr>
<tr>
<td>$+Y$</td>
<td>Aft facing surface</td>
</tr>
<tr>
<td>$-Y$</td>
<td>Forward (velocity vector) facing surface</td>
</tr>
<tr>
<td>$+Z$</td>
<td>Zenith facing surface</td>
</tr>
<tr>
<td>$-Z$</td>
<td>Nadir facing surface</td>
</tr>
<tr>
<td>$z$</td>
<td>Orbital altitude, km</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>Absorption coefficient</td>
</tr>
<tr>
<td>$\beta_e$</td>
<td>Beta angle, degrees</td>
</tr>
<tr>
<td>$\epsilon$</td>
<td>Emission coefficient</td>
</tr>
<tr>
<td>$\eta$</td>
<td>TEG efficiency, %</td>
</tr>
<tr>
<td>$\theta_{\text{HS}}$</td>
<td>Thermal resistance of heat sink, $(m^2K)/W$</td>
</tr>
<tr>
<td>$\theta_{\text{TEG}}$</td>
<td>Thermal resistance of TEG, $(m^2K)/W$</td>
</tr>
<tr>
<td>$\theta$</td>
<td>Orbital angle, degrees</td>
</tr>
<tr>
<td>$\lambda$</td>
<td>Wavelength, $m^2$</td>
</tr>
<tr>
<td>$\mu$</td>
<td>Thomson coefficient, $V/K$</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Density, $g/m^3$</td>
</tr>
<tr>
<td>$\sigma$</td>
<td>Boltzmann Constant, $W/(m^2K)$</td>
</tr>
</tbody>
</table>
Acknowledgments

First and foremost I would like to thank my advisor, Dr. Sven G. Bilén, for giving me the opportunity to work in his laboratory and for his support and guidance of my pursuit of a Master’s Degree in Aerospace Engineering; this thesis would have not been possible otherwise. Thanks to Dr. David Spencer, Dr. Michael Micci, and Robert Capuro for their time and efforts in reading and revising my thesis.

I would also like to thank the project manager Allen Kummer for his leadership in the Student Space Program Lab, his work and effort has developed and maintained an effective structure for all the subsystems of OSIRIS.

I’m also grateful for the leadership, intelligence, and dedication of my colleagues Michael Boeckel and Andre Coleman. Their support and guidance not only made possible the development of the OSIRIS Thermal Subsystem, but also the work presented in this thesis. Additional thanks to all other members of the Thermal Subsystem whose time and efforts went into selection and testing of the thermal control for OSIRIS.

I also want to thank Michael Conway for his help in network or computer related issues when I needed to work from home or helping me allocate more memory for my FEA simulations, or just finding bugs in my code.

Last but not least, I would like to give special thanks to Ignacio de Leon, who worked alongside with me through emails and Skype conference calls in learning the COMSOL software, and most importantly the development of thermal analysis of satellites in COMSOL, which is rarely used for this application.
Chapter 1. Introduction

A thermal control subsystem (TCS) is an essential element of any spacecraft design. Its primary purpose is to maintain the temperatures of all spacecraft components within allowable operational ranges during all mission phases. Proper thermal design ensures spacecraft reliability in performing its mission despite variations in the environment that the spacecraft encounters in space, such as heat inputs from the Sun, Earth, and the influence of full Sun-lit and eclipse phases. In addition, thermal radiation from internal components must be accounted to ensure relative temperature stability and to minimize temperature gradients.

All components of a spacecraft bus and its payload have temperature limitations. Table 1 lists typical temperature limits for common spacecraft components used in a mission design. They often encompass different temperature limitations depending on mode [14][15]:

- Operating temperature range: the maximum and minimum temperature limits between which components successfully and reliably meet their specified operating requirements.
- Turn-on temperature: the maximum and minimum temperature limits between which components are able to be turned on without experiencing any damage or malfunction.
- Non-operating temperature range: the maximum and minimum temperature limits within which components are able to survive while in a power off mode, and subsequently perform as required as required in the turn-on and operating modes.
Table 1. Typical Temperature Limits for Spacecraft Components.

<table>
<thead>
<tr>
<th>Component</th>
<th>Operating Temperature (°C)</th>
<th>Survival Temperature (°C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Batteries</td>
<td>−30 to 60</td>
<td>−50 to 60</td>
</tr>
<tr>
<td>General electronics</td>
<td>−40 to 85</td>
<td>−40 to 125</td>
</tr>
<tr>
<td>Infrared detectors</td>
<td>−269 to −173</td>
<td>−260 to 35</td>
</tr>
<tr>
<td>Solar panels</td>
<td>−100 to 125</td>
<td>−100 to 125</td>
</tr>
<tr>
<td>Solid-state particle detectors</td>
<td>−35 to 0</td>
<td>−35 to 35</td>
</tr>
</tbody>
</table>

Ideally, the design of the TCS must minimize mass and cost while optimizing its reliability. In order to achieve this there are two approaches to the thermal design [10]:

- **Passive Control**: includes the use of components such as space radiators, coatings/paints, multilayer insulation (MLI), heat pipes; and
- **Active Control**: includes the use of components such as heaters, pumped loop systems, Peltier elements.

In general, a thermal design utilizing only passive control composed of MLI, coatings, and paints is desirable over active control since it requires no power, is generally less expensive, and is lighter.

In a thermal design, size and complexity can come into play when minimizing cost and mass, and optimizing spacecraft performance. As spacecraft become smaller, the mass and power budgets decrease, thereby tightening design constraints. This makes passive control design more attractive since less power will be consumed, less heat will be generated by the electronics onboard the spacecraft, and passive designs are by their nature not complex and so are generally more reliable. The converse is typically true for larger spacecraft. However, active control design is necessary when one must control large heat dissipation and low temperature tolerances typical of electronics, thus adding to the complexity, mass, and cost of the spacecraft.

The overall goal for minimizing cost, mass, and power consumption and optimizing performance and reliability of a spacecraft has led to the development of a class of satellites known as nano-satellites.
One type of nanosatellite called a CubeSat has launched as secondary and tertiary payloads on a number of launch vehicles. The standard CubeSat has an envelope of 10×10×10 cm with a mass limit of 1.33 kilograms, and is referred to as a 1U CubeSat. CubeSats can increase in length up to a 3U CubeSat. Typically, CubeSats are deployed into Low Earth Orbit (LEO), which corresponds to orbital altitudes between 160 km and 2000 km. Advantages of these types of spacecraft include their reduced size, lower cost, and shorter time for design and development.

This project entails the research, development, and verification of a conceptual model and analytical model of the thermal controls for the Orbital Satellite for Investigating the Response of the Ionosphere to Stimulation and Space Weather (OSIRIS-3U), being developed by The Pennsylvania State University’s Student Space Program Laboratory, and shown in Figure 1. The OSIRIS-3U mission will provide in situ and remote-sensing measurements of the heated ionosphere, and will compare these results with ground-based measurements in order to better understand variable space weather and ionosphere irregularities [25]. The mass of OSIRIS-3U is targeted to be less than 4 kilograms, with dimensions of 10×10×30 cm. Its orbital parameters are targeted to be circular between altitudes of 300 and 400 kilometers and inclinations between 35° and 70°.

![Figure 1. ORISIS-3U CubeSat.](image)

Theoretical and mathematical development for the thermal control is discussed in detail in the following chapters, as well as the development of the analytical model using a single-node approximation.
for the prediction of worst-case-hot and worst-case-cold temperatures of the CubeSat. Additionally, verification of the analytical model is performed by developing computer simulations of the thermal environment in COMSOL Multiphysics, a finite element analysis (FEA) software, in order to verify that the model behaves as anticipated. Furthermore, studies are performed on the efficiency across low temperature gradients ($\Delta T$) of small Peltier junctions, or thermoelectric generators (TEGs), for energy harvesting on CubeSats.

This thesis provides contributions to the thermal design of OSIRIS-3U by understanding the theory behind the thermal environment of a CubeSat (Chapter 3), as well as the development of a geometric mathematical model (GMM) generated in MATLAB to calculate the incident heat flux, and a functional finite element model of the temperature variations of the CubeSat created in COMSOL Multiphysics using the Heat Transfer module (Chapter 4–5). Additional contributions include a theoretical understanding of TEGs (Chapter 3) and development of an analytical model of TEGs in MATLAB in order to determine the efficiency with which they can harvest additional energy for small spacecraft at low $\Delta T$.

Finally, results are provided for the GMM of the incident heat flux that different panels receive over one orbital period. Solution of the temperature variations over time for 20 orbital periods obtained from the finite element model is given and worst-case-cold and worst-case-hot temperatures are discussed and analyzed (Chapter 5). Lastly, results of the power output and efficiency of two commercial thermoelectric generators are given for the temperature variations of the CubeSat obtained from the Heat Transfer solver in COMSOL (Chapter 6).
Chapter 2. Thermal Control System of Small Satellites

CubeSats were developed to facilitate more space-based experiments by reducing satellite size, mass, cost, and development time, thereby providing more universities and small entities with the opportunity to build and fly satellites. CubeSats enable flight testing of new technologies and systems, much of which continue to get smaller in size. Though challenges are present, the interest in these CubeSats and other small spacecraft have fostered significant research in the use of these satellites that are readily deployable, flexible, and have autonomous networks, distributed computing, and power management capabilities [11] [24].

Challenges arise when technologies proven to work in large satellites are ported and integrated into small spacecraft and CubeSats [24]. The design challenges for the thermal control subsystem, passive or active, are limitations in mass, power budget, surface area, and attitude control for which the thermal engineers must account.

Conventional (i.e., larger) satellites have advantages over smaller satellites from a thermal control perspective due to their large mass and thermal inertia. Thermal inertia refers to the rate at which a material reaches thermal equilibrium and is dependent on the material’s thermal conductivity and volumetric heat capacity. For instance, the temperature extremes that any satellite can expect during its mission depend on the time spent in sunlight as well as in eclipse. Since larger spacecraft have greater thermal inertia compared to smaller satellites, during eclipse periods their temperature will decay more slowly; thus, their temperature will reach equilibrium more slowly compared to smaller satellites. Faster temperature decay in smaller satellites can result in components reaching temperatures below minimum operating limits. Additionally, due to smaller stored power capabilities, heaters must be used sparingly [7].
As stated previously, smaller spacecraft, in general, contain all of the primary subsystems that are part of larger spacecraft. However, volume and mass limitations do not allow them to incorporate heavy and bulky active controls such as louvers or radiators [7] used on large spacecraft. Another issue that arises with satellite downscaling is thermal balance control due to non-uniformities in size and location of electronics; placement of electronic components must be taken into careful consideration to minimize temperature gradients in internal electronic subassemblies [12]. Additionally, small spacecraft can experience high heat loads due to the compact integration of circuit boards, which makes it more difficult to provide a means of heat dissipation [11] [30]. The internal configuration of the satellite—e.g., how circuit boards and electronics are arranged—are configured such that internal heat loads are effectively dissipated [16].

Small satellites are typically low-Earth-orbiting satellites that experience a large number of thermal cycles and experience high heat inputs from solar radiation and Earth infrared. In order to prevent the satellite from heating too much, a thermal engineer can alter the optical properties of the satellite’s surface; however, the external surfaces of these small satellites generally are completely covered with solar cells to maximize power generation, which is already extremely limited. This coverage limits how much the material properties can be altered in order to maintain temperature within allowable limits [16] [22]. In addition, insulators such as MLI and thermal tapes can be used but, due to degradation and change in efficiency of insulating materials over time, their effectiveness in reducing the heat flows may cause the satellite to reach undesirable temperature limits [7]. Attitude control mechanisms also affect the thermal design of small spacecraft. For spin-stabilized satellites, it is general practice that internal components radiate the internal heat to the solar arrays, whereas for 3-axis stabilized satellites, radiators or exterior surfaces insulated from the thermal environment are implemented [7] [22].

Thermal design of small spacecraft has more limitations compared to that of larger spacecraft. Although passive thermal control is generally preferred for all spacecraft due to simplicity and cost, it is almost required in small spacecraft. A such, a thermal engineer must be able to test and simulate the
satellite to ensure proper thermal design that maintains all operating temperature limits within limited mass, power, and volume requirements. A thorough understanding of the thermal environment expected for the satellite is critical to ensure that the range of temperature limits are not reached and that temperature variations or thermal cycling is minimized [16].
Chapter 3. Thermal Analysis

Development of a thermal control system for all spacecraft in orbit begins with a theoretical understanding of the expected thermal environment. This chapter discusses the different types of heat transfer and heat sources that affect spacecraft in space. Additionally, a discussion of the different types of thermal controls commonly used in spacecraft is presented and, lastly, an understanding of thermoelectric generation theory is discussed. An implementation of TEGs in CubeSats is presented for energy harvesting applications.

