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*Dedicated to my loving parents
and
my supportive sisters and brother*

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NOMENCLATURE

A	Area	m ²
Across	cross section Area	m ²
a	Semi-major axis	km
cp	Specific heat capacity	J/kg.K
e	Eccentricity	
F_{ij}	View factor	
F_a	Albedo view factor	
F_e	Eclipse function	
fe	Eclipse fraction	
G	Universal gravitational constant 6.67×10^{-11}	m ³ /kg.s ²
H	Altitude of orbit	km
i	orbital inclination	deg
M	Mass (for Earth)	kg
m	Mass of spacecraft	kg
T_0	Orbital Period	s
T_e	Period of eclipse	s
A_f	albedo constant	
R	Radius of the Earth	km
S	Heat flux of solar radiation	
T	Temperature	°K
t	Time	s
α	Absorptivity of material	
β	Beta angle	deg
ε	Emissivity	
θ	Angle of incidence between surface and incident radiation	deg
σ	Stefann-Boltzmann constant	

v	True anomaly	deg
\emptyset	Orbit angle	deg
\emptyset_{es}	Angle at which eclipse begins	deg
\emptyset_{ee}	Angle at which eclipse ends	deg
ω	Argument of perigee	deg
Ω	Right ascension of the ascending node	deg
h	function of relative orbit radius	
F_s	step function for solar input	
\dot{Q}_{in}	input heat flux	w
\dot{Q}_{out}	output heat flux	w
\dot{Q}_a	albedo heat flux	w
\dot{Q}_{iR}	infrared heat flux	w
\dot{Q}_s	direct solar heat flux	w
RAAN	Right Ascension of the Ascending Node	deg
k	material conductivity	w/m.K

ABBREVIATIONS

NASA	National Aeronautics and Space Administration
ESA	European Space Agency
ISS	International Space Station
STK	Sytem Tool Kit software
MEO	Medium earth orbit
SSO	Sun-Synchronous Orbit
GTO	Geostationary Transfer Orbit
GEO	Geostationary Orbit
LEO	Low earth orbit
HEO	Highly elliptical orbit
TCS	Thermal control system
Al	Aluminium
SEET	Space Environment and Effects Tool

ABSTRACT

The document studies satellite's thermal control systems and thermal analysis of a cubesat in a low earth orbit.

The thermal control subsystem (TCS) maintains the CubeSat to be within the allowable operational and survival temperature limits.

In this document, the thermal analysis of the cubesat is done in two states: a steady state and a transient state, the temperatures were determined for the cubeSat with no thermal control. The resulting temperatures were then compared to typical temperature limits of the components to determine if they could survive and operate in some specific thermal environment. Two cases were investigated a hot case where the cubesat gain the maximum heat while exposing to direct sunlight and a cold case when the cubesat is in the eclipse.

Thermal analysis are very important before launching the cubesat in its orbit, it helps to approximate the changes of temperature that may the cubesat experience. In this work STK.11 software and Matlab R2015a also Microsoft EXCEL 2010 are used in the thermal analysis .

To ensure a successful mission and long lifetime to the cubesat, it is necessary within the scope of the study to design a Thermal Control System (TCS). This work summarizes the orbital analysis and thermal design and thermal analysis of a 3U cubesat launched in LEO orbit.

The methodology used in this document: single node model in the first part of the simulation, then a multi-node model, six surfaces of satellite taken as six nodes and all of the surfaces have different heat sources therefore different temperature in the second part of the simulation.

Resumé

Ce document étudier le système de contrôle thermique et l'analyse thermique d'un cubesat en orbite basse.

Le système de contrôle thermique assure que le cubesat fonctionne dans des limites de température opérationnelles.

Dans ce document l'analyse thermique ce fait dans deux états: un état stable, et un état transitoire, les températures ont été déterminées pour le cubesat sans contrôle thermique. Ensuite les températures résultantes ont été comparées avec les limites de températures des composants pour déterminer s'ils pouvaient survivre et fonctionner dans ce environnement . deux cas ont été étudiés , un cas chaud ça c'est quand le cubesat gagne le maximum de chaleur lorsqu'il est exposé à la lumière directe du soleil, et un cas froid lorsque le cubesat est dans la phase d'éclipse.

L'analyse thermique est très importante avant de lancer le cubesat dans son orbite, cette étape aide à rapprocher les changements de température que le cubesat peut être expérimenté. Dans ce travail, pour faire l'analyse thermique du cubesat , on a utilisé STK.11 software , Matlab R2015a et aussi Microsoft EXCEL 2010 .

Pour assurer une mission réussie et une longue durée de vie du cubesat, il est nécessaire dans le cadre de l'étude de concevoir un système de contrôle thermique (SCT). Ce travail résume l'analyse orbitale, analyse thermique et le système de contrôle thermique d'un 3U cubesat en orbite basse .

La methodologie utilisée dans ce document: modèle a noeud unique dans la première partie de la simulation, puis multi-noeuds model, dans ce model le cubesat a six faces donc il est converti chaque face en noeud , alors on a six noeuds, et toutes les surfaces ont des chaleurs différentes donc les températures sont différentes , dans la deuxième partie de la simulation.

ملخص

هذه المذكرة تدرس نظام التحكم الحراري ودراسة التبادل الحراري لقمر صناعي مكعب يدور في مدار منخفض.

نظام التحكم الحراري يعمل على ان يكون عمل القمر الصناعي المكعب في مجال حدود الحرارة المسموح و يبقى فيها التشغيل مستمرا.

في هذه المذكرة يتم دراسة التبادل الحراري للقمر الصناعي المكعب في حالتين: حالة مستقرة و حالة التغير مع الزمن ، و تم اعتبار ان القمر الصناعي المكعب يدور دون أي نوع من أنواع التحكم الحراري. ثم تمت مقارنة درجات الحرارة المحصل عليها مع درجات الحرارة للمكونات الداخلية للقمر الصناعي المكعب حتى نرى اذا ممكن لهذه المكونات العمل ضمن هذا المجال، و كانت الإجابة انه لا يمكن ذلك و هذا ما يستلزم استعمال نظام التحكم الحراري. المجال الذي نتحدث عليه تم تحديده بحالتين حرجيتين التي يمكن للقمر الصناعي المكعب الوصول اليها في هذا المدار : حالة الحرارة الشديدة حيث يستقبل القمر الصناعي اكبر كمية من الإشعاع الحرارة الممتصة ، نظرا للتعرض المباشر لأشعة الشمس، و حالة أخرى هي حالة البرودة الشديدة في غياب تام لأشعة الشمس أي خلال ظل الكرة الأرضية .

دراسة التبادل الحراري للقمر الصناعي المكعب خطوة مهمة جدا قبل اطلاق القمر الصناعي لمداره، فهذا يساعد على تقريب التغيرات التي تحدث لحرارة القمر الصناعي المكعب، في دراستنا استعملنا برنامج STK11 و ماطلاب MATLAB R2015a ، كما استعملنا برنامج EXCEL 2010 في بعض الحسابات الضرورية .

توصلنا بعد هذه الدراسة انه لضمان مهمة ناجحة و مدى حياة القمر الصناعي المكعب أطول ما يكون، من الضروري تصميم نظام حراري مناسب، و يجدر الإشارة انه استعملنا طريقتين : الأولى اعتبار القمر الصناعي المكعب نقطة مادية و الثانية اعتبار كل وجه من اجه القمر الصناعي نقطة و اعتبار التبادل الحراري الحاصل بين هذه النقط.

INTRODUCTION

Cubesats today are becoming the interest because of their low cost and fast producing, one of the problems that it may lead to the failure of the mission is the space environment, which is pretty sure is different from the ground environment, and the thermal heating fluxes in the space, precisely in the low earth orbit are: direct solar flux, albedo heat flux and finally the infrared heat flux.

In this thesis we study these entering and leaving fluxes in four chapters, it starts with a general chapter, where it contains generalities about satellites and their classification plus a brief history, then the second chapter it contains some basics of the science of heat transfer and the space environment, after that the thermal control system and how to do the thermal analysis and the orbital analysis plus the hardware used in satellites in chapter three, results of thermal analysis in chapter 4, with discussion.

Spacecraft thermal control is a very important step to mission success. The process of the thermal control for a spacecraft involves managing the energy entering and leaving the satellite to ensure that the components are remaining within an acceptable temperature range which leads to longer working lifetime [1]. This is substantiated during the development process by analysis, similarity studies, and testing.

We used in this thesis two methods of resolution of the thermal problem, one node which we consider the cubesat a lumped mass with an isothermal temperature, then we used a multi node model, which we consider each face of the cubesat as a node, so six nodes for our study, also the conduction between nodes is calculated, also calculations are for the two worst cases (over heating and under cooling).

The thermal tests are very expensive, and need a lot of time, personal experience and exact input data, so numerical simulations can help for estimating the real phenomenon, but also these simulations have a lot of errors. Now in general and from what we studied in this thesis we can say that the branch of thermal analysis is very large, and any change in the inputs or initial states can affect the temperature of the cubesat, for example the type of material, the beta angle, view factors ... all that can affect the temperature. However additional complexity can also introduce additional errors to the model.

CHAPTER 1: GENERALITIES ABOUT SATELLITES

1.1 Introduction

Spacecraft thermal control is integral to mission success. The process of thermal control for a spacecraft involves managing the energy entering and leaving the spacecraft to ensure that the components of the spacecraft remain within an acceptable temperature range.

In this chapter, we find general information about satellites and their orbits before entering in the thermal analysis.

1.2 Definition of « satellite »

A satellite is a moon, planet or space machine that orbits a planet or star. For example, Earth is a satellite because it orbits the sun, the same with the moon; it is a satellite because it orbits the Earth. Usually, the word "satellite" refers to a machine that is launched into space and moves around Earth or another body in space. [1]

So we distinguish two types of satellites natural and artificial and both have the movement of orbiting another body like planets or stars, in our study we focus on artificial satellites that made by humans and orbiting our earth, and we have different types of satellites depending on their application, because every mission has its own orbit and payload. [2]

CubeSats are one of the types of small satellites, they have gained much interest because of their small size and defined standards allow it to be a multi-purpose and low cost platform, flexible to accommodate different payloads and the ability to perform several missions. [3]

1.3 History

The story of artificial satellites begins well before the first launch of a satellite, Isaac Newton was the first to imagine how a man-made satellite could orbit Earth in (1687), He had the idea of a cannon firing a ball from the top of a very high mountain that if fired with enough power would describe the same curve over and over, it's not a concept to put a satellite in orbit but it was the very beginning at that time. [3]

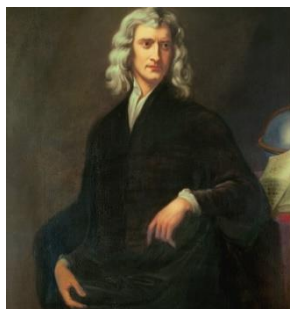


Figure 1 : Isaac newton [4]

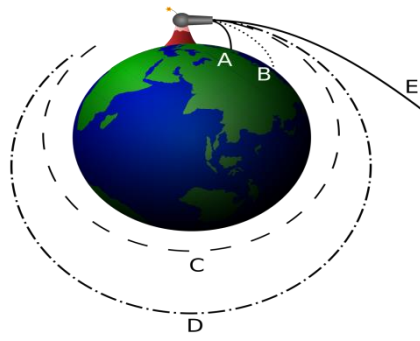


Figure 2 : cannon firing a ball from the top of a very high mountain [5]

After that Tsiolkovsky, in 1903, showed how satellites could be launched and calculated the required velocity in order to put a satellite into orbit [3].



Figure 3 : Tsiolkovsky [6]

Then it was the World Wars and it led to a period of high technological development, which started a race between the US and the Soviet Union to launch the first satellite. Then the Soviets won by launching Sputnik I in 1957 made of aluminum and put in low earth orbit and for 21 days mission duration , and leading to the begging of Space Age. Since then, almost 8500 satellites were launched by more than 40 countries [7].

1.4 Cubesats

Satellites can be classified based on their applications, orbits, mass etc. The satellites with less than 500 kg mass are called as small satellites [8]. For our topic we focus on the study of cubesats.

Table 1: Satellite classification by weight [8]

Quantity	Platform mass, kg
Large spacecraft	≥ 1000
Medium spacecraft	≥ 500 and < 1000
Small spacecraft	< 500
Minispacecraft	≥ 100 and < 500
Microspacecraft	≥ 10 and < 100
Nanospacecraft	≥ 1 and < 10
Picospacecraft	≥ 0.1 and < 1
Femtospacecraft	≤ 0.1

Cubesat is subset of nanosatellite category. The cubesat standard and configuration were developed by California Polytechnic State University and Stanford University as university education program satellite. [8]

As we said before a small satellite is generally considered to be any satellite that weighs less than 500 kg and to call the name cubesat it must has some control factors such as shape, size, and weight. [8]

CubeSats come in several sizes, which are based on the standard CubeSat “unit” referred to as a 1U. A 1U CubeSat is a 10 cm cube with a mass of approximately 1 to 1.33 kg.

- 1U Standard dimensions : 10 cm × 10 cm × 11 cm
- 3U Standard dimensions : 10 cm × 10 cm × 34 cm
- We can see figure 5 for other sizes
- We have to note that larger sizes have become popular, such as the 1.5U, 2U, 3U, 6U, and 12U.

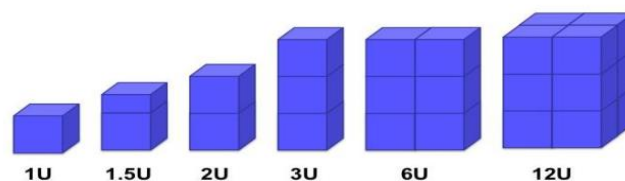


Figure 4: cubesats sizes with unit [9]

This type of small satellites provides affordable access to space for small companies, research institutes and universities. Their modular design means that subsystems are available and can be stacked together according to the needs of the mission. This allows CubeSat projects to be ready for flight extremely quickly and typically within one or two years, And commonly used in low Earth orbit, such as remote sensing and communications [10].

Over the past ten years, space agencies (ESA and NASA) have been putting more time and effort into its CubeSat activities. Roger Walker, head of ESA's CubeSat efforts (European Space Agency (ESA)), explains that «ESA is harnessing CubeSats as a fast, cheap method of testing promising European technologies in orbit ».ESA's first CubeSat project was GomX-3, a mission to demonstrate various signal monitoring technologies, which was deployed from the International space station (ISS) in October 2015,also NASA (National Aeronautics and Space Administration) launched its first pair of CubeSats designed for deep space Marco In 2018 [10] .

1.5 Satellite subsystems

The satellites in general are composed of two main parts: platform and the payload, as explained in the following figure .

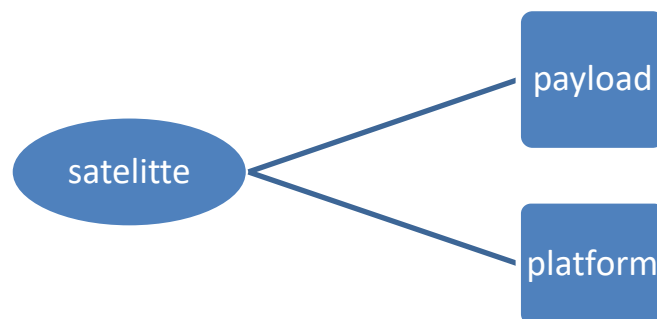


Figure 5: satellite components [11]

1.5.1 Payload

The mission of the satellite is closely related to the type of the payload on board: it is the payload of a satellite that defines its mission. It largely determines the cost, complexity and effectiveness of assignment [11].

Some payload examples [11]:

- Antennas, transponders and amplifiers of a satellite telecommunications
- Telescopes for an astronomy satellite
- Scientific instrumentation for a space probe
- The spacecraft for a crewed mission

1.5.2 Platform

It includes the equipment necessary for the proper functioning of the satellite during the expected lifetime: It ensures the survival of the satellite, Satellites systems are divided in several subsystems as can be seen below [2] [12]:

1. The mechanical structural subsystem, provides the framework for mounting other subsystems of the satellite and also an interface between the satellite and the launch vehicle
2. The propulsion subsystem is used to provide the thrusts required to impart the necessary velocity changes for the maneuvering during the lifetime of the satellite.
3. Thermal control subsystem: it is to maintain the satellite platform within its operating temperature limits for the type of equipment on board the satellite. It also ensures the desirable temperature distribution throughout the satellite structure.
4. Power supply subsystem: is to collect the solar energy, transform it to electrical power with the help of arrays of solar cells and distribute electrical power to other components and subsystems of the satellite. In addition, the satellite also has batteries, which provide standby electrical power during eclipse periods, during other emergency situations and also during the launch phase of the satellite when the solar arrays are not yet functional.
5. Telemetry, tracking and command (TT&C) subsystem: The tracking part determines the position of the spacecraft and follows its travel using angle, range and velocity information. The telemetry part gathers information on the health of various subsystems of the satellite encodes this information and then transmits the same. The command element receives and executes remote control commands to effect changes to the platform functions, configuration, position and velocity.
6. Attitude and orbit control subsystem: ensure that the satellite is in the correct location in space. It also provides attitude control, to prevent the satellite from tumbling in space and also to ensure that the antennae remain pointed at a fixed point on the Earth's surface.
7. Antenna subsystem: Antennas are used for both receiving signals from ground stations as well as for transmitting signals towards them.

1.6 Orbit and trajectory

Trajectory is a path traced by a moving body, while an orbit is a trajectory that is periodically repeated. The motion of different planets of the solar system around the sun and the motion of artificial satellites around Earth are examples of orbital motion [2].

1.6.1 Kepler's laws [2]

Johannes Kepler (1571-1630) discovered the laws of orbital motion, now called Kepler's laws, it describe how planetary bodies orbit the sun.

1st law: The orbit of an orbiting celestial body (i.e. Planet/comet/satellite) around another reference celestial body (i.e. Sun) is a conic section (ellipse, parabola, and hyperbola) with the reference celestial body's center of mass at one focus.

2nd law: A line joining an orbiting celestial body and the reference celestial body sweeps out equal areas in equal intervals of time.

3rd law: The Square of the period of a planet is proportional to the cube of its mean distance from the sun.

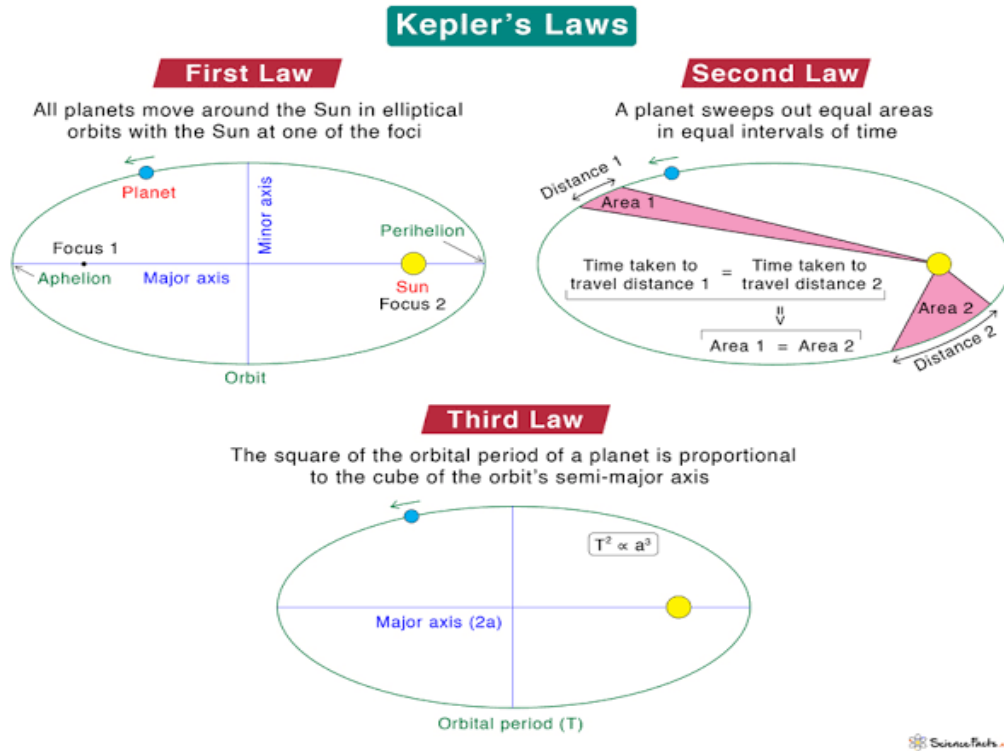


Figure 6: keplerian laws [2]

1.6.2 Newton's laws [13]

Isaac Newton (1642-1727) published his 'Principia Mathematical' in 1687. He described the universal gravitation and the three laws of motion.

Newton showed that the motions of objects on Earth and of celestial bodies are governed by the same set of natural laws by demonstrating the consistency between Kepler's laws of planetary motion and his theory of gravitation.

- 1st law: Everybody continues in its state of rest or of uniform motion in a straight line unless it is compelled to change that state by forces impressed upon it.
- 2nd law: The rate of change of momentum is proportional to the force impressed and is in the same direction as that force.
- 3rd law: To every action there is always opposed an equal reaction.

He also formulated his Law of Universal Gravitation by stating that any two bodies attract one another with a force proportional to the product of their masses and inversely proportional to the square of the distance between them.

$$F_g = -\frac{G M m}{r^2} \cdot \frac{r}{r} \text{ (N)} \quad (1.1)$$

F_g is the force on mass m due to mass M and r is the vector from M to m . The universal gravitational constant $G = 6.67428 \cdot 10^{-11} \frac{m^3}{kg \cdot s^2}$.

1.6.3 Orbit Determination [14]

Orbits are defined by orbital elements; the problem of defining the orbit is 6 degrees of freedom so there are exactly 6 parameters. These correspond to the three spatial dimensions which define position, plus the velocity in each of these dimensions. These can be described as orbital state vectors, but this is often an inconvenient way to represent an orbit, which is why Keplerian elements are commonly used instead.

The ones which describe the shape and size of the orbit are:

1. Semi major axis (a): Distance between the geometric centers of the orbital ellipse with the perigee, passing through the focal point where the center of mass resides.
2. Eccentricity (e): Shape of the ellipse, describing how flattened it is compared with a circle.

The ones which describe the orientation of the orbit plan:

1. Inclination (i) - Vertical tilt of the ellipse with respect to the reference plane, measured at the ascending node (where the orbit passes upward through the reference plane).
2. Right ascension of the ascending node (Ω) (RAAN) : Represents the angle between the vernal equinox and the point where the orbit crosses the equatorial plane (going north)

The ones which determine the position of the spatial body:

1. Argument of perapsis (ω): Defines the orientation of the ellipse (in which direction it is flattened compared to a circle) in the orbital plane, as an angle measured from the ascending node to the perigee.
2. True anomaly (ν): defines the angle in the plane of the ellipse, between perigee and the position of the orbiting object at any given time.

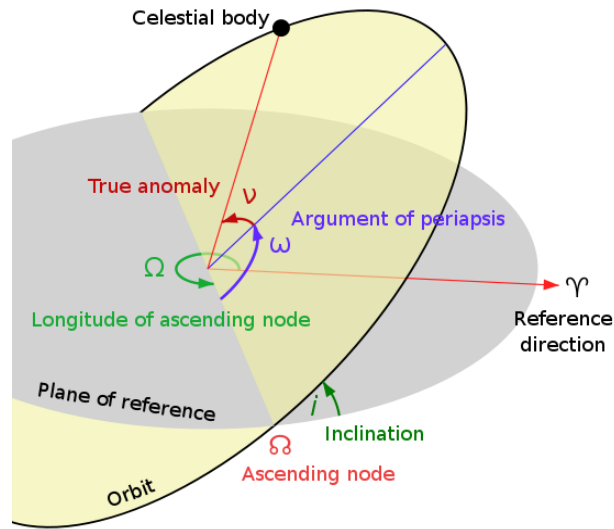


Figure 7: orbital elements [14]

1.6.4 Types of orbits [15]

Classification of satellites is not only based on their payload or sizes, they are also classified according to their orbits shapes and altitudes (distance from the Earth’s surface), the type of orbit affects the coverage and also the speed of the satellite.

