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Thermal Control of Turksat 3U Nanosatellite

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Abstract

This paper addresses design and analysis of 3U CubeSat satellite thermal control system. The main goal of this study is to ensure all the components will operate in their operating range of temperatures. A thermal control system of a 3U CubeSat at an altitude 600 km with an inclination of 98^o is presented. 3U CubeSat is designed for a circular, near sun synchronous Low Earth Orbit. The dimensions of 3U CubeSat is 10cm x 10cm x 30cm and a weight of 4kg. The thermal model of CubeSat was built in ThermXL. The thermal control analysis of 3U CubeSat's passive thermal control system has been conducted. A temperature distribution of the CubeSat was computed. Based on the results, the temperatures at hot case were within the range of operating but the temperatures at cold case were low. Therefore, the results show that temperatures are highly sensitive to surface coating of the CubeSat. Choosing the surface coating at the early stage of the design is recommended.

Key words: 3U CubeSat, low earth orbit, thermal design, thermal analysis, thermal control.

1. Introduction

Small satellites are typically low-Earth-orbiting satellites that experience a large number of thermal cycles and experience high heat inputs from solar radiation and Earth infrared [1]. In order to prevent the satellite from heating too much, the optical properties of the satellite's surface properly are selected. The main purpose of the thermal control makes sure that all the components are within their allowable temperature limits throughout the satellite mission. All the components on the CubeSat have to work from the beginning to end of the lifetime with the required performances. Thermal design of small spacecraft has more limitations compared to that of larger spacecraft. Although passive thermal control is generally preferred for all spacecraft due to simplicity and cost, it is almost required in small spacecraft. The thermal design is a key step to taken at the beginning of the program in order to maintain the temperature of the components within the allowable temperature range and minimize the mass/power requirements. The CubeSat temperature is controlled by selection of surface property and insulation selection. The infrared emissivity and solar absorptivity of areas around the solar panels are selected to ensure that the temperatures remain between the operating limits of the components [2].

The thermal design and analysis have been widely studied at various CubeSat and published in

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various documents and journals [3-7]. Onetto et al. [3] designed and analyzed the satellite on a Lower Earth Orbit (LEO). Moffitt et al. [4] addressed thermal modeling used during the design and analysis of the combat sentinel satellite. Tsai used a general thermal mathematical model for the entire satellite constructed from a combined conduction and radiation heat transfer equation with environmental heating and cooling as boundary conditions [5]. Bulut et al. [6] modeled CubeSat and the thermal analysis was performed by using ThermXL-spreadsheet-based thermal analysis tool. Bulut et al. [7] studied analytical investigation of a nanosatellite panel surface temperatures for different altitudes and panel combinations with the dimensions of 10 cm x 10 cm x 10 cm.

In this study, thermal design and analysis of the 3U CubeSat operating in LEO orbits has been presented.

2. Space Thermal Environment

The thermal inputs that contribute to the heating of a spacecraft in a Low Earth Orbit (LEO) are couplings between the environment and internal heat generation within internal components. The performance and operational lifetime of spacecraft are influenced by atmospheric environments and the near-Earth space [8]. For the CubeSat launching from the launcher at approximately 600 km altitude, the atmospheric pressure and drag is very small, hence aerodynamic heating and convective heat transfer is negligible. There are three main sources of heat for general spacecraft system operating the near-Earth environment. These are direct sunlight (solar), sunlight reflected off Earth (albedo), and Earth Infrared (IR) emitted by Earth [8]. The amount of external heat absorbed directly from solar energy is a function of an object's material properties and its orientation with respect to the Sun. Energy balance between the satellite and space are shown in Fig. 1. The temperature of the satellite is the result of a balance between absorbed and emitted energy of all of these sources.

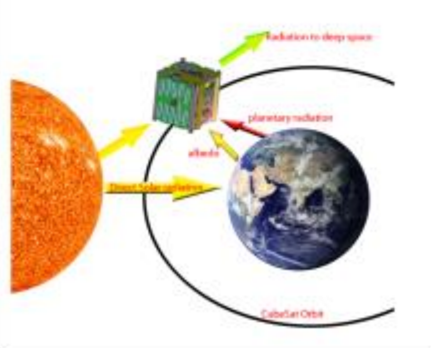

Figure 1. Thermal environments for CubeSat [8].

3. 3U Cubesat Model: Technical Overview

The model of the 3U CubeSat is shown in Fig.2. With dimensions are given as 10 cm width, 10 cm length, and 30 cm height. The primary structure consisted of Aluminum 6061. By design, the satellite structure is a 3U CubeSat with body-mounted solar panels. Figure 2 shows two views of the spacecraft. On the left view shows 3U CubeSat with outer solar panel. On the right view shows 3U Cubesat with structure.

