

# **Thermal Modeling, Analysis, and Control for ASTUSat**

**M.Sc. Thesis**

**By**

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# Thermal Modeling, Analysis, and Control for ASTUSat



*A Thesis Submitted to Thermal and Aerospace Engineering Program of School of Mechanical, Chemical, and Materials Engineering, Adama Science and Technology University in partial fulfillment of the requirements for the award of the degree of Master of Science in Thermal Engineering.*

By

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Adama, Ethiopia

March 2019

## Declaration

I hereby declare that this M.Sc. Thesis entitled “**Thermal Modeling, Analysis, and Control for ASTUSat**” in partial fulfillment of the requirements for the award of the degree of Master of Science in Thermal Engineering is an authentic record of my own work carried out from February 2018 to March 2019 under supervision of Dr. Gezahegn Habtamu, Assistant Professor of Thermal Engineering program, Adama Science and Technology University.

The matter embedded in this thesis has not been submitted for the award of degree or diploma in any other university. All relevant resource information used in this paper has been duly acknowledged.

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This is to certify that the above statement made by the candidate is correct to the best of our knowledge and belief. This has been submitted for examination with our approval.

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## Approval of Board of Examiners

We, the undersigned, members of the Board of Examiners of the final open defense by Habtamu Mengistu have read and evaluated his thesis entitled “Thermal Model, Analysis, and Control for ASTUSat” and examined the candidate. This is, therefore, to certify that thesis has been accepted in partial fulfilment of the requirement of the Degree of Masters of Science in Thermal Engineering.

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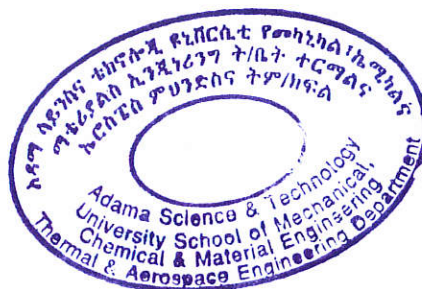
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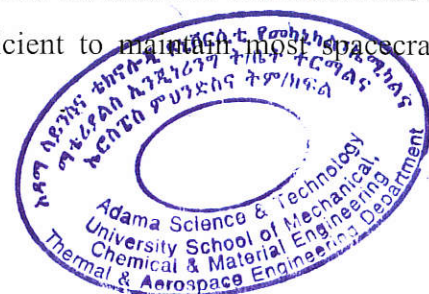


## Abstract

ASTUSat is the first 3U CubeSat in Ethiopia under development by Space Technology Institute of Adama Science and Technology University which have a technical mission of taking colored picture of the earth from low Earth orbit. The satellite will be deployed into orbit from the international space station which is located at an altitude of 400 km. The satellite contains multiple subsystems, from those subsystems thermal control subsystem is the one which maintains all components of the satellite within their operating temperature limit during all mission phases. The ASTUSat program creates the opportunity for graduate students to participate on different subsystem designs of the satellite.

The aim of this thesis is to develop thermal model, to perform thermal analysis, and to select the required thermal control system hardware's so that the satellite survives the space environment. In the thesis the main mission parameters of ASTUSat and the current spacecraft design are presented, in order to fix the boundary conditions and the thermal environment used for thermal modeling and analysis. In the document two thermal analysis were performed and presented based on the complexity of the models. The preliminary transient analysis is performed by developing a simplified equation from energy balance equation and solved using Euler explicit method in MATLAB environment. In the analysis two boundary conditions were identified which are labeled as the worst hot and cold scenario and the transient temperature of the satellite were predicted at these conditions. The results obtained reveled that the satellite temperature fluctuates between 86.6 and 89.3 °C during worst hot scenario and between 17.9 and 34.48 °C during worst cold scenario. The detail thermal modeling and analysis is a finite element model of the satellite developed within FEMAP TMG environment by including heat loads such as subsystems internal heat generation and environmental thermal radiations (infrared, albedo, solar flux). This model is solved and the transient temperature of satellite components are extracted and presented. The results showed that the CubeSat can reach temperature between +101.5 and -24.5 °C for worst hot scenario, and +65.5 and -41.3 °C for worst cold scenario. These results served as a basis for the development of thermal control system.

The Finite element model has been refined with different spacecraft configuration and solved to set the requirements applicable to the thermal control system of the satellite. The results showed that a basically passive thermal control technique is sufficient to maintain most spacecraft's



components within their temperature range when appropriate thermal tapes or coatings and thermal washers are provided.

Therefore, the thermal modeling results obtained in this thesis can easily be adjusted to the actual input data, and design arrangement of payloads and subsystems to generate thermal performance of spacecraft ASTUSat which is critically needed before conducting ground vacuum chamber thermal performance test of ASTUSat at the laboratory of Space Technology Institute of ASTU.

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## List of Symbols

|                 |                                                        |
|-----------------|--------------------------------------------------------|
| $A$             | surface area [m <sup>2</sup> ]                         |
| $A_c$           | cross sectional area [m <sup>2</sup> ]                 |
| $a$             | albedo coefficient                                     |
| $c_p$           | specific heat at constant pressure<br>[J/kg K]         |
| $\dot{E}_{st}$  | rate of energy stored [W]                              |
| $\dot{E}_g$     | rate of energy generation [W]                          |
| $\dot{E}_{in}$  | rate of energy transfers into the<br>system [W]        |
| $\dot{E}_{out}$ | rate of energy transfers out of the<br>system [W]      |
| $F$             | view factor                                            |
| $G$             | thermal conductance                                    |
| $h$             | universal Plank constant [J/s];<br>altitude [km]       |
| $k$             | thermal conductivity [W/m <sup>2</sup> K]              |
| $L$             | length [m]                                             |
| $L_c$           | characteristic length                                  |
| $m$             | mass [kg]                                              |
| $Q$             | total Absorbed thermal radiation [W]                   |
| $q'''$          | heat generation per unit volume<br>[W/m <sup>3</sup> ] |
| $q''$           | heat flux [W/m <sup>2</sup> ]                          |
| $q$             | heat transfer rate [W]                                 |
| $R$             | thermal Resistance [K/W]                               |
| $R_{t,c}$       | thermal contact resistance [K/W]                       |
| $r$             | radius of a sphere or circle [m]                       |
| $S$             | Solar flux [W/m <sup>2</sup> ]                         |

|            |                         |
|------------|-------------------------|
| $T$        | temperature [K]         |
| $T_\infty$ | deep space temperature; |
| $T_s$      | surface temperature [K] |
| $t$        | time [s]                |

### Greek letters

|               |                                                                 |
|---------------|-----------------------------------------------------------------|
| $\alpha$      | absorptivity                                                    |
| $\varepsilon$ | emissivity                                                      |
| $\beta$       | beta angle                                                      |
| $\rho$        | density                                                         |
| $\tau$        | transmittivity                                                  |
| $\sigma$      | Stefan-Boltzmann constant<br>[W/m <sup>2</sup> K <sup>4</sup> ] |
| $\lambda$     | wavelength [μm]                                                 |

### Subscripts

|        |             |
|--------|-------------|
| $a$    | albedo      |
| $con$  | connector   |
| $cond$ | conduction  |
| $cont$ | contact     |
| $e$    | Earth       |
| $emi$  | emitted     |
| $IR$   | infrared    |
| $rad$  | radiation   |
| $s$    | solar       |
| $sc$   | spacecraft  |
| $spa$  | spacer      |
| $sur$  | surrounding |
| $tot$  | total       |

## Abbreviations

|         |                                                     |
|---------|-----------------------------------------------------|
| 1U      | 1 Unit cubesat                                      |
| 3U      | 3 Unit cubesat                                      |
| ADCS    | Attitude Determination and Control Subsystem        |
| ADCS M  | Attitude Determination and Control Subsystem Module |
| ATCS    | Active Thermal Control System                       |
| ASTUSat | Adama Science and Technology University Satellite   |
| BCDR    | Battery Charge/Discharge Regulator                  |
| CanX    | Canadian Advanced Nanospace eXperiment              |
| CB      | Controller Board                                    |
| COMM    | COMMunication subsystem                             |
| COTS    | Commercial Of The Shelf                             |
| C&DH    | Command and Data Handling                           |
| DB      | Drive Board                                         |
| DSP     | Deployable Solar Panel                              |
| EPS     | Electrical Power Subsystem                          |
| ESA     | European Space Agency                               |
| EM      | ElectroMagnet                                       |
| FDM     | Finite Difference Method                            |
| FE      | Finite Element                                      |
| FEM     | Finite Element Method                               |
| FEMAP   | Finite Element Modeling And Postprocessing          |
| FSP     | Fixed Solar Panel                                   |
| GMM     | Geometrical Mathematical Model                      |
| ISS     | International Space Station                         |
| IR      | Infrared                                            |
| LEO     | Low Earth Orbit                                     |
| MLI     | Multi-Layer Insulation                              |
| MIST    | MIniature Student saTellite                         |
| NRCSD   | <u>Nanoracks</u> CubeSat Deployer                   |

|       |                                                   |
|-------|---------------------------------------------------|
| OBC   | On-Board Computer                                 |
| OUFTI | Orbiting Utility For Telecommunication Innovation |
| PCB   | Printed Circuit Board                             |
| PPM   | Pluggable Processor Module                        |
| P-POD | Poly Picosat Orbital Deployer                     |
| PTCS  | Passive Thermal Control System                    |
| SES   | Static Earth Sensor                               |
| SW    | Side Wall                                         |
| RW    | Reaction Wheel                                    |
| TCS   | Thermal Control System                            |
| TMG   | Thermal Model Generator                           |
| TMM   | Thermal Mathematical Model                        |
| TVAC  | Thermal Vacuum                                    |
| UHF   | Ultra-High Frequency                              |
| WCS   | Worst Cold Scenario                               |
| WHS   | Worst Hot Scenario                                |

# Chapter 1

## 1 Introduction

### 1.1 Introduction

In the last decade due to the advances in MEMS (Micro-Electro-Mechanical System) technology it becomes feasible in the development of small satellites that are small in size and mass compared to traditional ones. The main reason of miniaturizing and standardizing satellites is to reduce the time and money required to develop a satellite. Also, using small satellites creates the opportunity to use excess available space in huge launching vehicles. Satellites that weigh less than 500 kg are categorized as small satellites, and within this category they are further subdivided in to: minisatellites (between 100 and 500 kg), microsatellites (between 10 and 100 kg), nanosatellites (between 1 and 10 kg), and Picosatellites (between 0.1 and 1 kg) [1].

CubeSats are standardized small satellites that are categorized under nanosatellites and picosatellites. CubeSats standard was created by Stanford and California Polytechnic State Universities in 1999, and it specifies that a standard 1 unit (1U) is a 10 cm cube (10×10×11.35 cm) with a mass of up to 1.33 kg. A 1U CubeSat could either serve as a standalone satellite or could be combined together to build a larger satellite like 2U, 3U, 6U, 12U, and 27U CubeSats to increase their capabilities.

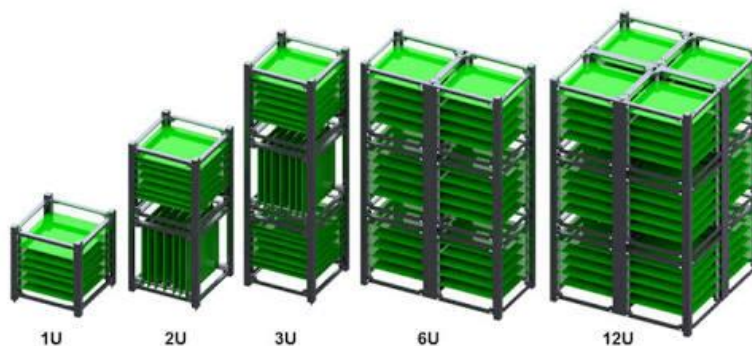


Figure 1-1: CubeSat specifications [2]

Since CubeSats are miniaturized satellites they are launched from the ground as a secondary payload by utilizing a small space available in huge launch vehicle's that has a primary mission (e.g. launch vehicles having a mission of launching normal size satellites) [3]. And they are put in orbit by deployers on the international space station, or launched as secondary payloads on a launch vehicle [4]. Usually CubeSats are deployed in to low Earth orbit (LEO) (i.e. with an altitude between 160 km to 2,000 km).

CubeSats were more popular for educational or technology demonstration platforms. But nowadays CubeSats are widely utilized for low-cost missions with high potential value in terms of science return and commercial revenue. These missions include Earth science (Support understanding and monitoring of Earth processes), space science (support the understanding of the solar system and the universe, for example, by monitoring the space radiation environment), to serve as technology demonstrators (reduce risk before implementing new technologies or new design concepts in a funded science campaign, for example, by demonstrating formation flight, aero-capture technologies and so on) or as a know-how transfer and training (provide means and hands-on experience of technical and managerial aspects, from design, construction, test, launch and orbital operations) [3].

### 1.1.1 ASTUSat

ASTUSat is the first 3U CubeSat in Ethiopia under development in Adama Science and Technology University Space Technology Institute with the technical objective of taking a better than 30 m<sup>2</sup>/pixel resolution color picture using a camera installed on the CubeSat platform. The development of this 3U CubeSat is based on the “**Purchase, Integrate/Assemble, Test and Launch**” principle of the standard/off-the-shelf CubeSat kits available in the market [5].

The objective of the ASTUSat mission is to allow ASTU-Space Technology Institute (STI) to demonstrate technology and learn challenging engineering concepts such as understanding the interdependent subsystems and how they fit into an integrated complex system during the design, development and testing of small spacecraft missions by using hand-on educational approach before the institute engaged on different projects [5]. Also, the satellite can be used for remote sensing application.

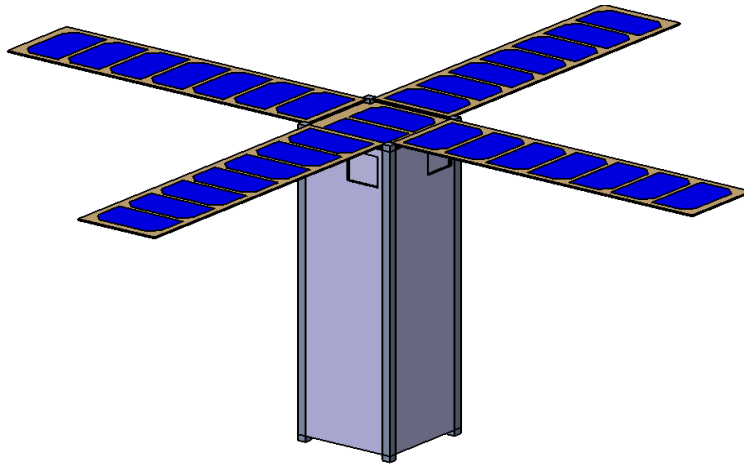


Figure 1-2: Simplified CATIA CAD model of ASTUSat

The CubeSat will be deployed into orbit from the international space station (ISS) at an approximate altitude of 400 km from the sea level. Once it is in orbit it will be powered by a battery that is charged by solar arrays on the outside of the structure. The attitude of the satellite will be controlled by an active attitude control system installed on-board in the spacecraft. It will use the camera installed for taking different part of the earth and a microcontroller will process and send the image to the telemetry system for transmittal to ground, and the ground station will receive the picture over a period of time.

Currently ASTUSat is under its design phase where different design analyses are performed, and based on the results the components of subsystems are selected by the design team. The satellite contains multiple components which are categorized in subsystems such as, structure, command and data handling (C&DH), Guidance, Navigation and Control (GNC), electrical power (EPS), communication (COMM), payload, and thermal control subsystems.

The subsystem which is directly related to this thesis is the thermal control subsystem. The function of the **thermal control subsystem (TCS)** is to keep all the satellite components within their operating temperature ranges during all mission phases. It is essential to guarantee the proper performance and success of the mission because if a component is subjected to temperature which are too high or too low, it could be damaged or its performance could be severely affected. In general, it is part of the spacecraft's system that protects the components from overheating, either by thermal insulation from external heat fluxes (such as the solar radiation, albedo, and planetary

infrared), or by proper heat removal from internal sources (such as the heat emitted by the internal electronic equipment). It also protects the equipment from undercooling, by thermal insulation from external sinks, by enhanced heat absorption from external sources, or by heat release from internal sources. In the design stage of TCS for a satellite, thermal analysis is performed to predict the mission thermal performance of the satellite and based on the results from the analysis different hardware of the TCS are selected [6].

Therefore, since ASTUSat's thermal control system is not designed yet, the aim of this thesis is to perform a detailed system level thermal analysis to obtain the mission temperature evolution and distribution of the satellite in order to observe the components that are in danger and select appropriate thermal controlling hardware so that the components work within their specified operating temperature limit.

## **1.2 Problem statement**

The ASTUSat contains multiple components within each subsystem, and each component has its own operating temperature limit which is specified by the manufacturer. Exceeding this temperature limit could severely affect the performance or damage the component. But when the satellite is in mission, it is exposed to heat loads like direct sunlight, albedo (sunlight reflected from the earth surface), Infrared of the earth, and dissipated electrical power by highly integrated electronic circuitry. These heat sources contribute as main heat sources during sunlit and causes overheating of components above their operating temperature range. Also, when the satellite is shadowed by the earth from the sun (eclipse) the space environment is so cold causes undercooling of components below their operating temperature range. To avoid such cases from occurring a thermal control system is required within the satellite in order to protect the satellite from the harsh space environment. Thus, to evaluate possible TCS strategies extensive radiative and conductive modelling needed to be performed for a variety of possible transient orbital parameters.

## **1.3 Objectives**

### **1.3.1 General objective**

The main objective of this thesis is to maintain the components of ASTUSat within their thermal design limits when the satellite is subjected to extreme low earth orbit environmental scenarios.

### **1.3.2 Specific objectives**

The specific objectives include:

- Developing a simplified transient thermal mathematical model for ASTUSat that simulates the average mission temperature of the satellite for worst hot and cold scenario heat loads.
- Developing detail finite element thermal model using CAD model of the ASTUSat and solving the model to predict the mission temperature of main components such as the payload, electronics stack, and structural components of the satellite.
- Analyzing the data and in case thermal requirements are not fully met and problems arise at system or component level the appropriate thermal control system components are recommended.

## **1.4 Scope and limitations of the work**

### **1.4.1 Scope**

This thesis mainly deals with developing two thermal models of ASTUSat and performing thermal analysis using MATLAB code and commercial software called FEMAP TMG. It also includes improving thermal control system of the satellite based on results gained from computer simulations.

### **1.4.2 Limitations**

A thermal model must be verified before flight, which is achieved through performing actual tests by using space environment simulating devices to confirm that the analysis method is valid and that the satellite components are indeed within their thermal design limits [7]. Since, the ASTUSat were under design phase when this paper is written thus no experimental tests were conducted to validate the analysis methods and thermal control system. But all the analysis methods were validated using simulations done in other CubeSat thermal control system design projects. Another limitation of this thesis is the materials thermal properties used for thermal analysis are all taken either from reference papers or by calculation (for composite materials) using averaging schemes instead of measuring the property of the material through experimental setups.

## 1.5 Methodology

The methodology followed to attain the thesis objectives are:

- Understanding the theory behind the low Earth orbit thermal environment and thermal control system design.
- Developing a simplified thermal mathematical model of ASTUSat from the spacecraft energy balance equation and identify the boundary conditions.
- Solving the simplified thermal mathematical model for the identified boundary conditions by writing a code within the MATLAB environment.
- Developing detail finite element thermal model of ASTUSat in the commercial software FEMAP TMG using the CATIA CAD model of the satellite.
- Solving the model and observe the mission temperature of main satellite components.
- Based on the results obtained select appropriate thermal control techniques that maintains the ASTUSat components within their operating temperature. Validate the thermal control technique by updating the finite element model with the selected thermal control techniques and resolve the model. Then, check whether the satellite is within its thermal design limits and if not change it with a thermal control technique or hardware that fits best for the satellite.

## 1.6 Thesis Outline

This thesis is organized as follows:

Chapter 2: Provides a theoretical basis by presenting background to heat transfer, thermal environment of the low earth orbit, spacecraft energy balance, and spacecraft thermal control system design which are very mandatory concepts for the work to be developed.

Chapter 3: The current ASTUSat design is discussed and the preliminary thermal analysis is performed for a simplified model by using MATLAB. In the analysis the average temperature evolution of the satellite is obtained and discussed.

Chapter 4: The detail finite element model of ASTUSat developed in FEMAP TMG software is discussed.

Chapter 5: The finite element model is solved to obtain the temperature distribution and transient temperature of the satellite components and subsystems. The results from the simulation are presented and discussed by comparing with the thermal design limits of the components. The finite element model is modified with new thermal control techniques to improve the thermal control system of the satellite. The model is solved again and the results are presented.

Chapter 6: Provides the conclusion, contributions, design recommendations and future work suggestions.

# Chapter 2

## 2 Literature Review

In this chapter, the international research publications on CubeSat thermal analysis and control system designs are discussed.

### 2.1 International CubeSat's Thermal control system design

In this section some of the researches conducted around the world about CubeSats thermal analysis and control system designs are discussed.

Compass-1 [8] is a 1U CubeSat developed by the students of the University of Applied Sciences in Aachen, Germany in 2004. Its mission is to take picture of the Earth at a sun synchronous LEO. Compass-1 has already been launched on April, 28<sup>th</sup>, 2008 by the Indian PSLV launcher. The thermal design and thermal analysis are done as a thesis in the university. for computational analysis ANSYS software which uses FEM to calculate nodal temperatures was chosen. but experimental analysis was not performed to validate computational analysis in the research. Their thermal model led them to use active thermal control hardware, as well as passive thermal control hardware. The active thermal control hardware selected is a heater that will attached to the battery of the satellite and activated when the temperature drops below a certain threshold level. For the passive thermal control black paint is selected and applied on external surfaces that are not occupied by solar cells, camera lens, and antennas.

OUFTI-1 [9] is the first 1U CubeSat developed at the University of Liège and even the first one ever made in Belgium in 2008. The key, innovative feature of OUFTI-1 is the use of the D-STAR amateur-radio digital-communication protocol. The thermal design and thermal analysis are performed in the university as a master's thesis by the student. In the thesis, the preliminary thermal analysis is performed by using a simulator developed in MATLAB Simulink environment, and more detailed thermal analysis is performed by using ESARAD radiation exchange package and ESATAN lumped finite difference thermal balance package. Also, some experimental measurements were taken to determine the conductivity and specific heat property of Li-Po battery, the emissivity of the aluminum frame's coatings, and the thermal contact resistance

between the top and bottom faces with the lateral frame around the screws. Therefore, their thermal model led them to use the combination of passive thermal control components (such as low emissive tapes, and Nylon washers combined with titanium spacers) and active thermal control components (heaters for the battery) in order to maintain OUFTI-1 components within their allowable temperature. But experimental analysis was not performed to validate computational analysis in the research.

The CanX-7 [10] nanosatellite is a 3U CubeSat developed at the Space Flight Laboratory in 2015. Its mission is to demonstrate the use of a mechanically deployed drag sail as a preventative approach to the space debris problem, as well as an Automatic Dependent Surveillance-Broadcast (ADS-B) receiver for aircraft tracking. The thermal control system analysis and design along with the design and analysis of Attitude Determination and control system (ADCS) of the CanX-7 CubeSat is performed in the University of Toronto as master's thesis. Throughout the thesis the satellite is modelled with Finite Difference Method (FDM) by Siemens'NX 8 software package for thermal analysis. Based on the temperature distribution gained from the thermal analysis, different types of thermal control tapes (which is passive thermal control component) are applied to the exterior surface of the spacecraft, an electric heater is selected for the battery (active thermal control component), and several modifications to the CanX-7 structural design were applied. In the thesis the finite difference analysis was not verified by thermal vacuum testing because the satellite assembly was not complete.

MIST [11] is a 3U CubeSat that is developed in KTH Royal Institute of Technology, Sweden in 2015. The satellite is aimed to host eight different payloads called Nanospace, Ratex-J, Morebac, SiC, Cubes LEGS, SEU and the last payload is a camera, whose applications are described in detail in reference [11]. The thermal control system design and analysis is performed as a master's thesis. For the thermal analysis, different MATLAB scripts were written for calculating thermal conductance between components and understanding the effect of radiation optical properties of the satellite's surfaces on the average temperature of the satellite. For more detail thermal analysis, the spacecraft was modeled with FEM using Siemens NX<sup>TM</sup> software package. Also, in the thesis the radiation exchange between two Printed Circuit Boards (PCB's) and the "wall" of the spacecraft was investigated both analytically and numerically by MATLAB and Siemens NX<sup>TM</sup> software respectively. It was noted from the thermal analysis the payloads are not within their

temperature limit. Therefore, from a passive thermal control technique MLI and from active thermal control technique heater was used to maintain the payloads within their operating temperature. In the thesis experimental analysis was not performed to validate the detail computational analysis.

SwissCube [12] nanosatellite is a 1U CubeSat developed at EPFL in Lausanne, Switzerland. SwissCube's mission is to study the nightglow phenomenon. SwissCube has its own developed structure which is fabricated by CNC milling and wire electrical discharge machining and uses an active attitude control with magnetotorquers, sun sensors, gyroscopes and magnetometers to stabilize the satellite. Concerning the thermal control subsystem, their simulations led them to use a 500 mW (25  $\Omega$ ) heater to keep the two batteries within their allowable range of temperatures. They are packed in an aluminum box to prevent deformation due to the vacuum environment and are fastened on a copper plate as the heater, all with epoxy resin. The copper plate is used because of its high thermal conductivity to conduct the generated heat to the batteries. The heater is thermostatically controlled to avoid the use of the OBC and is switched on under the 0 °C threshold.

The PiCPoT [13] nanosatellite is a 1U CubeSat developed at Politecnico di Torino University in Italy. The PiCPoT nanosatellite is developed as technology demonstration and its technical mission is taking pictures of the earth and transmitting them to the ground. The thermal analysis is performed by developing simulation code in MATLAB environment. From the simulation it is confirmed that passive thermal control system is enough to maintain most of the satellite's components within their temperature range. For the external surfaces that are not occupied by solar cells, antennas and cameras' lenses, tapes with Teflon substrate covered by a deposit of aluminum are used in order to improve the infrared emissivity of the satellite's external surface by keeping the solar absorptivity of the surface relatively low. For the internal configuration of the satellite ten layers-MLI blankets are used between the external and internal aluminum panels, and for panels of the internal structures' materials with emissivity equal to 0.8 is selected.

## **2.2 Conclusion from literature review**

From the literature review it can be concluded that thermal control system is mandatory for the success of any CubeSat's mission. Due to the time, budget, and personnel constraints placed on small satellites programs, designing and testing a spacecraft's Thermal Control System (TCS) usually comes as a last priority. Instead, it has been observed that CubeSat programs rely on thermal modeling to predict and analyze their spacecraft's on-orbit thermal performance. Usually thermal modeling and analysis of a CubeSat started from preliminary thermal models using simplified equation in MATLAB and it is followed by detail thermal modeling using modeling software that uses either finite element method or finite difference method such as Siemens NX<sup>TM</sup>, thermal desktop, FEMAP TMG, and so on. The thermal analysis of CubeSats usually are carried out for worst cold scenario (WCS) and worst hot scenario (WHS) experienced by the spacecraft. Based on the thermal modelling and analysis the CubeSat thermal control system's components are selected. The most common thermal control hardware's used for CubeSats are multi-layer insulations (MLI), coatings, tapes, and heaters.

Therefore, in this thesis the thermal modeling, analysis and selection of thermal control system's components are discussed for a spacecraft ASTUSat.

# Chapter 3

## 3 Preliminarily Thermal Analysis

In this chapter the preliminary thermal analysis of the ASTUSat is performed to obtain average transient mission temperature of the satellite. First the chapter give a brief overview about the ASTUSat subsystems including the internal heat dissipation of the electronics' and the orbit of the satellite. Next, the reference frames used to locate the satellite in orbit are discussed. Then, the environmental thermal radiations the satellite exposed to (specifically to its orbit) are studied, and methods used to compute the amount of environmental thermal radiations reaching the satellite are also discussed. After that energy balance is applied by assuming isothermal satellite and from it equations are derived that can be used to estimates the steady state temperature and transient temperature of the satellite. The equations are validated by applying them on 1U CubeSat called OUFTI-1. Finally, the boundary conditions for thermal analysis are identified and transient temperature of the ASTUSat are calculated and discussed.

### 3.1 Overview of ASTUSat

The following subsections provide a brief introduction to the subsystems of ASTUSat, components operating temperature range, and heat dissipation of components.

#### 3.1.1 Subsystem overview

The ASTUSat is composed of a number of subsystems including structure, the command and data handling (C&DH), Guidance, Navigation and Control (GNC), electrical power (EPS), communication (COMM), payload, and thermal control subsystem. The subsystem of integration, ground station, ground support equipment, and test support the mission also included.

The satellite uses the Pumpkin CubeSat Kit 3U structure made up of Aluminum 6061 and hard anodized rails [5]. The purpose of the structural subsystem for the ASTUSAT is to provide a simple well-made structure that will survive launch loads and a suitable environment for the operation of all subsystems throughout all phases of the mission life. While providing an easily accessible data and power bus for debugging and assembly of components. Moreover, the structural subsystem shall carry, support and mechanically align the spacecraft equipment. It shall also cage and protect

folded components during boost. Therefore, structural design shall aim for simple load paths, a maximization in the use of conventional materials, simplified interface and easy integration.

The command and data handling subsystem of the satellite is responsible for receiving, validating, decoding, and distributing commands to other subsystems as well as gathering, preparing, and storing housekeeping and mission data for downlink or onboard utilization. In general, the command and data handling subsystem also integrate additional functions such as computer health monitoring, security interfaces, and spacecraft timekeeping. The ASTUSat C&DH components are the CubeSat kit Motherboard (MB) which is the fifth generation Pumpkin's line of single-board computers (SBCs) designed for use in the CubeSat Kit and elsewhere. The motherboard has a socket to accommodate a Pluggable Processor Module (PPM). And From the available PPM the PPM D1 with the PIC24 processor is used [5].

