THERMAL MODELING OF NANOSAT

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ABSTRACT

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Advances in computer technologies and manufacturing processes allow creation of highly sophisticated components in compact platform. For example, a small scale satellite, such as the CubeSat, can now be used for scientific research in space rather than big scale project like the International Space Station (ISS). Recently a team of undergraduate and graduate students at SJSU has the opportunity to collaborate on designing and building a miniature size CubeSat with the dimension of 10x10x10 cm. Although the integration of compact electronics allows sophisticated scientific experiments and missions to be carried out in space, the thermal control options for such small spacecraft are limited. For example, because of its small size there is no room for dedicated radiator or insulation panels. To minimize mass of the thermal control system while keeping the electronics at safe operating conditions, this thesis aims at studying the external orbital radiation heat flux the CubeSat is expected to expose to and the steady state heat conduction of the internal electronics. If the operating temperature from these heating conditions causes issue, appropriate thermal control solutions will be presented.

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Chapter 1
Introduction

1.1 Introduction to CubeSat

Advances in computer technologies and manufacturing processes allow creation of highly sophisticated components in compact platform. This miniaturization of components allow small scale satellite, such as the CubeSat, to be used for scientific research in space rather than big scale project like the International Space Station (ISS). The CubeSat concept was originally developed in 1999 by Dr. Jordi Puig-Suari from the California Polytechnic State University and Professor Bob Twiggs from the Stanford University [1]. The goal of the CubeSat project was to reduce development time and to increase launch opportunity through standardized satellite buses, structures and subsystems. This allows academia and commercial entities to perform space research at affordable price. This also creates opportunities for many undergraduate and graduate students to have hands-on experience building small payload satellite that can be launch to space as a secondary payload. A typical size of a CubeSat is a 10 cm cube aluminum T6061 structure with a total mass of up to 1 kg and operates autonomously in orbit. This small satellite design model can easily be modified to accommodate different missions. Since its invention the project has grown into an international collaboration project of over 40 universities, high schools, and private firms. Under the coordination of the University of California at San Luis Obispo, 14 CubeSats were successfully launched using the standardize Poly-PicoSatellite Orbital Deployer (P-POD) on March 2005 using the Dnepr launch vehicle [1]. Another international collaboration project, called QB50, to put 50 CubeSats for multi-point, in-situ measurements in the lower thermosphere is also in development with a target launch date in 2014 [2].

Many students at San Jose State University (SJSU) had the opportunity to work on CubeSat in 2009 with Bob Twiggs, who was one of the pioneer creators of the CubeSat project. With appropriate funding, the aerospace engineering undergraduate students have been building the several versions of the CubeSat for their senior projects. However, because of the limited time available for the students to work on the project, most of the CubeSats are built with numerical modeling or experimental testing because the satellite is assumed to function as designed. The objective of this thesis is to perform numerical thermal modeling of a general CubeSat using different commercial applications. Thermal Desktop was used to model the external orbital heating rate. The result obtained from this analysis was then used as the wall boundary condition for the internal thermal analysis in Ansys Icepak.

There are three main operating phases concerning the thermal environment of the CubeSat are: (1) launch, (2) mission lifetime, and (3) reentry self-destruction. The purpose of this thesis is to perform a detailed analysis of the CubeSat in phase 2, the mission lifetime in orbit. Typically the CubeSats are deployed on a standard flight-proven Poly Picosatellite Orbital Deployer (P-POD) deployment system from CalPoly without further analysis except for qualitative tests to ensure mission safety. The assumption is that the CubeSat is safe inside the P-POD for transport and launch.

Without proper thermal protection system for reentry, the CubeSats are designed to be burned up in the atmosphere during reentry. Alternatively the CubeSat can be picked up by a larger satellite. Therefore phase 1 and phase 3 are not considered in this thermal analysis. It is the goal of this thesis to make sure the system is operating within safety limit by performing thermal simulation of the CubeSat to study the internal heat conduction and the external heat radiation.

A typical CubeSat consists of an anodized aluminum structure with space inside for mounting the hardware. Figure 1-1 shows an example of the 3-D CAD model of a CubeSat. Depending on the mission objective, there are usually several levels of embedded hardware with different power consumption. Details information on CubeSat subsystems are further discussed in Chapter 3.

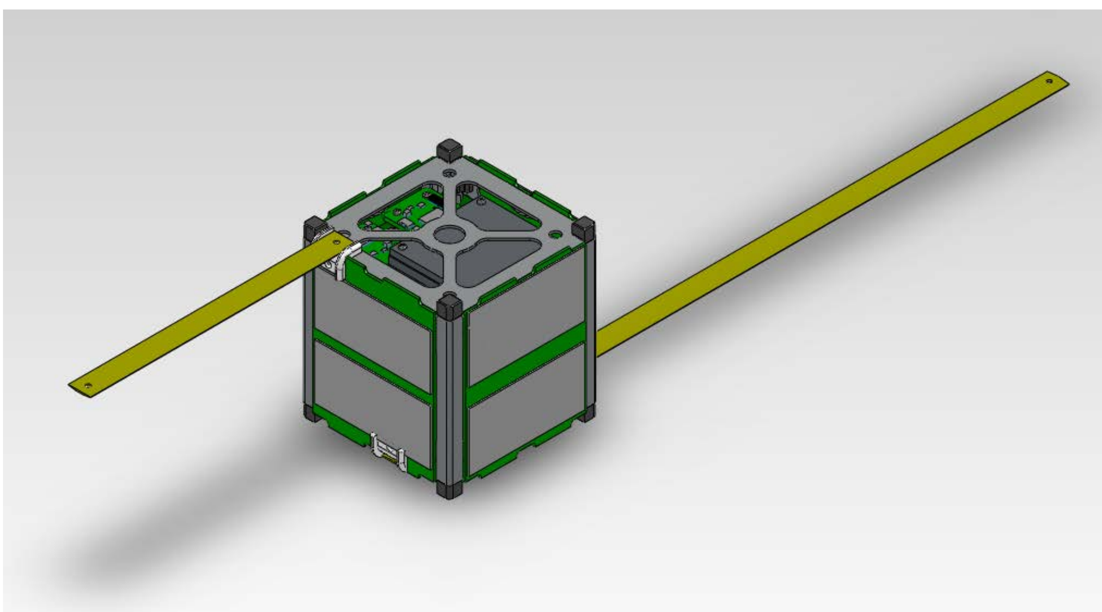


Figure 1-1 CubeSat Model

1.2 Motivations

There is currently a group of undergraduate and graduate students in aerospace engineering at San Jose State University collaborating to analyze, design and build a CubeSat with the purpose of testing communication and tracking modules as a technology demonstration. Although the design and assembly of the satellite has been completed, detailed analysis has not yet been performed. This creates the opportunity for graduate students to perform detailed analysis of the various aspect of the design such as structural and vibrational analysis, power system design and thermal analysis, etc. The motivation for this thesis is to perform detailed system-level thermal analysis of the CubeSat in Low Earth Orbit (LEO) at 400 km altitude to verify numerically that all electrical components are operating within the thermal design limits. This thesis also explored different design configurations to make sure the satellite will be able to operate safely in its environment.

1.3 Objectives

With the opportunity to build flight hardware with real access to space as a secondary payload, many educational institutions have been building the CubeSat with different mission objectives. The mission can range from measuring the temperature in the thermosphere to experimenting and validating different communication systems for future nano- and pico-satellite missions.

The thermal subsystem of a satellite depends on various parameters such as the geometry, structure, and onboard electronic components. For small scale satellite like the cubesat, the thermal control option is limited and often resorted to passive devices. To ensure components operability and mission success, a thermal analysis, either through numerical or experimental methods, is necessary to benchmark against components temperature limits. If the operating temperatures exceed the specified limits, different approaches are investigated to achieve proper thermal control.

The objective of this thesis is to perform thermal simulation of the CubeSat using Thermal Desktop and ANSYS Icepak to make sure internal electronics are operating within the various specified safety limits. The goal is to demonstrate numerical simulations can be used to predict and optimize the temperature distribution of the space system before actually building it. Electronics operating temperature typically range from -25^{o} to 85^{o}C, solar panels range from -85 to 100^{o}C, and batteries range from -40^{o} to 60^{o}C [3]. At any given time the CubeSat will have at most three faces facing the sun, while the other faces are in the shadow facing and absorbing the earth's albedo. If the electronics are not operating within safety range during flight operation, either active or passive thermal control, such as heater, isolator, heat pipes, or louvers, will be used. Alternatively the CubeSat can be set into hibernation until it gets to certain orbit position that would allow all the components to operate at a safe temperature. The analysis will include both internal heat transfer condition and external radiation analysis. At 400 km mission altitude and with the beta angle of 56 degrees, the CubeSat will spend about 65 minutes exposing to the sun, and about 35 minutes in the eclipse region for each orbital period. The steady state analyses are divided into two cases: (1) "hot case" for facing the sun and (2) "cold case" for the satellite during eclipse.

Chapter 2
Theoretical Background

2.1 Heat Transfer

When the electrical current flows through the electronic devices to power the system, a portion of the power generates heat as a by-product of normal operation. Sometimes the excess in the heat generated can damage the device, which is why proper thermal management and thermal analysis are necessary to ensure device operability and reliability. Heat is typically transfers in three ways: (1) convection, (2) conduction, and (3) radiation. However, since the density of air at very high altitudes is extremely low from the Low Earth Orbit (LEO) and beyond, the available air for natural cooling is negligible. Space heat transfer is then predominately control by conduction and radiation. Without natural convection to cool off the electronics, a system operating at the same power input in space is more likely to result in higher temperature compare to that operating at sea level.

For space application, the heat on the external of the spacecraft mainly comes from direct solar radiation from the sun. In general, if the CubeSat is tilted relatively to the sun, the solar flux can "see" at most three faces. The worst case scenario, in term of heat transfer, is when one side of the CubeSat faces directly to the sun and all the energy is either absorbed or reflected. This causes one side to be too hot, while the remaining 5 sides remain cool. If the thermal analysis for future mission shows that the concentrated heat on one side is too high, passive thermal control solution, such as heat pipes, can be used to redirect the heat from the hot side to the cold side. In addition to conduction and radiation, the CubeSat spins as it orbits the earth. This spinning effect, according to Gadalla [4], reduces the overall temperature on the external body. The primary goal of this thesis is to analyze the CubeSat thermal condition to make sure mission success, and to explore different design methods to keep the temperatures of the electronics with the specified operating temperature. A well design system would incorporate passive or active cooling mechanism, such as using thermal electric cooler (TEC) or heat pipes, to distribute the heat. The purposes of this thesis are to analyze thermal conditions of the CubeSat in space and explore possibilities of improvement for future missions. This section provides basic theoretical background to heat transfer and space related information relevant to the thermal analysis of the CubeSat. It will explain basic equations for conduction and radiation mode of heat transfer from the Fundamentals of Heat and Mass Transfer book by Incropera et el [5]. The explanation for space environment and orbital mechanics comes from the Space Mission Analysis and Design handbook by Larson et el [6, 7].