3.1 Heat Transfer

Energy balance is the governing equation for all types of systems. For the thermal analysis of a spacecraft, the energy balance begins with the First Law of Thermodynamics, which states that energy can be neither created nor destroyed, merely transferred, i.e.,

\[
\frac{dU}{dt} = Q - W, \tag{3.1}
\]

which simply states that the change in internal energy, \( U \), of a system is equal to the amount of heat added to the system, \( Q \), minus the amount of work done by the system to its surroundings, \( W \). For a spacecraft, the quantity \( Q \) represents the heat flux and \( U \) the internal energy of electronics. The work done by a spacecraft to its surroundings is zero. Therefore, for non-steady state systems, Equation 3.1 becomes

\[
\frac{dU}{dt} = Q = AdxpC_p \frac{dT}{dt}, \tag{3.2}
\]

where \( U \) and \( Q \) are described in terms of a material’s cross section \( (A) \), length \( (dx) \), density \( (\rho) \), specific heat capacity \( (C_p) \), and temperature \( (T) \) of the system [17].

The heat balance equation for a spacecraft is described by the heat flux entering the system minus the heat flux leaving the system, i.e.,
\[ Q_{\text{net}} = Q_{\text{in}} - Q_{\text{out}}, \]

where the net flow of thermal energy of the spacecraft is characterized by conduction, convection, and radiation \[14][17].

### 3.1.1 Convection

Convection is described as the thermal energy transfer between a solid surface and flowing fluid. The rate of heat flow \((q)\) for convection heat transfer is defined as

\[ q_{\text{conv}} = hA(T_{ss} - T_f), \]

where \(q\) is in units of watts, \(h\) is the convective heat transfer coefficient \((W/m^2K)\), \(A\) is the surface area in \(m^2\), and the temperature difference between the solid surface and flowing fluid is in kelvin. For satellite missions, convective heating can occur during the mission launch of manned spacecraft. However, due to the absence of flowing fluids in space (i.e., vacuum), convection is neglected \[10][14].

### 3.1.2 Conduction

Conduction is the thermal energy transfer within a material due to the motion or kinetic energy of the individual atoms interacting with each other within the material. For a one-dimensional slab (Figure 2), the rate of heat transfer by conduction is based upon the cross sectional area \((A)\) normal to the direction of heat transfer and the length of heat transfer \(dx\) as well as the difference in temperature given in kelvin, i.e.,

\[ q_{\text{cond}} = \frac{kA}{\Delta x} (T_1 - T_2), \]

where \(k\) is the thermal conductivity \((W/mK)\).
Figure 2. Steady State Heat Conduction for Rectangular Coordinate System [4].

Appendix A provides figures and equations for cylindrical and spherical conduction heat transfer.

For spacecraft, conduction plays a significant role in the dissipated heat produced by the onboard electronics, which can cause heat transfer within their individual circuit boards and integrated electronics, and also heat transfer through mated surfaces known as contact conductance [14].

3.1.3 Radiation

All objects with temperatures above absolute zero emit and absorb electromagnetic energy. This phenomenon is known as thermal radiation, which is governed by the amount of radiant energy emitted by a perfect emitter or black body, also known as Planck’s law:

\[ E_{B\lambda}(\lambda, T) = \frac{2\pi C_1}{\lambda^5 \left( e^{\frac{c_2}{\lambda T}} - 1 \right)}, \]

where \( E_B \) is the energy for black body, \( \lambda \) is the wavelength, \( C_1 = 0.59544 \times 10^{-16} \) Wm\(^2\) and \( C_2 = 14.388 \) µmK. Figure 3 illustrates Planck’s Law as a function of wavelength and temperature.
When integrating Equation 3.6 over all wavelengths, we obtain the total black body emissive power

\[ E_B(T) = \int_0^\infty E_B(\lambda, T) d\lambda, \]  

\[ E_B(T) = \sigma T^4 \left[ \frac{W}{m^2} \right], \]  

where \( \sigma \) is the Stefan–Boltzmann constant and \( T \) the temperature in kelvin. Equation 3.8 is known as Stefan–Boltzmann’s Law, which states that the total emissive power, \( E_B \), of a black body is proportional to the temperature to the fourth power [17].

### 3.1.4 Radiation Properties

Equations 3.6 and 3.8 have been developed for a black body that is an idealized object with its ability to absorb and emit all incident radiation. However, no object is a black body, rather they are real bodies that absorb, reflect, and transmit incident radiation. These radiation properties are illustrated in Figure 4. For all direction (\( \theta \)) and wavelength (\( \lambda \)) dependency, the energy balance requires these properties to be unity; therefore [17],

\[ \alpha(\theta, \lambda) + \rho(\theta, \lambda) + \tau(\theta, \lambda) = 1, \]
Furthermore, for an object that is opaque, no radiation gets transmitted through the object and 

\[ \tau = 0, \]

therefore,

\[ \alpha + \rho = 1. \]

Figure 4. Absorption, Reflection, and Transmission of Real Body [17].

As stated above, all objects above absolute zero emit electromagnetic radiation. As a result, a surface emissivity is defined as a dimensionless quantity of the ratio between the emissive power of a real body and black body both at the same temperature. The common notation used is

\[ \epsilon_\lambda(T, \lambda) = \frac{E_\lambda(T, \lambda)}{E_{B\lambda}(T, \lambda)}. \]  

3.10

For a black body, the emissivity is \( \epsilon = 1 \), and for a real body \( 0 < \epsilon < 1 \). Hence, the total emissive power for a real body becomes dependent on the emissivity [29]:

\[ E(T) = \epsilon \sigma T^4 \left[ \frac{W}{m^2} \right]. \]  

3.11

For an object that has achieved thermal equilibrium, the emissivity equals the absorptivity,
\[ \varepsilon(\theta, \lambda) = \alpha(\theta, \lambda), \]  

which is known as Kirchhoff’s Law. In order for the theory of Kirchhoff’s Law to hold true, the surface must be diffuse (surface properties are independent of direction) and gray (surface properties are independent of wavelength) [17].

### 3.1.5 View Factors

Thermal radiation on a single surface is discussed above. However, in an environment such as space, thermal analysis between two surfaces that can exchange radiative energy must be considered and analyzed. The process by which the amount of radiative energy that any two surfaces exchange is described by view factors, \( F_{i-j} \). For an enclosed surface, the view factor is defined as the fraction of radiative energy that leaves surface \( i \) in a straight path and intercepts surface \( j \). The view factors depend only on the geometry, size, orientation, and distance between surfaces [8] [17]. Consider the radiative energy exchange between two finite areas, \( A_i \) and \( A_j \), as shown in Figure 5. We begin by determining the view factors for the infinitesimal areas, \( dA_i \) and \( dA_j \) to be

\[ F_{dA_i-dA_j} = \frac{\cos \theta_i \cos \theta_j}{\pi R^2} dA_i dA_j, \]  

where \( \theta \) is the angle between the normal surface vector and distance \( R \). Integrating over the two areas, \( A_i \) and \( A_j \), the view factor between the two finite areas is defined to be [29]

\[ F_{A_i-A_j} = \frac{1}{A_i} \int_{A_i A_j} \frac{\cos \theta_i \cos \theta_j}{\pi R^2} dA_i dA_j, \]  

An important property that arises from Equation 3.14 is the law of reciprocity, which is stated mathematically as

\[ A_i F_{A_i-A_j} = A_j F_{A_j-A_i}, \]  

since view factors are defined as the fraction of energy leaving one surface and striking another. For \( N \) number of surfaces, the summation of the view factors should equal unity; therefore,
3.1.6 Radiative Exchange between Surfaces

For the purpose of this analysis we will assume the surfaces to be gray, diffuse, and isothermal (temperature within a surface remains constant), such that

\[ \epsilon = \alpha = 1 - \rho. \]  

3.17

The energy balance of the net heat flux at a surface is equal to the total energy leaving the surface minus the energy falling onto the surface; these energies correspond to the radiosity and irradiance \( (H) \), respectively [17]. The energy balance is written

\[ q_{\text{net}} = \epsilon E + \rho H - H, \]  

3.18

for \( \rho = 1 - \alpha \) and \( \epsilon = \alpha \), Equation 3.18 becomes

\[ q_{\text{net}} = \epsilon E - \epsilon H, \]  

3.19

where \( E \) is simply

\[ \sum_{j=1}^{N} F_{i-j} = 1. \]  

3.16
\[ E_B(T) = \sigma T^4 \left[ \frac{\text{W}}{\text{m}^2} \right]. \] \hspace{1cm} 3.20

For the radiosity term, \( J \), the total outgoing energy depends on the emissivity and reflectivity properties of the surface. Therefore, the radiosity leaving the surface at \( x_1 \) is defined as
\[ J(x_1) = eE(x_1) + \rho H(x_1). \] \hspace{1cm} 3.21

By making use of the definition of gray and diffuse surface from Equation 3.17, the radiosity becomes
\[ J(x_1) = eE(x_1) + (1 - \epsilon)H(x_1). \] \hspace{1cm} 3.22

The overall energy balance of the net heat flux of a surface in terms of the radiosity and irradiosity terms is then
\[ q_{\text{net}}(x_1) = eE(x_1) - eH(x_1) = J(x_1) - H(x_1). \] \hspace{1cm} 3.23

The incoming energy, \( H \), is now found by enclosing the surface and defining two differential areas, \( dA_1 \) and \( dA_2 \), in order to determine the view factors between the two surfaces integrated over the entire area [8][17]. The rate of radiant energy that leaves surface \( dA_1 \) and intercepts surface \( dA_2 \) is defined as
\[ J(x_1) dA_1 dF_{dA_1-dA_2}; \] \hspace{1cm} 3.24

thus, the total incoming energy is defined to be
\[ H(x_2) dA_2 = \int_A J(x_1) dA_1 dF_{dA_1-dA_2} + H_0(x_2) dA_2. \] \hspace{1cm} 3.25

Applying the law of reciprocity, the total incoming energy is
\[ H(x) = \int_A J(x_1) dF_{dA-dA_1} + H_0(x_2), \] \hspace{1cm} 3.26

where \( H_0 \) is radiant energy impinging on \( dA_2 \).
The preceding analysis illustrates the impact of view factors in thermal analysis, energy balance of heat inputs, and outputs of any system. For spacecraft, view factors play a major role in the heat inputs from Earth and its albedo, which are discussed in further detail in the following sections.

### 3.2 Thermal Environment of Spacecraft

The thermal inputs that contribute to the heating of a spacecraft in a Low Earth Orbit (LEO) are couplings between the environment and internal heat generation within internal components. These major contributions for thermal radiation are described as:

1. Direct solar radiation (Environment),
2. Earth albedo (Environment),
3. Earth infrared radiation (Environment), and
4. Internal heat generation.

Figure 6 illustrates all the heat fluxes for a spacecraft orbiting Earth. The environmental heat inputs that contribute to the spacecraft depend on the orbital parameters, size, shape, view factors, and surface properties of the spacecraft.

![Figure 6. Thermal Inputs for Spacecraft in LEO [14].](image)
3.2.1 Direct Solar Radiation

Direct solar radiation is the amount of radiation that is emitted by the Sun and strikes any object in space. Due to the distance between the Sun and the Earth, the solar rays are consider to be collimated light with a constant solar heat flux, $Q_s$, between 1300 W/m$^2$ at winter solstice and 1400 W/m$^2$ at summer solstice (in the Northern hemisphere) [14][17]. The heat flux describing the amount of direct solar radiation absorbed by LEO spacecraft is defined as

$$Q_{solar} = Q_s A_{prj} \alpha,$$

where $A_{prj}$ is the projected area of the surface facing the Sun, and $\alpha$ is the solar absorption coefficient of the surface.

3.2.2 Earth Albedo

As direct solar radiation gets absorbed by planet Earth, a fraction of this flux is reflected out into space, which is known as the Earth reflected solar energy or albedo, $a$. The albedo factor can vary between 18% and 55% depending on the location on Earth and properties of reflecting surfaces; clouds and snow have high albedo compared to water. For spacecraft in LEO, typical albedo values are between 30% and 35% [14] [15]. Therefore, the amount of reflected heat flux that is absorbed by a spacecraft is

$$Q_{alb} = Q_s a A_{prj} F_{s-planet} \alpha,$$

where $a$ is the albedo factor and $F_{s-planet}$ the view factor from the spacecraft surface to Earth. Additionally, Equation 3.28 is valid for orbital angles, $\theta$, between

$$0^\circ < \theta < 90^\circ.$$

Assuming the Earth to be spherical and a face of a spacecraft to be a flat plate with its normal vector pointing towards nadir, the view factor can be approximated by [21]

$$F_i = \left( \frac{R_e}{z + R_e} \right)^2,$$

where $R_e$ is the Earth's radius.
where $R_e$ is the radius of Earth and $z$ the orbital altitude. For all other flat surfaces with their normal vector being $90^\circ$ from nadir, the view factor is approximated by [21]

$$F_\perp = \frac{1}{2\pi} \left[ -2 \sin^{-1}\sqrt{1 - \left(\frac{R_e}{z + R_e}\right)^2} - \sin\left(2 \sin^{-1}\sqrt{1 - \left(\frac{R_e}{z + R_e}\right)^2}\right) \right] \text{(rad)}. \quad 3.30$$

Computer-generated values of the view factors for flat surfaces have been plotted as a function of the ratio between orbital altitude and radius of Earth, and for different $\gamma$ angles (angle between surface normal and nadir). References [2] [19] describe the development and derivation of planetary view factors. These values are provided in Figure 7.