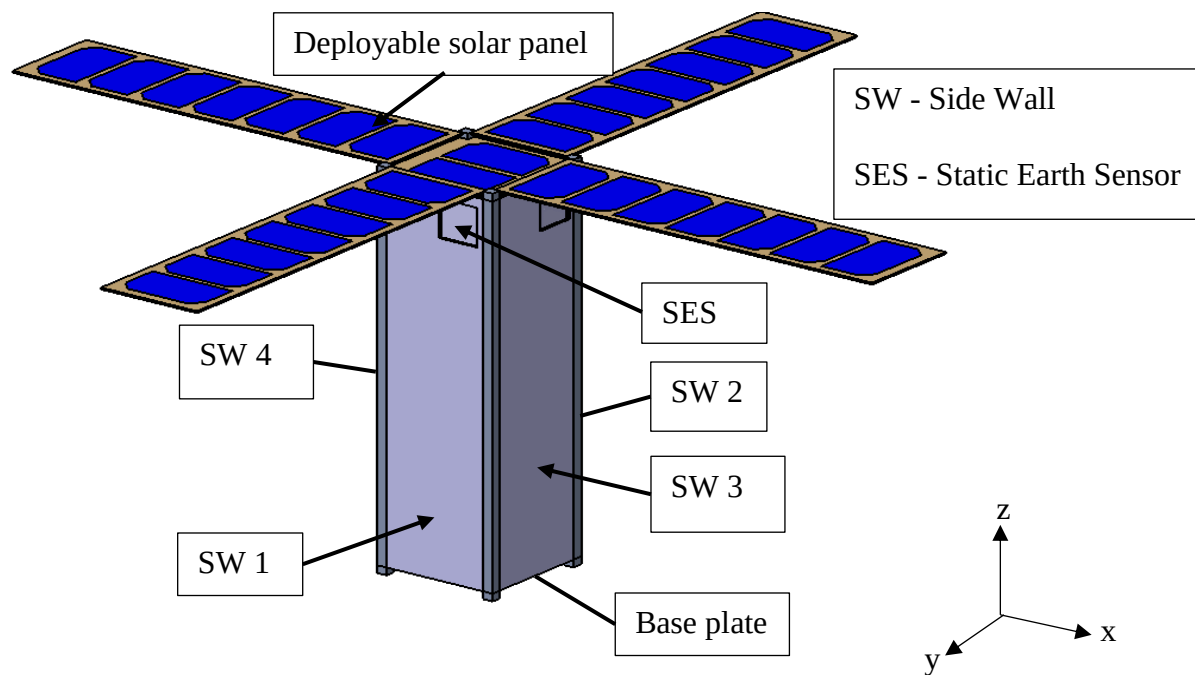


Figure 3-1: External feature of ASTUSat CATIA CAD model

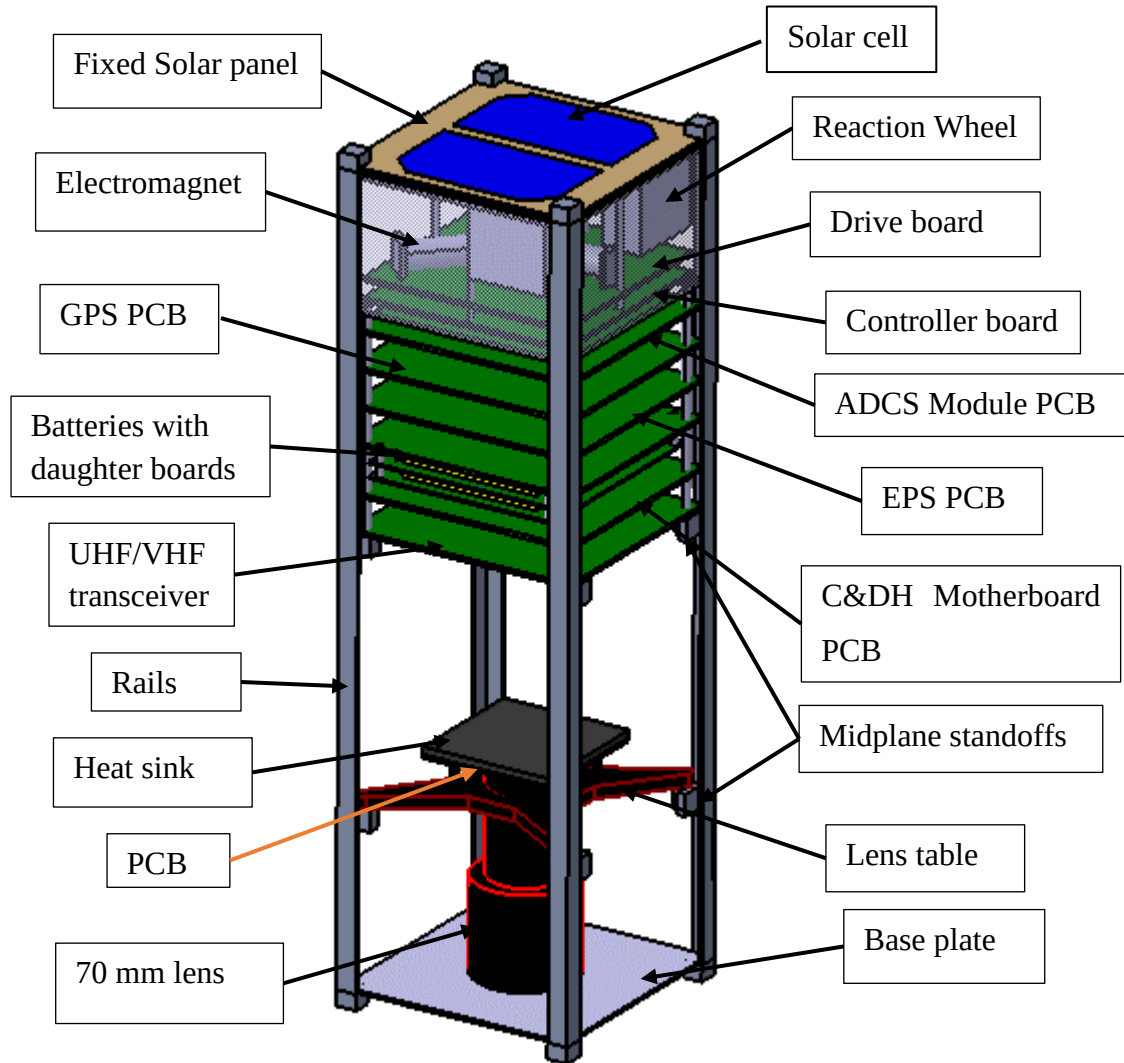


Figure 3-2: ASTUSat subsystem components CATIA CAD model

The Guidance, Navigation and Control (GNC) subsystem can be generally considered as a combination of the Orbit Determination and Control Subsystem (ODCS), which measures and maintains the position of the satellite's center of mass as a function of time, and the Attitude Determination and Control Subsystem (ADCS), which measures and maintains the satellite's orientation about its center of mass by using sensors such as star trackers, sun sensors, earth sensors, and magnetometers to determine spacecraft attitude and uses actuators such as reaction wheels, magnetorquers to orient the spacecraft in a desired direction. The satellite is integrated with a GPS receiver for ODCS. Since a precise orientation of the satellite is required the MAI-400

ADCS is used for ASTUSat mission which contains three reaction wheels, a 3-axis magnetometer, two IR Earth Horizon Sensor (IREHS), three electromagnets and an ADACS computer for a stand-alone, plug-and-play attitude control system for small satellites [5].

The other subsystem of the ASTUSat is the electrical power subsystem which consists a power source, energy storage, power distribution, regulation and control unit. The photovoltaic solar cells are the primary power sources for the satellite. The satellite uses four end hinged deployable solar panels and body fixed solar panel on the cover plate which have solar cells only on one side of the panel. Since the satellite is under design phase the number of solar cells required for each panel is not decided yet. Thus, based on the maximum number of solar cells each panel can hold (6 solar cells on each deployable solar panel and 2 solar cells on fixed solar panel) is used in this thesis. To store the collected energy by the solar cells the CubeSat uses a Clyde Space company lithium polymer battery. The 2S2P configuration which is two series battery cells mounted side-by-side on PCB that can be mounted as a daughter board on PC 104 sized, CubeSat kit compatible PCB the 2 daughter boards in parallel are used. The satellite is integrated with 3U EPS 2 motherboard to store the energy collected by the solar cells in the batteries and provide the bus with the required voltages [5].

The communication subsystem (COMM) is the interface between the satellite and the ground station, which allows the satellite to downlink its payload and housekeeping data to the operation center, to transmit operator commands to the CubeSat as well as to establish intersatellite communications. The ASTUSat communication subsystem contains three basic components: Li-1 Radio, VHF/UHF Transceiver, and CubeSat antennas [5].

Since the ASTUSat's mission is to take the best possible quality color pictures of the earth from the low earth orbit (LEO), its payload is the camera. The commercial NanoCam C1U camera is selected for the mission [5].

### **3.1.2 Thermal requirements**

As discussed previously, the thermal control system must maintain spacecraft components within their operational temperature ranges throughout the mission of the ASTUSat. The operating temperature limit of components is the range of temperature where the components function well. If a components temperature exceeded this range it might not work well or it could be damaged

permanently. The operating temperature range of ASTUSat components are listed in Table 3-1 which are determined based on the manufacturer's recommendations on their datasheets.

Table 3-1: ASTUSAT subsystem assembly operating temperature limit

| Subsystem         | Assembly                    | Qty | Mass [g] | Operating Temp. (°C) | Reference |
|-------------------|-----------------------------|-----|----------|----------------------|-----------|
| <b>C&amp;DH</b>   | Motherboard                 | 1   | 77       | -40 to +85           | [14]      |
|                   | PPM                         | 1   | 11       | -40 to +85           | [15]      |
| <b>EPS</b>        | Deployable Solar panel      | 4   | 3769     | -40 to +80           | [16]      |
|                   | Body fixed solar panel      | 1   | 40       | -40 to +80           |           |
|                   | Battery with daughter board | 2   | 142      | -10 to +50           |           |
|                   | EPS board                   | 1   | 75       | -40 to +85           |           |
| <b>GNC</b>        | Reaction wheel              | 3   | 270      | -40 to +85           | [17]      |
|                   | Electromagnet               | 3   | 54       | -40 to +85           |           |
|                   | Driver board                | 1   | 26       | -40 to +85           |           |
|                   | Controller board            | 1   | 47       | -40 to +80           |           |
|                   | Static Earth sensor         | 1   | 66       | —                    |           |
|                   | ADCS Housing                | 1   | 233      | -40 to +80           |           |
|                   | ADCS Interface Module       | 1   | 49       | -40 to +85           |           |
|                   | GPS receiver                | 1   | 106      | -40 to +80           | [18]      |
| <b>Comms</b>      | VHF/UHF transceiver         | 1   | 85       | -20 to +60           | [19]      |
| <b>Payload</b>    | Camera                      | 1   | 277      | 0 to +60             | [20]      |
| <b>Structure</b>  | Side wall                   | 1   | 447      | -40 to +80           | [21]      |
|                   | Cover plate                 | 1   | 50       | -40 to +80           |           |
|                   | Base plate                  | 1   | 50       | -40 to +80           |           |
| <b>Total mass</b> |                             |     | 3769     |                      |           |

### 3.1.3 Internal heat generation of subsystems

At all times during the mission, heat is dissipated by various electronics within the ASTUSat satellite bus. Internal heat dissipation can be obtained by assuming all of the consumed power is

dissipated in the form of heat. The power dissipated by the electronic components are summarized in the Table 3-2 based on the operating mode of the satellite.

Table 3-2: Power consumption of ASTUSat

| <b>Sub-system</b>             | <b>Assembly</b>     | <b>Standby mode [W]</b> | <b>Mission [W]</b> | <b>Reference</b> |
|-------------------------------|---------------------|-------------------------|--------------------|------------------|
| <b>C&amp;DH</b>               | Motherboard         | 0.5                     | 0.5                | [5]              |
| <b>EPS</b>                    | EPS board           | 0.1                     | 0.1                | [16]             |
| <b>ADACS</b>                  | Reaction wheel (1)  | 0                       | 0.45               | [17]             |
|                               | Electromagnet (1)   | 0                       | 0.45               |                  |
|                               | Driver board        | 0.15                    | 0.15               |                  |
|                               | Controller board    | 0.45                    | 0.45               |                  |
|                               | GPS board           | 0.25                    | 1.3                | [18]             |
| <b>COMM</b>                   | VHF/UHF transceiver | 0.2                     | 1.55               | [19]             |
| <b>Payload</b>                | Camera              | 0.38                    | 0.8                | [20]             |
| <b>Total dissipated power</b> |                     | 2.03                    | 5.75               |                  |

### 3.1.4 Orbit profile

ASTUSat will be deployed into orbit from International Space Station (ISS). CubeSat launched from ISS attain an elliptical orbit with an apogee 380-420 km and inclination of 51.6°, with the satellite life expectancy of 100 – 250 days [22]. But in order to determine this elliptical orbit the eccentricity, RAAN, apogee, and the argument of perigee are still unknown because the parameters are dependent on the day of deployment which is not fixed yet. Therefore, throughout this thesis it is assumed that the ASTUSat's orbit is circular orbit with inclination and altitude of 51.6° and 400 km respectively which is same as the ISS orbit. At this altitude the satellite takes 92.6 minutes to complete one full revolution around the earth.

The ASTUSat position and velocity vectors are computed using open source MATLAB functions in reference [23] "Algorithm 3-1 and 4-5". The functions are combined together to uses given values of inclination, RAAN, argument of perigee, maximum altitude, eccentricity and time as an input and computes the position vector, velocity vector and vector normal to both vectors (the MATLAB code is provided in Appendix A).

### 3.2 Defining satellite orientation and labeling external surfaces

In this thesis, two reference frames are used to locate the satellite in space which are the Geocentric reference frame, and Body reference frame. The Geocentric reference frame is an Earth-centered system of locating astronomical objects in the Solar System in three dimensions along the Cartesian X, Y, and Z axes and used in order to define size, shape, and orientation of the orbit of the satellite. The other reference frame is the body reference frame which is used to define the orientation of the satellite. The center of this reference frame is located at the geometric center of the CubeSat where the  $+z$  axis points towards the position vector,  $+y$  axis points toward the velocity vector, and  $+x$  axis points to the direction normal to both axes and it is illustrated in the figure below.

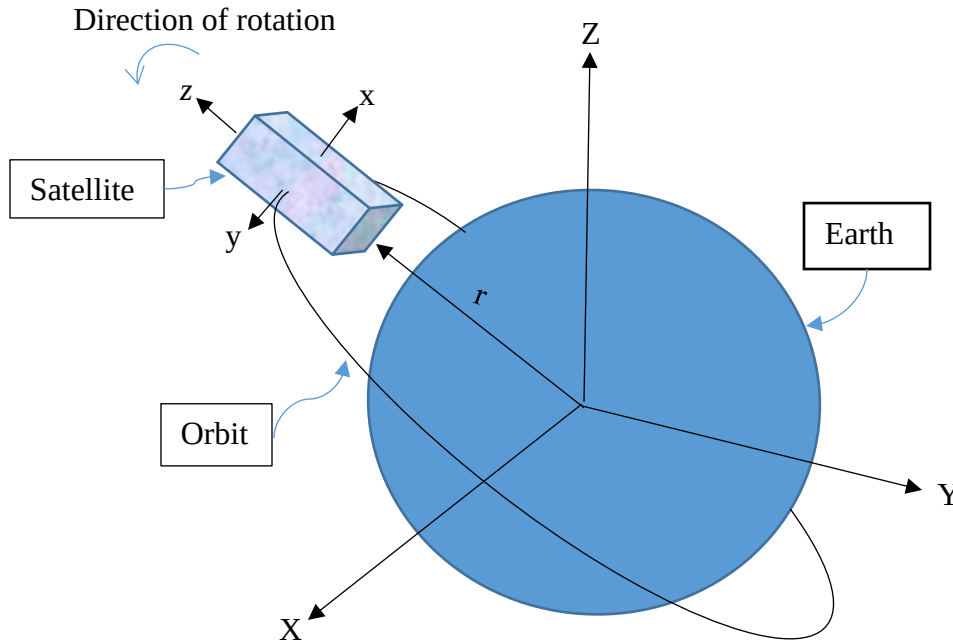


Figure 3-3: Reference frames

In this chapter, the ASTUSat is characterized by a simple mechanical architecture of six faced rectangular box as shown in Figure 3-3 with  $0.1 \times 0.1 \times 0.326$  m dimension (i.e. deployable solar panels are not considered). Each 6 faces of the CubeSat are labeled with numbers for identification purpose based on unit vectors normal to each face which are defined in body reference frame as

shown in Figure 3-1. The face which contains camera lens has to be oriented towards the earth which makes its unit vector points towards  $-z$  axis and it is labeled as “base plate”. The face opposite to the base plate which contains the cover plate and the fixed body solar panel is labeled as “FSP”. And the side wall’s face with normal unit vector pointing towards  $+y$  axis is labeled as “SW 1”, and to  $+x$  axis is labeled as “SW 3”. And finally faces which have opposite unit vectors to SW 1 and SW 3 are labeled as “SW 2” and “SW 1” respectively, as shown in Figure 3-1.

### 3.3 Environmental thermal radiation

The ASTUSAT is subjected to solar radiation, albedo, and earth infrared when it is in orbit. But these environmental radiations vary from one orbit to another and from season to season. Also, the incidence radiations also depend on the orientation of each external surface with respect to the thermal radiation sources. The next sub topics discuss about the environmental radiation of ASTUSAT orbit.

#### 3.3.1 Solar thermal radiation

Solar thermal radiation is a radiant energy emitted by the sun, particularly electromagnetic energy. The sun is a nearly spherical radiation source of  $1.39 \times 10^9$  m in diameter and emits approximately as a blackbody at 5777 K. Thus, the thermal radiated power intensity at the surface of the sun is calculated by using Stefan Boltzmann’s law:

$$S_o = \sigma T^4 = 5.670 \times 10^{-8} (5777)^4 = 63.15 \times 10^6 \frac{W}{m^2} \quad 3-1$$

Where:  $S_o$  is the radiated power intensity from sun’s surface [ $W/m^2$ ],  $T$  is surface temperature of the sun [K], and  $\sigma$  is the Stefan-Boltzmann constant ( $\sigma = 5.67 \times 10^{-8} W/m^2 K^4$ ).

Electromagnetic radiation follows the inverse square law. That is, if  $R_0$  is the radius of the sun, then the radiation intensity  $S$  at a distance  $R$  from the sun’s center is [23]:

$$S = S_o \left( \frac{R_o}{R} \right)^2 \quad 3-2$$

Because Earth’s orbit is elliptical, the intensity of sunlight reaching Earth varies approximately 3.5%, depending on Earth’s distance from the sun. At summer solstice, Earth is farthest from the

sun, and the intensity is at its minimum value of 1322 W/m<sup>2</sup>; at winter solstice, the intensity is at its maximum of 1414 W/m<sup>2</sup> [7].

Because the sun distance from the earth is extremely large, the rays coming from the sun are assumed to be parallel. The incident solar radiation on each face of the satellite is simply obtained by multiplying the incident solar flux with the dot product of unit vector normal to the surface and unit vector pointing from satellite to sun direction [9];

$$q''_{si} = S(\vec{n}_i \cdot \vec{r}_{sun}) \quad 3-3$$

From dot product rule, Equation 3-3 can be written in the following form

$$q''_{si} = S \cos(\gamma) \quad 3-4$$

Where:  $q''_{si}$  is incidence solar radiation on surface  $i$  per unit area [W/m<sup>2</sup>],  $S$  is the solar constant [W/m<sup>2</sup>],  $\vec{n}_i$  is unit vector normal to surface  $i$ ,  $\vec{r}_{sun}$  is unit vector pointing from satellite to sun direction,  $\gamma$  is the angle between the two-unit vectors

When solving Equation 3-4 two things should be considered:

- The incidence solar radiation on the satellite surface is zero when the satellite is in eclipse.
- The incidence solar radiation on the face of the satellite which is opposite to the face illuminated to solar radiation is zero.

The amount of solar radiation absorbed by the satellite's surface as a thermal radiation is calculated by simply multiplying Equation 3-4 with surface absorptivity factor and area of the satellite's surface.

$$q_{si} = \alpha_i S A_i \cos(\gamma) \quad 3-5$$

And,

$$Q_{stot} = \sum_{i=1}^n q_{si} \quad 3-6$$

Where:  $Q_{stot}$  is total absorbed solar radiation by the satellite [W],  $q_{si}$  is absorbed solar radiation by surface  $i$  [W],  $\alpha_i$  is absorptivity of surface  $i$ ,  $A_i$  is area of surface  $i$  [m<sup>2</sup>], and  $n$  is number of surfaces exposed to solar radiation.

### 3.3.2 Albedo

The albedo is expressed as the fraction of incident sunlight that is reflected back to space from the earth surface and environment, and it is highly variable. It is dimensionless and measured on a scale from 0 (corresponding to a black body that absorbs all incident radiation) to 1 (corresponding to a body that reflects all incident radiation). A satellite is always exposed to the reflected sunrays from the planet when it is not under eclipse. The albedo (the reflectivity of the planet) is greater over continental regions than oceanic regions and general increases with decreasing local solar-elevation angles and increasing cloud coverage [7].

The amount of reflected solar radiation that is incident on the satellite orbiting earth fluctuates continuously. Since the most extreme values for albedo only persist for very short periods of time (tens of seconds), it would be overly conservative to assume these extreme values occur continuously, and therefore, for thermal analysis an average value is used. The recommended albedo average value per orbit for thermal analysis is 0.4 during winter solstice and 0.19 during summer solstice for satellite orbiting earth with inclination 51.6° at 400 km altitude [24].

Assuming the earth surface as a diffuse surface, the amount of albedo incidence on the satellite's face per unit area is calculated by multiplying the incidence solar radiation flux with albedo coefficient and satellite earth view factor. Thus, the albedo can be expressed using the following equation [9].

$$q''_{ai} = \begin{cases} S a F_{i \rightarrow e} [\cos(0.9\theta)]^{1.5} & \text{when } -\frac{\pi}{2} < \theta \leq \pi/2 \\ 0 & \text{when } \pi/2 < \theta \leq 3\pi/2 \end{cases} \quad 3-7$$

Where:  $q''_{ai}$  is albedo incidence on surface  $i$  per unit area W/m<sup>2</sup>,  $S$  solar radiation intensity,  $a$  is the albedo factor,  $F_{i \rightarrow e}$  is surface  $i$  to earth view factor, and  $\theta$  is the angle between the earth sun vector and earth satellite vector. The 0.9 coefficient in the cosine means that even if the satellite has just come above the shadow part of the earth, reflected rays are still hitting the satellite. But the diffused albedo is zero for most of the shadow part of the earth.

To use Equation 3-7 the only term that is unknown is the satellite to earth view factor. Thus, from reference [25] simplified set of equations are used to calculate the view factor between a small planar surface tilted at an angle  $\psi$  with a sphere of radius  $r$ . Therefore, in this thesis these equations

of view factors are used to calculate the view factor by assuming that the spacecraft's surface as small planar surface and earth as spherical surface (Figure 3-4).

The surface is at a distance  $\|r_{sc}\| = h + r$  (km) from the center of the earth.

$$H = \frac{\|r_{sc}\|}{r} \quad 3-8$$

$$\phi = \cos^{-1}\left(\frac{1}{H}\right) \quad 3-9$$

Where:  $h$  is the altitude of the satellite from the earth surface,  $r$  is radius of the earth. Thus, the angle  $\phi$  is compared with  $\psi$  to determine whether all, part, or none of the earth is in view of the surface. And there are three cases each yield a different view factor from the surface to Earth.

Case 1 – The surface sees all of earth if:  $|\psi| \leq \phi$

$$F_{i \rightarrow e} = \frac{\cos \psi}{H^2} \quad 3-10$$

Case 2 -The surface sees part of Earth if:  $\psi > \phi$  and  $\psi < (\pi - \phi)$

$$F_{i \rightarrow e} = \frac{1}{\pi h^2} \left( \cos \psi \arccos y - x \sin \psi \sqrt{1 - y^2} \right) + \frac{1}{\pi} \arctan \left( \frac{\sin \psi \sqrt{1 - y^2}}{x} \right) \quad 3-11$$

$$\text{With: } x = \sqrt{H^2 - 1}, \quad y = -x \cot \psi$$

Case 3 – The surface does not see Earth

$$F_{i \rightarrow e} = 0 \quad 3-12$$

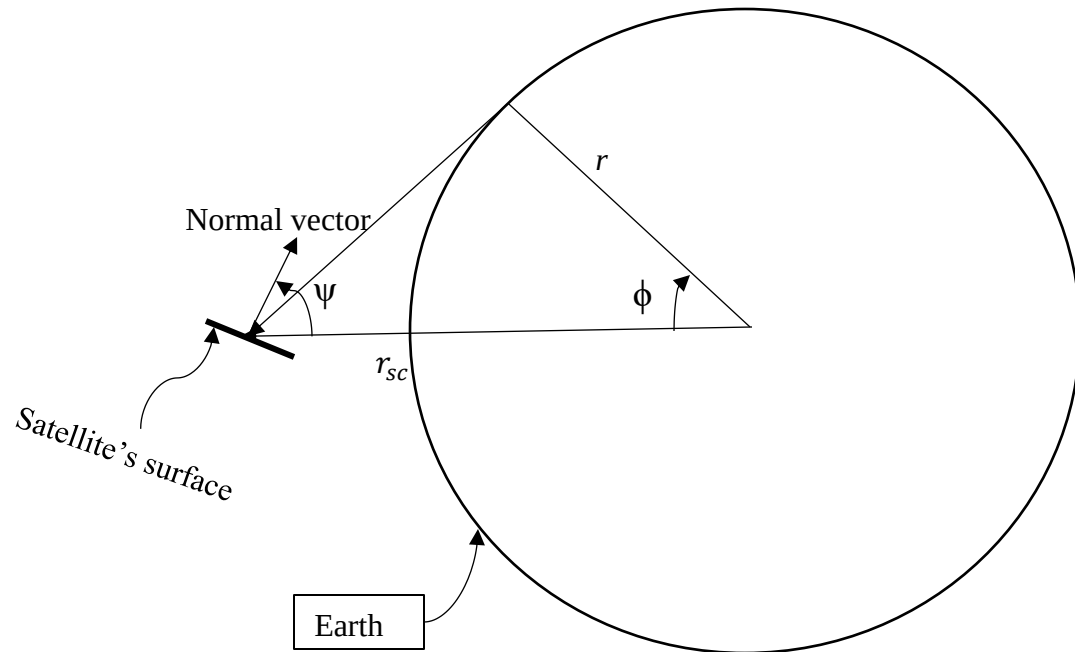


Figure 3-4: View factor between satellite's surface and Earth

Using the above three cases the view factor versus  $\psi$  is calculated and shown in the Figure 3-5:

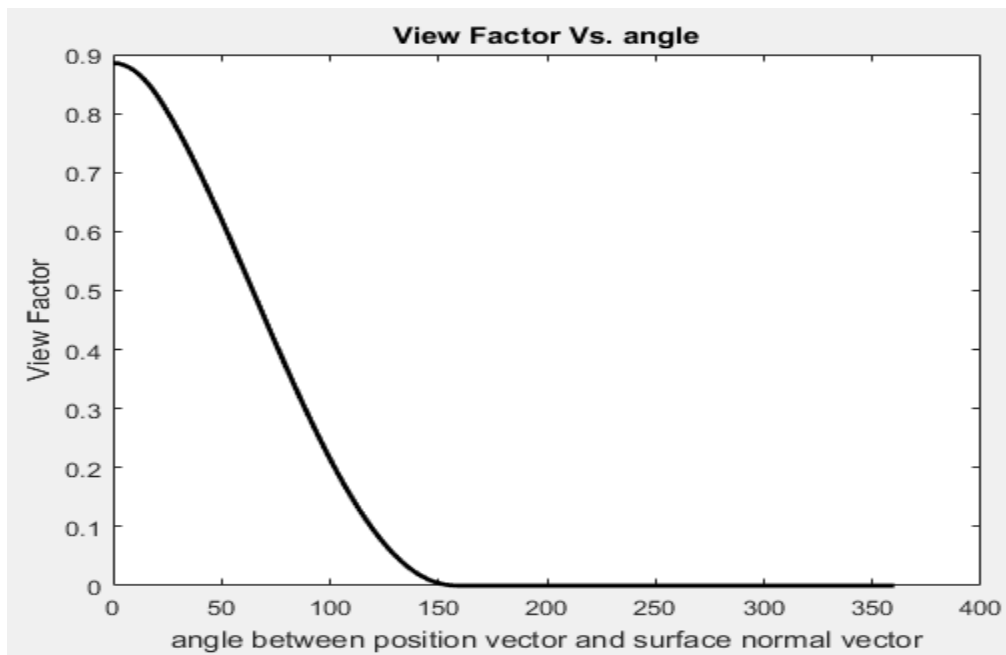


Figure 3-5: View factor between small flat surface and sphere

Now, the albedo radiation absorbed by the satellite's surface can be calculated using the following equation:

$$q_{a\ i} = \begin{cases} \alpha_i A_i S_a F_{i \rightarrow e} \cos \theta & \text{when } -\pi/2 < \theta < \pi/2 \\ 0 & \text{when } \pi/2 \leq \theta \leq 3\pi/2 \end{cases} \quad 3-13$$

And,

$$Q_{a\ tot} = \sum_{i=1}^n q_{a\ i} \quad 3-14$$

Where:  $Q_{a\ tot}$  is total absorbed albedo by the satellite [W],  $q_{a\ i}$  is the absorbed albedo by surface  $i$  [W],  $A_i$  is Surface area of the satellite [m<sup>2</sup>], and  $\alpha_i$  is surface  $i$  absorptivity factor

### 3.3.3 Earth infrared

The last environmental thermal radiation is the infrared emitted by the earth. The earth infrared varies extremely when the earth rotates around the sun, however these extreme values only persist for very short periods of time. Thus, for thermal analysis an average value is used. The recommended average earth infrared value per orbit for thermal analysis is 221.5 W/m<sup>2</sup> (or 250K earth temperature by considering earth as a blackbody) during winter solstice and 260 W/m<sup>2</sup> (260K) during summer solstice for satellite orbiting earth with inclination 51.6° at 400 km altitude [24]. From the selected values of Earth infrared and albedo coefficients it can be observed that the value of earth infrared is minimum during winter season and maximum during summer season which is vice versa for albedo coefficient. This occurs due to the fact that an Earth surface that is highly reflective will not absorb a lot of energy from solar radiation, causing it to be relatively cold and consequently the Earth IR will be lower.

In incidence infrared calculation, for the sake of simplicity it is assumed that the satellite will always absorb energy from Earth but will not expel energy to it. This simplification is balanced by assuming that the satellite radiates to space with an unobstructed view (i.e., ignoring the fact that Earth partially blocks the satellite's view of space). Now, to calculate the amount of IR radiation incidence on the satellite's  $i^{th}$  surface per unit area the following equation is used [9]:

$$q''_{e\ i} = IR \times F_{i \rightarrow e} \quad 3-15$$

Or

$$q''_{ei} = \sigma F_{i \rightarrow e} T_e^4 \quad 3-16$$

Where:  $q''_{ei}$  is incidence infrared on surface  $i$  [ $\text{W/m}^2$ ],  $IR$  is average infrared emitted by Earth due to its temperature [ $\text{W/m}^2$ ],  $T_e$  is Earth temperature [K], and  $F_{i \rightarrow e}$  is surface  $i$  to earth view factor

Since the amount of incidence infrared on each satellite's external surface is now can be calculated, the amount of Earth infrared absorbed by the  $i^{\text{th}}$  surface of the satellite surface can be calculated simply by multiplying Equation 3-15 with surface emissivity and surface area:

$$q_{ei} = \varepsilon_i A_i IR \times F_{i \rightarrow e} \quad 3-17$$

And the total absorbed IR by the satellite is given by:

$$Q_{IR \text{ tot}} = \sum_{i=1}^n q_{ei} \quad 3-18$$

Where:  $q_{ei}$  is absorbed earth infrared by surface  $i$  [W],  $Q_{IR \text{ tot}}$  is absorbed total infrared by the satellite,  $\varepsilon_i$  and  $A_i$  are the emissivity and area of surface  $i$  [W]

Therefore, in order to determine the incidence environmental fluxes on the six external faces of the satellite, MATLAB function is written by using the above equations (MATLAB code is provided in Appendix A). In this function when the satellite is in eclipse the solar flux and the albedo are automatically set to zero because the earth shields the satellite from the sun.

There is an accurate method used to determine whether the satellite is in eclipse or not, and it is called the line of sight (LOS) method, developed by Alfano in 1991 [23]. In this method if the line between the satellite and the sun intersects the earth then there is no line of sight otherwise it is in sunlight, for more detail refer [23]. Therefore, to determine whether the satellite is eclipsed or not a user defined MATLAB function code is written using the open source code 'ALGORITHM 12.3' in reference [23] which utilizes the LOS method.

### 3.4 Thermal energy balance

For the preliminary analysis, thermal mathematical model is developed for the satellite on the basis of closed system (thermal energy balance). The closed system is chosen because when the satellite is in orbit it becomes an isolated system which only exchanges heat energy between the

surrounding through radiation. Also, the satellite is assumed to be isothermal body. The thermal energy balance equation is expressed in the equation below:

$$\dot{E}_{st} = \dot{E}_{in} - \dot{E}_{out} + \dot{E}_g \quad 3-19$$

Where:  $\dot{E}_{st}$  is thermal energy stored in a system over time,  $\dot{E}_{in}$  and  $\dot{E}_{out}$  is the rate of energy entered and leaving the system respectively, and  $\dot{E}_g$  is the rate of heat energy generated in the system.

The thermal energy storage term  $\dot{E}_{st}$  represents the thermal mass of the satellite and is simply the change in thermal energy storage due to an environmental temperature change. The energy storage, is expressed by the following relation:

$$\dot{E}_{st} = mc_p \frac{dT}{dt} \quad 3-20$$

Where  $m$  is the satellite's mass,  $c_p$  is the specific heat, and  $dT/dt$  is the change in temperature with respect to time. It is clear that the energy storage is associated with the rate of change in the internal thermal energy.

The heat energy input term  $\dot{E}_{in}$  represents the rate of environmental thermal radiation absorbed by the satellite external surface which is responsible for increasing the overall temperature of the satellite. It can be defined mathematically as the summation of solar radiation, albedo, and Earth infrared absorbed by external surface of the satellite and expressed in the Equation 3-21:

$$\dot{E}_{in} = Q_{s\ tot} + Q_{a\ tot} + Q_{IR\ tot} \quad 3-21$$

Where:  $Q_{s\ tot}$ ,  $Q_{a\ tot}$ ,  $Q_{IR\ tot}$ , are solar radiation, albedo, and Earth Infrared absorbed by the satellite

From the energy balance equation, the term which is responsible for decreasing the overall temperature of the satellite is the heat energy output term,  $\dot{E}_{out}$ . It represents the rate of heat energy exchanged between the spacecraft and deep space through radiation. The deep space is the main source of the cold and can be seen as a black body emitting at 2.7K [26]. This temperature represents the radiation of the stars, the galaxies and the Cosmic Microwave Background. Thus,

the total amount of the rate of radiation exchanged between the satellite and deep space can be calculated using the Equation 3-22.

$$\dot{E}_{out} = \varepsilon A \sigma (T_s^4 - T_\infty^4) \quad 3-22$$

Where:  $\varepsilon$  is effective emissivity of the satellite,  $A$  is total surface Area,  $T_s$  is satellite temperature,  $T_\infty$  is deep space temperature (2.7K), and  $\sigma$  is the Stefan-Boltzmann constant ( $\sigma = 5.67 \times 10^{-8}$  W/m<sup>2</sup>.K<sup>4</sup>).

The forth term of the thermal energy balanced equation is the rate of heat energy generated within the satellite,  $\dot{E}_g$ .

