A THERMAL INVESTIGATION AND COMPARATIVE STUDY OF THE FORESAIL MISSIONS

Kartik Praful Naik

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A THERMAL INVESTIGATION AND COMPARATIVE STUDY OF THE FORESAIL MISSIONS

Kartik Praful Naik

School of Electrical Engineering

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Thesis supervisor:
Prof. Jaan Praks

Thesis Examiner:
Dr. Soheil Sadeghi
Cube Satellite (CubeSat) launches have been on the rise since its first launch in 2003. This popularity is mainly due to faster design process and lower launch costs. However, most CubeSats are launched into Low Earth Orbits (LEOs), with no missions to Geostationary Transfer Orbits (GTOs). However, many mission are planned for the next half-decade.

A major challenge to launch a CubeSat into a GTO is the thermal environment of the higher altitude orbits. These orbits are significantly colder due to reduced heating from Earth’s planetary and albedo radiations, and a possibility for longer eclipses due to the eccentricities of GTOs.

A thermal investigation of the thermal environment was done using the Foresail missions as examples, as the missions currently are set to fly the first missions to Polar LEO. The trajectories for the second Foresail mission are being evaluated, with the GTO being a strong contender. This thermal investigation is done through a comparative study of the two missions. The thermal effects of a few mission specific scenarios were also evaluated.

This provided a holistic thermal design of the first Foresail mission. A region specific thermal solution for the battery was analyzed. The various mission scenarios and their comparisons with the LEO mission, also formed a basis of the feasibility of various situations on the second mission. Moreover, the results, in part assessed the thermally feasibility to launch a 3U CubeSat into a GTO.

The results showed GTOs show larger magnitude of variation of thermal loads as compared to LEOs. However, these variations are more gradual due to the larger orbital periods. A 3U CubeSat can be launched into both, the LEO and GTO environments with passive thermal control. The properties of the thermal coats vary slightly. However, it is not possible to passively control the CubeSat if the eclipse occurred at the aphelion of the orbit.

Keywords: Space Systems, Thermal Design, Cube Satellite
Preface

I would like to thank Prof. Jaan Praks for this wonderful learning opportunity. During my stay at Aalto, I have learnt a lot about the functioning of a Satellite team that actually have built and launched satellites. This has been the first such experience.

I would like to thank Dr. Victoria Barabash for allowing me to pursue my thesis at Aalto University.

I would like to thank to the Foresail and Aalto -3 teams, as I have learnt much from the team members.

I would like to thank Dr. Soheil Sadeghi for agreeing to be my examiner and facilitating this thesis.
# Contents

Abstract  
Preface  
Contents  
Symbols, Abbreviations, Figures and Tables  

## 1 Introduction
1.1 Small Satellites and Cubesats  
1.2 Foresail Missions  
  1.2.1 Foresail-1  
  1.2.2 Foresail-2  
1.3 Problem Definition: Thermal Design  

## 2 Thermal Design Principles
2.1 Layout  
2.2 Heat Transfer in Space  
  2.2.1 Conduction vs Radiation  
  2.2.2 Thermal Couplings  
  2.2.3 Calculations  
2.3 Thermal Loads  
  2.3.1 External Loads  
  2.3.2 Internal Loads  
2.4 Thermal Design Examples  
  2.4.1 Holistic Thermal Design: FASTRAC  
  2.4.2 Holistic Thermal Design: GTOSat  
  2.4.3 Specific Thermal Design: NiCd Battery  
2.5 Thermal Control Methodology  

## 3 Implementation  
3.1 Layout  
3.2 Thermal Requirements  
3.3 Orbit Definition  
3.4 Thermal Load Analysis  
3.5 Simulation Setup  
  3.5.1 Summary  
  3.5.2 CAD Modelling  
  3.5.3 Idealization  
  3.5.4 Meshing and Material Properties  
  3.5.5 Thermal Couplings  
  3.5.6 Thermal Loads  
  3.5.7 Constraints
Symbols, Abbreviations, Figures and Tables

Symbols

Latin

\( Q \)  Heat Energy or Work (measured in Joules)
\( A \)  Area of cross-section
\( k \)  Thermal conductivity
\( T \)  Temperature

Greek

\( \sigma \)  Stefan-Boltzmann constant \((=5.67 \cdot 10^{-8} W m^{-2} K^{-4})\)
\( \kappa \)  Thermal diffusivity
\( \epsilon \)  Emissivity
\( \alpha \)  Absorptivity
\( \rho \)  Density

Operators

\( \dot{Q} \)  First time derivative of \( Q \)
\( \frac{d}{dt} \)  derivative with respect to variable \( t \)
\( \frac{\partial}{\partial x} \)  partial derivative with respect to variable \( x \)
### Abbreviations

<table>
<thead>
<tr>
<th>Abbreviation</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>‘∗’</td>
<td>Number or numeric denotation (used in this table)</td>
</tr>
<tr>
<td>Al-∗</td>
<td>Aluminum - Grade. For example, Al-6061</td>
</tr>
<tr>
<td>CAD</td>
<td>Computer Aided Design</td>
</tr>
<tr>
<td>COTS</td>
<td>Commericially Off The Shelf</td>
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<tr>
<td>CubeSat</td>
<td>Cube Satellite (Special class of Nano-satellite)</td>
</tr>
<tr>
<td>∗U</td>
<td>‘n’ unit CubeSat (For example: 3U = 3 Unit CubeSat)</td>
</tr>
<tr>
<td>PATE</td>
<td>PArticle TElescope (Foresail missions)</td>
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<tr>
<td>IR</td>
<td>Infrared</td>
</tr>
<tr>
<td>MLI</td>
<td>Multi-layered insulation</td>
</tr>
<tr>
<td>LEO</td>
<td>Low Earth Orbit</td>
</tr>
<tr>
<td>MEO</td>
<td>Middle Earth Orbit</td>
</tr>
<tr>
<td>GEO</td>
<td>Geostationary Earth Orbit</td>
</tr>
<tr>
<td>GTO</td>
<td>Geostationary Transfer Orbit</td>
</tr>
<tr>
<td>FS-∗</td>
<td>ForeSail Mission - Number (FS-1, FS-2, etc.)</td>
</tr>
<tr>
<td>PCB</td>
<td>Printed Circuit Board</td>
</tr>
<tr>
<td>NX, ANSYS, ESATAN</td>
<td>Official software names</td>
</tr>
<tr>
<td>∗-D</td>
<td>‘n’ dimensional. (2-D, 3-D, etc.)</td>
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<tr>
<td>FR4</td>
<td>A glass-reinforced epoxy laminate material</td>
</tr>
<tr>
<td>RAAN</td>
<td>Right Ascension of the Ascending Node</td>
</tr>
</tbody>
</table>
## List of Figures

1. The yellow line represents the total CubeSat launches since 2003. This number reached 932 by 2018 and continues to increase with an exponential trend [Nanosatellite Database, ].

2. (L) Render of the top isometric view of FS-1. (M) A composite render of the internal components, exterior render and wireframe model. (R) Placement of the three modules in the interior of the satellite. (Credit: Aalto University; Arno Alho)

3. Nanosatellite orbits at various altitudes. It is evident that almost all orbits are LEOs, save the 2 deep space orbits [Nanosatellite Database, ].

4. Ideal cases for conduction and radiation: (L) Perfect contact – Conduction (R) Small gap – Radiation.

5. Conductive coupling in a composite wall: (L) Snapshot showing the interface of the solar panel and the rails (structural frame) (R) Reduced thermal circuit.

6. Comparison of uncorrected and corrected estimates of albedo fluxes. It can be seen the there is a variable offset of 20 to 40 W/m² throughout.

7. NiCd battery thermal design; Pg 77, [Gilmore and Donabedian, 2003].

8. (L) FS-1’s polar sun-synchronous orbit; (R) FS-2’s highly elliptical GTO. Points marked in yellow color are the orbital positions used for transient thermal simulations in the subsequent section.

9. Thermal loads on plates. (Defined with respect to orbital references. For example, Nadir or Earth facing (Nadir +) and the opposite plate (Nadir -). This is explained in detail in the description.

10. Comparison of total thermal loads (incident external fluxes) for FS-1,2 for one GTO period.

11. FS-1 Design.


14. Elements chosen in each external face of the CubeSat.

15. Element-wise time variation of temperatures in FS-1; Temperature (y-axis) is plotted vs time (x-axis).

16. Element-wise time variation of total loads in FS-1; Thermal loads (y-axis) is plotted vs time (x-axis).


18. Element-wise time variation of temperatures in FS-2; Temperature (y-axis) is plotted vs time (x-axis).

19. Element-wise time variation of total loads in FS-2; Thermal loads (y-axis) is plotted vs time (x-axis).

20. Times at minimum temperatures in FS-2.

21. Temperature Distributions Profiles of FS-1 with Spin: (L) Exterior view (R) Interior View.
22 Element-wise time variation of total loads in FS-1 (Spinning mode). 34
23 Thermal Solution: Configuration depicting the layers of material setup for the thermal simulation of the battery. 36
24 Battery thermal simulation results: (L) Temperature distribution profile at the end of the simulation (R) Temperature variation of the various entities over 1000s. Here, the blue plot represents the metallic bracket temperature (reservoir) - Constant at -20 °C; the pink plot represents the battery temperature - Lowest 0.83 °C; and the red plot represents the maximum temperature of the test setup. 37
25 Temperature Distributions Profiles of FS-2 with deployable solar panels: (L) Exterior view (R) Interior View. 39
26 Element-wise time variation of total loads in FS-2 with solar panels: (L) Load variation elements places on the 6 surfaces; (R) Element placed in the surface shadowed by the deployed solar panel. 40
27 FS-2 GTO orbits: (L) Near-side eclipse (original assumption); (R) Far-side eclipse (Worst case). 41
28 Temperature Distributions Profiles of FS-2 in far-side eclipse GTO: (L) Exterior view (R) Interior View. 42
29 Element-wise time variation of total loads in FS-2 in the far-side GTO. 42
30 Temperature Distributions Profiles of FS-2 in MEO: (L) Exterior view (R) Interior View. 44
31 Element-wise time variation of total loads in FS-2 in MEO. 45
A1 Low Lunar Orbit propagated for thermal simulation of Lunar orbital loads. 52
A2 Temperature Distributions Profiles of the CubeSat in an LLO. 52
A3 Element-wise time variation in LLO: (L) Total Flux (R) Temperatures. 53
B1 Thermal simulation results in STK: (L) Mean equilibrium temperatures of the object in FS-2’s GTO; (R) Mean equilibrium temperatures of the object in FS-1’s LEO; Y axis represents the temperature, X represents time. 54
List of Tables

1. Results: Radiation and Conduction comparison ........................................ 7
2. FASTRAC Summary .............................................................................. 12
3. Dellingr CubeSat Summary ................................................................. 12
4. List of current and future CubeSat missions to the GTO, HEO and deep space (higher altitude missions than LEOs) ..................... 14
5. List of available thermal control methods with their respective TRL (Technology Readiness Level) and availability [Yusupov, 2018] .................. 15
6. Thermal Requirements (All temperatures in °C) .................................. 16
7. Orbital parameters of FS ...................................................................... 17
8. Thermal Properties of materials used .................................................... 22
9. Thermo-optical Properties of materials used ........................................ 23
10. Thermal Coupling Estimates for small, dissimilar interfaces ............. 23
11. Simulation Setup: Solver parameters ................................................... 24
12. Simulation Result Summary ................................................................. 31
13. FS-1 Simulation Result Summary: No spinning vs Spinning ........... 35
14. FS-2 Simulation Result Summary: With and without deployed solar panels .......................................................... 40
15. FS-2 Simulation Result Summary: Far-side and Near-side Eclipse GTOs 43
16. Total thermal flux comparison: LEO vs GTO (all values in W/m²) .... 44
17. FS-2 Simulation Result Summary: GTO vs MEO ............................... 45
18. Report Summary .................................................................................. 46
A1. Thermal Environment Comparison: Earth-bound vs Moon-bound orbits 51
1 Introduction

1.1 Small Satellites and Cubesats

Small satellites have gained popularity in the past decade among organizations invested in space research. They are categorized as satellites with a mass lower than 500 kg and have become popular for a simple reason – Smaller satellites in large numbers are often more useful and more economic than fewer, larger ones for the same purposes (like radio relay, scientific data gathering, etc.). Small satellites are further categorized according to their masses into Minisatellite (100–500 kg); Microsatellite (10–100 kg); Nanosatellite (1–10 kg); Picosatellite (0.1–1 kg) and Femtosatellite (0.01–0.1 kg) [Konecny, 2004].