![Figure 7. Earth View Factor for Planar Surfaces as a Function of the Ratio of Altitude ($h$) and Radius of Earth ($R_e$) for Different Angles Between Surface Normal and Nadir Vector ($\gamma$), [29].](image_url)
3.2.3 Earth Infrared Radiation

The Earth is considered to be a source at constant temperature. This source results in the planet emitting radiant energy in the infrared (IR) region of the spectrum, which is known as the Earth infrared radiation. The Earth infrared flux, $Q_{\text{IR}}$, varies between 220 W/m$^2$ and 270 W/m$^2$ for winter solstice and summer solstice, respectively. As with Earth albedo, view factors between surfaces must be accounted for\cite{10}[29], which are given in Equations 3.29 and 3.30. Therefore, the total IR radiation absorbed by the spacecraft is given by

$$Q_{\text{planet}} = Q_{\text{IR}}A_{prj}F_{s-\text{planet}}\alpha_{\text{IR}},$$

where $Q_{\text{IR}}$ is the Earth IR heat flux and $\alpha_{\text{IR}}$ is the infrared absorption coefficient of the surface\cite{14}[20].

3.2.4 Thermal Balance

In order to determine the temperature of a spacecraft, a thermal or energy balance must be implemented to account for all energy entering and leaving the system. Therefore, in steady state the mathematical expression of the energy balance of a spacecraft is as follows:

$$Q_{\text{sun}} + Q_{\text{alb}} + Q_{\text{IR}} + Q_{\text{int}} - Q_{\text{rad}} = 0,$$

where the first three terms are heat flux contribution from the Sun, albedo, and Earth IR, respectively, which is mathematically described in the previous section. For the last two terms, $Q_{\text{int}}$ refers to the internal heat generated by the spacecraft subsystems and $Q_{\text{rad}}$ accounts for the radiation emitted by the spacecraft into space, which is defined as\cite{20},\cite{29}

$$Q_{\text{rad}} = \varepsilon\sigma T^4 A_{\text{tot}},$$

where $\varepsilon$ is the surface emissivity and $A_{\text{tot}}$ is the total area of the spacecraft. Thus,

$$Q_{\text{sun}} + Q_{\text{alb}} + Q_{\text{IR}} + Q_{\text{int}} = \varepsilon\sigma T^4 A_{\text{tot}}.$$  

3.3 Thermal Control Components

Thermal control design is driven by the spacecraft payload, bus, orbital parameters, and heat inputs from the environment and heat generation from internal components. In order to address
uncertainties in orbital parameters, such as those that any spacecraft encounters during its mission timeline, a set of orbital parameters that cause the worst-case conditions of environmental inputs that the spacecraft is expected to see were selected. A set of worst-case-hot (WCH) and worst-case-cold (WCC) temperature limits can then be calculated and a thermal control design can be defined using passive and active controls.

3.3.1 Passive Control

Typically, a thermal control design starts by examining possible passive control elements. Passive controls are more practical for CubeSats due to their size, shape, and the limited power budget constraints. Common passive controls include the following [10][14][17]:

- **Space radiators**: are controls used to directly radiate waste heat out into space; they are commonly on the outer surfaces of the spacecraft in order to efficiently dissipate the heat away from the spacecraft. Radiator surfaces have low absorption and high emissive coefficients.

- **Heat pipes**: in order for radiators to work efficiently, flow paths for waste heat generated by internal components must be available. Heat pipes provide the means by which unwanted heat is conducted directly to the radiators. They function as a system wherein a fluid inside the pipe is vaporized when heat is applied at one end of the tube. Gases formed from the vaporization then flow down the pipe to the opposite end.

- **Coatings/paints**: white paint is the most common passive thermal control method employed on spacecraft. In order to maximize the heat lost in a radiator, a coating can be applied to the surface since they have high emissivity. Coatings can also be placed on the surface of circuit boards to provide even distribution of heat across the board. However, exposure to the environment causes the absorptivity of coatings to increase over time, thus reducing their effectiveness over the lifetime of the mission.
• Multilayer insulation (MLI): is used on spacecraft to minimize the radiative heat loss from
the spacecraft when trying to keep components from freezing during eclipses. They are
typically alternating layers of Mylar or Kapton separated by thin sheets of nylon, which
provide high thermal impedance.

3.3.2 Active Controls

Active controls become significant in larger spacecraft due to the larger amount of heat dissipated
from internal components, and when trying to control low temperature tolerances. Cost, mass, and
reliability impacts of active control elements must be considered when compared to passive controls. The
common active controls on small spacecraft include [10][14][17]:

• Heaters: are composed of electrical resistance elements commonly embedded within two
sheets of Kapton. Heat is then generated through what is known as the Joule effect to heat up
components and keep them from reaching undesired cold temperatures.

• Peltier elements are solid state active heat controllers that directly convert electricity into heat
through an electrified junction between two metals.

The CubeSat constraints of low power, size, mass, and cost as well as the LEO orbital environment favor
passive thermal control systems as the preferred design solution.

3.4 Thermoelectric Effect

The thermoelectric effect is a phenomenon in which thermal energy is converted directly into
electrical voltage, or electrical voltage into thermal energy [4] [31]. These effects are characterized as
follows:

• Seebeck effect: The direct conversion of heat into electricity though devices known as
thermoelectric generators (TEGs).

• Peltier effect: The direct conversion of electricity into heat through an electrified junction
between two metals.
Thomson effect: The heating or cooling of carrying conductor in the presence of a temperature gradient.

All three effects are related by

\[ P = ST', \quad 3.35 \]
\[ \mu = T \frac{dS}{dT'}, \quad 3.36 \]

where \( P \) is the Peltier coefficient in V, \( S \) is the seebeck coefficient in V/K, \( T \) is the temperature in K, and \( \mu \) is the Thomson coefficient in V/K [27].

The governing equations for the thermoelectric effect are given by the heat energy balance in terms of the heat flux, \( q \), and the flux of electric current, \( J \), as

\[ q = -k \nabla T + PJ, \quad 3.37 \]
\[ J = -\sigma \nabla V - \sigma S \nabla T \quad 3.38 \]
\[ \rho C_p \frac{\partial T}{\partial t} + \nabla \cdot q = Q, \quad 3.39 \]

where \( k \) is the thermal conductivity, \( \rho \) is the density, \( C_p \) is the heat capacity, \( \sigma \) electric conductivity, and \( Q \) the joule heating [27][31].

For typical energy harvesting applications, thermoelectric generators are of primary focus. These devices are solid-state systems made of thermally conductive materials as well as semiconductor p–n junctions as seen in Figure 8. Most thermoelectric generators contain \( N \) number of semiconductors that are wired electrically in series and thermally in parallel [1].
When a thermal gradient is applied to a TEG, it causes charge carriers within the semiconductor junction to diffuse from the hot side to the cold side, thus producing a thermoelectric voltage that is dependent on the Seebeck coefficient, $S$. The Seebeck coefficient measures the magnitude at which a thermoelectric voltage is created in the presence of a temperature difference. Since metals have small Seebeck coefficients, semiconductor materials can be doped in order to achieve higher Seebeck coefficients values and, in turn, obtain higher thermoelectric voltage from a TEG [4][31]

### 3.4.1 Thermoelectric Generation Theory

The power output produced in the presence of a temperature difference in a TEG is proportional to the efficiency ($\eta$) and the heat flux entering from the hot side ($Q_h$), i.e.,

$$P_o = \eta Q_h,$$  \hfill 3.40

with $\eta$ defined as

$$\eta = \frac{T_h - T_c}{R_C \left( \frac{1}{1 + ZT_M} - 1 \right)} = \frac{T_h - T_c}{T_R \left( \frac{1}{1 + ZT_M} + \left( \frac{T_c}{T_h} \right)^2 \right)}.$$  \hfill 3.41
where \( T_c \) and \( T_h \) correspond to the cold and hot temperatures of the TEG, respectively, and \( ZT_M \) is the Figure of Merit (\( ZT \)) of the thermoelectric (TE) material at an average temperature, \( T_m \) [4][9], given as

\[
ZT_m = \frac{S^2 (T_h + T_c)}{RKK} \frac{2}{2},
\]

where \( S \) is the Seebeck coefficient

\[
S = S_p - S_n,
\]

\( R \) is the TEG internal electrical resistance,

\[
R_{in} = N \left[ \frac{\rho_n L_n}{A_n} + \frac{\rho_p L_p}{A_p} \right] = \frac{2N\rho L}{A},
\]

\( K \) is the TEG thermal conductivity,

\[
K = N \left[ \frac{\lambda_n A_n}{L_n} + \frac{\lambda_p A_p}{L_p} \right] = \frac{2N\lambda A}{L},
\]

where \( N \) is the number of pair legs of p–n thermocouples, \( \rho \) is electrical resistance for p–n type junctions, \( \lambda \) is the thermal conductivity, \( A \) is the cross sectional area of p–n junctions, and \( L \) is the length of p–n junctions [4][9]. Figure 9 shows a TEG’s efficiency as a function of the temperature difference, \( \Delta T \), applied to the TEG for different Figure of Merit values. As seen in the figure, as \( \Delta T \) increases, the TEG efficiency increases as a function increasing \( ZT \) values.
Figure 9. TEG Efficiency as a Function of Temperature Difference, $\Delta T$, and Figure of Merit, $ZT$ [4].

Additionally, the TEG open-circuit voltage generation is proportional to the thermal gradient across the device [4][9][13],

$$V_{oc} = NS(T_h - T_c). \quad 3.46$$

Equation 3.46 implies that the larger the number of thermocouples in the TEG, the higher the voltage output produced. The relationship for current, $I$, is defined as [3]

$$I = \frac{V_{oc}}{R} = \frac{NS(T_h - T_c)}{R_{in} + R_{load}}, \quad 3.47$$

where $R_{load}$ is the external electrical resistance of the TEG. The power output can then be expressed in terms of current, electrical load resistance, $R_{load}$, and voltage output, $V_{oc}$ as

$$P_o = I^2 R_{load}. \quad 3.48$$

Furthermore, the heat flux from the hot side of the TEG is given by

$$Q_h = SIT_h + \frac{1}{2} I^2 R - K\Delta T, \quad 3.49$$

where the first term corresponds to the Peltier effect, the second term to Joule heat loss from current flow, and the last term is the thermal conduction through p–n junctions [9].
For a given TEG operating across a given $\Delta T$, optimal TEG efficiency can be achieved by minimizing the denominator ($RK$) in Equation 3.42. This minimum value is obtained when

$$RK = \left[\left(\rho_n \lambda_n\right)^{\frac{1}{2}} + \left(\rho_p \lambda_p\right)^{\frac{1}{2}}\right]^2.\tag{3.50}$$

This results in an increase in the Figure of Merit, thus, an increase in TEG efficiency and power output. However, optimal efficiency does not necessarily mean optimal power output since power is also related to electrical resistance. For optimal power output, the internal electrical resistance and load resistance in the TEG must be matched [3][9], which results in $P_o$ from Equation 3.40 becoming

$$P_{\text{max}} = \frac{V_{\text{oc}}^2}{4R_{\text{in}}}.\tag{3.51}$$

For effective performance, a constant heat source is required to maintain a $\Delta T$ across the device as well as a method of heat rejection at the cold side. Heat rejection can be achieved through methods such as liquid cooling, spray cooling, phase change, or applying a large thermal mass or heat sink to the cold side of the TEG. Heat sinks are the most common and least expensive methods used for heat rejection. Their effective performance is characterized by their thermal resistance,

$$\Theta_{\text{HS}} = \frac{T_{\text{HS}} - T_{\text{amb}}}{Q_h},\tag{3.52}$$

where low thermal resistance implies greater effectiveness of rejecting heat to the ambient and vice versa. When heat sinks are connected thermally to TEGs, the $\Delta T$ between the TEG and heat sink varies depending on their corresponding thermal resistances. For instance, if the TEG thermal resistance ($\Theta_{\text{TEG}}$) is greater than the heat sink thermal resistance, a large $\Delta T$ exists across the TEG. Conversely, if the thermal resistance of the TEG is low compared to the heat sink, then a small $\Delta T$ exists across the TEG. Low $\Delta T$ implies low efficiency, power, and voltage output as seen in Figure 10. For optimal performance, the TEG must be matched thermally to the heat sink in order to generate maximum power and voltage outputs [4].
Figure 10. TEG Efficiency, Power Output, and Heat Flux as a Function of the Ratio between Heat Sink Thermal Resistance and TEG Thermal Resistance [4].