Therefore, from the above discussions the energy balance equation can be written by substituting Equation 3-20, 3-21, and 3-22 in Equation 3-19.

$$mc_p \frac{dT}{dt} = Q_{s\ tot} + Q_{a\ tot} + Q_{IR\ tot} - \varepsilon A \sigma (T_s^4 - T_\infty^4) + \dot{E}_g \quad 3-23$$

To ascertain the level of thermal control needed to satisfy temperature requirements the following analytical methods have been conducted: a steady-state energy balance, and a transient energy balance.

### 3.4.1 Steady state thermal analysis

Steady state thermal analysis is performed by assuming the satellite temperature and the heat flux at a point in a body will not change with time, and due to the assumption of isothermal body the temperature and the heat flux will not vary from one point to another. In other word the energy storage term equals to zero in Equation 3-23:

$$0 = Q_{s\ tot} + Q_{a\ tot} + Q_{IR\ tot} - \varepsilon A \sigma (T_s^4 - T_\infty^4) + \dot{E}_g \quad 3-24$$

Now solving for  $T_s$

$$T_s = \left[ \frac{Q_{s\ tot} + Q_{a\ tot} + Q_{IR\ tot}}{\varepsilon \sigma A} + T_\infty^4 + \frac{\dot{E}_g}{\varepsilon \sigma A} \right]^{1/4} \quad 3-25$$

Where:  $T_s$  is the satellite temperature

The absorbed environmental fluxes in Equation 3-25 are the average value of environmental fluxes absorbed by the satellite throughout the orbit.

From Equation 3-25 we can observe for fixed value of internal heat dissipation, the temperature of the satellite is varied due to the effect of surface emissivity, and absorptivity. But any particular face of the ASTUSat is contains set of surfaces with a unique solar absorptivity and IR emissivity which depends on surface finish and material type. As we can see from Figure 3-1 no external face in its entirety is exclusively defined by a single set of optical properties. For this reason, a scheme was employed that average surface optical properties per face to obtain an effective optical property data set. Appendix B presents the averaging scheme used to obtain effective optical properties and the values of the effective optical properties of ASTUSat.

### 3.4.2 Transient Analysis

Due to the variation of environmental thermal radiation with respect to time and the effect of heat capacitance of the satellite, the amount of absorbed thermal radiation by the satellite is also varied with respect to time. Thus, in order to investigate the temperature evolution of the satellite through time the transient analysis is performed.

In this section, we are only interested in the evolution of the average temperature of the satellite through time, thus the lumped capacitance method is used. The essence of the lumped capacitance method is the assumption that the temperature of the body (satellite) is spatially uniform at any instant during the transient process. This assumption implies that temperature gradients within the satellite are negligible. It is certainly the simplest and most convenient method that can be used to solve transient heating and cooling problems. But in order the assumption of a uniform temperature distribution within the solid is reasonable the Biot number should be less than 0.1.

Where the Biot number is the ratio of an object's ability to conduct heat to the rate at which heat can be transmitted into or out of the object's surface. In other word, the biot number is nothing but the ratio of thermal resistance [27]. Thus, the ASTUSat biot number is the ratio of conductive resistance to radiative resistance and it can be expressed in the following form:

$$Bi = \frac{R_{cond}}{R_{rad}} = \frac{\left(\frac{L_c}{k}\right)}{\left[\frac{1}{\varepsilon\sigma(T_s + T_\infty)(T_s^2 + T_\infty^2)}\right]} \quad 3-26$$

Where:  $L_c$  is the characteristic length,  $k$  is the thermal conductivity,  $\varepsilon$  is the effective surface emissivity,  $T_s$  is the satellite surface temperature, and  $T_\infty$  is the deep space temperature.

In Equation 3-26 the thermal conductivity of Aluminum 6061 ( $k = 168 \text{ W/mK}$  [7]) is used to represent the overall material of the satellite because it is the most abundant material within the satellite. The characteristic length is also constant value which can be obtained by dividing the satellites total volume to total surface area. Therefore, the equation of Biot number is only dependent on the effective surface emissivity and surface temperature of the satellite.

The Equation 3-26 is solved in MATLAB for a variety of surface temperatures and effective surface emissivity values and shown in the Figure 3-6. The MATLAB code is available in Appendix A.

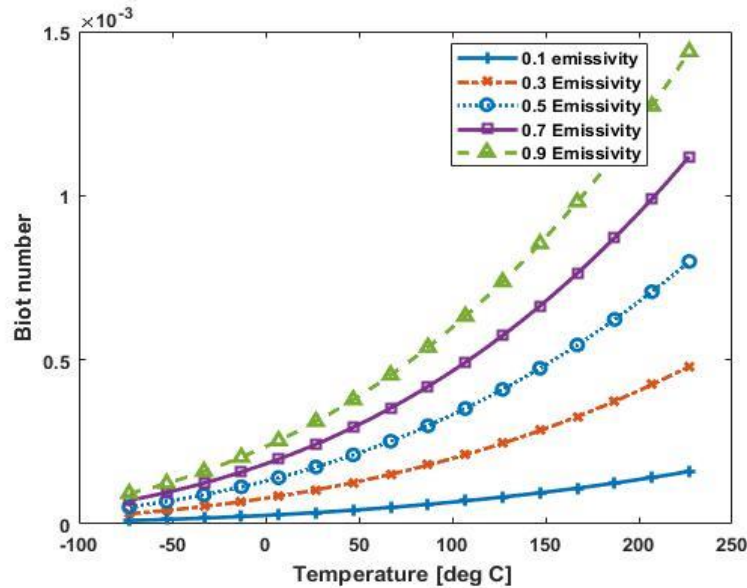


Figure 3-6: Biot number for multiple values of emissivity and temperature

In the Figure 3-6 for all those possible combinations of surface temperatures and emissivity the Biot number is much less than 0.1. Therefore, the assumption of isothermal satellite is valid to use for ASTUSat preliminary transient analysis.

Now to calculate the transient temperature of the satellite we use the energy balance Equation 3-23, and it can be expressed in the following form for a small value of  $\Delta t$  using the Euler's explicit method:

$$mc_p \frac{(T_2 - T_1)}{\Delta t} = Q_{s\ tot} + Q_{a\ tot} + Q_{IR\ tot} - \varepsilon A \sigma (T_1^4 - T_\infty^4) + \dot{E}_g \quad 3-27$$

Where:  $\Delta t$  is change in time,  $T_2$  is the temperature of the satellite at  $t + \Delta t$ ,  $T_1$  is the satellite temperature at  $t$ ,  $m$  is the total mass of the satellite,  $c_p$  is the specific heat of the satellite

For this preliminary thermal analysis, the total mass of the satellite is taken from Table 3-1 by subtracting the mass of the deployable solar panels. And the total specific heat of the satellite is assumed to be equal to the same as the specific heat of the material the structure is made off which is Aluminum 6061 ( $c_p = 961$  J/kgK [7]).

Now, solving Equation 3-27 for  $T_2$  we get an iterative equation:

$$T_2 = T_1 + \frac{\Delta t}{mc_p} (Q_{s\ tot} + Q_{a\ tot} + Q_{IR\ tot} - \varepsilon A \sigma (T_1^4 - T_\infty^4) + \dot{E}_g) \quad 3-28$$

### Solution procedures in MATLAB

So, for a given initial value temperature of the satellite, the temperature after  $\Delta t$  second will be obtained by using forward Euler explicit method. Thus, MATLAB program is written in this thesis that produce outputs like incidence environmental flux, steady state temperature, and transient temperature of the satellite for one orbit.

**Inputs:** The program requires inputs from the user listed below:

- Inclination
- Eccentricity
- Right ascension of the ascending node of the orbit (RAAN)
- Month [Either 6 or 12]
- Maximum altitude
- Spinning rate of the satellite [deg/orbit]
- Surface rotation angle from the velocity vector
- External surface area
- Thermal optical properties
- Total mass of the satellite
- Specific heat of the satellite

### Calculation method:

Using these inputs, the program computes the orbit and incidence fluxes on the six external faces of the satellite by using orbital equations for computing the position and velocity vectors, and

equations in section 3.3 to compute the incidence environmental fluxes. Then the program displays the orbit of the satellite and the incidence fluxes with respect to time for one orbit. After that the program computes each satellite's external face effective optical properties by using equations in Appendix B and the effective optical properties are multiplied by the averaged incidence environmental fluxes and surface area of the satellite to obtain the average absorbed environmental fluxes. Then using Equation 3-25 steady state temperature of the satellite is obtained. This temperature is used as initial temperature for starting the transient thermal analysis which reduces the amount of iterations required to obtain the steady state reached transient temperature of the satellite. The general flow chart of the program is given in Figure 3-7.

### **Convergence criteria**

The transient temperature is calculated in a cyclic repetition mode until the temperature at the end of the orbit is approximately equal to the temperature at the beginning of the orbit. If they are not equal, the initial temperature of the next orbit is updated with the final temperature of the previous orbit and the calculation performed again. Therefore, the transient temperature obtained from the program is the transient temperature of the satellite after reaching steady state.

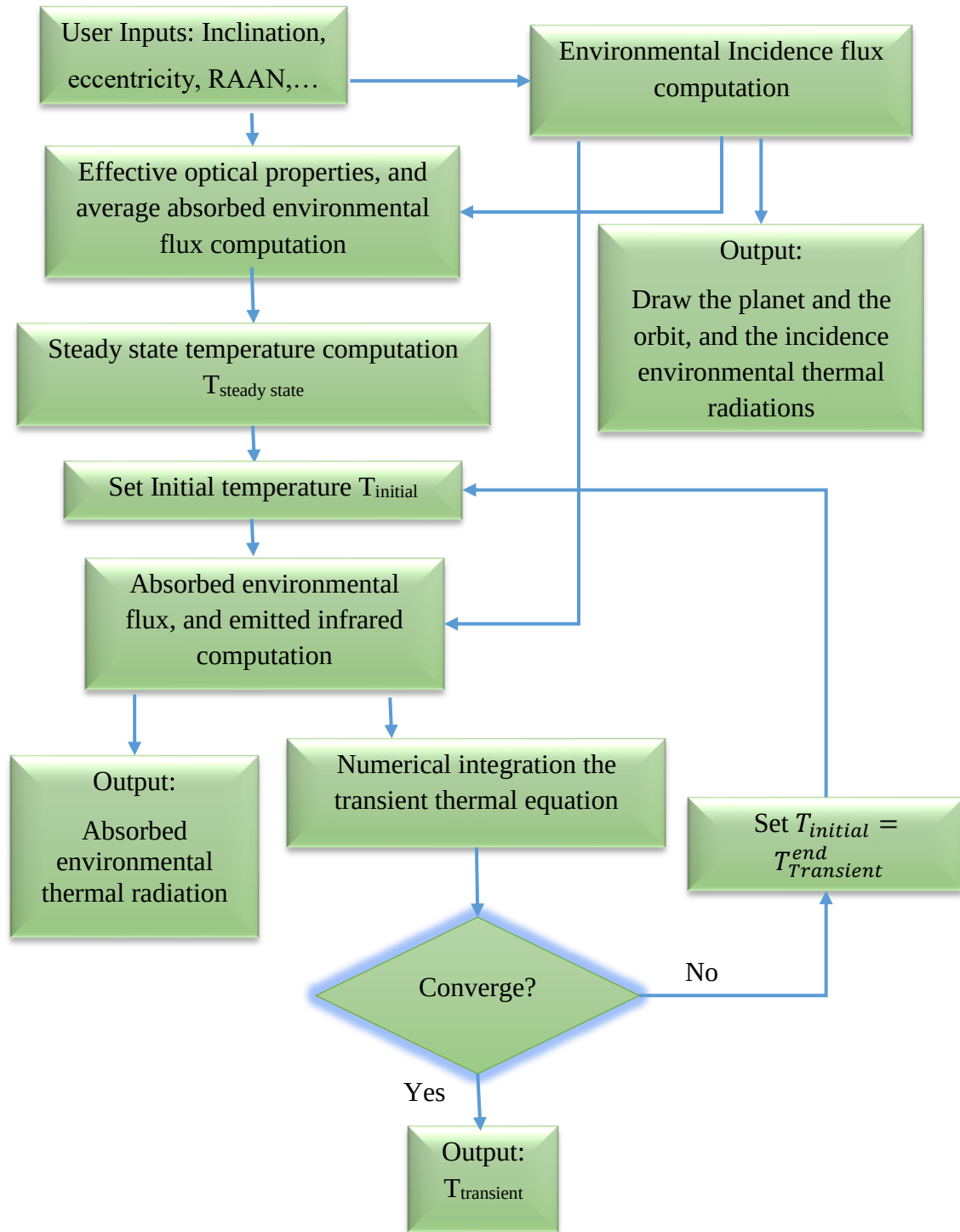


Figure 3-7: Flow chart of MATLAB program that solves the temperature of spacecraft ASTUSat

### 3.5 Validation

The MATLAB program written in this thesis was validated against the MATLAB/Simulink program written in reference [9]. The MATLAB/Simulink program is written in the document for performing preliminary analysis of 1U CubeSat called OUFTI-1. In the document, the CubeSat is modeled in the program as a non-rotating perfect black-body ( $\alpha = \varepsilon = 1$ ) cube pointing towards the sun on OUFTI-1's orbit ( $354 \times 1447 \text{ km} \times 71^\circ$ ), with a RAAN  $\Omega = 45^\circ$ , and argument of perigee  $\omega = 0^\circ$ . The main components heat capacities shown in Table 3-3 are also defined within the program as heat capacity of the cube. For the validation of the program in the document, the simplified model of OUFTI-1 is modeled and solved in ESATAN/ESARAD software's as well. Then the transient temperature, and the total absorbed radiation obtained from the two programs are plotted with respect of time (Figure 3-8).

Table 3-3: OUFTI-1s thermal heat capacity [9]

| Component       | Mass [g]      | Specific Heat [J/kgK] | Heat capacity [J/K] |
|-----------------|---------------|-----------------------|---------------------|
| aluminum frame  | 145           | 980                   | 142.1               |
| aluminum panels | 165           | 980                   | 161.7               |
| PCBs            | 350           | 1136                  | 397.6               |
| Antenna panel   | 33            | 980                   | 32.3                |
| Batteries       | 50            | 960                   | 42.3                |
| Total           | $\approx 698$ |                       | 776                 |

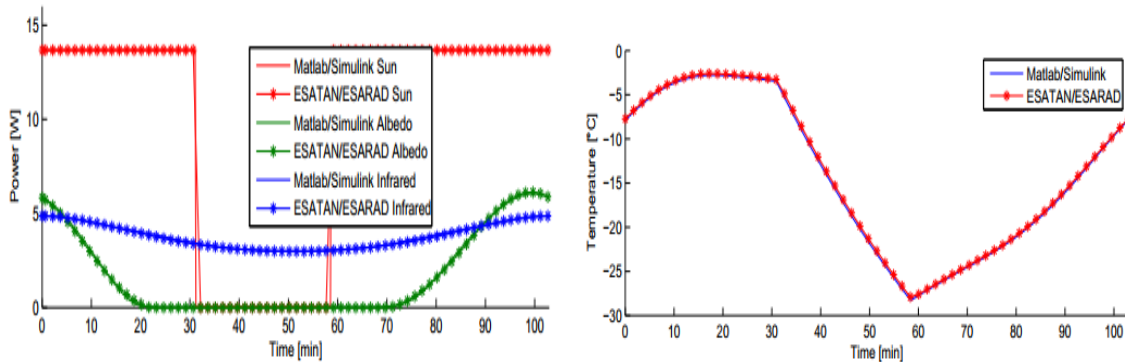


Figure 3-8: Total absorbed environmental power (left side) and transient temperature (right side) of the OUFTI-1 obtained from reference [9]

Now, the cube is modeled in the MATLAB code developed as a contribution of this thesis, and solved to get the transient temperature and the total absorbed environmental fluxes shown in Figure 3-9.

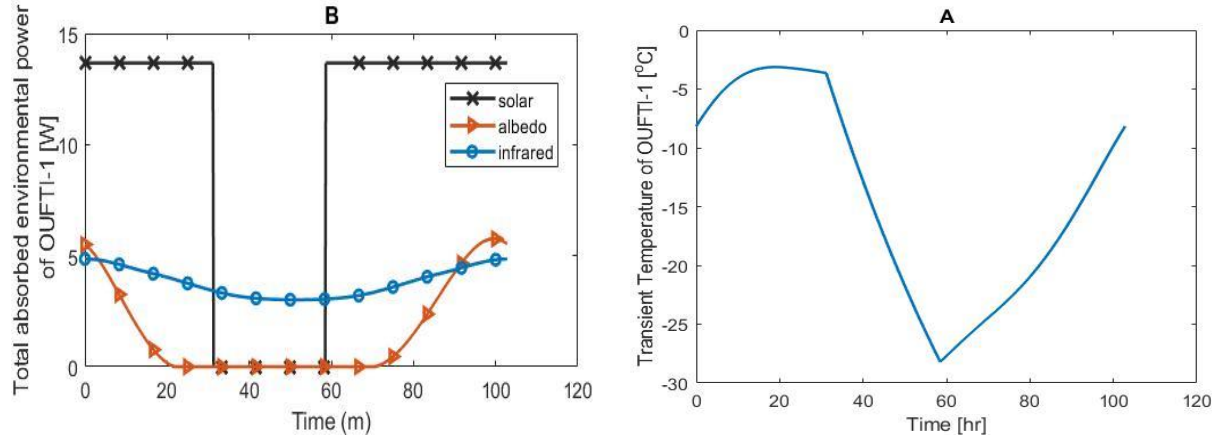


Figure 3-9: Transient temperature and total absorbed environmental power of OUFTI-1 calculated by the MATLAB program written as a contribution of this thesis

As we can see from the Figure 3-9 the results obtained from the MATLAB program do not show any significant difference between the results obtained from MATLAB/Simulink program results (Figure 3-8). Therefore, the MATLAB program written is valid to model and analyze the ASTUSat.

### 3.6 Results and Discussion

The thermal analysis is affected by the position of the earth on the ecliptic plane, defining the distance of the earth around the sun. As discussed in section 3.4. during winter solstice the incidence solar flux reach maximum value and during summer solstice reach minimum value. Thus, the thermal analysis is performed at these two extreme positions of the earth with respect to the sun, because it gives us the extreme temperature value of the satellite. Another parameter that affect the thermal analysis is the  $\beta$ -angle, which is the angle between the orbit plane and the Sun-Earth vector direction, and can vary from  $-90^\circ$  to  $90^\circ$  (in case of ASTUSat's orbit from  $-75^\circ$  to  $75^\circ$ ) (Figure 3-10). As the beta angle increases, the satellite is exposed to more sunlight per orbit, and eventually it will be in constant sunlight (in other words, there is no eclipse). And when the beta angle approaches  $0^\circ$  the eclipse time of the satellite increases and at  $0^\circ$  the satellite attains maximum eclipse.

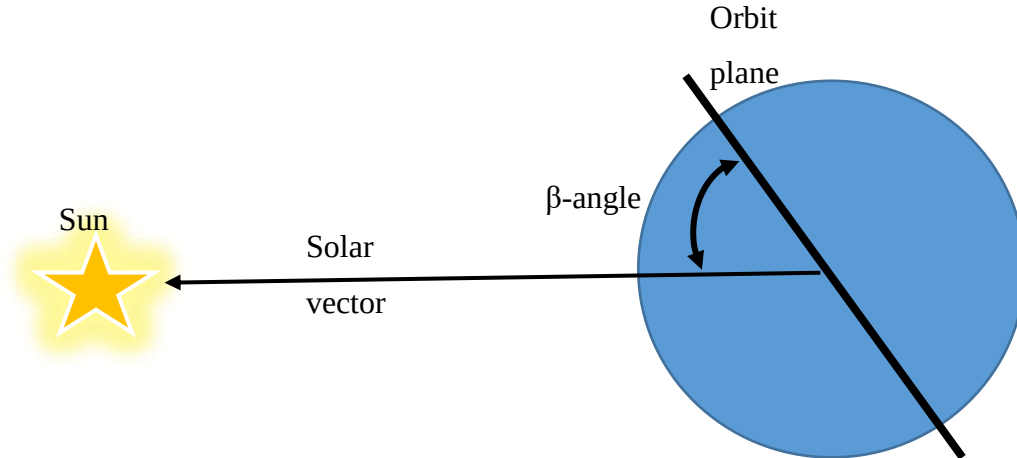


Figure 3-10: Satellite orbit beta angle

Another parameter that affect the temperature of the satellite is the orientation of the side wall from the velocity vector, as shown in Figure 3-11. When the unit vector normal to the side wall is at some angle  $\theta$  from the velocity vector the two faces of the side wall will be viewed from the sun for orbits with non-zero beta angle.

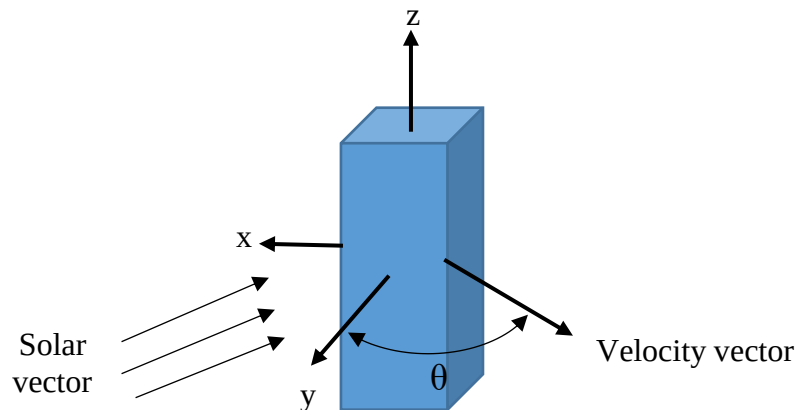


Figure 3-11: Side Wall orientation from the velocity vector

To determine the extreme conditions of the ASTUSat, the incidence environmental thermal radiations variation on each face of ASTUSat is computed and shown in Appendix D. Therefore, for thermal analysis of ASTUSat two boundary conditions, the worst hot and worst cold scenario are defined and the transient temperature evolutions of the satellite are calculated at these extreme conditions in the next sub topics;

### 3.6.1 Worst hot scenario (WHS)

At Worst hot scenario the satellite is exposed to maximum environmental heating flux and more sunlight per orbit. Thus, from Figure D-1 the satellite is subjected to such a kind of case when the earth is at winter solstice position, and the beta angle of the satellite's orbit is maximum (i.e.  $\beta=75^\circ$ ). The beta angle  $75^\circ$  maintains the satellite to be in sunlit throughout the revolution around the earth and it will expose the side wall faces to receive maximum solar flux. Also, in this scenario the maximum internal heat dissipation of the satellite is taken so that we can get the possible maximum temperature.

In this scenario the value of  $\theta$  (i.e. the angle between the unit vector normal to the side wall and the velocity vector) that extremizes the temperature of the satellite should be identified. Now, to find the value of  $\theta$  we substituted multiple values of  $\theta$  (i.e. between  $0-90^\circ$ ) within the MATLAB script and then the respective steady state temperature of the satellite is obtained and displayed in Figure 3-12. Therefore, from the figure we can see that the temperature of the satellite reaches peak value at  $\theta = 45^\circ$ .

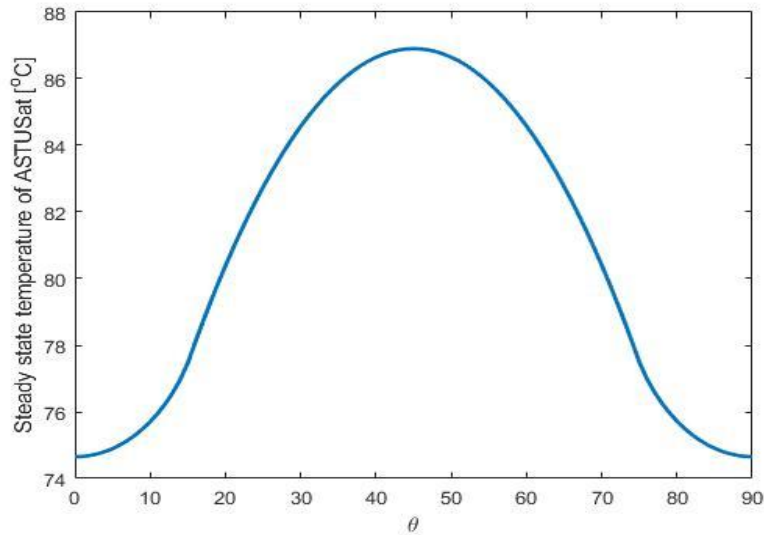


Figure 3-12: Steady state temperature of the spacecraft ASTUSat for different orientation of the satellite relative to the sun

Now, the WHS orbit of the satellite is fully defined and it is summarized in Table 3-4.

Table 3-4: WHS defining parameters

| Parameters                       | Unit             | Value |
|----------------------------------|------------------|-------|
| <b>Solar flux</b>                | W/m <sup>2</sup> | 1414  |
| <b>Albedo coefficient</b>        | —                | 0.4   |
| <b>Earth infrared</b>            | W/m <sup>2</sup> | 221.5 |
| <b>Altitude</b>                  | Km               | 400   |
| <b>Inclination</b>               | Degree           | 51.6  |
| <b>Beta angle</b>                | Degree           | 75    |
| <b>Side wall orientation</b>     | Degree           | 45    |
| <b>Internal heat dissipation</b> | W                | 5.75  |

Now, by substituting all values of Table 3-4 within the script the incident environmental thermal radiations on the six faces of the satellite, the total absorbed environmental thermal radiations and transient temperature of the satellite in WHS for one orbit after reaching its steady state temperature are computed. The incident environmental thermal radiations on each face of the satellite are shown on Appendix D in Figure D-13. The absorbed environmental thermal radiation and the isothermal temperature of the satellite are shown in Figure 3-13:

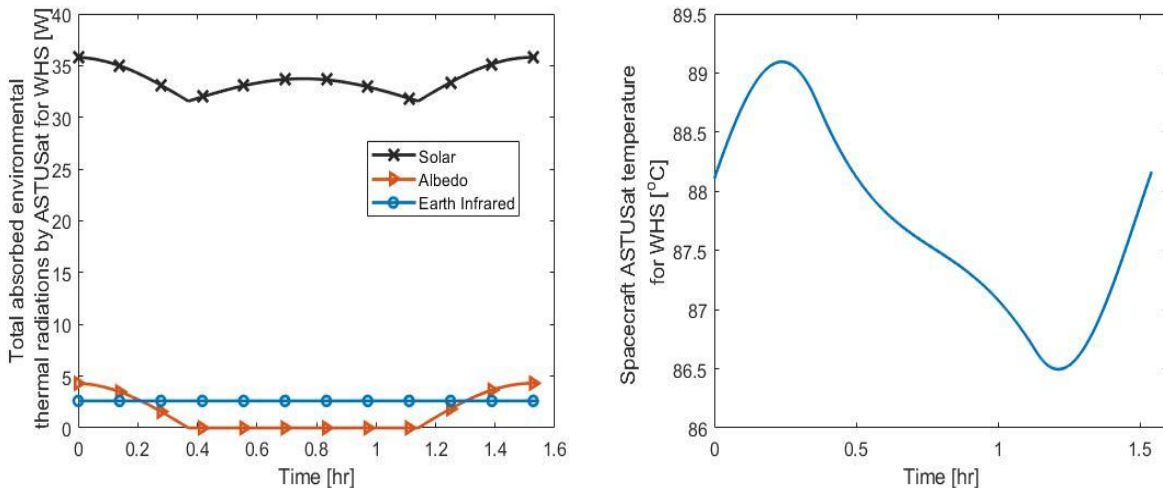


Figure 3-13: Total absorbed environmental thermal radiations and isothermal transient temperature for one orbit of ASTUSat during WHS

In Figure 3-13 we can see that the total solar radiation absorbed by the satellite reached peak value (35.61 W) at the beginning or at the end of the orbital time. This is because at this orbital position the three faces of the satellite (two side walls and fixed solar panel (FSP)) are exposed to solar radiation and due to high absorptivity of the FSP the total absorbed solar radiation is high. And the absorbed solar radiation is minimum at locations when the satellite crosses the equatorial plane because only two side walls are exposed to solar radiation at those locations. The albedo is also maximum at the beginning or ending of the orbit because the satellite is in daylit side and closer to subsolar point compared to other positions. But once the satellite crosses the equatorial plane and entered the dark night side of the earth the albedo is zero. The Earth infrared is constant (2.6 W) because it is not affected by the daylit side and dark night side of the earth and the satellite orientation do not change throughout the orbit.

From the Figure 3-13 we can see that the maximum temperature reached by the satellite is  $T_{\max} = 89.3\text{ }^{\circ}\text{C}$  which is not safe average temperature of the satellite according to the operating temperatures specified in Table 3-1.

In order to observe the effects of the external surface thermal optical properties on the overall satellite temperature, some surface finishes are selected according to their absorptivity per emissivity ratio and they are shown in Table 3-5.

Table 3-5: Surface finishes optical properties that can be used on the satellite [7]

| Surface finishes                                         | $\alpha/\epsilon$ | $\alpha$ | $\epsilon$ |
|----------------------------------------------------------|-------------------|----------|------------|
| <b>Chemglaze A276 white paint</b>                        | 0.26              | 0.23     | 0.88       |
| <b>Kapton, aluminized, 2 mil</b>                         | 0.45              | 0.34     | 0.75       |
| <b>Kapton, aluminized, silicon oxide coated, 0.5 mil</b> | 0.67              | 0.12     | 0.18       |
| <b>Kapton, aluminized, 5mil</b>                          | 0.74              | 0.61     | 0.83       |
| <b>Black anodized aluminum</b>                           | 1                 | 0.86     | 0.86       |
| <b>Kapton, black (carbon loaded), 1mill</b>              | 1.05              | 0.92     | 0.88       |
| <b>Tape, Aluminum</b>                                    | 2.5               | 0.1      | 0.04       |

The surface finishes in Table 3-5 are applied on the side wall of the satellite without changing the rails surface properties. Figure 3-10 shows the transient temperature of the satellite for each surface finishes specified in Table 3-5.

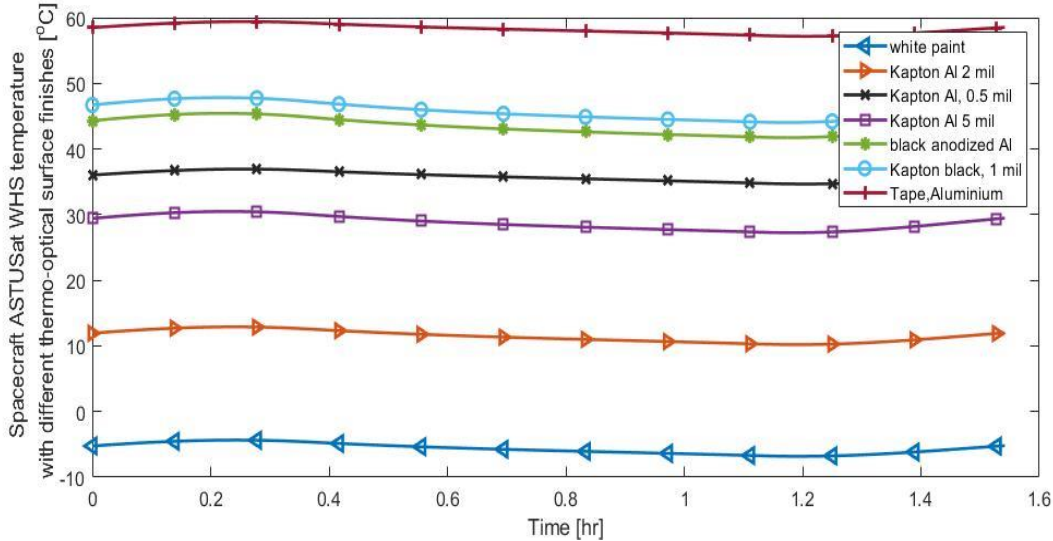


Figure 3-14: WHS transient temperature of ASTUSat for different side wall surface finishes verses time

From the figure we can see that as the surface  $\alpha/\epsilon$  increases the satellite temperature increases. This is because higher amount of solar radiation is absorbed and small amount of infrared emitted by the satellite as the ratio increases. But, from the figure there is exception with the Aluminized Kapton with silicon oxide coated which has higher temperature than the Aluminized Kapton 5mill even though the silicon oxide coated Kapton has smaller ratio. This is occurred because the emissivity of the silicon oxide coated Kapton has smaller emissivity which makes the satellite to emit infrared to the surrounding with smaller amount. Therefore, according to the Figure 3-13 the silicon oxide coated aluminized Kapton and the 5-mill aluminized Kapton maintain the ASTUSat within safe temperature during WHS.

