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Thermal Analysis of a 3U-Cubesat, a Case Study of Pakal Satellite

Gabriel Salazar-Salinas[1], Estefanía Botello-Ramírez[1], Edgar Avalos-Gauna[1]
[1]Universidad Panamericana, campus México
Augusto Rodin 498, 03920, Ciudad de México, México
0183645@up.edu.mx, 0204112@up.edu.mx, eavalos@up.edu.mx

Abstract - The current study presents the thermal control design of the 3U Cubesat "Pakal" from "Misión Colibrí". As part of the design requirements, a thermal analysis was carried out to study the effects of the space environment over the spacecraft. This was achieved by making a lumped parameter approach in the critical thermal cases and studying the effects of environmental heat fluxes on each critical component. The model was verified using Thermal Desktop software. Then a Passive Thermal Control (PTC) was proposed to keep critical components at their operating temperature. The proposed PTC consists of two types of passive control: thermal coatings and single-layer insulation.

Keywords: Cubesat, Thermal analysis, numerical modelling, heat transfer.

1. Introduction

CubeSats belong to a specific group of satellites built mostly for academic research. Their development can be traced back to the late 1990's. For a satellite to be considered as a CubeSat, it must follow certain criteria such as shape, size, and weight. CubeSat are a 10 cm cube with a mass of approximately 1 to 1.33 kg. This is what is called a 1U Cubesat (1-unit CubeSat) [1]. During their design, CubeSats are checked for thermal management, stress analysis, control, amongst other tests [2]. Their survival during launching, deployment and later while orbiting, highly depends on how accurate these assessments are [3]. The thermal design must ensure that all CubeSat subsystems work under acceptable temperature ranges [4]. Often, this analysis has been done using numerical modelling. Yet, this approach incurs in several hours of simulation until an acceptable parameter value is achieved [5]. As the number of parameters increases, the simulation model becomes more complex. Some of the parameters to consider during the heat transfer performance analysis of a Cubesat are the attitude of the orbit, inclination angle, orbiting speed, etc. The importance of the heat transfer analysis is that it helps in finding the best way to dissipate heat from the satellite. Once the analysis is complete, it is possible to choose between different thermal control techniques for the satellite. In the literature, it is possible two find two main types of thermal control for satellites: active control, which is based on components that require electrical power consumption such as electric heaters and coolers; and passive control consisting of different non-electrical components such as thermal coating surfaces and finishes to reduce or increase heat transfer [6].

Some examples of thermal analyses for CubeSats can be found in the literature. For instance, a thermal analysis for the UoMBSat-1 picosatellite was developed in [7]. It consists first of an analytical approach with a lumped parameter model that divides the picosatellite into 17 discrete nodes connected by a thermal network. Then a finite element model using ANSYS software was made to validate the results. Different values of the beta angle were tested to find the optimal conditions in which the thermal environment is such that thermal control is minimized. An important conclusion is that small satellites like UoMBSat-1 can consider launches with high beta angle values. Another example, the MIST 3U CubeSat study [8] provides the thermal analysis and control of the satellite. For this case, three scenarios were considered: "Hot Operational Case" (This is the worst scenario in which all the components have functional temperatures but they have to be limited to operating conditions), "Cold-Non Operational Case" (This is the worst scenario in which "none of the payloads are functional but still have to be maintained within the non-operational temperature limits"); and "Cold Operational Case" (In this scenario "all the payloads and subsystems are switched ON and are dissipating heat"). An analytical model was made in combination with a thermal model using software System-Thermica. Active thermal control was proposed as a thermal control approach to keep payloads and other components within their operating temperature range. In [9] the design of the power, propulsion, and thermal subsystems of a 3U CubeSat is presented. For the thermal design, steady state analytical calculations were made. Then a steady state simulation approach was made using SolidWorks software. Although the simulation validated the analytical results, the temperature magnitude resulted to be over other results reported in literature. For this case both passive and active thermal control were considered. In [10], a thermal model was developed for SwissCube. It is a 1U CubeSat launched in 2009 that has been sending different types of flight data for several years. The analysis consists of a simple thermal model for educational purposes dividing the cubesat into 2 nodes; an external node that represents the outer surfaces of the satellite and an internal node that represents the inner components. The analysis considered external heat sources from space and heat transfer between the internal and external nodes. The temperature results were validated with the temperature data collected from the CubeSat sensors during its flight operations.