3.4.2 TEG Application to CubeSats

TEGs provide energy harvesting properties through temperature differences and these devices have been used in space applications for powering spacecraft like the Voyager and Pioneer long-range probes as part of their Radioisotope Thermoelectric Generators (RTGs). Waste heat generated by a CubeSat can potentially be utilized for energy harvesting with the current TEG technology. However, many variables such as efficiency of the TEG, cost and mass for integration for the power gain at low ΔT need to be considered before determining the practicality that these devices presently have for energy harvesting applications.

For example, solar flux that strikes any surface in LEO space is about 1400 W/m². This solar flux can be reduced to as much as 30% due to losses in reflection and emission from external surfaces of the CubeSat, leaving about 980 W/m² available incident solar flux for power generation. Approximately 20% of the available incident solar flux striking solar cells is converted into electricity, leaving about 784 W/m² of incident solar flux available for power generation in a TEG. With this in mind, approximated power outputs for different TEG efficiencies, using Equation 3.40, are given in Table 2 for
the available solar heat flux, as well as Table 3 show the power output for the largest collection area for different CubeSat form factors, 10 cm\(^2\), 20 cm\(^2\), and 30 cm\(^2\) for 1U, 2U, and 3U CubeSats, respectively [5].

Table 2. Approximate TEG Power Output.

<table>
<thead>
<tr>
<th>TEG Efficiency</th>
<th>Approximate Power Output (W/m(^2))</th>
</tr>
</thead>
<tbody>
<tr>
<td>3%</td>
<td>23.5</td>
</tr>
<tr>
<td>5%</td>
<td>39.2</td>
</tr>
<tr>
<td>10%</td>
<td>78.5</td>
</tr>
<tr>
<td>15%</td>
<td>118</td>
</tr>
</tbody>
</table>

Table 3. TEG Power Output Based on CubeSat Size and TEG Efficiency.

<table>
<thead>
<tr>
<th>CubeSat Form-Factor</th>
<th>Power Output (W) for (\eta = 3%)</th>
<th>Power Output (W) for (\eta = 5%)</th>
<th>Power Output (W) for (\eta = 10%)</th>
<th>Power Output (W) for (\eta = 15%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1U</td>
<td>0.35</td>
<td>0.39</td>
<td>0.78</td>
<td>1.18</td>
</tr>
<tr>
<td>2U</td>
<td>0.71</td>
<td>0.78</td>
<td>1.57</td>
<td>2.35</td>
</tr>
<tr>
<td>3U</td>
<td>1.06</td>
<td>1.18</td>
<td>2.35</td>
<td>3.53</td>
</tr>
</tbody>
</table>

As shown in Table 2 and Table 3 above, minimal power output generated by TEGs for CubeSats require a TEG efficiency of more than 3%, this is one consideration when trying to implement TEGs into the design of a CubeSat. However, the ability to maintain a good thermal gradient becomes problematic since the CubeSat experiences temperature variations from all incoming environmental heat fluxes throughout the orbit during both light and eclipse phases.
Chapter 4. Preliminary Design Analysis

A spacecraft’s thermal environment is primarily dependent on the orbital parameters and orientation of the spacecraft. Due to uncertainty in these parameters, assumptions of worst-case orbital conditions for OSIRIS-3U must be identified in order to determine the temperatures limits that the CubeSat could reach during its mission. Temperature limits can be calculated and verified through analytical and computer simulated models, which can then serve to determine an effective thermal control system to maintain operating temperature limits for all components of OSIRIS-3U. This chapter presents the thermal requirements, assumptions, and methods for software modeling implemented for OSIRIS-3U. Lastly, a preliminary analysis of thermoelectric generators for OSIRIS-3U is discussed and two commercial TEGs are presented for analytical study of generated power output at low ΔT.

4.1 Thermal Requirements

As discussed previously, the goal of a thermal control system is to maintain temperatures of all spacecraft components within allowable operational ranges during all mission phases. The current thermal design for OSIRIS-3U consists of automatic control circuitry connected to thermistors and heaters integrated on the circuit boards containing critical components that require operation over a limited temperature range. OSIRIS-3U critical components are outlined in

Table 4.
Table 4. OSIRIS-3U Thermal Requirements of Critical Components.

<table>
<thead>
<tr>
<th>System</th>
<th>Components</th>
<th>Operating $T_{\text{min}}$(°C)</th>
<th>Operating $T_{\text{max}}$(°C)</th>
</tr>
</thead>
<tbody>
<tr>
<td>CDH</td>
<td>Microcontroller</td>
<td>−40</td>
<td>+85</td>
</tr>
<tr>
<td></td>
<td>NAND</td>
<td>−40</td>
<td>+85</td>
</tr>
<tr>
<td></td>
<td>SDRAM</td>
<td>−40</td>
<td>+85</td>
</tr>
<tr>
<td>COM</td>
<td>Beacon Power Amp</td>
<td>−65</td>
<td>+150</td>
</tr>
<tr>
<td></td>
<td>FPGA</td>
<td>−40</td>
<td>+85</td>
</tr>
<tr>
<td></td>
<td>Main Power AMP</td>
<td>−40</td>
<td>+85</td>
</tr>
<tr>
<td>GNC</td>
<td>GPS Module</td>
<td>−40</td>
<td>+85</td>
</tr>
<tr>
<td></td>
<td>Sun Sensors</td>
<td>−20</td>
<td>+80</td>
</tr>
<tr>
<td>PWR</td>
<td>LiFePO$_4$ batteries</td>
<td>−30</td>
<td>+60</td>
</tr>
<tr>
<td></td>
<td>Regulators (12 V)</td>
<td>−40</td>
<td>+125</td>
</tr>
</tbody>
</table>

The thermal subsystem is required to develop finite element analyses of the thermal environment expected by OSIRIS-3U during the mission. The results obtained from FEA models should serve as a basis for determining optimal placement of heaters and thermal sensors. The following sections summarize the development of FEA models for worst-case orbital conditions of OSIRIS-3U in COMSOL Multiphysics.

4.2 Assumptions

The thermal control system has been developed and analyzed for the OSIRIS-3U CubeSat in order to ensure that the spacecraft and its components will operate within the allowable temperature ranges. These allowable temperature ranges are based upon worst-case-hot and worst-case-cold temperatures that the CubeSat will experience during the mission. Due to the uncertainty in orbital parameters, the CubeSat has been assumed to be in a circular Earth orbit with altitudes between 300 km and 400 km. Furthermore, the 3U CubeSat has been oriented such that the face with the smallest area is
always pointed towards Earth (−Z). The −Y face always points in the direction of travel as seen in Figure 11.

![Figure 11. CubeSat Orientation with Respect to Earth.](image)

Table 5 provides values for environmental parameters that are taken into consideration when calculating the temperature of the CubeSat. The maximum heat flux values correspond to summer solstice, and minimum heat flux values correspond to winter solstice. In addition, constant parameters are listed below.

**Constant parameters:**

\[
R_e = 6371 \text{ km (radius of Earth)}
\]

\[
A_{sc} = 0.14 \text{ m}^2 \text{ (total area of CubeSat)}
\]

\[
\sigma = 5.67 \times 10^{-8} \frac{\text{W}}{\text{m}^2\text{K}} \text{ (Boltzmann constant)}
\]

<table>
<thead>
<tr>
<th></th>
<th>Internal Power (W)</th>
<th>Solar Flux (W/m²)</th>
<th>Albedo</th>
<th>Earth IR (W/m²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximum</td>
<td>3</td>
<td>1400</td>
<td>0.35</td>
<td>270</td>
</tr>
<tr>
<td>Minimum</td>
<td>0</td>
<td>1300</td>
<td>0.3</td>
<td>220</td>
</tr>
</tbody>
</table>

The view factors for these models were determined by making use of Equations 3.29 and 3.30 for parallel and perpendicular flat plate orientation with respect to Earth. These values were also checked for
consistency with the Earth view factor from Figure 7. Table 6 provides the values of view factors for each altitude.

Table 6. Calculated View Factors for the CubeSat at Different Orbital Altitudes.

<table>
<thead>
<tr>
<th>Altitude (km)</th>
<th>Calculated VF $\gamma = 0^\circ$</th>
<th>Estimated VF Figure 7 $\gamma = 0^\circ$</th>
<th>Calculated VF $\gamma = 90^\circ$</th>
<th>Estimated VF Figure 7 $\gamma = 90^\circ$</th>
</tr>
</thead>
<tbody>
<tr>
<td>300</td>
<td>0.91</td>
<td>0.89</td>
<td>0.31</td>
<td>0.28</td>
</tr>
<tr>
<td>400</td>
<td>0.88</td>
<td>0.90</td>
<td>0.29</td>
<td>0.28</td>
</tr>
</tbody>
</table>

4.3 Worst-Case Hot

Assuming an isothermal spacecraft, the hottest temperature a spacecraft will encounter in orbit will be at full sunlight phases, where solar, albedo, and Earth IR radiation are present. Additionally, the closer the spacecraft is to Earth, the larger Earth IR and albedo heat fluxes it will receive. Thus, the WCH scenario for OSIRIS-3U will correspond to an altitude of 300 km, with the maximum heat flux values given in Table 5. For this model the corresponding inclination is $i = 90^\circ$, where the CubeSat experiences no eclipse as seen from Figure 12.
Furthermore, as seen from Figure 11, the orientation of OSIRIS-3U with respect to Earth is such that $-Z$ is always facing Earth and $-Y$ is a face perpendicular to the velocity vector. Therefore, face $-X$ constantly experiences solar thermal inputs, $\pm Y$ and $-Z$ only experience albedo and Earth IR thermal inputs, as well as face $-Z$. The $+Z$ face is only exposed to deep space radiation such as star and moon shine, which will be neglected in these analyses.

### 4.4 Worst-Case Cold

The coldest temperature a spacecraft will encounter in orbit will be at eclipse phases, where only Earth IR radiation is present. Additionally, the farther away from Earth the spacecraft is, the less Earth IR radiation it will receive. For the WCC scenario, OSIRIS-3U is assumed to be at an altitude of 400 km with minimum heat input values from Table 5. Also, the orbit is at an inclination of $i = 0^\circ$; the orbital design for WCC case is illustrated in Figure 13.

![Diagram](image)

Figure 13. Worst-Case-Cold Orbit.
For this configuration OSIRIS-3U will receive only Earth IR and albedo on faces $-Z$, $\pm Y$, and $\pm Z$. Face $+X$ will be exposed only to sunlight as well as $\pm Y$ faces only during sun-illuminated orbital angles. Figure 14 shows the CubeSat with illuminated solar panels during a complete orbital period.

![Figure 14. Worst-Case-Cold Orbit with Illuminated Solar Panels.](image)

The amount of solar heat flux that strikes any surface depends on the projected area, since different projected areas are exposed to the solar heat flux throughout the orbit. In order to account for these changes one needs to determine a $\beta$ angle, which is the angle between the normal vector of the side of interest with the solar vector. The orbital angle ($\theta$) is used to determine the $\beta$ angle for faces that are exposed to solar heat inputs, which are shown Table 7.

<table>
<thead>
<tr>
<th>Face of Interest</th>
<th>Beta Angle ($\beta$)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$-X$</td>
<td>$\theta$</td>
</tr>
<tr>
<td>$+Y$</td>
<td>$\theta + 90^\circ$</td>
</tr>
<tr>
<td>$-Y$</td>
<td>$\theta - 90^\circ$</td>
</tr>
</tbody>
</table>
For WCC temperatures, the CubeSat will experience eclipse phases. Therefore, orbital angles at which the spacecraft enters and leaves Earth’s shadow (eclipse) need to be determined by calculating the angle beta $\beta_e$ from Figure 15:

$$\beta_e = \cos^{-1}\left(\frac{R_e}{R_e + z}\right),$$

where $R_e$ is the radius of Earth and $z$ the altitude of the orbit. Table 8 gives the estimated values of the angles the CubeSat enters and leaves eclipse.

![Figure 15. Determining Earth Beta Angle for Entering and Leaving Eclipse Phases.](image)

### Table 8. Orbital Angle Determination at which the CubeSat Enters and Leaves Eclipse.

<table>
<thead>
<tr>
<th>Altitude (km)</th>
<th>$\theta_{\text{enter}}$</th>
<th>$\theta_{\text{exit}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>400</td>
<td>109.8</td>
<td>250.2</td>
</tr>
</tbody>
</table>

### 4.5 Geometric Mathematical Model in MATLAB

For steady-state analysis, an analytical Geometric Mathematical Model (GMM) using a single-node approximation was developed in MATLAB (code provided in Appendices B & C) to determine the
thermal heat inputs that the CubeSat experiences for WCH and WCC scenarios. This model provides an average of the thermal heat inputs for CubeSats in LEO dependent on the spacecraft geometry and orbital parameters such as altitude, orbital position, and attitude. For better approximation of the total thermal heat input to a CubeSat, coefficients corresponding to emission and absorption of the material used in the CubeSat design must be accounted for.