### 3.6.2 Worst cold scenario (WCS)

The worst cold scenario orbit is the perfect opposite of the hot case, it corresponds to the case for which all parameters, both environmental and satellite related, are chosen in such a way that they all contribute to reach minimum temperature per orbit. Thus, from Figure D-1 the ASTUSat's

WCS orbit is when earth is at summer solstice, the beta angle equals to zero degree (maximum eclipse occurs), the internal heat dissipation of the satellite is minimum, and the side wall orientation from the velocity vector is zero. Thus, the parameters that define the WCS orbit are defined in the Table 3-6:

Table 3-6: WCS defining parameters

| <b>Parameters</b>                | <b>Unit</b>      | <b>Value</b> |
|----------------------------------|------------------|--------------|
| <b>Solar flux</b>                | W/m <sup>2</sup> | 1321         |
| <b>Albedo coefficient</b>        | —                | 0.19         |
| <b>Earth infrared</b>            | W/m <sup>2</sup> | 260          |
| <b>Altitude</b>                  | Km               | 400          |
| <b>Inclination</b>               | Degree           | 51.6         |
| <b>Beta angle</b>                | Degree           | 0            |
| <b>Side wall orientation</b>     | Degree           | 0            |
| <b>Internal heat dissipation</b> | W                | 2.03         |

Now, substituting all parameters in the MATLAB script we get the incidence environmental thermal radiations, the total absorbed environmental thermal radiations, and transient temperature of the satellite during WCS orbit. The incidence environmental thermal radiations are shown in Appendix D (Figure D-7), and the absorbed total environmental thermal radiations and the isothermal temperature of the satellite are shown in Figure 3-15.

From Figure 3-15 the ASTUSat absorbs less solar radiation during WCS compared to WHS. This is because the earth is located at the summer solstice. The solar radiation absorbed reached maximum value (25.4 W) when the satellites' two faces (i.e. fixed solar panel and one side wall) are exposed to solar radiation. It is also shown that when the satellite is in eclipse (i.e. when the satellite is shadowed by the earth from the sun from 0.47 to 1.08 hr.) the total solar radiation is zero. During WCS the absorbed albedo is higher than WHS because the satellite passes through the sub solar point. At the instant when the satellite passes through the sub solar point the satellite absorbs maximum albedo (5.9 W). The absorbed Earth infrared by the satellite do not change with

respect to time and it has a value of 3.05 W which is higher than from WHS because the incidence infrared during summer solstice is higher than winter solstice.

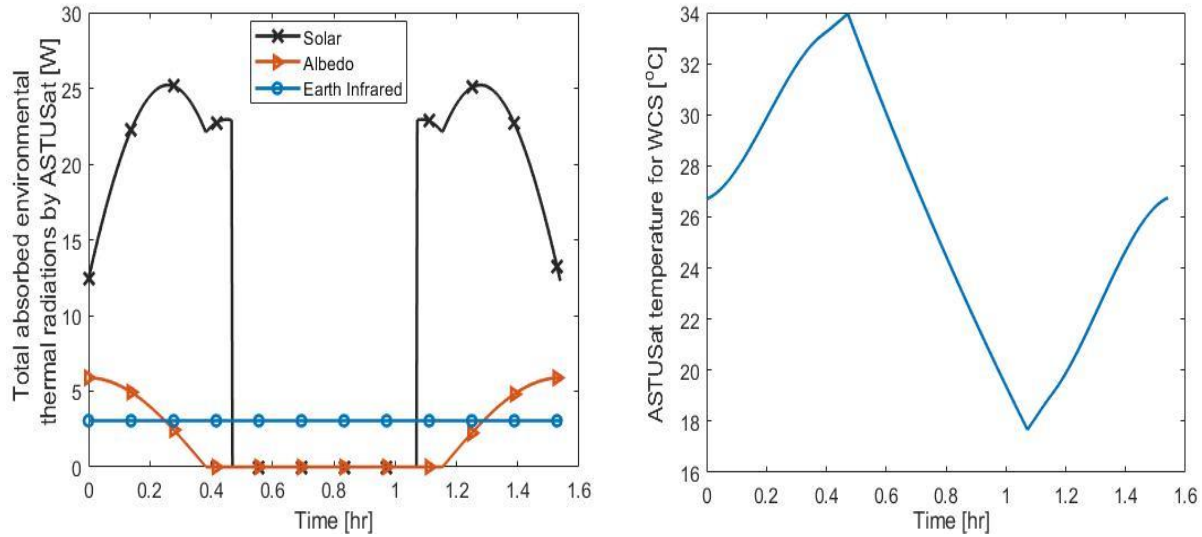


Figure 3-15: Total absorbed environmental thermal radiations and isothermal transient temperature for one orbit of ASTUSat during WCS

As we can see from Figure 3-15 the satellites' highest temperature 34 °C attained just before the satellite entered in to eclipse and the lowest temperature 17.64 °C is attained just after living the eclipse. Thus, according to the operating temperature of satellite's components in Table 3-1 all components are within their operating temperature range in the WCS.

Also, the transient temperatures of the satellite at this WCS is recalculated for surface finishes which are listed in Table 3-5, and shown in figure 3-16.

Here we also observe that the higher the optical properties ratio the higher the satellite temperature will get. But, one thing that is observed different here is that the satellite temperature dropped drastically during eclipse time for surface finishes which have maximum emissivity value (i.e. closer to 1). This is because when the satellite is in eclipse it is only subjected to weak planetary radiation (i.e. Earth infrared), thus for surface finishes having high emissivity value emit more radiation heat from the satellite and causes it to drop its temperature very fast. And that's why the temperature of the silicon oxide coated Kapton and the Tape aluminum temperature evolution is smoother than the rest of the surface finishes. Therefore, according to Figure 3-16 the 5-mil

aluminized Kapton do not maintain most of the ASTUSat components such as the UHF/VHF transvers board, camera, battery during WCS orbit. The silicon oxide coated aluminized Kapton maintain the satellite between the temperature range of  $-9.1^{\circ}\text{C}$  and  $1.5^{\circ}\text{C}$  which is only the payloads operating temperature is exceeded.

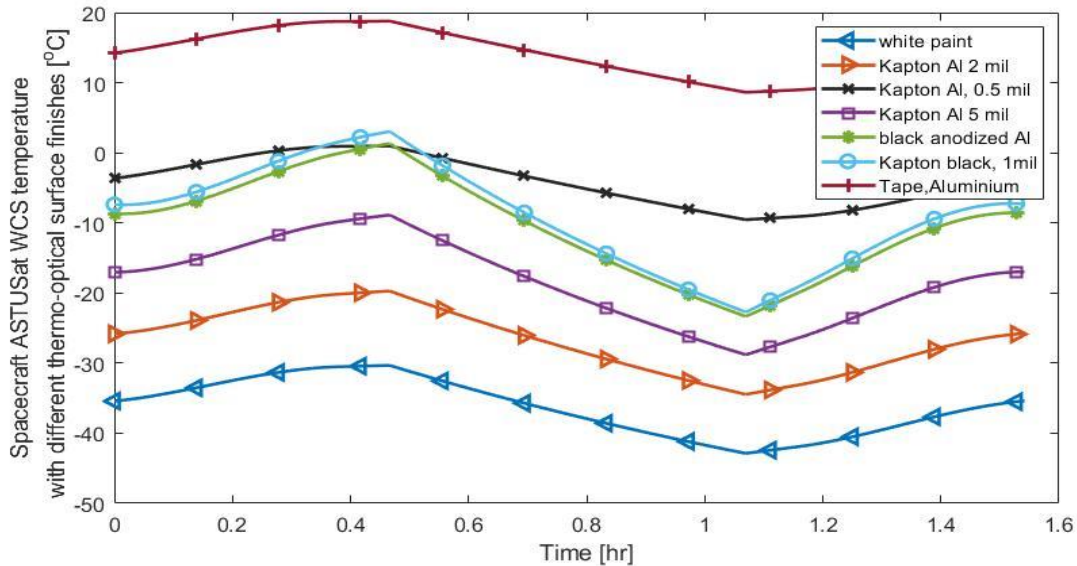


Figure 3-16: WCS transient temperature of ASTUSat for different side wall surface finishes

## Conclusion

In conclusion, the preliminary analysis gives a simple way to make a first estimation of the temperature limits the satellite might reach. The advantage of implementing bulk analysis is that it produces results that are reasonably accurate and fast first-order estimates. Due to the assumptions made this model has some major limitations. This approach does not consider the deployable solar panels that means the shielding effect, thermal coupling between hinges of solar panels with the satellite, and heat transfer related to the solar panels are not considered which makes the results gained not dependable. Another limitation is the reduction of the satellite to isothermal surface, and the assumption of the total power consumption of the satellite equal to the total heat generated within the satellite. The model gives no statements of local hot spots on outer and inside the spacecraft surfaces and the temperatures calculated might exceed locally prescribed temperature of the spacecraft's payload and subsystems. Thus, detail thermal modeling of the spacecraft ASTUSat is required to be analyzed.

# Chapter 4

## 4 Detail Thermal Modelling

The detail thermal modelling provides detail information about the temperature of the spacecraft components such as PCB, camera, etc. for the case of worst hot scenario and worst cold operating conditions. The finite element thermal model was developed using a commercial software FEMAP TMG for spacecraft ASTUSat. The chapter starts by giving the overall overview about the modelling process. Then, the thermal coupling, material thermal properties, internal heat generation definitions, and the boundary condition definitions within the model are discussed. Finally, the finite element thermal model of spacecraft ASTUSat is validated using the OUFTI-1's thermal performance.

### 4.1 Building finite element model

To simulate the spacecraft's components respective temperature evolution and distribution throughout the satellite components, a finite element model of the ASTUSat spacecraft is created in the FEMAP TMG software package and it is shown in the Figure 4-1 and 4-2.

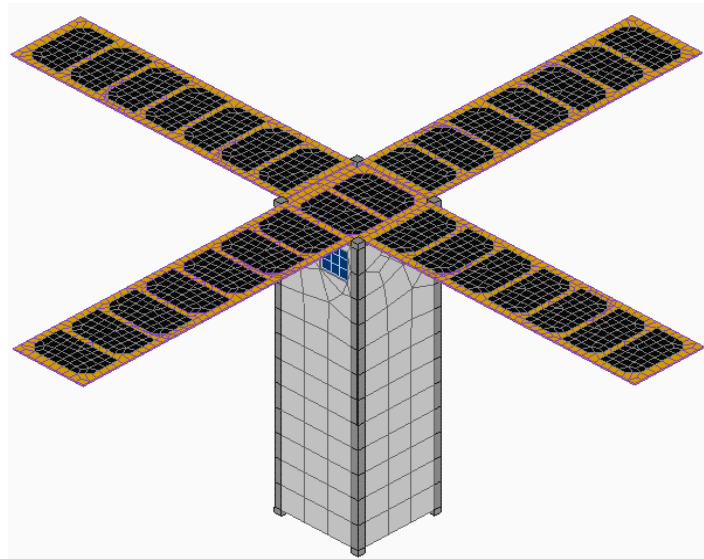


Figure 4-1: ASTUSat finite element model external feature developed in FEMAP TMG environment

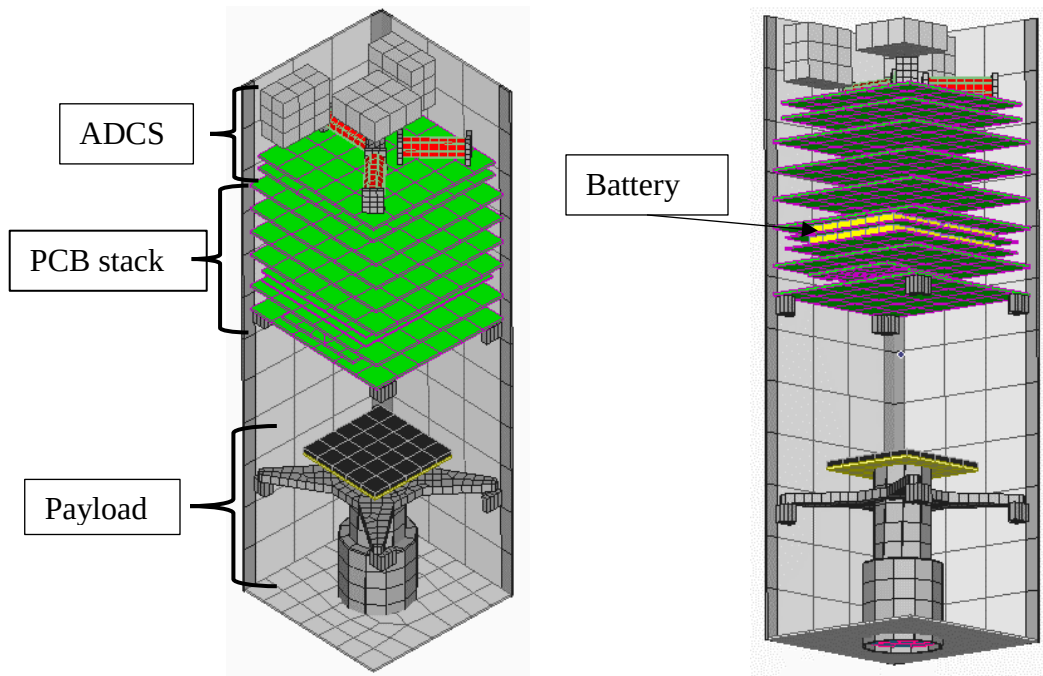


Figure 4-2: ASTUSat finite element model internal feature developed within FEMAP TMG environment

Before the FE (Finite Element) model is created, a simplified assembly CAD model (A model which do not contain the fasteners, and wire harnessing) of the ASTUSat is created in the CATIA V5 modeling software which is shown in the Figure 3-1 and 3-2 in the previous chapter. Then, the CAD model is imported in the FEMAP software environment and the finite element mesh is applied on it [Figure 4-1 and 4-2]. When developing the model, each satellite components are generally modelled as separate meshes from other components according to their shape and material properties. In the model, some irregular shape and voids of components are neglected to avoid node concentrations at those locations. Such a kind of model, minimize the number of nodes and elements of the overall model which minimizes the computational time. Also, for the same reason components like fasteners, connecting rods, spacers, and PCB connectors are not modeled explicitly. But, these components like other meshed components contribute in storing, conducting and radiating heat. Due to their small surface area and volume their contribution in radiative heat transfer is assumed to be negligible and their contribution in conductive heat transfer is considered. In order to account their conductive heat transfer, their respective thermal resistance is calculated in MATLAB and entered in the model as resistance thermal coupling.

Generally, the FE model contains 4907 and 3954 total number of nodes and elements respectively. To reduce the complexity of the model only 2D and 3D rectangular elements are used to model each assembled component of the satellite. The 2D elements are used to model components which have small thicknesses like structural panels and circuit boards because their temperature gradient along their thickness is almost negligible, and the 3D elements were used to model components like the heat sink, the batteries and the reaction wheels where temperature gradient along their thickness cannot be ignored.

## **4.2 Definition of thermal properties and thermal coupling**

To every element created in the model the corresponding thermal properties are assigned. The relevant properties are

- The density,
- The specific heat capacity,
- The thermal conductivity,
- The absorptivity,
- The emissivity,

In the following sub-sections, the thermal properties assigned to the model and the calculated thermal resistance are discussed in detail.

### **4.2.1 Structure**

The structure of ASTUSat is the standardized 3U structure of solid chassis walls made from aluminum 6061 material [5]. The structure contains the side walls, the cover plate, and base plates. On the side walls the rails are hard anodized to prevent friction during deployment and the rest of the structure is alodyned to enhance conductivity [21]. Table 4-1 listed the structural parts and their respective thermal properties.

Each side walls are connected to the cover plate and base plate by four M3 fastening screws (one at each side). These screws create a thermal contact phenomenon between the side walls and the cover and base plates. But it is hard to find the thermal contact conductance of screwed-joints because they are often developed within companies and rarely published. However, from Table 4-

2, which is taken from the spacecraft thermal control handbook by Gilmore [7], shows contact resistance data for various bolt size and plate thickness.

Table 4-1: Parts of the 3U structure and their thermal properties [7] and [28]

| Parts                        | Material and surface treatment     | $\rho$<br>$\left[\frac{kg}{m^3}\right]$ | $k$<br>$\left[\frac{W}{m K}\right]$ | $c_p$<br>$\left[\frac{J}{kg K}\right]$ | $\varepsilon$ | $\alpha$ |
|------------------------------|------------------------------------|-----------------------------------------|-------------------------------------|----------------------------------------|---------------|----------|
| <b>Side walls (SW)</b>       | Aluminum 6061, Alodined            | 2770                                    | 168                                 | 961                                    | 0.14          | 0.44     |
| <b>Rails</b>                 | Aluminum 6061, Black hard anodized | 2770                                    | 168                                 | 961                                    | 0.86          | 0.86     |
| <b>Cover and base plates</b> | Aluminum 6061, Alodined            | 2770                                    | 168                                 | 961                                    | 0.14          | 0.44     |

Table 4-2: Bolt thermal resistance estimate [7]

| Maximum Resistance versus Bolt Size and Plate Thickness ( $^{\circ}C/W$ bolt) <sup>a</sup> |                 |                  |                  |                  |                  |
|--------------------------------------------------------------------------------------------|-----------------|------------------|------------------|------------------|------------------|
| Steel Bolt                                                                                 |                 |                  |                  |                  |                  |
| Size                                                                                       | Shaft Diam (mm) | 1.57 mm Aluminum | 3.18 mm Aluminum | 6.35 mm Aluminum | 9.53 mm Aluminum |
| NC 4-40                                                                                    | 2.84            | 12.6             | —                | —                | —                |
| NC 6-32                                                                                    | 3.51            | 6.6              | 2.2              | —                | —                |
| NC 8-32                                                                                    | 4.17            | 4.5              | 1.5              | 0.75             | —                |
| NF 10-32                                                                                   | 4.83            | 3.0              | 1.0              | 0.50             | 0.30             |
| NF 1/4-28                                                                                  | 6.35            | 2.1              | 0.7              | 0.19             | 0.23             |
| NF 5/16-24                                                                                 | 7.92            | 1.5              | 0.5              | 0.25             | 0.17             |
| NF 3/8-24                                                                                  | 9.53            | —                | 0.4              | 0.19             | 0.13             |
| NF 7/16-20                                                                                 | 11.10           | —                | —                | 0.16             | 0.11             |
| NF 1/2-20                                                                                  | 12.70           | —                | —                | —                | 0.09             |

<sup>a</sup>Assumptions:

- Typical spacecraft bolted aluminum interface in vacuum
- Bare clean mill rolled surface finish
- Standard steel bolts torqued to specification
- Primary heat transfer through compressed area near bolt

Note: Confirmation measurements suggested for thermal-design purposes

Reference: NASA CR119933 June 1971 and other limited measurements

From Table 4-2, the contact resistance of fastening screws NC 4-40 (corresponding to M3 metric standard) for 1.57 mm thin plate surfaces which is 12.6 K/W taken for each connection of side wall with cover plate or base plate.

#### 4.2.2 The PCB stacks

The PCB stacks also called the bus of the satellite which is stacked PCBs one on the other by the connection of PC104 connectors and spacers. It consists the C&DH motherboard, battery pack boards, EPS board, GPS board, ADCS board, batteries, and VHF/UHF transceiver board. The order of stacks of PCBs are shown in the Figure 3-2.

Each PCB is made by alternating copper layers with insulating layers of FR4. The PCB thermal properties are strongly influenced by the number of layers and thickness of copper and FR4 present within the PCB. By using the manufacturers specified number of layers and thickness of PCB MATLAB script is written which calculates the thermal conductivity, the specific heat and the total density of each PCB using the equations in Appendix C. Since the detail design of the PCBs are not available, the common thickness used by most of the PCB manufacturers (0.035 mm copper layer) is used for the calculation. Therefore, the calculated properties of each PCB of the satellite bus are listed in Table 4-3.

Table 4-3: PCBs calculated average thermal properties

| PCB                                    | Layers | Thickness | $\rho$<br>$\left[\frac{kg}{m^3}\right]$ | $k$<br>$\left[\frac{W}{m K}\right]$ | $c_p$<br>$\left[\frac{J}{kg K}\right]$ | $\epsilon$ | $\alpha$ |
|----------------------------------------|--------|-----------|-----------------------------------------|-------------------------------------|----------------------------------------|------------|----------|
| <b>C&amp;DH motherboard</b>            | 6      | 1.6       | 2783.2                                  | 52.75                               | 505.69                                 | 0.8        | 0.6      |
| <b>PPM</b>                             | 4      | 1.6       | 2472.1                                  | 35.27                               | 529.22                                 | 0.8        | 0.6      |
| <b>GPS board</b>                       | 6      | 1.6       | 2783.2                                  | 52.75                               | 505.69                                 | 0.8        | 0.6      |
| <b>EPS board</b>                       | 2      | 1.6       | 2161.1                                  | 17.78                               | 559.5                                  | 0.8        | 0.6      |
| <b>ADCS board</b>                      | 2      | 1.6       | 2161.1                                  | 17.78                               | 559.5                                  | 0.8        | 0.6      |
| <b>VHF/UHF transceiver board</b>       | 4      | 1.6       | 2472.1                                  | 35.27                               | 529.22                                 | 0.8        | 0.6      |
| <b>battery pack boards</b>             | 2      | 0.8       | 2472.1                                  | 35.27                               | 529.22                                 | 0.8        | 0.6      |
| <b>Battery pack board (PC104 size)</b> | 2      | 1.6       | 2161.1                                  | 17.78                               | 559.5                                  | 0.8        | 0.6      |

The mass of the PCB's can be calculated by multiplying the calculated density with the volume of the PCB. But it is observed that the calculated mass of the PCB is less than from the one provided by the manufacturer. This is due to the assumptions about the design of the PCB's and neglecting components soldered on the PCB's. Thus, in order to compensate the mass of the PCB's non-

geometrical elements are added in the model for each PCB. The non-geometric element is an option in FEMAP TMG that creates new calculation point in the thermal model by defining the thermal capacitance of an element only. The thermal capacitances of the added elements are shown in the Table 4-4:

Table 4-4: Non-geometrical elements heat capacitance

| <b>PCB</b>                       | <b>Volume<br/>[m<sup>3</sup>] x 10<sup>-5</sup></b> | <b>Calculated<br/>mass [kg]</b> | <b>Actual mass<br/>[kg]</b> | <b>mass<br/>difference</b> | <b>Heat capacitance<br/>[J/K]</b> |
|----------------------------------|-----------------------------------------------------|---------------------------------|-----------------------------|----------------------------|-----------------------------------|
| <b>C&amp;DH</b>                  | 1.376                                               | 0.038                           | 0.077                       | 0.039                      | 19.72                             |
| <b>PPM</b>                       | 0.32                                                | 0.0079                          | 0.011                       | 0.0031                     | 1.64                              |
| <b>GPS</b>                       | 1.376                                               | 0.038                           | 0.106                       | 0.068                      | 34.4                              |
| <b>EPS</b>                       | 1.376                                               | 0.03                            | 0.075                       | 0.045                      | 25.2                              |
| <b>ADCS interface<br/>module</b> | 1.376                                               | 0.03                            | 0.049                       | 0.019                      | 10.63                             |
| <b>COMM</b>                      | 1.376                                               | 0.034                           | 0.085                       | 0.051                      | 27                                |

From Figure 4-5 there are two main heat flow paths from one PCB to another: through the spacers and through the connector. According to the PC/104 specification [21], the connector is made up of 104 phosphor bronze pins and only pins contributes to the heat transfer since the connector's housing height do not contact with the above PCB. Conduction through the M3 endless screws is neglected seeing the clearance with the PCB holes and spacers. A spacer is a small hollow cylinder with diameters of 3.3 mm (interior) and 4.5mm (exterior) made of 6061-T6 aluminum material. Thermal conductivities of phosphor bronze and 6061-T6 aluminum are given in Table 4-5.

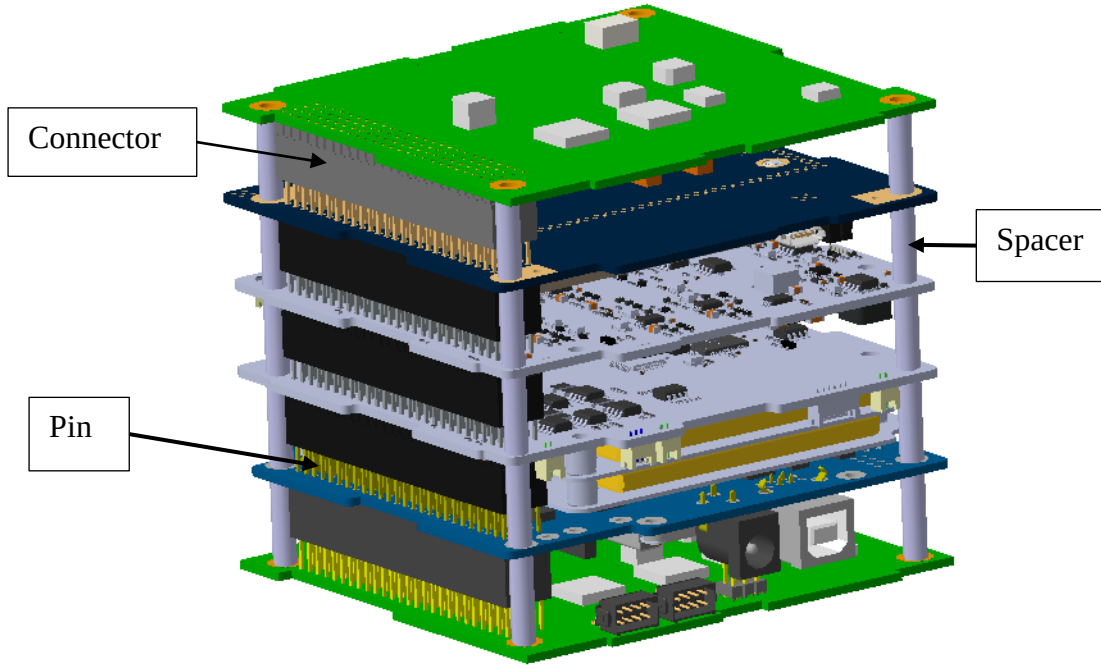


Figure 4-3: Detail PCB stack

Table 4-5: Thermal conductivity of pins and spacers

| Component     | Material         | k [W/mK] | Reference |
|---------------|------------------|----------|-----------|
| <b>Pin</b>    | Phosphor bronze  | 75       | [21]      |
| <b>Spacer</b> | 6061-T6 aluminum | 155.8    | [7]       |

The spacers and the connectors are not physically modeled in the software instead they are defined in terms of conductive links. Thus, by assuming the spacers and the connectors as 1D heat conducting bars their thermal resistance is calculated using the Equations 4-1 and 4-2 respectively:

$$R_{spa} = \frac{L_{spa}}{A_{spa} k_{spa}} \quad 4-1$$

$$R_{con} = \frac{L_{con}}{104 A_{con} k_{con}} \quad 4-2$$

Where:  $R$  is thermal resistance [K/W],  $A$  cross sectional area [m<sup>2</sup>],  $k$  is thermal conductivity,  $L$  is length, and the subscripts ' $spa$ ' and ' $con$ ' represent spacers and connectors respectively.

The spacers are connected with the PCBs through M3 endless screw which creates 12.6 K/W contact resistance between the PCB and spacer at both ends of the spacer according to Table 4-2. Since the contact resistance and spacer resistance are in series connection Equation 4-3 is used to find the total thermal resistance  $R_{spa,tot}$  of each spacer.

$$R_{spa,tot} = R_{spa} + 2R_{cont} \quad 4-3$$

But there are 4 spacers between two PCBs and one connector. Since, there is no interest in finding local hot spots on the PCB, the total thermal resistance between two PCBs is defined. The total resistance is thus the parallel connection of 4 spacers and a connector which can be calculated using Equation 4-4:

$$R_{tot} = \frac{1}{\frac{4}{R_{spa,tot}} + \frac{1}{R_{con}}} \quad 4-4$$

Therefore, by substituting the required values within Equation 4-3 and 4-4 the total thermal resistance between PCB's and other satellite components is calculated in MATLAB and the values are summarized in the Table 4-6.

From Figure 3-2, we can see that the communication board is fastened on the midplane standoffs by the M3 endless screws. Thus, 12.6 [K/W] thermal contact resistance is defined between the board and the midplane standoffs.

In the ASTUSat PCB stack, there are four Lithium polymer batteries which are integrated on daughter PCB's. Since the exact materials the batteries made of is not well known the specific heat 1350 J/kgK and the thermal conductivity 1.55 W/mK of the batteries are taken from the OUFTI-1 thermal design which uses a Lithium polymer battery as well [9]. The density of the batteries is calculated by dividing the mass of the batteries with volume of the batteries provided by the manufacturer and 2300 kg/m<sup>3</sup> is obtained. The batteries are foiled by the Kapton insulator which have an emissivity of 0.7 and absorptivity of 0.3. The thermal coupling between each battery and the PCBs is calculated and 0.1058 K/W value is obtained by assuming Kapton as an interfacing media between the battery and PCB and assuming in perfect contact with PCB and battery.

Table 4-6: Thermal resistance of spacers

| Spacers                                    | Length<br>[mm] | $R_{spa,tot}$<br>[K/W] | $R_{tot}$<br>[K/W] |
|--------------------------------------------|----------------|------------------------|--------------------|
| ADCS module-ADCS housing                   | 7.2            | 31.5                   | 7.9                |
| ADCS module-GPS board                      | 15             | 38.3                   | 2.7                |
| GPS board – EPS board                      | 15             | 38.3                   | 2.7                |
| EPS board – Power distributing board       | 15             | 38.3                   | 2.7                |
| Power distributing board – C&DH board      | 15             | 38.3                   | 2.7                |
| C&DH board – Comm board                    | 15             | 38.3                   | 2.7                |
| Power distributing board – Battery 1 board | 5              | 29.6                   | 7.4                |
| Battery 1 board – battery 2 board          | 5              | 29.6                   | 7.4                |
| C&DH board – PPM                           | 6              | 30.4                   | 7.6                |

#### 4.2.3 Solar panels

Each solar cell has a thickness of 0.1 mm and attached on the 1.6 mm thick 3-layer PCB which makes it 1.7 mm thick solar panel. The backsides of solar panels are often covered with copper fill and the underside of the cells is flooded with vias (Latin for path or way, is an electrical connection between layers) for thermal conductivity purposes [16]. The solar panels are modeled with 3 layered PCB with the calculated effective thermal properties shown in Table 4-7. The overall materials which made the solar panels are summarized in Table 4-7.

Table 4-7: components of the solar panel and their thermal properties

| Component   | Material         | $\rho$<br>$\left[\frac{kg}{m^3}\right]$ | $k$<br>$\left[\frac{W}{m K}\right]$ | $c_p$<br>$\left[\frac{J}{kg K}\right]$ | $\epsilon$ | $\alpha$ |
|-------------|------------------|-----------------------------------------|-------------------------------------|----------------------------------------|------------|----------|
| Solar cell  | Gallium arsenide | 5320                                    | 50                                  | 325                                    | 0.85       | 0.92     |
| Solar panel | Copper/FR4       | 2316.6                                  | 26.52                               | 553.3                                  | 0.88       | 0.92     |

In order to compensate the mass, a non-geometrical element is added on each deployable solar panel with heat capacitance of 171 J/K.

Each deployable solar panel are mounted to a cover plate using two 90° hinges. The hinges that connect the deployable solar panels with the cover plate are not physically modeled instead they are modeled as a rectangular bar made from stainless steel which has a thermal conductivity of 16 W/mK. The hinges have a heat transfer area of 20 x 2.2 mm<sup>2</sup> and heat traveling length 4.84 mm. Thus, the thermal coupling between the structure and the deployable solar panel (or thermal resistance of hinges) is 6.87 K/W per hinge. The connection between the fixed solar panel and the cover plate is by four M3 screws which have a total thermal contact conductance of 3.14 K/W.

#### 4.2.4 Attitude Determination and Control System

The ADCS model shown Figure 3-2 is a simple model created for the thermal analysis by discussing with the ADCS team of ASTUSat. Within the ADCS there are two PCB's: the drive board and the controller board. The PCB's are modeled by a materials composition of 2-layer and 6-layer copper and FR4 and their thermal properties are listed in Table 4-8. The electromagnets are a toroid of copper so it is modeled with thick copper layer and the reaction wheels are modeled with steel material. The thermal properties of each component of the ADCS are summarized in the Table 4-8:

Table 4-8: Components of ADCS and their respective thermal properties

| Component               | Material               | $\rho$<br>$\left[\frac{kg}{m^3}\right]$ | $k$<br>$\left[\frac{W}{m K}\right]$ | $c_p$<br>$\left[\frac{J}{kg K}\right]$ | $\epsilon$ | $\alpha$ | Reference |
|-------------------------|------------------------|-----------------------------------------|-------------------------------------|----------------------------------------|------------|----------|-----------|
| <b>Reaction wheel</b>   | Steel                  | 8000                                    | 16                                  | 500                                    | 0.47       | 0.14     | [7]       |
| <b>Electromagnets</b>   | Copper                 | 8920                                    | 400                                 | 100                                    | 0.02       | 0.32     |           |
| <b>ADCS housing</b>     | Alodined Aluminum 6061 | 2770                                    | 168                                 | 961                                    | 0.14       | 0.44     |           |
| <b>SES</b>              | Alodined Aluminum 6061 | 2770                                    | 168                                 | 961                                    | 0.14       | 0.44     |           |
| <b>Controller board</b> | 6-layer copper/FR4     | 2783.2                                  | 52.75                               | 505.69                                 | 0.8        | 0.6      | —         |
| <b>Drive board</b>      | 2-layer copper/FR4     | 2161.1                                  | 17.78                               | 559.5                                  | 0.8        | 0.6      | —         |

The electromagnets are modeled as a hollow circular bar of copper whose mass is equal to the actual mass. To compensate the mass of the ADCS components non-geometrical elements added on all components except the reaction wheel and Electromagnets. Thus, the thermal capacitance defining the non-geometrical elements is given in the Table 4-9 which is obtained by multiplying the mass difference gained with the specific heat of the materials.