The current study presents the numerical analysis and thermal control of the 3U CubeSat "Pakal" whose main objective is to measure atmospheric conditions in a low Earth orbit. This will be done by measuring the change in position of a sphere located in a box enclosure in the second unit. Due to mission specifications, there is limited power available. Thus, implementing active control will not be possible and therefore different passive thermal controls are proposed. The results of different types of passive control on a 3U CubeSat are presented considering different environmental conditions in space and analysed with their possible advantages and disadvantages specific to the mission.

2. Methodology

To design the thermal control, the space thermal environment must be taken into consideration. Space temperature is generally considered close to 3K (-270 °C) [11]. The analysis cases that were considered for the design consist of two different types of orbit scenarios: Perihelion and Aphelion, defined as follows:

Aphelion: is the point where an orbiting planet is farthest from the Sun. In this case, the Aphelion of the Earth occurs in July, where the Earth is 152.1 million kilometres from the Sun. At this point, space heat fluxes such as solar radiation, have their lowest value. During Aphelion, the satellite will be at the coldest possible scenario.

Perihelion: is the point where an orbiting planet is closest to the Sun. In this case, the Perihelion of the Earth occurs in January, where the Earth is 147.1 million kilometres from the Sun. At this point, space heat fluxes, such as solar radiation, have their highest value.

Moreover, during orbit transit there will be three main sources of heat in space that could heat the satellite directly: solar radiation, albedo and Earth's IR. Figure 1 shows a representation of the satellite's orbit including all different interacting heat sources.

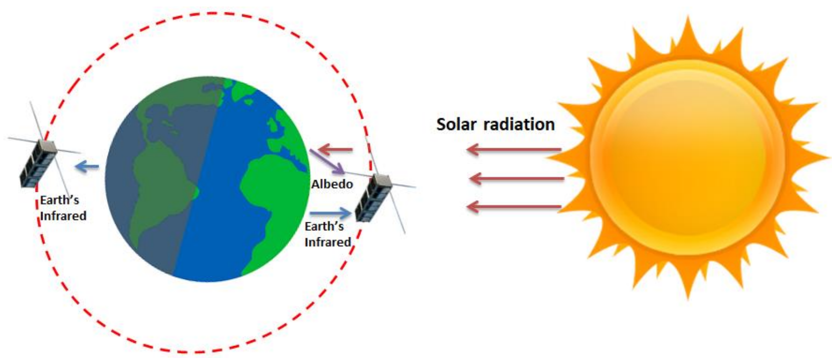

Figure 1: Heat sources in the orbit of a satellite around the Earth.

2.1 Heat Sources during Satellite's Orbiting

The energy that comes directly from the sun is the highest source of heat in space and it is also the most critical for a CubeSat while orbiting. The effect of the solar radiation in the spacecraft depends on the distance amongst the Earth, the Sun, and the satellite itself. For this reason, the solar radiation must be studied during Aphelion and Perihelion orbit points. Solar radiation can be expressed as [11]:

$$Q_s = \alpha G_s \mathrm{A} \cos\theta \quad (1)$$

Where Q_s is the heat radiated from the sun [W], α is the absorptivity of the surface, G_s is the solar flux in $\left[\frac{W}{m^2}\right]$ that is incident to a plane perpendicular to the vector's solar rays, A is the surface area $[m^2$] and θ is the angle between the solar vector and the normal vector of the surface. Additional to Sun's radiation, there is a second source of heat coming from Earth known as Albedo; Albedo is the amount of sunlight that is reflected from the Earth back to space. This heat source can be expressed as [11]:

$$Q_{alb} = G_s a \mathrm{A} F_{s-p} \alpha \; cos\varphi \quad (2)$$

Where Q_{alb} is the Albedo heat contribution to spacecraft $[W]$, G_s is the Solar flux in $\left[\frac{W}{m^2}\right]$, A corresponds to the area of the surface $[m^2]$, α is the solar absorptivity of the surface, F_{s-p} is the view factor from the surface to the Earth, a is the albedo factor and $-\pi/2 \leq \varphi \leq \pi/2$. The last heat source considered for the current study is the incident heat that is absorbed by Earth and then re-emitted as infrared energy to the space is known as Earth's IR. Not only the magnitude of heat depends on the local temperature of Earth's surface, but it also depends on the amount of clouds the surface has, as it decreases with more cloud coverage. The expression for this heat source is [8]:

$$Q_{IR} = \varepsilon q_{ir} A F_{s-p} \quad (3)$$

Where Q_{IR} is the Earth's heat contribution to spacecraft due to infrared radiation [W], ε is the infrared emissivity of the surface, q_{ir} is the Earth's infrared flux in $\left[\frac{W}{m^2}\right]$, A corresponds to the area of the surface $[m^2]$ and F_{s-p} is the view factor from the spacecraft to Earth.

2.2 Thermal Desktop and Nodalization

Thermal Desktop (TD) is a CAD-based geometric interface of the commercial on-orbit thermal analysis tool "Systems Improved Numerical Differencing Analyzer/Fluid Integrator" (SINDA/FLUINT) [12]. TD was used to model the Pakal 3U nanosatellite with its main internal components to validate the results of the analytical model and obtain more precise results that cover some assumptions made in the calculations. To perform the thermal analysis, the CubeSat was divided into a finite number of nodes that represent a region of the spacecraft with an associated thermal mass and temperature [13]. Once the spacecraft has been divided into a finite number of nodes, the heat transfer can be modelled. The total heat change with respect to time can be studied as the difference between the rate of heating entering each node and the rate of heating leaving each node. In this case, the heat inputs are the external heat fluxes (solar radiation, albedo and IR from the Earth) and the internal heat generation, and the heat output is the heat transferred as conduction and radiation between a given node and the rest of the model and the space. Let $Q_{ext,i}(t)$ be the sum of all external heating rates inputs over a node i and $Q_{int,i}(t)$ be the internal heat rate generation in that node and considering that the heat exchange of each node can be written as: $Q(t) = m_i C_i \Delta T_i$ where m_i is the mass of the node i $[Kg]$, C_i is the specific heat capacity of the node $\left[\frac{J}{kgK}\right]$ and T_i the temperature of node "i" $[K]$, then $\frac{dQ(t)}{dt}$for each node can be expressed as:

$$m_i C_i \frac{dT_i}{dt} = \dot{Q}_{int,i}(t) + \dot{Q}_{ext,i}(t) - \sum_{j=1}^{n} K_{ij}\left(T_i - T_j\right) - \sum_{j=1}^{n} R_{ij}\left(T_i^4 - T_j^4\right) \quad (4)$$

Where the term $\sum_{j=1}^{n} K_{ij}\left(T_i - T_j\right)$ represents conduction heat transfer between node "i" and the rest of the nodes in which K_{ij} is the total equivalent contact conductance matrix $\left[\frac{W}{K}\right]$ and the term $\sum_{j=1}^{n} R_{ij}\left(T_i^4 - T_j^4\right)$ represents radiation heat transfer between node "i" and the rest of the model in which R_{ij} is the radiation link coefficient matrix.

2.3 Assumptions, Boundary Conditions and Additional Considerations

Most of the geometry of the internal components was considered parallel rectangles (to simplify the calculation of the view factor). The payload was modelled as a small sphere inside the chamber without contact with the walls to analyse the total radiation effect over the sphere. Each internal electronic component had two internal heat dissipation values; the "maximum dissipated heat" when the satellite was in direct solar exposure and the "minimum dissipated heat" during the time of eclipse. For the external nodes, the solar panels and the structure walls were modelled as single nodes and their optical properties were weighted by considering the ratio of effective external area of walls and panels to total surface area of that face. The initial temperature of all the nodes of the model was considered 10°C and the temperature of space was taken as -270°C. The total analysis time was the equivalent of 10 orbits (55,450 seconds) since it was at this value when the simulation reached steady state. These boundary conditions were also considered for the model made on TD. Finally, the external heat conditions during Aphelion and Perihelion are shown in table 1. A thermal control design for a spacecraft is said to be "successful" if all of its components (especially mission-critical ones) maintain their temperatures in their operating range. The following operating temperatures were considered for the internal components (see Table 2). These temperatures were provided by the manufacturers used for the mission.