4.6 Software Modeling of OSIRIS-3U CubeSat

A time-dependent finite-element model of the temperature variations of OSIRIS-3U was created in COMSOL Multiphysics using the Heat Transfer module. The model included all parameters stated in Section 4.2 Assumptions. In COMSOL Multiphysics, the OSIRIS-3U structure was constructed comprised of 6061–T6 aluminum rails, FR4 solar panels, gallium-arsenide (GaAs) solar cells, FR4 circuit boards, and batteries as seen in Figure 16, the highlighted structures in each sub-image represent the different materials. Table 9 summarizes the thermal properties for all OSIRIS-3U components incorporated to the model. The batteries were assumed to be steel as the exact materials are viewed as by the manufacturer.
Figure 16. Exterior and Interior Structure of OSIRIS-3U in COMSOL Multiphysics, (a) Aluminum, (b) Solar panels, (c) Solar Cells, (d) Batteries and (e) Circuit boards.

Table 9. List of Material Thermal Properties for OSIRIS-3U.

<table>
<thead>
<tr>
<th>Material</th>
<th>Emissivity $\epsilon$</th>
<th>Absorptivity $\alpha$</th>
<th>Density kg/m$^2$</th>
<th>Thermal conductivity W/(mK)</th>
<th>Heat Capacity J/(kgK)</th>
</tr>
</thead>
<tbody>
<tr>
<td>FR4 Fiberglass</td>
<td>0.89</td>
<td>0.90</td>
<td>1900</td>
<td>0.30</td>
<td>1369</td>
</tr>
<tr>
<td>Solar Cells</td>
<td>0.85</td>
<td>0.90</td>
<td>5307</td>
<td>20</td>
<td>325</td>
</tr>
<tr>
<td>Aluminum 6061–T6</td>
<td>0.055</td>
<td>0.38</td>
<td>2700</td>
<td>201</td>
<td>900</td>
</tr>
<tr>
<td>Battery (Steel)</td>
<td>0.07</td>
<td>0.40</td>
<td>7850</td>
<td>44</td>
<td>475</td>
</tr>
</tbody>
</table>

In order to account for the CubeSat precession in orbit and different panels being exposed to sunlight at different orbital angles, a Heaviside step function, dependent on time, was incorporated in the
model. This step function, shown in Figure 17, which represents a value of zero and one for time-dependent solutions, acts as a switch to turn on solar and albedo heat fluxes on the panel of interest during sunlight periods, or turn it off during eclipse period. The period of one orbit at 300 km and 400 km is approximately 90.5 minutes and 92.5 minutes, respectively. Lastly, for both WCH and WCC cases, 20 orbital periods were simulated for the time-dependent solver.

![Figure 17. Heavyside Step Function in COMSOL Multiphysics as a function of time (seconds).](image)

### 4.7 Preliminary Analysis of Thermoelectric Generators for OSIRIS-3U

It is assumed that the heat source for the TEG is provided by the back side of a solar cell enabling the TEG, connected by vias, to provide a good thermal contact with the solar cell to drive sufficient thermal gradient across the device (note that mounting the hot side of the TEG through the FR4 that will be between the back of a solar cell and TEG will not effectively drive high thermal power since FR4 has a low thermal conductivity). A mathematical model has been developed in MATLAB using the information in Section 3.4.1 Thermoelectric Generation Theory to determine the power output of different commercial TEGs with an average $\Delta T$ achieved during sunlit periods of a 20-orbital-period simulation of OSIRIS-3U in COMSOL. An average temperature of the interior components is determined from simulations and set as the cold temperature $T_c$ of the TEG. An average heat source
temperature for a given solar panel is obtained from FEA models and set as $T_h$. $\Delta T$ is then computed and an average is obtained.

Due to small temperature differences ($\Delta T$), thermoelectric material properties are assumed to be constant. Typical material properties of TEGs provided by each manufacturer were used for these calculations and are given in Table 10. The first commercial traditional bulk TEG is from Custom Thermoelectric and the second is a thin film from Nextreme Thermal Solutions. Solutions of calculated TEG efficiencies and power output for different dimensions of semiconductors and number of thermocouples are developed and analyzed.

**Table 10. Material Properties for Commercial TEGs.**

<table>
<thead>
<tr>
<th>Material Properties</th>
<th>Traditional Bulk (Custom TE)</th>
<th>Thin Film Superlattice (Nextreme)</th>
</tr>
</thead>
<tbody>
<tr>
<td>TE element Height, mm</td>
<td>2.0</td>
<td>0.07</td>
</tr>
<tr>
<td>TE element width, mm</td>
<td>1.0</td>
<td>0.10</td>
</tr>
<tr>
<td>Device $ZT$</td>
<td>0.73</td>
<td>0.36</td>
</tr>
<tr>
<td>Thermal Conductivity, W/(mK)</td>
<td>1.4</td>
<td>0.75</td>
</tr>
<tr>
<td>Avg. Seebeck Coeff, $\mu$V/K</td>
<td>220</td>
<td>293</td>
</tr>
<tr>
<td>N# couples</td>
<td>254</td>
<td>144</td>
</tr>
</tbody>
</table>
Chapter 5. Results

Thermal analysis for OSIRIS-3U requires calculation and verification of temperature limits reached with the assumptions discussed in Chapter 4. This chapter presents results of the steady-state temperatures and verification through FEA models developed in COMSOL Multiphysics for worst-case orbital conditions. For FEA models, the maximum and minimum temperature limits are presented for 20 orbital periods of OSIRIS-3U. Additionally, results of the analytical solution of the efficiency and output power generation of two commercial TEGs is presented by using low temperature gradients achieved during sun-lit periods of the FEA models of OSIRIS-3U.

5.1 Theoretical Steady-State Temperatures of OSIRIS-3U CubeSat

A thermal energy balance was performed to determine the steady-state temperatures of OSIRIS-3U for WCH and WCC cases; these results reference section 3.2.4 Thermal Balance. The thermal energy balance incorporates the material thermal properties given in Table 9 and assumes that 50% of each solar panel is covered by solar cells; Table 11 summarizes the surface areas of the different materials exposed to all incoming heat fluxes. Additionally, for steady-state solutions the hot case assumes the CubeSat is placed in sunlight forever, and the cold case assumes the CubeSat to be placed in shadow forever.

<table>
<thead>
<tr>
<th>Material</th>
<th>Surface Area cm²</th>
</tr>
</thead>
<tbody>
<tr>
<td>Smallest FR4 Fiberglass</td>
<td>100</td>
</tr>
<tr>
<td>Longest FR4 Fiberglass</td>
<td>150</td>
</tr>
<tr>
<td>Solar Cells</td>
<td>150</td>
</tr>
<tr>
<td>Aluminum 6061–T6</td>
<td>30</td>
</tr>
</tbody>
</table>

Table 11. Exposed Surface Area of OSIRIS-3U for Different Materials.

Hot Case:

- Altitude: 300 km
- Incoming heat flux: solar, albedo, and Earth IR
- Internal power dissipation: 3 W

Using the steady-state equations in Section 3.2, Thermal Environment of Spacecraft to solve for solar, albedo, and Earth IR heat fluxes for each material, the total power absorbed is

\[ Q_{\text{total}} = 100 \text{ W}, \]

with an average calculated value of emissivity for all materials, \( \epsilon = 0.57 \), and the total area of the CubeSat, \( A_{\text{tot}} = 0.14 \text{ m}^2 \), the temperature is calculated to be

\[ Q_{\text{total}} = \epsilon \sigma T^4 A_{\text{tot}}, \]
\[ T = 113 \, ^{\circ}\text{C} \]

**Cold Case:**
- Altitude: 400 km
- Incoming heat flux: Earth IR

Applying the same methods from the hot case, the total power absorbed is

\[ Q_{\text{total}} = 10 \text{ W} \]

and the temperature for worst-case-cold case is

\[ T = -56 \, ^{\circ}\text{C}. \]

Overall, the steady-state temperatures for WCH and WCC cases give an initial maximum and minimum temperature limits that OSIRIS-3U can experience. For a transient or time-dependent model, the temperature limits will be expected to lie between these calculated theoretical values, \( T_{\text{max}} = 113 \, ^{\circ}\text{C} \) and \( T_{\text{min}} = -56 \, ^{\circ}\text{C} \). A more detailed analysis of these calculations is provided in [6].

### 5.2 Geometric Mathematical Model

As discussed in Section 4.5, Geometric Mathematical Model in, an average thermal heat inputs for WCH and WCC cases have been calculated using a GMM for OSIRIS-3U. The results for the WCH case are shown in Figure 18. As seen in this figure, it is expected for the thermal heat inputs to be constant over all orbital angles since the CubeSat in a worst-case-hot orbit is never exposed to eclipse
phases. The maximum heat input is received by panel \(-X\) because this panel is constantly facing the sun at \(i = 90^\circ\), and it also receives constant Earth IR and albedo. Additionally, the difference in heat input received by panels \(-Z\) and \(+X\) is caused by the change in surface area and view factors. Panels \(\pm Y\) receive constant heat inputs because they only receive constant Earth IR and albedo. Panel \(+Z\), which always faces deep space, does not experience any thermal heat inputs, consequently the plot shows zero heat input for panel \(+Z\).

![Thermal Heat Inputs of OSIRIS 3U](image)

**Figure 18. Average Thermal Heat Inputs for Worst-Case-Hot Case.**

Similarly, a GMM for WCC was generated to determine the average thermal heat inputs the CubeSat experiences. These results are show in Figure 19, where the top plot shows heat inputs for panels that do not receive solar heat inputs. Panels \(\pm X\) receive the same amount of heat flux throughout the orbit since they only experience constant Earth IR and albedo. Also, the reason for the small
The difference between panels $\pm X$ and $-Z$ is due to the different view factors between panels, since $-Z$ is the Earth-facing panel the view factor is higher than panels $\pm X$.

**Figure 19. Average Thermal Heat Inputs for Worst-Case-Cold Case.**

The bottom plot from Figure 19 illustrates the heat inputs for all other CubeSat panels that, in addition to Earth IR and albedo, also experience solar heat inputs. Panels $\pm Y$ experience greater heat inputs compared to all other panels due to larger surface area and greater time exposed to solar heat inputs. Furthermore, during sunlight periods half of the time panel $+Y$ experiences solar flux and the other half of the time the opposite panel, $-Y$, experiences solar flux. The minimum heat inputs that the CubeSat experiences (reference Figure 19) is during eclipse phases, where Earth is assumed to be at a constant temperature; therefore, a constant heat input close to 2 W is present on each panel except panel $+Z$ because it is facing away from Earth as it enters Earth’s shadow.
5.3 Transient Model of OSIRIS-3U

As described in Section 4.6, a finite element model of OSIRIS-3U was created in COMSOL Multiphysics to obtain temperature variations over 20 orbital periods. The results for WCH and WCC average temperatures for OSIRIS-3U are summarized in the following sections.

5.3.1 Worst Case Hot

The WCH scenario corresponds to all parameters both spacecraft and environmental that will cause OSIRIS-3U to achieve maximum temperatures. For this model, an average surface temperature over 20 orbital periods for each panel was extracted from COMSOL, as well as a volumetric average temperature of the interior components. These temperature evolutions can be seen in Figure 20 and Figure 21; the maximum and minimum temperatures have been described in Table 12 and Table 13.

Table 12. WCH Average Max and Min Temperature Values for Panels of OSIRIS-3U.

<table>
<thead>
<tr>
<th></th>
<th>Panel +X</th>
<th>Panel −X</th>
<th>Panel +Y</th>
<th>Panel −Y</th>
<th>Panel +Z</th>
<th>Panel −Z</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max ºC</td>
<td>43</td>
<td>110</td>
<td>52</td>
<td>53</td>
<td>43</td>
<td>63</td>
</tr>
<tr>
<td>Min ºC</td>
<td>−1.1</td>
<td>7.6</td>
<td>−0.86</td>
<td>−0.79</td>
<td>−4.1</td>
<td>0.51</td>
</tr>
</tbody>
</table>

Table 13. WCH Average Max and Min Temperature Values of Interior Components of OSIRIS-3U.

