Thermal Control Subsystem for CubeSat in Low Earth Orbit

Harsh Vardhan Mishra[1]

[1]Department of Chemical Engineering, Birla Institute of Technology and Science, Pilani, Rajasthan, India

---***---

Abstract – *The purpose of the paper is to present the thermal control subsystem for a satellite at 580km altitude in the Low Earth Orbit and at 98.1° inclination. Miniaturization of components enabled small scale satellite projects, such as the CubeSat, to be used for scientific research in space. Although the integration of compact electronics allowed sophisticated scientific experiments and missions to be carried out in space, the thermal control options for such small satellites were limited. For example, due to CubeSat's small size there was no room for dedicated a radiator or insulation panels. To minimize the mass of the thermal control subsystem while keeping the electronics at safe operating conditions, this paper presents a study of the thermal environment and thermal control system of the satellite. Numerical analysis and proposed subsystem is also provided in the paper.*

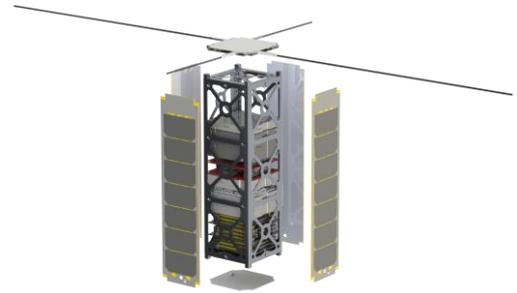*Key Words*: **CubeSat, inclination, passive thermal control, multi-layer insulation, temperature, altitude**

1. INTRODUCTION

Advances in computer technology and manufacturing processes allow creation of highly sophisticated components in compact form. This miniaturization of components allow small scale satellites, 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 California Polytechnic State University and Professor Bob Twiggs from the Stanford University. 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 opportunity for many undergraduate and graduate students to have hands-on experience building small payload satellite that can be launched to space as a secondary payload. A typical size of a 1U satellite is a 10cm cube aluminium T6061 structure with a total mass of 1 kg and operates autonomously in orbit. This small satellite design model can easily be modified to accommodate different missions. The project does a study on a 3U satellite with a total mass of 4 kg.

There are three main operating phases concerning the thermal environment of the CubeSat are: (1) launch, (2) mission lifetime and (3) re-entry self-destruction. Depending on the mission objective, there are usually several levels of embedded hardware with different power consumption.

Fig-1: 3U CubeSat [4]

The thermal design of a satellite is very important to ensure the health and longevity of a spacecraft in Earth orbit. The objective of the Thermal Subsystem is to keep every component in the determined temperature range.

The thermal control subsystem design is configured for simple and reliable temperature control that provides the flexibility to accommodate variations in the spacecraft heat load.

The approach to thermal control of the spacecraft uses conventional passive techniques such as selective placement of power dissipating components, application of surface finishes, and regulation of conductive heat paths. The passive design is augmented with heaters for certain components (particularly with narrow allowable temperature limits) and with louvers. The thermal control is maintained by minimum heat transfer between major components of the spacecraft. Designing and modeling an effective spacecraft Thermal Control System (TCS) requires an understanding of the top level mission objectives as well as the physical interaction between components and their operational environment. The mission objectives translate into the system, assembly and component-level thermal control requirements. Thermal control is accomplished through a variety of methods both active and passive that are used to maintain component temperatures within their operational or survival range.

Most small and nano-satellites strive to use passive thermal control methods. Passive thermal control can achieved through the use of special surface finishes and proper orientation of heat producing and heat rejecting components. Some spacecraft components are extremely sensitive to changes in their thermal environment and may require additional active thermal control mechanisms to maintain their thermal environment.

The paper presents a reliable thermal control method for a satellite (CubeSat) at 98.1° inclination and 580km height i.e. in low earth orbit. The incident solar radiation depends on the angle of inclination and height in the low earth orbit. The time spent in eclipse and in solar radiation depends on the orientation of the satellite. Radiation and conduction rates are two factors which decide the thermal control method used. Passive thermal control is used for the satellite. Active thermal control is used to supplement the passive control method used. Paper presents the laws used in calculating the radiation and conduction rates and calculation of the time spent in eclipse and in solar radiation. Paper also presents the thermal control method which is suitable for the CubeSat.