Table 1: Environmental conditions in Aphelion and Perihelion.

	Aphelion	Perihelion
Solar flux $\left[\frac{W}{m^2}\right]$	1,323	1,414
Albedo	0.25	0.5
Earth's IR $\left[\frac{W}{m^2}\right]$	220	275

Table 2: Operating temperature range of the main components

Components	**Operating temperatures**
UHF Transceiver	-35°C to 80°C
Cube Computer (Cube ADCS)	-10°C to 60°C
Batteries	-20°C to 60°C
EPS (Electrical Power System)	-20°C to 60°C
OBC (On Board computer)	0°C to 70°C
Solar Panels	-40°C to 85°C

The satellite and its components were divided into 18 nodes. The node's distributions are as follows: Nodes 1 to 6 correspond to solar panels & structure. Node 7 corresponds to the transceiver. Nodes 8 and 9 are the battery and EPS. Node 10 will represent the OBC. Nodes 11 to 16 are for the payload chamber. Node 17 will be for the payload sphere. And finally, node 18 corresponds to the cube's computer. Once all nodes are set in place, the whole geometry is meshed, and simulations are carried out. Figure 3 shows the nodal subdivision considered for the analytical model and the TD model after a simulation run. The results will be better explained in section 3.

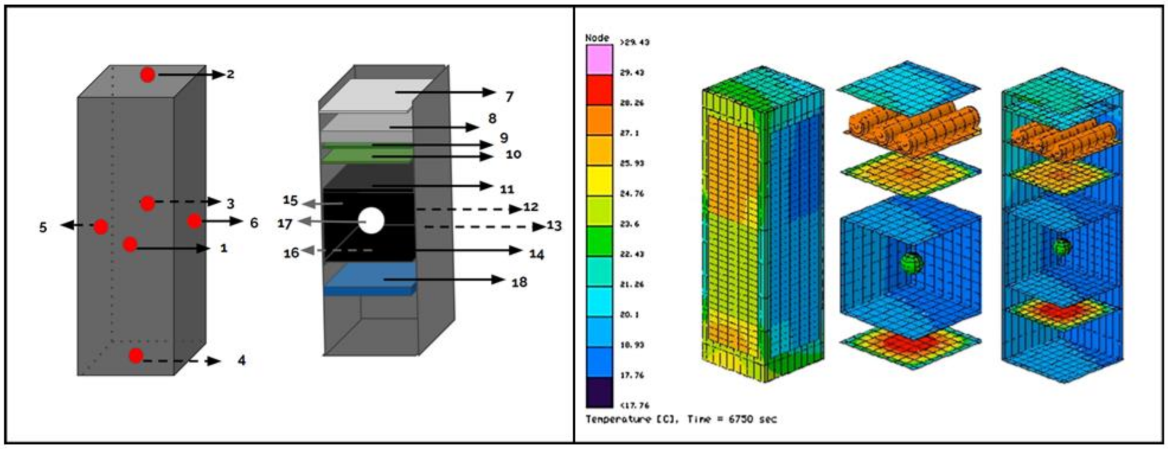

Figure 3: Nodal subdivision of the model (left) and Thermal Desktop model after a simulation run (right)

2.4 Passive Thermal Control

For the heat dissipation, two main types of Passive Thermal Control (PTC) components were considered: paints and insulation coatings. Each thermal control is defined on the current study as follows:

Thermal paints are a typical passive thermal control in combination with coatings that will cover the external effective area to change the impact of external radiation on the spacecraft. There are two different types of paints: white paints; which are generally used to reflect most of the incoming external radiation (and therefore have low solar absorptivity and high emissivity) and black paints that function as a "black body" because they have high solar absorptivity and high infrared emissivity values, these are typically used inside the spacecraft to facilitate heat transfer between components. An important aspect to consider when using thermal paints is the degradation of the surface paint due to ultraviolet radiation and atomic oxygen [13][14].