We have 5 types of orbits according to their altitudes as showed in the table below:

Table 2: types of orbits and their orbits [15]

	LEO	SSO	MEO	GTO	GEO
Altitude (km)	160-1500	600-800	5000-20,000	Up to 35,786	35,786
Period (min)	Up to 128	96-100	720	630	436

1. Low Earth Orbit (LEO) Satellites: the satellite orbiting in an altitude between 160–1,500 kilometers above the Earth’s surface, with a short orbital period, between 90 and 120 minutes, meaning they can travel around the planet up to 16 times a day. This makes them particularly well-suited to all types of remote sensing, high-resolution earth observation, and scientific research, as data can be acquired and transmitted rapidly, this proximity to the Earth; it makes the satellite have a smaller coverage area than other satellite types. Often, groups of LEO spacecraft, known as satellite constellations, launched together so they can cover huge areas simultaneously by working together.
2. Sun-Synchronous Orbit (SSO) Satellites: those satellites go from north to south across the Polar Regions at an altitude of 600 to 800 km above the Earth. The orbital inclination and altitude of SSO spacecraft are calibrated so that they always cross any given location at precisely the same local solar time. Thus, the lighting conditions are consistent for imaging, making this type of satellite ideal for earth observation and environmental monitoring. But because of their lower orbital altitude, SSO

type of spacecraft can only cover a smaller region at once and need more machines to do so continuously.

3. Medium Earth Orbit (MEO) Satellites: typically at an altitude of about 5,000 to 20,000 kilometers. It is the orbit of Positioning and navigation services, like GPS. Recently, high-throughput satellite (HTS) MEO constellations have been put into operation to enable low-latency data communication to service providers, commercial and government organizations. With their longer orbital period (usually between 2 and 12 hours), this type of satellites offer a happy medium between coverage area and data transmission rates.
4. Geostationary Transfer Orbit (GTO) Satellites: Spacecraft are not always placed directly into their orbit when launched from Earth into space such as Falcon 9. Rockets carrying payload to GEO drop it off at transfer orbits, which are halfway points on the path to its final position. Then a satellite's engine fires to reach its destination orbit and adjust its inclination. This shortcut allows the machine to reach geostationary orbit with minimal resources.
5. Geostationary Orbit (GEO) Satellites: positioned 35,786 kilometers above Earth's surface, precisely over the equator. Three satellites orbiting in GEO can give nearly worldwide coverage thanks to the huge area they cover on Earth. GEO satellite appear motionless from the ground because their orbital period is identical to Earth's rotation = 23 hours, 56 minutes, and 4 seconds. This allows a terrestrial antenna to always point toward the same device in space. That's why this type of satellites is perfect for communication services and meteorology.

There are other types but less common include the highly elliptical orbit (HEO), polar orbit, and Lagrange point (L-point).

CHAPTER 2: HEAT TRANSFER AND SPACE ENVIRONMENT

2.1 Introduction

It is obvious that the satellite will experience different heating fluxes caused by different heat transfer modes during its lifetime, and it is very important to know the basics of heat transfer concepts before studying the space environment in orbits plus the thermal analysis and control of satellites in that environment[16].

After we have seen in the previous chapter what is a satellite and its orbit, now in this chapter we find the basics in heat transfer, and then we enter in to the space environment.

2.2 Thermodynamics and heat transfer [17]

The energy transfer is always from the higher temperature body or environment to the lower temperature one, and the energy transfer stops when the two bodies (environments) reach the same temperature. So we can say that heat is the form of energy that can be transferred from one system to another as a result of temperature difference. The science that deals with the determination of the rates of such energy transfers is heat transfer.

The thermodynamic analysis allow us to determine heat transfer experienced for any system at any process because it is concerned with the transfer of heat in a system from one equilibrium state to another without giving an indication of how long this process takes. A thermodynamic analysis simply tells us how much heat must be transferred to realize a specified change of state to satisfy the conservation of energy principle.

2.2.1 First law of thermodynamics

The first law of thermodynamics, also known as the conservation of energy principle, «the total energy in a system remains constant, although it may be converted from one form to another » Another common phrasing in books is that « energy can neither be created nor destroyed , only altered in form (in a "closed system") » [17].

We can also say that energy will always be conserved. If heat energy is provided to the system by the surrounding then system may provide the work energy to the surrounding and similarly if work is being done upon the system then system may deliver the energy in terms of heat to the surrounding i.e. heat energy transfer and work energy transfer across the system boundaries will be zero. In other words, the total energy of the universe remains constant, the following equation (2-1) is a mathematical writing of the first law of thermodynamics [17]:

$$\Delta U = Q - W \quad (w) \quad (2-1)$$

To understand this equation we can take the example of a gas enclosed in cylinder with a movable piston as showed in (figure 8) , the walls of the cylinder is separation the gas from the outside, the movable piston do work by expanding against the force holding the piston, so here the gas does work W as it expands, and /or absorbs heat Q from the surroundings through walls, then this corresponds to a net flow of energy $W-Q$ across the boundary (walls).and in order to conserve the total energy U , the rate must be a counterbalancing change as showed in the equation (2-1) [17].

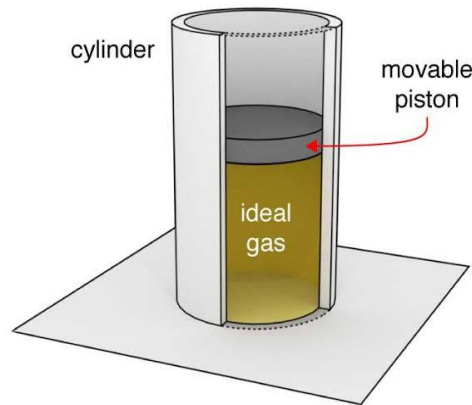


Figure 8: gas enclosed in cylinder with a movable piston [17].

Another explanation of the previous example is in the (figure 9) where: Q heat energy enters the system from surrounding and work energy W leaves the system [17].

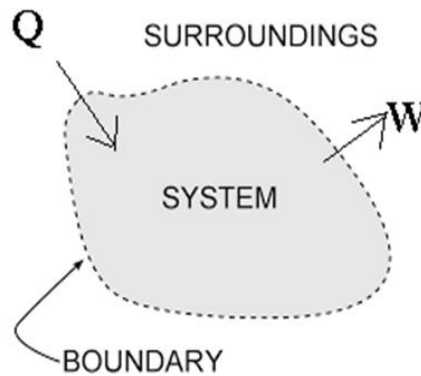


Figure 9: first law of the thermodynamic for closed system [17]

If the state of the system does not change during the process, so it is steady which mean: energy and mass in = energy and mass out [17].

$$\dot{Q}_{in} = \dot{Q}_{out} \quad (w) \quad (2-2)$$

heat transfer can be done without work process, and In this case In the absence of any work interactions, the change in the energy content of a closed system is equal to the net heat transfer [17] :

$$\frac{dE}{dt} = m c \frac{dT}{dt} \quad (2-3)$$

Where m is the mass, and c is the specific heat and dT/dt is the change of temperature.

2.3 modes of heat transfer [17]

Heat transfer occurs in three modes:

- Convection
- Conduction
- Radiation

We illustrate those modes with (figure 10) to get a brief idea about those modes of heat transfer.



Figure 10: modes of heat transfer [18]

2.3.1 Convection

Convection is one of the modes of heat transfer and we can see it in our daily life, it can be defined as: « The heat transfer between a solid surface and a surrounding fluid in motion at different temperatures».

For a better explanation of this mode of heat transfer, we can take a hot cup of water and leave it on a table, its gets cold at some point; we illustrate this example in the (figure11) [19].

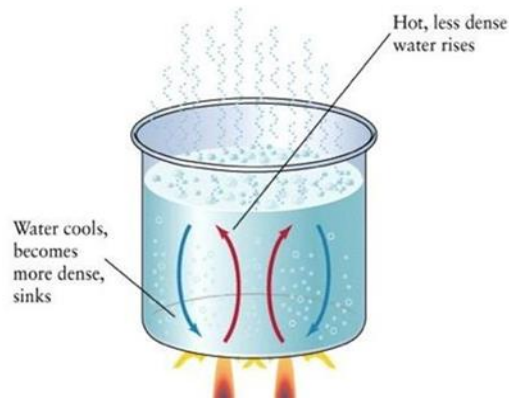


Figure 11: illustration of convection heat transfer mode [19]

From this example we can describe two phenomena's [19]:

1. Conduction between the wall and the very near particle of water
2. The density of hot water decrease by increase of temperature so it move up to the low temperature region and mixes with cold fluid by mixing motion.

Also we must note that:

- The faster the fluid motion, the greater the convection heat transfers.
- In the absence of any motion of fluid, the heat transfer between the solid surface and the surrounding fluid is pure conduction .

Convection has two types [19]:

1. Natural convection : we can call it also « free convection » , and this happen alone without external force induced by density differences due to the variation of temperature in the fluid like the example we had before is natural convection.
2. Forced convection : it is when the fluid is forced to flow over the surface like using fan , pumps ...

The basic mathematical equation for convection heat transfer is expressed by Newton's law of cooling is in equation (2-4):

$$Q = h A (T_s - T_\infty) \quad (2-4)$$

Where:

- T_s is the surface temperature
- T_∞ is the fluid temperature
- A is the surface area of the solid.

• h is the convection heat transfer coefficient in (W /m^2k)

The coefficient h is not a property of the fluid, it is determined experimentally and it depends on all the variables influencing convection such as the surface geometry, the nature of fluid motion, the properties of the fluid, and the fluid velocity. (Table 3) represents some values of h [20]:

Table 3: Typical Values of the Convective Heat Transfer Coefficient [21]

Type of Convection	Convective Heat Transfer Coefficient h ($W/m^2 K$)
Air, free	2,5–25
Air, forced	10–500
Liquids, forced	100–15,000
Boiling water	2,500–25,000
Condensing water vapor	5,000–100,000

For space applications and exactly in thermal control measures of satellites , we find this kind of heat transfer inside the satellite like in : heat pipes and pumped fluid loops , also the exterior surfaces of the satellite are exposed to free convection from the ambient atmosphere in orbits that are below 180 km [22].

2.3.2 Conduction

This is another mode of heat transfer (figure 12) is a representation of this mode ,and it happen through a solid or a fluid due to a temperature difference between the warm and the cold side, in other words it is the transfer of kinetic energy from one molecule to the adjacent molecule this kinetic energy always go from the high temperature molecule (showed in red) to low temperature molecule (showed in blue) [23].

- In solids, conduction may be attributed to atomic activity in the form of lattice vibrations and energy transport by the free electrons.
- In fluids, conduction occurs due to the collisions and diffusion of the molecules during their motion.

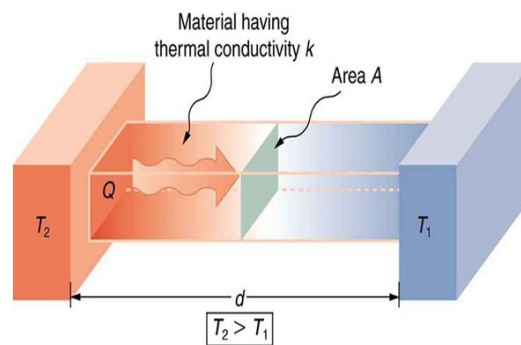


Figure 12: conduction heat transfer [24]

The basic equation for thermal conduction is the Fourier's law and it say that: « the rate of flow of heat through a simple homogenous solid is directly proportional to the area of the section at right angles to the direction of heat flow, and to change of temperature with respect to the length of the path of the heat flow » [21].

$$q \propto dT/dx \quad (2-5)$$

$$q = -k \frac{dT}{dx} \quad (2-6)$$

Where:

- k is thermal conductivity (W/m.K)
- dT/dx is temperature gradient
- q is heat flux (W/sq. m)

2.3.2.1 Assumptions of Fourier's Law [21]

1. Steady state conduction \rightarrow temp. Does not change with time.
2. Uni-directional heat flow.
3. Constant temp gradient and linear temp profile.
4. No internal heat generation.
5. Bounding surfaces are isothermal in character.
6. Isotropic and homogeneous material.

2.3.2.2 Thermal Conductivity (k) [23]

Thermal conductivity is the measure of the ability of a material to conduct heat, and it is one of the transport properties of a material, its unit is W/m°C or W/m.K, and for:

- For Solids and Liquids: $k = f(T)$.
- For gases/vapor, $k = f(p, T)$.

We must note that:

- Metals are best conductors.
- Alloying of metals will reduce « k »
- « k » decreases with increase in temperature in most of metals except: Aluminum and Uranium.
- Gases with high molecular weight have small « k ».
- For most materials, the variation of thermal conductivity with temperature, is given by the following expression :

$$k = k_0(1+bT) \quad (2-7)$$

Where:

- k_0 is thermal conductivity at 0°C
- b is a constant, and it is called the temperature coefficient of thermal conductivity.

Table 4: The values of the coefficient of thermal conductivity for various substances at atmospheric pressure and moderate temperatures [25].

Substance	$t, ^\circ\text{C}$	$\lambda, \text{W/mK}$	Substance	$t, ^\circ\text{C}$	$\lambda, \text{W/mK}$
<i>Metals</i>			<i>Liquid</i>		
Silver	0	429	Mercury	0	7.82
Copper	0	403	Water	20	0.599
Iron	0	86.5	Acetone	16	0.190
Tin	0	68.2	Ethyl alcohol	20	0.167
Lead	0	35.6			
<i>Nonmetallic materials</i>			<i>Gases</i>		
Sodium chloride	0	6.9	Hydrogen	0	0.1655
Tourmaline	0	4.6	Helium	0	0.1411
Glass	18	0.4 – 1.0	Oxygen	0	0.0239
Wood	18	0.16 – 0.25	Nitrogen	-3	0.0237
Asbestos	18	0.12	Air	4	0.0226

For satellites we find this kind of heat transfer inside the satellite between components.

2.3.3 Thermal Radiation

This the last mode of Heat transfer, it is thermal radiation it happen by electromagnetic waves. Contrary to heat conduction, where the transfer is by the movement of molecules, atoms or electrons, i.e. a transfer medium is required, radiation does not need a medium and thus can even occur in vacuum which mean without any contact between objects [26].

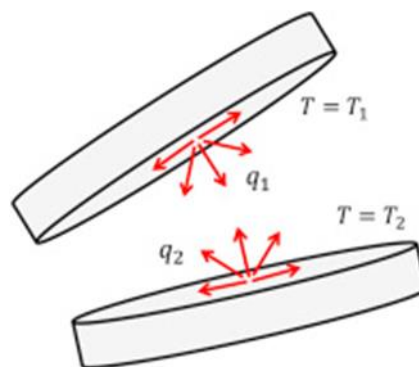


Figure 13: radiation heat transfer [27]

The wavelength of electromagnetic waves transferring heat varies between 0.8 and $400 \mu\text{m}$ are called ultra-red light. The wavelength range of visible light is between 0.35 and $0.75 \mu\text{m}$. At low temperatures the part of visible radiation is so low that it cannot be detected by human eyes. At high temperatures the part of

visible light is increasing and can be detecting by our eyes (e.g. the glowing filament of an electric bulb) [26].

Radiation waves will be fully or partially reflected, transmitted or absorbed at the surface of a body. The ratio of absorbed to total radiation is α , the ratio of transmitted vs. Total is τ and the ratio of reflected vs total radiation is ρ . α is also called absorptivity. The sum of the three ratios is always 1 as we se in equation (2-8) [26].

$$\alpha + \rho + \tau = 1 \quad (2-8)$$

2.3.3.1 Blackbody Radiation [28]

A blackbody is defined as a perfect emitter and absorber of radiation at a specified temperature and wavelength, no surface can emit more energy than a blackbody.

We can call a blackbody as diffuse emitter which means it emits radiation uniformly in all direction beside that it absorbs all incident radiation regardless of wavelength and direction.

The radiation energy emitted by a blackbody per unit time and per unit surface area can be determined from the Stefan-Boltzmann Law given by Equation (2-9).

$$E_b = \sigma T^4 \quad (2-9)$$

- E_b Is called the blackbody emissive power.
- σ is the constant Stefan-Boltzmann and it equals to $5.67 \times 10^{-8} \text{W/m}^2 \cdot \text{K}^4$
- T is the absolute temperature of the surface.

The maximum rate of radiation that can be emitted from a surface at an absolute temperature T_s expesed as it is showed in equation (2-10):

$$\dot{Q}_{emit} = \sigma A_s T_s^4 \quad (2-10)$$

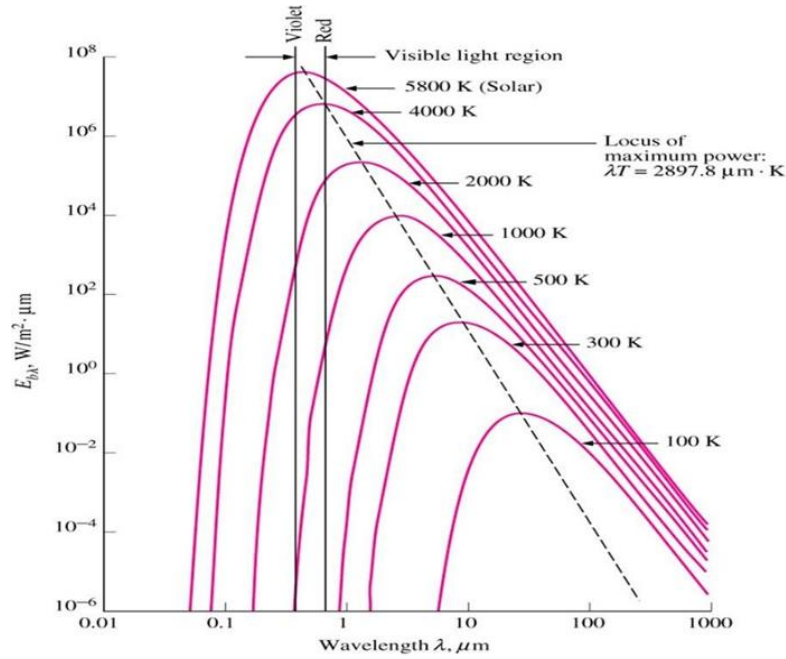


Figure 14: Variation of blackbody emissive power with wavelength [28]

2.3.3.2 Radiation Properties [28]

A blackbody can be a reference in describing the emission and absorption characteristics of real surfaces.

Diffuse surface: is a surface which its properties are independent of direction.

Gray surface: is a surface which its properties are independent from wavelength.

2.3.3.2.1 Emissivity [28]

The emissivity of a surface is defined as the ratio of the radiation emitted by the surface to the radiation emitted by a blackbody at the same temperature. Thus: $0 \leq \epsilon \leq 1$

The emissivity of a surface is not a constant; it is a function of temperature of the surface and wavelength and the direction of the emitted radiation, $\epsilon = \epsilon(T, \lambda, \theta)$ where θ is the angle between the direction and the normal of the surface.

We note that for a blackbody $\epsilon = 1$.

2.3.3.2.2 Absorptivity, Reflectivity, and Transmissivity [28]

The radiation energy incident on a surface per unit area per unit time is called irradiation, G .

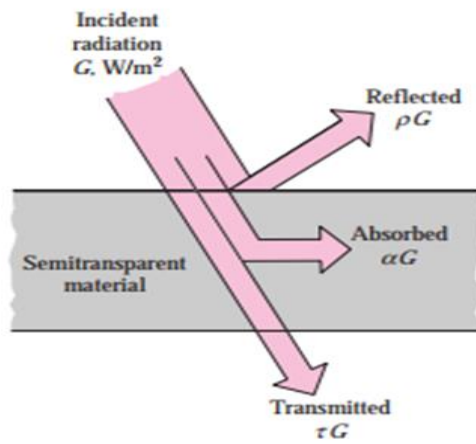


Figure 15: transmission, absorption, reflection [28]

Absorptivity α : is the fraction of irradiation absorbed by the surface, It is the ratio of absorbed radiation (G_{abs}) to incident radiation (G).

Its value range: $0 \leq \alpha \leq 1$

Reflectivity ρ : is the fraction of irradiation reflected by the surface, it is the ratio of reflected radiation (G_{ref}) to incident radiation (G).

Its value range: $0 \leq \rho \leq 1$

Transmissivity τ : is the fraction of irradiation transmitted through the surface. It is the ratio of transmitted radiation (G_{tr}) to incident radiation (G).

Its value range: $0 \leq \tau \leq 1$

Radiosity J : total radiation energy streaming from a surface, per unit area per unit time. It is the summation of the reflected and the emitted radiation.

- The first law of thermodynamics requires that the sum of the absorbed, reflected, and transmitted radiation energy be equal to the incident radiation. That is : $G_{abs} + G_{ref} + G_{tr} = G$

Dividing each term of this relation by G we obtain equation (2-8): $\alpha + \rho + \tau = 1$

An opaque surface: is a surface which does not allow transmission of radiation through it. Wood, metal, concrete are some examples of opaque materials.

For opaque surfaces, $\tau=0$, and thus above equation reduces to (2-11):

$$\alpha + \rho = 1 \quad (2-11)$$

2.3.3 .2.3 Real body radiation

The radiation emitted by all real surfaces is less than the radiation emitted by a blackbody at the same temperature, and is expressed as it is in (2-12) [29]:

$$\dot{Q}_{emit} = \varepsilon \sigma A_s T_s^4 \quad (2-12)$$

Where: ε is the emissivity of the surface [29]

The emissivities of some surfaces are given in Table 5:

Table 5: The values of emissivities of surfaces at 300°K [29]

Materials	Emissivity (ε)
Black body	1
Skin, human	0.95–0.98
Burnt skin	0.97
Human bone	0.96
Plaster	0.98
Black Silicone Paint	0.93
Plant leaves	0.96
Water	0.95–0.96
White paper	0.93
Asphalt	0.93
Concrete	0.85
Wood	0.8–0.95
Aluminum Foil	0.04
Silver Polished	0.02–0.03

2.3.4 Radiation heat transfer between two surfaces [29]

When a surface of emissivity ε and surface area A_s at an absolute temperature T_s is completely enclosed by a much larger (or black) surface at absolute temperature T_{large} separated by a gas (such as air) that does not intervene with radiation, the net rate of radiation heat transfer between these two surfaces is given by equation below (2-13) :

$$\dot{Q}_{rad} = \varepsilon \sigma A_s (T_s^4 - T_{large}^4) \quad (2-13)$$

In this special case, the emissivity and the surface area of the surrounding surface do not have any effect on the net radiation heat transfer.

And for the general cases radiation exchange depends on temperature and geometric as well the surface proprieties like material, smoothness, and curvature it depends also on wavelength and direction.

2.3.4.1 The view factor

Accounting for all effects is extremely complicated, so we define a new parameter called « the view factor », which is a purely geometric quantity and is independent of the surface properties and temperature. It is also called the shape factor, configuration factor, and angle factor [30].

The view factor is note with capital F like that F_{ij} the letters i, j are referred to bodies i and j [30].

- the view factor from a surface to itself is zero :

$$F_{ii} = 0 \quad (2-14)$$

- The value of the view factor is between zero and one.

From the above definition of view factors, we can develop a general expression for F_{ij} ; we consider the arbitrarily oriented surfaces A_i and A_j . Elemental areas on each surface, dA_i and dA_j are connected by a line of length R , which forms the polar angles θ_i and θ_j , respectively, with the surface normal n_i and n_j so we write the view factor like shown in equation (2-15) :

$$F_{ij} = \frac{\int_{A_i} \int_{A_j} I_{r+e,i} \frac{\cos\theta_i \cos\theta_j}{R^2} dA_i dA_j}{\pi \int_{A_i} I_{r+e,i} dA_i} \quad (2-15)$$

$I_{r+e,i}$ Is the intensity of radiation leaving surface i by emission and reflection

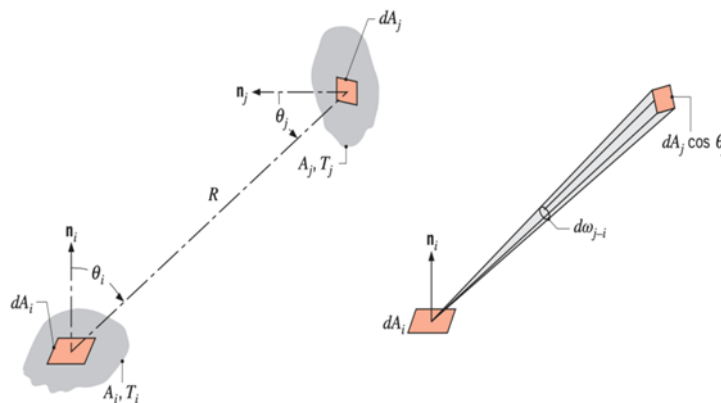


Figure 16: geometry for view factor [30]

Because of the complexity of the equation (2-15) the values View factors for hundreds of common geometries are evaluated and the results are given in analytical, graphical, and tabular form in several publications.