Table 4-9: Non-geometrical elements heat capacitance

| <b>Components</b>           | <b>Volume<br/>[m<sup>3</sup>] x 10<sup>-5</sup></b> | <b>Calculated<br/>mass [kg]</b> | <b>Actual<br/>mass [kg]</b> | <b>mass<br/>difference</b> | <b>Heat<br/>capacitance<br/>[J/K]</b> |
|-----------------------------|-----------------------------------------------------|---------------------------------|-----------------------------|----------------------------|---------------------------------------|
| <b>SES</b>                  | 0.136                                               | 0.0038                          | 0.033                       | 0.0292                     | 28.1                                  |
| <b>ADCS housing</b>         | 4.36                                                | 0.121                           | 0.231                       | 0.11                       | 105.71                                |
| <b>Drive board</b>          | 1.21                                                | 0.026                           | 0.026                       | 0                          | 0                                     |
| <b>Controller<br/>board</b> | 1.21                                                | 0.034                           | 0.047                       | 0.013                      | 6.6                                   |

The thermal coupling of the spacers connecting the electromagnet drive board with the ADCS housing, and the spacers connecting the electromagnet drive board with the controller board is calculated and summarized in the Table 4-10. Since there is no data available about the type of material the spacers are actually made of and the type of screws used to connect the spacers with the components of the ADCS, the same material type, cross sectional area, and screw type are used for spacers used for PCB stacks.

Table 4-10: Thermal resistance of spacers

| <b>Spacers</b>                                               | <b>Length<br/>[mm]</b> | <b>R<sub>tot</sub><br/>[K/W]</b> |
|--------------------------------------------------------------|------------------------|----------------------------------|
| <b>Electromagnet drive board – top plate of ADCS housing</b> | 40                     | 8.7                              |
| <b>Electromagnet drive board – controller board</b>          | 6.4                    | 1.4                              |

In the model the electromagnets are fixed on the electromagnet drive board. Presently, the connection of the drive board and the electromagnets connection is not clear perfect connection is assumed (thermal contact resistance is neglected). The same assumption is made for the connection between the reaction wheels and the ADCS housing.

#### 4.2.5 Payload/Camera

The NanoCam C1U camera roughly contains camera lens, lens housing, lens table, PCB, and heat sink and it is shown in Figure 3-2. The camera lens is made from specially made glass material and covered with lens housing. The lens mounts in the lens table, which is the main load bearing part of the system. The image sensor is located at the bottom side of the camera printed circuit board (PCB), above the lens table and interfaces to the image processor and memory components on the bottom side of the board. The important part to know about the NanoCam is that the plate above the PCB is a 3 mm special-made aluminum heat sink [20]. Since the materials which made the lens housing, lens table, and heat sink are not available in the datasheet provided by the company. Thus, the materials used in MIST project are taken from reference [11]. But for the PCB the 4-layer FR4/copper layer is assumed and its properties are calculated and displayed in Table 4-11. In Table 4-11 the materials of the components of the NanoCam are summarized.

Table 4-11: Components of NanoCam and their respective thermal properties

| <b>Parts</b>        | <b>Material</b>    | $\rho$<br>$\left[\frac{kg}{m^3}\right]$ | $k$<br>$\left[\frac{W}{m K}\right]$ | $c_p$<br>$\left[\frac{J}{kg K}\right]$ | $\varepsilon$ | $\alpha$ |
|---------------------|--------------------|-----------------------------------------|-------------------------------------|----------------------------------------|---------------|----------|
| <b>PCB</b>          | 4-layer FR4/copper | 2472.1                                  | 35.27                               | 529.22                                 | 0.8           | 0.6      |
| <b>Heat sink</b>    | Aluminum 6061      | 2700                                    | 168                                 | 961                                    | 0.8           | 0.9      |
| <b>Lens</b>         | Glass              | 2700                                    | 0.8                                 | 800                                    | 0.83          | 0.75     |
| <b>Lens housing</b> | Brass              | 8050                                    | 116                                 | 380                                    | 0.8           | 0.9      |
| <b>Lens table</b>   | Aluminum 7075      | 2770                                    | 175                                 | 900                                    | 0.8           | 0.9      |

In the datasheet of the Nanocam the mass of the 70 mm lens with the lens housing is specified to be 200g and the rest of the camera components weigh 77 g. Thus, non-geometrical element is defined for the lens that compensate the weight and it is shown in Table 4-12:

Table 4-12: Non-geometrical elements heat capacitance

| <b>Components</b>                                      | <b>Volume<br/>[m<sup>3</sup>] x10<sup>-5</sup></b> | <b>Calculated<br/>mass [kg]</b> | <b>Actual<br/>mass [kg]</b> | <b>Mass<br/>difference [kg]</b> | <b>Heat<br/>capacitance</b> |
|--------------------------------------------------------|----------------------------------------------------|---------------------------------|-----------------------------|---------------------------------|-----------------------------|
| <b>Lens table, PCB,<br/>heat sink, and<br/>spacers</b> | 2.82                                               | 0.077                           | 0.077                       | 0                               | 0                           |
| <b>Lens and<br/>housing</b>                            | 1.93                                               | 0.123                           | 0.2                         | 0.077                           | 61.6                        |

From the assembly CAD model of ASTUSat (Figure 3-2), we can see that heat conducted from the camera to the structure or vice versa is through the connection between the bottom cover plate and camera lens housing, and also through the connection between lens table and SW 1 and 2. The lens table is bolted on four midplane standoffs made from aluminums 6061 where two of the midplane standoffs connects the lens table with SW 1 and the other two connect SW 2. Since the midplane standoffs are connected with side wall through the M3 screw the thermal contact resistance between these two materials is 12.6 K/W from Table 4-2. Also, the thermal resistance of 13 mm spacers between the lens table and the camera PCB is calculated and have a value of 2.8 K/W is obtained. For this model connection between the lens table and lens housing, the lens housing and lens, and the lens housing and PCB is assumed to be in perfect connection.

### 4.3 Internal heat generation definition

The components that dissipate energy were discussed in section 3.1.3. for two different operating conditions. Thus, the heat dissipated within each component is defined in the model for each PCB mesh model. Since the local hot spots of PCBs are not known the thermal loads listed in Table 3-2 are defined within the entire elements that modeled the component generating heat.

### 4.4 Boundary conditions definition

In addition to the heat that is exchanged among the spacecraft components through conductive heat transfer, there is also radiative heat transfer which occurs between the space environment and the spacecraft, and between all spacecraft components that are in view with each other. FEMAP TMG has an inbuilt radiation modeling options that will detect enclosures, calculate view factors,

perform shadowing checks, calculate diffuse and reflected radiative exchanges and compute radiative heating within the model. Thus, radiation modeling of ASTUSat in TMG is divided into two main types: orbital heating definition, and satellite radiation definition.

#### **4.4.1 Orbital heating definition**

The FEMAP TMG software has the orbital heating simulation object to request solar, albedo, and planet view factor calculations for selected elements within a defined orbit. Thus, to simulate the orbital heating on the satellite the thermal engineer is required to specify the illuminated elements to orbital heating, the calculation positions, the environmental heat fluxes, orbit and the orientation of the satellite.

Now to simulate the ASTUSat orbital heating, the elements that are subjected to environmental heating (i.e. the elements modeling the external surfaces of the satellite) are only requested within the software to compute orbital view factor calculation instead of requesting for all elements of the model. Such a kind of method saves the thermal engineer a lot of time during solving the model. And all orbital view factor computations are performed at 14 calculation points on orbit. The parameters like the environmental fluxes, orbit, and orientation of the satellite are defined within the software using the same parameters used in the preliminary analysis. Thus, Figure 4-4 and 4-5 summarizes the orbital heating definition using the orbit visualization option within the software. In the figures, the satellite is displayed at each calculation points requested for view factor calculation and the environmental fluxes and orbital parameters are shown at the left corner of these figures.

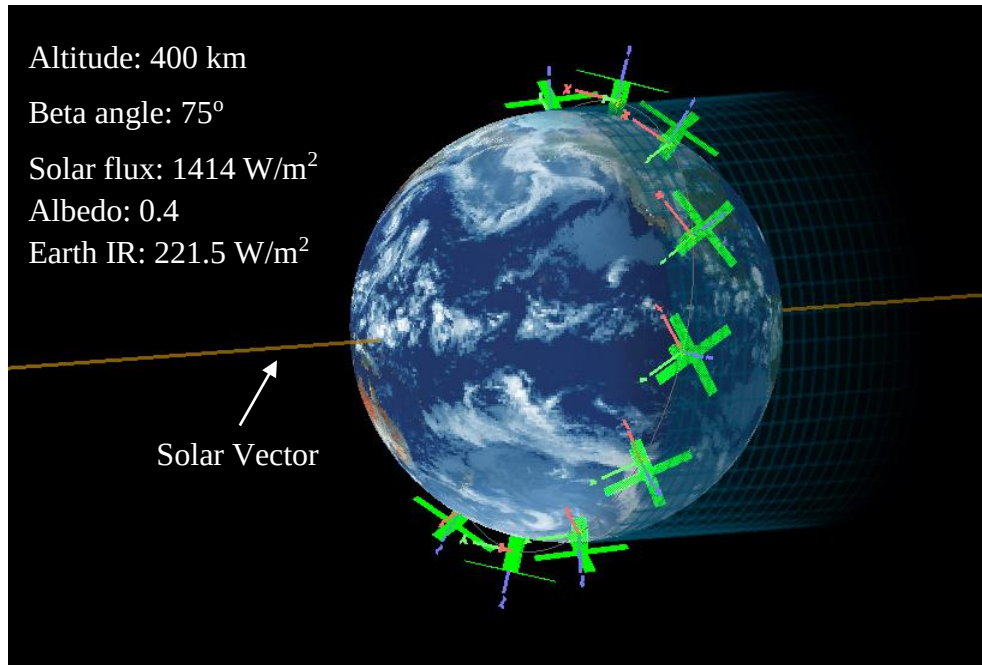


Figure 4-4: WHS boundary conditions (FEMAP TMG)

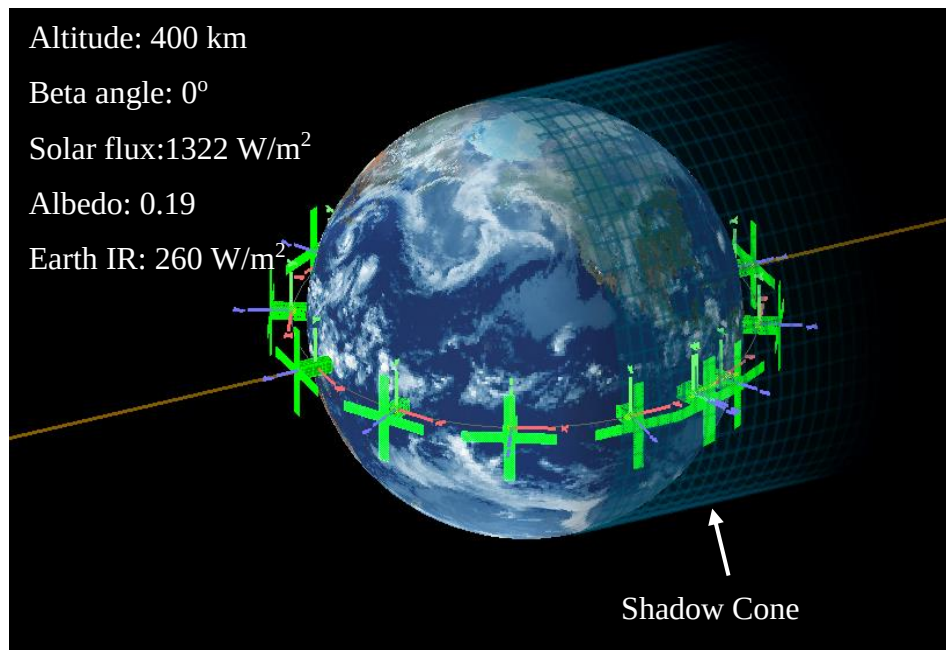


Figure 4-5: WCS boundary conditions (FEMAP TMG)

#### 4.4.2 Satellite radiation definition

Satellite radiation is a term used in this thesis to define either the rejection of infrared by the satellite to the surrounding space or the infrared radiation exchanged between components of the

satellite that are in view with each other. Thus, in the model two radiation requests are defined in order to determine the satellite radiation in efficient way. The first radiation request contains elements that belongs to the external surface of the satellite which are in view with the space environment and labeled as “external radiation”. This radiation request minimizes the computational time of the solver by reducing unwanted computations for elements that are not in view with the deep space. To correctly model the radiation exchange with the environment, the radiative environment temperature is set to -270.45°C. The other radiation request is defined to determine the radiation exchange between the internal components of the satellite.

Finally, the model’s initial temperature is set to be 25 °C and the steady state temperature of the satellite is calculated for the average heat fluxes around the orbit. Then, the steady state temperature of the satellite is set to be initial condition for the transient temperature computation.

The transient temperature is to be calculated using backward method which is an implicit differencing scheme at time step  $\Delta t = 60$  s. The model’s iteration time is set to start from time 0 to 33321.6 s orbital time (i.e. 6 orbits) and if the temperature difference between the beginning and the end of one orbit is less than 0.1 °C the transient temperature is said to be converged.

## **4.5 Validation**

Now in order to validate the modeling process used for modeling ASTUSat the OUFTI-1 model discussed in section 3-5 (previous chapter) is modeled in the FEMAP TMG. The 10x10x10 mm cube and each side of the cube is 1.25 mm thick CAD model is created in CATIA V5. Then, the model is imported in FEMAP TMG software environment and meshed with 218 nodes, and 216 2D elements. Within the model two non-geometrical elements are defined, one to compensate the thermal capacitance of the structure, and one to model the entire internal electronics including the batteries. The orbital heating and the satellite radiation to space is also defined within the model using the parameters mentioned in section 3-5. Finally, it is solved to obtain the transient temperature and total absorbed environmental radiation which are shown in Figure 4-6 and 4-7.

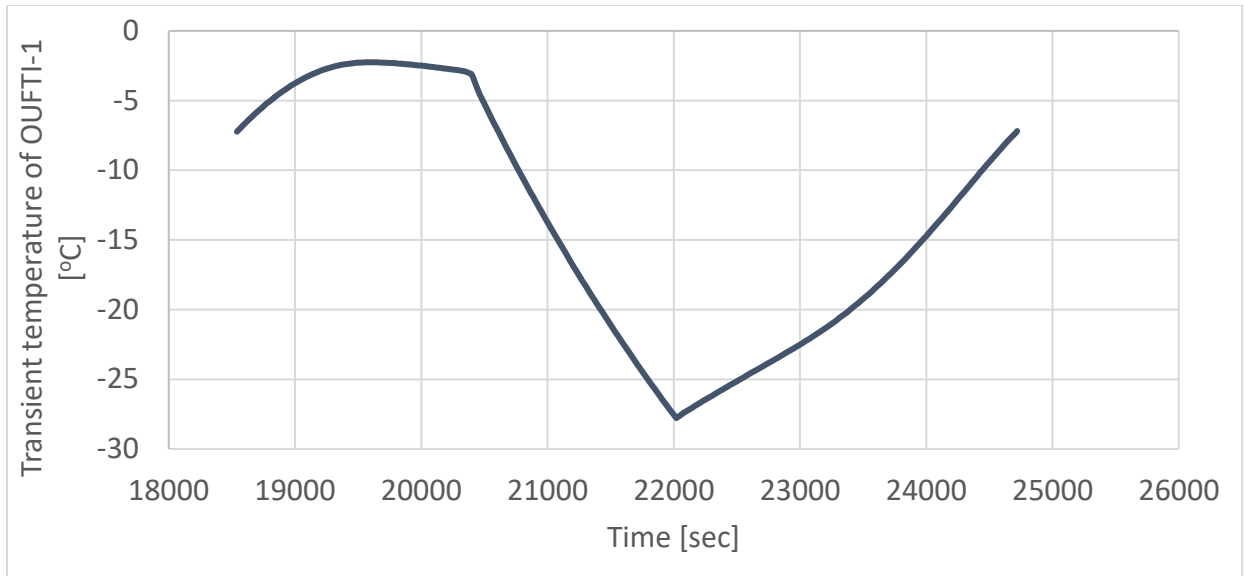


Figure 4-6: Transient temperature of OUFTI-1 calculated by FEMAP TMG

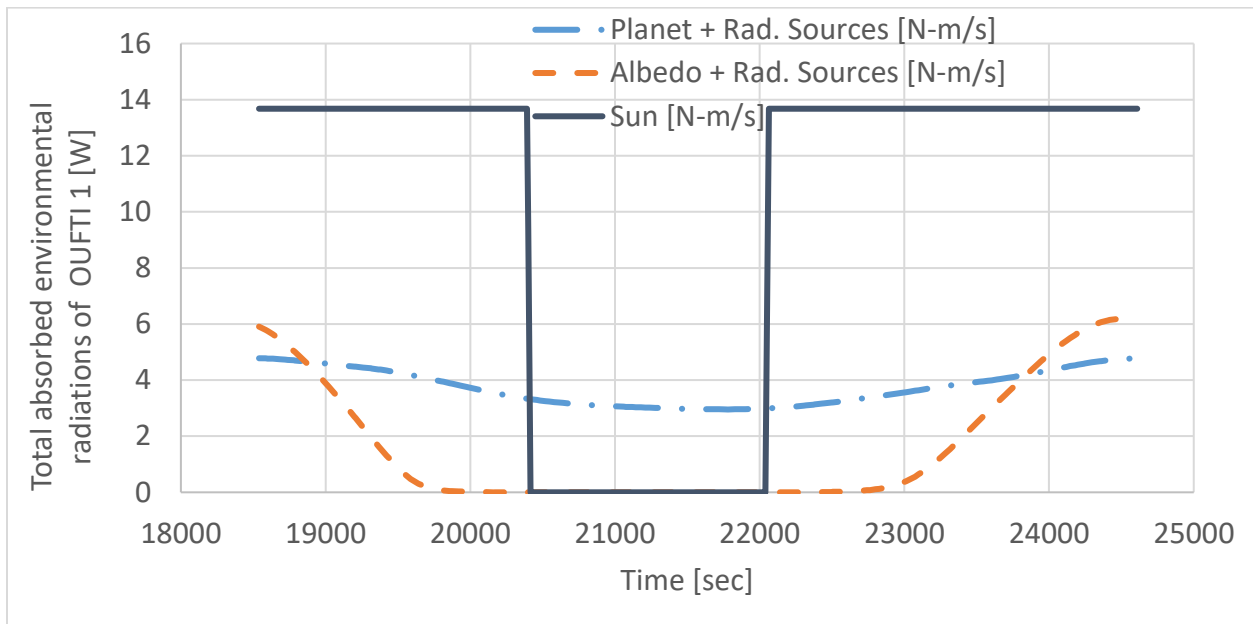


Figure 4-7: Total Absorbed environmental radiation of OUFTI-1 calculated by FEMAP TMG

Therefore, from the results obtained no significant difference is observed between the results obtained from the FEMAP TMG (shown in Figure 4-6 and 4-7) and the results obtained from ESATAN/ESARAD software (shown in figure 3-8). This implies that the method used to model the ASTUSat is a valid approach.

# Chapter 5

## 5 Results and Discussion

The FE model developed in the previous chapter is solved to observe whether the satellite components are within their operating temperature or not. Thus, in this chapter the results obtained by solving the model are presented and discussed. Then, the thermal control system needed to maintain almost all components within their operating temperature is discussed. After that, the FE model is resolved by updating it with the thermal properties selected for the purpose of thermal control and the results are discussed.

### 5.1 Results and discussions

Now the ASTUSat finite element model is solved and the transient temperature of the satellite components are extracted from FEMAP TMG software and discussed in this section. The results obtained will help to observe components that are in danger.

#### 5.1.1 Worst hot scenario (WHS)

As we have discussed in the previous chapter there is no eclipse in this scenario which leads to maximum possible temperature reached by the satellite. Thus, the FE model of ASTUSat is solved for WHS and the temperature is converged at the 3<sup>rd</sup> orbit (i.e. from time 11207.2 to 16660.8 sec iteration time). Thus, the total absorbed environmental thermal radiations and component's temperature evolution are extracted and shown in Figure 5-1 and 5-2 respectively.

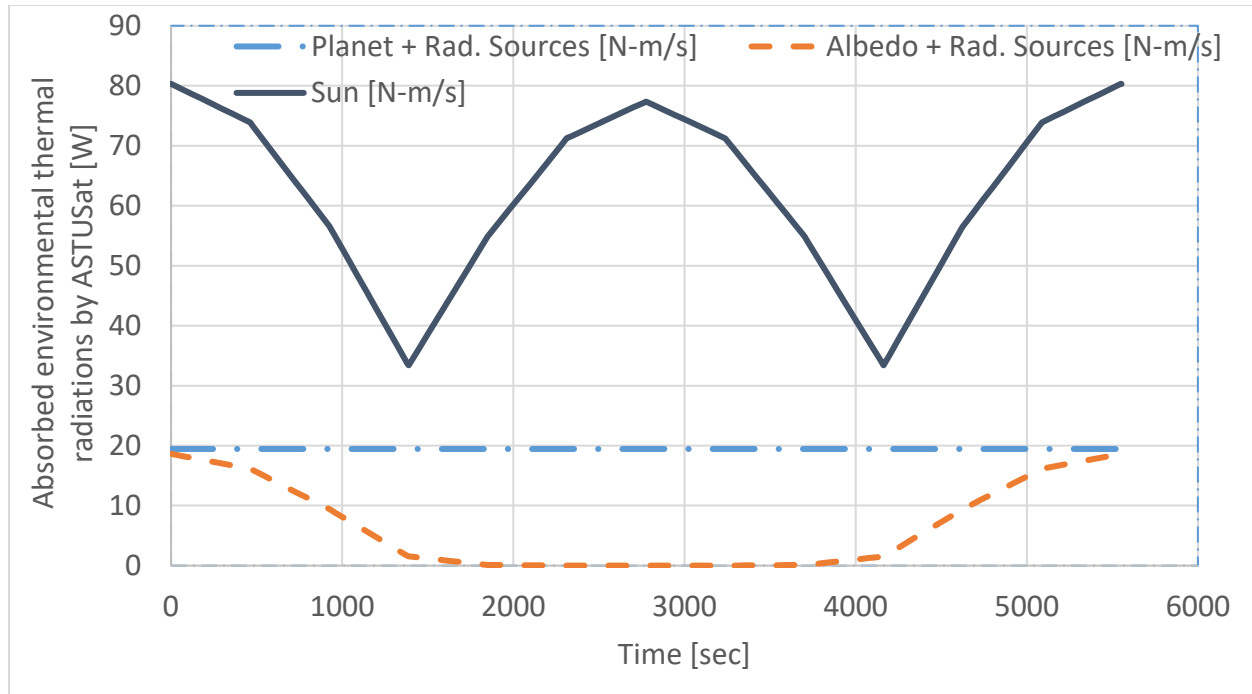


Figure 5-1: Total absorbed environmental thermal radiations by ASTUSat for one orbit during WHS calculated by FEMAP TMG

As we can see from Figure 5-1, all absorbed environmental thermal radiations obtained in this detail thermal analysis have similar profile but have higher value than the one obtained in the preliminary thermal analysis. This is due to the consideration of deployable solar panels (DSP) in the detail thermal analysis. From the figure the solar radiation absorbed by the satellite reached maximum value 80.3 W at the beginning or end of the orbit because it is the only position where the angle between the unit vector normal to the fixed solar panel (FSP) and the solar vector become maximum which means maximum area of the FSP and DSP are visible to the sun. The Earth infrared has constant value throughout the orbit which is similar to the one obtained in the preliminary thermal analysis but higher value (i.e. 19.4 W) because the consideration of deployable solar panels.

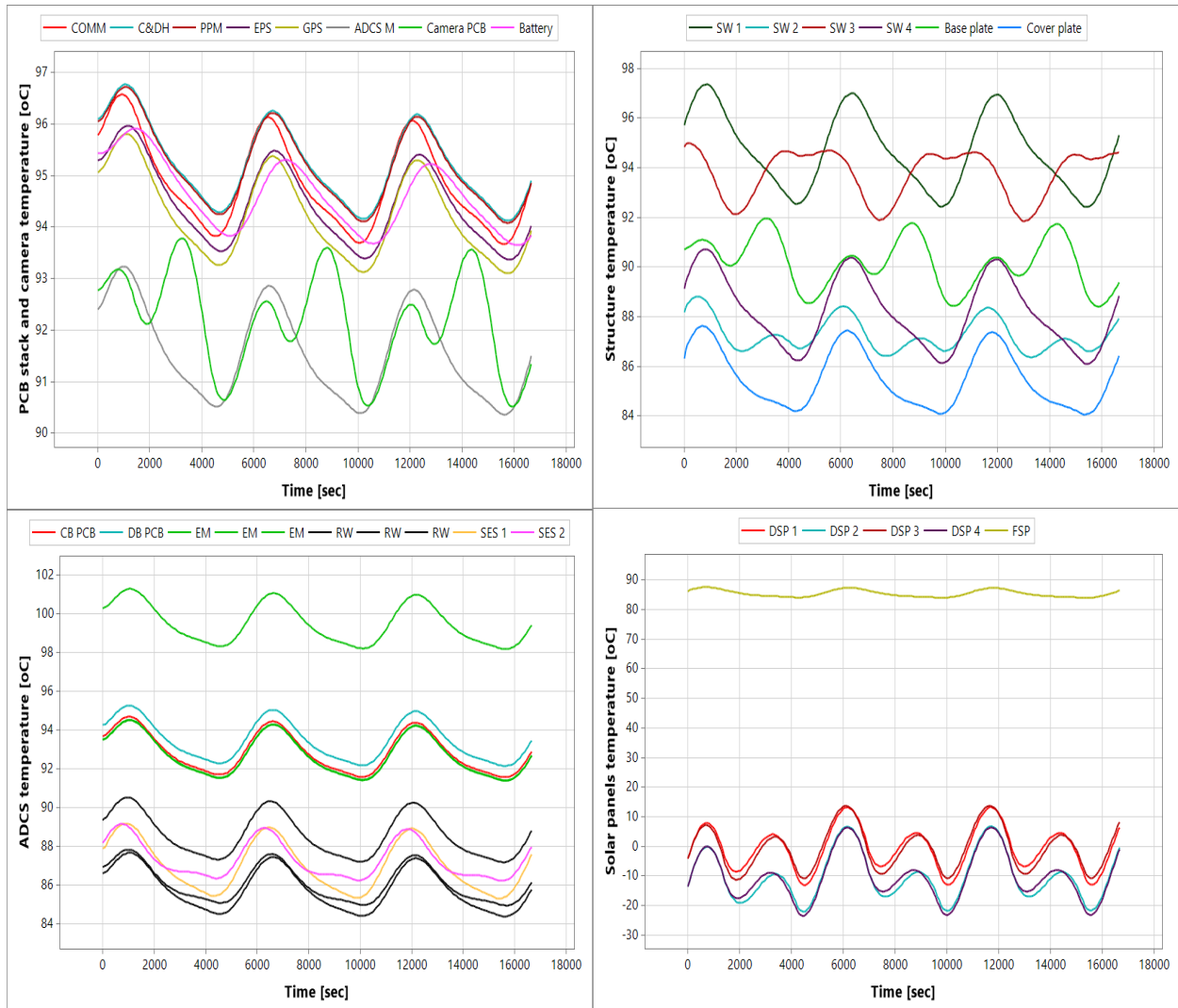


Figure 5-2: WHS transient temperature of ASTUSat's components

From the simulation the highest temperature is 101.6 °C which is reached by electromagnet with heat dissipation. The SW 1 and SW 3 have higher temperature than SW 2 and SW 4 because these two faces are viewed by the sun throughout the orbit. Also, deployable solar panels that are in view with SW 1 and SW 3 (i.e. DSP 1 and DSP 3) have higher temperature than the DSP 2 and DSP 4 due to the effect of the reflected solar radiation from the side wall of the satellite.

From the Figure 5-2 we observe that all except the deployable solar panels exceeded the operating temperature of the components specified on Table 3-1. Thus, according to this simulation, the satellite will not operate if it is subjected to such a kind of scenario unless the right thermal control system is designed for it.

### 5.1.2 Worst cold scenario (WCS)

The model is also solved for WCS with zero beta angle to determine the minimum temperature reached by the satellite components. The model is converged at the 3<sup>rd</sup> orbit which is same as WHS. The absorbed environmental thermal radiations and the temperature evolutions of main satellite components obtained from the simulation are shown only for one orbit in Figure 5-3 and 5-4:

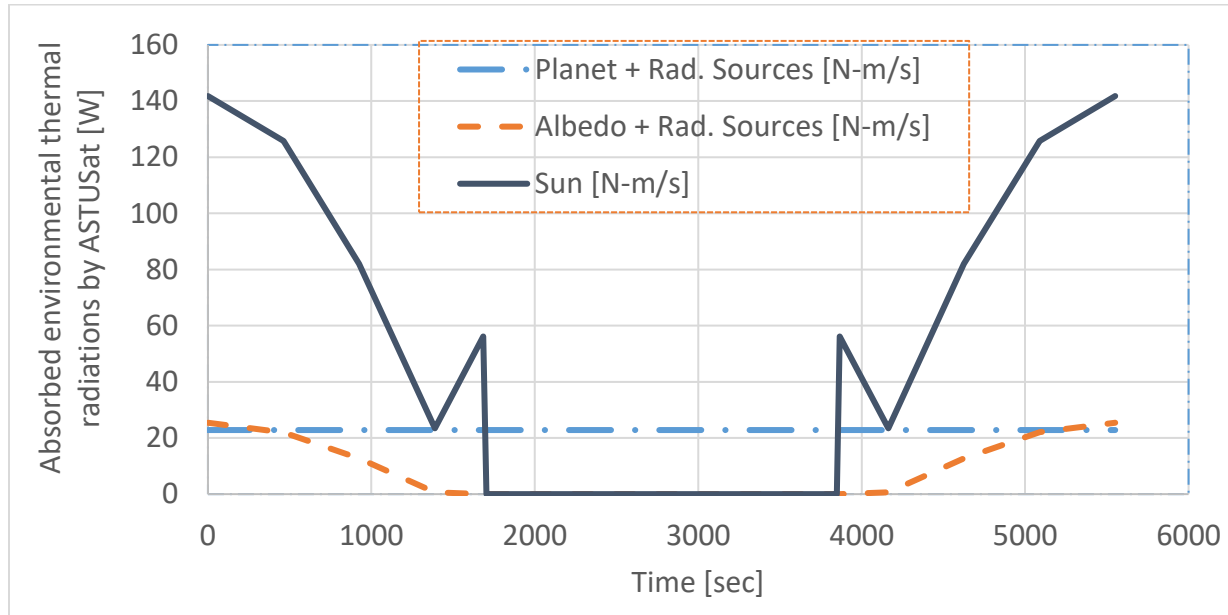


Figure 5-3: Total absorbed environmental thermal radiations by ASTUSat for one orbit during WCS calculated by FEMAP TMG

As we can see from Figure 5-3, the direct solar radiation from the sun reached maximum value of 141 W at sub solar position of the satellite which is higher than the WHS. This is because at this location both the deployable solar panels and fixed solar panel are perpendicular to the solar vector. At the same location the albedo is high because the satellite is at sub solar position. The solar radiation becomes zero when the satellite is in eclipse. The satellite absorbs 22.8 W Earth infrared throughout the orbit which is slightly higher than WHS.