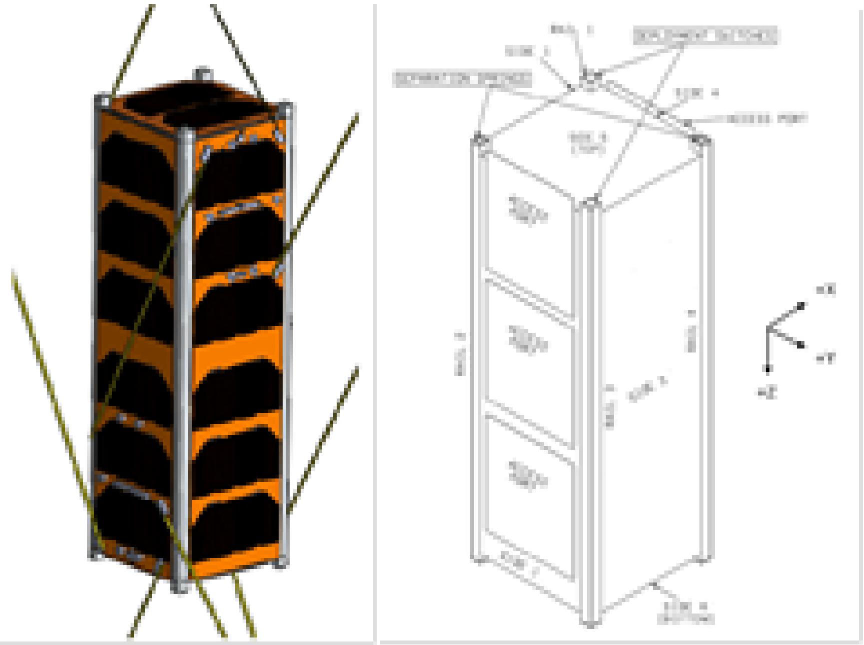

Figure 2. 3U CubeSat (left) with body-mounted panels covering the outer walls and 3U structure (right).

The faces of 3U CubeSat is shown in Fig.3. 3U CubeSat is the combination of three units Cube in the same structure. First Cube has 5 faces including side faces and top face. Second Cube has 4 faces including side faces. Third Cube has 5 faces including side faces and bottom face. All faces are covered with solar cells.

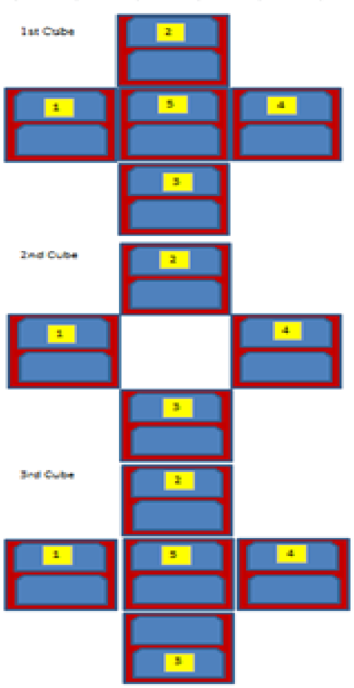

Figure 3. Faces of 3U CubeSat.

In the model of the 3U CubeSat in Fig. 3, all side is covered with solar cells and coatings (aluminum or black paint). On this side the communication antenna is mounted. The temperature requirements for each subsystem are shown in Table 1.

Table 1. Operating temperature limits.

Components	Tmin(^{o}C)	Tmax(^{o}C)	Note
Electronics	-40	+85	
LiPo Batteries	0	+45	charge
	-20	+60	discharge
Main Structure	-40	+85	
Solar cells	-100	+100	

Table 2 provides approximate solar flux, Earth albedo, and Earth IR values for the average, hot, and cold cases. A more detailed description of each source is available in Gilmore's Spacecraft Thermal Control Handbook [9]. Depending on the time of year, the integrated, radiative flux in low earth orbit from the sun can vary from 1414 W/m^2 during the winter solstice and to 1323 W/m^2 in the summer solstice. This variation is due to varying distances between the Earth and sun at certain times during the year [10].

Table 2. Environmental heat sources: average, hot and cold cases.

Parameter	Hot Case	Cold Case
Orbital Parameters	permanently illuminated	max eclipse time
Solar Flux	1414 - 1323 W/m^2	0 W/m^2
Earth Albedo	494.6 W/m^2 (a=0.35)- 330.75 W/m^2 (a=0.25)	0 W/m2
Earth IR	260 K [260 W/m^2]	250 K [220 W/m^2]
Internal dissipation	full (3.8 W)	full (3.8 W)

Material properties for covered surfaces are shown in Table 3.

Table 3. Metarial properties.