2.3.4.2 RADIATION HEAT TRANSFER between surfaces [29]

Radiation exchange between surfaces is complicated because of reflection. This can be simplified when surfaces are assumed to be black surfaces.

The net radiation between two surfaces can be expressed as:

$$\dot{Q}_{12} = A_1 F_{12} E_{b1} - A_2 F_{21} E_{b2} \quad (\text{W}) \quad (2-16)$$

Applying reciprocity rules $A_1 F_{12} = A_2 F_{21}$ we get:

$$\dot{Q}_{12} = A_1 F_{12} \sigma (T_1^4 - T_2^4) \quad (\text{W}) \quad (2-17)$$

2.4 Thermal environment

The satellite is exposed during its lifetime to many heating fluxes from ground to orbit [31], at ground heat sources are: satellite heat generation and the conditions of the surrounding air, then the launch vehicle experience some heat changes during ascent caused by friction in the upper atmosphere [16], and other heating fluxes while orbiting.

We mostly focus on orbital heating, because it is the most environment that the satellite spent his lifetime to do his mission, also these fluxes are considered as radiation fluxes, and it depend on the orbital parameters, size, shape, view factors and surface properties of the spacecraft [31].

The low earth orbit CubeSat will be exposed to direct solar radiation, sunlight reflected off the Earth (albedo), and infrared energy emitted by the Earth as shown in Figure below . The Sun is the source of thermal radiation so at 1 AU, the average intensity of solar radiation is approximately 1367 W/m². The albedo of the Earth varies with the land, ocean, cloud cover, and latitude [31]. While the amount of sunlight and albedo a spacecraft is exposed to depend on its orbit (sunlight phase or eclipse), a spacecraft will always be exposed to the Earth's infrared radiation. These three heat fluxes make the satellite in dangerous cases at some points and it is necessary to ensure that the satellite can resisted the temperatures resulting from environmental heating.

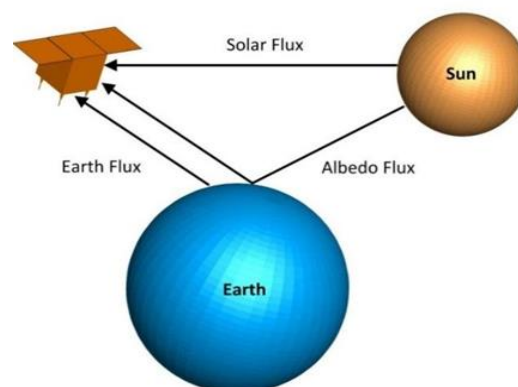


Figure 17: orbital heating fluxes [31]

2.4.1 Direct solar radiation

The direct solar flux is by far the most intense heat input for a satellite [32], The radiation is nearly constant, and is equal along all directions, so we can say that the sun is the main heat power source and it can be approximated as a blackbody at 5762° K (to simplify the thermal modeling) emitting primarily shortwave radiation [3], because of the elliptic orbit of earth the sunlight reaching earth varies according to the distance from the sun [46]. so the value of the solar constant, the intensity of incident energy at 1 AU (Earth's mean distance from the sun), is equal to $S=1367 \text{ W/m}^2$ at 1 AU this value called the solar constant is the « rate at which solar energy is incident on a surface normal to the Sun's rays at the outer edge of the atmosphere » in W/m^2 [16]. Earth is furthest from the sun and intensity is at minimum value of 1322 W/m^2 at summer solstice, and at winter solstice it is at maximum value of 1414 W/m^2 [31].

The heat absorbed by the spacecraft due to the direct solar flux can be analytically written as [34]:

$$\dot{Q}_s = \alpha S A \cos \theta \text{ (W)} \quad (2-18)$$

\dot{Q}_s Is the solar flux in (w), total area A, whose normal vector forms an angle θ with the solar rays, and α is the material solar absorptance of the surface.

2.4.2 Albedo – Reflected Solar Radiation

Albedo is defined as: « The fraction of the incident radiation that is reflected from the surface » [33], the amount of albedo radiation incident upon a spacecraft is a function of spacecraft orientation and orbit and can be a significant source of radiation when the spacecraft is near the Earth. Albedo is considered to be in the same spectrum as solar radiation, and is calculated as a fraction of the solar constant [16]. The heat flux due to albedo that is incident on a surface depends on the surface absorptivity α , the solar constant S, the albedo factor A_f and it is dependent upon the surface properties of the Earth [16], and the view factor F between the surface and the Earth, the analytical equation is: [34]:

$$\dot{Q}_a = \alpha S A F_{bp} A_f \cos \theta \text{ (w)} \quad (2-19)$$

F_{bp} Is view factor from earth to satellite.

The albedo value ranges from 0 to 1. The value of 0 refers to a blackbody, a theoretical media that absorbs 100% of the incident radiation. Albedo ranging from 0.1–0.2 refers to dark-colored, rough soil surfaces, while the values around 0.4–0.5 represent smooth, light-colored soil surfaces. The albedo of snow cover, especially the fresh, deep snow, can reach as high as 0.9. The value of 1 refers to an ideal reflector surface (an absolute white surface) in which all the energy falling on the surface is reflected. The mean albedo of the earth system is 0.36 ± 0.06 [33].

2.4.3 Earth IR

Incident sunlight on the Earth that is not reflected as albedo is absorbed by the Earth and reemitted as infrared energy. The IR energy emitted by the Earth can vary with many factors such as the local temperature of the Earth's surface, and the amount of cloud cover, also variation with season and latitude [31]. These localized variations can be significant, but they are far less severe than variations in albedo. The Earth IR intensities of interest for satellites are the long-term averages, so the variations are not of great concern [16].

Heat flux absorbed in the IR range is treated differently than that of the visible range and requires that the emitting body's temperature be included in the equation [51], the analytical equation is shown as: [34]:

$$\dot{Q}_{iR} = A \varepsilon_b F_{bp} \varepsilon_p \sigma T_p^4 \quad (2-20)$$

ε_b Emissivity of satellite, ε_p and T_p^4 are respectively emissivity and temperature of earth, and σ is the stephan-boltzman constant $\sigma = 5,67 \cdot 10^{-8} \text{ W/m}^2 \text{ K}$.

2.4.4 Internal Radiation

Internal radiation accounts for all of the radiation heat transfer that occurs between components inside the spacecraft this internal radiation may vary around the orbit and at different times during the mission phases. For satellites without moving parts, the electrical-power draw for components will be converted entirely to heat so heat depends only on the power of the components [31].

2.4.5 Solar eclipse: umbra penumbra antumbra [35]

The eclipse phase that we talked about in the introduction of the space environment part, it has degrees of shadowiness called umbra, penumbra and antumbra described as it follows:

The umbra is a shadow's dark core. Imagine a light source and an object casting a shadow. If you are standing within the umbra, you will not be able to see any part of the light source as the object blocks all direct light rays.

The penumbra is a half-shadow that occurs when a light source is only partly covered by an object for example, when the Moon obscures part of the Sun's disk.

The antumbra is the lighter area of a shadow that appears beyond the umbra, at a certain distance from the object casting the shadow. It only exists if the light source has a larger diameter than the object.

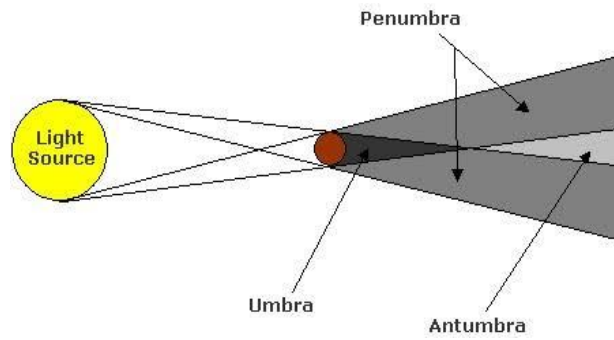


Figure 18: representation of umbra, penumbra and antumbra [35]

In our study the light source is sun , the object which is exposed to light is our planet earth in we are going to consider all of this degrees of shadawnes as one phase and we call it eclipse, of course it has an angle of start and angle of end , we take it in consideration in the transient analysis.

CHAPTER 3: SATELLITE THERMAL ANALYSIS AND HARDWARE

3.1 Introduction

Satellite thermal control system ensures that the satellite will be operating safely during the mission and this by maintaining the temperatures of satellite components and surfaces within allowable limits in its orbit by heating and cooling. For this purpose active and passive thermal control methods are used.

Before choosing those methods of thermal control we have to find the interval of temperatures of the satellite in specifies conditions and characteristics, and that by doing an orbital analysis (in our study we used system tool kit (STK.11) firstly then thermal analysis (done numerically with matlab R2015a) to protect our satellite in extreme temperatures while orbiting , and it is exposed to sunlight or in the eclipse, so the thermal analysis has a maximum range of temperature (hot case) and a minimum range of temperature (cold case). And it is clear for thermal engineers to define a hot case (max) and a cold case (min) to make their calculations easier and obvious but the condition is that the parameters chosen for each case must be realistically experience during the satellite lifetime. Hot and cold case input parameters can include the solar vector, albedo factor, component dissipation, beta angle, and altitude [16].

For conductivity k , specific heat c_p , density ρ , and beginning and end-of-life as input parameters they can be changed for a hot case versus a cold case , so we can put an average value for the analysis instead of a separate hot and cold case values, in order to reduce the number of variables in the analysis [16].

In this chapter we defined our orbit parameters and the two cases, and the thermal nodes; we must note that the thermal analysis is done by solving the energy balance equation.

3.2 Thermal mathematical model

Thermal control of a satellite on orbit is done mathematically by balancing the energy absorbed from the environment and generated internally to the energy stored and emitted by the satellite, by following the law of conservation of energy [31], this equation is represented as:

$$\dot{Q}_{in} + \dot{Q}_{generated} = \dot{Q}_{stored} + \dot{Q}_{out} \quad (w) \quad (3-1)$$

The heat entering the satellite comes from the external heating fluxes experienced in LEO, expressed in (W), and is shown by equation below:

$$\dot{Q}_{in} = \dot{Q}_s + \dot{Q}_a + \dot{Q}_{iR} \quad (3-2)$$

The heat generated is the total heat dissipated by the satellite components.

Stored heat is also called the heat capacity, thermal mass or thermal capacitance . The heat storage capability of a material is represented by both the heat capacity and the specific heat c_p , and from equation (2-3) [36], we can write our stored heat inside the satellite as:

$$\dot{Q}_{stored} = m c \frac{dT}{dt} \quad (3-3)$$

Heat emitted from the satellite occurs through radiation from the satellite external surfaces to outer space, and according to the Stefan-Boltzmann law, all surfaces above absolute zero emit radiation, using equation (2-10), and concedring the emissivity of the satellite's surfaces we get:

$$\dot{Q}_{out} = \varepsilon \sigma A_s T_s^4 \quad (3-4)$$

3.3 thermal cases

We said in the introduction that thermal engineers study two cases so that they can have an idea about the temperature range, these cases are the hottest case (maximum temperature), then the coldest case (minimum temperature), explained below:

3.3.1. Hot Case

For a hot case analysis, is for the case when the satellite experience the highest temperature during its mission. For parameters in this case we choose the highest values corresponding to the highest temperatures. So during the warmest time of year we can choose the highest solar vector S of about 1419 W/m^2 and Albedo factors of about 0.35 , as the highest values in this time of the year. Any other properties that are approximated in the analysis should also be chosen to result in higher temperatures [16].

3.3.2. Cold Case

For the cold case analysis parameters are selected to result in temperatures as low as the satellite may realistically experience during its mission. Which means the coldest time of the year so we choose the lowest solar vector S of about 1317 W/m^2 and the lowest albedo factors A_f of 0.2 . Any other approximations made in the analysis should also be chosen to result in low temperatures [16].

3.4 geometric mathematical models [48]

The geometry of the satellite is complex sometimes, that's why we need a geometry model for easier calculations, we must note that the satellite must be discretized to a number of isothermal parts called « nodes », and we have two models:

1. Single node model: a single node means to consider the satellite as an isothermal lumped mass, in this model the satellite is represented as a sphere in a circular orbit around Earth but it has the same surface as the real satellite, so that we develop a baseline understanding the thermal dynamic environment of the low earth orbit (LEO).
2. Six-Node model: is used to remove some of the assumptions in the single-node analysis, and increase the accuracy of the temperatures. Instead of a single, spherical node model, this models each face of the satellite as its own node to compose a "box". This introduces some complexity over the single-node analysis, namely the consideration of view factors, and conduction between nodes. These nodes are still considered lumped masses, so internal temperature gradients are ignored.

3.5 effect of orbital motion on thermal control

Thermal control of the satellite is affected by the orbital motion; the main parameter that is effective is the beta angle, and the eclipse phase that we talked about previously [37].

3.5.1 Beta Angle

Beta angle is defined as the angle between the orbit plane and the vector from the Sun as shown in (figure 19) [37].

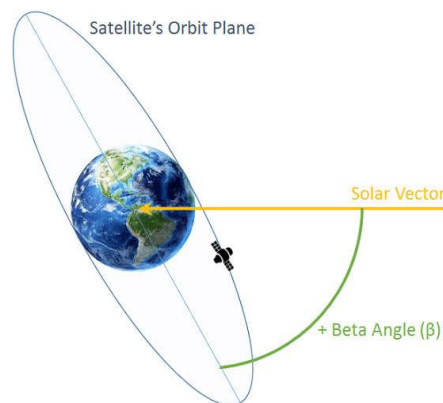


Figure 19: beta angle [37]

Depending on orbital inclination, beta can take values within the range ± 90 degrees. For many purposes, however, it is necessary to consider only the magnitude of beta (not its sign). This is because differences between positive and negative beta have to do solely with whether the satellite appears to move in its orbit clockwise or counter-clockwise when viewed from the perspective of the Sun [37].

The degree of orbital shadowing (eclipse), which a satellite in LEO experiences is determined by the satellite beta angle. A satellite launched into an initial beta angle of 0 degrees ($\beta=0^\circ$) for the orbiting satellite, this makes the satellite spend a maximum possible time in Earth's shadow as a consequence the solar

absorption is reduced . If the satellite is launched into an orbit follows the terminator, results in a beta angle of 90 degrees ($\beta=90^\circ$) and the satellite is in sunlight 100% of the time [37].

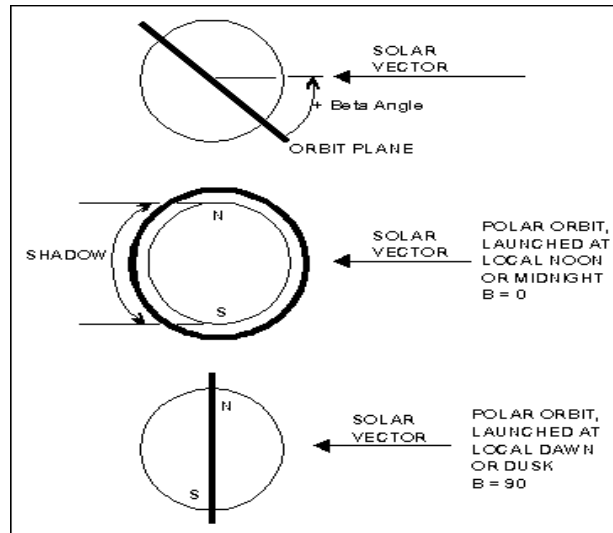


Figure 20: orbit plane vs polar orbit [37]

3.5.2 Eclipse duration

We must study the eclipse duration, and we can calculate it for circular orbits, it has relation with the time to take by the satellite to complete one orbit around the earth which means the orbital period and, and the beta angle ($|\beta| \leq \frac{\pi}{2}$), and finally (h) a function of relative orbit radius and it is calculated as :

$$h = \frac{H+R}{R} \tag{3-5}$$

Where H is orbit altitude, R is earth's radius and it equals to 6370 km, h is the function of relative orbit radius.

As the beta angle decreases the orbit plane will eventually intersect the horizon of the Earth at which point eclipse will be a factor to consider. This angle is defined as the critical beta angle: [22]

$$\beta_{max} = \frac{\pi}{2} - \cos^{-1}\left(\frac{1}{h}\right) \tag{3-6}$$

The analytical equation of orbital period is:

$$P = 2\pi \sqrt{\frac{(H+R)^3}{GM}} \quad (s) \tag{3-7}$$

where H is orbit altitude, R is earth's radius and it equals to 6370 km, h is the function of relative orbit radius, G is the universal gravitation constant it equals to $6,67 \cdot 10^{-11} \frac{N \cdot m^2}{kg^2}$ M is the mass of the earth it equals to $5.97 \cdot 10^{24}$ Kg.

The relative eclipse period $\frac{T_e}{T_0}$ (eclipse duration divided by orbital period), T_e is eclipse duration, and this report is expressed as:

$$\frac{T_e}{P} = \frac{1}{\pi} \cos^{-1} \left(\frac{\sqrt{h^2 - 1}}{h \cos \beta} \right) \quad \text{with } \beta < \beta_{max} \quad (3-8)$$

This is for eclipse duration, for eclipse start angle and end angle, it calculated like in the equation below:

$$\begin{cases} \phi_{es} = \pi - \cos^{-1} \left(\frac{\sqrt{h^2 - 1}}{h \cos \beta} \right) \\ \phi_{ee} = \pi + \cos^{-1} \left(\frac{\sqrt{h^2 - 1}}{h \cos \beta} \right) \end{cases} \quad (3-9)$$

3.6 Thermal analysis

after choosing the orbit and the type of satellite, the problem can be solved, and for that it has two states, the steady state (it can be resolved by hand), the transient state (it more complex and it need a numerical method to be solved many softwares can do thermal analysis like THERMICA, SINDA, ESATAN...) we solved it by a MATLAB code. And we have this diagram as a clear explanation, and all calculation are done will there is no internal heat inside the satellite:

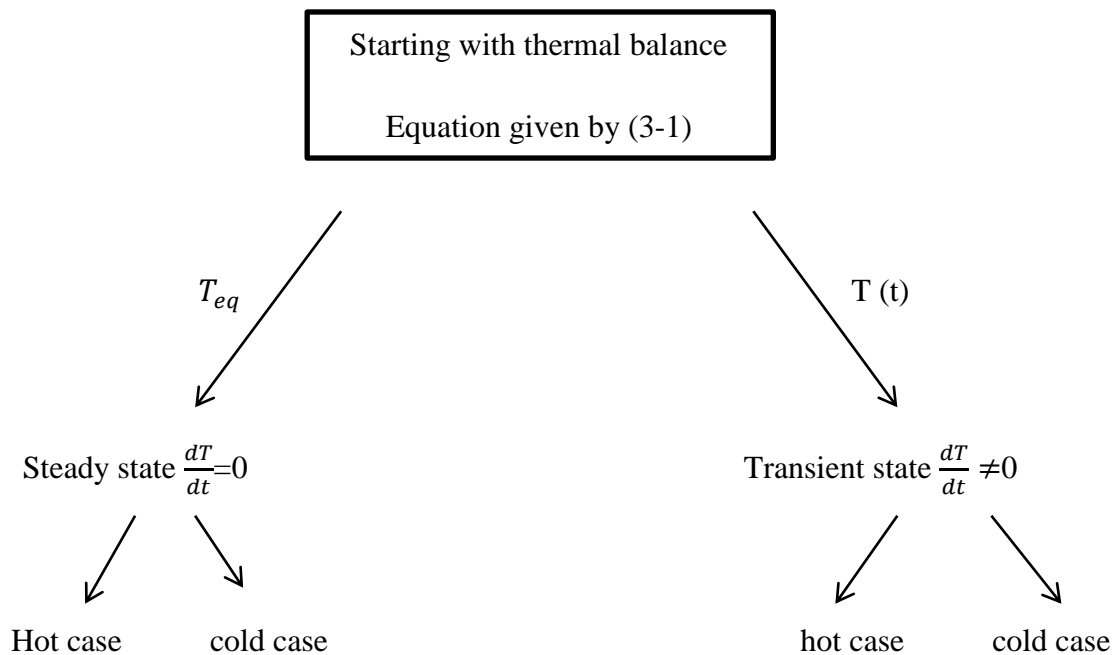


Figure 21: thermal analysis diagram

3.6.1 Steady State

In steady state analysis calculations, the heat flux in the satellite and the heat flux out of the satellite are constant . Steady state, by definition, means that the temperature and heat flux at a point in a body will

not change with time . However, the temperature may vary from one point to another. For this state we use equations (2-18), (2-19), and (2-20) to find the heat fluxes along the orbit. And the temperature of the cubesat we will use the energy balance equation (3-1); all changes in equations are in the next chapter [38].

3.6.2 Transient state

A realistic practical thermal design for a satellite is generally based on transient considerations .However, the starting point for a transient analysis calculation is the temperature distribution found from the steady state calculation [38] [48].

The transient calculation will result in the satellite temperature history at successive time intervals [38]. For a transient model, the temperature will typically vary with time as well as location which means we take in consideration eclipse start angle and end angle.

The transient thermal analysis uses the same equations (2-18), (2-19), (2-20) but with a tiny modification but it matter for the calculations.

1. For solar heat flux it will be calculated in the transient sate :

$$\dot{Q}_s = \alpha S A \cos \theta * F_s \quad (3-10)$$

$$\text{If } \phi_{es} < \phi < \phi_{ee} \quad F_s = 0 \quad \text{else} \quad F_s = 1$$

ϕ_{es} And ϕ_{ee} is the start angle and stop angles of the eclipse respectively, and F_s is the step function for solar heat flux.

2. For albedo heat flux also there is a modification because the direction of sunlight is reflected off a planet changes as the spacecraft orbits the planet. To account for this, the cosine term in Equation (2-19) is put in terms of the beta angle and orbit angle, this modification depends on the altitude of the satellite, which means low orbits less than 320 km and high orbits [39].
- For low orbits the cosine term in Equation (2-19) is modified by using the law of spherical cosines, where :

$$\cos \theta = \cos \phi \cos \beta \quad (3-11)$$

Where ϕ is the spacecraft angular position in its orbit, and β the orbit solar angle, the albedo heat flux will be:

$$\dot{Q}_a = \alpha S A F_{bp} A_f \cos \theta * F_a \quad (3-11)$$

F_a Is the albedo function and it is calculated with [39]:

$$F_a = \cos \phi \cos \beta F_s \quad (3-12)$$

If $-\frac{\pi}{2} < \phi < \frac{\pi}{2}$ $F_e = 1$ else $F_e = 0$

- For high orbits :

$$\dot{Q}_a = \alpha S A F_{bp} A_f \cos \theta^* F_a \quad (3-13)$$

Where F_a is:

$$F_a = \left(\frac{1+\cos\phi}{2}\right)^2 \left[1 - \left(\frac{\phi}{\phi_{es}}\right)^2\right] \cos \beta F_s \quad (3-14)$$

If $-\phi_{es} < \phi < \phi_{es}$ $F_s = 0$ else $F_s = 1$.

Using the thermal energy balance equation (3-1), and using equation (3-3), and considering that there is no heat generated we get the transient state equation:

$$\dot{Q}_{in} - \dot{Q}_{out} = m c \frac{dT}{dt} \quad (3-15)$$

Then using equation (3-2) we get:

$$\dot{Q}_s + \dot{Q}_a + \dot{Q}_{iR} - \dot{Q}_{out} = m c \frac{dT}{dt} \quad (3-16)$$

Now replacing with the heat fluxes equations (3-10), (3-13), (2-20) and equation (3-4) then rearrangement we get:

$$m c \frac{dT}{dt} = \alpha S A \cos \theta^* F_s + \alpha S A F_{bp} A_f \cos \theta^* F_a + A \varepsilon_b F_{bp} \varepsilon_p \sigma T_p^4 - \varepsilon \sigma A_s T_s^4 \quad (3-17)$$

Now to get temperature in every point we need to solve this equation (3-17).but it this can happen only numerically because of the non-linear terms, so An Euler or Runge-Kutta method can be used to solve this differential equation (3-17).