CubeSats are a standardized class of Nanosatellites, that are designed as modular unit volumes (1U = A cubic volume of side length 100 mm) [Mehrparvar et al., 2014], [Lee et al., 2009]. Since the first CubeSat launch in 2003, there has been an exponential increase in the number of launches per year. Fig 1 shows this trend. Over 1100 nanosatellites have been launched as of January 2019 [Nanosatellite Database, ]. The main reason for this success is the low cost of deployment. They can be launched in large numbers and are also often launched as piggybacks, minimizing the risk on the primary payload of the launch vehicle; and reducing mission and launch costs. Moreover, they usually use COTS (Commercially Off The Shelf) components, further reducing the costs.

Figure 1: The yellow line represents the total CubeSat launches since 2003. This number reached 932 by 2018 and continues to increase with an exponential trend [Nanosatellite Database, ].
1.2 Foresail Missions

Aalto University, having launched the first Finnish Satellite into space, has a keen interest and expertise in CubeSat missions. The Foresail missions are among the ongoing projects at Aalto. Currently, two Foresail missions have been planned, the first of which is set to launch by early 2020.

1.2.1 Foresail-1

Foresail-1 is a CubeSat mission of Finnish Centre of Excellence in Research of Sustainable Space hosting two payloads – the PArticle TElescope (PATE) from University of Turku, and the plasma brake, a de-orbiting device, from Finnish Meteorological Institute. The spacecraft platform is designed, manufacture and assembled by the Aalto University team.

The PATE is a particle detector, capable to measure electrons and protons, their energies and pitch angles. The electrons are monitored with a range of 80 to 1600 keV with a single channel, while the protons are monitored with a range of 8 to 30 MeV in two energy channels. The secondary payload is the Electrostatic Plasma Brake, developed by the Finnish Meteorological Institute. The device is designed to lower the orbit altitude of the satellite and de-orbit the satellite [Foresail Satellite, ].

Figure 2: (L) Render of the top isometric view of FS -1. (M) A composite render of the internal components, exterior render and wireframe model. (R) Placement of the three modules in the interior of the satellite. (Credit: Aalto University; Arno Alho)

Foresail-1 (FS-1) is a 3U CubeSat with four deployable antennas and four (undeveloped) solar panels, mounted on the four long faces of the satellite. As of April 2019, the first prototype was manufactured and tested for mechanical vibrations. Fig
illustrates the renders of the model and its interior. As shown, there are three principal modules in the interior of the structural frame: The payloads, PATE and MASTER; and the Avionics module. All modules are encased in Aluminum, and bolted in place with connections to the structural frame.

1.2.2 Foresail-2

The Foresail-2 (FS-1) mission is planned with two payloads on-board provided by the same science teams (University of Turku and the Finnish Meteorological Institute) to be flown in a higher altitude orbit. This mission is still in its mission definition and design phase and thus, multiple orbit scenarios are being evaluated. The most favorable scenario for now is a GTO (Geostationary Transfer Orbit). However, MEOs (Medium Earth Orbits) and other high-altitude orbits are also being investigated, to meet the scientific requirements of the payloads. As the mission design is in its early phases, the exact mission and scientific requirements will not be discussed in this thesis.
1.3 Problem Definition: Thermal Design

Thermal design refers to the design, or selection of a suitable method to control the temperatures of various regions of the satellite to ensure smooth operations at best, and survival at least. CubeSats being small entities, have tight mass and power budgets, along with budget and space constraints to allow active or large thermal systems. Since there usually is a small metallic region exposed to the external environment, conventional MLIs (Multi-Layered Insulations) are avoided as well. Thus, the satellite relies mainly on the thermal control obtained from the change in thermo-optical properties, with the use of paints and coatings. Smaller active control methods like patch heaters, conformal coatings, etc. can be used for region-specific thermal control. This is the main challenge posed to the thermal engineer in CubeSats [Dinh, 2012].

The goal of the thesis is to provide a comparison between the thermal environments of a polar LEO and a GTO, using the Foresail missions as an example. It also provides an assessment on a few thermal scenarios that are based on the Foresail missions but can be generalized to a large extent. This is done by thermal analysis. The results of which are then used to develop a comparative study of FS-1 and FS-2 by placing the 3U CubeSat in their respective mission scenarios. This comparison is needed to understand the difference between the thermal environments of the LEO and the GTO. Fig 3 shows the altitudes of the launched CubeSats in the past few years. There has been no satellite launched into high altitude orbits, especially a GTO. This makes it particularly challenging as there are no thermal designs with flight heritage and thus, a thorough comparison with a LEO mission - with high flight heritage - in an important step towards GTO mission design.

![Nanosatellite orbits at various altitudes. It is evident that almost all orbits are LEOs, save the 2 deep space orbits](Nanosatellite Database, )
The comparative analysis will also form a basis to determine, in part, the feasibility of launching a 3U CubeSat into a GTO. Our approach is to evaluate the effects of the different thermal environments by studying the incident thermal loads – The GTO has larger, but much slower thermal load variations (shown in section 3.4). This is followed by the validation of this hypothesis by numerical thermal simulations in section 4. Finally, thermally sensitive regions are found, and specific thermal control measures are suggested. In sections 5 and 6, some prospective mission scenarios’ of FS-1 and FS-2 respectively are investigated for their thermal effects.
2 Thermal Design Principles

2.1 Layout

This section briefly explains the underlying theory of the thermal design process described in the subsequent parts. The theory first explains the heat transfer in orbit, underlining the differences between the ground and space environments. It then proceeds to explain the thermal loads on the satellite in orbit. In section 2.4, a few thermal design examples are studied to get a general idea of the thermal control strategies used in similar missions. Finally, a summary of thermal control methods and their applicability for our mission is described in section 2.5.

2.2 Heat Transfer in Space

Most of space is vacuum at a temperature of 2.7 K, making it a thermal reservoir. Thus, inside the satellite, almost all heat transfer takes place through conduction or radiation with space (except for convection in small amounts when propulsion or heat pipes are used).

2.2.1 Conduction vs Radiation

Conduction takes place between any surfaces in direct contact whereas radiation does not require contact. Moreover, the direction of the conductive heat flow is in the direction of the temperature differential. The steady and transient conduction equations in one dimensional flow (most commonly observed form) are [Bergman et al., 2011]:

\[ \dot{Q} = k \cdot A \cdot \frac{\partial T}{\partial x}, \]  
\[ \frac{\partial T}{\partial t} = \kappa \cdot \frac{\partial^2 T}{\partial x^2}, \]  

where, \( \dot{Q} \) is rate of heat transfer; \( k \) is thermal conductivity of the medium; \( A \) is Area of cross-section of the heat flow; \( T \) is Temperature; \( x \) is linear dimension; \( \frac{\partial T}{\partial x} \) is gradient of temperature; \( \kappa = \frac{k}{\rho c_p} \) = thermal diffusivity; \( \rho \) is density of material; \( c \) is specific heat capacity.

On the other hand, the radiation flow depends on the difference of the fourth powers of the respective temperatures of the surfaces. The net radiative heat transfer law is given by the Stefan-Boltzmann equation [Bergman et al., 2011],

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

where, \( \epsilon \) is emissivity; \( \sigma \) is Stefan-Boltzmann constant \( = 5.67 \cdot 10^{-8} \text{Wm}^{-2}\text{K}^{-4} \); \( A \) is Area of emitting surface; \( T_1 \) is Temperature of emitting surface; \( T_2 \) is Temperature of surroundings. All the heat transfer from the external environment to the satellite body is due to incident radiation. There can be no conduction or convection due to the absence of a medium for the heat to flow. On the other hand, a significant amount of heat in transferred within the body is through conduction. There is no
convection whatsoever as there is an absence of a flowing medium (unless in the case of propulsion). Moreover, there is negligible radiative heat transfer among internal components.

To support this claim, both modes of heat transfer are studied between similar unit metal surfaces: The first, with perfect contact to allow pure conduction; the second, with a small distance of vacuum in between to surfaces to allow pure radiative heat transfer. This distance is assumed small to avoid losses due to distance (intensity of radiation varied inversely with the square of distance from the source) and view factors (Radiation is diffuse and is spread in all directions. The further the distance from the source, the smaller amount of radiation leaving the source incident on the target). This setup is depicted in Fig 4.

Figure 4: Ideal cases for conduction and radiation: (L) Perfect contact – Conduction (R) Small gap – Radiation.

A unit area was assumed for ease of calculations. A temperature differential of 5 K (in the case of radiation, $T_1 = 300$ K and $T_2 = 295$ K) was assumed in both cases, and Aluminum was assumed as a material with emissive properties of anodized Al ($\epsilon = 0.8$). Here, the emissivity ($\epsilon$) refers to effectiveness of the surface to radiate its heat, compared to a black-body (ideal radiator $\epsilon = 0.8$). Equation 3 is used calculate the radiative heat transfer which assumes the Kirchhoff’s law of radiation [Trefil, 2003]. This mean that, as the surfaces are so close, all radiation emitted by surface one is absorbed by surface, with no losses. The results are tabulated in Table 1.

<table>
<thead>
<tr>
<th>Type</th>
<th>Heat transfer</th>
<th>Remark</th>
<th>[Thermal Properties, ]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Conduction</td>
<td>1025 W</td>
<td>Over unit length, $K_a l = 205$</td>
<td></td>
</tr>
<tr>
<td>Radiation</td>
<td>26.87 W</td>
<td>View factor = 1</td>
<td></td>
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Table 1: Results: Radiation and Conduction comparison.

The ratio of radiative to conductive heat transfer using the aforementioned conditions is negligible (ratio = 2.62 %). Moreover, in the interior of the spacecraft, the view factors are much lower due to obstructions, further reducing radiative heat exchange.
2.2.2 Thermal Couplings

Thermal couplings are an important means of defining the thermal connection between two bodies or faces of bodies. It has been established that most of the heat transferred internally is due to conduction and most of the external heat transfers are through radiation. Thermal coupling calculations are an integral part of thermal analysis.