	Material Properties	
	Al 6061 T6	Solar Cells
Emmisivity	0,08	0,81
Absorptivity	0,379	0,91

4. Analytical Solutions

The objective of the thermal analysis is to maintain all subsystems and components of a spacecraft within their operating temperature limits for all mission phases [9, 11-12]. The thermal control is to balance the thermal energy of the satellite to ensure all the components remain within their acceptable temperature limits during the worst hot and cold cases. External and internal heat generation must be properly balanced with the excess heat radiated to space. An energy balance analysis between the space environment and cube satellite can be used to determine whether or not the satellite has enough radiative area to maintain its temperature within acceptable limits for the hot case. In addition, it can be used to size survival heat power to maintain the temperature within acceptable limits for the cold case [13]. The steady-state temperatures use a basic energy balance. The equilibrium temperature is obtained from condition $Q_{in}=Q_{out}$. The effects includes in the calculation are internal dissipation (Q_{id}), Solar radiation (Q_{Sun}), Albedo (Q_{albedo}), Earth radiation ($Q_{EarthIR}$) and radiation from the body to space.

The heat balance equation for node i coupled with nodes j though n is written as [4, 9]

$$(Mc)_i \frac{dT_i}{dt} = Q_i^d + (\dot{Q}_{Sun} + \dot{Q}_{albedo} + \dot{Q}_{EarthIR})_i - \sum_j \Im_{ij} A_i^r (\sigma T_i^4 - \sigma T_{jr}^4) - \sum_j K_{ij} (T_i - T_{jk}) \tag{1}$$

The equilibrium temperature can be obtained by equating the heat rate inputs, where

$$\dot{Q}_{albedo} = a \alpha f_e A_s S \tag{2}$$

$$\dot{Q}_{EarthIR} = \varepsilon f_e q_e A_s \tag{3}$$

$$\dot{Q}_{Sun} = \alpha A_c S \tag{4}$$

$$f_e = \frac{1}{2}(1 - \cos \rho) \tag{5}$$

$$\rho = \sin^{-1}\left(\frac{R}{R+h}\right) \tag{6}$$

5. Thermal Analysis Cases

The main goal for the thermal analysis is to model 3U CubeSat temperature distribution in orbit [14]. The thermal environment and the external heat loads are determined from the specific orbit for the mission, the orientation of the satellite, the surface properties, and the size of the system. From these, the absolute worst hot and cold case conditions are determined [13]. In the hot case, 3U CubeSat is in sunlight. The satellite receives the maximum environmental fluxes, orbit illuminated, electrical power entirely dissipated into heat. In the cold case, 3U CubeSat is in the shadow of the Earth and is not in view of any portion of the Sun's radiation. There is no direct solar, albedo energy intake, which also means, that no electric power can be generated by the solar cells [15]. The Earth IR radiation is only the external load.

6. Result and Discussion

A steady state analysis was performed and the temperature results for hot case and cold case on the model are listed in Table 4. As expected, the hottest surface is the one that directed towards the sun and the coldest one that watching only the cold space. The face that watching of the earth is in the intermediate condition. It can be seen from Table 4 that the maximum temperatures vary between 38,6 ^{o}C and 13,5 ^{o}C. The highest temperature is 38,6 ^{o}C. The temperatures of cube satellite subsystem are below the cold limits for eclipse equilibrium temperature. With minimum temperature values vary between T_{min}=-90,3 ^{o}C and T_{min}=-86,2 ^{o}C. The lowest temperature is -90,3 ^{o}C. The cube satellite is too cold.

Table 4. Temperature results

Cube	**Face**	**Max** ^{o}**C**	**Min** ^{o}**C**
1st	1	31,6	-90,3
	2	17,2	-90,3
	3	13,5	-90,3
	4	17,2	-90,3
	5	16,6	-90,3
2nd	1	36,9	-88,8
	2	19,9	-88,8
	3	15,4	-88,8
	4	19,9	-88,8
3rd	1	38,6	-86,2
	2	24,3	-86,2
	3	20,6	-86,2
	4	24,3	-86,2
	5	32,3	-86,2

Conclusions

In this paper, thermal design and analysis of the 3U CubeSat operating in LEO orbits has been presented. The main advantage of the simplified model was that solutions could be obtained rapidly and easily.

3U CubeSat components need to be maintained within their acceptable temperature limits in order to remain operational. It was a goal to provide an adequate control because of the low available thermal power. Therefore, the thermal control would be as passive as possible to minimize mass and power requirements.

The results show that surface coating of 3U CubeSat affects the temperatures. Therefore, the surface coating at the early stage of the design is recommended.

Results of the analysis involving only radiation of the satellite were represented by a three unit simple cube. To keep the cube satellite warm in the cold case, electrical heaters needs to be implemented.

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