Table-1: Design Temperature of satellite components

Component	T_{min} (°C)	T_{max} (°C)
Battery (Charge)	0	45
Battery (Discharge)	-20	60
CPU	-40	85
Micro-spectrometer	-20	40
Magnetometer	-40	85

2. 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 400km altitude, the atmospheric drag and pressure 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 Earth (black body radiation from Earth). Solar activity varies daily and its variation between solar maxima and minima often dictates the design of the spacecraft.
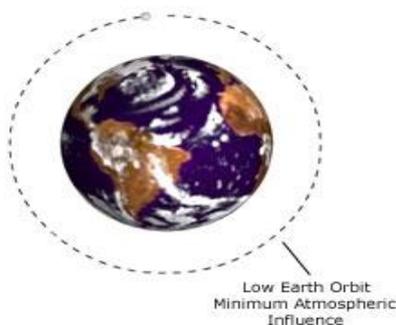

Fig-2: Low Earth Orbit [5]

LEO environment characteristics:

A. Earth rarified atmosphere, with extremely low density and pressure, but very high temperature

B. Earth magnetic field (due to relative motion of Earth's iron nucleus). A magnetometer measure the field strength and, when used in a three-axis triad, magnetic field direction, what can be used to know the spacecraft attitude (if its position is known). By using magnetic coils to interact with the geomagnetic field, it is possible to control attitude in small satellites

C. Solar Radiation. Mainly electromagnetic radiation (EMR) in amount of $1360 W/m^i$ where i =2.[1]

Table-2: Satellite orbital environment

i=2	Solar Constant (W/m^i)	Earth Emission (W/m^i)	Albedo Coefficient
Cold Case	1321.0	275.0	0.35
Hot Case	1423.0	201.0	0.25

3. Radiation and Conduction:

There are three mechanisms by which thermal energy is transported:

1. Convection
2. Conduction
3. Radiation

In space convection can be neglected. The surrounding temperature is 2.7K, very close to absolute zero. Then there is no transfer between hot and cold air as on the Earth. The vacuum is also cause for this problem, because there is no difference in density of gases.
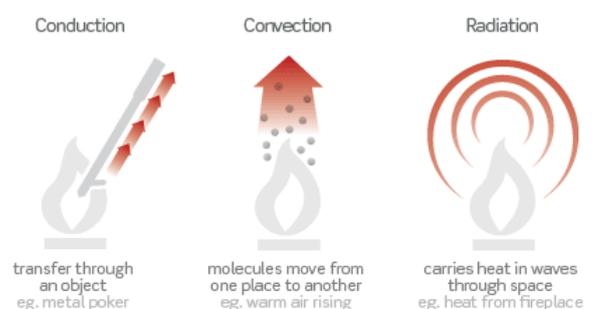

Fig-3: Conduction, Convection and Radiation [6]

3.1 Conduction

In definition of heat transferred through conduction, the following formula is used:

$$P_{cond} = (\lambda.A.\Delta T)/L$$

where λ is thermal conductivity[J/s.m.K]
    A is cross-sectional area [m^i where i=2]
    L represents the length of the conduction path[m]

ΔT is difference temperature between the two bodies, T1 and T2

Thermal resistance is defined by formula

$$R_{th\_cond} = \Delta T/P_{cond} \text{ [K/W]}$$
$$= L/\lambda.A \text{ [K/W]}$$

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 follows:

$$Q_{x}'' = -k.(dT/dx)$$

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

3.2 Radiation:

All matter and space contains electromagnetic radiation. A particle of electromagnetic energy is a photon, and heat transfer by radiation can be viewed either in terms of electromagnetic waves or in terms of photons.

For calculation of amount of heat transferred through radiation is valid Stefan-Boltzmann law:

$$P_{rad} = \varepsilon.\sigma.A.T^4$$
$$Q_r = \varepsilon.\sigma.F.A. (T_1^4 - T_2^4)$$

where Q_r = amount of heat transfer by radiation (W)
    ε = emissivity of the radiation surface (reflective =0, absorptive =1)
    F = shape factor between surface
    A = surface area of the object [m^i where i=2]

The thermal resistance for radiation can be defined as:

$$R_{th\_rad} = \Delta T/P_{rad} \text{ [K/W]}$$

4. Temperature Equilibrium:

A temperature equilibrium will be always reached after the solar irradiation has begun. The temperature equilibrium is obtained when the absorbed power Q_a is equal to emitted power by radiation Q_e.

$$Q_a = Q_e$$

Using the energy equilibrium of equal incoming and outgoing heat, the outgoing heat is solely influenced by the radiation through all the six side areas, while the incoming heat is divided up into the component of solar, IR and albedo influence, as well as the internal dissipation. It is also assumed that the entire external heat energy input (solar, IR and albedo) is received through the projected area of one spacecraft side facing the Earth.

5. Calculation of time in eclipse and sunlight:

The orbital heating rate simulation is simplified by assuming that the CubeSat travels in circular orbit.