Heat transfer coefficient: As conductive and radiative coefficients have different units; a normalized constant is introduced into play. Heat transfer coefficient is the ratio of the heat flux and the difference of temperatures in the bodies (thermodynamic driving force). Mathematically, it is expressed as [Trefil, 2003]

\[ h = \frac{\partial \dot{Q}}{\partial T} \cdot \frac{1}{\Delta T}, \]  

(4)

where, \( A \) is area of cross-section of the heat flow; \( \Delta T \) is temperature difference between the bodies/surfaces. Heat transfer coefficient being a measure of overall heat transfer, is sometimes more useful in simulations as surfaces can be assigned with one thermal coupling instead of multiple thermal couplings for various modes of the heat transfer.

2.2.3 Calculations

They are two main cases: First, when the area of the conductive coupling is uniform, with an interface or similar or dissimilar materials – This case is straightforward, one can treat it as simple series connection. Thus, the thermal resistances directly add up as [Trefil, 2003]

\[ R = R_1 + R_2 + ... R_n = \frac{l_1}{k_1 \cdot A_1} + \frac{l_2}{k_2 \cdot A_2} + ... \frac{l_n}{k_n \cdot A_n}, \]  

(5)

where, \( R \) is thermal resistivities, and its relation to thermal conductivity is given by \( k = R^{-1} \); \( k_n \) is thermal conductivity of the \( nth \) body; \( l_n \) is length of the \( nth \) body; \( A_n \) is cross-sectional area of the \( nth \) body.

Second, when the area is not uniform. This occurs mainly when there are empty spaces in the between. For instance, in the structural frame, the rails are attached to the edges of the solar panels. This is illustrated Fig 5.

Once the interface is reduced to thermal circuits, it can be solved using the laws for series-parallel flow. In this case, it is reduced to:

\[ R = R_{SolarPanel} + ((R_{Rail1})^{-1} + (R_{Rail2})^{-1} + (R_{Vacuum})^{-1})^{-1}, \]  

(6)

where, \( R_{SolarPanel} \) is thermal resistivity of the solar panel; \( R_{Rail} \) is thermal resistivity of the rails; and \( R_{Vacuum} \) is thermal resistivity of vacuum. As conduction in vacuum is negligible, \((R_{Vacuum})^{-1})^{-1} = 0 \) can be assumed.

However, in most CAD and simulation software, specific surfaces at the interface can be selected and thermally connected with couplings. This is one of the major advantages of Finite Element Analysis, used by most simulation software. Finite
Figure 5: Conductive coupling in a composite wall: (L) Snapshot showing the interface of the solar panel and the rails (structural frame) (R) Reduced thermal circuit.

Element Analysis is a numerical method in which a body is divided into smaller elements. The governing equations are then solved individually for each elements and then combined for the solution. Since the each element interacts only with its neighbour, the boundary conditions are more straightforward. [Reddy, 1993].

2.3 Thermal Loads

Thermal loads are of two kinds: Internal and External. Internal loads are due to heat dissipation in the electronics, thermal control or propulsion elements of the satellite. The external loads are due to the space environment. Importantly, thermal loads are measured as fluxes or intensities with units of $W/m^2$ for external loads, and as heating power with units of $W$ for internal loads.

2.3.1 External Loads

External loads due to incident radiations on the satellite that cause heating in the satellite. Thus, apart from aerodynamic heating at low altitudes, all heating in done due to incident thermal Electromagnetic (EM) radiations. These radiations are particularly in close to the Infrared (IR) and Microwave part of the spectrum. This is because all heating due incident radiation is dependent only on its thermo-optical property – absorptivity – which in turn, depends on the wavelength of the incident radiation. Most of known materials tend to absorb heat at higher wavelengths (IR region) only. The heat absorbed at other wavelengths on the spectrum are negligible. Thus, cosmic rays, which are mainly in the gamma spectrum can cause other damages, but do not heat the satellite significantly. Furthermore, for the particle plasma radiations also have negligible heating effects for the densities found in the vicinity of planets. Thus, space thermal engineers tend to classify the space thermal environment into three primary types [Gilmore and Donabedian, 2003]:
1. **Direct Solar:**
   This is the EM radiation directly from the sun. This radiation varies with the Earth’s position in its orbit around the Sun, varying from 1322 W/m² at the summer solstice to 1414 W/m² at the winter solstice. This is average to 1367 W/m² for 1 A.U. Notably, the variation of direct solar fluxes is negligible even for highly elliptical orbits like that of FS-2 [Gilmore and Donabedian, 2003].

2. **Planetary:**
   Each planetary body is thermal equilibrium with incoming radiation from the sun and emits its own blackbody radiation, usually in the IR region of the spectrum. This value for earth is estimated at 236 W/m² close to the surface. It varies inversely with the square of the distance from Earth α1/d². All other celestial bodies (including the moon) are too far away to have a considerable influence in Earth-bound orbits. Notably, Earth planetary radiations do not show large variations in orbits with same altitudes [Gilmore and Donabedian, 2003].

3. **Albedo:**
   Albedo is the reflected solar radiation from a planetary body. Since it is reflected by the body, it also varies inversely with the square of distance from the body, and disappears during eclipse times. Earth Albedo is highly variable even in orbits with the same altitude. This is due to several factors like clouds, surface density, etc. which causes variation in the reflectivity of Earth [Gilmore and Donabedian, 2003]. The common assumption made in books is to use a constant albedo and its variation of the albedo flux is purely based on the view factor due to cosine of the zenith angle of the satellite in orbit only. However, a more corrected estimate of variation of the Albedo is given by:
   \[ \rho(\epsilon) = \rho_{\epsilon=0} + C_4 \epsilon^4 + C_3 \epsilon^3 + C_2 \epsilon^2 + C_1 \epsilon, \]
   where, \( \epsilon \) is orbital zenith angle or Sun-Earth-satellite angle; \( C_4 = 4.9115 \times 10^{-9}, C_3 = 6.0372 \times 10^{-8}, C_2 = -2.1793 \times 10^{-5}, C_1 = 1.3798 \times 10^{-3} \) [Rickman, 2014]. The albedo variation and its corrected estimate is plotted in Fig 6. This shows the corrected albedo to have a slightly greater value.

2.3.2 **Internal Loads**

Internal heat loads are due to heat dissipation from electronic components. Apart from a small amount of heat dissipated by the antennas, most of the generated power is dissipated as heat during operation. As this is yet not fixed and varies with the specific times in the mission, it can considered a constant source for a worst case thermal condition [Meseguer et al., 2012].

---

\(^{1}\)Ratio of the reflected to incident solar radiation
2.4 Thermal Design Examples

2.4.1 Holistic Thermal Design: FASTRAC

FASTRAC, or 'Formation Autonomy Spacecraft with Thrust, Relative Navigation, Attitude, and Crosslink program' were a set of twin satellites designed by University of Texas, at Austin. The satellites, having a mass of 25 kg, fall under the microsatellite category. This was used as an important example to understand the thermal design procedure adapted from beginning to end for a low budget (0.1 M USD) satellite design, manufacturing and launch [Diaz-Aguado et al., 2006].

The satellite has mission requirements to launch into a LEO with altitudes of 300 to 700 km. Since using a passive thermal control system largely saves on costs, it was one of the mission design drivers. Initially, a thermal orbital analysis was done to identify the worst hot and cold cases of the orbital environment by the calculation of the thermal loads. The limiting subsystem temperature range was that of the COM module, limited to 5 to 65°C. Passive thermal control using black fiber glass and MLI blanket was then tested with a thermal simulation using ABAQUS (simulation software). After the validation through simulation, a thermal cycling test was carried out in a vacuum chamber. A cyclic temperature variation from -100 to 90°C was done and the COM was specifically studied for temperature limits [Dinh, 2012]. [Muñoz et al., 2012],[Campbell, 2006].

The summary of FASTRAC mission is shown in Table 2.

**Inference:** The general thermal design strategy at the beginning is to check the feasibility of implementing a passive thermal control system in the holistic design of small satellite. This saves on costs, manufacturing time and reduces risks due to reduced complexity of the system. This is done using a top-to-bottom approach,
especially if it is one of the design drivers of the missions, like the FS-2. This means that a holistic design is first tested, and then component level thermal control is to be designed.

### 2.4.2 Holistic Thermal Design: GTOSat

GTOSat is a CubeSat designed by NASA, developed on the Dellingr platform, to be launched into a GTO. The Dellingr X is a flight-ready platform to use for Cubesat missions. GTOSat has chosen the Dellingr 6U platform for its mission. Moreover, the region of space as well as the scientific requirements of the GTOSat and FS-2 show some similarities. Thus, it is relevant to be looked into.

The Dellingr CubeSat mission summary is presented in Table 3.

<table>
<thead>
<tr>
<th>Size</th>
<th>6U CubeSat</th>
</tr>
</thead>
<tbody>
<tr>
<td>Status</td>
<td>Launched, Operational (With minor failures)</td>
</tr>
<tr>
<td>Orbit</td>
<td>410-km LEO, Deployed from ISS</td>
</tr>
<tr>
<td>Thermal Control Type</td>
<td>Passive</td>
</tr>
<tr>
<td>Strategy</td>
<td>Coatings (region specific) and Louvers(Custom)</td>
</tr>
<tr>
<td>Ambient Range</td>
<td>0 to 65</td>
</tr>
<tr>
<td>Structure Material</td>
<td>Al 6061</td>
</tr>
</tbody>
</table>

Table 3: Dellingr CubeSat Summary.

The GTOSat is a 6U CubeSat that is proposed to be launched in 2021. The larger size allows a higher flexibility of implementing various control methods including those for thermal control. The salient features of the thermal design used are listed below [Yusupov, 2018], [Clagett et al., 2017]:

1. Interior coatings with low emissivity materials.
2. Regulated heat conduction from powered components to the base-plate radiating to space.
3. Metal behind solar cells was coated with high emissivity Teflon impregnated anodize.
4. Batteries use internal heaters and radiator.
5. The magnetometer was anodized for high emissivity.

6. Thermal louvers with bimetallic springs were implemented successfully, and passed the test of 12900 cycles at a range of 32 to 55 °C and 8 thermal-vacuum tests at a range from -20 to 85 °C.

**Inference:** Region specific thermal control in CubeSat is mainly done by coatings and surface finishes, modifying the thermo-optical properties as required. If excess heat needs to be rejected, heat regulation is a better option compared to a pure metallic connection.

A thermal louver will be available in the market, as an added option if the mission switches to a 6U CubeSat configuration. With flight heritage and high TRL (Technology Readiness Level), this is a useful option to have.

### 2.4.3 Specific Thermal Design: NiCd Battery

NiCd batteries are widely used in the space industry. The thermal designs used for it, describe the region specific design methodology. Smaller NiCd batteries are usually placed inside the spacecraft and painted with a black conformal coating to radiate out the waste heat generated by charge and discharge inefficiencies. However, in some cases, they maybe conducted out (regulated actively or passively), and use radiators and thermostatically controlled heaters to do so, as shown in Fig 7. This design is used for larger spacecraft missions, where the excess heat is conducted to the Aluminum fins (radiators), by semi-actively regulating its path. This is done by adding insulation around the path, forcing heat flow in the required direction. Patch heaters are placed on the other end of the batteries, allowing the thermal control in worst cold case situations.

![NiCd battery thermal design](image)

**Figure 7:** NiCd battery thermal design; Pg 77, [Gilmore and Donabedian, 2003].

**Inference:** The region specific thermal design is much more detailed than the holistic design - The analysis must account for most of the smaller components as well, which may have been excluded in the idealization process.

The major thermally sensitive avionic components are batteries, as they have a small temperature limit (operational) range. Active heating is usually done with the use
of patch heaters for batteries. Other options like a filament heater are less common. Also, the waste heat from the battery have other ways of being disposed, like the use of a passively regulated conduction path.