2.1.1 Conduction Heat Transfer

Conduction is the process by which heat is transferred through the solid, liquid, or gas from a high energy source to a relatively lower energy source. Conduction heat transfer can occur within a material, or from two or more contacting bodies. It is governed by Fourier's Law as follow:

$$q_x'' = -k\frac{dT}{dx} \tag{2.1}$$

where Q is the heat flow rate (W), k is the thermal conductivity (W/mK) of the material, dT/dx is the temperature differential over the length. Unlike convective heat transfer mode, the conduction resistance inside the solid does not change with altitude.

The thermal resistance is a measure of the resistance of heat flow between contacting bodies through the thermal interface material, such as thermal pads or adhesive bonding materials. Thermal resistance, often expressed as °C/W or K/W, is found by

$$\theta_{cond} = \frac{\Delta T}{q_x} = \frac{L}{lA_C} \tag{2.2}$$

where A_C is the cross-sectional area available for conduction. One can quickly calculate the expected operating temperature of the device if the ambient temperature, the thermal resistance and the power consumption are known. For example, if the thermal resistance of a die package is 2.5 °C/W, the ambient temperature is 25°C and the chip consumes 2 W of power, then the expected temperature of the device is 30°C.

2.1.2 Radiation Heat Transfer

Radiation heat transfer occurs by radiating the heat between two or more surfaces through space by electromagnetic waves. It is dependent on the temperature and the coating of the radiating surface. The radiation heat transfer is governed by Stefan-Boltzmann's Law as follow:

$$E = \sigma T^4 \tag{2.3}$$

where σ is the Stefan-Boltzmann constant, 5.67×10^{-8} W/m^{2}K^4.

The amount of energy transfer through radiation between two bodies having temperature of T1 and T2 is found by

$$q_r = \varepsilon\sigma F_{1,2} A(T_1^4 - T_2^4) \quad (2.4)$$

where:

q_r = amount of heat transfer by radiation (W)

ε = emissivity of the radiation surface (reflective = 0, absorptive = 1)

$F_{1,2}$ shape factor between surface are of body 1 and body 2 (≤ 1.0)

Unless the difference temperature between two or more bodies is high, radiation heat transfer is often very low. For example, the radiation energy between the sun and the spacecraft is taken into consideration for the external analysis of the CubeSat. However, the radiation between the internal electronics can be neglected because the maximum heat dissipation is only a fraction of 1 W of power.

2.2 Space Thermal Environment

The performance and operational lifetime of space systems are strongly influenced by the near-Earth space and atmospheric environments. For the CubeSat launching from the International Space Station (ISS) at approximately 400 km altitude, the atmospheric pressure and drag is very small, and hence aerodynamic heating and convective heat transfer is negligible. There are three main sources of heat for general spacecraft systems operating in the near-Earth environment, which includes the radiated heat from the sun, the albedo (the reflection of solar radiation) and planetary heating from the Earth (black-body radiation of the Earth) [7, 8]. This section provides basic information about the space environment and how it relates to the thermal simulation of the CubeSat.

Solar activity varies daily and its variation between the solar maxima and minima often dictates the design of the spacecraft. Most CubeSats, however, are designed to be in orbit for only a short period of time, and hence, the average solar flux over an extended period of time is sufficient for the radiation analysis. The solar flux used for this analysis is 0.873547 W/in^2 (1354 W/m^2). Another source of radiation is the reflected sun-light off the earth surface, also called albedo. The average albedo of the earth is about 0.3. Table 2-1 [7] provides a summary of the range of Direct Solar, Reflected Solar (Albedo), and Planetary Infrared for the planet Earth. The thermal simulations are performed on two worse case scenarios: (1) “Hot Case” during direct sunlight and (2) “Cold Case” during eclipse. The values used for the “Hot Case” simulation are approximately the values in the “Mean” column, and the values used for the “Cold Case” are in the “Eclipse” column. From Table 2-1 it can be seen that the solar intensity at approximately 1 AU from the sun is about 1371 W/m^2. The fraction of the reflected solar radiation, also referred to as albedo, is about 0.3.

Table 2-1 LEO Thermal Environment (Adapted from [5], [7] and [9])

	Perihelion	**Aphelion**	**Mean (Hot Case)**	**Eclipse (Cold Case)**
Direct Solar	1414 W/m^2	1323 W/m^2	1371 W/m^2	0
Albedo (average)	0.30+/-0.01	0.30+/-0.01	0.30+/-0.01	0.25
Planetary IR (average)	234 +/-7 W/m^2	234+/-7 W/m^2	234 +/7 W/m^2	220 W/m^2

An important parameter to consider when performing thermal analysis for the spacecraft and satellite is that space vehicle often spins as it orbits the earth. This spinning effect causes the vehicle to receive heat on one side and dissipated heat through the remaining sides. The combine effect of a rotating spacecraft and space solar heating causes the temperature variation on the external body, depending on the orbit and position of spacecraft relative to the sun [10]. The space thermal environment for general spacecraft in the earth orbit is shown in Figure 2-1 below.

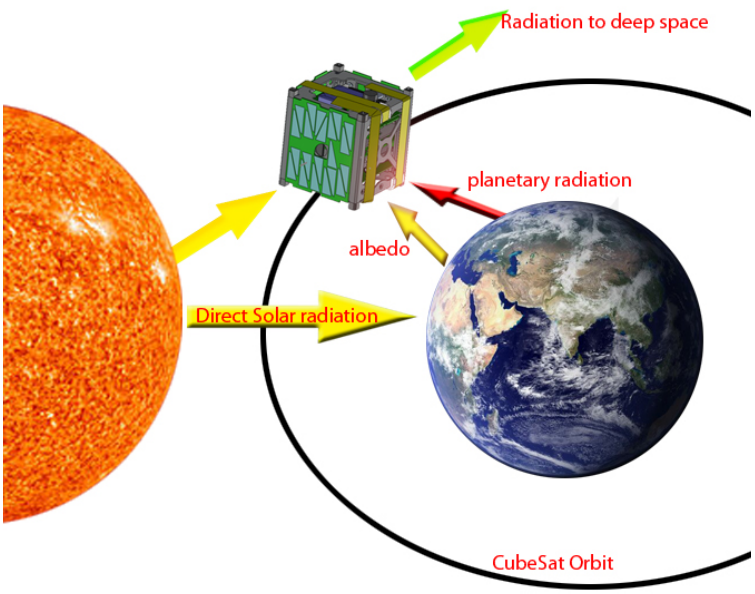


Figure 2-1 Space thermal environment for CubeSat

From a generalized heat balance equation for conservation of energy:

$$Q_{in} = Q_{out} \tag{2.5}$$

$$Q_{external} + Q_{internal} = Q_{radiated} \tag{2.6}$$

Where $Q_{external}$ is the environmental heat absorbed, $Q_{internal}$ is the power dissipation by the internal electronics, and $Q_{radiated}$ is the heat rejected from the spacecraft to deep space.

Assuming the satellite is a black sphere, which means the absorptivity equals the reflectivity ($\alpha = \varepsilon = 1$), and that the satellite is in direct sunlight with the effect of planetary heating and albedo, the satellite energy balance for the hot case calculation can be shown as follows

$$\boldsymbol{A_{solar}q_s + FA_{surface}\sigma\overline{T}_E^4 + FA_{surface}aq_s + Q = A_{surface}\sigma\overline{T}^4} \tag{2.7}$$

where:

$A_{surface} = 4\pi r^2$, $A_{solar} = A_{albedo} = A_{planetary} = \pi r^2$,

F is the view factor (F = ½ for a spherical spacecraft assumption, and F = 1/6 for one side of the CubeSat (cube shape)), a = 0.3 is the earth's albedo,

Q = 0 for internal heating.

Solving for the average surface temperature $\overline{T}$ of the CubeSat, Equation (4.5) becomes

$$\overline{T}^4 = \frac{q_s}{\sigma}\left(\frac{1}{4} + Fa\right) + F\overline{T}_E^4 = \frac{1371}{5.67 * 10^{-8}}\left(\frac{1}{4} + 0.15 * 0.3\right) + 0.15 * 255^4$$

For $q_s = 1371\ W/m^2$, $\sigma = 5.67\text{x}10^{-8}$ W/m^{2}K^4, F = 0.15, a = 0.3, and $\overline{T}_E^4 = 255K$, the CubeSat equilibrium temperature is about 297 K or 24 °C. Similarly, when the satellite is in eclipsed, the term $A_{solar}q_s$ in Equation (2.7) becomes zero and the cold case calculation for the average surface temperature is about 198 K or -75 °C.

2.3 Orbital Mechanics

This section provides fundamental theory dealing with orbital mechanics to help aid the understanding of orbital heating simulation study. First, an important variable for this analysis is the beta angle, which determines the amount of time the spacecraft exposing to direct sunlight. Beta angle is defined as the angle between the orbit plane and the vector from the sun of any Earth-orbiting object [8], see Figure 2-2 below. The beta angle varies between +90° and -90°, depending on the direction of the spacecraft. Viewing from the Sun, the beta angle is positive if the satellite is revolving counter clockwise and negative if revolving clockwise. The satellite is exposed to more sunlight per orbit as beta angle is increased, and eventually reach constant sunlight exposure when beta angle is at 90 degrees. At high beta angle the satellite might get overheated if it is not properly controlled. As will be shown later in the Results and Discussion section for the analysis of the CubeSat for two beta angle, at 30 and 90 degrees, the heating rate for beta angle equals 90 degrees is higher compare to the heating rate for beta angle equals 30 degrees. On the other hand, the spacecraft would spend a shorter amount of time in sunlight and a longer amount of time in eclipse when the beta angle is small. The spacecraft might need heating to keep the electronic components within the various operating ranges. For small satellite that has a low power budget, such as the CubeSat, the power subsystem may not carry enough power to keep all the components warm. It is therefore best to keep the CubeSat's beta angle at the range where it will receive a sufficient amount of time in sunlight and eclipse.