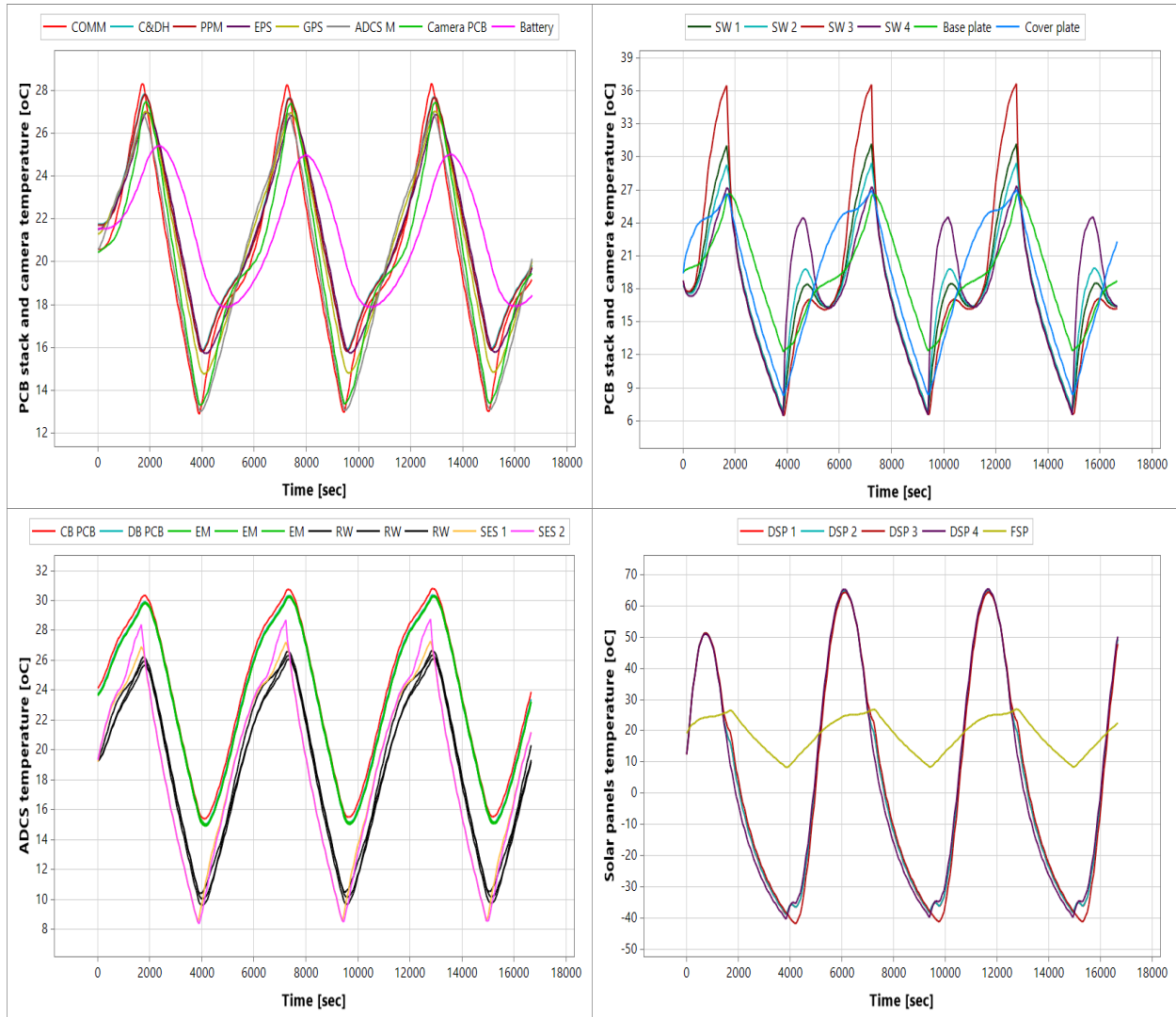


Figure 5-4: WCS transient temperature of ASTUSat's components

In Figure 5-4 the components temperature rises when the satellite is in sunlit and drops when the satellite is in eclipse. And these temperature fluctuations are dependent on the thermal optical properties and the orientation of the faces according to the heating source (i.e. sun, and earth). In this scenario the deployable solar panels reach the highest temperature 65.5 °C and lowest temperature -41.6 °C. This is due to high absorptivity and emissivity of the deployable solar panels (in both faces) which makes them to absorb maximum high thermal radiation during sunlit and release maximum thermal radiation during eclipse. Another thing different observed in this scenario is all electromagnets have approximately equal temperature because no heat generation is assigned for any of them. In general, from the figure we can see that all components are within

their operating temperature range except the deployable solar panels. The deployable solar panels are below their operating temperature  $-40^{\circ}\text{C}$  for small period of time.

## **5.2 Thermal control system design**

There are two ways of spacecraft thermal control techniques: Passive thermal control system (PTCS), and Active thermal control system (ATCS) [7].

Passive thermal control systems consume no power during operation which ensures its low mass and cost. It is highly reliable due to its design simplicity, implementation and testing. The only drawback of PTCS is its low heat transport capability except for heat pipes when compared with ATCS. The passive thermal control can be achieved by use of insulation blankets, sun shields, radiating fins, straps, surface thermo-optical properties, thermal coatings, heat sinks, and phase-change materials [7].

When the thermal control for a satellite is done by using power it is called Active Thermal Control System (ATCS). ATCS can handle large heat loads and also provides a degree of control over how to manage these loads. The ATCS can be achieved by the use of heaters, louvers, heat pipes, thermoelectric coolers, cryogenic coolers, and pumped fluid loops (PFLs) [7]. In most cases of small spacecraft designers are keen to use active thermal systems for temperature sensitive devices (such as batteries, cameras and electronics).

In order to maintain the components of the ASTUSat within their operating temperature passive thermal control is generally preferred due to limitations of power, mass, and volume. According to the current design of the ASTUSat the thermal control system is made up of the thermal control tapes applied to the exterior surfaces of the spacecraft, and thermal washers in the components between joints. Each of these aspects are described in the following sections.

As we have seen in section 5-1, due to high absorptivity and small emissivity of the alodined aluminum raised the temperature of the satellite above the operating temperature of components in WHS. So, to improve the external heat transfer, the optical properties of the external body surfaces are modified that are not occupied by solar cells, sensors, and cameras' lenses. To meet this design point, the feasible solutions identified are surface treatments (e.g. oxidation), surface coatings (e.g. paints) or surface overlay by means of specific tapes. Surface treatments are usually complex and expensive, oxidation is not electrically conductive, and it could lead to a major design

limitation if not specifically required. Coatings, usually black or white, can be either electrically conductive or not. But similar performance can be achieved with the use of specific tapes. Since paints are more critical in terms of outgassing, the use of a specific tape overlay is recommended. Tapes can be applied easily to any surface of the satellite and they can be easily removed in case of design changes. Therefore, through an iterative process using the finite element analysis the side wall external surfaces coated with alodined aluminum surface finish are changed by using silicon oxide coated aluminized Kapton tape ( $\epsilon = 0.18$ , and  $\alpha = 0.12$ ). This tape mainly selected is to reduce the amount of thermal radiation absorbed during WHS and reduce the amount of thermal radiation released to space during WCS.

From the simulation, the internal face of the side wall and the base plate  $\epsilon = 0.88$  has been selected. This value improves the internal radiation heat exchange between components by releasing and absorbing infrared. Specially, during WHS the portion of the side wall facing the sun becomes hot surfaces releases higher infrared and it is absorbed by components with low temperature electronics and the other portion of the side wall facing the deep space. Thus, such a kind of emissivity of a surface can be achieved by using tapes like black Kapton tape, or paints like carbon black paint or Chemglaze A276 white paint with reasonable outgassing level.

In the simulation, it is observed that the amount of heat conducted from the camera during WCS should be minimum to avoid temperature drop below its minimum operating temperature (0 °C). Thus, to isolate the camera from conductive heat transfer washers with small thermal conductivity are used between the lens table and mid plane standoffs. The 2 mm thickness nylon washers ( $k = 0.24 \text{ W/mK}$ ) are selected which creates 116 K/W thermal resistance between each four connections. Also, there should be no physical connection between the lens housing and the base plate to avoid further conductive heat transfer.

### **5.2.1 Results and discussions after the application of thermal control system**

In this section, the results obtained from the FE model updated by the properties selected for the thermal control purpose are discussed. The model is solved for WHS and WCS and the results are provided in the Figure 5-5 and 5-6.

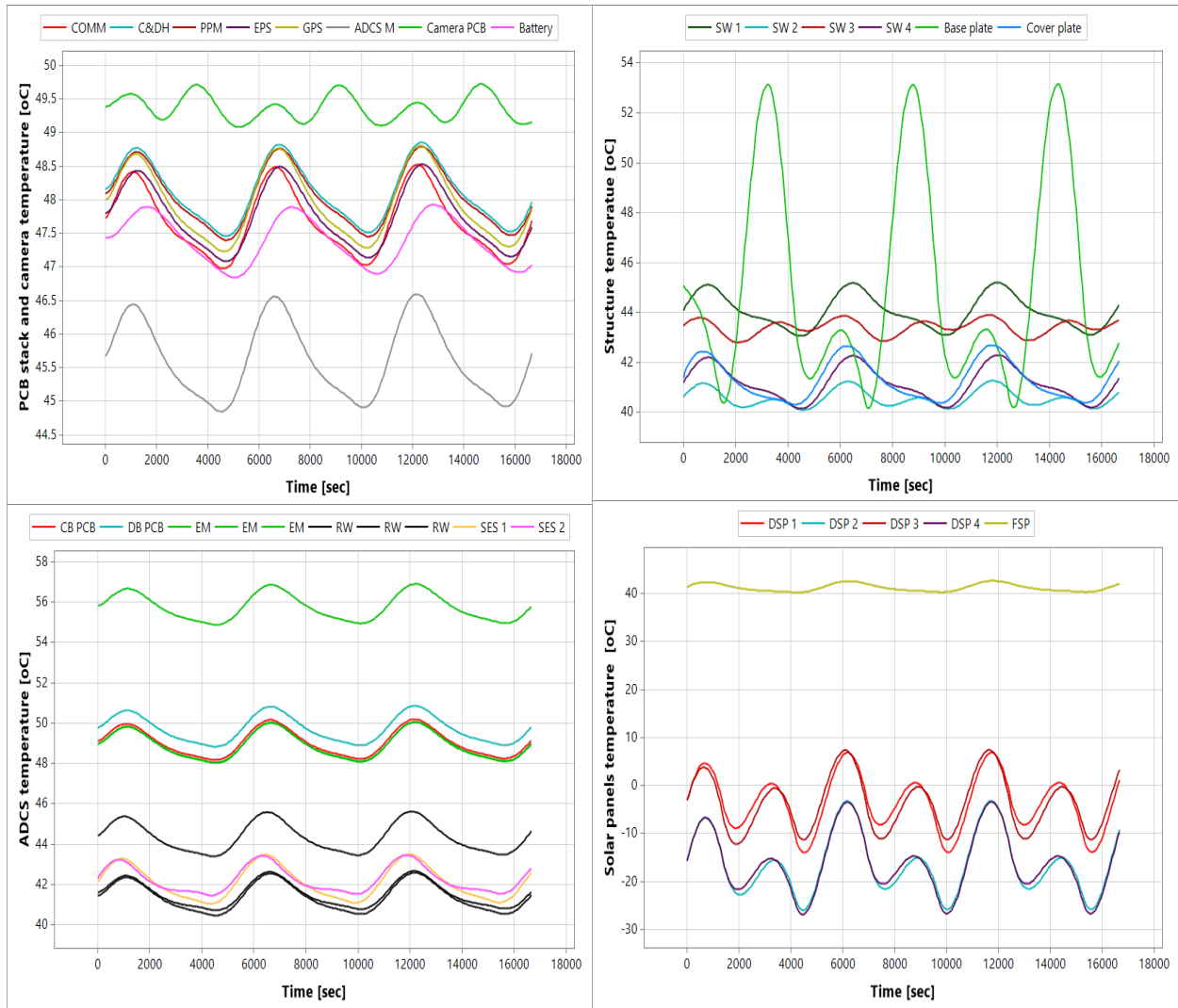


Figure 5-5: WHS transient temperature of ASTUSat's components after the application of passive thermal control system

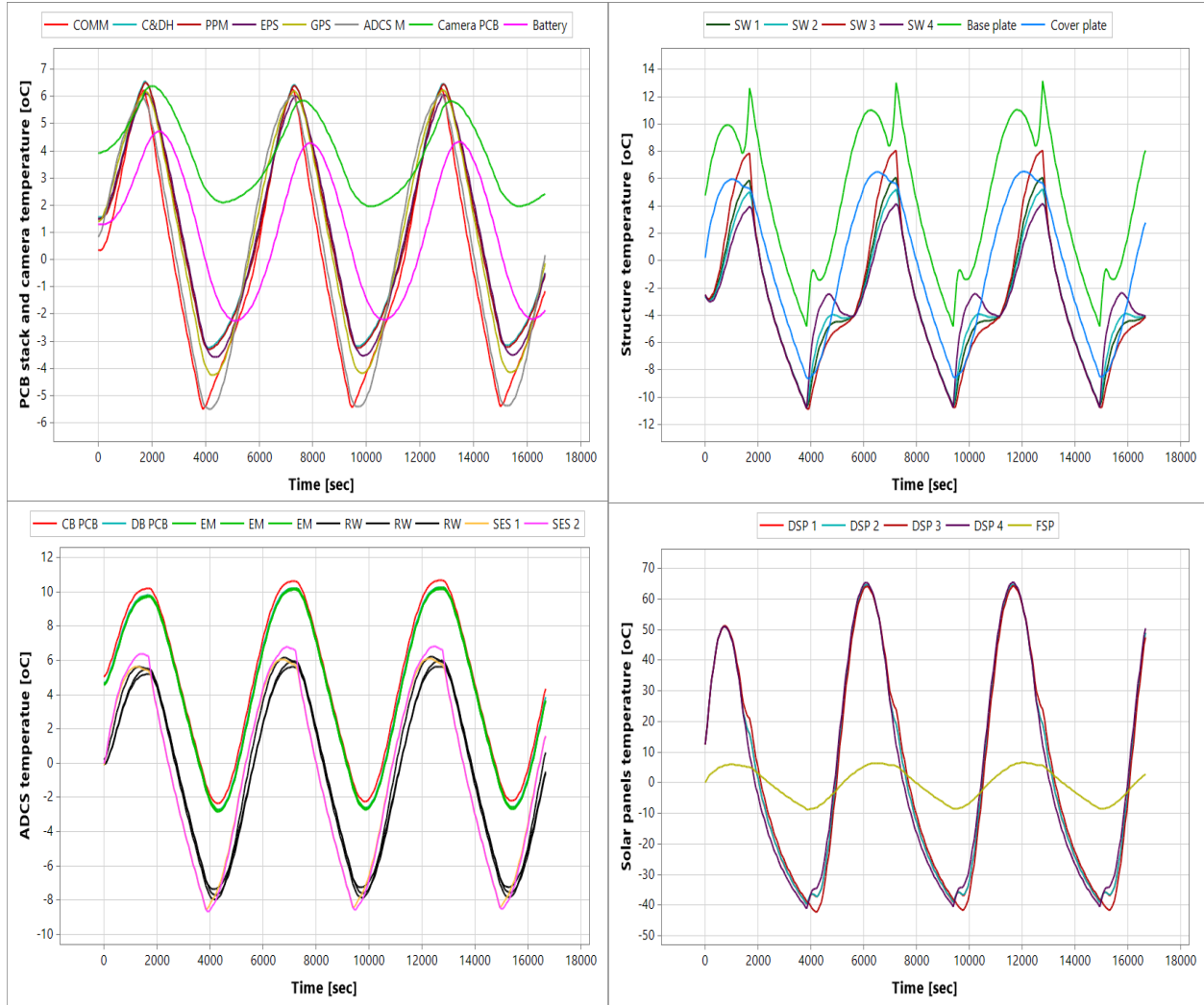


Figure 5-6: WCS transient temperature of ASTUSat's components after the application of passive thermal control system

As we can see from the Figure 5-5 and 5-6, the temperature of the critical satellite components was brought within their specific operating temperatures necessary for proper operation. So, by the application of only passive thermal control techniques the satellite maximum temperature is dropped from 101.6 °C to 56.9 °C during WHS. The deployable solar panels still remain below their operating temperature during WCS for short period of time. Also, the deployable solar panels pass by 2 °C the maximum operating temperature during WHS for short period of time when the beta angle is zero (not shown here). Thus, the solar cells may not operate as efficiently as expected at those short periods of time because of their cold and hot temperature. Although, when the number of solar cells required for the satellite is determined future work will be required to

determine a more appropriate method to maintain the temperature of the solar cells within operating temperature range during all operating phase.

Therefore, the results showed that passive control is sufficient to maintain most of the satellite's components within their temperature range with appropriate thermal coating or tapes overlay solutions. However, the selected thermal control system must be verified through thermal analysis again when the satellite design is fully completed before being accepted for use.

# Chapter 6

## 6 Conclusion and Recommendations

### 6.1 Conclusion

In this thesis two thermal models of ASTUSat with different complexity have been developed to reveal the temperature of the payload and subsystems of the spacecraft for worst hot scenario (WHS) and worst cold scenario (WCS) when the spacecraft is orbiting the Earth in a circular orbit.

In the preliminary thermal analysis, the WHS and WCS of the satellite in orbit operation are identified and the average transient temperature of the satellite is calculated at those scenarios. The model used in the preliminary thermal analysis was a simple thermal mathematical model (TMM) which was derived from the spacecraft energy balance equation. The TMM was solved along with orbital equations in MATLAB environment by using Euler's explicit method to obtain the transient temperature of ASTUSat. From the results, it was observed that the satellite temperature is between 86.6 and 89.3 °C during WHS which is not in the safe operating temperature range of most subsystems and components of the ASTUSat. Then, some surface finishes were selected according to their optical properties and applied at the side wall of the satellite and the analysis was performed again. From the results it was observed that the silicon oxide coated aluminized Kapton surface finish maintains the CubeSat within good thermal performance during WHS and WCS compared to other surface finishes.

The detail thermal modeling and analysis were performed in order to obtain both temperature distribution and temperature evolution of ASTUSat's components and subsystem for WHS and WCS. The detail thermal model was developed in FEMAP TMG environment. The model was created using finite element method and it includes detail physical and thermal properties of the spacecraft. The model is solved and based on the results obtained from the thermal analysis it was observed that, in the WHS simulations all components were not within their operating temperature range. This problem can be solved by using passive thermal control techniques like taping the external surface of the side wall with low absorptivity and emissivity tape. Also, from the detail thermal analysis it was observed that the internal heat radiation exchange within the satellite can

be improved by covering the internal side of the side wall and base plate with high emissivity tapes or paints. During WCS the camera falls below the operating temperature during eclipse due to high conductive heat loss from the camera. But this problem can be solved by using thermal washers at each joint where the camera is mounted.

## **6.2 Recommendations**

The thermal models and analysis performed in this thesis were not for actual orbits (i.e. circular orbit was assumed) because the day of deployment is not fixed. To obtain more realistic thermal simulation of ASTUSat the thermal models should be updated by the actual orbit of the satellite once the day of deployment is fixed. In addition, the ASTUSat is still in the design phase and there are components of the satellite that are not fully defined yet. Once the design phase is completed and all components are well defined, the detailed thermal model should be updated by including components that were not previously included in the model. Thus, any changes made to the internal layout of the component's boxes and electrical components and changes to external surface finish should be incorporated into the model, to ensure an accurate model and accurate temperature predictions for the final satellite design. In the model, every mission mode heating load case should be included in the analysis as well, rather than just for the two worst cold and hot scenarios. Since, the conduction values, optical properties, and material properties used in this thesis study are estimates, their inaccuracy can contribute to errors in the model.

The model could be improved by determining exact input values through laboratory testing of materials and conductance, though this process can be complex and time consuming. And, using the results gained from the new model, the proposed thermal control technique in this thesis needs to be vetted on a system level as well as subsystem level before being approved for use in the final satellite design.

Therefore, the thermal modeling results obtained in this thesis can easily be adjusted to the actual input data, and design arrangement of payloads and subsystems to generate thermal performance of spacecraft ASTUSat which is critically needed before conducting ground vacuum chamber thermal performance test of ASTUSat at the laboratory of Space Technology Institute of ASTU.

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# Appendix A.

## A. MATLAB Codes

### Biot number calculation

```
%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Biot number calculation                                     %%
% HABTAMU MENGISTU                                         %%
% 2018-09-01                                              %%
% Adama Science and Technology University                 %%
% %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

close all; clear all; clc;
sigma = 5.67e-8; % Stephen Boltzmann Constant
Tinf = 2.7; % Deep space Temperature(K)
k = 175; % Thermal conductivity

%% Calculation of biot number

L = 0.1; % satellites width and length
h_sat = 0.326; % Satellite height
Ts = 200:500; % Satellite surface temp.
emi = [0.1,0.3,0.5,0.7,0.9]; % Emissivity value
Lc = L*h_sat/(4*h_sat+2*L); % Characteristic length
Rcon = Lc/k; % conduction resistance

biot = zeros(length(emi),length(Ts));

for i = 1:length(emi)
    for j= 1:length(Ts)
        Rrad = 1/(emi(i)*sigma*(Ts(j)...
            +Tinf)*(Ts(j)^2+Tinf^2)); % Radiation resistance
        biot(i,j) = Rcon/Rrad; % Biot number
    end
end

%% plotting surface temperature Vs Biot number

figure
Ts = Ts-273;
for i = 1:length(emi)
    plot(Ts,biot(i,:), 'linewidth',2,...
```

```

        'MarkerIndices',1:20:length(biot(i,:)))
    hold on
end
xlabel('Temperature [deg C] ')
ylabel('Biot number')
hold off
legend('0.1 emissivity','0.3 Emissivity'...
    , '0.5 Emissivity','0.7 Emissivity','0.9 Emissivity')

```

---

## **Satellite position vector calculating function**

```

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Satellite position calculating function      %%
% Reference: Orbital Mechanics for Engineering %%
%           Students,3rd edition            %%
% Modified by: HABTAMU MENGISTU             %%
% 2018-10-02                               %%
% Adama Science and Technology University    %%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
function [r_geocen, V_geocen, rxv]= keplere(inc, raan,...
    arg_peri,max_alt,ecc,t)

% this function calculates the geocentric position of a
% spacecraft in earth orbit.
% inputs:
% inc = inclination (deg),
% raan = right ascension of ascending node (deg)
% arg_peri = argument of perigee (deg)
% max_alt = maximum altitude (km)
% ecc = eccentricity
% t = time (s)
%% constants
mu = 398601;           % Earth's gravitational parameter (km)
Re = 6378.14;          % Radius of Earth (km)

a = (Re + max_alt)./(1+ecc);    % semi-major axis (km)
incr = deg2rad(inc);           % inclination (rad)
raanr = deg2rad(raan);         % raan (rad)
arg_perir = deg2rad(arg_peri); % Argument of perigee (rad)

P = 2.* pi.*sqrt(a^3/mu);       % Orbit period (s)
M = 2.*pi.*t./P;               % Mean Anomaly (radians)
%% Set an error tolerance:
error = 1.e-8;
%% Select a starting value for eccentric anomaly (E)
if M < pi
    E = M + ecc/2;

```

```

else
    E = M - ecc/2;
end
%% Iteration until E is within the error tolerance
ratio = 1;
while abs(ratio) > error
    ratio = (E - ecc*sin(E) - M)/(1 - ecc*cos(E));
    E = E - ratio;
end
r_pfmag = a.*(1-ecc.*cos(E));
%% distance from focus (peri-focal coords)

r_pf = a.*(cos(E)-ecc),(sqrt(1-ecc.^2).*sin(E)),0];
v_pf = sqrt(mu.*a)./r_pfmag.*[-sin(E),sqrt(1-ecc.^2).*cos(E),0];

%% Rotation matrix (transform from peri-focal to geocentric
coordinate)
rotmat(1,1) = cos(raanr).*cos(arg_perir)-sin(raanr)...
    .*sin(arg_perir).*cos(incr);
rotmat(1,2) = -cos(raanr).*sin(arg_perir)-sin(raanr)...
    .*cos(arg_perir).*cos(incr);
rotmat(1,3) = sin(raanr).*sin(incr);
rotmat(2,1) = sin(raanr).*cos(arg_perir)+cos(raanr)...
    .*sin(arg_perir).*cos(incr);
rotmat(2,2) = -sin(raanr).*sin(arg_perir)+cos(raanr)...
    .*cos(arg_perir).*cos(incr);
rotmat(2,3) = -cos(raanr).*sin(incr);
rotmat(3,1) = sin(arg_perir).*sin(incr);
rotmat(3,2) = cos(arg_perir).*sin(incr);
rotmat(3,3) = cos(incr);
%% satellite position (geocentric coordinate)

r_geocen = transpose(rotmat*transpose(r_pf));

%% satellite velocity (geocentric coordinate)
V_geocen = transpose(rotmat*transpose(v_pf));

%% cross product of satellite position
% and satellite velocity
rxv = cross(r_geocen,V_geocen);
end

```

## **Environmental incidence thermal radiation calculating function**

```
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Incidence environmental heat flux determining %%
%% function                                     %%
%% HABTAMU MENGISTU                           %%
%% 2018-10-02                                 %%
%% Adama Science and Technology University     %%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
function [Qs,Qa,Qir] = external_heat_flux(inc,...
    Altitude,ecc,month,dt,no_period,raan,...
    spinning,Surface_rotation)
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% This function calculates the external incidence heat fluxs
% on each spacecraft surface (W/m^2)
% Input-
%     Rsc => Geocentric position vector of the spacecraft
%
%     Vsc => Velocity vector of the spacecraft
%
%     RxV => cross product of Rsc and Vsc
%
%     Rsun => geocentric equatorial position vector of
%            the sun
%     Te => Earth surface Temperature (k)
%
%     a => Albedo factor
%
%     Es => Solar constant (W/m^2)
% Output-
%     Qs => Incidence Direct solar radiation on a surface
%           per m^2
%
%     Qa => Incidence albedo radiation on a surface per m^2
%
%     Qir => Incidence Earth IR on a satellite surface (W/m2)
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% constants
sigma = 5.6704e-8;          % Stefan Boltzmann Constant
%                               (W/(m^2K))
Re = 6378.14;               % Radius of Earth
arg_peri = 0;               % Argument of perigee

P = period(Altitude,ecc);   % Orbit period (s)
t = 0:dt:no_period*P;       % time discretization of a
%                               period
```

```

jd = julian_date(2019,month,...
    21,0,0,0); % Julian date
r_sun = solar_position(jd); % Position of the sun
%% Determining the radiation according to the year
if month == 12
    Es = 1414; % Incidence solar radiation
    a = 0.4; % Albedo coefficient
    Te = 250; % Earth temperature

else
    Es = 1322; % Incidence solar radiation
    a = 0.19;
    Te = 260; % Earth temperature
end
%% Variable initialization
Rsun = r_sun;
Rsc = zeros(length(t),3);
Vsc = zeros(length(t),3);
RxV = zeros(length(t),3);

Qs = zeros(6,length(t));
Qa = zeros(6,length(t));
Qir = zeros(6,length(t));
%% Calculation
for j = 1:length(t)
    % Position, Velocity, and cross product of the two
    [Rsc(j,:),Vsc(j,:),RxV(j,:)] = keplere(inc,...
        raan,arg_peri,Altitude,ecc,t(j));
    % Unit vector normal to each surface
    n = Surf_unit_Vec(Rsc(j,:),Vsc(j,:),RxV(j,:),ecc,...
        Altitude,t(j), spinning, Surface_rotation);

    for i = 1:6

        gamma_m = acos(dot(n(i,:),Rsc(j,:))/...
            (norm(n(i,:))*norm(Rsc(j,:))));
        % View factor calculation
        VF = satellite_earth_VF(Altitude,gamma_m);
        light_switch(j) = shadow(Rsc(j,:), Rsun);
        S = Rsc(j,:)-Rsun; % Sun-satellite vector
        % angle between sun-satellite vector and ...
        % unit vector normal to the satellite surface
        teeta = acos(dot(S,n(i,:))/(norm(S)*norm(n(i,:))));
        % angle between earth-sun vector and
        % earth-satellite vector
        phii = acos(dot(Rsun,Rsc(j,:))/(norm(Rsun) ...
            *norm(Rsc(j,:))));
    end
end

```

```

% Incidence Direct solar radiation on a surface (W)
Qs(i,j) = Es*cos(teeta);
%Qs(i,j) = Es;
% Incidence albedo radiation on a surface (W)
Qa(i,j) = Es*a*VF*cos(phii);
% Incidence Earth IR on a satellite surface (W)
Qir(i,j) = sigma*VF*Te.^4;

if light_switch(j)==0
    % satellite is in a shadow ie. Qs = 0 and Qa = 0
    Qs(i,j) = 0;
    Qa(i,j) = 0;
end

if cos(teeta)<=0
    % satellite surface is not viewed from the sun
    Qs(i,j) =0;
end
if Qa(i,j) < 0
    % Albedo heat flux is zero for phii >= pi/2
    Qa(i,j) =0;
end
end
end
figure(1)
%% Draw the planet
[xx, yy, zz] = sphere(100);
surf(Re*xx, Re*yy, Re*zz);
colormap(light_gray)
caxis([-Re/100 Re/100])
shading interp

%% Draw and label the X, Y and Z axes
unitsun = r_sun/norm(r_sun); % unit vector of the sun
line([0 2*Re], [0 0], [0 0]); text(2*Re, 0,...
    0, 'X (Vernal equinox)')
line( [0 0], [0 2*Re], [0 0]); text( 0, 2*Re, 0, 'Y')
line( [0 0], [0 0], [0 2*Re]); text( 0, 0, 2*Re, 'Z')
line( [0 2*Re*unitsun(1,1)], [0 2*Re*unitsun(1,2)],...
    [0 2*Re*unitsun(1,3)]);
text(2*Re*unitsun(1,1),2*Re*unitsun(1,2),...
    2*Re*unitsun(1,3), 'Sun Vector')
% Plot the orbit, draw a radial to the starting point
% and label the starting point (o) and the final point (f)
hold on
plot3( Rsc(1:floor(length(t)),1),...