<table>
<thead>
<tr>
<th></th>
<th>COM</th>
<th>GNC</th>
<th>PWR</th>
<th>BATT</th>
<th>SCI</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max ºC</td>
<td>62</td>
<td>68</td>
<td>71</td>
<td>72</td>
<td>69</td>
</tr>
<tr>
<td>Min ºC</td>
<td>−0.036</td>
<td>−0.033</td>
<td>−0.028</td>
<td>0.022</td>
<td>−0.027</td>
</tr>
</tbody>
</table>
As seen in Figure 20 and Figure 21, the temperatures of the CubeSat reach steady state at around 16 hours. Overall, the maximum temperature that the CubeSat experience is 110 °C by panel $-X$ and the minimum temperature is $-4.1$ °C by panel $+Z$. These results are as expected since, as stated previously for WCH, the CubeSat is always illuminated with panel $-X$ receiving all solar radiation and Earth IR and albedo. Furthermore, panel $+Z$ always faces deep space resulting in the overall lowest temperature of the CubeSat with the temperature being influenced by means of conduction within the CubeSat.
For interior components as seen from Figure 21, the minimum temperature is approximately 
\(-0.03 \, ^\circ\text{C}\) for all circuit boards including the battery, indicating no concern for minimum operating 
temperature limits as described in Table 4. However, the maximum temperature of circuit boards and battery reach up to +72 \, ^\circ\text{C},
which is of concern for the battery due to the maximum operating temperature limit of 60 \, ^\circ\text{C}
from Table 4. Lastly, the results obtained for worst case hot of OSIRIS-3U show overall temperature
limits between +110 \, ^\circ\text{C} and \(-4.4 \, ^\circ\text{C}\), which suggest no concerns for minimum allowable operating
temperature of internal components, but definitely a number of concerns for maximum operating
temperatures, especially the battery, which is a very critical component for all CubeSats.

5.3.2 Worst-Case-Hot OSIRIS-3U Temperature Profiles

Temperature profiles of OSIRIS-3U modeled in COMSOL Multiphysics for WCH case were
obtained for different time periods of the transient solution and are illustrated below. Figure 22 shows a
surface and sliced temperature profile of OSIRIS-3U at 3700 seconds. Looking at the surface temperature
profile, the model shows that panel \(-X\) is the hottest panel, as expected, since it is constantly receiving
solar, albedo, and Earth IR radiation. Panel \(+Z\) indicates the lowest temperature since it does not receive
any radiation besides radiation from deep space that is so small and negligible. The sliced temperature
profile shows that the heat is being radiated across the interior of OSIRIS-3U by means of surface to
surface radiation moving from higher temperatures to decreasing temperatures across the \(X\) axis. Overall,
the interior surfaces show temperature limits that fall in between the exterior surface temperature limits,
as expected.
Figure 22. Surface (Left) and Slice (Right) COMSOL temperature profile of OSIRIS-3U at 3700s.

Figure 23 illustrates the surface and slice temperature profile of OSIRIS-3U near the end of the 20-orbital-period simulations. As it is evident, the overall temperature limits are higher compared to Figure 22 since enough time has elapsed and a steady-state temperature has been reached in the simulation. Once again, panel –X remains at a higher temperature compared to all other panels as expected. The sliced profile shows a significant increase in heat being radiated across the X axis, and its temperature limits fall in between the temperature limits of the surface temperature, as anticipated.
5.3.3 Worst Case Cold

The WCC case corresponds to all parameters that will influence OSIRIS-3U to obtain the minimal temperature variations over 20 orbital periods. For this model, the CubeSat experience eclipse phases and assumes all parameters described in Section 4.4 Worst-Case Cold; the average temperature evolutions are shown in Figure 24 and Figure 25. Minimum and maximum temperatures are given in Table 14 and Table 15.

Table 14. WCC Average Max and Min Temperature Values for Panels of OSIRIS-3U.

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Min °C</td>
<td>$-10$</td>
<td>$-10$</td>
<td>$-8.0$</td>
<td>$-11$</td>
<td>$-12$</td>
<td>$-6.5$</td>
</tr>
<tr>
<td>Max °C</td>
<td>$-0.86$</td>
<td>$-0.86$</td>
<td>$29$</td>
<td>$24$</td>
<td>$25$</td>
<td>$9.9$</td>
</tr>
</tbody>
</table>

Table 15. WCC Average Max and Min Temperature Values for Interior Components of OSIRIS-3U.
<table>
<thead>
<tr>
<th></th>
<th>COM</th>
<th>GNC</th>
<th>PWR</th>
<th>BATT</th>
<th>SCI</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Min °C</strong></td>
<td>−4.0</td>
<td>−3.4</td>
<td>−3.3</td>
<td>−2.4</td>
<td>−3.9</td>
</tr>
<tr>
<td><strong>Max °C</strong></td>
<td>7.6</td>
<td>2.6</td>
<td>2.2</td>
<td>3.0</td>
<td>3.7</td>
</tr>
</tbody>
</table>
Figure 24. WCC Average Temperatures for OSIRIS-3U.
Figure 25. WCC Average Temperatures of Interior Components of OSIRIS-3U.
Assuming the CubeSat starts orbiting at the subsolar point (point closest to the Sun) with the satellite orientation described in Section 4.4 Worst-Case Cold, then the results in Figure 24 effectively show the variations between sunlight and eclipse periods for which the rise in temperatures of the panels corresponds to the direct solar heating of panels exposed to the sun in addition to IR and albedo radiation. The drop in temperatures corresponds to two factors. The first factor is that, while the +Y panel receives direct solar radiation, the −Y panel does not for half of the time in sunlight periods and vise versa; thus, one panel increases in temperature while the other drops, which is clearly illustrated in the results in Figure 24. The second and major factor in the decreases in temperature is the influence of eclipse where only Earth IR radiation is present.

The overall maximum temperature achieved by the CubeSat is +29 °C by panel −Y, which is expected since it is the panel with the largest surface area that starts receiving a combination of direct solar, Earth IR, and albedo radiation. The minimum temperature is −12 °C by panel +Z, which is caused by the fact that, once the CubeSat enters the Earth’s shadow, panel +Z points directly away from Earth, thereby only being influenced by conductive heat transfer within OSIRIS-3U. Results for panels ±X show same temperature variations, which is also expected since they receive the same amount of Earth IR and albedo radiation and both have the same surface area and view factors.

Figure 25 illustrates results of the temperature variations of the four interior circuit boards and battery. It should be noted that in COMSOL Multiphysics the heating of internal components occurs by surface-to-surface radiation. Therefore, since the solar panels provide the heat to internal components, the closer the boards are to these surfaces the greater amount of thermal heat they will receive. As a result, since the COM board is oriented closest and parallel to panel +Y and SCI is closest and parallel to panel −Y, they both display the highest temperature values compared to all other internal components. Lastly, the results for WCC case has shown temperature limits between +29 °C and −12 °C, which will require little to no thermal control since these ranges fall between allowable operating temperature ranges of components characterized in
Table 4.

5.3.4 Worst-Case-Cold OSIRIS-3U Temperature Profiles

Temperature profiles of OSIRIS-3U modeled in COMSOL Multiphysics for the WCC were obtained for different time periods of the transient solution and are illustrated below. Figure 26 illustrates the surface and sliced temperature profile of OSIRIS-3U after the first hour of one orbital period. The temperature limits are low as expected since at time 3600 seconds the CubeSat is expected to be in the Earth’s shadow. Panel +Y experiences the highest temperature because, from the start of the simulation, this panel starts receiving solar, albedo, and Earth IR radiation; by the time it enters eclipse this panel is the hottest panel compared to all other panels. Panel –Z also experiences the highest temperature due to direct Earth IR radiation being absorbed since it is the nadir-facing surface. Lastly, the interior of the CubeSat demonstrates the heat being radiated across the Y axis with the highest temperature values near the +Y panel and lowest near –Y panel, as expected.

![Figure 26. Surface (Left) and Slice (Right) COMSOL Temperature Profile of OSIRIS-3U at 1 Hour.](image)
Figure 30 shows another surface and slice temperature profile of OSIRIS-3U after ten orbital periods. The temperature limits are higher compared to Figure 26 since at this time the CubeSat is in the sunlit region. Once again, the temperature profile of the exterior surface and the interior are as expected since the sun-side panels $+Z$ and $-Y$ receive direct solar radiation. The coldest panels are $\pm X$ and $-Z$ for the reason that they only experience albedo and Earth IR radiation.

![Figure 27. Surface (Left) and Slice (Right) COMSOL Temperature Profile of OSIRIS-3U at 15 Hours.](image)

Figure 28 shows the surface and slice temperature profile of OSIRIS-3U at the end of 20 orbital periods. At this time interval the CubeSat is expected to have reached steady state, which is shown in the plots of the temperature evolution of total simulated time in Section 5.3.3 Worst Case Cold. The sliced profile shows the interior distribution of heat across all axes with the highest temperature near the solar panels with high temperature values and low internal temperature near the coldest solar panels.
5.4 TEG Analytical Results

Two commercial thermoelectric modules have been analyzed to determine the practicality of these devices for energy harvesting on CubeSats. Results obtained from the time-dependent solver of the temperature variations of OSIRIS-3U have been used to determine an average temperature gradient necessary for these TEGs to harvest energy. The average temperature gradient was obtained during sunlight periods of the COMSOL simulations discussed in Section 5.3.3 Worst Case Cold; these COMSOL results provided an average internal temperature of 275 K, which was set as an initial estimate of the cold temperature, $T_C$, for the TEG. Initial analyses was performed using the temperature variation of panel $+Y$ since it experiences the hottest temperature throughout the sunlit periods, the higher the temperature the greater temperature gradient that can exist between the solar panel providing the heat and the internal surfaces of OSIRIS-3U. With this in mind, the average surface temperature of panel $+Y$ was obtained and set as an initial hot temperature, $T_h$. The minimum, maximum, and average temperature
gradient between $T_h$ and $T_c$ were calculated for the 20 orbital period simulations and provided in Table 16.

Table 16. Temperature Gradient Obtained from COMSOL Simulation of 400-km Orbit.

<table>
<thead>
<tr>
<th>$\Delta T_{\text{min}}$ (K)</th>
<th>$\Delta T_{\text{max}}$ (K)</th>
<th>$\Delta T_{\text{avg}}$ (K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.0034</td>
<td>28</td>
<td>13</td>
</tr>
</tbody>
</table>

For $\Delta T_{\text{avg}}$, MATLAB was used to determine the power outputs generated by the two TEGs. With the information specified in Table 10, the power output as a function of load resistance, $R_{\text{load}}$, was calculated and results are presented in Figure 29.

![Graph showing calculated output power vs. load resistance for two TEGs](image-url)
As expected, the maximum power output for both thermoelectric modules occurs at matched load conditions, i.e., when the internal resistance of the TEG is equal to the load resistance. The traditional bulk thermoelectric device from Custom Thermoelectric gives the highest power output; this is because of the low internal resistance produced by the small length-to-TE-area ratio of the TE elements. The thin-film TEG from Nextreme delivers significantly lower power output compared to the traditional bulk TEG.

In terms of load resistance, higher output power is achieved with lower values of load resistance in the traditional bulk TEG compared to the thin film TEGs.

Figure 30 shows the output power per unit area of the TEGs for different load resistance with the same temperature gradient as $\Delta T_{\text{avg}}$. The results obtained show that thin-film TEG delivers significantly higher output power densities compared to the traditional bulk TEG.

![Figure 30. Output Power per Unit Area of TEG Device as a Function of Load Resistance.](image)
Maximum power outputs are achieved in TEGs when the load resistance matches the internal resistance. Figure 31 shows the maximum power output per unit area of both TEGs for a range of temperature differences, $\Delta T$. The relationship between the maximum power output per unit area and temperature difference is parabolic. It is clear that the output power increases with increasing temperature gradient, and that the smaller thermoelectric devices will deliver greater power output per unit area of the module.

![Graph showing maximum power output per unit area of TEG vs. increasing temperature gradient.]

**Figure 31. Maximum Power Output per Unit Area of TEG vs. Increasing Temperature Gradient.**

To evaluate the above thermoelectric results, Figure 32 is a Nextreme-provided performance comparison of various TEG technologies via a plot of the power density per unit area of TEG as a function of the output voltage per watt flowing through the device. From all of these results discussed it is evident that thin-film TEG modules deliver higher power densities compared to the traditional TEGs. Additionally, for these devices the efficiency of generating power is greatly influenced by the figure of
merit ($ZT$) of the material and the temperature gradient. With the minimum and maximum temperature gradients obtained from COMSOL simulations and summarized in Table 16, the efficiencies of both devices were calculated and are plotted in Figure 33. As expected, the TEG efficiency increases with increasing temperature differences and $ZT$ values. One important thing to notice is that the low $\Delta T$ produces TEG efficiencies lower than 1%.

Figure 32. Performance Comparison of Various TEG Technologies [23].
Figure 33. Calculated TEGs Efficiency vs. Temperature Differences.
Chapter 6. Discussion of Results

6.1 Discussion of Thermal Results and Thermal Design for OSIRIS-3U

A successful model of OSIRIS-3U for temperature determination of worst-case orbital conditions was created. The analytical solutions yielded steady-state temperature limits between +113 °C and −56 °C; for verification of these calculations it was expected that for time-dependent solutions the temperature limits for WCH and WCC cases will fall well within the predicted values. As discussed in Chapter 5, the time-dependent solution did result in minimum temperatures well above that calculated and for maximum temperature a little less than calculated.