### 2.5 Thermal Control Methodology

In this section, some proposed missions and thermal control strategies will be studied and then the available thermal control methods and their utility for CubeSat missions will be pointed out. Current high altitude missions set to launch are listed in Table 4 [Blum et al., 2018], [Rahman, 2017], [Thomsen et al., 2015] [Maximenko and Niiler, 2006]. Here, passive thermal control strategy refers to modification of spacecraft thermo-optical properties with surface finishes or coatings. Thus, only radiative properties are affected. It is also important to note that these are thermal control strategies used for the Holistic design. Specific components may be controlled by active methods like heaters, thermal pipes, etc. as well.

<table>
<thead>
<tr>
<th>CubeSat</th>
<th>Orbit</th>
<th>Size</th>
<th>Thermal Strategy</th>
<th>Launch</th>
</tr>
</thead>
<tbody>
<tr>
<td>GTOSat</td>
<td>GTO</td>
<td>6U</td>
<td>Thermal Louvres</td>
<td>Dec ’20 or ’21</td>
</tr>
<tr>
<td>SpectroCube</td>
<td>GTO</td>
<td>3U</td>
<td>Passive</td>
<td>Dec ’20</td>
</tr>
<tr>
<td>Orbital Factory</td>
<td>GTO</td>
<td>1U</td>
<td>Insulation Coating</td>
<td>Dec ’19</td>
</tr>
<tr>
<td>LACCE</td>
<td>GTO</td>
<td>3U</td>
<td>NA</td>
<td>was ’18 (Delay)</td>
</tr>
<tr>
<td>ADE</td>
<td>GTO</td>
<td>1U</td>
<td>Passive</td>
<td>Dec ’19</td>
</tr>
<tr>
<td>Shields-1</td>
<td>GTO</td>
<td>3U</td>
<td>Shielding vaults</td>
<td>was ’18 (Delay)</td>
</tr>
<tr>
<td>mDot</td>
<td>HEO</td>
<td>6U</td>
<td>Insulation, Coating</td>
<td>Dec ’20</td>
</tr>
<tr>
<td>MarsCO</td>
<td>Mars (DS)</td>
<td>6U</td>
<td>Insulation, Radiator</td>
<td>May ’18</td>
</tr>
</tbody>
</table>

Table 4: List of current and future CubeSat missions to the GTO, HEO and deep space (higher altitude missions than LEOs).

**Inference:** Most CubeSat missions use passive control strategies, as it saves on cost and complexity. This is also hints that a GTO CubeSat mission with passive thermal control is feasible.
The major thermal control methods and their availability is listed out in Table 5.

<table>
<thead>
<tr>
<th>Technology</th>
<th>TRL</th>
<th>Availability</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Passive Systems</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>MLI Blanket</td>
<td>9</td>
<td>MODERATE</td>
</tr>
<tr>
<td>Thermal coatings (Tape)</td>
<td>9</td>
<td>MODERATE</td>
</tr>
<tr>
<td>Sun Shields</td>
<td>7</td>
<td>LOW</td>
</tr>
<tr>
<td>Thermal Finishes</td>
<td>9</td>
<td>MODERATE</td>
</tr>
<tr>
<td>Metal Thermal Straps</td>
<td>9</td>
<td>MODERATE</td>
</tr>
<tr>
<td>Composite Thermal Straps</td>
<td>7</td>
<td>MODERATE</td>
</tr>
<tr>
<td>Passive Thermal Louvers</td>
<td>9</td>
<td>LOW</td>
</tr>
<tr>
<td>Deployable Radiators</td>
<td>6</td>
<td>LOW</td>
</tr>
<tr>
<td>Passive Heat Pipes</td>
<td>6</td>
<td>LOW</td>
</tr>
<tr>
<td>Storage Units</td>
<td>8</td>
<td>LOW</td>
</tr>
<tr>
<td><strong>Active Systems</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>FEATS</td>
<td>6</td>
<td>LOW</td>
</tr>
<tr>
<td>Electrical Heaters</td>
<td>9</td>
<td>LOW</td>
</tr>
<tr>
<td>Mini Cryocoolers</td>
<td>6</td>
<td>MODERATE</td>
</tr>
<tr>
<td>Patch heater</td>
<td>9</td>
<td>HIGH</td>
</tr>
<tr>
<td><strong>Emerging systems</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fluid Loops</td>
<td>3</td>
<td>LOW</td>
</tr>
<tr>
<td>Deployable Passive Radiators</td>
<td>5</td>
<td>LOW</td>
</tr>
</tbody>
</table>

Table 5: List of available thermal control methods with their respective TRL (Technology Readiness Level) and availability [Yusupov, 2018].

This provides a basic idea of the available thermal control methods to choose from. The availability is based on the ease of acquiring the item, and is based on the current demand and the number of companies selling high TRL products commercially.
3 Implementation

3.1 Layout
The focus of this thesis is to perform a comparative thermal study of the LEO-GTO environments through simulations. The results of this study will be used to comment on the feasibility of the launching a 3-U CubeSat into certain higher altitude orbits – with a focus on GTOs. This comparative study is done using the Foresail missions as an example as their mission aims fit the points of comparison aptly.

The brief layout of the procedure employed for this thermal study is as follows:
The thermal requirements are first tabulated in section 3.2. Then, the thermal environments in space are studied, which is followed by the comparison of thermal loads of the two missions (compared in MATLAB) in section 3.4. This is undertaken to understand and distinguish between the thermal scenarios of both missions. This is followed transient thermal simulation of the CAD model, explained in section 3.5.

3.2 Thermal Requirements
Thermal requirements are defined by the survival and operational (for active avionic components) temperature limits of the components. They are also driven by specific thermally sensitive components only. These include avionics, propulsion and other active components (if applicable). The thermal requirements of the FS missions are listed in Table 6.

<table>
<thead>
<tr>
<th>Component</th>
<th>Temperature Limits</th>
</tr>
</thead>
<tbody>
<tr>
<td>Battery</td>
<td>-5 to 25</td>
</tr>
<tr>
<td>Antennas</td>
<td>-80 to 120</td>
</tr>
<tr>
<td>Solar Arrays</td>
<td>-150 to 180</td>
</tr>
<tr>
<td>Onboard Electronics</td>
<td>-25 to 50</td>
</tr>
<tr>
<td>Payloads</td>
<td>-10 to 35</td>
</tr>
<tr>
<td>Magnetotorquers</td>
<td>-10 to 50</td>
</tr>
</tbody>
</table>

Table 6: Thermal Requirements (All temperatures in °C).

3.3 Orbit Definition
To compare the LEO-GTO thermal environments, the primary focus of the thesis, the following orbits were used: A polar sun-synchronous, near-circular LEO used by FS-1; A highly elliptical GTO used by FS-2. The orbit parameters for the FS mission comparisons are tabulated in Table 7 [Yusupov, 2018].

The aforementioned orbits are illustrated in Fig 8. The images were obtained from the NX’s orbit propagator.
Table 7: Orbital parameters of FS

<table>
<thead>
<tr>
<th>Orbital Parameter</th>
<th>LEO(FS-1)</th>
<th>GTO(FS-2)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Apogee Altitude</td>
<td>400 km</td>
<td>35786 km</td>
</tr>
<tr>
<td>Eccentricity (e)</td>
<td>0</td>
<td>0.73</td>
</tr>
<tr>
<td>Inclination (i)</td>
<td>96.2°</td>
<td>27°</td>
</tr>
<tr>
<td>Argument of Periapsis</td>
<td>90°</td>
<td>180°</td>
</tr>
<tr>
<td>RAAN</td>
<td>0°</td>
<td>7.5°</td>
</tr>
</tbody>
</table>

Figure 8: (L) FS-1’s polar sun-synchronous orbit; (R) FS-2’s highly elliptical GTO. Points marked in yellow color are the orbital positions used for transient thermal simulations in the subsequent section.

3.4 Thermal Load Analysis

In this section, the thermal loads of the Foresail (FS) missions are calculated and then plotted for comparison. The thermal load variation for each face plate of the 3U CubeSat is shown in the Figs 9.

In Fig 9(a), the navy blue line (on top) shows the thermal loads on Solar pointing plate, which is constantly receiving direct solar flux; the yellow line shows the thermal loads on Earth(Nadir)-pointing plate from Earth’s radiations which is constant for an inertially stable satellite(in a circular). There is negligible Albedo in this case as the satellite orbit is inclined at 91°, positioning it at the start of the eclipse cone; the blue line shows the thermal loads on the plate pointing in the direction normal to the sun and position vector, which varies as a cosine function, as it is cyclically visible to the Earth. The purple, red and green lines show the thermal loads on the plates opposite the Earth-pointing, Sun-pointing and Normal-pointing plates - They receive no thermal loads during the orbit.

In Fig 9(b), the blue line shows the thermal loads on the plate pointing to the Earth. It has negligible loads for most of the orbit due to a large distance from the Earth, except when in eclipse (towards the end); the orange line shows the thermal loads on the plate opposite to the nadir pointing plate, which peaks when it has solar...
and albedo loads; the yellow line shows the thermal loads on the plate in the direction of the orbital normal. It does not face any source directly and thus has low thermal loads - essentially making it a radiator plate; the purple line shows the thermal loads on the plate opposite to orbital normal plate, which reaches a peak similar to the orange line, when the sum or solar and albedo loads are maximum. However, it is much lower as it faces the sources at a smaller angle in comparison; The green line shown the sum of thermal loads on the other two plates (along velocity). This is done as the thermal loads on these plates were complimentary to each other, making the sum easier to represent.

Both plots use different references to describe the plates as using the sun direction is more useful in the case of FS-1, as its reference direction is constant in a geosynchronous LEO. Both the LEO and GTO have distinct peaks when the satellite is mid-way in the orbit (orbit propagation starts from True Anomaly = 0). However, we are more concerned with the total thermal loads on the satellite, which are discussed in sub-sections 3.4.0.1 and 3.4.0.2.

3.4.0.1 Comparison

The net thermal loads (sum of the thermal loads on all surfaces) of both orbits are compared in Fig 10. This comparison is done for the duration of orbital period of the FS-2 – as this period is much larger than that of FS-1 – making it a more logical comparison.

In Fig 10, the purple line shows the net thermal load variation in the LEO; and the yellow line shows the net thermal load variation in the GTO. The GTO has a much larger magnitude of variation as seen, but has a much lower average of net thermal load (1240.2 W/m$^2$) as compared to that of the LEO (1980.8 W/m$^2$). The peak total load of the LEO (2231.4 W/m$^2$) is also greater than that of the GTO (1847.4 W/m$^2$). This suggests that the GTO is a much colder environment.
3.4.0.2 Inference

It is conspicuous that FS-2’s GTO has a much larger variation in the thermal loads throughout the orbit and much lower average thermal load (averaged over the whole orbit). Notably, the variations in the GTO are much slower (with respect to time) and smoother. Thus, the plot does not necessarily mean that larger thermal stress in induced in due to FS-2’s orbital environment as the larger variation in magnitude is compensated with slower changes. However, and more importantly, the GTO is definitely a much colder environment as compared to FS-1’s polar LEO. This results have also been verified in Appendix B using a steady state thermal simulation in STK (Systems tool Kit).