```

```

Rsc(1:floor(length(t)),2),...
Rsc(1:floor(length(t)),3),'k','LineWidth',3)
line([0 Rsc(1,1)], [0 Rsc(1,2)], [0 Rsc(1,3)])
text( Rsc(1,1), Rsc(1,2), Rsc(1,3), 'o')
text( Rsc(end,1), Rsc(end,2), Rsc(end,3), 'f')
view([1,1,.4])
hold off
end

```

---

## **Preliminarily thermal analysis**

```

%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% Preliminarily thermal analysis code for WHS    %%
%% HABTAMU MENGISTU                               %%
%% 2018-10-02                                       %%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
clear all; close all; clc;
%% Inputs
% Inputs for orbit definition
inc = 51.6;           % Inclination [Deg]
ecc = 0;              % eccentricity
RAAN = 180;           % RAAN [Deg]
month =12;            % month
h = 400;              % Altitude [km]
P = period(h,ecc);    % Period calculation [sec]
% Inputs for surface orientation
s = 0;               % Spinning rate [deg/orbit]
r = 45;              % Side wall orientation [deg]
% Inputs for thermal analysis
sigma = 5.67e-8;      % Stephen Boltzmann Constant
Tinf = 2.7;           % Deep space Temperature[K]
Egen = 5.75;          % Internal heat generation
area = 0.1.*[0.1,0.1,0.326, 0.326,...
    0.326,0.326];     % the 6 faces areas [m2]
A = [0.0028,0.027,0.003,0.004,0.001,...
    0.009];           % External surface area [m2]
emi = [0.86,0.14,0.85,0.88,0.83,0.14,...
    0.14];            % Surface emissivity
alp = [0.86,0.44,0.92,0.92,0.75,0.44,...
    0.44];            % Surface absorptivity
m = 2.103;            % mass [kg]
Cp = 961;             % specific heat [J/kg.K]
k = 175;              % Thermal conductivity [W/mK]

```

```

% Time discretization
No_p = 1;                                % number of orbits
dt = 10;                                % Time step [sec]
t = 0:dt:P;                             % Time vector [sec]
time = t./3600;                          % time in hour

%% Variable Initialization
Ttra_WHS = zeros(1,length(t));
qs_abs= zeros(1,length(t));             % Total absorbed solar [W]
qa_abs = zeros(1,length(t));            % Total absorbed albedo [W]
qir_abs = zeros(1,length(t));           % Total absorbed Infrared [W]
Qtot_out = zeros(1,length(t));          % Total emitted thermal [W]
Qtot_in = zeros(1,length(t));           % Total absorbed thermal [W]
surf_emi = zeros(1,6);                  % Effective surface emissivity
surf_abs = zeros(1,6);                  % Effective surface Absorptivity

%% Calculating incidence environmental thermal radiation
[qs,qa, qir] = external_heat_flux(inc,h,ecc,month,...
    dt,No_p,RAAN,s,r);
% Incidence thermal radiation plotting
Q = zeros(6,length(t),3);
Q(:, :, 1) = qs; Q(:, :, 2) = qa; Q(:, :, 3)=qir;
S = strings(1,3);S(1,1)="Solar [W/m^{2}]";
S(1,2)="Albedo [W/m^{2}]";S(1,3)="Infrared [W/m^{2}]";
for p = 1:3
    figure
    plot(time(1:555),Q(1,1:555,p),'b-+', 'LineWidth',...
        2, 'MarkerIndices',1:50:length(qs(1,1:555)))
    hold on
    plot(time(1:555),Q(2,1:555,p),'g-*', 'LineWidth',...
        2, 'MarkerIndices',1:50:length(qs(1,1:555)))
    hold on
    plot(time(1:555),Q(3,1:555,p),'y-s', 'LineWidth',...
        2, 'MarkerIndices',1:50:length(qs(1,1:555)))
    hold on
    plot(time(1:555),Q(4,1:555,p),'m-o', 'LineWidth',...
        2, 'MarkerIndices',1:50:length(qs(1,1:555)))
    hold on
    plot(time(1:555),Q(5,1:555,p),'r-^', 'LineWidth',...
        2, 'MarkerIndices',1:50:length(qs(1,1:555)))
    hold on
    plot(time(1:555),Q(6,1:555,p),'c-<', 'LineWidth',...
        2, 'MarkerIndices',1:50:length(qs(1,1:555)))
    xlabel('Time (hr)')
    ylabel(S(1,p))
    legend('Base plate','FSP','SW 1','SW 2',...
        'SW 3', 'SW 4')

```

```

    hold off
end
%% Effective surface optical property computation
% Effective surface emissivity calculation for 6 surfaces
surf_emi(3:6) = (2*A(1)*emi(1)+A(2)*emi(2))/(2*A(1)+A(2));
surf_emi(2) = (2*A(3)*emi(3) +A(4)*emi(4))/(2*A(3)+A(4));
surf_emi(1) = (A(5)*emi(5)+A(6)*emi(6))/(A(5)+A(6));
emi_ave = sum(area.*surf_emi)...
    /sum(area); % Total effective surface emissivity
% Effective surface absorptivity calculation for 6 surfaces
surf_abs(3:6) = (2*A(1)*alp(1)+A(2)*alp(2))/(2*A(1)+A(2));
surf_abs(2) = (2*A(3)*alp(3) +A(4)*alp(4))/(2*A(3)+A(4));
surf_abs(1) = (A(5)*alp(5)+A(6)*alp(6))/(A(5)+A(6));
abs_ave = sum(area.*surf_abs)...
    /sum(area); % Total effective surface absorptivity
% Average absorbed environmental flux computation
qs_ave = mean(qs'); % Average solar [W/m2]
qa_ave = mean(qa'); % Average albedo [W/m2]
qir_ave = mean(qir'); % Average Infrared [W/m2]

qs_sta = sum((surf_abs.*area).*qs_ave);
qa_sta = sum((surf_abs.*area).*qa_ave);
qir_sta = sum((surf_emi.*area).*qir_ave);

%% Steady state temperature computation
TWHS = ((qs_sta+qa_sta+qir_sta)/(sigma*emi_ave*sum(area))...
    + Tinf^4+Egen/(sigma*emi_ave*sum(area)))^(1/4);
Tin = TWHS;
err = 25;

%% Numerical integration of the transient thermal equation
while err >=0.1
    for j = 1:length(t)

        % Absorbed solar, albedo, and Infrared calculation
        qs_abs(1,j) = sum((surf_abs.*area).*qs(:,j)');
        qa_abs(1,j) = sum((surf_abs.*area).*qa(:,j)');
        qir_abs(1,j) = sum((surf_emi.*area).*qir(:,j)');
        Qtot_in(1,j) = qs_abs(1,j)+qa_abs(1,j)+qir_abs(1,j);
        % Emitted Total radiation calculation
        Qtot_out(1,j) = emi_ave(1)*sigma*sum(area)...
            *(Tin^4-Tinf^4);
        % Calculation of temperature at time t+dt
        Ttra_WHS(1,j) = Tin +dt*(Qtot_in(1,j)...
            -Qtot_out(1,j)+Egen)/(m*Cp);
        Tin = Ttra_WHS(1,j);
    end
end

```

```

    end
    err = abs(Ttra_WHS(1,1)-Ttra_WHS(1,end));

end

Ttra_WHS = Ttra_WHS-273;
TWHS = TWHS - 273;
TWHS
%% Total absorbed thermal radiation plotting
Qtot = zeros(3,length(t));
Qtot(1,:) = qs_abs; Qtot(2,:)=qa_abs; Qtot(3,:)=qir_abs;

figure
for p = 1:3
    plot(time(1:555),Qtot(p,1:555),'LineWidth',2)
    hold on
end
    xlabel('Time [hr]')
    ylabel('Total absorbed environmental theramal...
    radiations by ASTUSat for WHS [W]')
    legend('Solar','Albedo', 'Earth Infrared')
hold off

%% Plotting transient temperature
figure
plot(time,Ttra_WHS(1,:),'linewidth',2)
ylabel('Spacecraft ASTUSat temperature for WHS [^oC]')
xlabel('time [hr]')

hold off

```

---

## **PCB's Thermal property calculation**

```

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% PCB's thermal Property calculation                                %%
%% HABTAMU MENGISTU                                                %%
%% 2018-11-15                                                       %%
%% Adama Science and Technology university                          %%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
clc
clear all
close all
%% PCB Thermal property calculator
% User inputs

```

```

n = [6,4,6,2,2,4,6,6,2];           % number of copper layer
s =n-1;                             % number of FR4 layer
t = 1.6;                             % total thickness of PCB
[mm]
t_Cu = 0.035;                       % thickness of copper [mm]
t_FR4 = (t-n.*t_Cu)./s;             % thickness of FR4 [mm]
V = [0.0086,0.002,0.0086,0.0086,0.0086,0.0086,...
     0.008,0.008,0.0086].*0.0016;   % Volume of the PCB [m^3]
mass_pcb = [77,11,77,75,49,85,24,...
            47,80].*10^-3;          % mass of the PCB [kg]
% Thermal conductivity [W/m.k] of PCB stackup's
K_Cu = 400;                          % Copper thermal
conductivity
Kxy_FR4 = 0.29;                     % FR4 thermal conductivity
% Density of PCB stackup's [kg/m^3]
raw_Cu = 8960;                      % Copper density
raw_FR4 = 1850;                     % FR4 density
% Specific heat [J/kg.K]
C_Cu = 376.8;                       % Copper specific heat
C_FR4 = 600;                        % FR4 specific heat
% Volume fraction
x_Cu = n.*t_Cu/t;                   % copper in PCB
x_FR4 = 1-x_Cu;                     % FR4 in PCB
% Density PCB [kg/m^3]
density_PCB = mass_pcb./V
% Mass fraction
x_m_cu = (x_Cu.*raw_Cu)./(...
         (density_PCB));             % Copper in PCB
x_m_FR4 = 1-x_m_cu;                 % FR4 in PCB
% Thermal conductivity [W/m.K]
K_PCB = x_Cu.*K_Cu + x_FR4.*Kxy_FR4
% Specific heat [J/kg.K]
cp_PCB = x_m_cu.*C_Cu + x_m_FR4.*C_FR4

```

---

## **Thermal resistance calculation**

```

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%% PCB's thermal Resistance calculation                                %%
%% HABTAMU MENGISTU                                                  %%
%% 2018-11-15                                                         %%
%% Adama Science and Technology university                            %%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
close all
clear all
clc

```

```

%% Thermal resistance of connectors

% user inputs
Apin = 0.508*10^-6;           % Area of pins [m^2]
kpin = 75;                   % Thermal conductivity of pins [W/mK]
Lcon = 0.015;                % Length of connectors [m]
R_conectors = Lcon/(104*Apin...
    *kpin)                   % Connectors thermal resistance [K/W]

%% Thermal resistance of bus spacers

% User inputs
do = 4.5*10^-3;              % Outer diameter of spacer [m]
di = 3.3*10^-3;              % Inner diameter of spacer [m]
Lspa = [7.2,15,15,15,15,15,5,5,...
    6]*10^-3;                % Length of spacers [m]
kspa = 155.8;                % Thermal conductivity of spacers
[W/mK]
% Thermal resistance calculation
Aspa = pi*(do^2-di^2)/4;     % Area of spacers [m^2]
R_contact = 12.6;            % Contact resistance [K/W]
R_spacer = Lspa./(Aspa*...
    kspa);                   % thermal resistance of a spacer [K/W]
R_Spacer_tot = R_spacer + ...
    2*R_contact;            % total resistance of each spacer
[K/W]
Rtot = zeros(1,9);           % Total resistance between two PCB's
[K/W]
Rtot(2:6) = 1./(4./R_Spacer_tot(2:6) + 1/R_conectors);
Rtot(1) = R_Spacer_tot(1)./4;
Rtot(7:9) = R_Spacer_tot(7:9)./4;
Rtot'

%% Thermal coupling between batteries and PCBs

Abat = 0.002;                % Surface area of the battery [m^2]
kkap = 0.12;                 % Thermal conductivity of Kapton
[W/mK]
tkap = 2.54 *10^-5;          % thickness of kapton 1 mil
R_battery_pcb = tkap/(Abat...
    *kkap)                   % Thermal resistance [K/W]

%% thermal resistance calculation of camera spacers

doc = 4.5*10^-3;             % Outer diameter of spacer [m]
dic = 3.3*10^-3;             % Inner diameter of spacer [m]

```

```

Lspac = 13*10^-3;           % length of spacer [m]
kspac = 16;                 % Thermal conductivity of spacers
[W/mK]
% thermal resistance calculation of spacer
Aspac = pi*(doc^2-...
    dic^2)/4;               % cross sectional area of spacer [m^2]
R_camera = Lspac/(4*Aspac...
    *kspac)                 % Thermal resistance [K/W]

%% Thermal resistance calculation of ADCS spacers

% length of spacers [m]
Lele = 0.04;               % spacers between DB and ADCS housing
Lcon = 0.0064;             % spacers between DB and CB
% Thermal resistance calculation [K/W]
R_DB_ADCS_H = Lele/(4*...
    Aspa*kspa)              % Between DB and ADCS housing
R_DB_CB = Lcon/(4*Aspa...
    *kspa)                  % Between DB and CB

```

## Appendix B.

### B. Effective Optical Properties

#### Effective optical properties

Effective surface emissivity and absorptivity of a surface containing various optical properties are calculated using the following equations

$$\varepsilon_{ave} = \frac{\varepsilon_1 A_1 + \varepsilon_2 A_2 + \varepsilon_3 A_3 + \dots}{A_{tot}} \quad 6-1$$

$$\alpha_{ave} = \frac{\alpha_1 A_1 + \alpha_2 A_2 + \alpha_3 A_3 + \dots}{A_{tot}} \quad 6-2$$

Where:  $\varepsilon_{ave}$  and  $\alpha_{ave}$  are effective surface emissivity and absorptivity respectively,  $A_{tot}$  is total Surface area,  $\varepsilon_1$  is emissivity of the surface at a particular area  $A_1$

Table B-1: External surface optical properties of ASTUSat without the application of thermal control surface finishes

| Surfaces               | Parts       | Quantity | A (m <sup>2</sup> ) | $\varepsilon$ | $\alpha$ |
|------------------------|-------------|----------|---------------------|---------------|----------|
| <b>Side wall faces</b> | Rails       | 8        | 0.003               | 0.86          | 0.86     |
|                        | Side walls  | 4        | 0.027               | 0.14          | 0.44     |
| <b>Top</b>             | Solar cell  | 2        | 0.003               | 0.85          | 0.92     |
|                        | Solar Panel | 1        | 0.004               | 0.88          | 0.92     |
| <b>Bottom</b>          | Camera lens | 1        | 0.001               | 0.83          | 0.75     |
|                        | Base plate  | 1        | 0.009               | 0.14          | 0.44     |

The effective optical properties calculated for each six faces of ASTUSat using Equation B-1 and B-2 are shown in Table B-2.

Table B-2: Effective surface optical properties on the six surfaces of ASTUSat

| <b>Surfaces</b>            | <b><math>A_{tot}</math> (m<sup>2</sup>)</b> | <b><math>\epsilon_{ave}</math></b> | <b><math>\alpha_{ave}</math></b> |
|----------------------------|---------------------------------------------|------------------------------------|----------------------------------|
| <b>Bottom</b>              | 0.01                                        | 0.2090                             | 0.4710                           |
| <b>Top</b>                 | 0.01                                        | 0.8620                             | 0.9200                           |
| <b>Side wall<br/>faces</b> | 0.033                                       | 0.2637                             | 0.5121                           |
|                            | 0.033                                       | 0.2637                             | 0.5121                           |
|                            | 0.033                                       | 0.2637                             | 0.5121                           |
|                            | 0.033                                       | 0.2637                             | 0.5121                           |
| <b>Total</b>               | 0.152                                       | 0.2998                             | 0.5365                           |

## Appendix C.

### C. Average Thermal Properties Computing Schemes

#### Thermal properties of PCB

In many thermal calculations, it is convenient to treat PCB which is a layered structure as a homogeneous material, with effective specific heat, effective density, and two effective thermal conductivities: one describing heat flow within the plane ( $K_{in-plane}$ ) and another for heat flow through the thickness of the plane ( $K_{through}$ ). The thermal conductivity within the plane is only used in this thesis because all PCBs are meshed 2D mesh which assumes heat is constant through the thickness of the PCB. The equations for calculating each of these conductivities, given the values for the thickness and thermal conductivity of each layer, are presented below.

$$k_{in-plane} = \frac{\sum layers k_i f_i \delta_i}{\delta_y} \quad 6-3$$

And

$$k_{through} = \frac{\delta_y}{\sum layers \frac{\delta_i}{k_i f_i}} \quad 6-4$$

Where:  $k_i$  is the thermal conductivity of the i-layer,  $f_i$  is the average cross-section area occupied by the i-layer,  $\delta_i$  is the i-layer thickness in y-direction, and  $\delta_y$  is the total thickness of the PCB.

By assuming constant thickness of the copper and FR4 layers, and setting area fraction  $f_i = 1$  we can express the effective thermal conductance in terms of volume fractions:

$$x_{vol,cu} = \frac{n.t_{cu}}{t_{PCB}} \quad 6-5$$

$$x_{vol,FR4} = 1 - x_{vol,cu} \quad 6-6$$

$$k_{PCB} = x_{vol,cu} k_{cu} + x_{vol,FR4} k_{FR4} \quad 6-7$$

Where:  $x_{vol}$  volume fraction,  $t$  is thickness of layers,  $t_{PCB}$  is thickness of PCB,  $k_{PCB}$  is thermal conductivity of PCB. The subscripts ‘cu’ and ‘FR4’ indicate copper and FR4 materials.

The effective specific heat of the PCB depends on the mass fraction of copper and mass fraction of FR4 presented in the PCB. The mass fraction of the two materials is expressed in terms of volume fraction of the materials.

$$x_{m,Cu} = \frac{x_{vol,Cu} \rho_{Cu}}{x_{vol,Cu} \rho_{Cu} + x_{vol,FR4} \rho_{FR4}} \quad 6-8$$

$$x_{m,FR4} = 1 - x_{m,Cu} \quad 6-9$$

$$c_{p,PCB} = x_{m,Cu} c_{p,Cu} + x_{m,FR4} c_{p,FR4} \quad 6-10$$

Where:  $x_m$  is mass fraction, and  $c_{p,PCB}$  is specific heat of PCB

Finally, the effective density of the PCB  $\rho_{PCB}$  can be calculated using the following equation

$$\rho_{PCB} = x_{vol,Cu} \rho_{Cu} + x_{vol,FR4} \rho_{FR4} \quad 6-11$$

## Appendix D.

### D. Incidence Environmental Fluxes

The average incidence environmental thermal radiations in a day on the six faces of the satellite ASTUSat is computed using MATLAB program and they are shown from Figure D-1 to D-3 for one year

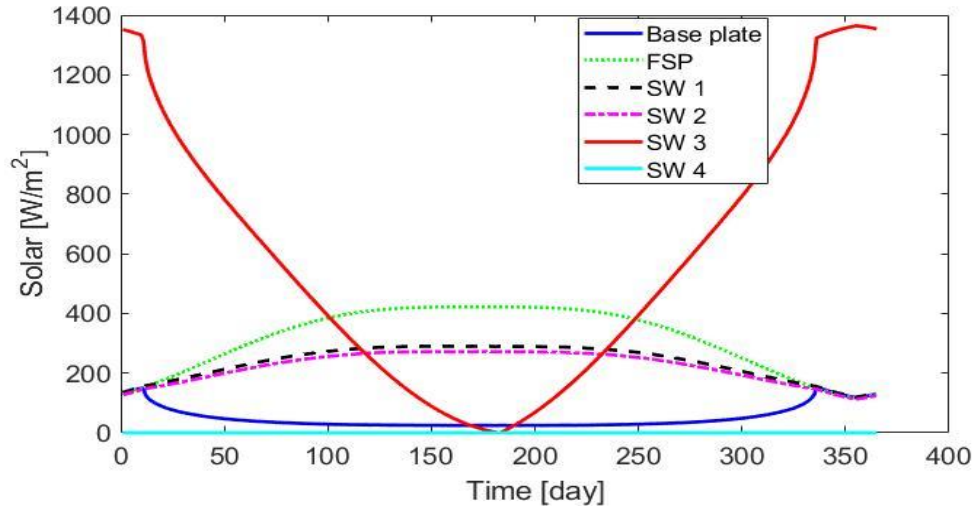


Figure D-1: Variation of average incidence solar thermal radiation on the 6 faces of ASTUSat within a year

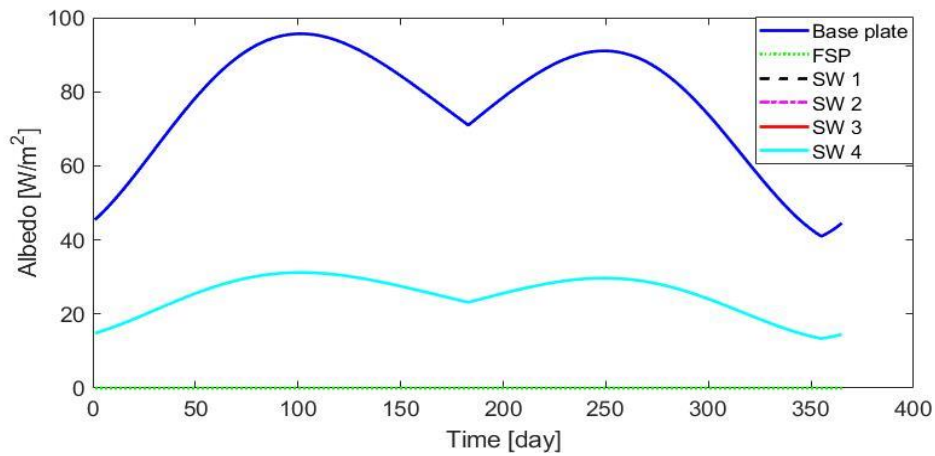


Figure D-2: Variation of average incidence albedo on the 6 faces of ASTUSat within a year

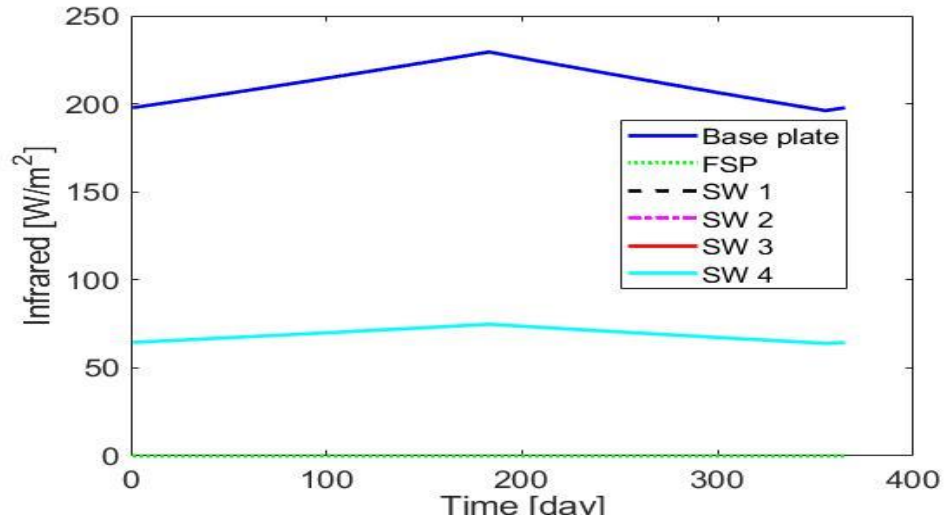


Figure D-3: Variation of average incidence Earth infrared on the 6 faces of ASTUSat within a year

Incidence environmental thermal radiations on the six faces of the satellite ASTUSat computed for various months using MATLAB program and they are shown from Figure D-4 to D-15:

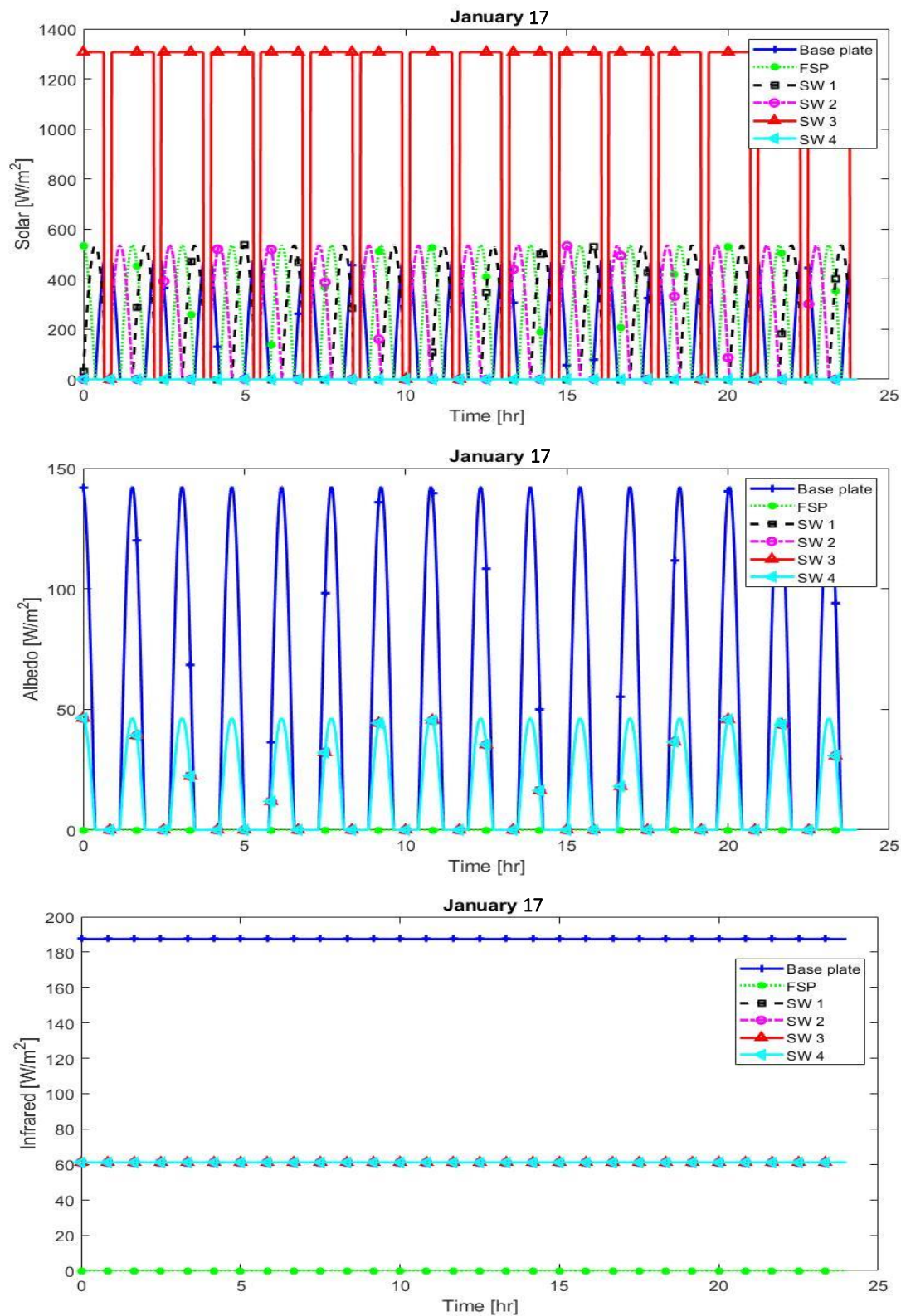


Figure D-4: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in January 17

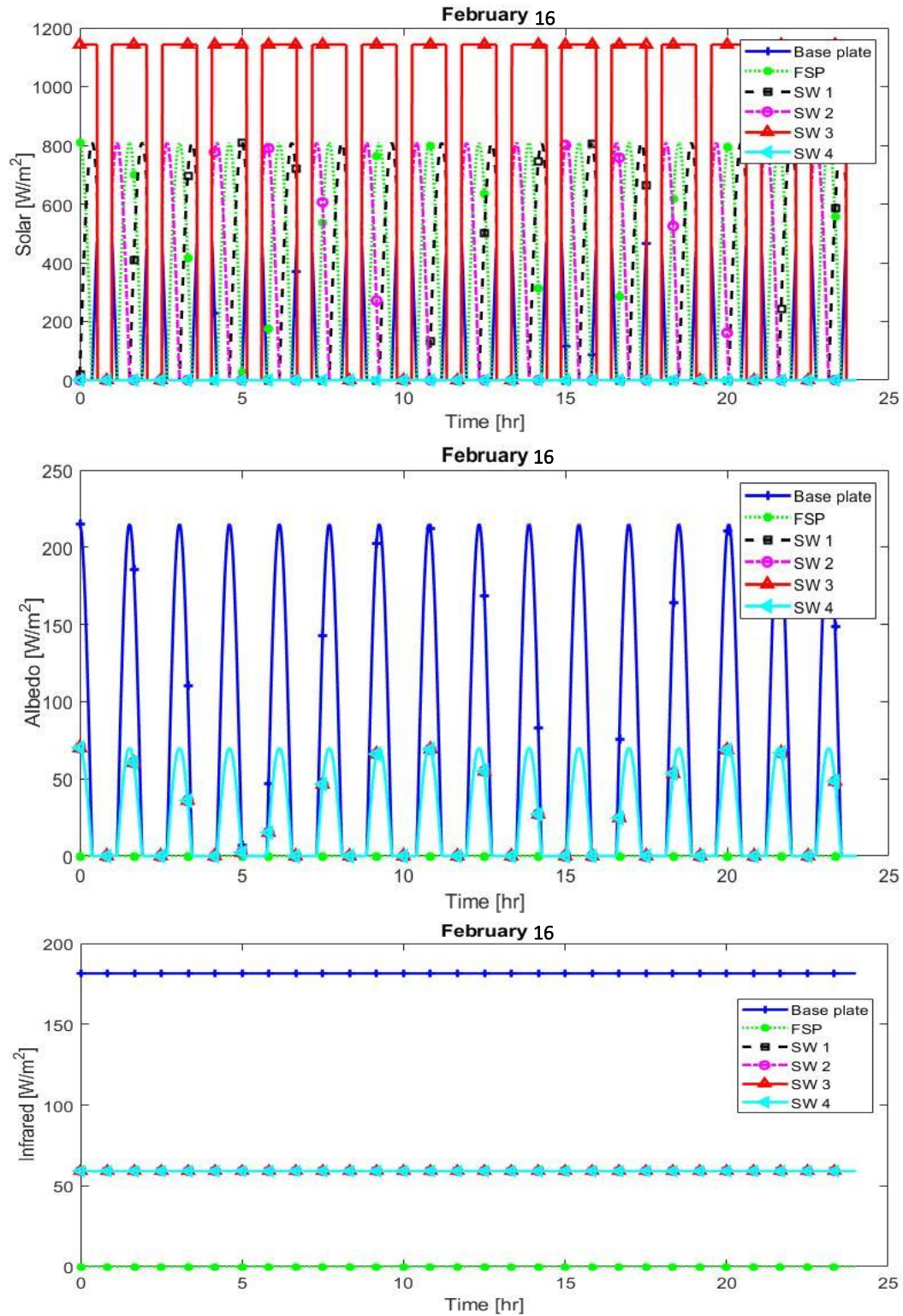


Figure D-5: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in February 16

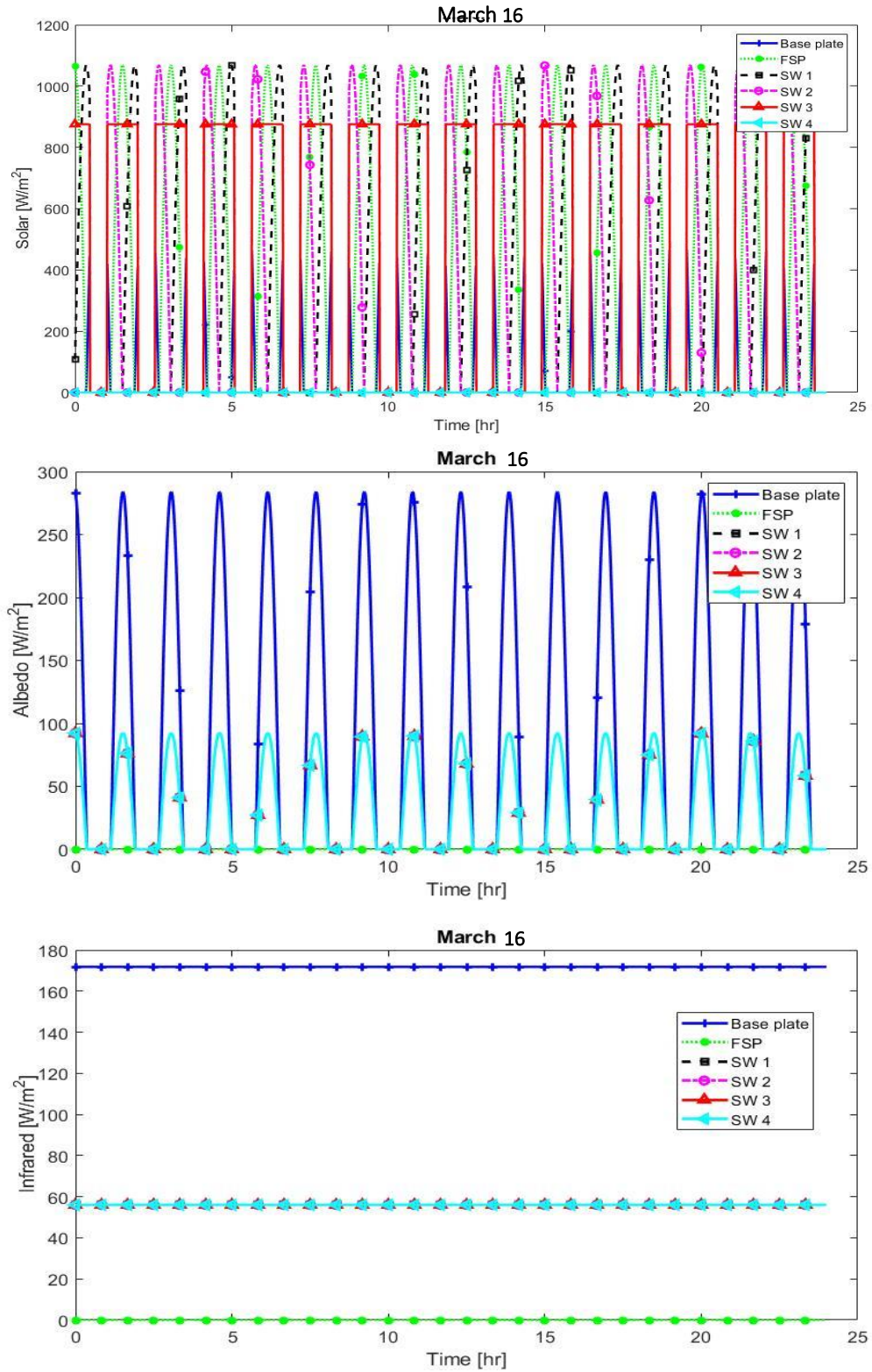


Figure D-6: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in March 16

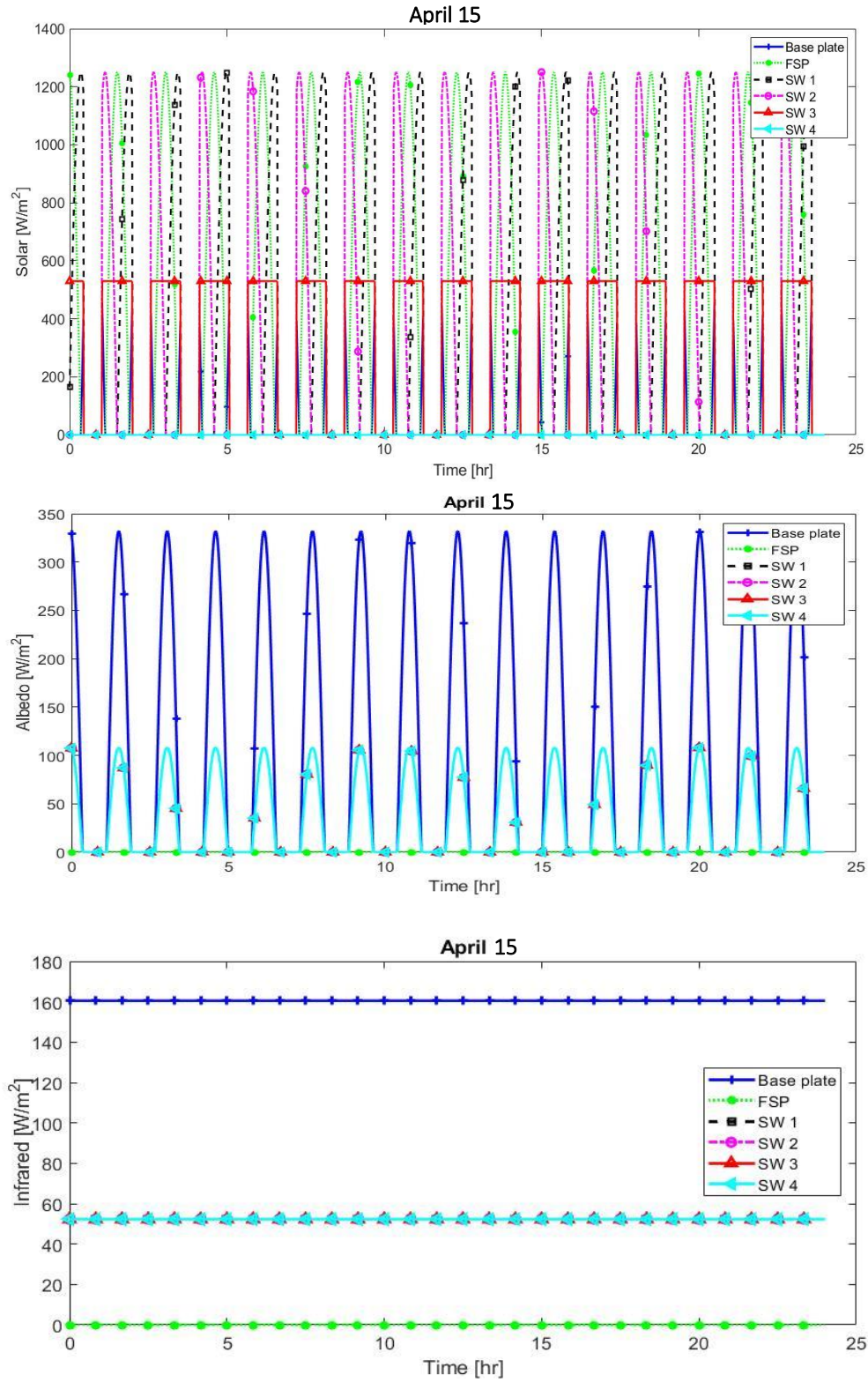


Figure D-7: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in April 15

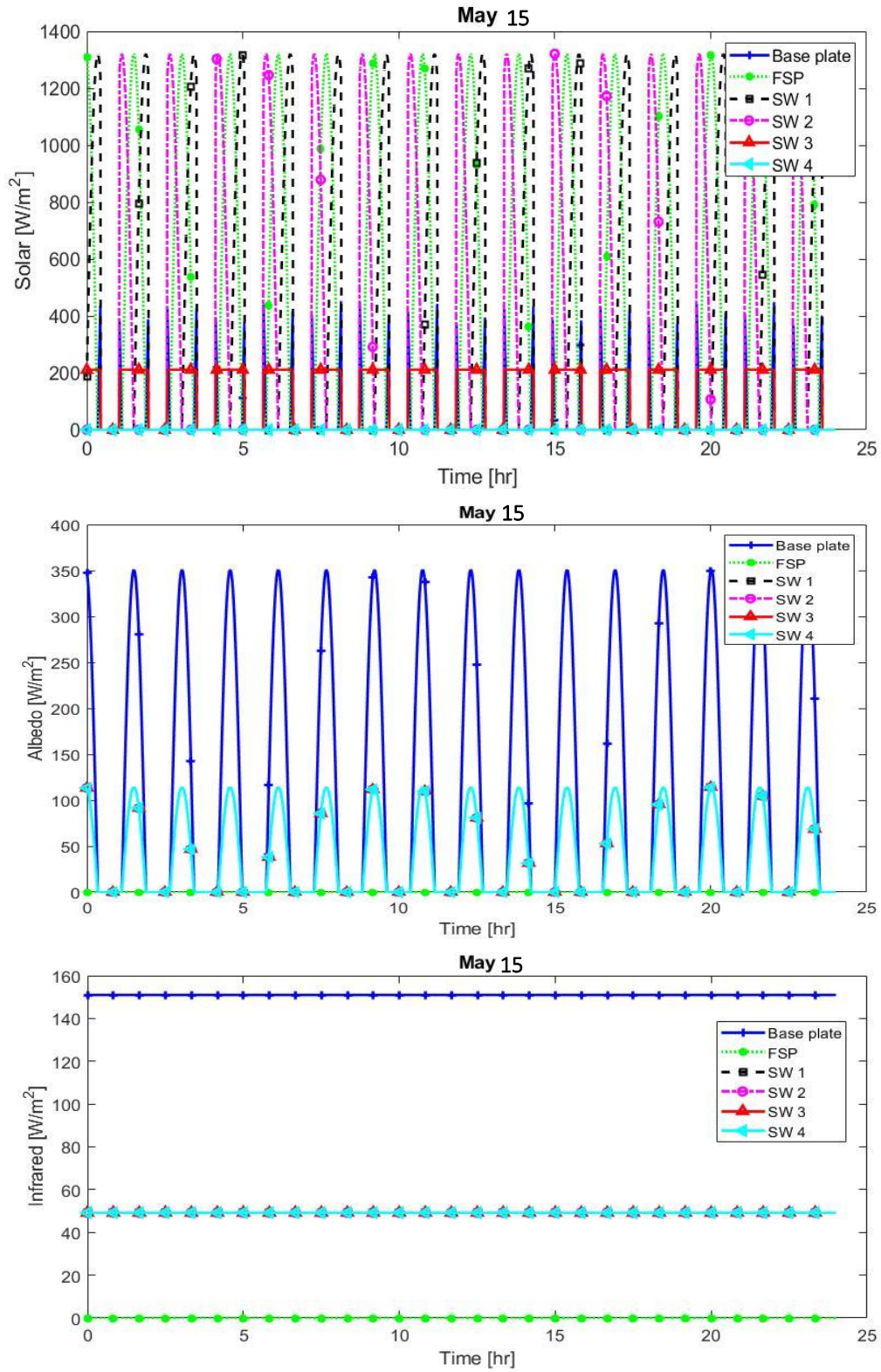


Figure D-8: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in May 15

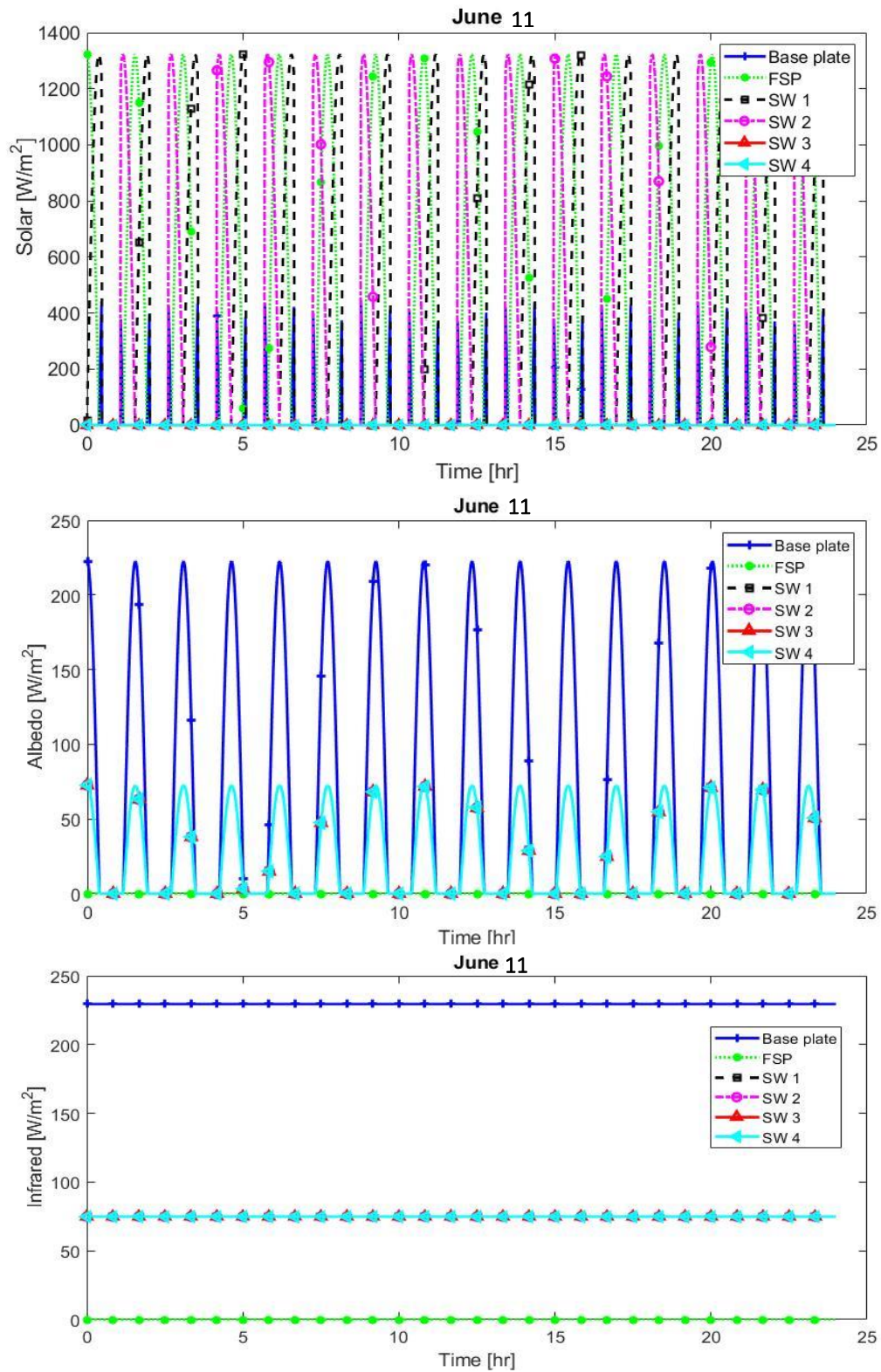


Figure D-9: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in June 11

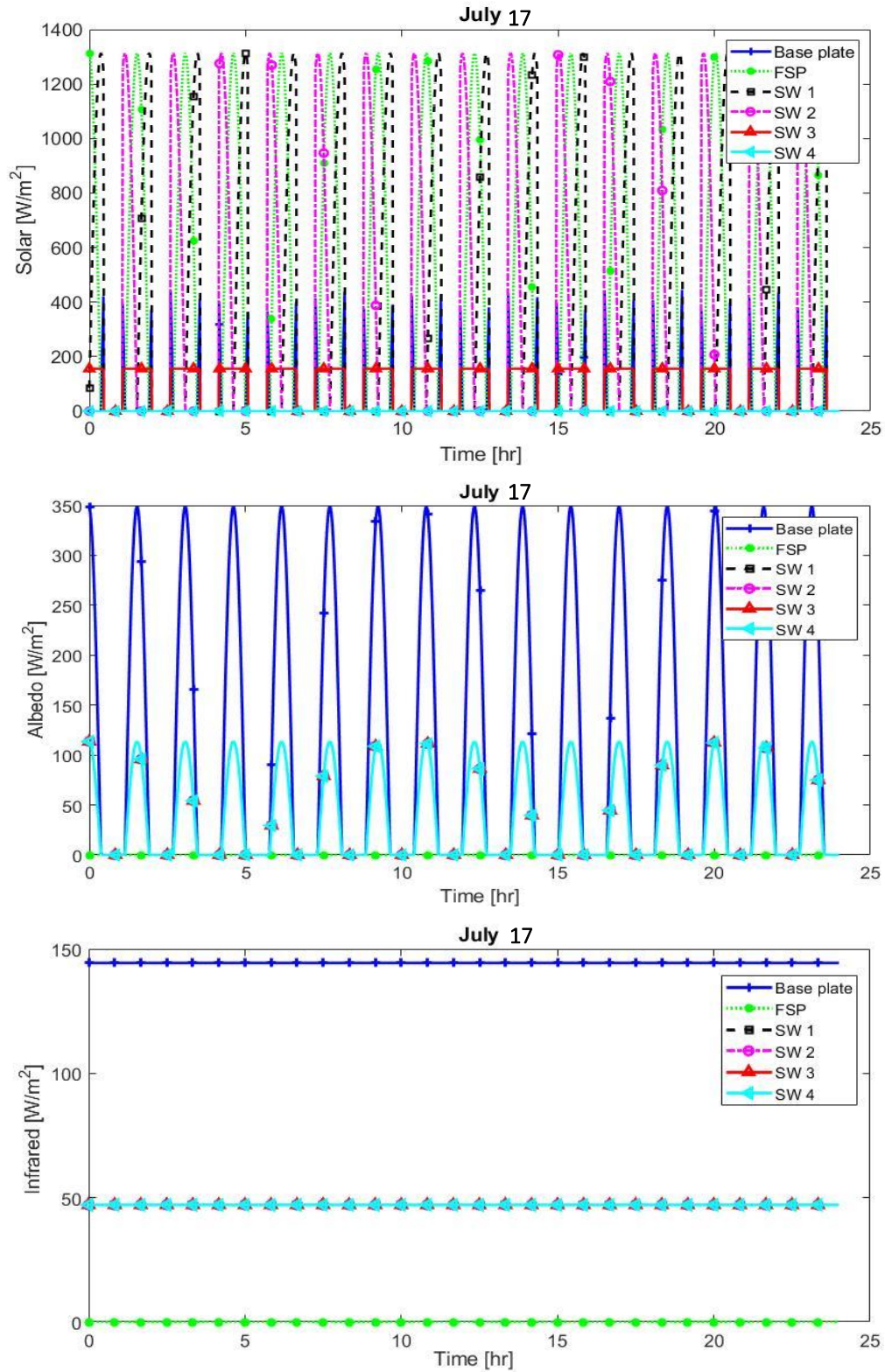


Figure D-10: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in July 17

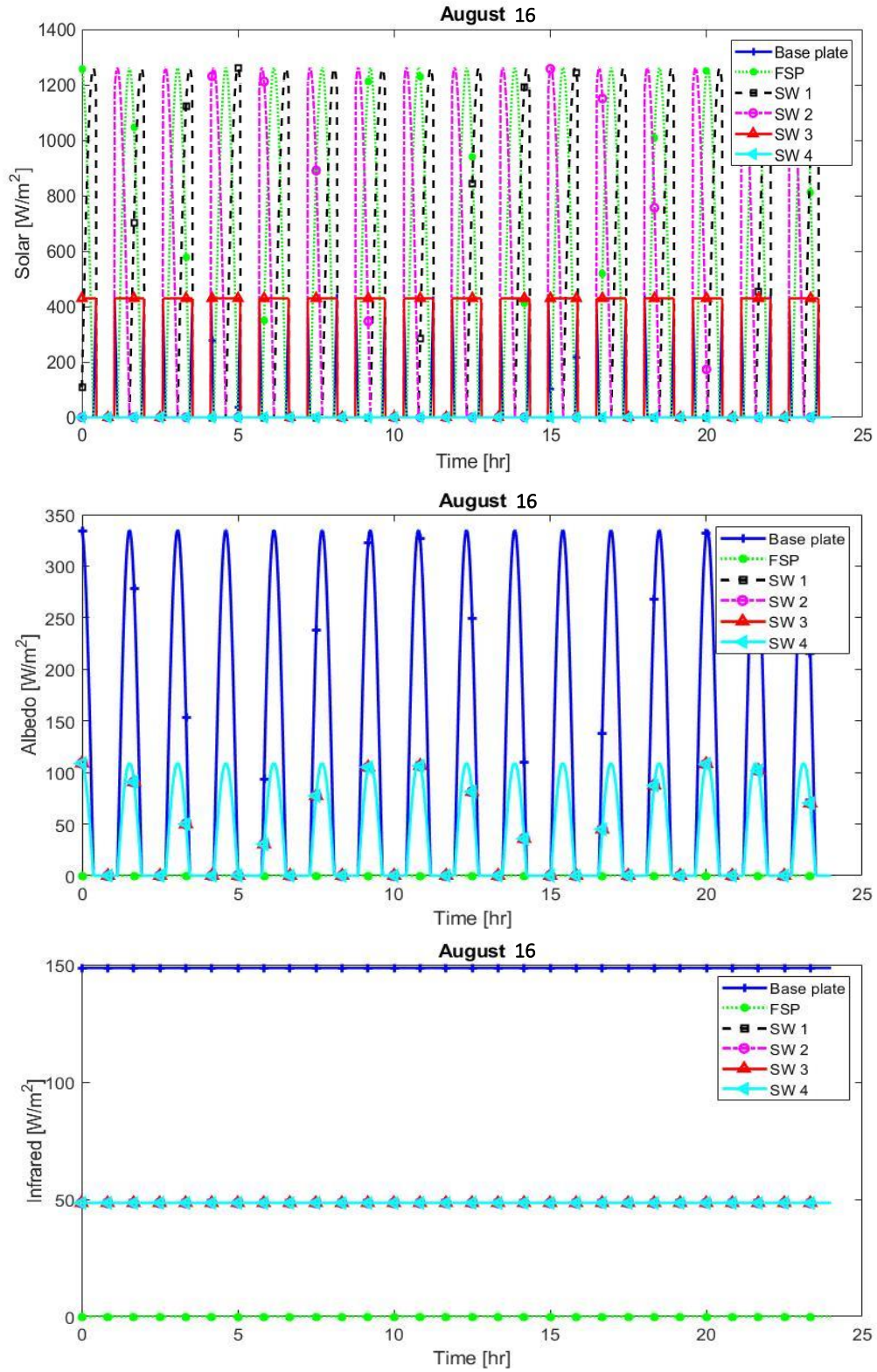


Figure D-11: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in August 16

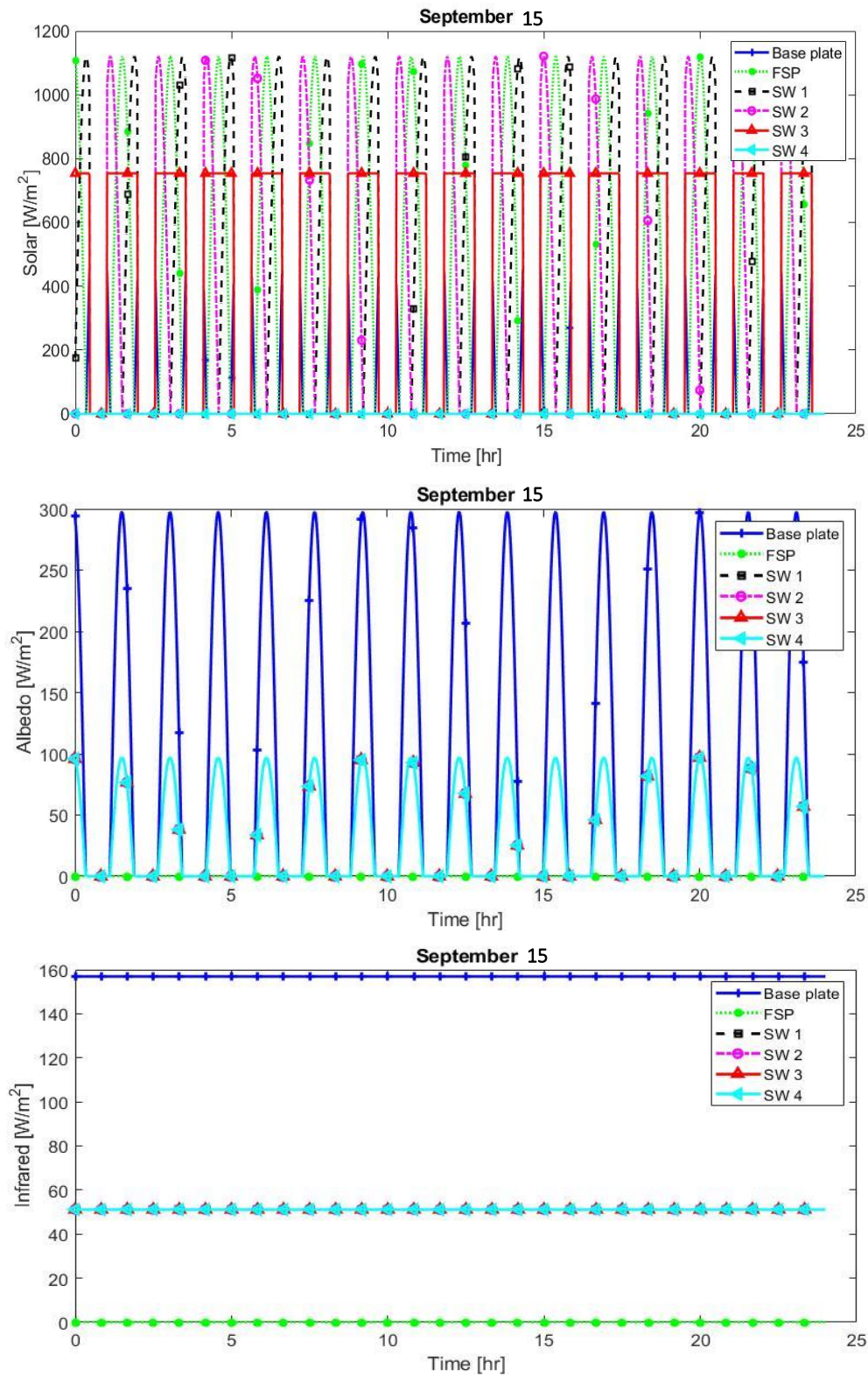


Figure D-12: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in September 15

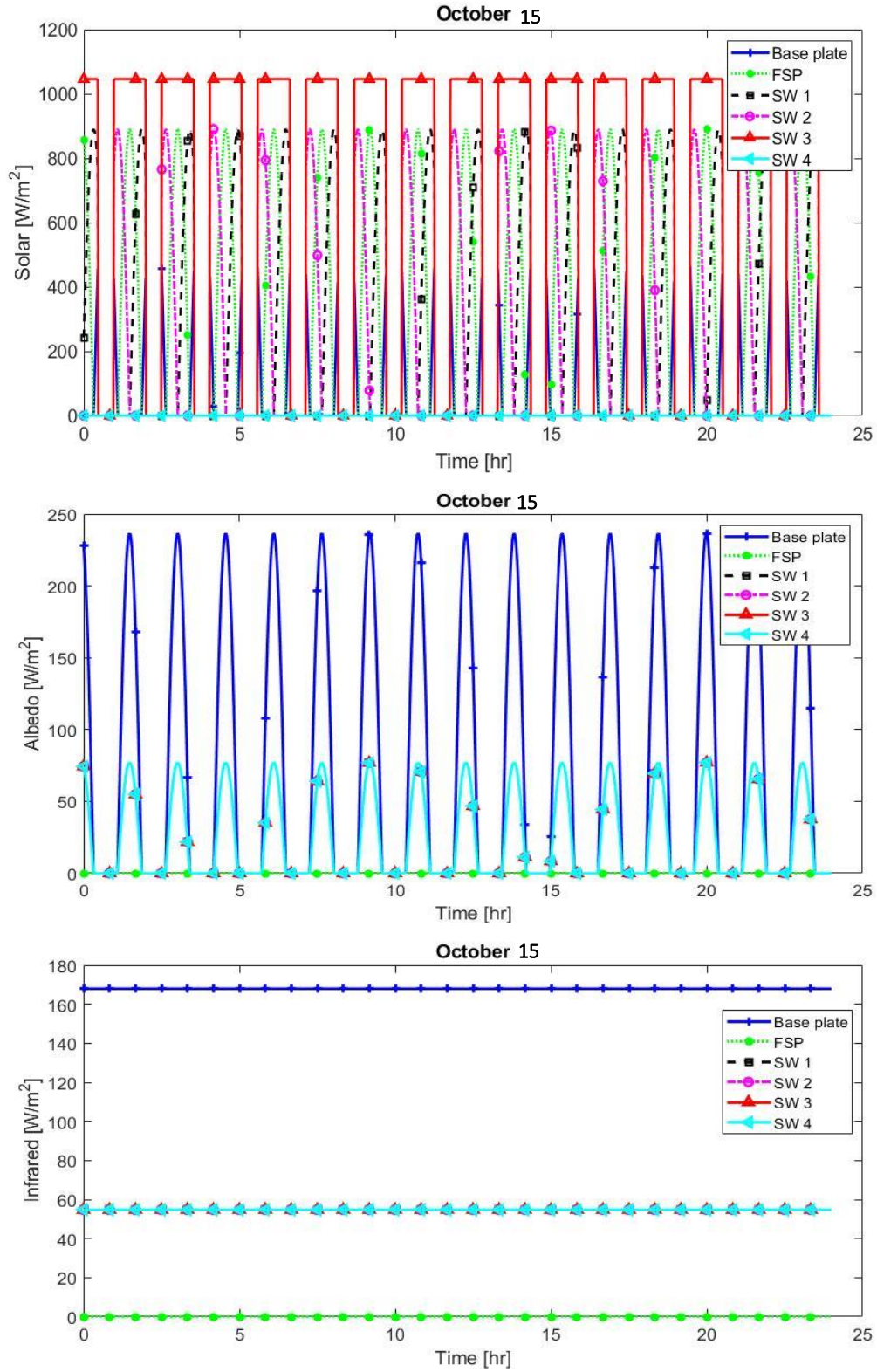


Figure D-13: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in October 15

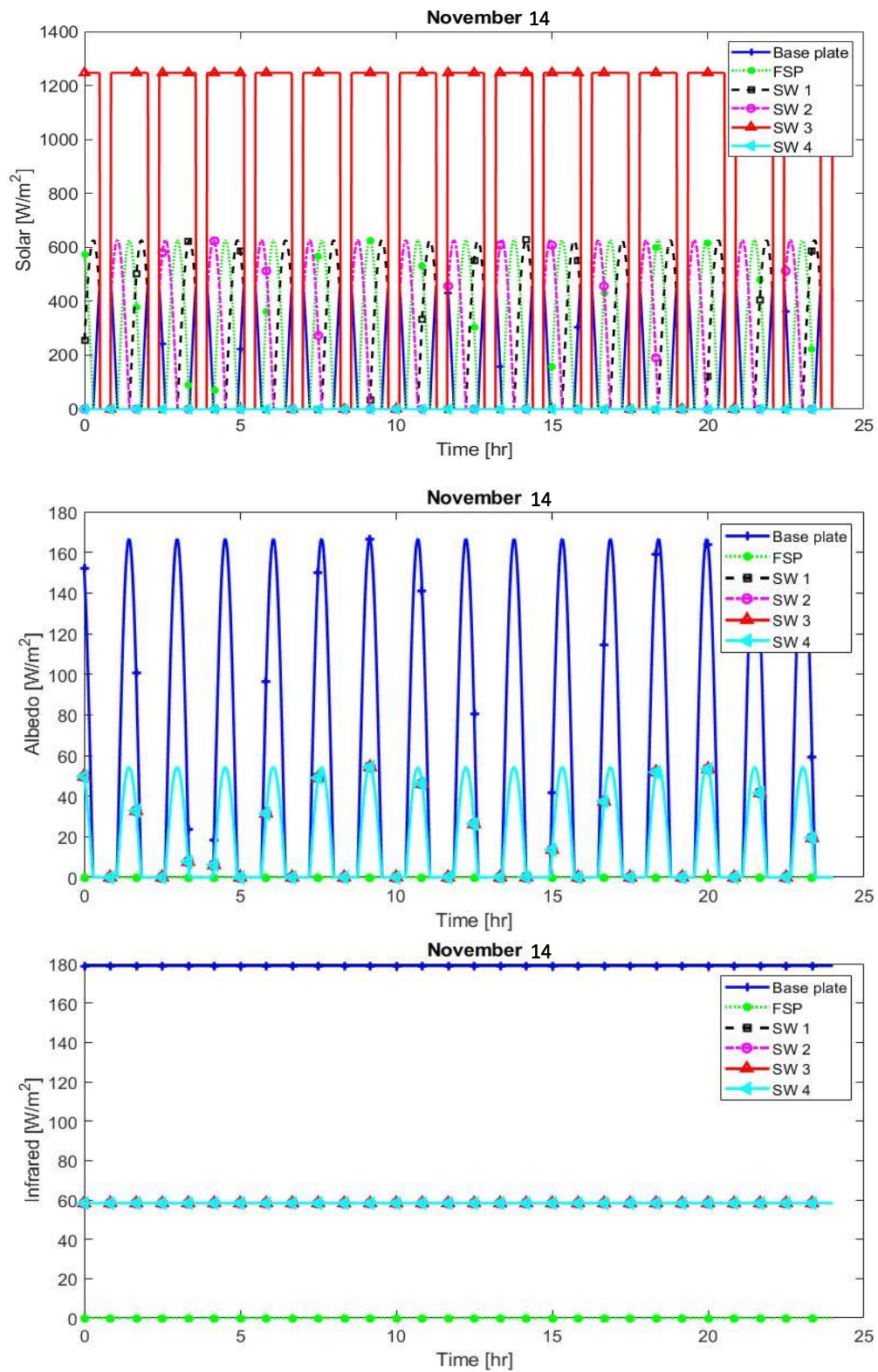


Figure D-14: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in November 14

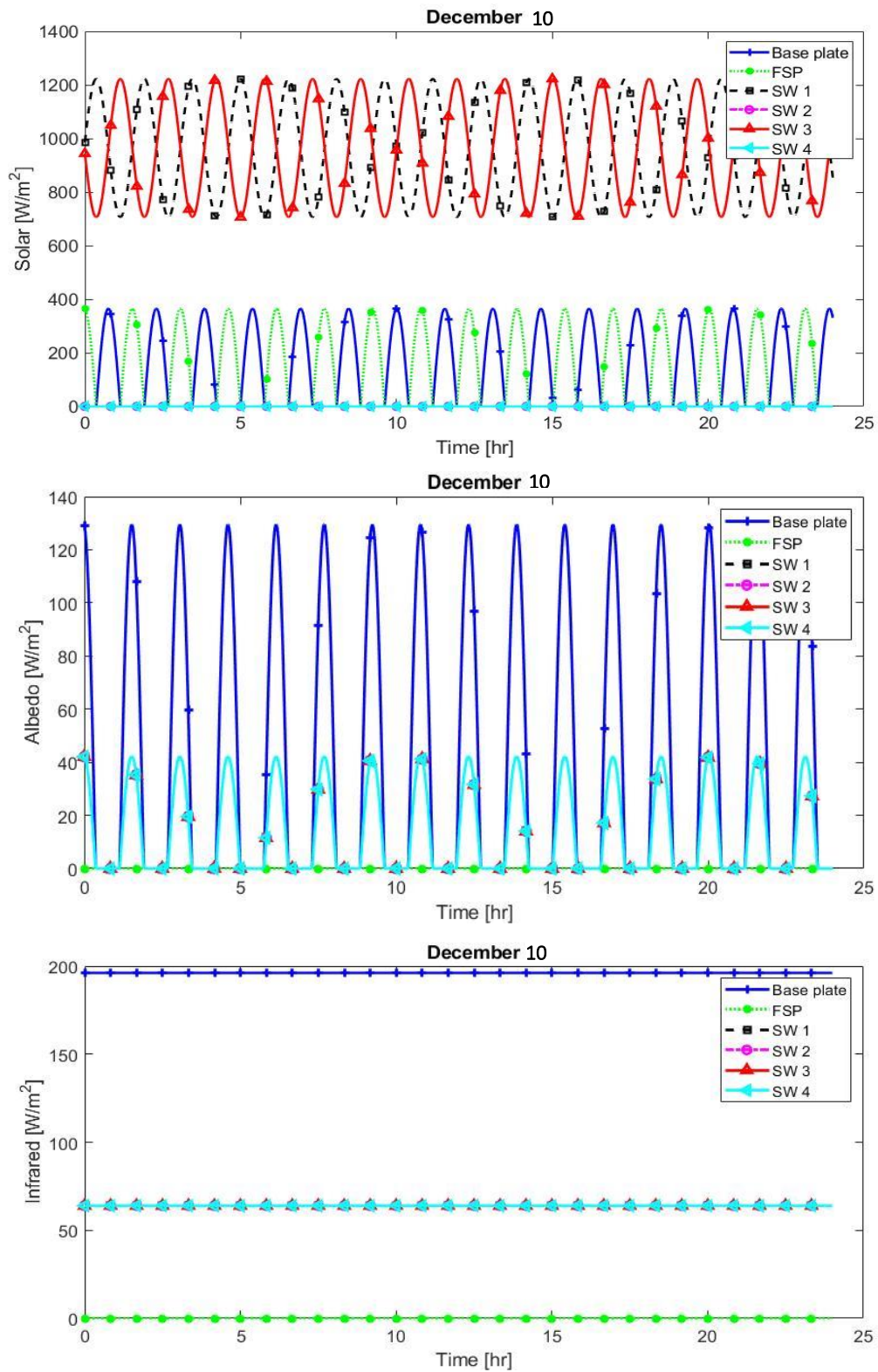


Figure D-15: The incidence solar, albedo, and infrared fluxes on each external surface of ASTUSat in December 10