The time-dependent analyses indicated that for the WCC orbit OSIRIS-3U will experience temperature variations between +29 °C and −12 °C. These temperature limits fall within acceptable operating temperature limits of major components of OSIRIS-3U, it was concluded that it will be beneficial to have some thermal control of the sun sensors since their minimum operating temperature is −20 °C.

For the WCH orbit, the time-dependent solution indicated that OSIRIS-3U will experience temperature ranges between +109 °C and −4.1 °C. The minimum temperature limit falls between acceptable minimum operating temperatures of OSIRIS-3U components. However, the maximum temperature limit falls outside of the range of maximum operating temperature of major components, most critically the batteries, but due to mission requirements of OSIRIS-3U it is necessary to use ground stations that are available in the northern hemisphere imposing a constraint in the maximum orbital inclination of \(i = 70°\). Thus, the temperature limits will fall well below the simulated results of a polar orbit (\(i = 90°\)).

The thermal results obtained from analytical calculations as well as a time-dependent FEA simulated models served as a basis for determining a thermal design of OSIRIS-3U. The thermal design
was driven by the limited mass, power budget, and board space available for thermal control of OSIRIS-3U. With all of these factors, the Thermal Subsystem determined that an Independently Operating Thermal Control will effectively regulate the temperature of OSIRIS-3U. This type of thermal control will read temperature data and issue heater commands by controllers operating independently from the flight computer. The down side to this method is the implementation of separate controllers, which can take up precious board space. However, in order to avoid this disadvantage, a trade study was performed to determine the type of hardware that will minimize board space and circuitry of the thermal control system.

The trade study performed on the type of temperature sensor suitable for OSIRIS-3U concluded that the two highest scoring sensors that comply with primarily mass, power, and space constraints were the US Sensor Thermistor and the TMP122 (Appendix D). The TMP122 ended up being substituted with a TMP175 since it contained communication format incompatibilities with the flight computer. Therefore, all subsystems will use the thermistors except for the sun sensor, which will use a TMP175 due to the lack of board space. In order to regulate the temperature and heaters placed in OSIRIS-3U, an ADC128 analog-to-digital converter connected to the thermal controller was chosen. Table 17 lists the placement of hardware chosen for the thermal control of OSIRIS-3U.

<table>
<thead>
<tr>
<th>System</th>
<th>Components</th>
<th>Thermistors</th>
<th>TMP175</th>
<th>Heaters</th>
</tr>
</thead>
<tbody>
<tr>
<td>CDH</td>
<td>Microcontroller</td>
<td>1</td>
<td>0</td>
<td>1</td>
</tr>
<tr>
<td></td>
<td>NAND</td>
<td>1</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td></td>
<td>SDRAM</td>
<td>2</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td>COMM</td>
<td>Power Amp</td>
<td>1</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td></td>
<td>FPGA</td>
<td>1</td>
<td>0</td>
<td>1</td>
</tr>
<tr>
<td></td>
<td>Main Power AMP</td>
<td>1</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>GNC</td>
<td>GPS Module</td>
<td>1</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td></td>
<td>Sun Sensors</td>
<td>0</td>
<td>6</td>
<td>6</td>
</tr>
<tr>
<td>PWR</td>
<td>LiFePO₄ batteries</td>
<td>2</td>
<td>0</td>
<td>2</td>
</tr>
<tr>
<td></td>
<td>Regulators (12V)</td>
<td>4</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td></td>
<td><strong>Total</strong></td>
<td><strong>20</strong></td>
<td><strong>6</strong></td>
<td><strong>10</strong></td>
</tr>
</tbody>
</table>
The total mass of hardware components that contributes to the overall thermal design of OSIRIS-3U is estimated to be about 1% of the 4 kg total mass of OSIRIS-3U (without consideration of additional batteries needed); this is summarized in Table 18. Lastly, with the thermal control design chosen, the total power consumption (not including heaters) is estimated to be about 0.13 W; this is summarized in Table 19.

Table 18. Estimated Total Mass of the Thermal Control System.

<table>
<thead>
<tr>
<th>Component</th>
<th>Number of Components</th>
<th>Mass per Unit (g)</th>
</tr>
</thead>
<tbody>
<tr>
<td>ADC128</td>
<td>4</td>
<td>0.0025</td>
</tr>
<tr>
<td>Thermistor</td>
<td>20</td>
<td>0.01</td>
</tr>
<tr>
<td>TMP175</td>
<td>6</td>
<td>0.01</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td></td>
<td><strong>2.05</strong></td>
</tr>
</tbody>
</table>

Table 19. Expected Power of the Thermal Control System.

<table>
<thead>
<tr>
<th>Component</th>
<th>Number of Components</th>
<th>Voltage (V)</th>
<th>Power (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>ADC128</td>
<td>4</td>
<td>3.3</td>
<td>0.05</td>
</tr>
<tr>
<td>Heaters</td>
<td>10</td>
<td>4.2</td>
<td>TBD</td>
</tr>
<tr>
<td>Thermistors</td>
<td>20</td>
<td>5</td>
<td>0.08</td>
</tr>
<tr>
<td>TMP175</td>
<td>6</td>
<td>3.3</td>
<td>9.90×10⁻⁴</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td></td>
<td></td>
<td><strong>0.13</strong></td>
</tr>
</tbody>
</table>

6.2 Discussion of TEGs Use on CubeSats

Section 5.4 TEG Analytical Results discussed an analytical solution to the energy harvesting of two different commercial TEGs at an average temperature gradient of $\Delta T_{\text{avg}} = 13$ K. The results showed that the thin-film TEG modules produced higher output power per unit area, implying that higher power outputs can be achieved with the smaller devices when placed in series over a given surface area. The power generation is shown to be in the milliwatt range and increases with increasing temperature difference. In terms of efficiency, with the low temperature difference these devices are below 1% for the thin film modules and a little more than 1% for the traditional bulk. As stated in Section 3.4.2 TEG Application to CubeSat, for practical use of TEGs in a 3U CubeSat a minimum of 3% TEG efficiency will
be required. Due to the limited temperature difference available in these small spacecraft in orbit, it is currently impractical to use these current devices for power generation.

Due to size, mass, space, and cost constraints for CubeSats, a comparison of these variables are given in Table 20 for the two TEGs. Assuming the largest area of the different CubeSat form factors can be covered with TEGs, the total number of TEGs is given with the total mass, cost, and total maximum power generation in Table 21. Although the thin-film TEGs provide the highest maximum power output and low mass for the different form factors compared to the traditional bulk TEG, the cost is significantly higher. Clearly, this type of TEG device, although viable for larger amount of power generation, other variables like total implementation cost makes it currently not practical for use in CubeSats.

Table 20. Comparison of Variables for Different TEGs.

<table>
<thead>
<tr>
<th></th>
<th>Footprint Area (cm²)</th>
<th>Mass per Unit (g)</th>
<th>Cost per Unit ($)</th>
<th>Max Power Density (mW/cm²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Traditional Bulk</td>
<td>9</td>
<td>11</td>
<td>49</td>
<td>4.9</td>
</tr>
<tr>
<td>(Custom TE)</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Thin Film</td>
<td>0.10</td>
<td>0.016</td>
<td>90</td>
<td>97</td>
</tr>
<tr>
<td>Nextreme</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 21. Power Gain Comparison for Different CubeSat Form Factors.

<table>
<thead>
<tr>
<th>CubeSat Form Factor</th>
<th>Number of TEGs</th>
<th>Total Mass (g)</th>
<th>Total Cost ($)</th>
<th>Max Power Output (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Traditional Bulk</td>
<td>1U</td>
<td>6</td>
<td>67</td>
<td>0.26</td>
</tr>
<tr>
<td>(Custom TE)</td>
<td>2U</td>
<td>12</td>
<td>134</td>
<td>0.53</td>
</tr>
<tr>
<td></td>
<td>3U</td>
<td>18</td>
<td>202</td>
<td>0.79</td>
</tr>
<tr>
<td>Thin Film</td>
<td>1U</td>
<td>1000</td>
<td>16</td>
<td>9.7</td>
</tr>
<tr>
<td>(Nextreme)</td>
<td>2U</td>
<td>2000</td>
<td>32</td>
<td>19</td>
</tr>
<tr>
<td></td>
<td>3U</td>
<td>3000</td>
<td>48</td>
<td>29</td>
</tr>
</tbody>
</table>

The traditional TEG, although its cost is significantly lower than the thin-film TEGs, the total mass for each configuration is high with not enough power harvesting capability. For a large scale CubeSat mission like the Boeing Colony 2 bus, which has deployable solar panels, it can be semi-affordable and beneficial to implement these devices. But for CubeSats like OSIRIS-3U, there is not
enough gain when compared to all other variables like mass, cost, and board space. Another issue that arises with the implementation of TEGs in CubeSats is the ability to maintain a good thermal gradient across the device. As discussed in Section 3.4.1 Thermoelectric Generation Theory, this can be achieved by adding thermal mass or using heat sinks. Either option increases the mass as well as cost, and heat sinks become complex since a good balance must exist between the thermal resistance of the TEG and heat sink.
Chapter 7. Conclusion and Future Work

Research and development of the thermal control system for OSIRIS-3U and viability of utilizing thermoelectric generators on CubeSats has been presented. A theoretical understanding of the thermal environment that OSIRIS-3U will experience has been summarized and worst-case orbital scenarios have been determined to calculate maximum and minimum temperature limits. For the WCH, the CubeSat was assumed to be in a sun-synchronous polar orbit and for the WCC, the CubeSat was at 0° inclination orbit. Steady-state analytical solutions anticipated the CubeSat to have temperature limits between +113 °C and −56 °C. Time-dependent COMSOL FEA models of OSIRIS-3U was created and simulated for 20 orbital periods. The results demonstrated temperature ranges in between the analytical results as expected for both worst case orbits given in Table 22.

<table>
<thead>
<tr>
<th></th>
<th>WCC</th>
<th>WCH</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max °C</td>
<td>29</td>
<td>110</td>
</tr>
<tr>
<td>Min °C</td>
<td>-12</td>
<td>-4.1</td>
</tr>
</tbody>
</table>

The solution for WCC temperature limits demonstrates that OSIRIS-3U will maintain all operating temperature limits of its critical components. However, the WCH solutions showed that the CubeSat reaches temperatures above the operating temperature of critical components, most critically the batteries. But due to mission requirements, OSIRIS-3U is expected to reach a maximum orbital inclination of 70°, therefore, the CubeSat was assumed to reach temperature limits below the calculated and simulated results. For validation of the OSIRIS-3U simulation results, we compare these limits with SwissCube which is currently orbiting the Earth, their current life tracking website reports maximum and minimum temperature limits of +44 °C and −37 °C, respectively[26], which favorably compare with the results presented here.
Using the aid of the analytical and FEA thermal results, an Independently Operating Thermal Control system was determined appropriate for OSIRIS-3U. The system’s main components include thermistors, TMP175 sensors, and an ADC128 controller, all distributed around critical components of all subsystems (reference Table 17). Lastly, the overall thermal control design was estimated to have a total mass of 2.05 g and total power consumption of 0.13 W.

In the analysis of TEGs, the results indicate that thin-film TEG devices deliver higher power density compared to traditional TEGs, but due to the low temperature gradient expected in a CubeSat the efficiency of these devices falls below 1%. The results also demonstrated that, when comparing variables like mass, cost, and available space for integration of these devices over the power harvesting provided, the viability of utilizing current-technology TEGs in CubeSats is not practical. Ongoing future work on these devices includes experimental testing of both traditional and thin-film TEGs to investigate each devices performance under conditions similar to those inside an orbiting CubeSat. Furthermore, although these devices currently are not favorable for the CubeSat application, this topic should be revisited as advances occur in the efficiencies of these devices increases for low $\Delta T$. Currently, there is ongoing research that includes the development of the “nighttime solar cell”, which is investigating the possibility of integrating TEGs in solar cells to generate power during both sunlit and night time periods [18]. In addition, the *Space TEG Feasibility Study* is being conducted in which TEG efficiencies are expected to increase up to values of 25–30% [28].