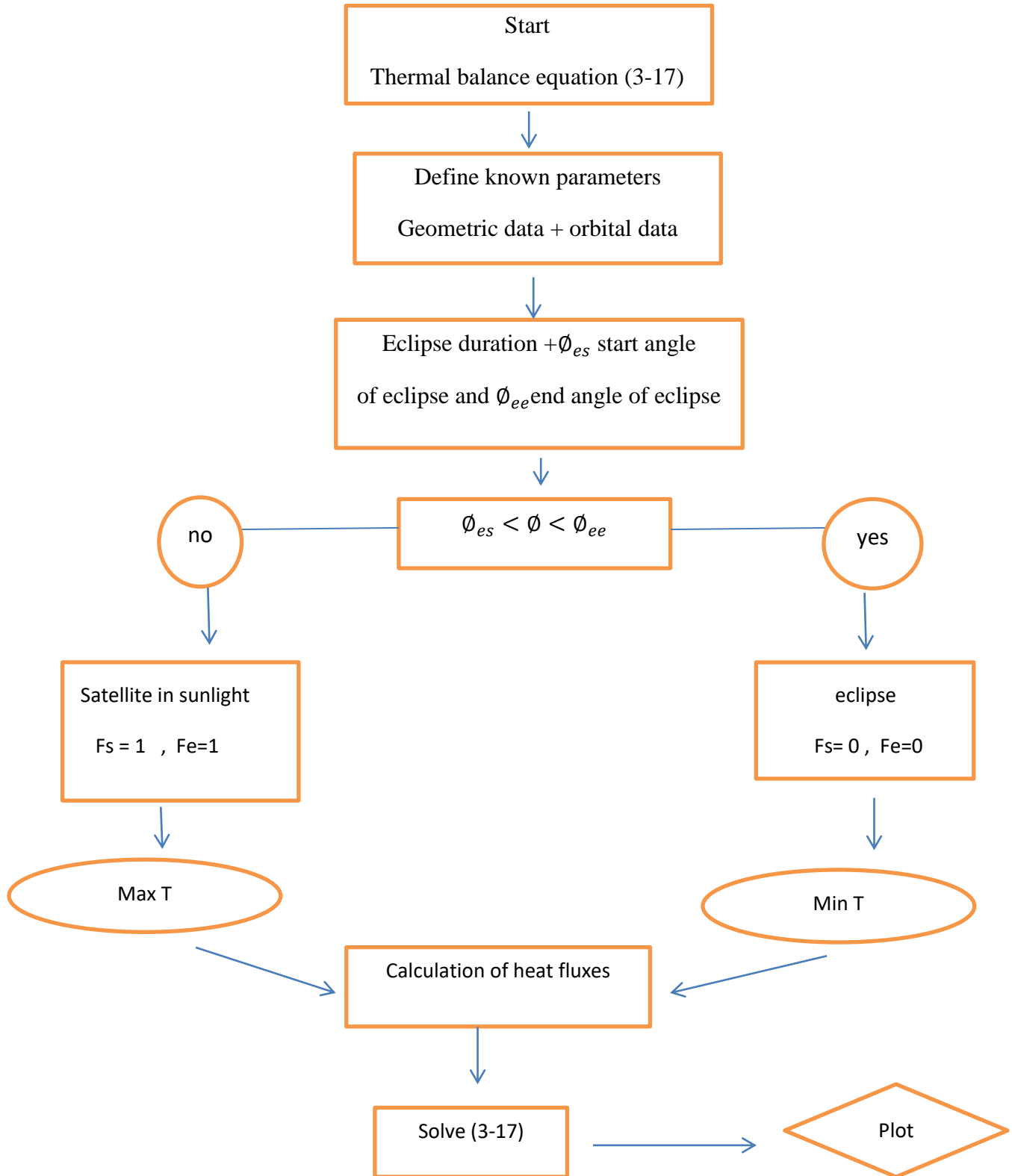
An iterative process was used in MATLAB to calculate the temperature of the CubeSat at various points along its orbit, see it in the thermal analysis part.

It should be noted that for the analysis presented here it is helpful to express dT/dt as $dT/d\phi$, since F_s and F_a are functions of ϕ . dT/dt can be converted to $dT/d\phi$ using the following equation:

$$\frac{t}{P} = \frac{\phi}{2\pi} \quad (3-18)$$

3.6.3 summary organigram

Now after knowing all the details of the study , we can summarize that in the following organigram, starting with thermal balance equation and we end with solving this equation, and finally plotting this results to make it easy to see the changes of the temperature during this orbit.



3.7 Orbital analysis

When analyzing the thermal environment of a spacecraft, certain orbital elements must first be determined including how long the spacecraft will be exposed to the radiations previously described, plus the six keplerian orbital parameters (seen in chapter 1).

Orbital analysis was done with System’s Tool Kit (STK) software (Analytical Graphics, Inc.), which allows thermal engineers to design and analyze dynamic simulations on land, on sea, in air, and, for this project’s purposes, in space. A free educational license of STK 11 with Space Environment and Effects Tool (SEET) was used to complete the orbital analysis. SEET evaluates the effects of the near-Earth space environment on the satellite including radiation, the geomagnetic field, particle impacts, and temperature. STK was selected because it is the top software packages for orbital analysis at a very low cost. For the purposes of our analysis, the ISS orbit was simulated using STK for one day, more details about STK and all the calculations are the numerical part [38].

3.8 thermal analysis processes

We are going to represent our work plan as diagram so that it is easy to read it and understand our plane.

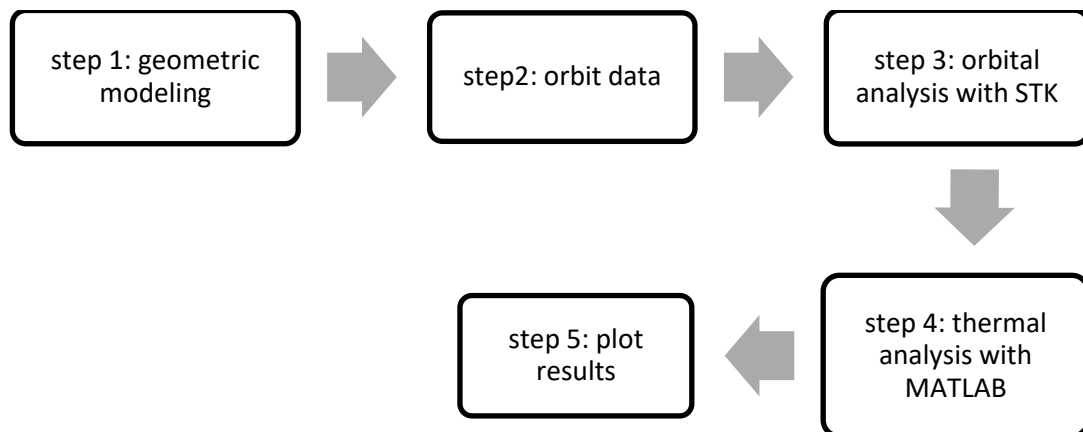


Figure 22 : steps of thermal analysis

1. Step 1 :geometric modeling : which means we define the satellite shape (spherical or cubical) , size (cubesat : 1U, 2U, 3U ..), mass in kg , material proprieties (absorptivity , emissivity , specific heat ..)
2. Step 2: orbit data: it means we done the information of the orbit like altitude, and keplerian parameters.
3. Step 3: orbital analysis: we set time of simulation (one orbit, half of day, 24 hours ...), plus the information that we decided in step 1 and 2, and we get the eclipse duration and other orbital information.
4. Step 4: thermal analysis: after having the orbital information, we write a code MATLAB to calculate heat fluxes and the satellite temperature using equations mentioned previously.

5. Step 5: plot results: after having all that, we represent our results in graphs then discuss them.

3.9 thermal control in previous mission

A very helpful article i found for a previous LEO orbit mission and it is published in 2001 where Kwang-bog shin, Chun-Gon Kim, Chang-Sun Hong, and Ho-hyung Lee, from the department of Aerospace Engineering, Korea published their analysis of the transient temperature of the Korea Multi-Purpose Satellite (KOMPSAT) solar array to show how the temperature of the solar array varied during its orbit [40].

Figure 25 is taken from the search of the Korean team and it shows the time evolution of the array's temperature, or transient temperature of the array. Different cases are analyzed; the difference is the proprieties of composite materials.

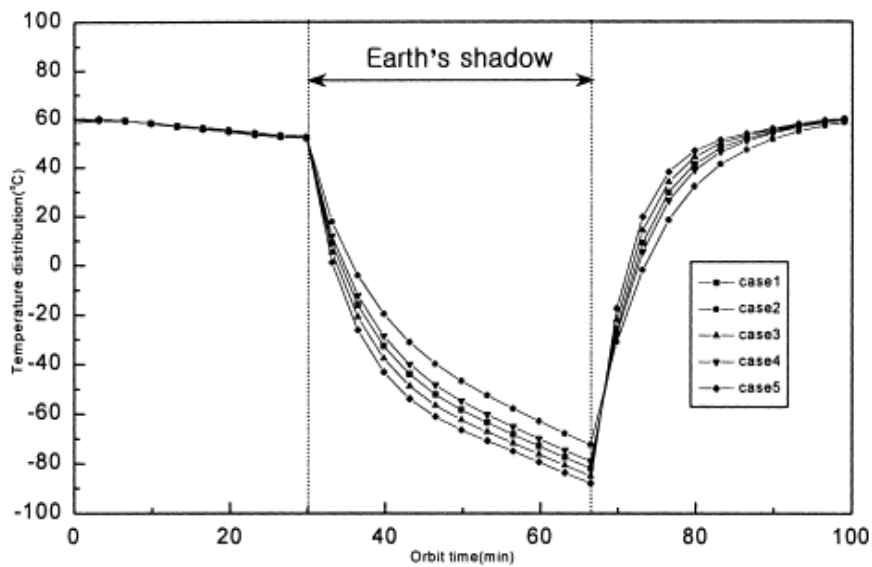


Figure 23: Time evolution of the temperature of the KOMPSAT solar array [40]

The moments the solar array would be in eclipse is clearly indicated by the large decrease in temperature. At approximately 67 minutes into the array's orbit, there is a large increase in temperature, which indicates the moment the array begins to transit back into sunlight. The highest temperature is about 60°C and the lowest temperature of the cubesat in the eclipse phase is -80°C.

3.10 thermal control systems and its goal

It is so important to use the thermal control system, it make the satellite while orbiting operate in a safe range of temperature, and this lead to a successful mission.

Usually we use two types of thermal control for satellites: passive and active thermal control. And for small satellite, thermal design is primarily passive, like using MLI, low absorption/emission ratio surface properties and other ones [31].

TCS is to prevent overheating and undercooling in every part of equipment, at all phases of the spacecraft mission (mostly within the space environment). The typical solution adopted in TCS to avoid overheating (which cause permanent damage), is to choose cover materials with appropriate thermo-optical properties to keep the system basically cool, and to compensate the eventual undercooling (particularly at eclipses) by means of distributed electrical heaters. Undercooling, usually do not cause permanent damage but just a dormant non-operational state (which may be critical to the mission, however) .

3.10.1 Passive thermal control

Passive thermal control is a means of controlling spacecraft and its component temperatures without using equipment such as fluid loops or other active components this definition according to NASA web site [41].

3.10.1.1 Multi-Layer Insulation (MLI)

One of the most common thermal control components on spacecraft are multilayer insulation (MLI) and single-layer radiation barriers, Thermal insulation is crucial for cryogenic and infrared satellite missions. These types are the most efficient insulation material for space applications .

The MLI blanket is composed of a number of low-remittance sheets combined with low-conduction netting layers ,the outer cover can be made from Teflon, aluminized Kapton, black Kapton, or Beta cloth, which is a Teflon coated glass fabric .The blanket assembly is typically secured to the satellite by bonding or using Velcro strips. Grounding straps are added to reduce the electrostatic charge on the insulation during orbit [16].

Materials used for fabrication of an MLI blanket should never be handled with bare hands and should never be exposed to uncontrolled and corrosive environments in order to avoid contamination and material degradation. Also avoid harsh pulling or unnecessarily wrinkling because this can cause stress in the layers and defects that may not appear until launch [16].

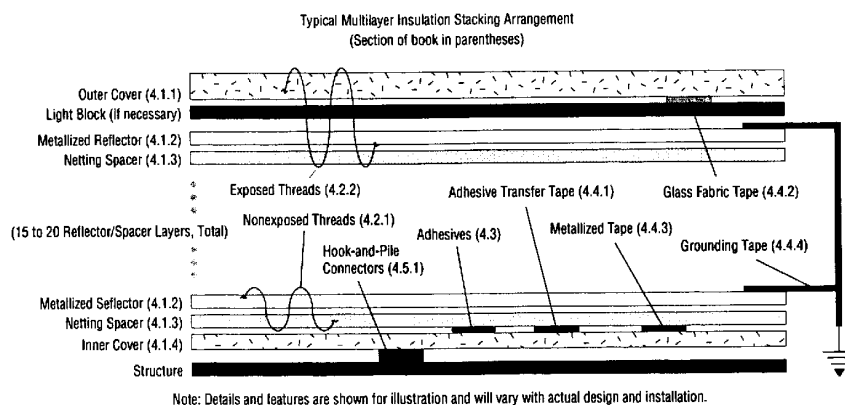


Figure 24: Typical MLI blanket structure [41]

Heat transfer through MLI is by conduction and radiation. As we add more layers the heat transfer with radiation is minimized, and with minimum contact between these layers we ensure the embossing of the sheets, on the other hand conduction heat transfer is minimized by using low density material [31].

3.10.1.2 Surface coatings and finishes

In space, external satellite surface paints or coatings are influenced by environmental effects, namely, Atomic Oxygen (ATOX), molecular contamination, and Ultraviolet radiation (UV) [41].

using paints is necessary for the degradation of physical and thermo-optical properties of the components and this degradation is dependent on mission duration and orbit altitude , also solar absorptivity (α) and infrared emissivity (ϵ) of materials control the thermal radiation of the satellite .

3.10.1.3 Thermal Straps

Thermal straps are used on space missions to conduct the heat from inaccessible regions of the spacecraft and radiate into space. Thermal straps come in different shapes and lengths according to the requirement. They are comprised of thin wires of high thermal conductivity metals or foils which make them flexible and efficient thermal links [41].

3.10.1.4 Radiators

A radiator is a surface that disperses more heat than any other surface because of its thermo-optical properties and temperature. Satellite emitted heat is rejected to space by infrared (IR) radiation from its surfaces [34].

The simplest and most common radiators are the existing panels in the exterior surfaces of the satellite. For example, an exterior aluminum panel can serve as a structural panel as well as a radiator [16].

3.10.1.5 Heat pipes

Heat pipe transfers heat from one position to another, with the use of a closed two-phase liquid-flow loop and evaporator. A heat pipe is capable of creating isothermal surfaces; it can change the flux density of a heat flow as a thermal transformer and work as a thermal control system in several different ways [34].

A heat pipe is a sealed pipe or tube with an interior wick that is saturated with a liquid, and a vapor space. As heat is applied to one end, a differential pressure is created that drives the vapor to the cooler end of the pipe where it condenses back into the wick.

Where heat is being applied called the evaporator, capillary pressure head drives the condensed liquid back from the cooler end, called the condenser, to the heated end causing the cycle to repeat [31].

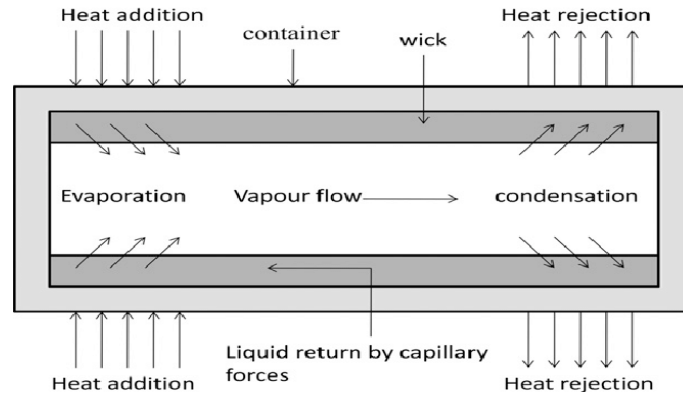


Figure 25: heat pipe schematic [34]

3.10.2 Active thermal control

Active thermal control is the second method used in TCS and it refers to mechanical or thermoelectric systems used for thermal control and to maintain required temperatures with the use of power and may have moving parts, we use this type only when it is not possible to use passive systems alone[40].

3.10.2.1 Heaters

Heaters are sometimes needed in a thermal design to support components in cold conditions and they are one of the simplest devices to use for maintain satellite components in the required temperature range, also to warm up components that are dormant before their activation [34].

The most common type of heater used on spacecraft is the patch heater; it consists of an electrical-resistance element sandwiched between two sheets of flexible electrically insulating material, such as Kapton.

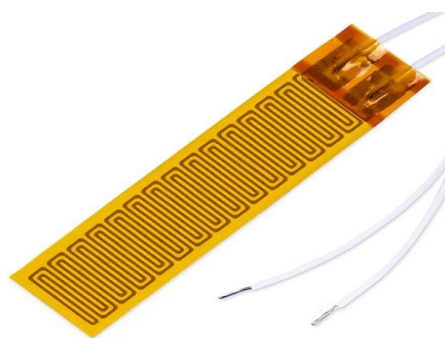


Figure 26: kapton heater [41]

Polyimide or kapton heaters provide the thinnest profile of all flexible heaters, offer precise heat distribution, and excellent tensile strength; they can maintain temperatures from $-328^{\circ}\text{F}/73, 15^{\circ}\text{K}$ to $329^{\circ}\text{F}/438, 15^{\circ}\text{K}$. They are also self-extinguishing and can maintain their original shape regardless of temperature exposure [41].

Heaters are controlled by a thermostat or temperature sensor and used in cold environments to maintain battery temperature; typically the component with the narrowest temperature limits .

The 1U CubeSats Compass-1, MASAT-1, and OUTFI-1 each required an electrical heater attached to the battery in addition to passive control for the entire spacecraft system to maintain thermal regulation in eclipses [41].

Another type of heater is the cartridge heater, is often used to heat blocks of material or high-temperature components , it is a tube-shaped, joule heating element used in the process heating industry[41]



Figure 27: cartridge heater [41]

3.10.2.2 Louvers

Louvers are active thermal-control devices that have been most commonly placed over external radiators, they may also be used to modulate heat transfer between internal spacecraft surfaces, or from internal surfaces directly to space through openings in the spacecraft wall , the most used louver assembly is the bimetallic, spring-actuated, rectangular-blade (venetian-blind) type. Hydraulically activated louvers and pinwheel louvers are used less often today than in the past [22].

Louver radiator assemblies consist of five main parts : baseplate (surface of low absorptance-to-emittance ratio that covers the critical set of components whose temperature is being controlled) , blades (give variable-radiation characteristics to the baseplate) , actuators (drive the blades according to the perceived baseplate temperature, and generally coated black to enhance the interchange of radiation) , sensing elements, and structural elements[22].

3.10.2.3 Fluid Loops

A pumped fluid loop is capable of achieving heat transfer between multiple locations via forced fluid convective cooling. Mechanically pumped fluid loops are not the best choice for small spacecraft because of the high power consumption and mass [41], but limited use in robotic space missions over the past 30 years due to reliability concerns, but are increasingly being looked at to solve complex thermal control problems .

3.10.2.4 Cryocoolers

Cryogenic coolers, or cryocoolers, are used on instruments or subsystems requiring cryogenic cooling, such as high precision IR sensors. Further, the use of cryocoolers is associated with longer instrument lifetimes, low vibration, high thermodynamic efficiency, low mass and supply cooling temperatures less than 50 K[41] .

CHAPTER 4: NUMERICAL RESULTS AND DISCUSSION

4.1 introductions

In this chapter, we are going to specify the orbit and necessary parameters of the satellite, then present the results obtained for the thermal analysis.

The thermal control analysis it has an orbital analysis which is connected with thermal analysis. So the orbital analysis is done with STK.11 software, then the thermal analysis which is done by writing a code in MATLAB R2015a.

So for simplification we are going to use one node model to calculate temperatures of the two cases for the steady state and transient state, then we talk about the multi-node model.

4.2 One node analysis for 3U cubesat

One node model is to consider the satellite as a isothermal lumped mass, in this model the satellite is represented as a sphere in a circular orbit around Earth, it serves as a stepping stone for more development of a more accurate, multiple node model (with different parts allowing different temperatures) [42]. Also we consider that there is no internal heat inside the satellite.

4.2.1 Analytical One-Node Sinusoidal Solution

Due to the non-linear terms in the energy inputs and the non-linear term in the energy output, the energy balance equation can be solved numerically. And we know that for low altitude orbits: we have a highest value of temperature in sunshine, and lowest temperature in eclipse, so that cosine modulation on average can be a suitable first approximation and linearized [42].

$$F_e = F_a = \frac{(1+\cos\phi)}{2} \quad (4-1)$$

For the temperature variations, it is suggested that equation (3-17):

$$T(t) = T_m + \Delta T(t) \quad (4-2)$$

Then after some transients, must develop a periodic solution and using equation (3-18):

$$T(\phi) = T_m + T_a \cos(\phi - \varphi) \quad (4-3)$$

Where T_m is the mean temperature value, T_a is the amplitude of the temperature oscillation, and the phase lag, φ . Linearizing the energy balance equation, we obtain:

$$T_m = \left[\frac{F_{bp} \epsilon_p}{F_{b0}} T_p^4 + \frac{\alpha S A \cos \theta + \alpha S A F_{bp} A_f}{2 \epsilon \sigma F_{b0} A_s} \right]^{\frac{1}{4}} \quad (4-4)$$

F_{b0} Is view factor of unity?

$$T_a = \frac{\alpha S A \cos \theta + \alpha S A F_{bp} A_f}{2 \left(\frac{\pi m c_p \sin \varphi}{T_0} \right) + \epsilon \sigma F_{b0} A_s 4 T_m^3} \quad (4-5)$$

$$\varphi = \tan^{-1} \left(\frac{\pi m c_p}{2 \epsilon \sigma F_{b0} A_s T_0 T_m^3} \right) \quad (4-6)$$

Linear one-node approximation is a very helpful guide during preliminary attempts to spacecraft thermal control.

4.3 Numerical analysis for the one node model

4.3.1 Geometry model

We suppose a 3U cubesat (10 cm × 10 cm × 34 cm) weights 3 kg, its total surface is 0, 14 m² an example of a 3U cubesat represented in the figure below.

We simplified the cubesat as a sphere that has the same total area as the original cubesat of 3 kg mass, so we consider a sphere of total area = 0.14 m², (sphere surface area is $4\pi r^2$ r is the radius of sphere) and cross section area = the total area /4 = 0.14/4= 0.035 m², the sphere is represented in the figure below.

So we plot a sphere (figure 28) in MATLAB for better vision.

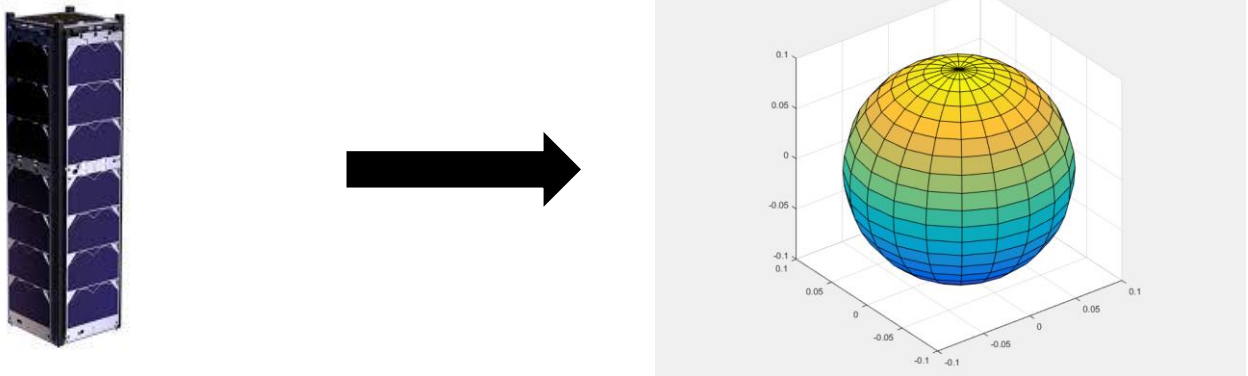


Figure 28: geometric modeling: from the original shape to a spherical model (sphere plotted with MATLAB)

considering the structure of the satellite is with aluminum 6061-T6, this is the value assumed for the specific heat ($c_p=879$ J/Kg.K), and its temperature is about 20°C (=293, 1°K), for absorptivity and emissivity of the surface of the cubesat: we consider that the surface of the cubesat is 56 % solar cells + 44 % aluminum as shown in (figure29).