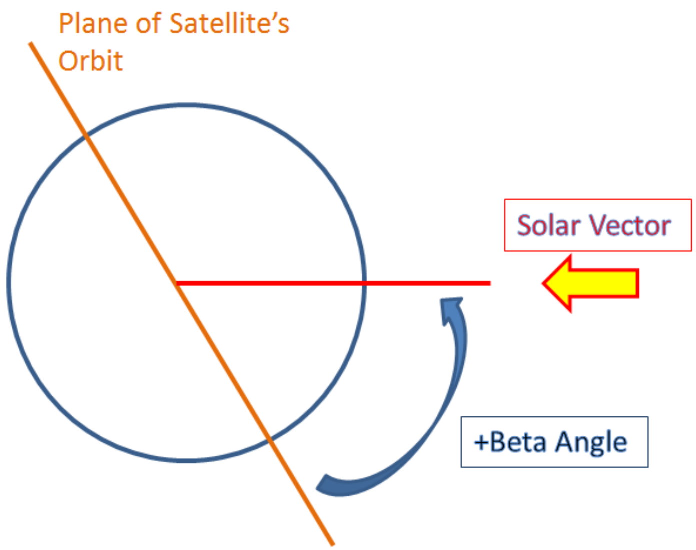


Figure 2-2 Beta Angle

Another important parameter for simulating the heating condition is the mission altitude. The orbital heating rate simulation is simplified by assuming that the CubeSat travels in a circular orbit at 400 km altitude above the earth. Using the standard values for earth gravitational constant and radius, the orbit's semi-major axis a, orbital velocity v and period *P* are calculated as follow:

$$\mu_{Earth} = 3.986 * 10^5 \frac{km^3}{s^2} \tag{2.8}$$

$$R_{Earth} = 6378\ km \tag{2.9}$$

$$a_{mission} = R_{Earth} + h_{mission} = 6378 + 400 = 6778km \tag{2.10}$$

$$\boldsymbol{v} = \sqrt{\frac{\boldsymbol{\mu_{Earth}}}{\boldsymbol{a_{mission}}}} = \sqrt{\frac{\mathbf{3.986*10^5}}{\mathbf{6778}}} = \mathbf{7.67\ km/s} \tag{2.11}$$

$$P = 2\pi\sqrt{\frac{a_{misison}^3}{\mu_{Earth}}} = 2\pi\sqrt{\frac{6778^3}{3.986*10^5}} = 5535\ seconds \tag{2.12}$$

The number of orbits the satellite travels in one day, or the mean motion (n) of the satellite is calculated as:

$$\boldsymbol{n = \frac{86400}{P} = \frac{86400}{5431.01} = 15.9\ revs/day} \tag{2.13}$$

Using the above equations, general orbital parameters various orbital heights are calculated and plotted in Figure 2-3 below. As can be seen from the figure, the orbital velocity varies from 7 to 8 km/s and the mean motion varies from 12 to 17 orbits per day for a general satellite in LEO.

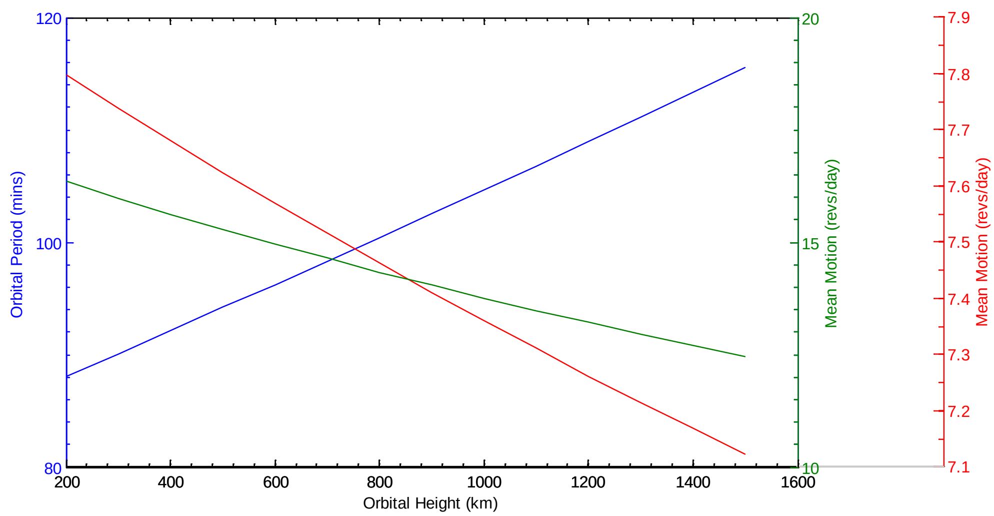


Figure 2-3 Orbital Parameters for Low Earth Orbit

The next step is to verify analytical results with the result from Thermal Desktop. The calculated period in the equation above is about the same as the result shown in

Figure 2-4. It should be noted that the calculation for orbital period depends only on the mission altitude and not the beta angle.

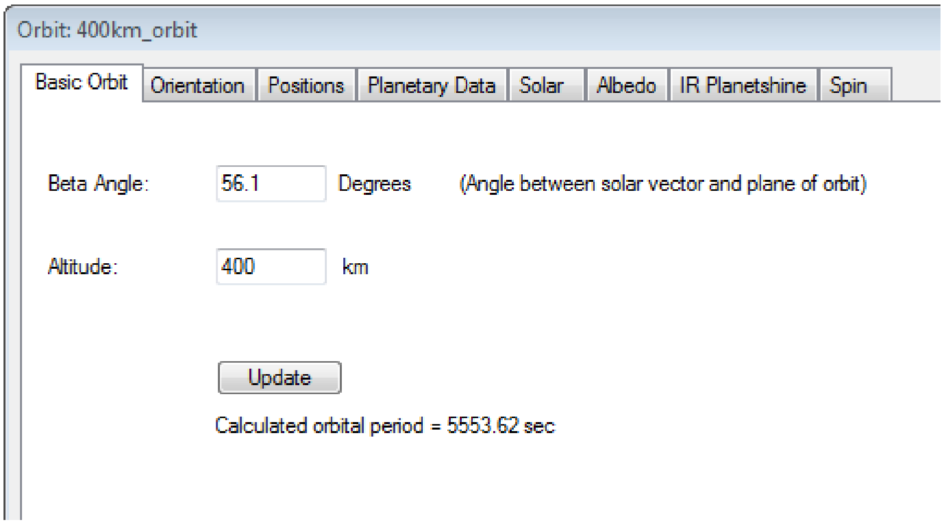


Figure 2-4 Thermal Desktop Orbit Input Options

Next, the Earth's angular radius at mission altitude is given by:

$$\rho = \sin^{-1}\left(\frac{R_{Earth}}{R_{Earth}+h_{mission}}\right) = \sin^{-1}\left(\frac{6378}{6378+400}\right) = 70.2\ deg. \qquad (2.14)$$

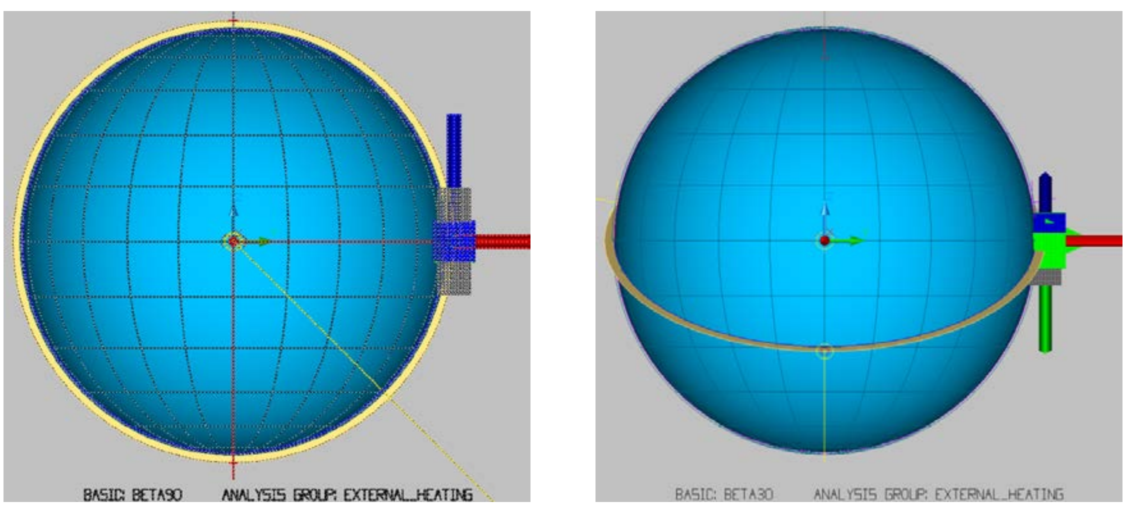
The maximum time of eclipse (TE) and the time in sunlight (TS) are given by:

$$TE = \frac{2\rho}{360^{o}}P = \frac{2*70.2^{o}}{360^{o}}5553.6 = 2165.9\ seconds = 36.1\ minutes \qquad (2.15)$$

$$TS = P - TE = 5553.6 - 2165.9 = 3387.7\ seconds = 56.5\ minutes \qquad (2.16)$$

The time of eclipse and time in sunlight helps determine the expected thermal conditions of the spacecraft during these time intervals. For comparison, the orbital heating rate simulation was performed for three beta angles: (1) beta = 56.1^{o}, (1) beta =

90° and (2) beta = 30°. The simulation for beta angle of 56.1 degrees was intended for the main mission in which further analysis were performed in subsequent sections. The simulation for beta angle of 90 and 30 degrees were provided for comparison purpose. Figure 2-5a shows the CubeSat in circular orbit for beta angle of 90 degrees. It can be seen that the CubeSat is always in direct sunlight at this beta angle, which means that the heat flux is expected to be constant. For numerical simulation without the spinning effect, the result in later section should show that the side directly facing the sun would receive a significantly higher heat flux compare to the other sides that were in eclipse. Figure 2-5b shows the circular orbit of the CubeSat for beta angle equals 30 degrees. Both vehicle orientations are viewed from the sun perspective. As will be shown later in the Results and Discussions section, the heat flux in this case is not constant because the spacecraft will spend portion of its period in eclipse.



(a) (b)

Figure 2-5 (a) CubeSat Orbit for beta angle = 90 deg. (b) beta angle = 30 deg.

Another important parameter for mission analysis is the rate of decay, which is the rate at which the satellite de-orbits from the initial height before falling back and burn up in the atmosphere. This helps determine the satellite lifetime and whether or not the satellite can operate for the duration of the mission requirement timeframe. The rate of decay is influenced by the atmospheric drag and gravitational forces. The value of 2.2 has widely been used for the drag coefficient for spacecraft operating in LEO, although it is expected that this value tend to increase as the altitude decreases [9]**.** The Satellite ToolKit (STK) application from AGI was used for the mission analysis of the CubeSat. Using 400 km for altitude, 56.1 degrees for beta angle and 2.2 for drag coefficient as the input parameters, STK calculated that the satellite have about 7 months mission lifetime, as shown in Figure 2-6.

