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Design and Calculation Analysis for Thermal Control System of CubeSat Modeling

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A B S T R A C T

Thermal control subsystem (TCS) as one of the satellite subsystems has a function for maintain a thermal control for over all the components of the satellite within their required temperature limits for alls mission phase. This objective can be reached with keeping the balance of energy, among incoming energy and discharge energy. In this paper temperature distribution (worst hot case and worst cold case) of nanosatellite in low earth orbit using some data assumptions for calculating the spherical, solar array and sphere of the satellite are used. For the sphere of the satellite is not describes in this paper. Temperature distribution of a cubsat was computed using SCDE (Spacecraft Control Design Engineering) based on the Microsoft Excel. It was created self by author using some references. The thermal analysis results show that average the maximum and minimum temperature's based on low earth satellite's altitude (400 km to 700 km) are 32.62°C and -75.32°C for the spherical of the satellite, and 67.15°C and -75.12°C for the solar array of the satellite.

Keywords: Thermal, Temperature, Satellite, Spherical, Solar Array.

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1. Introduction

The main goal of Thermal Control Subsystem is to keeping all subsystem of the satellite in the allowable
range at all mission phase. It is necessary subsystem for a satellite, even a nanosatellite has thermal control subsystem (Janson et. al., 1993; Nazari and Emami, 2008). The thermal analysis and control provide the necessary means to control the temperature of the satellite during the harsh conditions in space. Spacecraft thermal control is a process of energy management in which environmental heating plays a major role (Wertz and Larson, 1991). Both of the thermal design roles are some parameters such as thermo optical properties of thermal surfaces, specific heat and thermal conductivity of materials, components power dissipations and thermal boundary conditions (direct solar flux, albedo reflected radiation, infrared radiation reflected from earth and heat generated by satellite's internal electronics), and too some orbitals elements data. Many thermal analyses have done with thermal models using both of software analysis or a thermal calculation analysis using the energy balance equations. All of them are gives us many experiences for satellite thermal control system analyzing. Nazari and Emami (2008) studied thermal control techniques about power consumption for observation satellite with compare between passive and active thermal control analysis. Bulut et al., (2010) reported thermal design and analysis for hot and cold cases temperature of nanosatellite using ThermXL software. Da Silva et al., (2014) reported thermal design control and analysis for payload temperatures of the Amazonia-1 satellite in three critical cases and verified using SINDA/FLUINT thermal analyzer. Nobari and Novinzade (2009) presented the thermal analysis of the satellite using the first law of thermodynamics where in elements number of thermal subsystem network can be solved with the energy balance equations. Arslantas et al., (2017) presented the thermal analysis simulation model using six nodes and thermal physical parameters tolerance value based on ECSS standard for calculating the maximum and minimum temperatures of the thermal model surface of the satellite.

In this study, the model is constructed using the energy balance equations to compute the equilibrium temperature of the satellite using mathematical model technique. The development of nanosatellites is currently a trend technology in the area of space science and engineering research (government, universities, industry, aerospace, militery, etc). The cubesat under this study is a cube shaped satellites have a mass approximately 1 kg based on the standard (Mehrparvar, A., 2014), and surface dimension of 10 x 10 x 10 cm each. The geometry of cubesat model was built in Femap. Femap is finite element modelling and post-processing software that can be used to create geometry or import CAD geometry from other model software. The cubesat model also design approach is using passive thermal control with consideration of the simplicity, cost, the reliability, the limited mass and the lower power consumption (Bulut et al., 2010). The passive thermal control too is mostly defined by its thermo optical properties. The

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highest and lowest temperatures on cubesat surface during on the orbit can be calculated with consideration the effect of thermo optical properties in order to find out how much the temperature that occurs on the satellite. The thermal analysis of cubesat in this study is used some data such as sun emissivity, temperature absolut of sun surface, radius of sun and radius of earth for radiation calculation, because this parameter will be used in spherical, solar array and sphere for satellite (Solar Constant in Appendix B). In orbital heat fluxes, solar radiation has the maximum value of 1428 W/m2, and the minimum value of 1316 W/m2. The radiation of a black body is used at 288°K (emissivity of sun value is 1) (Abouel-Fotouh et al., 2006). This paper describse of the thermal design and calculation analysis of cubesat using the mathematical model in equation energy balance approach in SCDE (Spacecraft Control Design Engineering) based on Microsoft Excel for predicting worst hot and cold case distribution temperatures on the satellite.

Thermal control system of the satellite consists of active and passive thermal control elements in order to maintain the satellite components and structure within a controlled range of temperature throughout the mission of the spacecraft, from the Beginning of Life (BOL) to the End of Life (EOL) (da Silva et al., 2014). The passive thermal controls are usually uses OSR (Optical Solar Reflector) and MLI (Multi Layer Insulation), and active thermal control are usually use heater and heat pipe. The heaters function to maintain the temperature of the battery and earth sensor (Tetsuo et al., 2007), and in part of a thermal subsystem, a heater is too have responsible for maintaining ideal temperature range for the different satellite subsystems (Nobari and Novinzade, 2009).