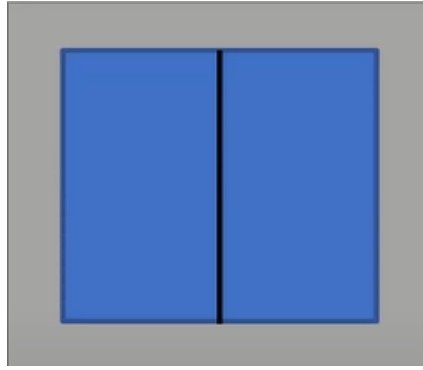


Figure 29: surface of the cubesat (in blue 56% solar cells + in grey 44% aluminum)

With those hypotheses absorptivity and emissivity values as shown in the (table6):

Table 6: absorptivity and emissivity of the surface of the cubesat

	Occupancy	Absorptivity α	Emissivity ϵ
Solar cells	56%	0,92	0,8
Aluminum	44%	0,255	0,025
Average	100%	0,627	0,459

For simplifications we use the average values of absorptivity and emissivity.

4.3.2 Orbit data

It is proposed that the CubeSat will be launched from the International Space Station (ISS). So it is assumed the CubeSat will have the same orbital parameters as the ISS. We choose this orbit because all its parameters are known, and anyone can get information about it in numerous sources .Having this vehicle, a report with the orbital values was generated, in the STK simulation values were extracted than 1 day, but only one orbit will be shown in the results.

Table 7: orbital parameters of the international space station. [43]

Elements	Values	Elements	Values
Inclination (degree)	51.6	Eccentricity	1.518×10^{-4}
Apogee Altitude (km)	410.83	Apogee (km)	6788.93
Perigee Altitude (km)	408.18	Perigee (km)	6786.88
Average Altitude (km)	409.8	Average Radius R_{ISS} (km)	6787.905
Orbital Period (min)	92.6151	Revolutions per Day	15.5482

The CubeSat will have an altitude of 400 km, inclination of 51.6° , and orbital period of approximately 90 minutes. Figures bellows shows the trajectory for the CubeSat. The orbit's beta angle is assumed to be the same as the orbit's inclination. The portion of the orbit in yellow indicates when the CubeSat is exposed to direct sunlight, while the portion in blue indicates eclipse phase.

(Figure 30 and 31) below are taken from STK and it represents the orbit in 3D and 2D respectively, and the satellite is showed as a point.

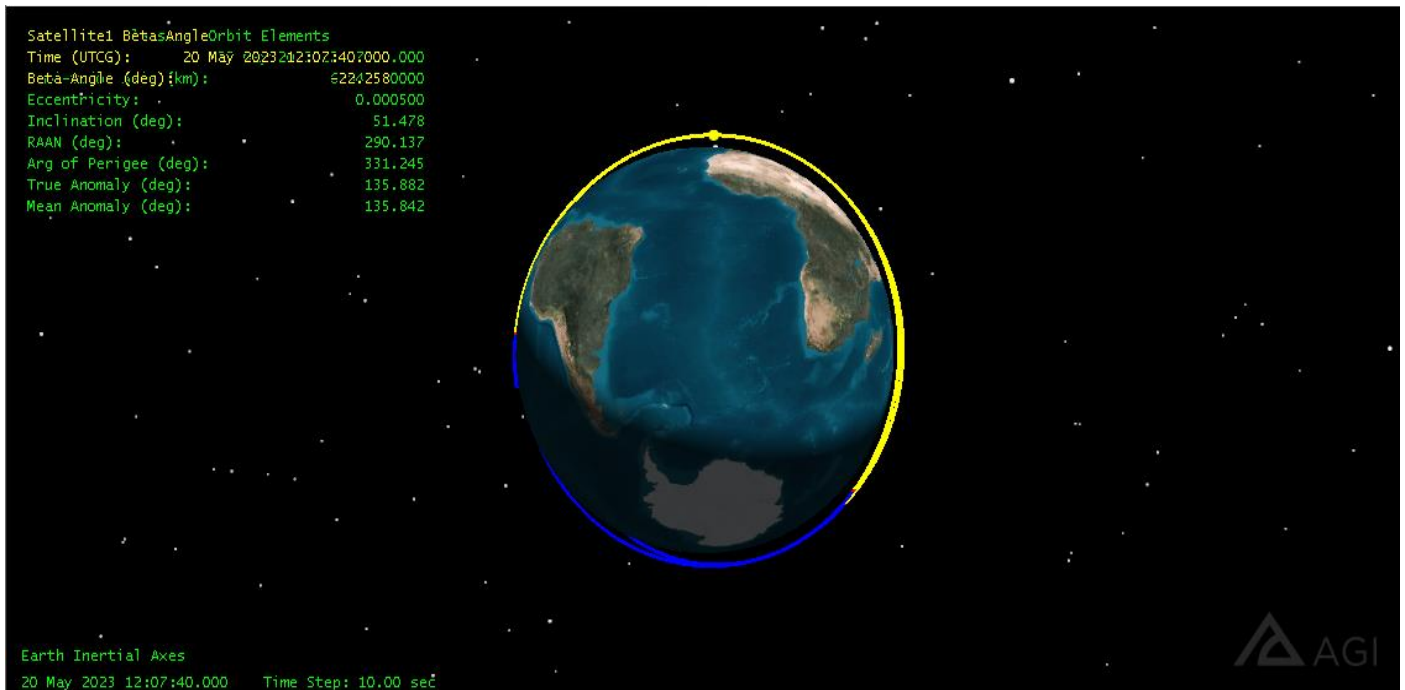


Figure 30: 3D view of our low earth orbit

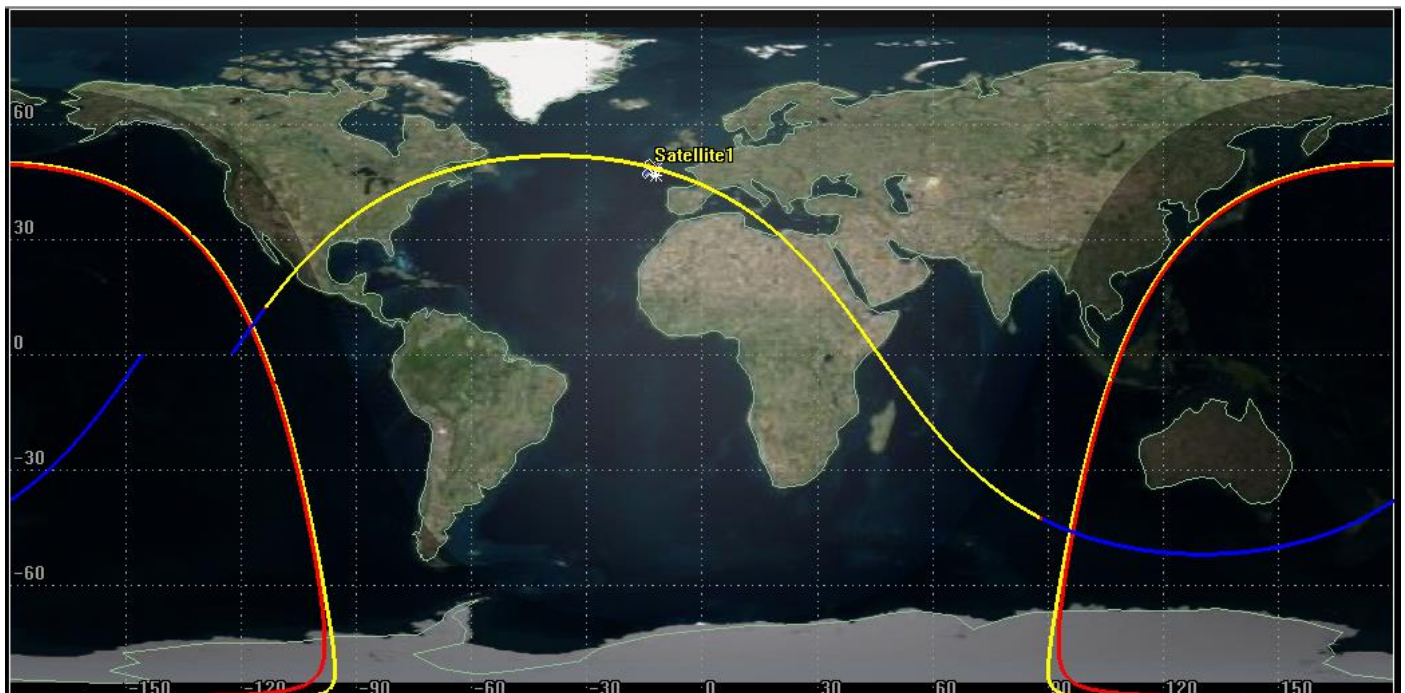


Figure 31: 2 D view of our low earth orbit

4.3.3 Orbital analysis

For a space mission we should study many domains and analyses different conditions and their effect on the mission. So that we can design the most perfect spacecraft.

In our thesis we are studying the thermal control for the spacecraft, and we have chosen for that STK (system tool kit) software for the orbital analysis.

4.3.3.1 STK software

Systems Tool Kit is a modeling, simulation, analysis, and operations software we can say that it is a multi-physics software application from Analytical Graphics, Inc (AGI) (an Ansys company) that make things easier for engineers and scientists to perform complex analyses to evaluate their performance in real or simulated time for space applications. And it named it by its initials (STK) [44].

STK was created in 1989 by the three founder of Analytical Graphics, Inc (Paul Graziani, Scott Reynolds, and Jim Poland) they named it Satellite Tool Kit (STK) then after developments the name has changed to Systems Tool Kit in 2020 to reflect its applicability in land, sea, air, and space systems.

By using STK the user can:

- Select, build, or import vehicle and sensor models to represent a system (space, land, air ...)
- Simulate the system proposed in real-time
- Collaborate with other platforms to create a more complete project.

An additional advantage that STK offers is being able to calculate thermal loads; however, to access this environmental module it is necessary to have a license with a cost [45]. This is important to mention, these costs cannot be easily covered by public institutions throughout the country, so it is necessary to develop, as a country, its tools that can be used without restrictions for use. Academic, non-commercial, non-governmental, non-military.

4.3.3.2 STK interface

Once STK is installed, we can run it, first thing we get this window (figure 32), it says that opening a scenario (you have a scenario created before) or creating a new scenario and the last one is training and tutorials (you can get manuals how to use it online):

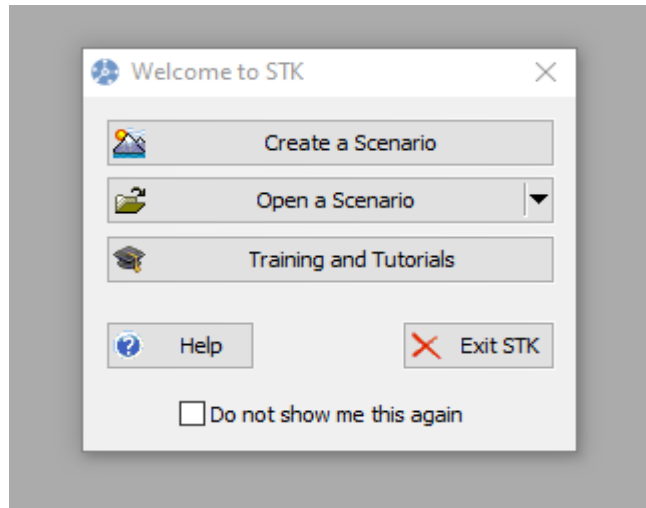


Figure 32: welcome window of STK

Each analysis or design within STK is called scenario. Within each scenario any number of satellites, aircrafts, targets, ships, and other objects can be created it is the first thing we have to decide for each scenario, (figure 33) below show us object we can select.

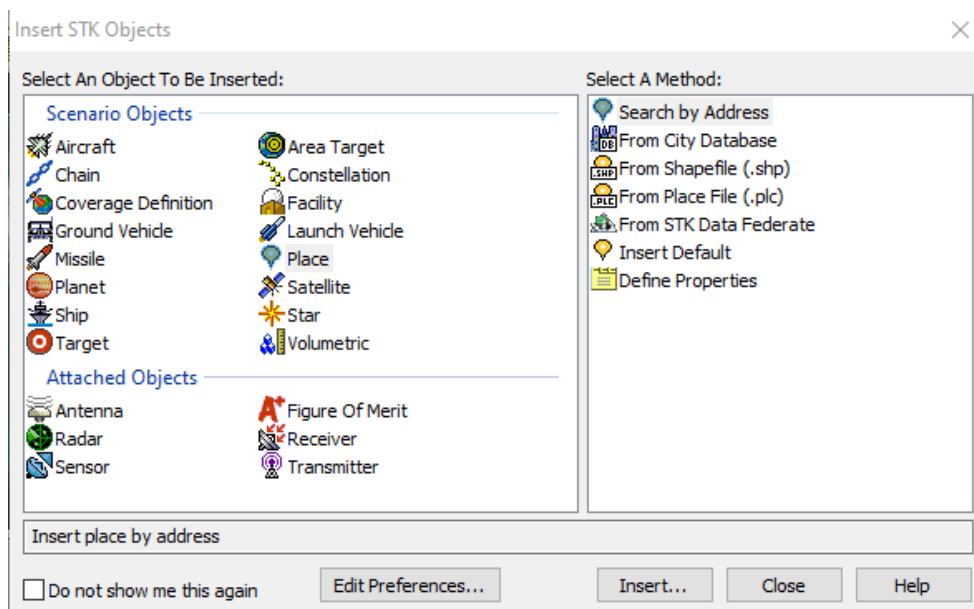


Figure 33: STK objects window

The STK interface is standard GUI (graphical user interface) display with toolbars and 2D map and 3D map windows, (figure 34) show all the interface.

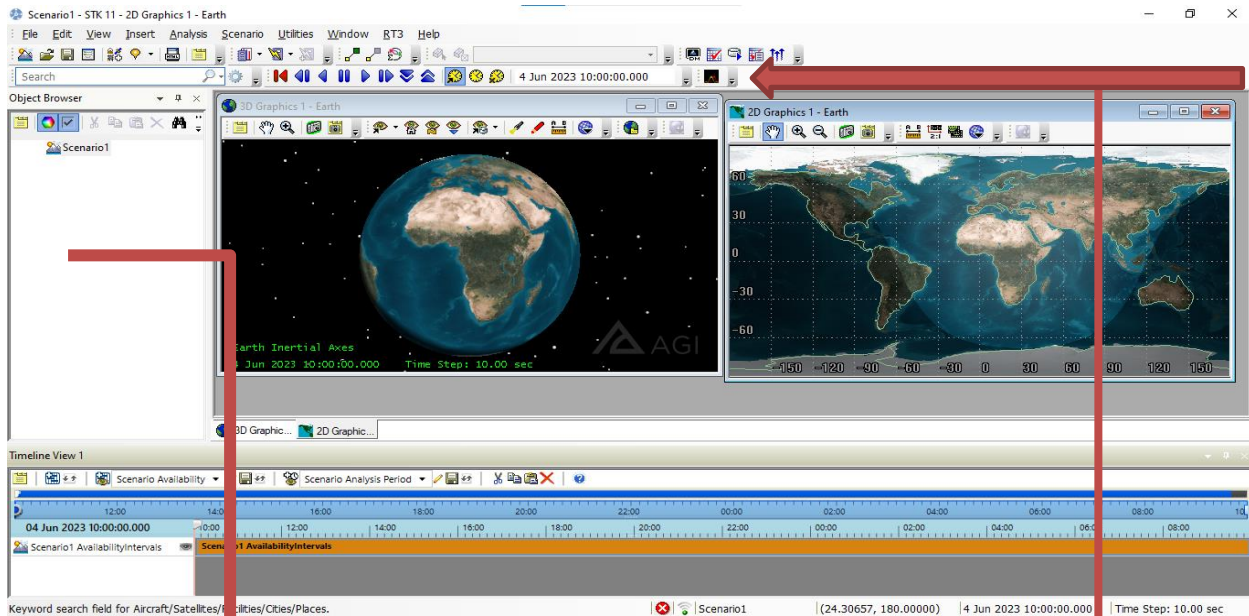


Figure 34: STK interface



Figure 35: object browser



Figure 36: animation bare and the scenario time

Scenario time is settable, we can reset any moment.

The upper part contain the toolbar showed in(figure 37), animation bare(is for animating the scenario , we can fast time or relent it) and the scenario time , plus a very important tool it's the called report and graph manager , it is showed in black .

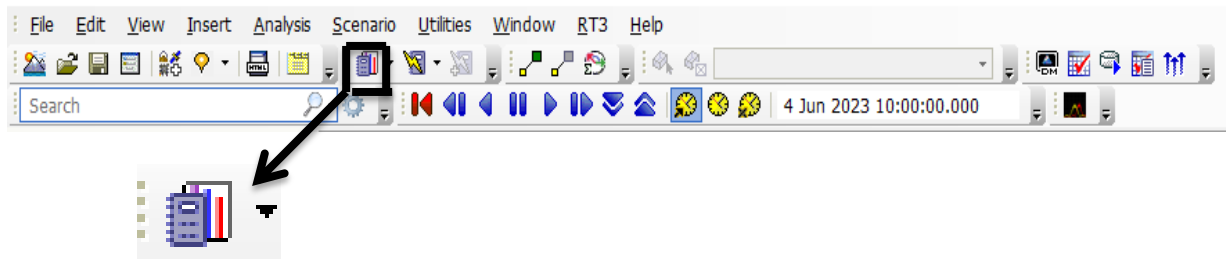


Figure 37: report and graph manager tool

Report and graph manager tool allows us to get reports and graphs for the analysis we have done.

4.3.3.3 Create a satellite

As mentioned before we run STK.11 , select new scenario we named our scenario ISS , then we select a satellite from the STK object window we named it satellite 1, then we select orbit wizard , then insert and you do any necessary changes as shown in (figure 38).

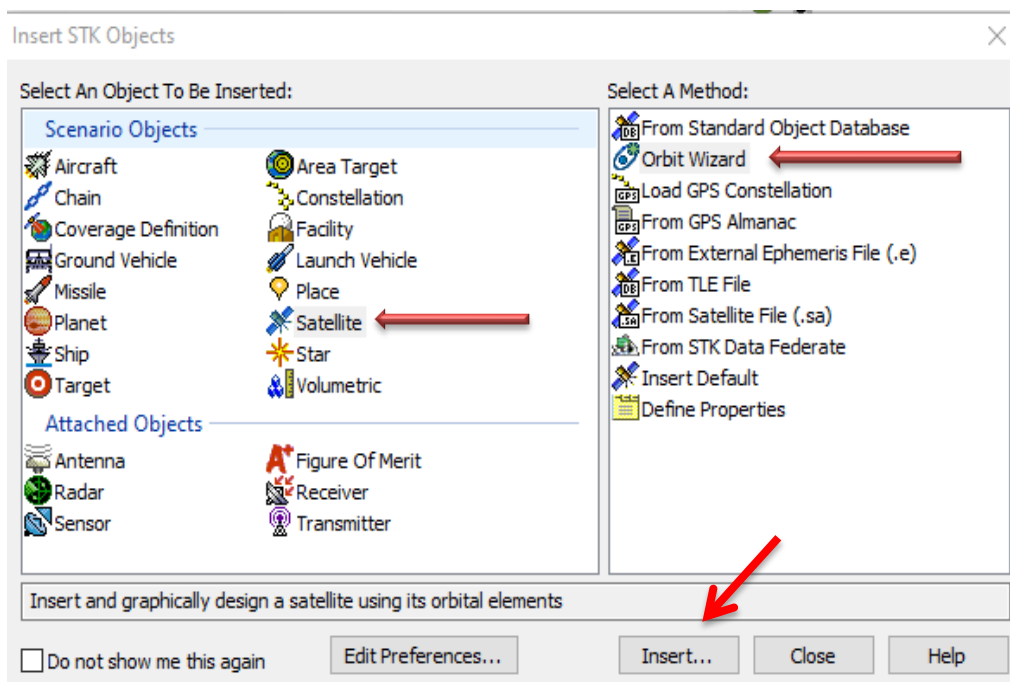


Figure 38: create a satellite

Or just select the satellite, then you will find it in the object browser, double click on it, from the new window we click on orbit to change orbital parameters (figure 39).

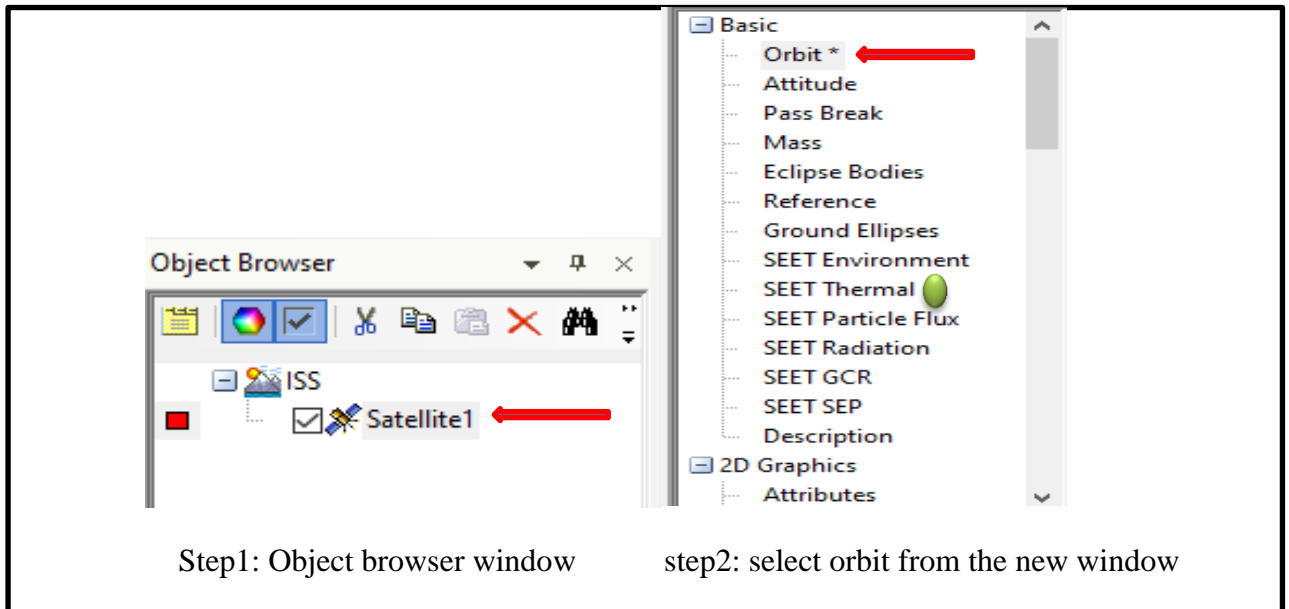


Figure 39 : change orbital parameters

Our orbit parameters are the same of the ISS see (table 7), so we enter those parameters, and step size:

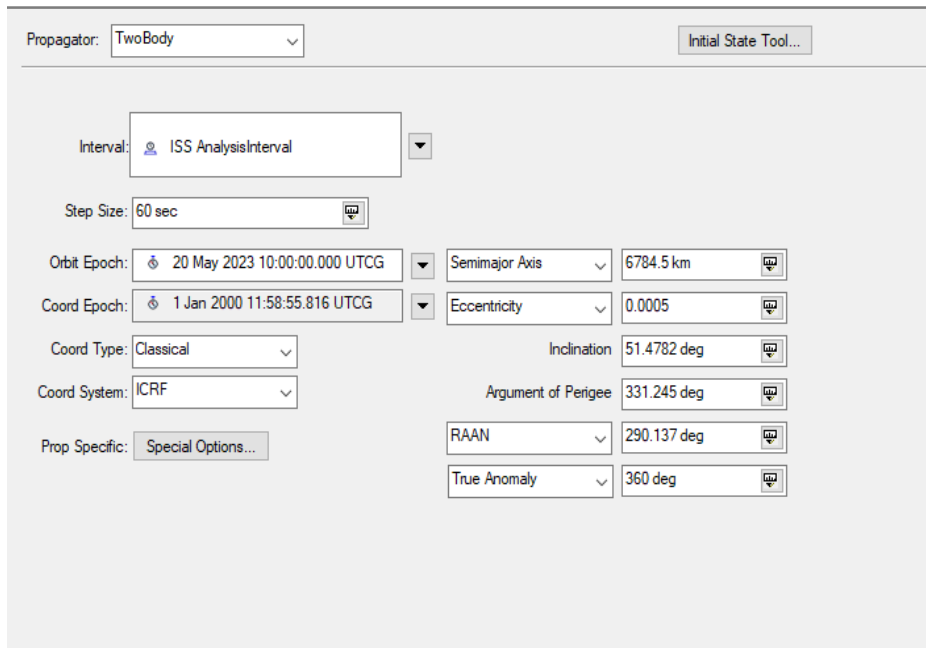


Figure 40 : orbital parameters for satellite1

Now we can add some thermal parameters to the analysis by clicking on (SEET thermal) showed with green point in the basic list ,we consider that the satellite doesn't have any internal dissipation heat , the absorptivity and emissivity values see (table 6),average value of earth albedo is 0.3 (figure 41) then we apply and ok.