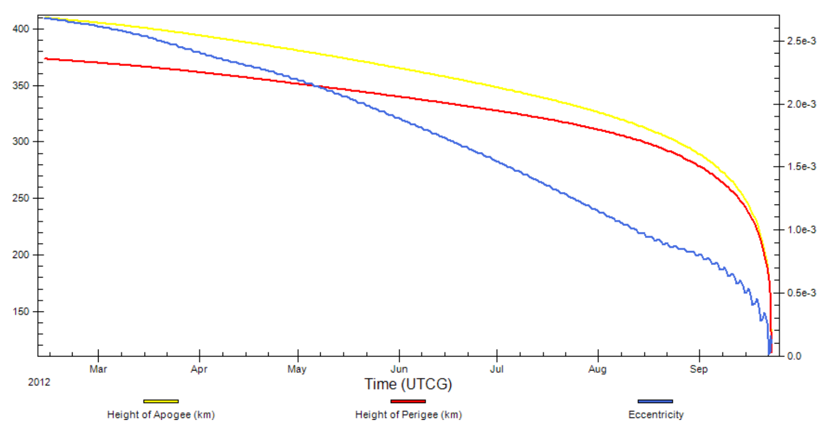


Figure 2-6 CubeSat Lifetime – Apogee, Perigee and Eccentricity Variation

Figure 2-6 shows the apogee, perigee, and eccentricity variation for the 400 km circular orbit satellite from the STK simulation. It can be seen that the rate of decay is slow initially but quickly reach an abrupt end once the altitude of the satellite falls below

200 km. STK also have the capability of analyzing the heat transfer on the satellite but the appropriate license was not available for the analysis. An alternative approach, provided in the next section, was used in this thesis.

Chapter 3

CubeSat System Design and Power Budget Analysis

3.1 Subsystems Overview

Since the specifications for CubeSat standard are fairly limited in size and weight and that most CubeSat are only in space for a short duration due to budgetary cost and other constraints, this type of satellite utilized mostly passive control systems with relatively low power usage. The main subsystems of a typical CubeSat usually consisted of (1) passive attitude control using the hysteresis antenna, (2) Pumpkin CubeSat Kit for structure, (3) radio communication system, (4) battery and solar cells for power and (4) custom electrical hardware for power distribution, data handling, communication and control. An exploded view of the 3-D CAD model is shown in Figure 3-1 (Simplified version of TechEdSat model).

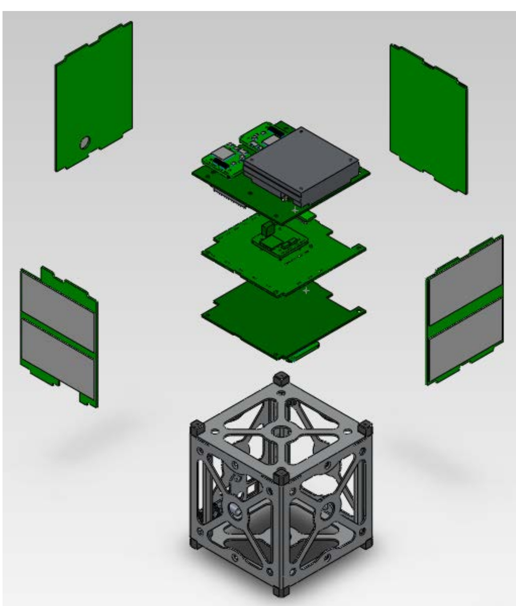


Figure 3-1 Example of a CubeSat Model – Exploded View

3.2 Attitude Control Subsystem

Upon ejection from the P-POD, the CubeSat is expected to be rotating and spinning arbitrarily in its designated orbit of around 400 km attitude. A passive attitude control subsystem using the hysteresis material and permanent magnet are used to dampen the potential high rational spin rate. The basic idea is that a passively control satellite will be aligned with the Earth's magnetic field through the use of a magnet installed on the satellite.

3.3 Structural Subsystem

Most CubeSat utilized a commercial-off-the-self (COTS) 6061 Aluminum frame structure from the Pumpkin CubeSat Kit. The advantages of using this structural kit are that (1) it adhered to the CubeSat Standards, (2) can easily be fitted into the standard Poly-PicoSatellite Orbital Deployer (P-POD) and (3) has been used successfully on other CubeSat project from different university. With its proven success in withstanding the strong vibration environment during launch, using this kit allows the CubeSat designers to focus on other design aspects. Although the external interface is restricted by the CubeSat Standards, the internal configuration can vary depending on the payloads and mission objectives. As with any space vehicle, stress and vibration test are still necessary to quantify the structural integrity during transport and launch to ensure that the integrated components will be able to withstand the expected load and vibration, it's beyond the scope of this this thesis.

3.4 Communication Subsystems

Typically CubeSat designers have been using the amateur radio frequency for communication. With the objective of testing for the best communication systems for general purpose pico-satellite, the CubeSat has a total of three communication systems; a HAM radio beacon, OrbComm Q1000 and Iridium 9602 modem.

The OrbComm Q1000 modem is a satellite-based communication system provided by the Orbcomm Communication Network. It has the operating temperature of -40 °C to +85 °C. This system uses 150 MHz for uplink and 138 MHz for downlink. Since this is a COTS product and the design parameters related to the thermal simulation is not known, the author decided to exclude this component from the thermal simulation. Through analytical and numerical solutions, it is assumed that if the average temperature inside the CubeSat is within the specified temperature limits of the component, then the component is assumed to be operable. An actual experiment is needed to determine the operating temperature limits based on power input.

3.5 Power and Electrical Subsystems (EPS)

As an overall passive system, the CubeSat uses very little power. The electricity mainly comes from an onboard battery pack and is recharged by using several arrays of high density solar cells installed on the external surface of the satellite. The electrical power is controlled and regulated by the main power distribution unit (MPDU). The MPDU has a dedicated microcontroller to control the on and off states of the onboard components.

Table 3-1 shows a typical power budget of the Cubesat by estimating the approximate power consumption for the various subsystems for both active and passive state from different sources [6, 7]. The active state describes the satellite operating in the sun with most of the onboard subsystems turned on for communication and data handling. The standby state describes the operating state during eclipse, where only crucial components are powered.

Table 3-1 Power Budget for CubeSat

Component	**Power (W)**		**Operating Limit**
	Active	**Standby**	
Solar Panels	N/A	N/A	-150°C - 110°C
Battery	N/A	N/A	-10°C - 60°C
Chip 4	1.5	1.3	110°C Max.
Chip 1, 2 and 3	0.15	0.02	110°C Max.
Quake Q1000 (Active cycle = 0.5 second)	24	0.84	-40°C - 85°C
Iridum 9602 (active cycle = 0.5 second)	7.5	0.975	-40°C - 85°C

Chapter 4

Thermal Modeling Approach

4.1 External Analysis Approach

The radiation heating in the space environment can be either helpful or harmful to the CubeSat or spacecraft in general [7]. In some cases, the Sun's thermal energy and the Earth's albedo can help to warm electronic components to its normal operating temperature of about 20 ^{o}C. In other cases, the heat generated from onboard computing combined with solar energy may cause overheating issue. Initial attempt to simultaneously simulate the external radiation and internal heat conduction was not successful because of the temporary limited software license available. The problem was decoupled into two cases: (1) internal heat conduction and radiation of the electronics using Icepak and (2) external heat radiation using Thermal Desktop. A combination of SolidWorks and AutoCAD were used to generate the computer model of the CubeSat. This decoupled analysis approach assumed that the external heating does not cause any change to the internal environment and vice-versa. The assumption would be valid since the radiation effect is only significant when there is a high temperature differential between different objects. Otherwise it would cause about 5 to 10% error if the heat conduction from the external surface to internal surface of the case is significant.

Generally the outer surface area of a CubeSat is usually covered on four sides with solar cells in order to generate as much power as possible. Since the solar cells behave as a flat absorber, designing it on the sides reduces the heat load on the surface of the case that does not face the sun. As mentioned in the previous chapter, the CubeSat spins as it orbits the Earth, which creates a relieving effect that lower the overal temperature as the surface is continually absorbing heat from the sun and radiating heat to deep space. This is difficult to solve numerically in the steady state analysis. An alternative approach used in this thesis was to calculate the heat flux on each side individually and compute the average of the heat flux that can be uniformly applied to all sides.

Another point to note is that solar cells can be tricky to model thermally since portion of the absorbed energy is converted to electricity and not heat [11]. For example, a solar cells may convert 15% of the incident energy into electricity and another 40% into heat, leaving 45% of the energy reflected to nearby surfaces. If the value of absorptivity of 40% is used, then the amount of energy absorbed as heat is correct, but the reflected energy has an extra 15% of the incident energy. The solution to this is just ignored the 15% of the energy converted to electricity, and used 40% absorbeb and 45% reflected energy. Table 4-1 provides the optical properties used in the analysis.

Table 4-1 Optical Properties

Description/Composition	$\boldsymbol{\alpha_s}$	$\boldsymbol{\varepsilon^b}$	$\boldsymbol{\alpha_s/\varepsilon^b}$
Anodized Aluminum	0.14	0.84	0.17
Solar Cells	0.40	0.45	0.89
Tedlar White	0.39	0.87	0.448

4.2 Internal Analysis Approach

Ansys Icepak was used to perform the internal thermal simulation. The internal simulation focused on the main heat dissipation of each electrical component. The analyses were performed on the component level as well as system level simulation. For example, the BeagleBoard shown in Figure 4-1 that is typically used to build the CubeSat was performed separately on the component level. The system level simulation is discussed in details in the next chapter.

The BeagleBoard has the main heat source coming from the Texas Instrument's OMAP3530 720 MHz chip. All the connection interfaces, such as S-Video, Stereo In/Out, etc., were excluded from the analysis because the heats generated from these components are very small compare to the main chip. The BeagleBoard is designed for low powered application, and hence, does not require a heat sink. The PCB has the overall dimension of 76.2x78.7x1.6mm [1]. It is made up of multi-layered dielectric materials, such as FR4, and several layers of copper planes. From the BeagleBoard documentation [12], this board has 6 internal layers and the standard trace material of Cu-Pure was assumed for the simulation. The PCB board from BeagleBoard is a single overall layer board in which components are mounted on one side and the copper patters are mounted and soldered on the opposite side. From the thermal modeling point of view, however, the PCB was treated as a homogeneous material with an orthotropic conductivity related to the copper layers in the substrate. The thermal physical properties used in this steady state heat conduction are provided in Table 5.1 below.