3.4.0.3 Internal loads

Internal loads were assumed to be a 7 W for the simulations (carried out in the later sections). This is the initial estimate provided, as is considered a constant source for both, to study a worst-case condition and to avoid complexity of implementing accurate thermal loads of the step form. Moreover, apart from minimal power being radiated out through the antenna, all the power that is consumed is dissipated as heat. The only exception to this case is when the battery is charging.
3.5 Simulation Setup

3.5.1 Summary

All simulations in sections 4, 5 and 6 were carried out in Siemens NX’s Spacecraft Thermal Systems’ software environment. The software was chosen over its competitors ANSYS and ESATAN due to its ability to calculate and input orbital thermal loads with its own orbit propagator, and to solve transient simulations effectively. Also, the degree of the detail is high for complex simulations, making it an apt choice for many engineering design companies.

The simulation setup includes CAD modeling, Idealization, Meshing and Thermal Property Definition, Thermal loading, Thermal couplings and Constraints and Solver setups.

3.5.2 CAD Modelling

Illustrated in Fig 11 is the detailed CAD model of the FS-1, and the engineering model prototype [Praks, 2018]. It consists of two Rails, that constitute the structural frame of the Satellite bus. It encompasses three ‘box’ structures: Two payloads (PATE and MASTER) and the Avionics box. Since the payloads are designed by separate entities, it is assumed to be thermally robust. Moreover, it does not contain any batteries. Notably, electronics are the most sensitive elements on the satellite, the battery being the most sensitive of them. The satellite is then covered with on the four long surfaces with solar panels on each side, and metallic (Aluminum) top and bottom plates. This model was developed in SolidWorks\(^2\) and imported into Siemens NX.

3.5.3 Idealization

Idealization of the CAD model pertains to removal of certain details to run the simulation reasonably. This means that the simulation should not exceed a certain number of iterations (in this case, 10000) or time (a few hours) for the solution to converge.

Idealization is an iterative procedure – The desirable solutions must be as accurate and as reasonable as possible. The process follows the primary principle – The processing time decreases with the increase in the size and complexity of the elements. Hence, the primary aim is to reduce features in the CAD model that drastically decrease the size of elements. The important idealizations made are listed below:

- The structural frame (Rails) were divided into simpler cuboidal elements, to allow simple 2D meshing.
- All screw holes were removed.
- All PCBs, when used, were blank. (All modelled electronics on it were excluded).

\(^2\)The CAD model was not developed as part of this thesis work.
• The Payloads were modelled as simple cuboids.

• Isolated elements such as the antennas are removed and can be simulated separately.

3.5.4 Meshing and Material Properties

Most of the elements were meshed as 2-D (quad) elements. The four-noded elements are best suited for rectangular surfaces, which are in abundance in the CubeSat. 2-D meshing was used as in most cases the temperature the thickness dimension was considerably small compared to the other dimensions (< 10%). Thus, the temperature variation along this dimension can be neglected. However, the mesh is accurately representative of the thermal inertia of the right thickness. This is because the 2-D meshes need a thickness definition to accurately represent a body. Notably, all 2-D meshes were modelled as center surfaces - A plane parallel to the major area is created in the middle (at half the thickness). Only the internal modules are modelled with a 3-D mesh to study the temperature distribution in all three directions. The meshed model is shown in Fig 12.

The CubeSat is primarily made of Aluminum. Al - 7075 Alloy was used for simulations, but most of the other space-grade alloys have similar thermal properties as well - Giving it a potential to be used. The magneto-torquer coils are made of copper, and
all PCBs are made of FR4\(^3\), whereas the cells are made of GaAs (Gallium Arsenide Triple Junction). The thermal property overview of the materials in use is given in Table 8. [Azar and Graebner, 1996], [MatWeb database, a], [MatWeb database, b], [IOFFE database, ].

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Solar Cell</td>
<td>GaAs</td>
<td>5317</td>
<td>5.7</td>
<td>0.056</td>
<td>330</td>
</tr>
<tr>
<td>2</td>
<td>Elec. boards</td>
<td>FR4</td>
<td>1850</td>
<td>2.4</td>
<td>0.29 to 0.343</td>
<td>1330</td>
</tr>
<tr>
<td>3</td>
<td>Frame</td>
<td>Al 7075</td>
<td>2700</td>
<td>23.6</td>
<td>130</td>
<td>960</td>
</tr>
<tr>
<td>4</td>
<td>Coils</td>
<td>Copper</td>
<td>8834</td>
<td>16.4</td>
<td>385</td>
<td>387</td>
</tr>
</tbody>
</table>

Table 8: Thermal Properties of materials used.

3.5.4.1 Radiation

Only the Aluminum on the frame (rails, top and bottom plates- shown in Fig 11), the Solar panel PCBs and the Solar cells are exposed to the external environment. The thermo-optical properties of the solar cells are fixed, but the other those of the other two materials can be altered by choosing the thermal coating or surface finishes on their respective surfaces. This can be an iterative procedure as the aim is to change the thermo-optical properties to optimally maintain the ambient temperatures. It was decided that a Kapton coating (tape) will be used on the solar panels and the type of surface finish of the Aluminum will be sought through simulations. These are tabulated in Table 9 [Gilmore and Donabedian, 2003].

\(^3\)A composite material composed of woven fiberglass cloth with an epoxy resin binder
<table>
<thead>
<tr>
<th>Component</th>
<th>Coat</th>
<th>Emissivity</th>
<th>Absorptivity</th>
</tr>
</thead>
<tbody>
<tr>
<td>Solar Cell</td>
<td>-</td>
<td>0.92</td>
<td>-</td>
</tr>
<tr>
<td>Elec. boards</td>
<td>Kapton</td>
<td>0.65</td>
<td>0.82</td>
</tr>
<tr>
<td>Frame</td>
<td>Black Anodized</td>
<td>0.31</td>
<td>0.45</td>
</tr>
</tbody>
</table>

Table 9: Thermo-optical Properties of materials used.

### 3.5.5 Thermal Couplings

Thermal coupling calculation methods were shown in the earlier sections. In most cases, the interface was in between the same material. Assigning the thermal couplings in these cases is fairly straightforward - Material thermal conduction at the interface area only.\(^4\).

On the other hand, in the case of dissimilar metal interface, thermal couplings are needed to be calculated manually. As explained before, it is simpler to assign a thermal coupling by calculating the heat transfer coefficient than the net conductance. The thermal couplings for major interfaces are tabulated in Table 10.

<table>
<thead>
<tr>
<th>Interface</th>
<th>Value (W/m(^2)K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Copper - FR4(Coil-PCB)</td>
<td>58.1</td>
</tr>
<tr>
<td>Al-FR4(Panel-Frame)</td>
<td>13.4</td>
</tr>
<tr>
<td>GaAs-FR4(Cell-Panel)</td>
<td>&lt;0.01</td>
</tr>
</tbody>
</table>

Table 10: Thermal Coupling Estimates for small, dissimilar interfaces.

It can be seen than most of these couplings have a fairly small value as atleast one of the two materials are poor conductors. Thus, the main heat flow is through the metallic structure and the casing of the internal modules.

### 3.5.6 Thermal Loads

Thermal loads have been discussed in detail in the previous section and earlier in this section. However, for the simulation, the thermal loads are calculated by the software. This is much more accurate than manually assigning the thermal loads as the software obtains the loads from an internal thermal model (based on ESATAN) after propagating the orbit.

### 3.5.7 Constraints

In this case, the constraints is solely to assign the temperature of the surrounding thermal reservoir (space vacuum). Thus, a uniform and constant temperature of 2.7 K is assigned to the surroundings (vacuum).

\(^4\)Assuming there is a perfect thermal contact for ideal cases
3.5.8 Solver Setup

The solver parameters are summarized in Table 11. Notably, all subsequent simulations involving orbital loads used the same setup.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Comment</th>
</tr>
</thead>
<tbody>
<tr>
<td>Software</td>
<td>Simens NX</td>
</tr>
<tr>
<td>Solver (Environment)</td>
<td>Spacecraft Systems Thermal</td>
</tr>
<tr>
<td>Type of Simulation</td>
<td>Transient</td>
</tr>
<tr>
<td>Orbital Positions</td>
<td>20</td>
</tr>
<tr>
<td>Simulation Time</td>
<td>&gt;2 Orbital periods</td>
</tr>
<tr>
<td>View factor solver</td>
<td>Monte-Carlo</td>
</tr>
<tr>
<td>Initial conditions</td>
<td>290 K (uniform)</td>
</tr>
</tbody>
</table>

Table 11: Simulation Setup: Solver parameters.
4 Thermal Simulation Results: FS-1 vs. FS-2

This section illustrates, describes and discusses the results of simulating the FS-1 CubeSat structure in the two FS mission orbits. This comparative study will also form the basis of evaluating the feasibility of launching a 3U CubeSat into a GTO – The thermal feasibility being an important driver.

4.1 Summary

The thermal simulation results validate the hypothesis made in the earlier sections – The GTO has a colder environment and has a gradual change between the extremes. FS-1 mission, when sun-synchronous, has no eclipse time and is constantly closer to the Earth, having large heating loads. However, there is a possibility of a small eclipses to occur in the other options of mission orbits to be considered. Thus, a large emissivity, lower absorptivity surface finish will suffice to maintain the ambient temperature in favorable ranges. The results of the polar orbit show that the satellite can reach a steady soon as the input and output thermal energies are essentially the same through the orbit (except for orbital decay and perturbations, which can be considered negligible for a short mission).

FS-2 is colder mainly because of its the combined lowering of incident fluxes due to view factors and distance from the Earth. The latter reduces albedo and Earth IR loads considerably, which is a significant part of the total thermal loads. The ambient temperatures are maintained by slightly increasing the absorptivity of the surface finish. Notably, there is no drastic decrease in temperatures – Making a partially passive thermal [Specific elements of the satellite like the Battery may require active heating] control system feasible for the satellite. FS-2 has a mean eclipse times of approximately 31 minutes.

4.2 Results

4.2.1 Foresail-1

The results are of three main categories based on what is being studied: Temperature distribution; Element-wise time variation plots of temperature and total thermal loads; and studying the times at minimum and maximum temperatures for different regions of the satellite.

4.2.1.1 Temperature Distribution Profiles

The temperature distribution for a time-invariant - In some cases, like a polar sun-synchronous LEO, the profile becomes time invariant – Having reached a pseudo steady state] profile is shown in Fig 13.
Figure 13: Temperature Distributions Profiles of FS-1: (L) Exterior view (R) Interior View. Ambient range within acceptable limits.

4.2.1.2 Element-wise time variation

Six elements (One element on each external face of the CubeSat) were chosen for both (FS-1 and FS-2) simulations. The temperature and total flux variations on these elements were then plotted to study the trends of variations throughout the iterations (throughout the simulation time). The elements selected for plotting are displayed in Fig 14.

Figure 14: Elements chosen in each external face of the CubeSat.
### 4.2.1.3 Global Legend for element-wise time iteration plots

Element-wise time iterations of either temperatures or thermal loads are plotted to study the simulation results in the latter parts of sections 4, 5 and 6. In each case, the objective is to plot and compare the variation on the temperature or thermal loads on each of the six surfaces of the satellite, over time. In each of the plots, the Red line shows the variation on the bottom plate; the Blue line shows the variation on the top plate; the Green, Yellow, Purple and Pink lines show the variation over the other four surfaces with mounted solar panels.