In conclusion, a thermal subsystem’s primary goal is to maintain temperatures of all spacecraft components within allowable operational ranges during all mission phases. The design of a thermal control begins with understanding, calculating, and modeling the thermal environment that a spacecraft will experience during its mission. Therefore, successful models were created and analyzed for the thermal environment expected for OSIRIS-3U, as well as the development of a thermal control with the aid of these results. There is still future work to be done in order to ensure proper thermal design for reliability of OSIRIS-3U in performing all mission requirements, including a more detailed FEA model in
COMSOL where critical components are added to internal circuit boards and simulating their temperature evolution.
Appendix A

For a cylindrical coordinate system, the steady state-heat conduction equation is described as [4]:

\[ Q_{\text{cond,cyl}} = \frac{2\pi k L (T_1 - T_2)}{\ln \left( \frac{D_o}{D_i} \right)} \]

For spherical coordinate system the steady state heat conduction equation is described as [4]:

\[ Q_{\text{cond,sph}} = \frac{4\pi k R_i R_o (T_1 - T_2)}{R_o - R_i} \]
Appendix B

MATLAB Code for GMM WCH

cclc;
clear all;

%%%%%%%%%% Constants %%%%%%%%%%
Re=6371;                    % Radius of Earth (km)
z=300;                      % Orbital Altitude (km)
beta_e=acosd(Re/(Re+z));    % Beta Earth angle (deg)
rho=0.3;                    % Albedo coefficient
Q_sun=1400;                 % Solar Heat Flux (W/m^2)
Q_ir=270;                   % Earth IR Heat Flux (W/m^2)
alpha=0.9;                  % Abs coefficient for FR4
emi=0.8;                    % Emission Coefficient for FR4
sigma=5.67e-8;              % Plank's Constant (W/(m^2K^4)

%% Calculating View Factors %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
VF_nadir =((Re/(Re+z))^2; % Earth Facing

a=sqrt(1-VF_nadir);
b=2*asin(a);
c=sin(b);

VF_walls = (pi-b-c)/(2*pi); % Perpendicular to nadir

%% Beta Angles for Each Face %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% with Respect to Sun %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
theta= 0:360;
beta1=theta+180;
beta2=theta;
beta3=theta;
beta4=theta-90;
beta5=theta;
beta6=theta+90;

% CubeSat Area %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

area12=0.1*0.1; % Area for Panels 1,2 (m^2)
area3456=0.1*0.3; % Area for Panels 3,4,5,6 (m^2)
Asc=0.14; % Total CubeSat Area (m^2)

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 1 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_1=zeros(1,361);
Qalbedo_1=Q_sun*rho*VF_nadir*ones(1,361);
Qir_1=Q_ir*VF_nadir*ones(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 2 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_2 = zeros(1,361);
Qalbedo_2=zeros(1,361);
Qir_2=zeros(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 3 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_3=Q_sun*ones(1,361);
Qalbedo_3=Q_sun*rho*VF_walls*ones(1,361);
Qir_3=Q_ir*VF_walls*ones(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 4 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_4=zeros(1,361);
Qalbedo_4=Q_sun*rho*VF_walls*ones(1,361);
Qir_4=Q_ir*VF_walls*ones(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 5 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_5=zeros(1,361);
Qalbedo_5=Q_sun*rho*VF_walls*ones(1,361);
Qir_5=Q_ir*VF_walls*ones(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 6 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_6 = zeros(1,361);
Qalbedo_6=Q_sun*rho*VF_walls*ones(1,361);
Qir_6=Q_ir*VF_walls*ones(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Total Heat Inputs for Each Panel %
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Panel_1=(Qsolar_1+Qalbedo_1+Qir_1)*area12; % (W)
Panel_2=(Qsolar_2+Qalbedo_2+Qir_2)*area12; % (W)
Panel_3=(Qsolar_3+Qalbedo_3+Qir_3)*area3456; % (W)
Panel_4=(Qsolar_4+Qalbedo_4+Qir_4)*area3456; % (W)
Panel_5=(Qsolar_5+Qalbedo_5+Qir_5)*area3456; % (W)
Panel_6=(Qsolar_6+Qalbedo_6+Qir_6)*area3456; % (W)

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Plots of Heat Inputs for Individual Panels %%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
figure
subplot(2,1,1)
plot(theta,Panel_1,theta,Panel_3, theta,Panel_5);
hleg1 = legend('Panel 1','Panel 3', 'Panel 5...', 'Location','NorthEastOutside');
title('Thermal Heat Inputs of OSIRIS-3U')
ylabel('Environmental Heat Flux (W)')
axis([0 365 0 60])

subplot(2,1,2)
plot(theta,Panel_2,theta,Panel_4,theta,Panel_6);
hleg2 = legend('Panel 2','Panel 4', 'Panel 6...', 'Location','NorthEastOutside');
xlabel('Orbital Angle')
ylabel('Environmental Heat Flux (W)')
axis([0 360 -1 8])
Appendix C

MATLAB Code for GMM WCC

```matlab
clc;
clear all;

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Constants
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Re=6371;                    % Radius of Earth (km)
z=400;                      % Orbital Altitude (km)
beta_e=acosd(Re/(Re+z));    % Beta Earth angle (deg)
rho=0.3;                    % Albedo coefficient
Q_sun=1300;                 % Solar Heat Flux (W/m^2)
Q_ir=220;                   % Earth IR Heat Flux (W/m^2)
alpha=0.9;                  % Abs coefficent for FR4
emi=0.8;                    % Emission Coefficient for FR4
sigma=5.67e-8;              % Plank's Constant (W/(m^2K^4)

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Calculating View Factors
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
VF_nadir =(Re/(Re+z))^2;    % Earth Facing
a=sqrt(1-VF_nadir);
b=2*asin(a);
c=sin(b);
VF_walls = (pi-b-c)/(2*pi); % Perpendicular to nadir

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Beta Angles for Each Face
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
theta= 0:360;
beta1=theta+180;
beta2=theta;
beta3=theta;
beta4=theta-90;
beta5=theta;
beta6=theta+90;

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% CubeSat Area
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%```
area12=0.1*0.1;            % Area for Panels 1,2 (m^2)
area3456=0.1*0.3;          % Area for Panels 3,4,5,6 (m^2)
Asc=0.14;                  % Total CubeSat Area (m^2)

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 1 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_1=zeros(1,361);
Qalbedo_1=Q_sun*rho*VF_nadir*cosd(theta);
for i=1:361;
    if (i>=90) && (i<=270);
        Qalbedo_1(i)=0;
    else
        end
end
Qir_1=Q_ir*VF_nadir*ones(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 2 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_2 = Q_sun*cosd(theta);
for i=1:361;
    if Qsolar_2(i)<0.0;
        Qsolar_2(i)=0;
    else
        end
end
Qalbedo_2=zeros(1,361);
Qir_2=zeros(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Heat Flux Input Panel 3 %%%%%%%%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_3=zeros(1,361);
Qalbedo_3=Q_sun*rho*VF_walls*cosd(beta3);
for i=1:361;
    if i>=90 && i<=270;
        Qalbedo_3(i)=0;
    else
        end
end
Qir_3=Q_ir*VF_walls*ones(1,361);
%% Calculating Heat Flux Input Panel 4 %%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_4=Q_sun*cosd(beta4);
for i=1:361;
    if i>=180-beta_e && i<=180+beta_e;
        Qsolar_4(i)=0.0;
    elseif Qsolar_4(i)<0
        Qsolar_4(i)=0;
    end
end
Qalbedo_4=Q_sun*rho*VF_walls*cosd(theta);
for i=1:361;
    if i>=90 && i<=270;
        Qalbedo_4(i)=0;
    else
        Qalbedo_4(i)=0;
    end
end
Qir_4=Q_ir*VF_walls*ones(1,361);

%% Calculating Heat Flux Input Panel 5 %%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_5=zeros(1,361);
Qalbedo_5=Q_sun*rho*VF_walls*cosd(beta5);
for i=1:361;
    if i>=90 && i<=270;
        Qalbedo_5(i)=0;
    else
        Qalbedo_5(i)=0;
    end
end
Qir_5=Q_ir*VF_walls*ones(1,361);

%% Calculating Heat Flux Input Panel 6 %%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Qsolar_6 = Q_sun*cosd(beta6);
for i=1:361;
    if i>=180-beta_e && i<=180+beta_e;
        Qsolar_6(i)=0;
    elseif Qsolar_6(i)<0
        Qsolar_6(i)=0;
    end
end
Qalbedo_6=Q_sun*rho*VF_walls*cosd(theta);
for i=1:361;
    if i>=90 && i<=270;
        Qalbedo_6(i)=0;
    else
        end
end

Qir_6=Q_ir*VF_walls*ones(1,361);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Calculating Total Heat Inputs for Each Panel %%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Panel_1=(Qsolar_1+Qalbedo_1+Qir_1)*area12;   % (W)
Panel_2=(Qsolar_2+Qalbedo_2+Qir_2)*area12;   % (W)
Panel_3=(Qsolar_3+Qalbedo_3+Qir_3)*area3456; % (W)
Panel_4=(Qsolar_4+Qalbedo_4+Qir_4)*area3456; % (W)
Panel_5=(Qsolar_5+Qalbedo_5+Qir_5)*area3456; % (W)
Panel_6=(Qsolar_6+Qalbedo_6+Qir_6)*area3456; % (W)

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Plots of Heat Inputs for Individual Panels %%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%
figure
subplot(2,1,1)
plot(theta,Panel_1,theta,Panel_3, theta,Panel_5);
hlen = legend('Panel 1','Panel 3', 'Panel 5',...
'Location','NorthEastOutside');
title('Thermal Heat Inputs of OSIRIS-3U')
ylabel('Environmental Heat Flux (W)')
axis([0 365 2 7])

subplot(2,1,2)
plot(theta,Panel_2,theta,Panel_4,theta,Panel_6);
hlen2 = legend('Panel 2','Panel 4', 'Panel 6',...
'Location','NorthEastOutside');
xlabel('Orbital Angle')
ylabel('Environmental Heat Flux (W)')
axis([0 365 -1 50])
Appendix D

Trade Study of Thermal Design Hardware

0-Metric no favorable  1-Metrics in row and column  2-Metric in row is favorable
consider equal, and metric than column metric.

Table 23. Pair-Wise Comparison Matrix of Design Metrics.

<table>
<thead>
<tr>
<th>Temperature Range</th>
<th>Precision</th>
<th>Heritage</th>
<th>Footprint</th>
<th>Sensor Size</th>
<th>Mass</th>
<th>Power Dissipation</th>
<th>Removability</th>
<th>Number of Leads</th>
<th>Durability</th>
<th>Lifetime</th>
<th>0</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Temperature Range</td>
<td>1</td>
<td>1</td>
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<td>0</td>
<td>2</td>
<td>1</td>
<td>1</td>
<td>2</td>
<td>0</td>
<td>0</td>
<td>2</td>
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</tr>
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<td>2</td>
<td>1</td>
<td>2</td>
<td>1</td>
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<td>1</td>
<td>2</td>
<td>0</td>
<td>1</td>
<td>0.095</td>
</tr>
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<td>Heritage</td>
<td>0</td>
<td>0</td>
<td>1</td>
<td>1</td>
<td>2</td>
<td>0</td>
<td>1</td>
<td>2</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>0.091</td>
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<tr>
<td>Footprint</td>
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<td>1</td>
<td>1</td>
<td>1</td>
<td>2</td>
<td>1</td>
<td>2</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>2</td>
<td>0.096</td>
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<td>Sensor Size</td>
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<td>0</td>
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<td>0</td>
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<td>2</td>
<td>2</td>
<td>0</td>
<td>0</td>
<td>0.085</td>
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<td>2</td>
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<td>2</td>
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<td>Power Dissipation</td>
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<td>1</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>0</td>
<td>0</td>
<td>0.081</td>
</tr>
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<td>Removability</td>
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<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>0</td>
<td>0.081</td>
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<tr>
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<td>1</td>
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<td>1</td>
<td>1</td>
<td>1</td>
<td>0</td>
<td>0</td>
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<td>1</td>
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</tr>
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<td>1</td>
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<td>1</td>
<td>2</td>
<td>2</td>
<td>1</td>
<td>1</td>
<td>2</td>
<td>0.098</td>
</tr>
</tbody>
</table>
## Scale Performance of Hardware

1- Poor   2-Average   3-Good   4-Very Good   5- Excellent

**Table 24. Pugh Matrix for Determining Best Thermal Sensor.**

<table>
<thead>
<tr>
<th>Metrics</th>
<th>Weights</th>
<th>Temperatures Range</th>
<th>Precision</th>
<th>Heritage</th>
<th>Footprint</th>
<th>Sensor Size</th>
<th>Mass</th>
<th>Power Dissipation</th>
<th>Removability</th>
<th>Number of Leads</th>
<th>Durability</th>
<th>Lifetime</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>Raw Score</td>
<td>Weighted Score</td>
<td>Raw Score</td>
<td>Weighted Score</td>
<td>Raw Score</td>
<td>Weighted Score</td>
<td>Raw Score</td>
<td>Weighted Score</td>
<td>Raw Score</td>
<td>Weighted Score</td>
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<td>0.278</td>
<td>3</td>
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<td>0.185</td>
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<td>1</td>
<td>0.091</td>
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<td>Standard Disc Style</td>
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<td>3</td>
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<td>US Sensor Thermistor</td>
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<td>2</td>
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<td>0.255</td>
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<td>Raw Score</td>
<td>Weighted Score</td>
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<td>Weighted Score</td>
<td>Raw Score</td>
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<tr>
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<td>1</td>
<td>0.098</td>
</tr>
</tbody>
</table>

| Final Score      | 2.977   | 2.834              | 2.940       | 2.336      | 3.000      | 2.892       |

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References


[27] The Thermoelectric Effect, COMSOL Multiphysics.