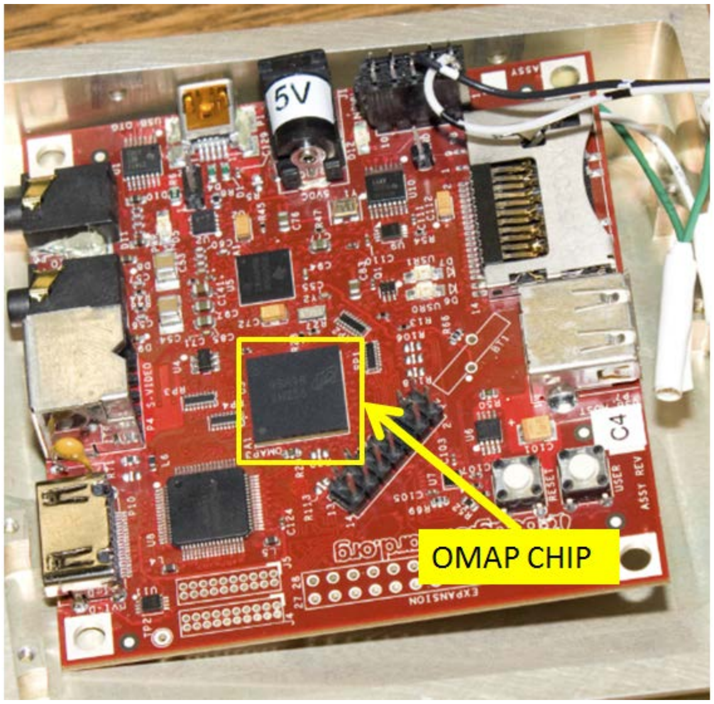


Figure 4-1 BeagleBoard top side components

Table 4-2 Thermal Physical Properties

Description/Composition	Cond. [W/in/K]	Density [kg/in^3]	Cp [J/kg/K]
Aluminum (Alloy 2024-T6)	4.489*	0.04539	873.372*
FR4 2 oz Copper	0.44958	0	0
OMAP Chip	0	0.0327741	837.32

Note: * indicates property is temperature dependent.

The steady state heat conduction simulation assumed a constant heat load of 1.2 W applied on the OMAP chip. Figure 4-2 shows that 1.2 W heat load would produce a temperature of approximately 310K (37 ^{o}C), which is within the safety limit of the electronic. This is expected because the BeagleBoard is designed for low powered applications. However, printed circuit boards generally endure high heat load and it is difficult to provide heat dissipation capability for components such as voltage regulators. A solution to this is to use certain parts of the case, such as the back face, for heat dissipation. A high conductive material can be used to connect the hot component to the back face to reduce the heat load.

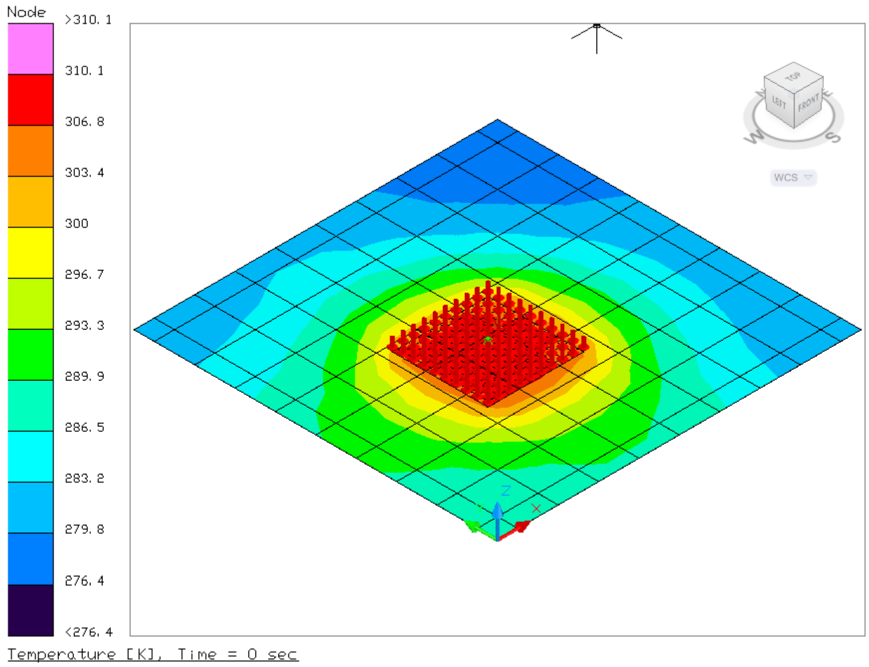


Figure 4-2 BeagleBoard Heat Conduction

An experiment was conducted to validate the validity of the numerical solution. The experiment consisted of applying a specified heat load to a copper block to simulate the heating condition of the microcontroller under full load. A temperature profile of a 1.2 W applied to a 35 mm^2 copper block is shown in Figure 4-3. Two thermocouples were directly attached to the surface of the copper block to measure the temperatures profile, denoted as T_source1 and T_source2 in Figure 4-3. After about 40 minutes, the experimental model reached a maximum temperature of about 39°C. Comparing to the simulated model in Figure 4-2, the temperature difference between the two models is about 2^{o}C. This experiment showed that numerical method can be used for preliminary thermal design and analysis with reasonable accuracy.

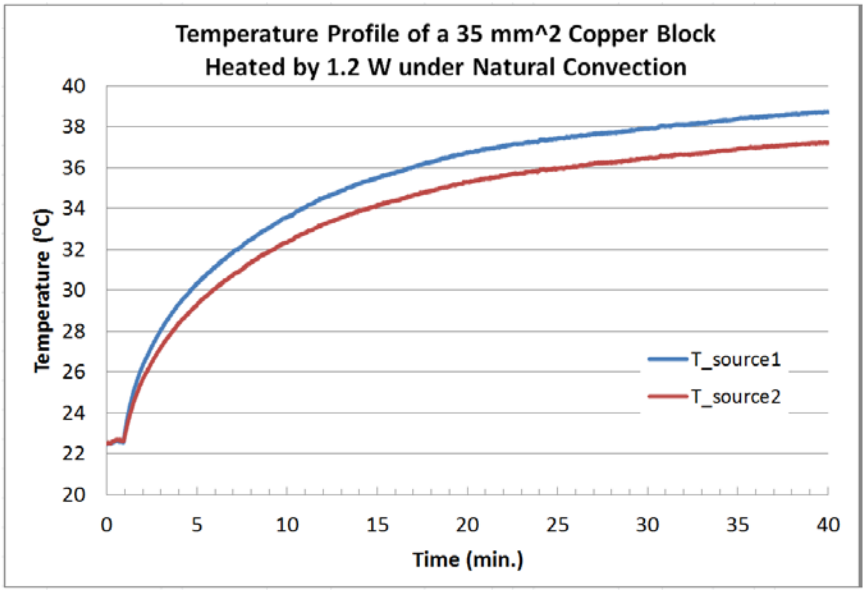


Figure 4-3 1.2 W power dissipation through CU block under natural convection

Chapter 5

Thermal Modeling Results and Discussion

5.1 External Analysis with Thermal Desktop

This section explains the orbital heating result for the external surfaces using Thermal Desktop. Figure 5-1 shows the setup of a CubeSat at 400 km circular orbit at 56.1^{o} inclination angle. The external faces of the CubeSat were modeled as surfaces covered with solar panels. This simplification approach greatly reduced the complexity of the thermal model. The internal heat dissipation, as mentioned previously, was not taken into consideration for this external analysis. The thermo-optical properties of solar cells were selected based on the assumption that part of the incident energy is converted into electricity, part of it is reflected off the surface, and the rest is heating up the components. A more detailed setup is provided in Appendix B.

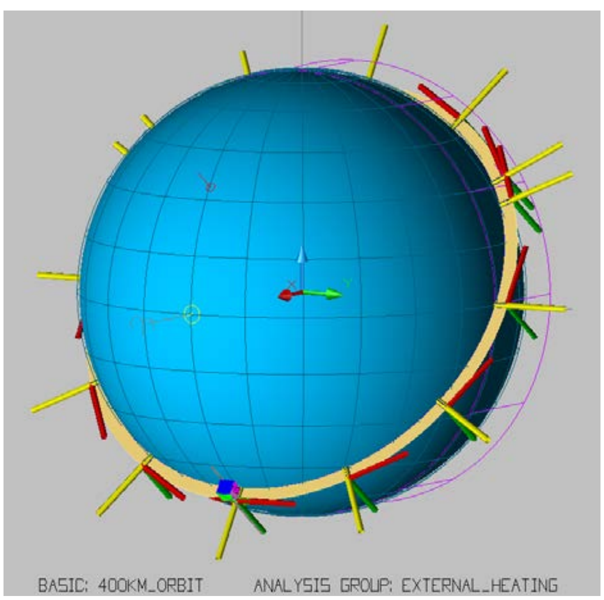


Figure 5-1 External heating at 400 km altitude

Figure 5-2 shows the external total absorbed heat flux profile of the CubeSat. As can be seen from the temperature profile, the heat flux on the side facing the sun was about twice the heat flux on the surrounding sides. Figure 5-3 shows the temperature variation on the sides by probing monitoring points about the center of the front and right side faces. This plot shows that the heat flux on the sides of the CubeSat is about 100 W/m^2 during its period in the sun, and the heat flux is about 60 W/m^2 during its period in eclipse. It should be noted that the time periods during sunlight and eclipse, as shown in Figure 5-3, was congruent with the calculations in Chapter 2. Although the hysteresis antenna was included in the thermal model, it has negligible effect on the overall heat flux and can often be omitted.