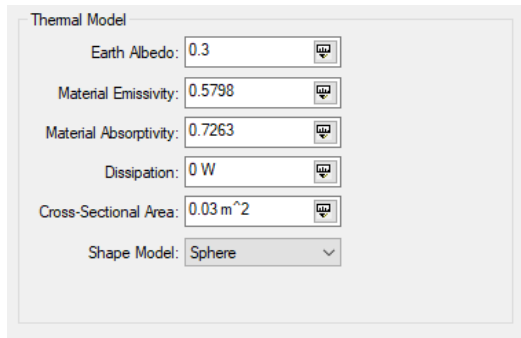


Figure 41 : thermal model window

Now edit colors and what to see in the home screen. We go to the 2D graphics list, then lighting we select red for penumbra phase, yellow when satellite is exposed to direct sunlight, and blue for umbra, and we select them to be viewed in maps:

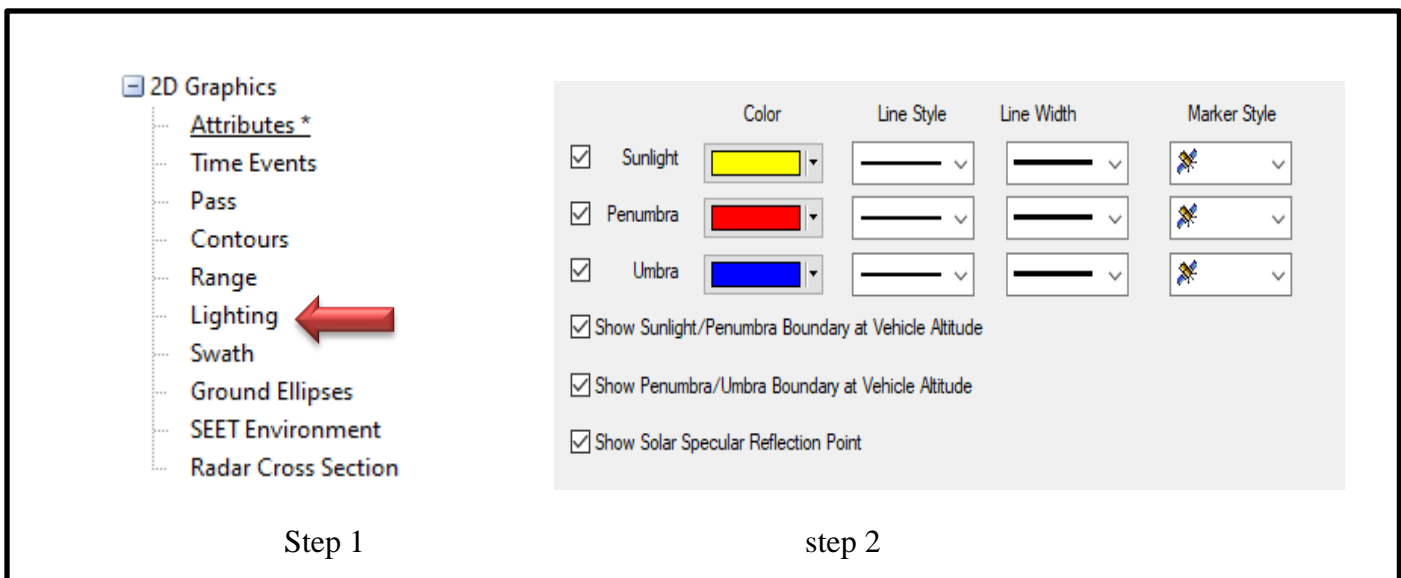


Figure 42: colors choice for umbra ,penumbra and sunlight

Now , we want to show the orbital parameter on 3D map, we go to the 3D graphics list, then Data display and we select classical orbit elements also we chose the color :

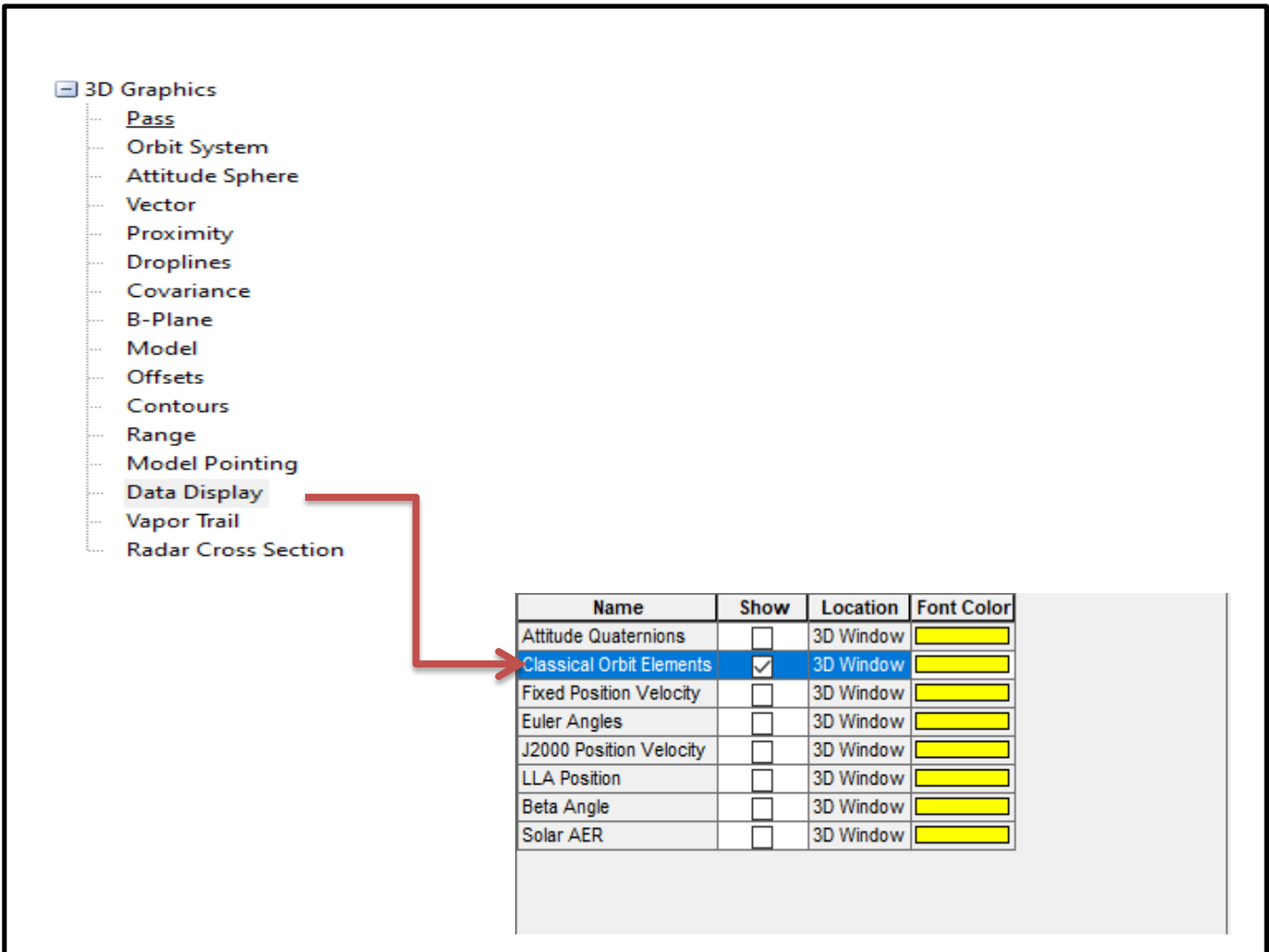


Figure 43 : show classical orbit elements with yellow on home screen

Apply then OK.

The final view on 3D and 2D maps are in the next page.

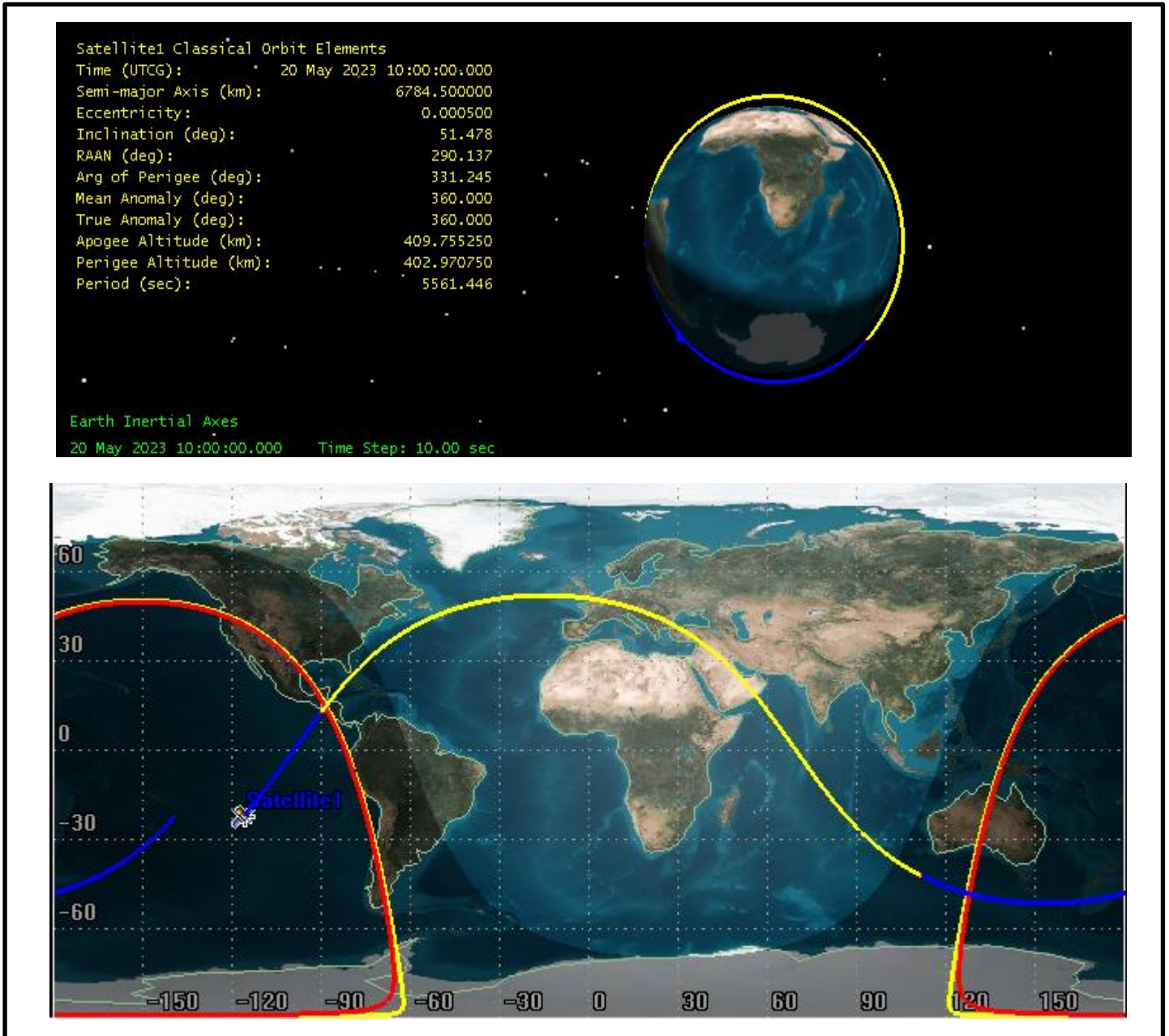


Figure 44 : 2D and 3D maps after setting all the orbital parameters

Like that we created a scenario with the name ISS and a satellite named satellite 1, we showed the sunlight phase and eclipse phase.

4.3.3.4 Results of the orbital analysis

After creating the scenario, and the satellite, with necessary information for orbit and geometry and other thermal data. And we used the report and graph manager tool, to get our results.

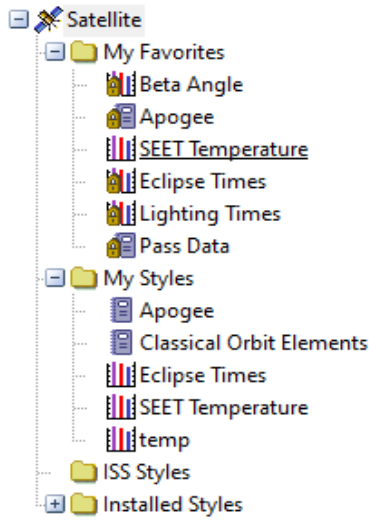


Figure 45 : report and graph manager list

- **Report for classical orbit parameters**

The report was too large so we represent it in parts showed in figure 46:

Time (UTCG)	Semi-major Axis (km)	Eccentricity	Inclination (deg)	RAAN (deg)	Arg of Perigee (deg)	Mean Anomaly (deg)	True Anomaly (deg)
20 May 2023 10:00:00.000	6784.500000	0.000500	51.478	290.137	331.245	360.000	360.000
20 May 2023 10:01:00.000	6784.500000	0.000500	51.478	290.137	331.245	3.884	3.888
20 May 2023 10:02:00.000	6784.500000	0.000500	51.478	290.137	331.245	7.768	7.776
20 May 2023 10:03:00.000	6784.500000	0.000500	51.478	290.137	331.245	11.652	11.663
20 May 2023 10:04:00.000	6784.500000	0.000500	51.478	290.137	331.245	15.536	15.551
20 May 2023 10:05:00.000	6784.500000	0.000500	51.478	290.137	331.245	19.419	19.438
20 May 2023 10:06:00.000	6784.500000	0.000500	51.478	290.137	331.245	23.303	23.326
20 May 2023 10:07:00.000	6784.500000	0.000500	51.478	290.137	331.245	27.187	27.213
20 May 2023 10:08:00.000	6784.500000	0.000500	51.478	290.137	331.245	31.071	31.101
20 May 2023 10:09:00.000	6784.500000	0.000500	51.478	290.137	331.245	34.955	34.988
20 May 2023 10:10:00.000	6784.500000	0.000500	51.478	290.137	331.245	38.839	38.875
20 May 2023 10:11:00.000	6784.500000	0.000500	51.478	290.137	331.245	42.723	42.762

Apogee Altitude (km)	Perigee Altitude (km)	Period (sec)
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446
409.755250	402.970750	5561.446

Figure 46 : report about classical orbit parameters and the period

And now we show this report in graph for better understanding.

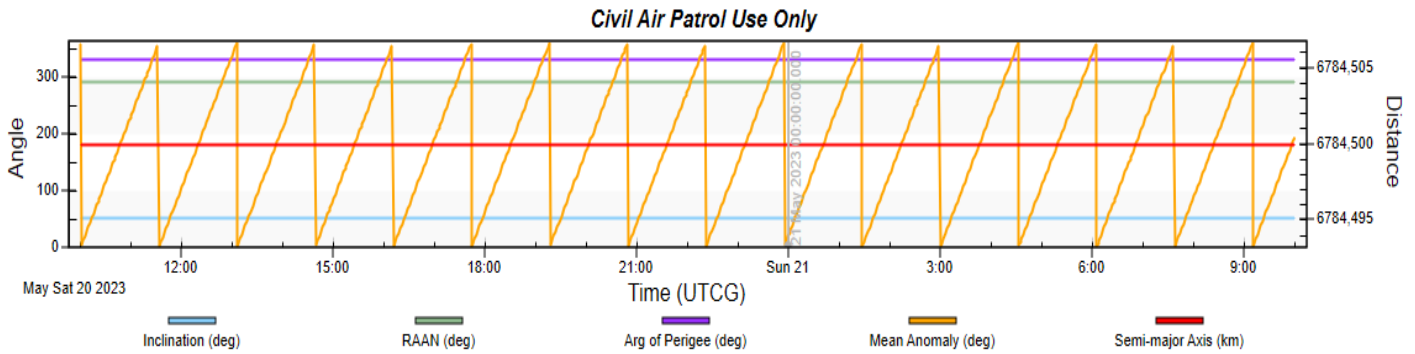


Figure 47 : graphical representation of the orbital parameters

From the report and the graph it is clear that all the parameters don't change with time, except mean anomaly and true anomaly because they represent the position of the satellite in specific time while orbiting, it can't be fixed obviously!!

The period of the orbit is 5561,446 seconds it equals 92, 6 minutes.

- **lightning**

Information about lightning times are represented in the graph below (figure 48), lightning it means sunlight and eclipse (umbra and penumbra).

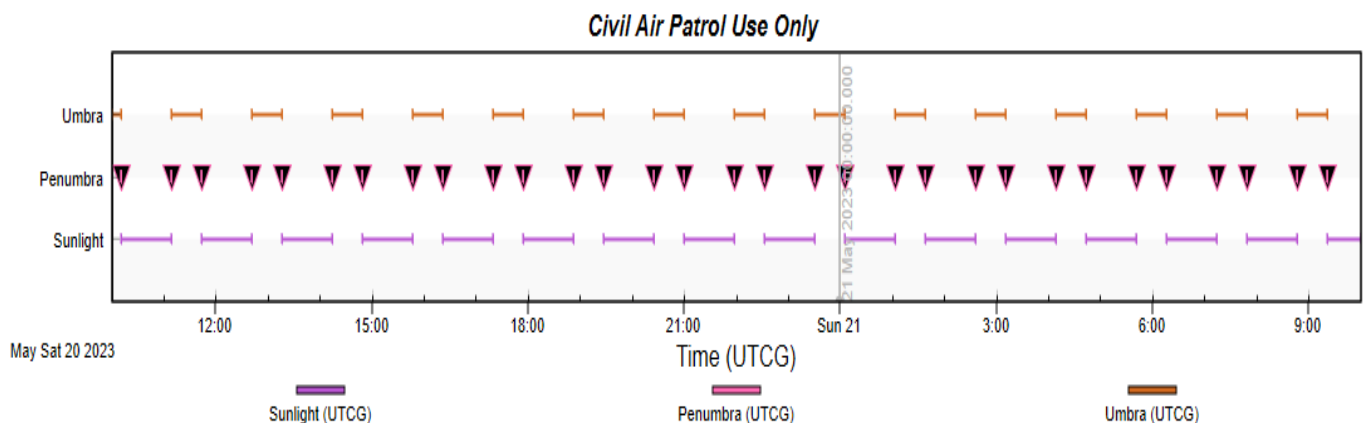


Figure 48 : graph for lightning times while orbiting

This figure is a good representation for the lightning times, we can see that sunlight period is the longest then it comes umbra(full darkness or eclipse figure (49) it last the half of the sunlight period (we will prove that later) and penumbra lasts for few minutes.

Civil Air Patrol Use Only

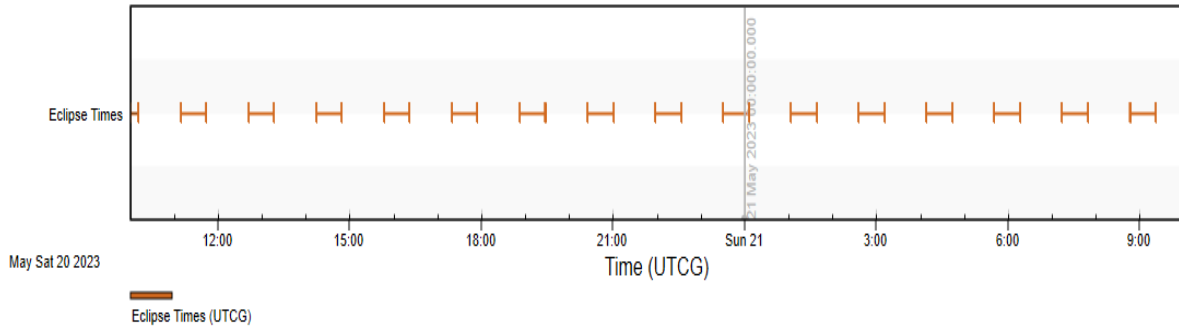


Figure 49 : graph for eclipse times while orbiting

And about start and end time of shadows is represented in the figure below plus the time of max shadow (total darkness) it means the coldest period while orbiting , it very important to know those times .

Penumbra Start Time (UTCG)	Umbra Start Time (UTCG)	Umbra Stop Time (UTCG)	Penumbra Stop Time (UTCG)	Obstruction	Duration (sec)
No Data	20 May 2023 10:00:00.000	20 May 2023 10:11:19.356	20 May 2023 10:11:28.189	Earth	688.189
20 May 2023 11:08:53.164	20 May 2023 11:09:02.110	20 May 2023 11:44:01.661	20 May 2023 11:44:10.490	Earth	2117.325
20 May 2023 12:41:35.265	20 May 2023 12:41:44.209	20 May 2023 13:16:43.967	20 May 2023 13:16:52.794	Earth	2117.529
20 May 2023 14:14:17.366	20 May 2023 14:14:26.307	20 May 2023 14:49:26.271	20 May 2023 14:49:35.093	Earth	2117.727
20 May 2023 15:46:59.470	20 May 2023 15:47:08.403	20 May 2023 16:22:08.575	20 May 2023 16:22:17.396	Earth	2117.926
20 May 2023 17:19:41.568	20 May 2023 17:19:50.500	20 May 2023 17:54:50.878	20 May 2023 17:54:59.693	Earth	2118.125
20 May 2023 18:52:23.669	20 May 2023 18:52:32.599	20 May 2023 19:27:33.180	20 May 2023 19:27:41.994	Earth	2118.325
20 May 2023 20:25:05.768	20 May 2023 20:25:14.694	20 May 2023 21:00:15.482	20 May 2023 21:00:24.293	Earth	2118.525

Time of Max Shadow (UTCG)
20 May 2023 10:00:00.000
20 May 2023 11:09:02.110
20 May 2023 12:41:44.209
20 May 2023 14:14:26.307
20 May 2023 15:47:08.403
20 May 2023 17:19:50.500
20 May 2023 18:52:32.599
20 May 2023 20:25:14.694
20 May 2023 21:57:56.791
20 May 2023 23:30:38.887
21 May 2023 01:03:20.983
21 May 2023 02:36:03.079
21 May 2023 04:08:45.175
21 May 2023 05:41:27.272
21 May 2023 07:14:09.367
21 May 2023 08:46:51.462

Figure 50 : start and stop times of penumbra and umbra plus time of max shadow

Maximum duration of eclipse is 2118,525 seconds its approximaly 35 min, and the rest is sunlight phase.

The period of the orbit is 92, 6 min from the orbit parameters results and eclipse is 35 min, so confirmed that eclipse time is half of the sunlight time.

- **Solar flux**

From the theoretical study, solar flux has relation with the time that the cubesat send 100% in direct sunlight, and there's no solar flux 0% in eclipse phase. And this is what we had as results from STK it confirmed that. More information in the thermal analysis part.