2. Methodology

For the thermal analysis of a satellite the following data must be known as in the design process, meanwhile some data provision such as Stefan Boltzmann, earth radius, solar flux, and earth IR emission. The thermalization analysis process in this study could be implemented as the following process in Fig.1.

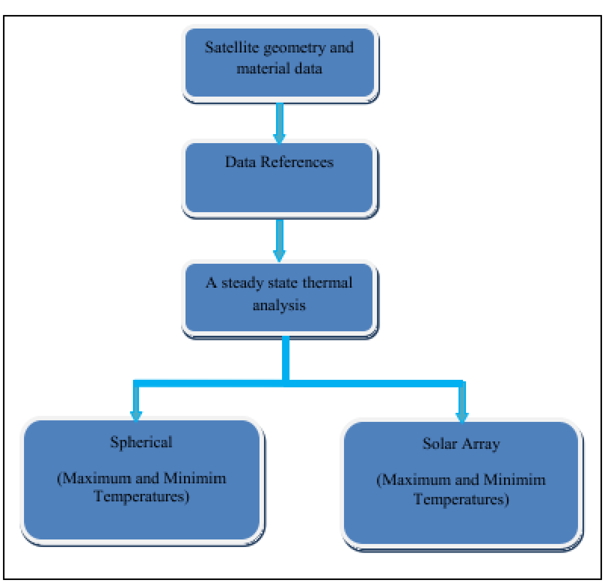

Figure-1. Athermalization calculation process

The geometry model of the cubesat in this study is presented by Fig.2, with a cube shape platform. The structure material uses an aluminum 7075-T651 with thermalphysical properties are presented in Table.1, in order to thermal analysis each that the thermal spherical spacecraft with an aluminum material panel have absorptivity and emissivity value are 0.6 and 0.8. For solar array panel are use absorptivity and emissivity on the top surface value are 0.68, and solar panel on the bottom surface used absorptivity and emissivity value are 0.6, whereas both the absorptivity and emissivity for sphere of the satellite have value are 0.6 and 0.8. Using SCDE for thermal spherical, solar array and sphere calculation was computed easily with derified all formula in Microsoft Excel. Using some data provitions constants and both data when we need, the worst hot and cold case can be calculated.

Table 1: Thermalphysical properties

Properties	Value	Unit
Conductivity, k	0.20139	W/mm°C
Specific Heat, Cp	77.28	J/Ton°C
Mass Density, ρ	2,59E+01	Ton/mm3

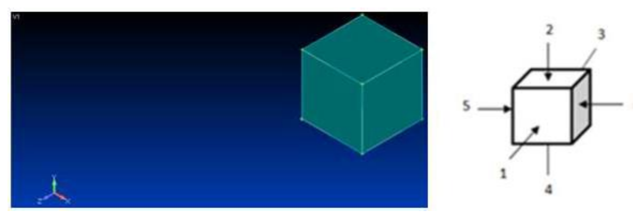

Figure-2. Cubesat model and configuration

3. Result and Discussion

Thermal control system in this study is managed using passive thermal control system with some considerations of an orbit environment for the cubesat, and appropriate the structure surface treatment by controling thermal properties is used. The objective of the thermal control system is to keep the temperature of the satellite and its components within the allowed temperature ranges, and therefore for extremely severe temperatures ranging when satellite orbiting in space between -150°C to +150°C, and basically often use the electronics and structure components in satellite based on COTS (commercial off-the-shelf) within temperatures operation range are between -20°C and +80°C. It's still allowable in the satellite platform. The aim of thermal design is to provide the comfortable environment for all the components in the cubesat. The thermal control of the cubesat in orbit is usually achieved by balancing the thermal energy, dissipated by the internal electronics components and the energy absorbed from its environment with the energy emitted by a satellite. Its mean conservation of energy can be described as equation (1) (Czernik S., 2004).

$$Q_{in} = Q_{out}$$

$$Q_{sun} + Q_{albedo} + Q_{earth} + Q_{P} = Q_{sat-earth} + Q_{sat-space} \qquad (1)$$

The temperature of an orbiting satellite with internal power dissipation QW, available area surface of the satellite A, and satellite cross sectional AC, the radiation properties of its surface and orbital geometry are determined by its radiation environment (Hass and Schach, 1959). All of them are considered for a spherical satellite thermal calculation. The maximum and minimum temperatures for the spherical satellite thermal can be calculated by equation (2) and equation (3) (Col. John E. Keesee, In Lecture Notes).

$$T_{max} = \left[\frac{(A_C G_s.\alpha) + (A.F.q_{1max}.\varepsilon) + (G_s.a.\alpha.K_a.A.F) + Q_W}{A.\sigma.\varepsilon}\right]^{1/4} \quad (2)$$

$$T_{min} = \left[\frac{(A.F.q_{1min}.\varepsilon + Q_W)}{A.\sigma.\varepsilon}\right]^{1/4} \quad (3)$$

Where F is the view factor of cylinders (= (1- cos ρ)/2), the ρ is the angular radius of the earth , the Ac and A are the area of cylinders (m2) and the satellite surface area (m2), the Gs is solar flux (W/m2) with a blackbody is equal to 1418 W/m2, the σ and ε are absorptivity and emissivity of satellite surface, the σ is Stefan Boltzmann's constant, equal to 5.67x10-8 W/m2K4, the a is albedo, equal to 0.3 up to 0.35. The q1min and q1max are the minimum and maximum Earth IR emission, equal to 216 W/m2 and 258 W/m2, the Ka is the albedo reflection factor, and the Qw is power dissipation on satellite.