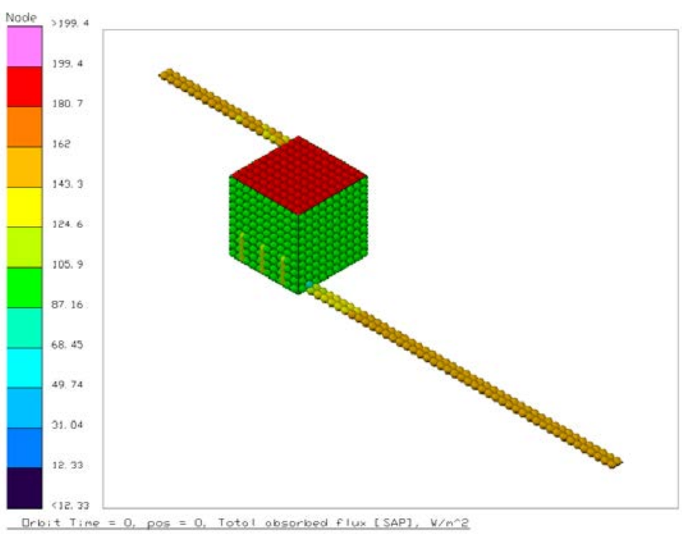


Figure 5-2 External heat flux

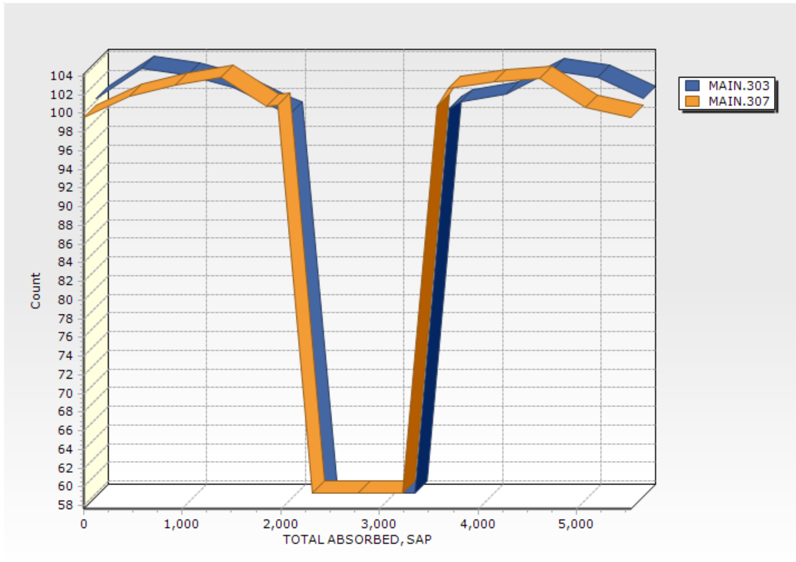


Figure 5-3 Total Absorbed Heat Flux for 400 km Altitude

5.2 Internal Simulation with Ansys Icepak

Ansys Icepak was used for modeling the internal components because the software has been well known for its ability to produce acceptable results in the electronics industry. The Icepak software uses the Ansys Fluent as the back-end solver to solve for the 3-D Navier-Stokes equations for the mass, momentum and energy using finite volume technique. The simplified internal components of the CubeSat were created using Ansys Icepak. The simulation model was built based on the measured dimensions from the 3-D model of the Pumpkin CubeSat kit. The computational domain, also called “Cabinet” in Icepak, was created using the internal diameter of the CubeSat kit model. The additional objects, such as the PCB board or the chips, were added to the approximate coordinates on the CubeSat 3-D assembly model with approximate thermal physical and optical properties that closely match the model. The vacuum space environment is simulated by turning off the “flow” option in Icepak, and a very low thermal conductivity value is assigned to the ambient computation domain. The convergence criterion was set at 1e-7 for the energy equation and an under-relaxation value of 0.7 was set for the momentum equation. The initial and boundary conditions were chosen to closely match the expected operating condition of the internal electronics. The model definition and mesh were passed to the Ansys FLUENT for computational fluid dynamic simulation, and the resulting data are post-processed using the ANSYS Icepak user interface.

The internal simulation was divided into two cases: (1) hot case and (2) cold case. The hot case is when the satellite is in direct sunlight, and the cold case is when the satellite is in eclipse. Figure 5-4 shows the temperature distribution of various components for the hot case analysis.

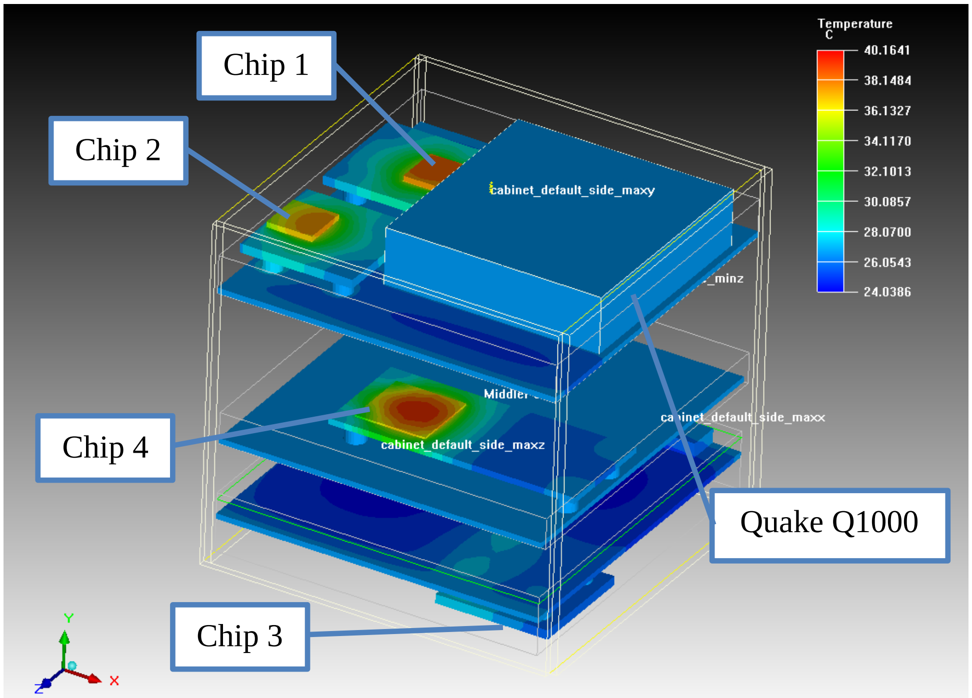


Figure 5-4 CubeSat Temperature Profile – Hot Case Analysis

Even though the same power of 0.15 W was applied to Chip 1 through 3, Figure 5-4 shows that the temperatures of the components varied slightly depending on the location of the chip relative to the other heating sources. The Quake Q1000 is a propriety commercial product that does not disclosed the power consumption of the internal components. Since this product has an operating temperature range from -40°C to +85°C and the CubeSat internal environment is within the range, it was assumed that the Quake Q1000 would be fully functional without further analysis. To study what effect it has on the other component, the overall system of the Quake Q1000 was modeled as a radiating block with the surface temperature assumed to be at 40°C. This caused the temperature on Chip 1 to be slightly higher than that of Chip 2. The temperature rise of the overall system is about 15°C.

Figure 5-5 shows the temperature profile of the internal components for the "cold case" simulation. With the initial operating temperature of -75°C, the temperature rise for this case is about 57°C, which is almost 4 times higher comparing to the "hot case". The variation of temperature between Chip 1 and Chip 2 is much higher than the "hot case". The simulation for this case represents a worse case analysis that might not accurately predict the temperature distribution of a real case. For example, as the CubeSat orbits from direct sunlight to eclipse area, the temperatures of the component has a temperature that are higher and slowly cooling down, which is not possible to model using the steady state method.

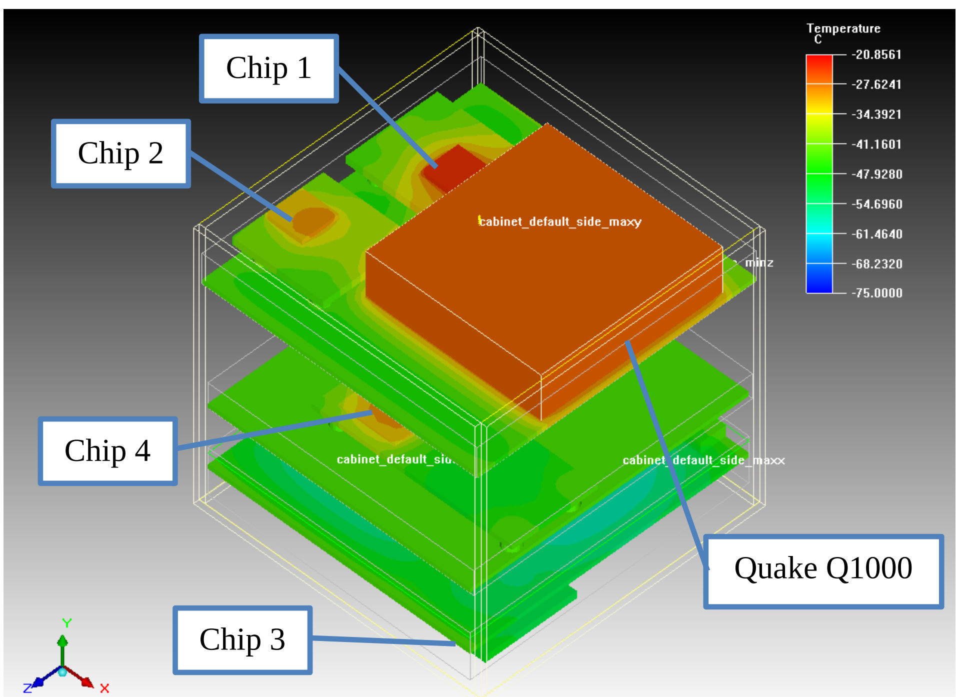


Figure 5-5 CubeSat Temperature Profile – Cold Case Analysis

Table 5-1 shows the tabulated temperatures of the main components for both the "hot case" and the "cold case" analysis. As mentioned previously the same power consumption was used for both cases, while actual experiment might have different power setting for different cases. When the CubeSat is in direct sunlight and the solar panels are charging the battery, the electrical components are more likely to operate at full power compare to the eclipse case. Based on the "cold case" analysis, an electric tape heater might be necessary to keep the internal components within the safety limits while the CubeSat is in the eclipse region.

Table 5-1 Components Temperature

Component	**Power (W)**	**"Hot Case" Temperature (^oC)**	**"Cold Case" Temperature (^oC)**
Chip 1	0.15	39.6	-34.7
Chip 2	0.15	39.2	-20.7
Chip 3	0.15	38.4	-28.6
Chip 4	0.2	40.2	-27.3
Quake Q1000	N/A	40 [reference temp.]	40 [reference temp.]

Chapter 6

Design of Experiments Simulation

The previous chapter discussed the analysis of a CubeSat operating at a specific altitude and orbit. This chapter explores how different design parameters, such as inclination angle or altitude, effect the thermal management of the CubeSat. For example, by adding the two solar panels on the sides of the CubeSat to collect more solar energy, as shown in Figure 6-1, the orbital heating is slightly different compare to only having solar panels on the sides. The view displayed the total absorbed flux using the sum of all heating rate sources (solar, albedo, and planetshine). Unlike the uniform heat flux on the surrounding faces, the top and bottom surfaces with the solar cells attached has a non-uniform behavior. This is because these two sides experiences some radiation from the back of the solar cells hitting the top and bottom surfaces of the case, and vice versa.