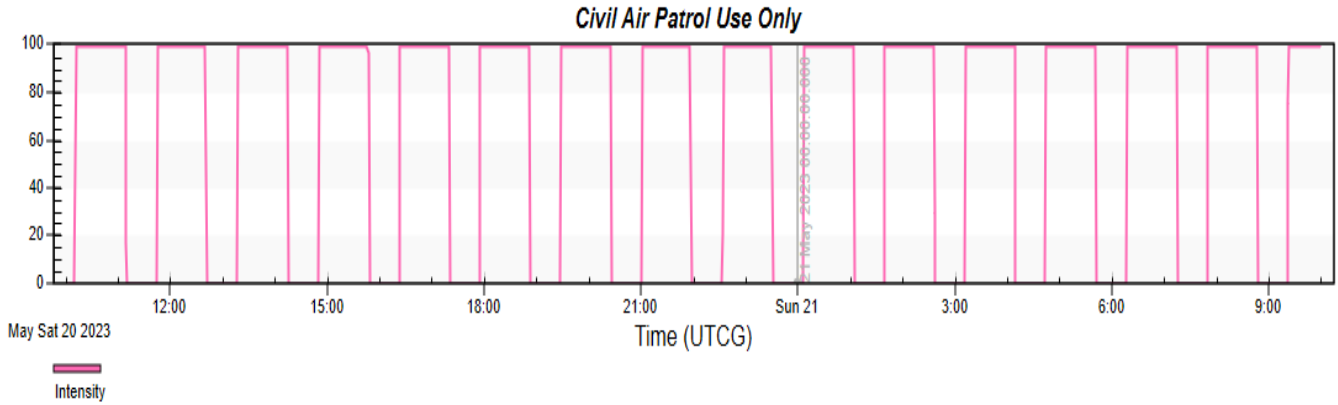


Figure 51 : time evolution of solar flux

- **Satellite temperature**

The graph of the satellite temperature found in (figure 52) and reports in (figure 53) and (figure 54) representing max and min values of temperatures, and it is seen that the maximum of temperature the satellite remains is while exposing to direct sunlight and it is about 53, 78°C /326, 93°K and the minimum temperature is in the max shadow time it is about -80.162°C/192, 98°K.

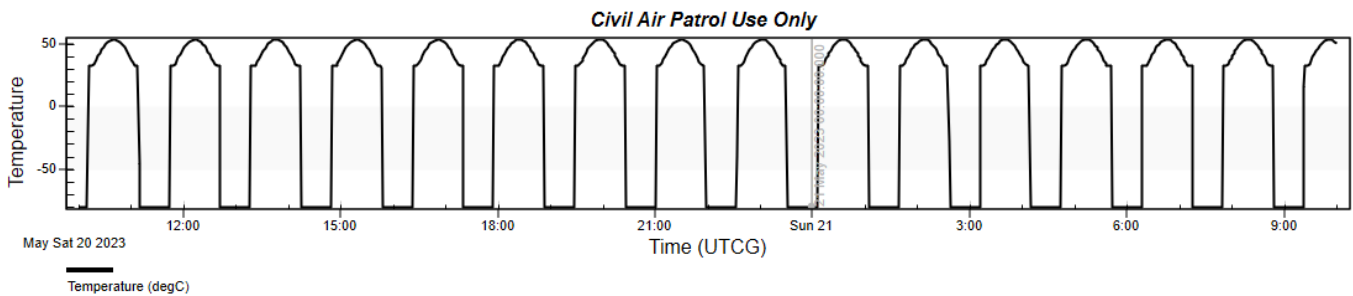


Figure 52 : time evolution of satellite temperature with STK

Time (UTCG)	Temperature (degC)	Solar Flux (W/m^2)	Solar Intensity (%)
20 May 2023 10:00:00.000	-80.162	1333.310261	0.0000
20 May 2023 10:01:00.000	-80.162	1333.306050	0.0000
20 May 2023 10:02:00.000	-80.163	1333.301975	0.0000
20 May 2023 10:03:00.000	-80.165	1333.298053	0.0000
20 May 2023 10:04:00.000	-80.167	1333.294301	0.0000
20 May 2023 10:05:00.000	-80.169	1333.290734	0.0000
20 May 2023 10:06:00.000	-80.172	1333.287367	0.0000
20 May 2023 10:07:00.000	-80.175	1333.284214	0.0000
20 May 2023 10:08:00.000	-80.179	1333.281287	0.0000
20 May 2023 10:09:00.000	-80.182	1333.278599	0.0000
20 May 2023 10:10:00.000	-80.187	1333.276159	0.0000
20 May 2023 10:11:00.000	-80.191	1333.273978	0.0000
20 May 2023 10:12:00.000	32.857	1333.272064	100.0000

Figure 53 : satellite temperature report shows the minimum temperature in °C

We will detail this more in the thermal analysis part, taken albedo in consideration and IR of earth.

20 May 2023 10:28:00.000	47.553	1333.279514	100.0000
20 May 2023 10:29:00.000	48.516	1333.282159	100.0000
20 May 2023 10:30:00.000	49.403	1333.285002	100.0000
20 May 2023 10:31:00.000	50.214	1333.288029	100.0000
20 May 2023 10:32:00.000	50.945	1333.291223	100.0000
20 May 2023 10:33:00.000	51.596	1333.294569	100.0000
20 May 2023 10:34:00.000	52.164	1333.298049	100.0000
20 May 2023 10:35:00.000	52.649	1333.301645	100.0000
20 May 2023 10:36:00.000	53.049	1333.305341	100.0000
20 May 2023 10:37:00.000	53.363	1333.309116	100.0000
20 May 2023 10:38:00.000	53.592	1333.312953	100.0000
20 May 2023 10:39:00.000	53.733	1333.316831	100.0000
20 May 2023 10:40:00.000	53.788	1333.320731	100.0000
20 May 2023 10:41:00.000	53.756	1333.324634	100.0000
20 May 2023 10:42:00.000	53.637	1333.328521	100.0000
20 May 2023 10:43:00.000	53.431	1333.332371	100.0000
20 May 2023 10:44:00.000	53.139	1333.336166	100.0000
20 May 2023 10:45:00.000	52.762	1333.339886	100.0000
20 May 2023 10:46:00.000	52.299	1333.343512	100.0000
20 May 2023 10:47:00.000	51.753	1333.347027	100.0000
20 May 2023 10:48:00.000	51.125	1333.350413	100.0000

Figure 54 : max temperature in °C of the satellite taken from the same report

4.3.3.5 Integrating STK with other applications

STK can be connected or controlled from other applications. Both integrations techniques can make use of the connect scripting language to accomplish the task [46].

STK can be driven from a script that is run from the STK internal web browser in the free version of the tool. To control STK from an external source requires the STK integration module [46]. We can find examples in the STK help files.

When talking about integration we talk about STK Programming Interface, this Programming Interface offers a wide variety of options to automate and customize STK and to integrate its technology into other applications...

The STK Programming Interface help can be accessed from the Help menu in STK.

STK provides vehicle dynamics, sensors, communications, visualization..., and other applications provide numerical analysis, advanced plotting capabilities and a programming environment.

The application that we are using for thermal analysis is MATLAB. While searching and studying we found that it can be a connection between the two application (STK and MATLAB), but we didn't use this connection for the results, but it is so fun. And for more we found that we can connect between STK and EXCEL too!!

- The connect command used is: object model: root.executecommand (command).
- Code snippets are available in the STK programming interface help.
- **Integration STK with excel**

The integrating with EXCEL can be by creating buttons , each button do its job , starting with launch STK ,creating a new scenario and ending with creating the satellite , all that from EXCEL only (figure 55).

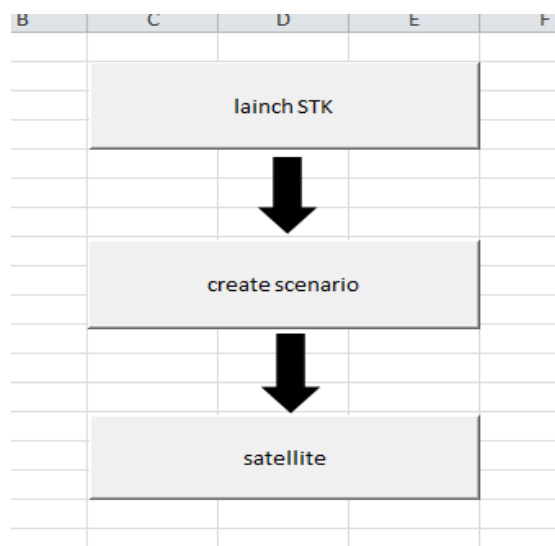


Figure 55 : Excel buttons to start STK

The first button is to run STK ,second button is to create a scenario it has all information about scenario time and the name of scenario , the last one is to create a satellite with keplerian orbit parameters . Scripts to create these are from the STK programming interface.

- **Integration STK with MATLAB**

This integration is so helpful so that we can automate, extend or visualize the analysis wanted [46].

- Automate: access the capabilities of both tools without the need to switch between application sessions.

- extend: STK and MATLAB can be used together to extend or augment the types of analysis that you can perform with just one of them alone.

- visualize: MATLAB users can use STK's 2D and 3D visualization to view any geo-referenced or geometric MATLAB data, such as position or attitude information ...

There are two interfaces to connect STK and MATLAB:

- MexConnect - this interface communicates over TCP/IP
- COM - this interface communicates over a COM connection

To choose the best interface to use, we have to get a plan to accomplish with the STK MATLAB integration and then expand the decision tree and follow the steps provided [44].

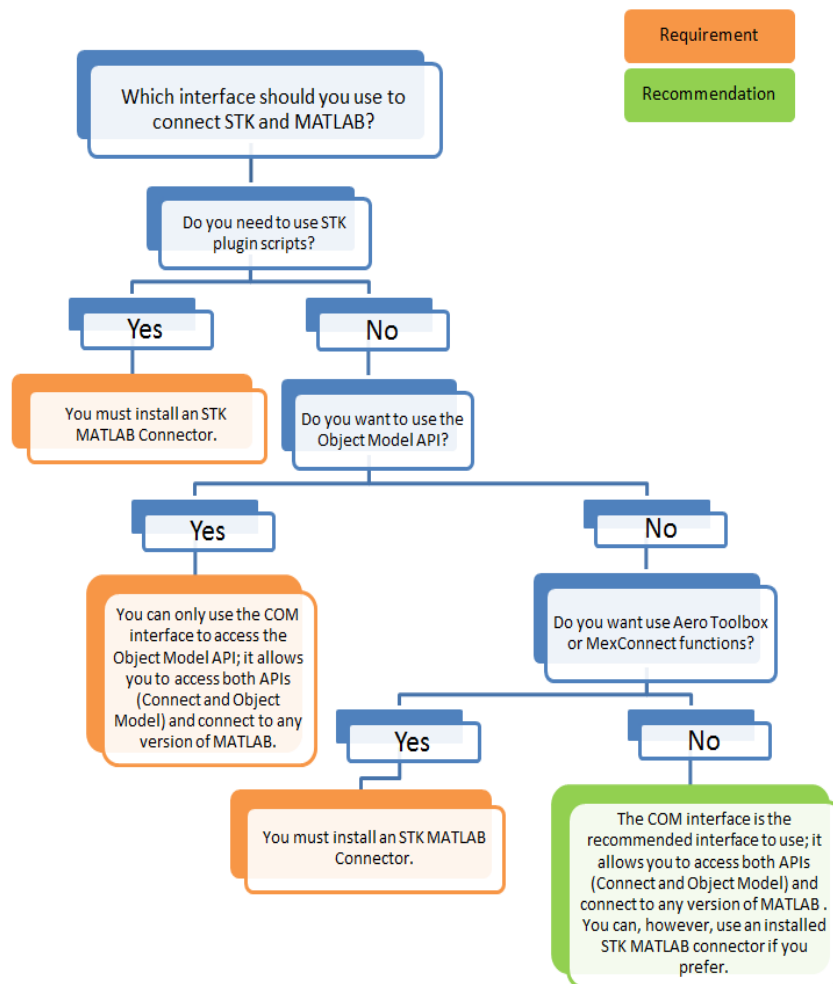


Figure 56 : interface decision tree [46]

After that we do the first step with integrating, opening MATLAB any version and then we type these 2 lines to initialize and launch STK.

```
App = actxserver ('STK11.application');
```

root = app.Personality2;

Then typing a script to create a scenario and a satellite, you will find it in the annex.

- My example to connect STK with MATLAB R2015a

```

Command Window
>> app = actxserver('STK11.application');
root = app.Personality2;
scenario = root.Children.New('eScenario','leosat');
scenario.SetTimePeriod('Today','+24hr')
root.ExecuteCommand('Animate * Reset')
satellite = scenario.Children.New('eSatellite','LeoSat');
root.ExecuteCommand(['SetState */Satellite/LeoSat Classical TwoBody "',sc
% IAgSatellite satellite: Satellite object
lighting = satellite.Graphics.Lighting;
%Settings for vehicle in sunlight
sunlight = lighting.Sunlight;
sunlight.Visible = true;
sunlight.Color = 65535; %yellow
sunlight.LineWidth = 3;
%Settings for vehicle in penumbra
penumbra = lighting.Penumbra;
penumbra.Visible = true;
penumbra.Color = 42495; %orange
penumbra.LineWidth = 2;
%Settings for vehicle in umbra
umbra = lighting.Umbra;
umbra.Visible = true;
umbra.Color = 255; %red
umbra.LineWidth = 1;
root.CloseScenario;
fx ans =
  
```

The last line is to close the scenario, if you want to keep changing don't type that line.

4.3.4 Thermal analysis

It mentioned before that the thermal analysis has two states, steady state and transient analysis .and we have these constants:

Table 8: hot case parameters and cold case parameters

	hot case	cold case
s	1414	0
IR	261	189
albedo	0,35	0,2

For area we have the following table :

Table 9 : area values

Total area (m ²)	0,14
Cross section area (m ²)	0,035

Table 10: some parameters that we need in calculation.

Stephan Boltzmann constant (w/m ² K)	5,67E-08
Mass of the cubesat (kg)	3
Specific heat(J/Kg.K)	879

We must note that the values below are the range of temperatures that satellite can experience without TCS, so when using the right TCS, temperature of the satellite will be in the allowable range of temperatures while orbiting.

Another note: the infrared constant is equal to $\epsilon_p \sigma T_p^4$, in calculation; we are going to use these constants

4.3.4.1 Steady thermal analysis

To obtain a general idea of the temperature limits that our 3U CubeSat will experience, for that we study two cases a hot case and it means the highest temperature of the cubesat while exposing to direct sunlight and a cold case which means the lowest temperature of the cubesat while it is in the eclipse phase .The difference between the hot and cold cases is that there is zero solar radiation and albedo radiation in the cold case. Knowing these two temperatures is important for the success of the mission and to make the cubesat lifetime longer .all the numerical calculation of this steady state are done with EXCEL, and we assume that there is no internal heat dissipation inside the satellite.

- **Hot case**

To determine the temperature of the CubeSat in this case, the energy balance described by Equation (3-1) is used.

Since a steady state is being considered, energy will not vary with time, and there is no heat dissipation so the energy balance will be:

$$\dot{Q}_{in} = \dot{Q}_{out} \quad (4-7)$$

And \dot{Q}_{in} is given by equation (3-2), and \dot{Q}_{out} is given by equation (3-4), so equation (4-7) will be:

$$\dot{Q}_s + \dot{Q}_a + \dot{Q}_{iR} = \epsilon \sigma A_s T_s^4 \quad (4-8)$$

Now replacing with equations (2-28), (2-19), (2-20) we get:

$$\alpha S A \cos \theta + \alpha S A F_{bp} A_f \cos \theta + A \epsilon_b F_{bp} \epsilon_p \sigma T_p^4 = \epsilon \sigma A_s T_s^4 \quad (4-9)$$

For spherical geometry we set that:

$$A \cos \theta = \text{the cross section area} = A_{\text{cross}} \quad (4-10)$$

Equation (4-9) becomes:

$$\alpha S_{\text{Across}} + \alpha S_{\text{Across}} F_{bp} A_f + A_{\text{Across}} \epsilon_b F_{bp} \epsilon_p \sigma T_p^4 = \epsilon \sigma A_s T_s^4 \quad (4-11)$$

F_{bp} Is calculated with this equation:

$$F_{bp} = \left(\frac{R}{R+H} \right)^2 = 0,8853 \quad (4-12)$$

Where $R = 6370\text{km}$ and $H=400\text{ km}$. and T_p is the earth's temperature 288°K . Emissivities and absorptivity values are in the table (4-1), we set emissivity of earth is 0.6 and the other constants are in tables (4-2), (4-3), (4-4).

After rearrangement of equation (4-5), we get the equation (4-8), and we solve it using EXCEL.

$$T_s^4 = \frac{\alpha S_{\text{Across}} + A \epsilon_b F_{bp} \epsilon_p \sigma T_p^4 + \alpha S_{\text{Af}} A_{\text{Across}} F_{bp}}{\epsilon A \sigma} \quad (4-13)$$

$$T_s = \left(\frac{\alpha S_{\text{Across}} + A \epsilon_b F_{bp} \epsilon_p \sigma T_p^4 + \alpha S_{\text{Af}} A_{\text{Across}} F_{bp}}{\epsilon A \sigma} \right)^{\frac{1}{4}} \quad (4-14)$$

For easier calculation, we calculate heat fluxes alone then we solve the equation (4-8), like represented in the figure:

steady state: hot case					
some constantes for calculations				hot case calculations	
A tot	Across		alpha	epsilon	Qs
0,14	0,03		0,627	0,459	26,597 w
solar constante		albedo constant		Qa	8,2413 w
hot case	cold case	hot	cold	QiR	14,848 w
1414	1318	0,35	0,2	Q tot	49,687 w
infrared		view factor		temperature	
hot	cold	0,8853	6E-08	T in °K	341,73
261	189			T in °C	68,727

Figure 57 : hot case calculations

The CubeSat temperature obtained for the hot case is approximately $69^\circ\text{C}/342^\circ\text{K}$.

- **Cold case**

The calculations for the cold case followed the same steps as that for the hot case. The main difference is that the solar radiation and albedo are zero. The albedo is zero because there is no reflected solar radiation when the CubeSat is in eclipse phase. The only radiation the CubeSat is exposed to and absorbs is Earth's infrared.

First we are going to use equation (4-7), then we put albedo heat equals zero and solar heat is equal zero too, and only infrared absorption, so equation (4-8) will be:

$$\dot{Q}_{iR} = \epsilon \sigma A_s T_s^4 \quad (4-15)$$

With rearrangement of equation (4-15), we get:

$$T_s^4 = \frac{\dot{Q}_{iR}}{\epsilon A \sigma} \quad (4-16)$$

Replacing \dot{Q}_{iR} with its equation (2-20), we get equation (4-17), and this is the equation that we are going to solve using EXCEL and the parameters of the cold case:

$$T_s = \left(\frac{A \epsilon_b F_{bp} \epsilon_p \sigma T_p^4}{\epsilon A \sigma} \right)^{\frac{1}{4}} \quad (4-17)$$

Results are in the (figure 58), we must note that we used the IR constant during the eclipse, so we take the minimum values to get the lowest temperature it can be:

steady state: cold case					
some constantes for calculations					
A tot	Across		alpha	epsilon	
0,14	0,03		0,627	0,459	
solar constante		albedo constant			
hot case	cold case	hot	cold		
1414	0	0,35	0,2		
infrared		view factor			
hot	cold	0,8853		sig	
261	189			6E-08	
cold case calculations					
Qs	0	w			
Qa	0	w			
QiR	10,752	w			
Qtot	10,752	w			
↓					
temperature					
T in °K	233,07				
T in °C	-39,927				

Figure 58 : cold case calculations

The CubeSat temperature obtained for the cold case is approximately $-40^{\circ}\text{C}/233^{\circ}\text{K}$.

- **commentaire :**

The calculated temperatures give the survival temperature limits for a CubeSat with no TCS; the temperatures are between 69°C and -40 °C. (Figure 59) gives some typical temperature limits for various spacecraft components.

Component or Subsystem	Operating Temperature (°C)	Survival Temperature (°C)
General electronics	-10 to 45	-30 to 60
Batteries	0 to 10	-5 to 20
Infrared detectors	-269 to -173	-269 to 35
Solid-state particle detectors	-35 to 0	-35 to 35
Motors	0 to 50	-20 to 70
Solar panels	-100 to 125	-100 to 125

Figure 59 : temperature limits for some spacecraft components [47]

When the hot and cold case temperatures are compared to the operational and survival temperatures for the components in (figure 59), it can be seen that except for the solar panels, the CubeSat will be too cold for the components when the spacecraft is in eclipse. The only components that can survive both cases are the solar panels. And here we can see the important of using TCS.

4.3.4.2 Transient thermal analysis

The temperatures calculated in the steady thermal analysis shows temperature limits that the CubeSat can experience. Therefore, the CubeSat at any point along its orbit is expected to have a temperature that is between those values.

For the transient thermal analysis the same mission profile as that in steady analysis is assumed. In addition, the trajectory of the CubeSat is assumed to be as shown the orbital analysis part. And equation (3-17) is solved with a MATLAB code.

- **eclipse**

In this part , we must know the position of the cubesat if it is exposed to sunlight or it is in the eclipse, which means we have to know the start angle of eclipse and stop angle too, to that we use equations (3-5), (3-6), (3-9), we used those equation in MATLAB code to obtain the results.

The period of the orbit is $T_0 = 5561,446$ seconds it equals 92, 6 minutes, taken from the orbital analysis.

Results are shown in figure taken from MATLAB:

beta	0.0000 + 1.0000i
beta_max	70.2059
G	6.6700e-11
h	1.0628
H	400
M	5.9700e+24
phi	1x97 double
phi_e	250.2091
phi_s	109.7909
R	6370
t	1x97 double
T0	5.5614e+03
Te	2.1692e+03

Figure 60 : eclipse duration and its start and stop angles taken from MATLAB

The angles at which eclipse starts and ends are 109, 7° and 250, 2° respectively. For summarize as we can see in table (11):

Table 11: eclipse information

Eclipse duration	2169,2 (s) ~ 36 min
Eclipse start angle	109,7°
Eclipse stop angle	250,2°

- **heat fluxes**

For this part we used MATLAB code for calculations, and results are for one orbit duration.

For the solar heat flux we see from the figure (61) that the cubesat during the orbit it receives is 30 w of energy when it is exposed to sunlight and zero when it is eclipse.

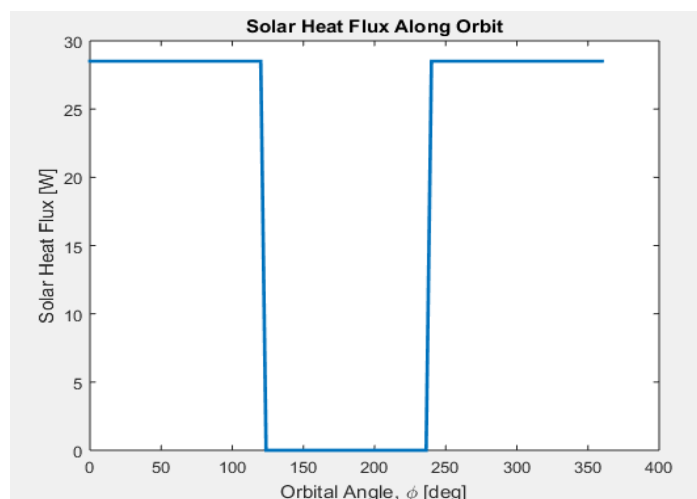


Figure 61 : solar heat flux along orbit

The heat absorbed resulting from albedo plotted in figure (62) , and it shows that during eclipse albedo is equal to zero , and it reach a maximum while the cubesat is exposed to direct sunlight. We notice also that

the eclipse for albedo variation is not like the solar fluxes, if there is solar flux it's if it is shadow it zero, but for albedo we see that there is some albedo in shadow in the penumbra but it is less than it is in the direct sun obviously.

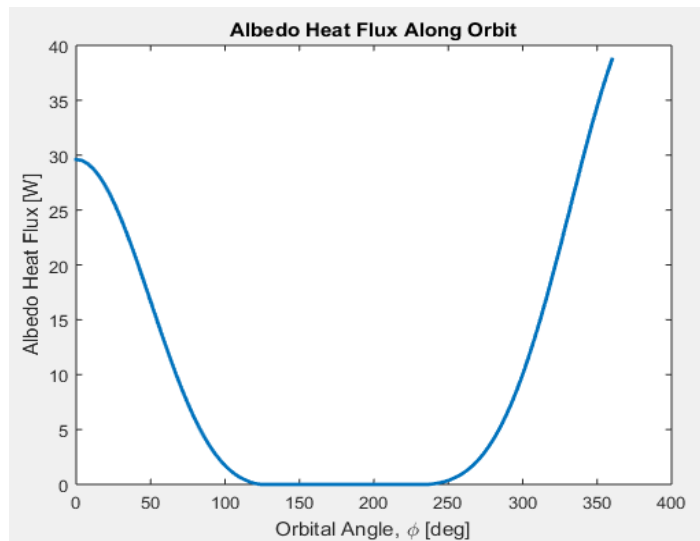


Figure 62 : albedo flux along orbit

Infrared flux is represented in figure (63); it shows that infrared absorbed is always the same value during all the orbit and it is approximately 13,5 watt.