For the LEO simulations in sections 4 and 5, the Purple line shows the variation on Sun-pointing plate; the Green line shows the variation on the plate that opposite to the Sun-pointing plate; the Yellow line shows the variation on Earth-pointing plate; and the Pink line shows the variation on the plate that opposite to the Earth-pointing plate.

The time variation of the temperatures of the six elements are shown in Fig 15. The description of the plots are given in sub-section 4.2.1.3.

**Inference:** In Fig 15, it can be clearly seen that all the elements are reaching steady state quickly. The purple line, showing the element facing the sun reaches the highest steady state temperature (59.6°C) and the element facing in the opposite direction, reaches the lowest steady state temperature (-8.9°C). The nadir and normal facing surfaces have a more moderate temperature value. As the steady temperatures are reached within a few orbits, thermally, the case of the FS-1's polar LEO is the same as a stationary satellite at a 400 km altitude, with one surface always facing the sun.

The time variation total thermal fluxes of the six elements are shown in Fig 16. The

---

**Figure 15:** Element-wise time variation of temperatures in FS-1; Temperature (y-axis) is plotted vs time(x-axis).

**Inference:** The thermal loads show little to no variation with time. The highest
absorbed flux is on the sun-facing side, with a constant 850 W/m$^2$. Apart from that plate and the nadir plate, the other plates of the satellite are not illuminated. The small variations shown maybe due perturbations or propagator convergence errors.

Figure 16: Element-wise time variation of total loads in FS-1; Thermal loads (y-axis) is plotted vs time(x-axis).

4.2.1.4 Time at Maximum and Minimum Temperatures

The software is also able to plot the time at minimum and maximum temperatures reached. This is important to realize the time at which specific thermal control methods might be required. Since the internal temperatures are closer to the minimum limits of the avionics than the maximum, studying the times at minimum is pertinent to our case. This is compared for both cases in sub-section 4.2.2.3.

4.2.2 Foresail-2

Results are presented in a similar way as that of of FS -1 to facilitate the comparative study.

4.2.2.1 Temperature Distribution Profiles

The temperature distribution profiles for FS-2 is shown in Fig 17. In this case, there is no time-invariance and thus, the temperature distribution of the last iteration (20th orbital position is shown). The variation of temperatures can be studied with time varying plots in the subsequent section.
4.2.2.2 Element-wise time variation

As shown for FS-1, the same chosen (six) elements are used for plotting the variation of temperatures and total fluxes across the iterations of the simulations (time variation). The time variation of the temperatures of the six elements are shown in Fig 18. The description of the plots are given in sub-section 4.2.1.3.

**Inference:** Fig 18 shows that there are larger but gradual change in the temperatures. In this case, the surface opposite the Earth-facing (Nadir) is illuminated by the sun the most. The temperature of this element (Blue Line) peaks when it at the aphelion of orbit. However, at this time, the thermal loads on the other surfaces are lower due to the distance from the earth. Thus, the other temperatures have low temperatures. The element facing the earth (green line) shows a peak temperature slightly before the aphelion, when the albedo loads peak.

The temperatures vary in cyclic manner, according to the orbital period. Thus, there is not net heating or cooling after each orbit.

The time variation total thermal fluxes of the six elements are shown in Fig 19. The description of the plots are given in sub-section 4.2.1.3.

**Inference:** Fig 19 shows that the basic trends followed by the element temperatures their respective loads are similar. The navy blue line (top plate element) and orange line (element on plate opposite the direction of velocity) are not illuminated in the orbit except for small peaks during the eclipse from Earth’s IR loads. The Green line (element on nadir facing plate) peaks at just before eclipse when the sum of the earth loads and the albedo loads is the highest. This is so because the albedo loads
Figure 18: Element-wise time variation of temperatures in FS-2; Temperature (y-axis) is plotted vs time (x-axis).

disappear during eclipse and is most of the orbit as it is too far from the Earth. The purple line (element on plate facing the direction of velocity) peaks right after the eclipse for the same reason. The total fluxes follow the same trend as the temperature profiles (It is the cause the temperature profiles follow the trend). There are no observable discrepancies in the thermal loads.

Figure 19: Element-wise time variation of total loads in FS-2; Thermal loads (y-axis) is plotted vs time (x-axis).

4.2.2.3 Time at Maximum and Minimum Temperatures

The time at minimum or maximum is fairly irrelevant for a steady state case, as the temperature soon becomes invariant. However, in the case of FS-2, where the
tendency of the satellite is to cool, it is important to note the time at which minimums can occur. The spatial distribution of times at minimum for the case of FS-2 is shown in Fig 20.

**Inference:** The times shown in orange/red are close to eclipse end times. This shows that the satellite reaches its minimum temperature towards the end of the eclipse. This is due to the thermal inertia of the satellite, giving it some time to cool down through emitting radiations.

### 4.3 Discussion

The important comparison parameters of FS-1 and FS-2 are summarized in Table 12. Notably, the thermo-optical property ratios are slightly different. This slight change was required to get a desired result of similar temperature ambient ranges.

<table>
<thead>
<tr>
<th>CASE</th>
<th>Unit</th>
<th>FS-1</th>
<th>FS-2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Min. Temp</td>
<td>°C</td>
<td>73.87</td>
<td>68.76</td>
</tr>
<tr>
<td>Max. Temp</td>
<td>°C</td>
<td>-18.27</td>
<td>-29.55</td>
</tr>
<tr>
<td>Ambient Range</td>
<td>°C</td>
<td>9.23-16.84</td>
<td>9.36-17.32</td>
</tr>
<tr>
<td>Abs/Emm</td>
<td>-</td>
<td>0.65/0.82</td>
<td>0.85/0.8</td>
</tr>
</tbody>
</table>

Table 12: Simulation Result Summary.

This set of simulations show that the GTO is different from a the FS-1’s LEO, but not significantly. The maximum and minimum temperatures show that the
GTO is definitely a colder environment. Even though the direct solar load is almost constant (with small eclipse orbits), the distance from the Earth significantly reduces the albedo and planetary fluxes form the Earth. Thus, a slightly higher absorbing material is required to ensure maximum intake of heat from the incident fluxes. The LEO, even with an eclipse has an advantage of fast variation of loads, and decreasing the chances of failure as the particular thermal scenario is constantly changing (For example, an eclipse cannot last for long. Thus, surviving an eclipse in an LEO is easier). However, the GTO has the advantage of slow variations of thermal loads to minimize thermal stresses. This minimized damage to avionic solders and other parts of the satellite with large thermal coefficients [Yusupov, 2018]. Moreover, the ambient ranges suggest that the 3U CubeSat can be pseudo-passively controlled (thermally), opening the possibility of launching a 3U CubeSat into a GTO if so required.
5 Specific Mission Scenarios: FS -1

This section investigates specific mission scenarios of the Foresail missions. Two primary scenarios are analyzed for FS-1: The effects of spinning in orbit; and a thermal solution for the batteries in the Avionics casing. The thermal investigations, like the rest of the report, be through comparative analysis with the closest ideal situation.

5.1 Effect of Spinning

5.1.1 Background

FS-1 has two payloads. The secondary payload is a plasma brake instrument and an experiment to measure the coulomb drag force in orbit. This instrument is essentially a 300 meter-long thin tether that is unwound from the satellite as it spins. In the process of unwinding, the satellite spins about an axis pointing to the sun, ensuring that at least one panel is always sun-pointing. This is partly a scientific requirement of the mission. This section investigates the thermal effects of the spinning of the satellite during this process.

5.1.2 Simulation Setup

To appropriately compare the results, similar thermal models are used, with and without spinning of the satellite. Thus, the FS-1’s thermal simulation parameters are used and re-run with a single attitude constantly changing. The spin rate is set to two complete rotations per orbit about the sun-pointing vector. The thermo-optical properties and materials remain the same.

5.1.3 Results

Temperature distribution profiles and load variations over time are studied.

5.1.3.1 Temperature Distribution Profiles

The temperature distribution profiles for FS-1 in spinning is shown in Fig 21. The temperature reach near equilibrium, like in FS-1. The profile shows the distribution for the final orbital position.

5.1.3.2 Element-wise time variation

The time variation total thermal fluxes of the six elements are shown in Fig 22. Fig 22 shows that the description of the plots are given in sub-section 4.2.1.3.

**Inference:** The total flux on the sun-facing surface (orange line) is constant. The total flux on the element on the surface opposite the sun-facing surface (blue line) is constant, and equal to 0, as it receives no thermal loads (like the original FS-1 orbit). However, due to the spinning of the satellite, the thermal fluxes on the
remaining surfaces vary from 0 to nearly 200 W/m$^2$ cyclically with rotation (two full rotations per orbit). However, the total flux incident on the satellite (sum of fluxes on all 6 surfaces combined) remains the same for spinning and normal operation.

Figure 22: Element-wise time variation of total loads in FS-1 (Spinning mode).
5.1.3.3 Discussion

The major results of the FS-1 thermal simulation of the orbit with the spinning CubeSat are summarized in Table 13. The results indicate the minimum exterior temperatures are reduced and the ambient range in increased slightly.

<table>
<thead>
<tr>
<th>CASE</th>
<th>Unit</th>
<th>FS-1</th>
<th>FS-1 Spin</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max. Temp</td>
<td>°C</td>
<td>73.87</td>
<td>73.87</td>
</tr>
<tr>
<td>Min. Temp</td>
<td>°C</td>
<td>-18.27</td>
<td>-22.25</td>
</tr>
<tr>
<td>Ambient Range</td>
<td>°C</td>
<td>9.23-16.84</td>
<td>9.37-17.24</td>
</tr>
<tr>
<td>Abs/Emm</td>
<td>-</td>
<td>0.65/0.82</td>
<td>0.65/0.82</td>
</tr>
</tbody>
</table>

Table 13: FS-1 Simulation Result Summary: No spinning vs Spinning.

The net thermal load on the satellite remains the same. However, spinning causes the loads to get distributed over multiple surface rather than stay nearly constant at a few surfaces of the CubeSat. The emitted radiation remains the same as the all surfaces emit the same amount of radiation, unless their thermo-optical properties are changed. However, the holistic heating of the satellite is in rapid changes, causing the extremity of the temperature ranges to decrease. This is similar to the case of difference between a stationary object receiving heat from two directions, and the same object moving around the heat source to cause cyclic heating. The object will be slightly colder in the latter case as it does not stay in one position (or orientation) long enough to absorb considerable heat.

The ambient range is increased due to the same reason - The same maximum exterior temperatures suggests that the net heating effect is the same due to same thermal loads. However, the distributed loads gives an overall higher input load to the interior. These temperatures are retained by the interior due to thermal inertia.

In summary, though the cases of spinning is thermally different, it does not pose as a threat for the CubeSat operations. This is evident from the small magnitude of the difference in results.

5.2 Thermal Solution: Battery

The battery (Li ion) is the sole thermally sensitive avionic component of the satellite. This is usually the case in most missions as the temperature ranges of the on-board electronics is much more robust, with the exceptions of missions with propulsion systems - Some propellants like hydrazine have small temperature ranges (freezing point = -2 °C).

To ensure the optimal operations of the battery, it must operate in positive temperature ranges (0 to 25 °C) [Karam, 1998]. A thermal solution using a Kapton tape insulation and a patch heater was tested with a worst thermal case simulation.
5.2.1 Background

The batteries are mounted on PCB which is placed on the lowest bracket on the avionics module. The batteries are attached to the PCB with metal straps for power connections and then soldered. The bracket itself cannot have a slots (cutouts) made for the batteries to stay in place. This design was tried and excluded in as the batteries were short circuited and physically damaged in the previous vibration test. Hence, the slots were removed.