Multi-Layer Insulation *(MLI) Systems* consist of a series of layers (or a single layer in some cases) with reflective and absorbing characteristics that help the spacecraft to avoid any type of radiation. Compared to paints and coatings, insulation systems have a longer service life due to the type of material and optical properties. Furthermore, degradation caused by UV radiation and atomic oxygen is reduced relative to paints [13].

3. Results and Discussion

The results are divided into two parts: spacecraft temperature results during Perihelion (Figure 4) and results during Aphelion (Figure 5). Each part also considers three different situations, first the temperature ranges without thermal control in place, second, temperature results with passive thermal control I (thermal paints on external surfaces), and temperature results with passive thermal control II (a Kapton single layer insulation on external surfaces). In each section, the maximum and minimum temperature results of the components (listed in table 2) are shown as red markers in Figure 4 and Figure 5. The temperature ranges are presented as bar plots to facilitate comparison.

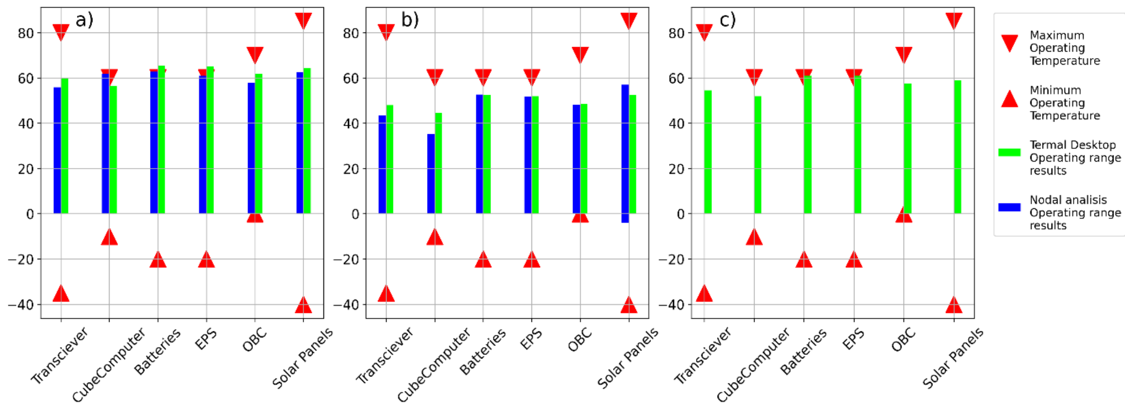

Figure 4: Thermal operating range results of the satellite components during Perihelion with different thermal control situations: a) Nonthermal control, b) Paintings, and c) Multi-layer insulation (MLI)

Figure 4 (a) shows that without thermal control during perihelion, there would be 4 components that exceed or are close to reach their maximum operating temperature which are Batteries, EPS, OBC and Cube ADCS. Following with the analysis, multiple surface paint options were studied and tested during the simulations. The main goal was to have all the components in their operating temperature range in the two worst scenarios, the paints presented in the current study are white paint A276 and black paint Z306 whose parameters are: Solar absorptivity of 0.28 and infrared emissivity of 0.88 for paint A276; and Solar absorptivity of 0.92 and infrared emissivity of 0.89 for paint Z306. The black paint was implemented on the inside of the structure's faces and the white on the + X and -Y faces (nodes 6 and 1 respectively) which are the ones that receive the most solar radiation, in order to dissipate and reflect heat from external sources. No paintings were implemented on the rest of the faces of the spacecraft. By implementing the use of paints, all the components are at their operating temperature range (Figure 4 (b)). Furthermore, this PTC configuration may be the most appropriate for the spacecraft. The external white paint A276 has an adequate degradation behaviour and response due to UV radiation in the presence of atomic oxygen for a long period of time [15]. Since most CubeSat missions are planned to operate for a long time (particularly in this case, the satellite is planned to operate for up to one year), degradation due to ultraviolet radiation will not be a major factor.