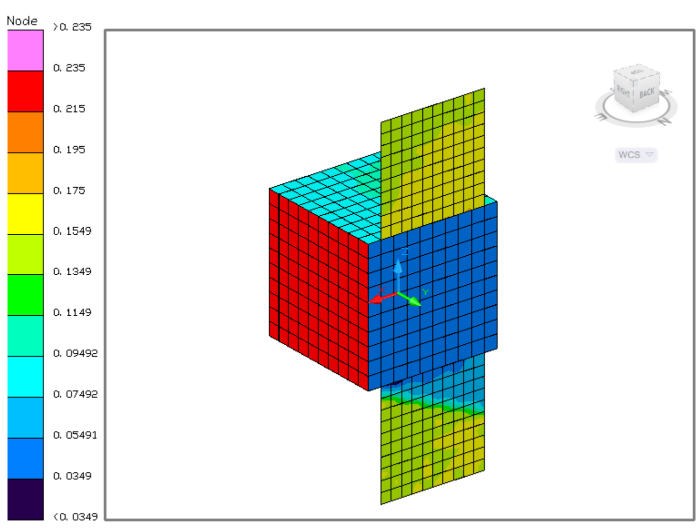


Figure 6-1 CubeSat Total Absorbed Flux

Another parameter that influences the thermal performance of the CubeSat is the beta angle. To study the effect of beta angle on the spacecraft, two simulations were performed for beta angles of 90^{o} and 30^{o}, respectively. The spinning effect was not included in this simulation; this allowed the author to study how the beta angle affects the thermal performance without the bias influence from other factors. Figure 6-2 shows the orbital heat flux for the 90^{o} beta angle. Each line on the graph represents a node on each surface of the spacecraft. As expected, the heat flux remains constant throughout its period because its position and orientation relative to the sun does not change. The heat flux for 30^{o} beta angle, however, does change with time. This is because portion of its time is spent in eclipsed, while the rest of the time it faces sunlight. The changes in orbit cycles also results in changes in thermal cycles.

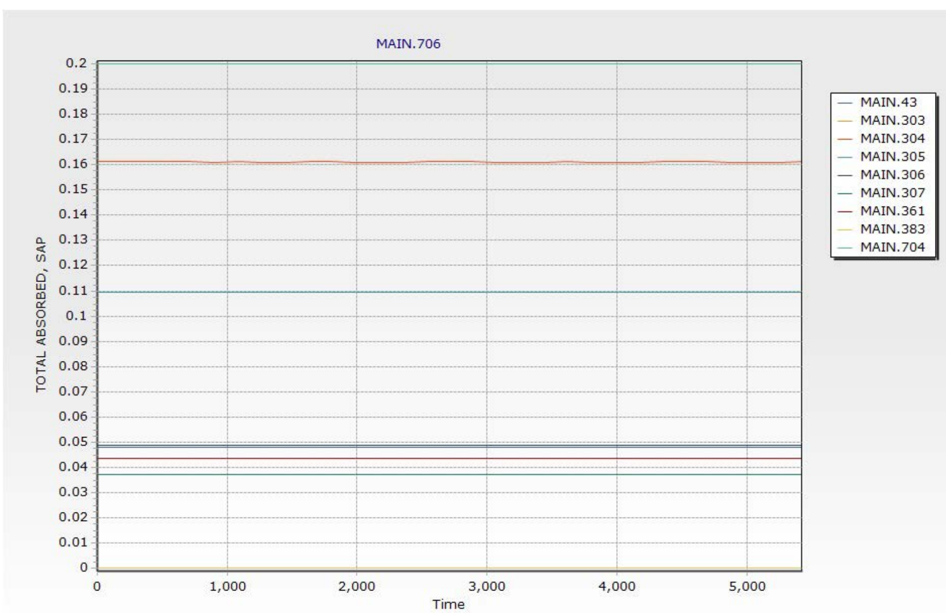


Figure 6-2 Total Absorbed Heat Flux for beta = 90 deg

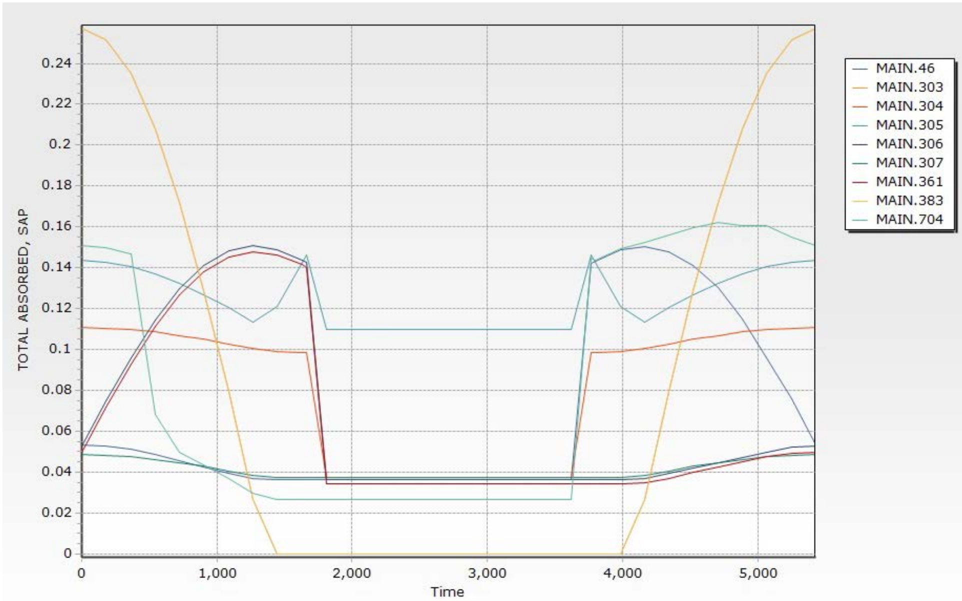


Figure 6-3 Total Absorbed Heat Flux for beta = 30 deg

As can be seen from Figure 6-3, the total absorbed heat flux remains constant from approximately 1,700 to 3,800 seconds. During these 2,100 seconds, or 35 minutes, the CubeSat is facing the back side of the Earth and does not see the heat flux from the sun. This estimate from the graph is about the same as the calculated value using the equations provided in the Orbital Mechanics section. The graph shows that the CubeSat's heat flux increases and/or decreases in periodic pattern with respect to the position of the spacecraft in orbit. The thermal analyst can use this information to design the thermal control system that meets both operating conditions. This information can also be used to calculate how much energy is required during eclipse, and how much power the solar cells can generate during sunlit. Combining all the information can result in a design that is optimized for mass as well as power. This chapter showed that the beta angle has a major influence on the external heating rate of a satellite in LEO.

Chapter 7

Conclusions and Future Work

This thesis analyzed the thermal performance of the nano-satellite in the Low Earth Orbit at 300 and 400 km altitude. High fidelity level models were used to predict the temperatures variation on the external body and the internal electrical components. The results showed that the components did not overheat in the hot case simulation because onboard components were mostly low-power devices. Looking ahead, the electronics will continue to shrink while adding more power and functionality. It is expected that future design of CubeSat will include high-power components to accommodate more sophisticated missions. Thermal control will be necessary to ensure system reliability. From the thermal analysis aspect, thermal control options available for miniature size spacecraft, like the CubeSat, is limited. Active thermal control system would take up too much space, as well as electrical power. On the other hand, passive thermal control system might not be adequate for certain operating conditions of the spacecraft. Therefore it is important to continually update the thermal model as the design changes take place to determine the best possible thermal control option. This process would make sure the system functions properly, and that the overall system is well optimized for mass and power budgets.

Although numerical simulation can be used as a guide for designing the CubeSat, actual thermal testing of the system is still needed to ensure that the system can operate without failure. Current numerical simulation methods are created based on a simplified model of the system and should only be used to compliment testing because numerical model, due to software limitation, cannot simulate all physical conditions correctly. For example, the Ansys Icepak used for internal electronic simulation was designed for the earth environment, and lack the option to simulate the vacuum environment of space. The space environment was simulated by switching off all the flow parameters, and assigned a very low thermal conductivity for the surrounding ambient domain, simulating a model at very high altitude. The spinning effect of the satellite was not simulated in the steady state analysis. It is expected that as the satellite spins while travels in its designated orbit, the heat exchange between the hot surface and the cold deep space reduces the overall maximum temperature of the vehicle. Another limiting factor in thermal modeling is that some manufacturers did not provide sufficient data to control a numerical model. These components were omitted in the numerical model, but actual testing will provide valuable data on the temperature distribution of the overall system.

The aluminum case structure that is typically used for building CubeSat can be optimized to reduce weight, while still maintaining the structural integrity. Since the CubeSat weight specification is limited to 1 kg, further structural study is needed to optimize for the weight. The structural optimization can be couple with thermal optimization to determine the best design.

The key issues of thermal design are performance, reliability and cost. As electronic devices become more complicated and practical applications require more power, thermal analysis using CFD tools become indispensable tool for the design engineers. With sufficient funding, university students can gain valuable hand-on experience building the CubeSat model and launching it to space. The CubeSat model can be used to effectively conduct scientific experiments.