Using equation (2) and equation (3), we can do hand calculation for both of steady state temperatures the satellite (Tmax and Tmin) with some data required as, radiation from the sun with effective area exposed to solar radiation is 0.06 m2 and solar emitted heat flux is 1418 W/m2. Radiation and albedo from the earth with infrared load is assumed to be 258 W/m2, the albedo coefficient is 35% and maximum power dissipation is 1.8 Watts. The spherical maximum temperature for satellite can be calculation with equation 2 and its results is 58.07°C. Otherwise, when radiation and albedo from the earth with infrared minimum load is assumed to be 216 W/m2 and minimum power dissipation is 1 Watt, the spherical minium temperature for satellite can be calculation using equation 3 and its results is -37.24°C, respectively.

Most satellites are used solar arrays as a primary power source for long lifetime the satellite operation in orbits. Furtherrmore, solar arrays are to be assisted by batteries that used as secondary power source to store and deliver energy. This store energy is needed by satellite when the operation of the satellite systems and charging the battery to supply power when in eclipse periods (McGuire, et al., 2016). For this energy, it must be considered also the amount of solar array temperatures using equation (4) and equation (5) (Col. John E. Keesee, In Lecture Notes).

$$T_{max(SA)} = \left[\frac{(G_s.\alpha_t) + (F_p.q_{1max}.\varepsilon_b) + (G_s.a.\alpha_b.K_a.F_p) - Gs\eta}{\sigma.(\varepsilon_b + \varepsilon_t)}\right]^{1/4} \quad (4)$$

$$T_{min(SA)} = \left[\frac{(F_p.q_{1min}.\varepsilon_b)}{\sigma.(\varepsilon_b + \varepsilon_t)}\right]^{1/4} \quad (5)$$

Where Fp is the view factors (Fp = sin2 ρ), the εt and the εb are the IR emissivity on the top and the bottom surface of solar arrays, the σt and the σb are the solar absorptivity on the top and the bottom surface of solar arrays, and the η is the solar array efficiency.

Using equation (4) and equation (5), we can do hand calculation for both of steady state temperatures the satellite (Tmax and Tmin) with some data required such as solar absorptivity and IR emissivity on the top surface of solar array are 0.68 and 0.88, and solar absorptivity and IR emissivity on the bottom surface of solar array are 0.6 and 0.8 with solar emitted heat flux is 1418 W/m2. Radiation and albedo from the earth with infrared load is assumed to be 258 W/m2, the albedo coefficient is 35% and maximum power required during daylight is 10 Watts. Finally the maximum solar array temperature for satellite can be calculated using equation 2 and its results is 68.43°C. Otherwise, when radiation and albedo from the earth with infrared minimum load is assumed to be 216 W/m2 and minimum power required during eclipse is 5 Watts, the minium solar array temperature for satellite can be calculated using equation 3 and its results is -72.96°C. The goal of the implemented calculation by SCDE is to calculate the temperature profile of the satellite with Low Earth Orbit. Furthermore, we can describse some amount of the temperatures for satellite based on the satellite altitude in orbit (400 km to 700 km) is plotted in Table 2.

Table 2. An example of a table.

Altitude (km)	Spherical Temperature		Solar Array Temperature	
	Max	**Min**	**Max**	**Min**
400	35.05°C	-72.30°C	68.43°C	-72.96°C
500	33.31°C	-74.45°C	67.56°C	-74.42°C
600	31.76°C	-76.38°C	66.71°C	-75.85°C
700	30.36°C	-78.15°C	65.89°C	-77.25°C
Average	32.62°C	-75.32°C	67.15°C	-75.12°C

4. Conclusion

Based on altitude, the satellites in low earth orbit have an experience a different maximum and minimum temperature environment and the usage of thermo optical porperties properly have effect to the satellite. In this study, thermal control of the satellite model can be solved using the energy balance equations to compute the temperature gradients, the result of the thermal analysis both for spherical and solar array of the satellite based on satellite altitude have turned out of extremly for used in the satellite. The analysis results show that the average the maximum and minimum temperature's based on low earth satellite's altitude (400 km to 700 km) are 32.62°C and -75.32°C for the spherical of the satellite, and 67.15°C and -75.12°C for the solar array of the satellite. The calculations show that the minimum temperature for solar array is need thermal active control system, and battery is needed for store the power energy solar array.

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