5.2.1.1 Problem Recognition:

A thermal solution need to be designed to maintain the satellite at optimal operational ranges in the case of an eclipse. This has be done considering the direct contact to the metal (lower bracket), which is transporting the heat to be dissipated out.

5.2.1.2 Solution:

Removal of the slots made the metal in contact with the batteries thinner. The thermal solution is depicted in Fig 23. The configuration consists of two added layers of insulation (above the patch heater and the below the batteries). This isolates the heat of the battery to a large extent and avoids heat losses to the metal. A patch heater is added on top of the batteries (Cu wire patches) to switch on the active heating in worst conditions.

Figure 23: Thermal Solution: Configuration depicting the layers of material setup for the thermal simulation of the battery.
5.2.2 Simulation Setup

5.2.2.1 Objective

To check whether the battery temperatures are within allowable ranges for the duration of the eclipse for the assumed worst case loads and constraints.

5.2.2.2 Worst case definition

A worst case temperature of the -20 °C was assumed on the metallic bracket (made of Al 7075). The metal was also assumed to maintain the temperature, as most of the heat gotten in transport to the exterior of the satellite to be dissipated through radiation. Thus, it acts as a thermal reservoir. The simulation was run for 1000s, assuming this to be an eclipse. These extreme measures were simulated to consider safety margins.

5.2.2.3 Thermal load

The patch heater was constantly supplying 1 W power. This heat load is supplied uniformly over the area of the patch.

5.2.3 Results and Discussion

The temperature variation of the batteries with time and the distribution profile is shown in Fig 24. For the duration of the simulation, the battery temperature does not dip below 0 °C. It is can be seen that 1 W of power is sufficient to keep the battery at in the required positive range for the duration of 1000 s.

Figure 24: Battery thermal simulation results: (L) Temperature distribution profile at the end of the simulation (R) Temperature variation of the various entities over 1000s. Here, the blue plot represents the metallic bracket temperature (reservoir) - Constant at - 20 °C; the pink plot represents the battery temperature - Lowest 0.83 °C; and the red plot represents the maximum temperature of the test setup.
6 Specific Mission Scenarios: FS -2

Following the theme of the previous section, a few FS-2 mission scenarios are investigated in this section: The effect of adding deployable solar panels; Effect of eclipse occurring on the far side of the orbit; and the evaluation of a MEO mission.

6.1 Effect of deployable solar panels

6.1.1 Background

FS-2 will have similar payloads as FS-1. The scientific objectives may differ based on the region of space being analyzed. However, there is one major difference in FS-2 – Unlike FS-1, FS -2 cannot rely on magnetic torquer coils for unreeling the tether for the secondary payload or attitude control otherwise. This is because the distance of the GTO places the satellite much further away from Earth, making Earth’s magnetic field much weaker at most positions in the orbit. Thus, it cannot be used by the coils for any control of the CubeSat’s attitude. This adds a propulsion system requirement for the platform of the satellite for the mission. Thus, a large power requirement is added to the already tight power budget. This might require additional power generation and thus, the thermal implications of the addition of a set of deployed solar panels was studied.

6.1.2 Simulation Setup

Like the previous studies, similar thermal models were used to accurately compare the effect of adding a solar panel. The CAD model was modified to add two deployed solar panels held to the main body by four hinges each. Each solar panel was roughly the same size as the body mounted panels and meshed with the same material. The exposed rear part of the solar panel was coated with Kapton to avoid overheating of the panels during operation. The hinges were made of Teflon, one of the few space-grade plastics, that isolate the heat from the solar panels well [Silverman, 1995].
6.1.3 Results

Temperature distribution profiles and load variations over time are studied for FS-2 with deployed solar panels.

6.1.3.1 Temperature Distribution Profiles

The temperature distribution profiles for the FS-2 thermal model with added solar panels is shown in Fig 25. The time variations of the temperatures are similar to the FS-2 orbit, and hence has a cyclic variation. The temperature profile at the last iteration of the simulation is shown.

![Temperature Distributions Profiles of FS-2 with deployable solar panels: (L) Exterior view (R) Interior View.](image)

Inference: There is much larger total flux, as a much larger area faces the sun directly. The total flux average is increased to 1823 W/m², which is closer to the FS-1 polar sun-synchronous orbit. It is also important to note that the total heat fluxes on the shadowed surface is reduced from an average of 620 W/m² to a negligible amount.

6.1.3.2 Element-wise time variation

The time variation total thermal fluxes are shown in Fig 26. The load variation on an element placed on the +X surface (surface shadowed by the deployed solar panel) is also plotted. The average total flux of this surface is 8 W/m² which is negligible. Inference: There is much larger total flux, as a much larger area faces the sun directly. The total flux average is increased to 1823 W/m², which is closer to the FS-1 polar sun-synchronous orbit. It is also important to note that the total heat fluxes on the shadowed surface is reduced from an average of 620 W/m² to a negligible amount.
6.1.3.3 Discussion

The major thermal parameters are tabulated in Table 14. The increase of the maximum temperatures is attributed by the net increase in the thermal loads. The deployment causes a significant decrease of net thermal loads on the body mounted surfaces. However, the ambient range is increased. This can be because the hinges are partially, not completely thermally isolating the deployed panels from the satellite. Decreasing the number of hinges in the possible design and adding a surface finish with a low $\alpha/\epsilon$ value on the rear surface of the deployed solar panels can be considered. The latter is particularly for the avoid over-heating of the solar cells diminish performance [Li et al., 2011]. Nevertheless, the CubeSat can still be thermally controlled with the given thermal configuration.

### Table 14: FS-2 Simulation Result Summary: With and without deployed solar panels.

<table>
<thead>
<tr>
<th>CASE</th>
<th>Unit</th>
<th>FS-2 Solar Panel</th>
<th>FS-2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max. Temp</td>
<td>°C</td>
<td>84.68</td>
<td>68.76</td>
</tr>
<tr>
<td>Min. Temp</td>
<td>°C</td>
<td>-28.57</td>
<td>-29.55</td>
</tr>
<tr>
<td>Ambient Range</td>
<td>°C</td>
<td>13.32-17.45</td>
<td>9.36-17.32</td>
</tr>
<tr>
<td>Abs/Emm</td>
<td>-</td>
<td>0.85/0.8</td>
<td>0.85/0.8</td>
</tr>
</tbody>
</table>

6.2 Far-side eclipse

6.2.1 Background

FS-2’s GTO uses an important assumption that the eclipse occurs at the perihelion of the orbit. This is thermally most favorable, as the eclipse time is the shortest and the satellite spends a large fraction of the orbit illuminated by the Sun. Far-side eclipses, or the case when the eclipse occurs at the aphelion of the orbit, also needs to be considered. This is done to evaluate the flexibility of the mission.
trajectory from the thermal point of view, and also to evaluate the worst case condition of orbit precession.

6.2.2 Simulation Setup

An identical thermal model was simulated. The argument of perigee was changed between $0^\circ$ and $180^\circ$ to compare the near-side and far-side eclipse scenarios. These orbits are compared in Fig 27. The eclipse zones are shown to provide a clearer comparison.

![Figure 27: FS-2 GTO orbits: (L)Near-side eclipse(original assumption); (R)Far-side eclipse(Worst case).](image)

6.2.3 Results

Temperature distribution profiles and load variations over time are studied for FS-2’s GTO with eclipse at the aphelion.

6.2.3.1 Temperature Distribution Profiles

The temperature distribution profiles for the FS-2’s GTO with a far-side eclipse is shown in Fig 25. The time variations of the temperatures are similar to the FS-2 orbit, and hence has a cyclic variation. The temperature profile at the last iteration of the simulation is shown.

6.2.3.2 Element-wise time variation

The time variation total thermal fluxes for this case is shown in Fig 29. **Inference:** The magnitude of maximum and minimum thermal loads conspicuously remain the same as the orbit is the same. However, the time variations of the thermal loads are cyclic with a smaller period, with more abrupt variations. This can be seen from the steep peaks of the graphs. The thermal loads also diminish to negligible loads at some parts of the eclipse time. This is because the only source of heat during eclipse is Earth’s planetary radiations, which vary inversely with the square of...
Figure 28: Temperature Distributions Profiles of FS-2 in far-side eclipse GTO: (L) Exterior view (R) Interior View.

distance form the surface as mentioned earlier. This is negligible when the distance is as high as the apogee. (Altitude 35000 km).

Figure 29: Element-wise time variation of total loads in FS-2 in the far-side GTO.
6.2.3.3 Discussion

The major thermal parameters are tabulated in Table 15 for comparing the GTOs with different eclipse types (best and worst cases).

<table>
<thead>
<tr>
<th>CASE</th>
<th>Unit</th>
<th>FS-2 Far-Side</th>
<th>FS-2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max. Temp</td>
<td>°C</td>
<td>65.90</td>
<td>68.76</td>
</tr>
<tr>
<td>Min. Temp</td>
<td>°C</td>
<td>-34.61</td>
<td>-29.55</td>
</tr>
<tr>
<td>Ambient Range</td>
<td>°C</td>
<td>-1.71-12.42</td>
<td>9.36-17.32</td>
</tr>
<tr>
<td>Abs/Emm</td>
<td>-</td>
<td>0.85/0.8</td>
<td>0.85/0.8</td>
</tr>
</tbody>
</table>

Table 15: FS-2 Simulation Result Summary: Far-side and Near-side Eclipse GTOs.

The orbits clearly point out that the GTO with an eclipse completely at the aphelion can have much longer eclipse. The results suggest that this eclipse make the satellite much colder, with the extremes of the temperature ranges of the exterior and interior decreasing considerably. The most alarming observation is that the ambient range has a significantly lower temperature. This is attributed to the satellite being in a longer and harsher eclipse condition (no thermal heating at all). The induced thermal stresses may be catastrophic and thus, a more comprehensive thermal control system - including active elements - would be a requirement for smooth operations of the avionics.

Notably, the change in RAAN or the orbit to completely invert the eclipse is a very slow process. Given that the mission life is 6 to 12 months depending on various requirements, it is not likely to occur if started from an orbit with a more favorable eclipse condition [Kozai, 1960].

6.3 MEO Mission

6.3.1 Background

Discussing the scientific requirements of FS missions is not in the scope of this thesis. However, the payloads need to study a high-altitude region, which may not necessarily be a GTO. Thus, an MEO was thermally evaluated. This was also done to prove the hypothesis that the thermal environments of the MEOs can be estimated by studying the trends of thermal loads: The magnitude of variation of thermal loads will be higher than a LEO and lower than a GTO, and the variation of the loads in the orbit will be gentler than a LEO and faster than a GTO.

6.3.2 Simulation Setup

The same thermal model was simulated in an MEO orbit. The orbit parameters are the same as used in the GTO will the apogee altitude decreased to 10000 km (MEO range). Thus, the simulation setup was straightforward with minimal changes apart from the apogee of the orbit.
6.3.3 Results

Temperature distribution profiles and load variations over time are studied for the MEO mission.

6.3.3.1 Temperature Distribution Profiles

The temperature distribution profiles for the FS-2 GTO with a far-side eclipse is shown in Fig 30. The thermal load variation has the same magnitude, as the orbit remains the same. However, the variation of the loads are faster as the loads are abruptly low during the eclipse.

Figure 30: Temperature Distributions Profiles of FS-2 in MEO: (L) Exterior view (R) Interior View.