An alternative to paints is the implementation of layers of insulation. Their purpose is to isolate the internal components from the external environment and hence to make them meet their temperature requirements. The current study considers the implementation of a single layer insulation. For this thermal control, a 0.5 mil aluminized Kapton layer was chosen, which protects against short term atomic oxygen and long term ultraviolet radiation. Insulation was applied over all the external surfaces of the spacecraft. Its implementation was carried out only in the TD model since the software presents facilities to do that compared to a nodal analysis. The numerical results can be observed in Figure 4 (c). It is possible to observe that when MLI is implemented at perihelion, there are two components (batteries and EPS) that get close to exceed their maximum operating temperature range. However, all other components meet their operational requirements and overall, none of the components are over their minimum operating temperature. As perihelion is only one possible scenario, aphelion must be analysed first to test the effectiveness of the proposed PTC configurations.

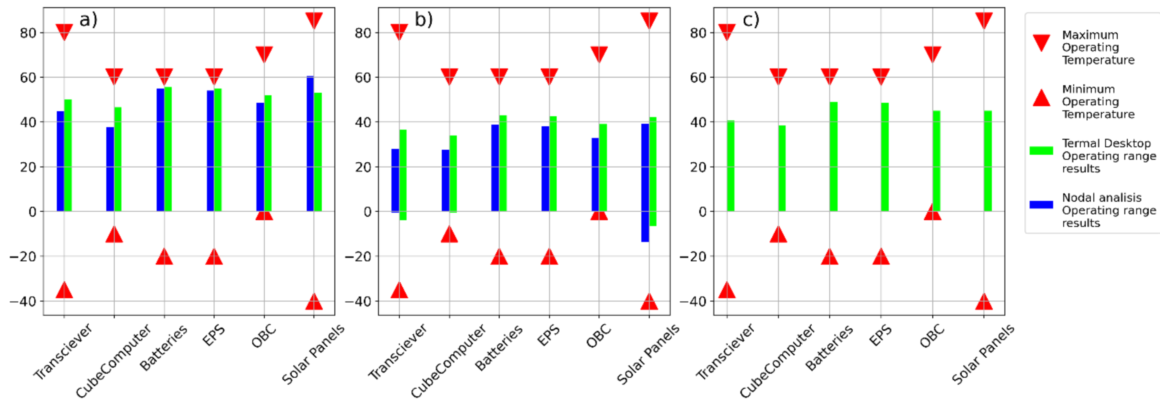

Figure 5: Thermal operating range results of the satellite components during Aphelion with different thermal control situations: a) Nonthermal control, b) Paintings, and c) Multi-layer insulation (MLI)

Likewise Figure 4, the numerical and nodal results during aphelion can be observed in Figure 5. Figure 5 (a) shows the effects of not using any thermal control during Aphelion. From the results, it is possible to observe that there would be 2 components that are close to reaching their maximum operating temperature. These components are batteries and EPS. Since in both scenarios without a thermal control (i.e., Aphelion and Perihelion) these components are close or even exceed their operating temperature, it can be concluded that a suitable thermal control must be implemented. Figure 5 (b) shows the resulting temperature ranges with the implementation of paintings. It is possible to observe that no component exceed its operating temperature range. Yet, the minimum temperature of the OBC is close to the minimum operation range value, so this PTC setting may pose a risk in this scenario during the mission. This must be considered while further testing to avoid any inconvenience. Finally, Figure 5 (c) shows that when MLI is implemented in the aphelion, all components are in their operating temperature range and since the spacecraft is isolated from the external environment, the minimum operating temperature requirement is not a big issue. Thus, this PTC setting is a suitable alternative for this scenario. Yet, Figure 4 (c), showed that there would be a risk if MLI is implemented because in hot cases such as perihelion, some components could exceed their maximum operating temperature. One possible alternative for the satellite's thermal management control would be the implementation of both approaches. Yet, this will need further testing.