Using the standard values for Earth gravitational constant and radius, the orbit's semi-major axis a, orbit velocity and period P are calculated as follows:

$\mu_{Earth} = 3.986 * 10^5 \text{ km3/s2}$
$R_{Earth} = 6378 \text{ km}$
$a_{mission} = 6378+580 = 6958 \text{km}$
$\vartheta = \sqrt{(\mu/a)} = 7.57 \text{ km/s}$
$P = 2\pi\sqrt{(a^i/\mu)}$ where i=3
Earth's angular radius at mission altitude is given by
$\rho = \sin^i(R/a) = 66.44°$
where i=-1

The maximum time of eclipse (TE) and the time in sunlight (TS) are given by
$TE = (2\rho/360)*P = 2130.74$ seconds
= 35.51 minutes

$TS = P – TE = 3645.36$ seconds = 60.76 minutes

The time of eclipse and time in sunlight helps determine the expected thermal conditions of the spacecraft during these time intervals. [2]

5.1 Wein's displacement law:

The wavelength λ_{max} for which the radiated power per unit of wavelength is maximal, is related to the temperature of the thermal emitter as

$\lambda_{max} * T = $ constant = 0.28978 [cm/K]

5.2 Solar beta angle:

The solar beta angle is another important variable affecting the solar radiation effects. The beta angle is useful to describe the thermal environment of a satellite in low Earth orbits, which is defined as the minimum angle between the orbit plane and the solar vector. Consequently, the analyses were performed on a full range of beta angles, from -90° to 90° to ensure that requirements are met. [3]

6. Proposed Thermal Control Subsystem:

The paper presents a thermal control subsystem for a satellite i.e. a CubeSat at 580 km altitude and 98.1° inclination. The system consists of passive thermal control methods along with some active control methods for emergency situations. Passive control methods consist of

Multi-Layer Insulation (MLI), paints, coatings that change the thermo-optical properties of the external surface, thermal washers and doublers. Some active control method like heat exchange via fluid like ammonia and louvres can also be used for thermal regulation of the CubeSat environment. Keeping the batteries and thermal sensors at suitable places can be used for thermal regulation of sensitive components. Keeping sensitive components towards the middle of the satellite can be used for their thermal regulation. Thermistors can be used for temperature detection.

There will be significant conduction heat loss through components to structure panels and to the space if there is no thermal control along the heat transfer path. This may cause some component temperatures, especially the batteries, to fall below their allowable temperature limits. Unfortunately, due to the inherent limitations of picosatellites, traditional active thermal control methods, and even semi-passive method are no longer practicable. Thus, the only feasible thermal control method is a passive technology such as surface finishing, painting, or an application of thermal isolators. In order to fulfill the system requirements, besides, the selection of thermal control materials must conform to the following standards:

1. Material shall be selected with low-outgrassing characteristics to avoid the contamination on other spacecraft equipment

2. All electrically conductive layers of thermal finishes shall be grounded to the vehicle structure

3. Thermal finishes and materials that have low degradations in solar absorptance shall be emphasized

7. Advantages of the System:

A. System design is passive heat transfer system

B. System is passive because they operate without any additional power or control in order to maintain a thermal balance

C. MLI reduces the amount of heat lost when the satellite is not in sunlight

D. MLI is thermal blanket used in various satellites because of low mass, low volume and simplicity

E. Thermistors are cheap and effective

F. Louvres change the heat rejection capability to space as a function of temperature

G. Active systems are complex than passive systems

8. Conclusion:

The paper presents an effective thermal control subsystem for a CubeSat at 580 km altitude in the Low Earth Orbit and at 98.1° inclination. Passive thermal control system is effective because of it's cost effectiveness and low physical properties. Multi-layer insulation is used in the paper for passive thermal control but paints and some active control system such as louvres and heat exchange fluids can also be used for better and safer thermal control of the components. Keeping components at appropriate places can help regulate thermal control of temperature sensitive parts of the satellite.

References:

[1]http://webserver.dmt.upm.es/~isidoro/tc3/Space%20environment.pdf

[2]https://www.crtech.com/sites/default/files/publications/Dihn.S12_0.pdf

[3]Lee-Her Hu, Ming-Shong Chang, Jih-Run Tsai "Thermal Control Design and Analysis for a Picosatellite – Yamsat" Transactions of the Aeronautical and Astronautical Society of the Republic of China (2003)

[4] www.wvspacegrant.org/programs/cubesat/

[5]www.polaris.iastate.edu/EveningStar/Unit4/unit4_sub3.htm

[6] www.insulfoam.co.nz/rvalue.php