6.3.3.2 Element-wise time variation

The time variation total thermal fluxes of the six surfaces is shown in Fig 31.

**Inference:** The total fluxes in the MEO shown cyclic and smaller variations as compared to the GTO. The maximum and minimum total fluxes on the CubeSat for the GTO and MEO are compared in Table 16. It is magnitude of variation in the GTO is larger than the MEO. The minimum flux remains the same as the perigee parameters of both orbits are identical.

<table>
<thead>
<tr>
<th>Case</th>
<th>GTO</th>
<th>MEO</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximum Flux</td>
<td>1778</td>
<td>1452</td>
</tr>
<tr>
<td>Minimum Flux</td>
<td>389</td>
<td>389</td>
</tr>
</tbody>
</table>

Table 16: Total thermal flux comparison: LEO vs GTO (all values in W/m²).
Figure 31: Element-wise time variation of total loads in FS-2 in MEO.

6.3.3.3 Discussion

The major thermal parameters are tabulated in Table 17.

<table>
<thead>
<tr>
<th>CASE</th>
<th>Unit</th>
<th>FS-2 MEO</th>
<th>FS-2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max. Temp</td>
<td>°C</td>
<td>69.86</td>
<td>68.76</td>
</tr>
<tr>
<td>Min. Temp</td>
<td>°C</td>
<td>-14.74</td>
<td>-29.55</td>
</tr>
<tr>
<td>Ambient Range</td>
<td>°C</td>
<td>9.2-16.9</td>
<td>9.36-17.32</td>
</tr>
<tr>
<td>Abs/Emm</td>
<td></td>
<td>0.85/0.8</td>
<td>0.85/0.8</td>
</tr>
</tbody>
</table>

Table 17: FS-2 Simulation Result Summary: GTO vs MEO

LEO have smaller variations in fluxes, but the variations are rapid. GTOs have larger variations, but the variations are gentler. MEOs, lies in the middle, both in the altitude and thermal parameters. It is has a moderated amount of flux, that vary moderately compared to the LEO and GTO. This moderation in seen with the more moderate minimum and maximum temperatures in the profile. Thus, the MEO is a thermally favorable mission with respect to thermal stresses induced as compared to the GTO. Conspicuously, the mission is feasible too.
7 Conclusion

A thermal investigation on the various orbital environments and thermal scenarios were done with respect to the Foresail missions. Initially, the thermal loads were studied to compare a polar sun-synchronous low altitude circular orbit (FS-1) to a Geostationary highly elliptical transfer orbit (FS-2). From the load analysis a hypothesis arose: GTOs have small average incident heat fluxes, making the environment colder in general. The thermal loads vary more. However, they vary gradually, inducing low thermal stresses due to the slower changes.

This was followed by a literature survey of other similar satellite missions. The study of the missions provided three important conclusions: First, the thermal design of a mission is done on two levels – System level (holistic design) and Region-specific design. A holistic design is initially done to ensure the feasibility of the configuration, after which thermally sensitive regions are identified and controlled; Second, the thermal control strategies that exist, and the ones that are adaptable for a CubeSat; Third, a CubeSat has not yet been successfully launched in an environment like that of FS-2’s GTO.

The thermal simulations of the various scenarios, their primary results and their respective feasibility conclusions are listed in Table 18. The feasibility refers to the scope of changes in the 3U CubeSat configuration that is needed for launching the satellite successfully. ‘Moderate’ and ‘Low’ refer to large changes in the configuration – For example, increasing the size to a 6U CubeSat to accommodate new thermal control methods, or increase the mass and thus, the thermal inertia.

<table>
<thead>
<tr>
<th>Mission</th>
<th>Case</th>
<th>Ext. Range</th>
<th>Ambient</th>
<th>Feasibility</th>
</tr>
</thead>
<tbody>
<tr>
<td>FS-1</td>
<td>LEO</td>
<td>-18.2 to 73.8</td>
<td>9.2 to 16.8</td>
<td>High</td>
</tr>
<tr>
<td></td>
<td>Spin mode</td>
<td>-22.5 to 73.8</td>
<td>9.3 to 17.4</td>
<td>High</td>
</tr>
<tr>
<td></td>
<td>Battery</td>
<td>NA</td>
<td>NA</td>
<td>High</td>
</tr>
<tr>
<td>FS-2</td>
<td>GTO</td>
<td>-29.5 to 68.7</td>
<td>9.3 to 17.3</td>
<td>Mod. High</td>
</tr>
<tr>
<td></td>
<td>Far-side GTO</td>
<td>-34.6 to 65.9</td>
<td>-1.7 to 12.42</td>
<td>Moderate</td>
</tr>
<tr>
<td></td>
<td>MEO</td>
<td>-14.7 to 69.8</td>
<td>9.2 to 16.9</td>
<td>High</td>
</tr>
<tr>
<td>FS-3</td>
<td>Lunar</td>
<td>-57.2 to 112.9</td>
<td>NA</td>
<td>Low</td>
</tr>
</tbody>
</table>

Table 18: Report Summary

LEO missions have flight heritage with many nanosatellites orbiting the low altitude region. Spinning is mission specific to FS-1, which analyzed and cleared for thermal control. A region-specific battery solution was suggested and implemented. A thermal chamber test (to be performed in the future) can confirm its utility for the mission.

GTO missions is considered a venture, as no CubeSat orbit the higher altitude region. The thermal analysis shows the thermal environment maybe more severe but can be controlled passively by modifying the thermo-optical properties of the satellite. GTO missions are proven feasible, but the argument of perigee or the placement of the eclipse is an important factor to consider during mission design.
The lunar environment (studied and analyzed in Appendix A) shows that a lunar CubeSat mission cannot use the given configuration. Significant modifications, on the external and internal areas may be required to fly a Nanosatellite class CubeSat to the moon. Higher altitude lunar orbits may be an option, but the stability of the orbit is then brought into question.
References


A Lunar Thermal Environment

Background
In this section, the thermal environment in the lunar sphere or influence in studied. To aptly compare the results, a 3U CubeSat is placed in a Low Lunar Orbit (LLO) or 400 km altitude. Deep space CubeSat missions are not far from reality after the success of the MarsCO. A prospective mission for deep space in the far future is a possibility. The Moon, the closest planetary body is a conspicuous choice for a CubeSat mission given the limitations of the satellite’s propulsion and communication systems due to a constrained mass and power budget. Thus, a prospective lunar mission was preliminarily assessed.

Theory: Lunar Environment
The lunar space receives similar solar radiations as that of Earth-bound orbits. The effects of Earth’s planetary IR and albedo radiations are negligible in lunar orbits due to the distance of separation. However, Low Lunar Orbits – The few stable orbits, allowing a mission life of atleast 3 months – receive a large, highly variable set of radiations from the lunar surface [Parker and Anderson, 2013], [Folta and Quinn, 2006]. The thermal load variations of each type are compared for the cases of Earth-bound and Lunar orbits in Table A1 [Gilmore and Donabedian, 2003], [Racca, 1995].

<table>
<thead>
<tr>
<th>Type of Load</th>
<th>Earth Orbit</th>
<th>Lunar Orbit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Direct Solar (W/m²)</td>
<td>1322 to 1414</td>
<td>1316 to 1421</td>
</tr>
<tr>
<td>Solar flux average</td>
<td>1367</td>
<td>1368</td>
</tr>
<tr>
<td>Albedo (ratio)</td>
<td>0.09 to 0.40</td>
<td>0.02 to 0.13</td>
</tr>
<tr>
<td>Albedo average</td>
<td>0.3</td>
<td>0.1</td>
</tr>
<tr>
<td>Planetary IR (W/m²)</td>
<td>182 to 298</td>
<td>5.2 to 1268</td>
</tr>
<tr>
<td>Planetary average</td>
<td>236</td>
<td>NA</td>
</tr>
</tbody>
</table>

Table A1: Thermal Environment Comparison: Earth-bound vs Moon-bound orbits

Simulation Setup
The thermal simulation was setup to accommodate the orbital thermal loads of the lunar orbit. A 400km-altitude Low Lunar orbit with small inclination was chosen. Only circular frozen orbits are stable at small altitudes. The orbit propagated for the simulation is shown in Fig . Due to a higher processor load, smaller number of orbital positions were used in orbit, compared to Earth orbit simulations carried in the earlier sections.
The external temperatures and thermal loads are presented in this section. The satellite interior is the same as that of the FS missions but is not studied in this thesis. The interior of the deep space mission may be completely different (for example, a large propulsion module and few payloads), making its study inconsequential for the scope of this thesis. However, the exterior surface provides a preliminary idea of the harsh thermal environment for lunar missions.

The temperature profile of the CubeSat in a lunar orbit is shown in Fig A2.

Figure A2: Temperature Distributions Profiles of the CubeSat in an LLO
The temperature and total flux variations on the six surfaces of the CubeSat in shown in Fig A3.

Figure A3: Element-wise time variation in LLO: (L) Total Flux (R) Temperatures

**Inference** It can be seen that the temperature and thermal load variations are steep compared to that of Earth-bound orbits. The total thermal loads on the satellite vary from over 2050 W/m$^2$ to below 120 W/m$^2$, in short short time periods. The orbital periods for a low lunar orbit is much shorter due to the smaller size of the primary body. This attributes to large thermal stresses inducing much more extreme temperature ranges as shown.

**Discussion**

The external temperature variations are much more severe, even compared to the far-side eclipse GTO (worst case Earth bound orbit). The temperature ranges from **-57.22 to 112.9 °C**. This is due to a much more extreme thermal loading. Given the small orbital period and the large thermal stresses induced, the temperatures cannot be controlled by iterating with the thermo-optical properties of the exterior. Thermal insulation is used when the thermal stresses are high instead. However, a major design change will be required when active thermal control elements are implemented.
B Steady state analysis using STK

Summary

In this section, the results of the thermal loads are verified using a steady state thermal analysis. A steady state analysis simulate the thermal loads on the satellite at multiple positions on the orbit and the satellite achieves thermal equilibrium with the environment at these positions. Thus, it is similar to the case if the satellite were stationary for a long time in that particular position in the orbit. This was done using a spherical model in Systems Tool Kit (STK). STK has a limited number of shapes that can be propagated in the orbit, and a small spherical model is perfectly representative of the thermal loads acting the object in orbit. However, STK was not used for the all the other simulations as complex models cannot be used for simulation and a transient simulation cannot be run. Even though steady analysis can be useful to study the thermal loads, a more detailed thermal model simulated in transient conditions is necessary to the effects of these loads on a particular satellite. The simulations were done to compare the loads of the polar LEO and the GTO, the respective orbits discussed in section 3.4.

Results

The results of the simulations are presented in Fig B1.

Figure B1: Thermal simulation results in STK: (L) Mean equilibrium temperatures of the object in FS-2’s GTO; (R) Mean equilibrium temperatures of the object in FS-1’s LEO; Y axis represents the temperature, X represents time.
**Inference:** The equilibrium temperatures are representative of the net thermal loads incident on the satellite (The higher the thermal loads, the greater the equilibrium temperature and vice versa). It can be seen that the temperatures in FS-1’s LEO vary only from **-5 to 55°C**, whereas the temperatures in FS-2’s GTO vary from **-7 to 94°C** - A much larger variation. It can also be seen that the variation in the LEO is more steep, whereas the variation in the GTO is gradual and smooth. The temperature ranges also suggest that the GTO environment is much colder. This verifies the results of the thermal load analysis to a large extent.