4. Conclusions

The current case study was performed to achieve a heat balance in the critical cases of the orbit and thus design a thermal control model comparing different approaches. Using a numerical simulation software to model the satellite allowed to consider important geometric aspects and distribution of materials of the model and thus obtain acceptable and accurate results. As shown in section 3, thermal paints present a risk with the minimum operating temperature in cold cases such as aphelion. Meanwhile, MLI does with the maximum operating temperature in hot cases such as perihelion. After analysing both scenarios, it is proposed that both MLI coatings and paints are implemented. The possible solution could be used on different faces of the CubeSat to heat it sufficiently during the orbit shadow time and dissipate most of the heat during the time with direct solar exposure. This approach will also help for the internal components to meet their operating temperature ranges, especially when not enough power is available to consider a thermal heater during cold cases.

References

[1] A. G. C. Guerra, D. Nodar-López, and R. Tubó-Pardavila, "Thermal analysis of the electronics of a CubeSat mission," *Appl. Therm. Eng.*, 2018, [Online]. Available: http://arxiv.org/abs/1803.10468.

[2] V. M. Chau and H. B. Vo, "Structural Dynamics Analysis of 3-U CubeSat," *Appl. Mech. Mater.*, vol. 894, pp. 164–170, 2019, doi: 10.4028/www.scientific.net/amm.894.164.

[3] N. Athirah, M. Afendi, K. Hafizan, N. A. M. Amin, and M. S. A. Majid, "Stress and thermal analysis of cubesat structure," *Appl. Mech. Mater.*, vol. 554, pp. 426–430, 2014, doi: 10.4028/www.scientific.net/AMM.554.426.

[4] P. Wallace, A. Kalapura, and S. I. Kim, "Thermal modelling and analysis of a cube satellite, EIRSAT-1: Steady analysis," in *2018 AIAA SPACE and Astronautics Forum and Exposition*, 2018, doi: 10.2514/6.2018-5285.

[5] E. Escobar, M. Diaz, and J. C. Zagal, "Evolutionary design of a satellite thermal control system: Real experiments for a CubeSat mission," *Appl. Therm. Eng.*, vol. 105, pp. 490–500, 2016, doi: 10.1016/j.applthermaleng.2016.03.024.

[6] N. A. and S. Administration, "NASA State of the Art of Small Spacecraft Technology," *Thermal Control*, 2020. https://www.nasa.gov/smallsat-institute/sst-soa-2020/thermal-control.

[7] M. Bonnici, P. Mollicone, M. Fenech, and M. Anthony, "Analytical and numerical models for thermal related design of a new pico- satellite," *Appl. Therm. Eng.*, vol. 159, no. October 2018, p. 113908, 2019, doi: 10.1016/j.applthermaleng.2019.113908.

[8] S. Chandrashekar and T. Karlsson, "Thermal Analysis and Control of MIST CubeSat," KTH Royal Institute of Technology Master, 2017.

[9] J. Claricoats and S. M. Dakka, "Design of power, propulsion, and thermal sub-systems for a 3U CubeSat measuring Earth's radiation imbalance," *Aerospace*, vol. 5, no. 63, 2018, doi: 10.3390/aerospace5020063.

[10] S. Rossi and A. Ivanov, "Thermal model for a CubeSat: A simple and easy model from Swisscube's thermla flicht data," in *International Astronautical Conference*, 2013.

[11] J. Meseguer, I. Pérez-Grande, and A. Sanz-Andrés, *Spacecraft thermal control*, 1st Ed. Cambridge, UK: Woodhead Publishing, 2012.

[12] S. J. Kang and H. U. Oh, "On-orbit thermal design and validation of 1 U standardized CubeSat of STEP cube lab," *Int. J. Aerosp. Eng.*, vol. 2016, 2016, doi: 10.1155/2016/4213189.

[13] D. G. Gilmore, *Spacecraft Thermal Control Handbook Volume I: Fundamental Technologies*, 1st Ed. California, USA: The Aerospace Press, 2002.

[14] R. Karam, *Satellite Thermal Control for Systems Engineers*, 1st Ed. Cambridge, UK: American Institute of Aeronautics and Astronautics, Inc., 1998.

[15] A. Anvari, F. Farhani, and K. S. Niaki, "Comparative study on space qualified paints used for thermal control of a small satellite," *Iran. J. Chem. Eng.*, vol. 6, no. 2, 2009.