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APPENDIX A – STK Setup

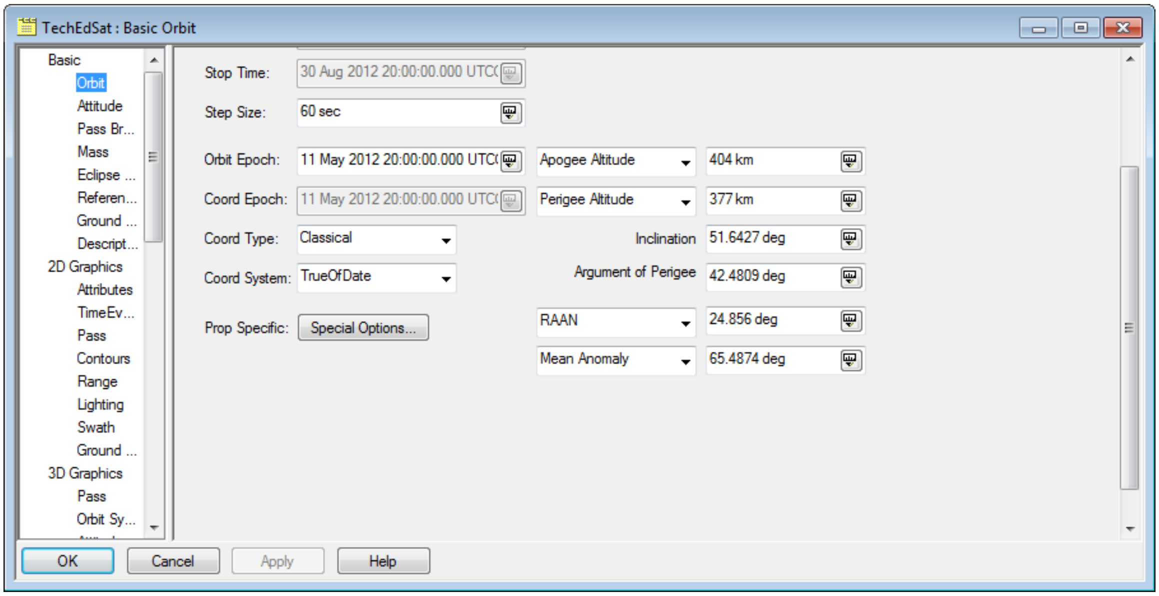


Figure A-1 Satellite orbit setup in STK

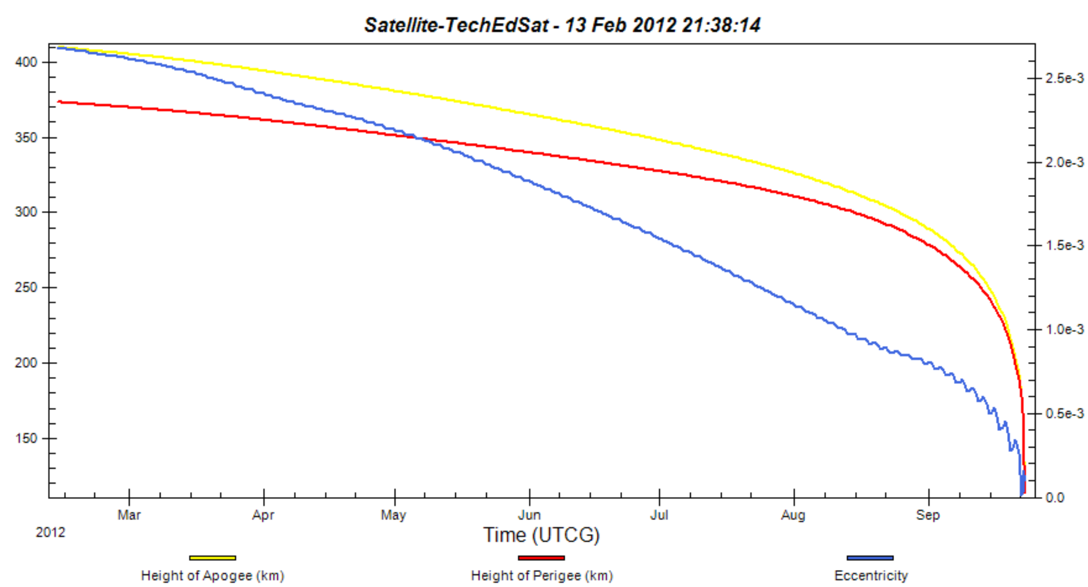
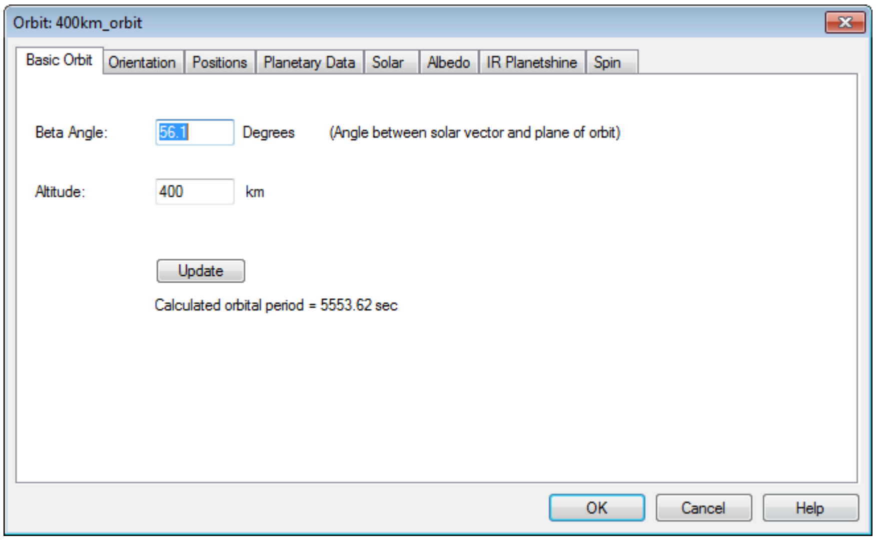


Figure A-2 CubeSat Lifetime – Apogee, Perigee and Eccentricity Variation

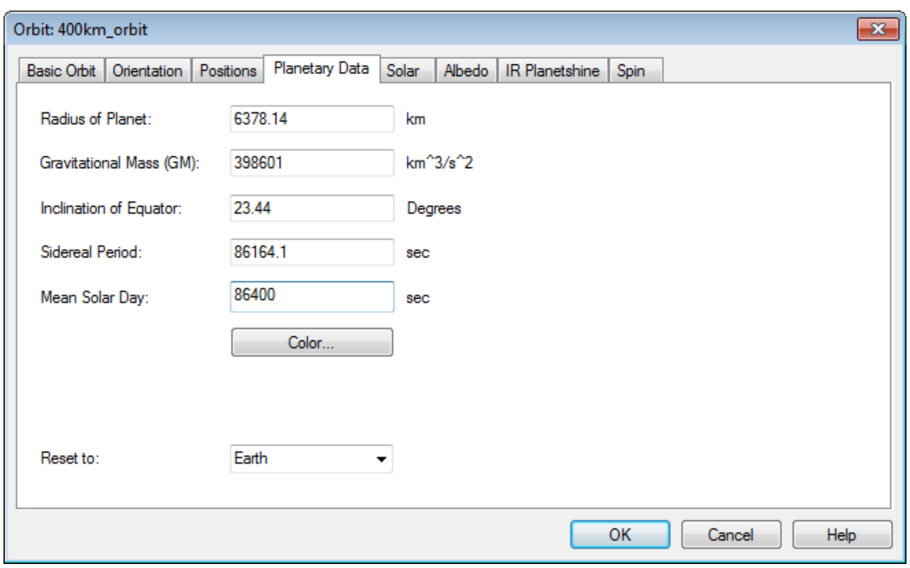
APPENDIX B – Thermal Desktop Setup Guide

Thermal Desktop from C&R Technologies was used for the external orbital heating rate. It allows heat transfer calculations to be performed at any specified mission altitude. Below are some general steps used to set up the external analysis.

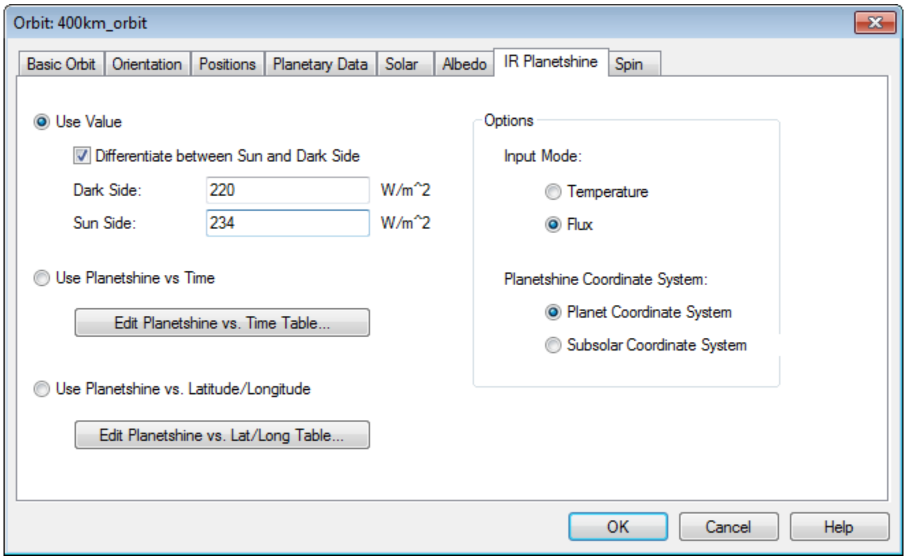
1. Enter the information for the beta angle and mission altitude.



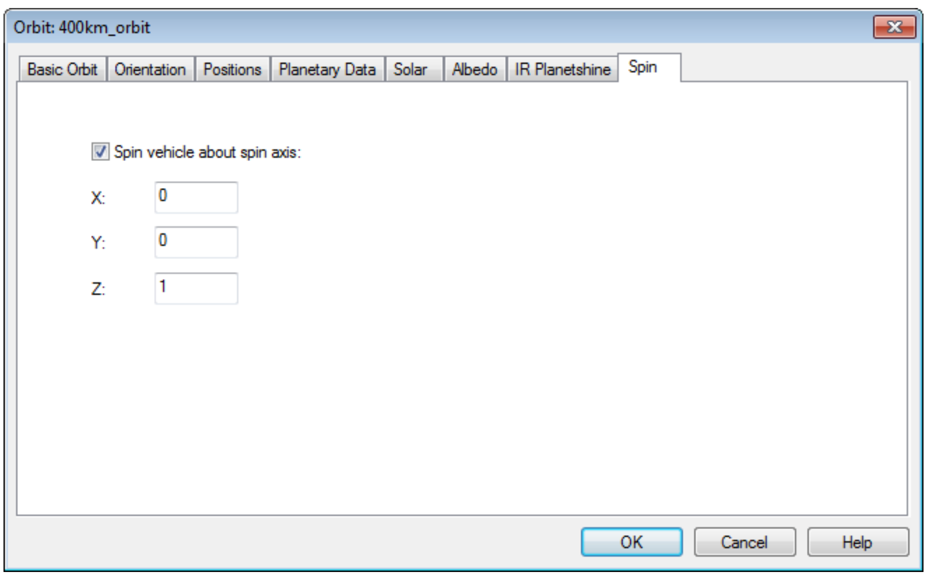
2. The orientation and positions of the satellite can be changed under the Orientation and Positions tabs. For CubeSat operating in LEO, the Earth parameters were used under Planetary Data tab.



3. Next, enter the heat flux values for the hot case and cold case.



4. If the satellite spins as it orbit the earth, the option can be set under Spin tab.



5. The last step is to double check the thermal and optical properties for the simulation. It should be noted that not all the optical properties shown in the figure below were included in the actual simulation.