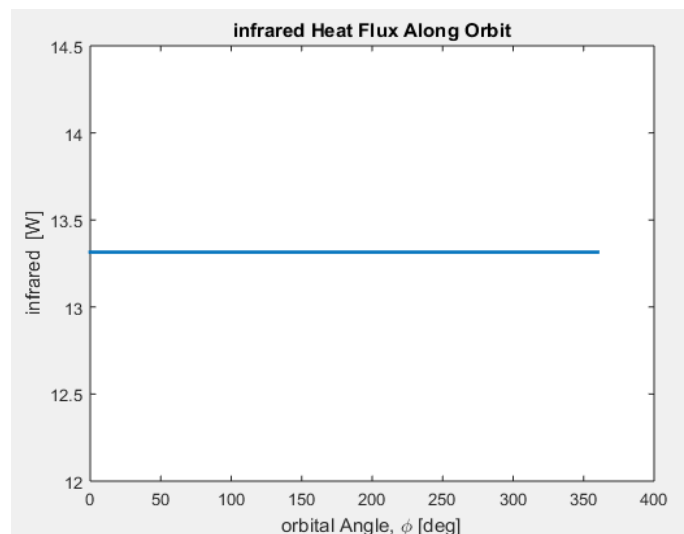


Figure 63 : infrared flux along orbit

- **temperature of satellite**

The temperatures calculated in steady state predict the upper and lower temperature limits of the CubeSat and the temperature at any point along the orbit of the CubeSat would exist somewhere between those limits.

And we take the values calculated in the steady state as initial conditions for the transient state.

In the theoretical we said that the differential equations can be solved with range kutta method or Euler method, however, an alternative method was used. An iterative process was used in MATLAB to calculate the temperature of the CubeSat at various points along its orbit .First, an initial temperature was assumed. In this case, the hot case temperature of 69 °C, which was calculated in the hot case of the steady state, was assumed. This initial temperature was substituted into the temperature term on the right hand side of Equation (3-17), and the differential $dT/d\phi$ was calculated. The differential $dT/d\phi$ indicated how much the temperature changed from one angle to the next. This change in temperature was then added to the initial temperature, which yielded a new temperature. This process was repeated with the new temperature acting as the initial temperature.

The calcutions are done with $dT/d\phi$, wich mean arording to the position of the cubesat we calculat the new temperature, then we convert angle to time .to convert from angle to time or from time to angle we use equation (3-18).

1. hot case

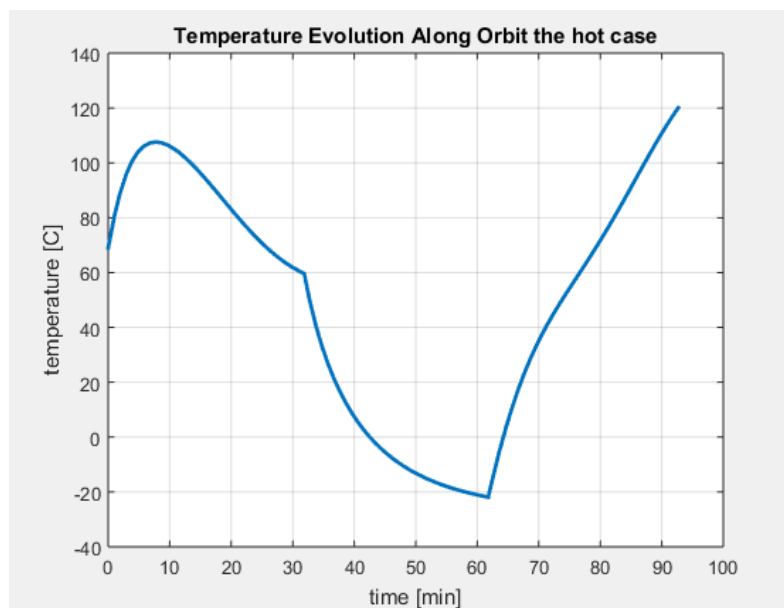


Figure 64 : the cubesat temperature changes along orbit for the hot case

From this figure (64), we see that the highest temperature the cubesat can get is 120°C/393°K, and the lowest hot case temperature is about -20°C/253°K.

2. cold case

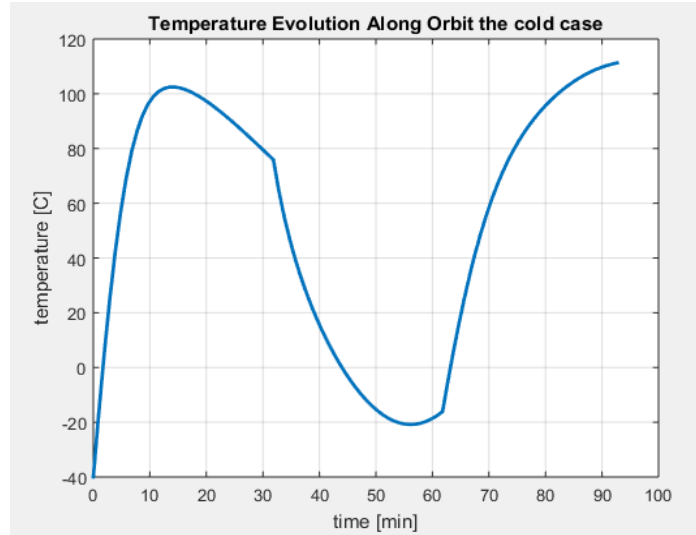


Figure 65 : the cubesat temperature changes along orbit for the cold case

From this figure (65), we see that the highest temperature the cubesat can get is about $85^{\circ}\text{C}/358^{\circ}\text{K}$, and the lowest cold case temperature is about $-40^{\circ}\text{C}/233^{\circ}\text{K}$.

3. commentaire

The (table 12) is a conclusion of the transient state temperature while the cubesat is orbiting, so the limits are $[-40^{\circ}\text{C}, 120^{\circ}\text{C}]$.

Table 12: transient state summary

Transient state : one node model results			
Hot case		Cold case	
highest temperature	lowest temperature	highest temperature	lowest temperature
$120^{\circ}\text{C}/393^{\circ}\text{K}$	$-22^{\circ}\text{C}/251^{\circ}\text{K}$	$85^{\circ}\text{C}/358^{\circ}\text{K}$	$-40^{\circ}\text{C}/233^{\circ}\text{K}$

4.3.4.3 Thermal analysis comments

The maximum and minimum values of the transient state do not agree perfectly with the temperatures calculated using the steady state model. In fact, it shows that the CubeSat will experience a temperature higher than it is founded in the steady state it is 120°C . For the steady state case minimum temperature was -40°C , and in the transient it can't assume lower than in temperature in steady state.

Comparing the results from our thermal analysis using one node model and the results of the Korean satellite (figure 23), we see that both analysis follow the same general changes.

4.4 six-node analysis for 3U cubesat [57]

Since it is a 3U cubesat we consider each side as a lumped mass (one node each side so there is 6 nodes connected to each other) each side is with a surface area equal to what it would actually be on a 3U CubeSat, all sides are assumed to have the same thermo-optical properties as the single-node model.

A key point to remember when doing the mathematical modeling of thermal problems is that it is nonsense to start demanding great accuracy in the solution when there is not such accuracy in the input parameters and constraints. Without specific experimental tests, there are big uncertainties even in materials properties, like thermal conductivity of metal alloys, entrance and blocking effects in convection, and particularly uncertainty in thermo-optical properties.

Usually the one node thermal model of the satellite results in temperatures higher than actual, because it has very low accuracy because it does not consider conduction between surfaces and properties of each panel. But in the six-node model the resulting range of temperatures from the analysis are more accurate.

The advantage of multi-node model is that it allows to consider heat transfer between the parts (by conduction through the joints, and radiation when they see each other, since convection will be absent in the vacuum of space, or even in pressurized boxes under microgravity).

4.4.1 the geometric model

The geometric model, is the real cubesat represented by a cube with the real surface, so each side is considered as a node, so we obtain six nodes, the geometric model is shown in (figure 66).

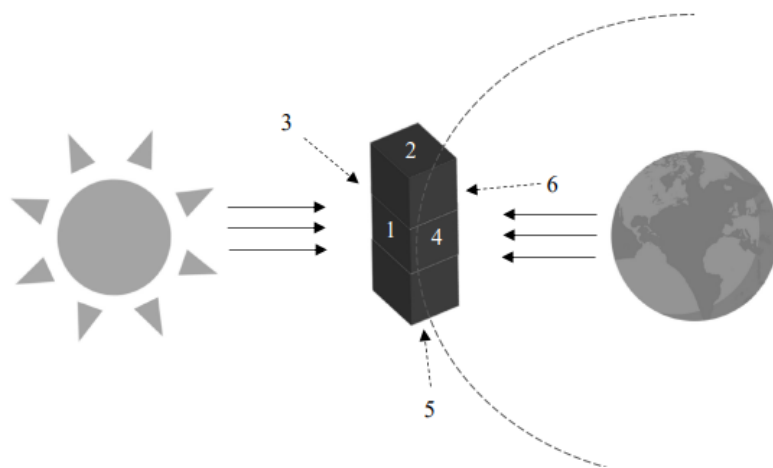


Figure 66 : six node geometric model

4.4.2 The thermal energy balance [57]

The first equation is obviously the thermal balance equation noted (3-15), but we said before there are 6 nodes, each side is represented as a node, also the heat transfer between nodes, so the thermal energy balance for node i in an N-node spacecraft may be written as:

$$(mcp)_i \frac{dT_i}{dt} = \dot{Q}_{in} - \dot{Q}_{out} + \sum_j^N \dot{Q}_{con\ i,j} + \sum_j^N \dot{Q}_{rad\ i,j} \quad (4-18)$$

As it is shown in the equation (4-18), there is an term refer to the conduction between nodes $\sum_j^N \dot{Q}_{con\ i,j}$ and the radiation too by the term $\sum_j^N \dot{Q}_{rad\ i,j}$, also these terms are called node coupling.

Notice that, in many computer packages, two external nodes are added, the planet (or moon) and the background, and then there is no explicit infrared input-from-planet term, and node-own-emission term, because they are included in the summation for node radiation exchanges.

4.4.3 Node coupling [57]

The heat input to node i by conduction coming from the other nodes, can always be written as:

$$\sum_j^N \dot{Q}_{con\ i,j} = \sum_j^N \frac{k_{ij,eff} A_{ij,eff}}{L_{ij,eff}} (T_j - T_i) \quad (4-19)$$

Where conduction coupling, is stated as an effective conductivity of the materials implied ($k_{ij,eff}$), times an effective heat-flow area ($A_{ij,eff}$), divided by an effective distance between nodes $L_{ij,eff}$, can be noted as G_{ij} .

The heat input to node i by radiation, coming from the other nodes, may be written as:

$$\sum_j^N \dot{Q}_{rad\ i,j} = \sum_j^N \sigma R_{ij} (T_j^4 - T_i^4) \quad (4-20)$$

Where R_{ij} is the radiative coupling, which coincides with the view factor time's area.

4.4.4 Node matrix [57]

A node matrix of N·N thermal couplings amongst the N nodes and thermal capacities are included in the diagonal; e.g. see (table):

- Radiative couplings between node i and node j in the upper triangular side of the matrix, R_{ij} .
- Thermal capacities of each node i in the diagonal of the matrix, $m_i c_{pi}$
- Conductive couplings between node i and node j in the lower triangular side of the matrix, G_{ij} .

Table 13: matrix of thermal couplings and thermal capacities

i/j	1	2	3	4	5	6
1	$m_1 c_1$	R_{12}	R_{13}	R_{14}	R_{15}	R_{16}
2	G_{21}	$m_2 c_2$	R_{23}	R_{24}	R_{25}	R_{26}
3	G_{31}	G_{32}	$m_3 c_3$	R_{34}	R_{35}	R_{36}
4	G_{41}	G_{42}	G_{43}	$m_4 c_4$	R_{45}	R_{46}
5	G_{51}	G_{52}	G_{53}	G_{54}	$m_5 c_5$	R_{56}
6	G_{61}	G_{62}	G_{63}	G_{64}	G_{65}	$m_6 c_6$

4.4.5 Analytical six -nodes sinusoidal solution

In the one-node model developed above, the input thermal loads follow an up-and-down pattern (high gains when under sunshine, low gains under eclipse), so that a cosine modulation over the average may be a suitable first approximation. We may linearize the problem and get an analytical result.

In the one-node case, the energy equation was given by equation (3-15):

$$\dot{Q}_{in} - \dot{Q}_{out} = m c \frac{dT}{dt}$$

(Figure 67) represented the node coupling between node (i) and the other nodes, and each node we do the same coupling representation.

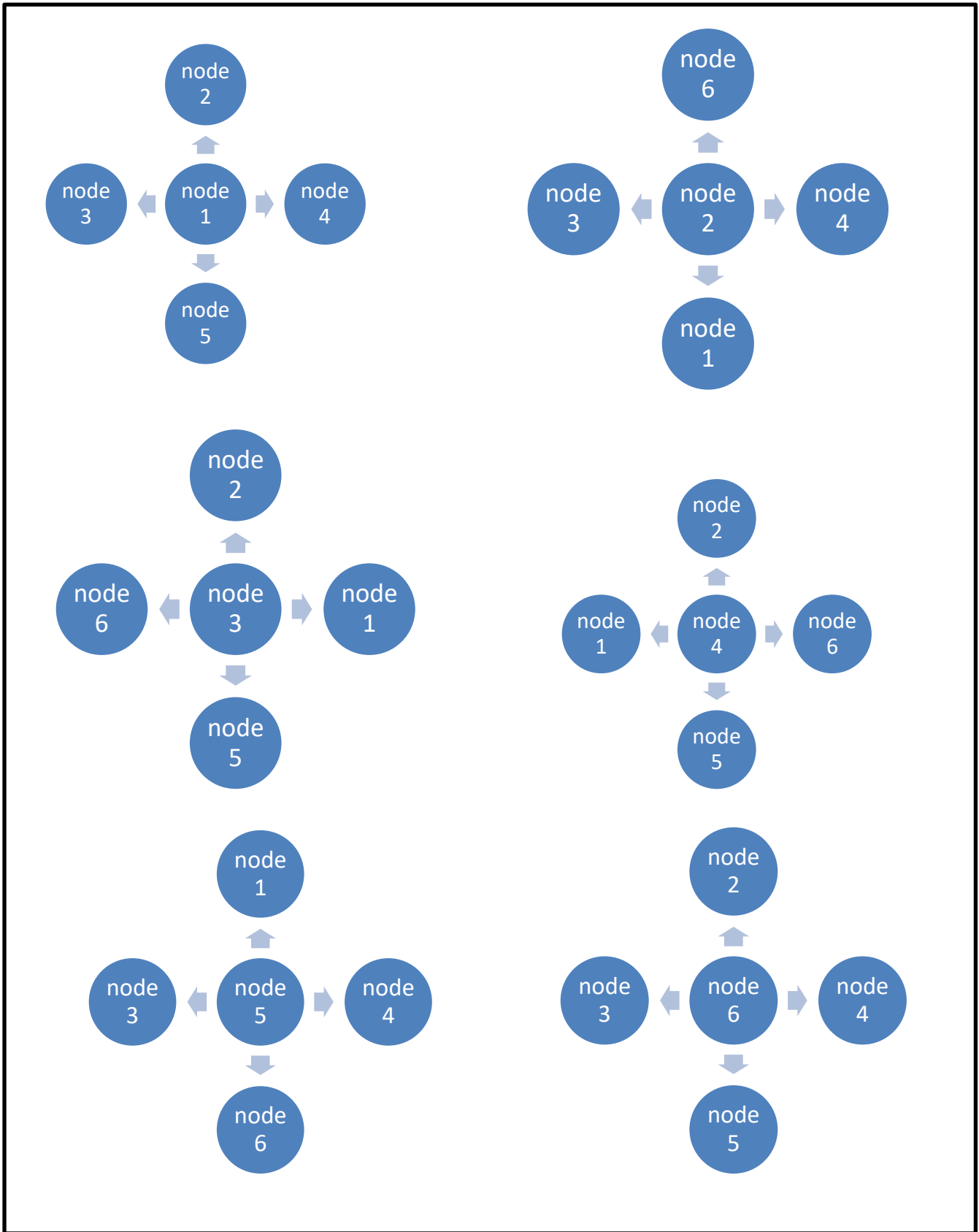


Figure 67 : representation of the coupling between node (i) and other nodes

Now with six- nodes we can introduce conductive and radiative couplings between them. We intend to make the same setting the eclipse and albedo factors $F_e = F_a = (1 + \cos\phi)/2$. Now the energy balance equation at each node is:

$$\begin{aligned}
(mcp)_1 \frac{dT_1}{dt} &= (\dot{Q}_{in} - \dot{Q}_{out})_1 + \dot{Q}_{con 1,2} + \dot{Q}_{con 1,3} + \dot{Q}_{con 1,4} + \dot{Q}_{con 1,5} \\
(mcp)_2 \frac{dT_2}{dt} &= (\dot{Q}_{in} - \dot{Q}_{out})_2 + \dot{Q}_{con 1,2} + \dot{Q}_{con 2,3} + \dot{Q}_{con 2,6} + \dot{Q}_{con 2,4} \\
(mcp)_3 \frac{dT_3}{dt} &= (\dot{Q}_{in} - \dot{Q}_{out})_3 + \dot{Q}_{con 3,1} + \dot{Q}_{con 3,2} + \dot{Q}_{con 3,5} + \dot{Q}_{con 3,6} \\
(mcp)_4 \frac{dT_4}{dt} &= (\dot{Q}_{in} - \dot{Q}_{out})_4 + \dot{Q}_{con 4,1} + \dot{Q}_{con 4,2} + \dot{Q}_{con 4,5} + \dot{Q}_{con 4,6} \\
(mcp)_5 \frac{dT_5}{dt} &= (\dot{Q}_{in} - \dot{Q}_{out})_5 + \dot{Q}_{con 5,1} + \dot{Q}_{con 5,3} + \dot{Q}_{con 5,4} + \dot{Q}_{con 5,6} \\
(mcp)_6 \frac{dT_6}{dt} &= (\dot{Q}_{in} - \dot{Q}_{out})_6 + \dot{Q}_{con 6,2} + \dot{Q}_{con 6,3} + \dot{Q}_{con 6,4} + \dot{Q}_{con 6,5}
\end{aligned} \tag{4-21}$$

And the answer we are looking for is, like it is in the one node but this time for six-node:

$$\begin{aligned}
T_1(\varnothing) &= T_{1m} + T_{1a} \cos(\varnothing - \varphi_1) \\
T_2(\varnothing) &= T_{2m} + T_{2a} \cos(\varnothing - \varphi_2) \\
T_3(\varnothing) &= T_{3m} + T_{3a} \cos(\varnothing - \varphi_3) \\
T_4(\varnothing) &= T_{4m} + T_{4a} \cos(\varnothing - \varphi_4) \\
T_5(\varnothing) &= T_{5m} + T_{5a} \cos(\varnothing - \varphi_5) \\
T_6(\varnothing) &= T_{6m} + T_{6a} \cos(\varnothing - \varphi_6)
\end{aligned} \tag{4-22}$$

When we expand all terms in (4-22) with this sinusoidal model, and we group in the independent-variable orthogonal functions, we get equations of the form:

$$\begin{aligned}
C_{10} + C_{1c} \cos(\varnothing) + C_{1s} \sin(\varnothing) &= 0 \\
C_{20} + C_{2c} \cos(\varnothing) + C_{2s} \sin(\varnothing) &= 0 \\
C_{30} + C_{3c} \cos(\varnothing) + C_{3s} \sin(\varnothing) &= 0 \\
C_{40} + C_{4c} \cos(\varnothing) + C_{4s} \sin(\varnothing) &= 0 \\
C_{50} + C_{5c} \cos(\varnothing) + C_{5s} \sin(\varnothing) &= 0 \\
C_{60} + C_{6c} \cos(\varnothing) + C_{6s} \sin(\varnothing) &= 0
\end{aligned} \tag{4-24}$$

Where the independent-term coefficients (C10 and C20), the cosine coefficients (C1c, C6c), and the sine coefficients (C1s ... C6s), are dependent on the geometry and material data.

By resolving these equations we get the analytical solution of six-node model.

4.4.6 Thermal analysis of six node model with STK

In this model, we consider, satellite's one face is directly looking to Sun and opposite face is directly looking to Earth while orbiting.

So we start to do the orbital analysis of side 1 , we consider it as a flat plat and its normal pointed to sun .

First step we open the previous scenario in the STK application , it named as ISS like we said previously, and we find the satellite that we created it before , the name is satellite 1, now we add new satellite and we name it satellite2, but this time we set it as a flat plat not a sphere like we did with satellite1.the orbital parameters are the same as the previous orbital simulation, and the same as ISS , we can find it in table (7).

The new object browser is as shown in figure (68).

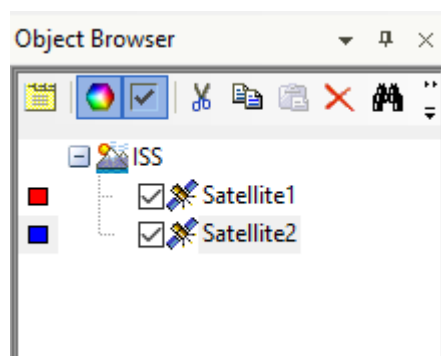


Figure 68 : object browser show both satellite1 and satellite 2.

Now we set the thermal parameters, we keep same values as satellite1, and we set the shape as a plat the normal is pointed to sun, the cross section area is the area of face 1 which considered as a rectangle the area is equal to 0.34 m², all that is in figure (69).

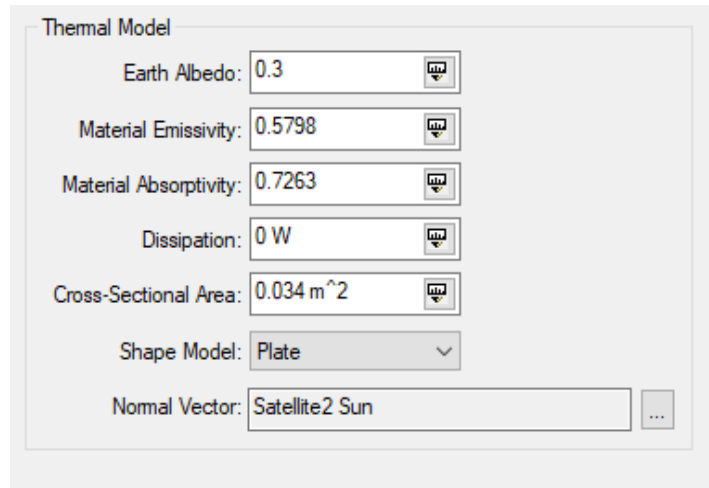


Figure 69 : thermal model of satellite2

Now we rename the satellite2, and we name it side1, and we chose the colors of eclipse phase, and sunlight phase as it is explained previously in the orbital analysis for one node.

So now we can study the temperature of all sides, the object browser is in (figure 70). (Table 14) show the direction of each side with their cross section area, working with same (figure 66). All sides are pointing to sun.

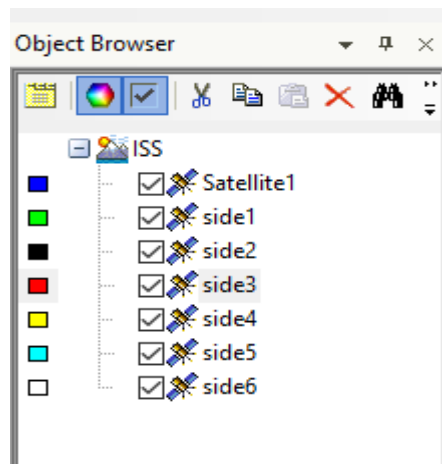


Figure 70 : object browser after creating all the sides

Table 14: the six sides with their cross section area

side	Normal directed to	Cross section area(m ²)
Side 1	sun	0.034
Side 2	North	0.01
Side 3	Zenith	0.034
Side 4	Nadir	0.034
Side 5	South	0.01
Side6	earth	0.034

- **Temperature of side1**

(Figure 71) represent the temperature changes of side 1 and the values of the maximum and minimum are in (figure 72). from these figure we can see that that the minimum temperature is in the eclipse-81°C/192°K and the maximum 145°C/418°K is when exposed to direct sunlight , this is a very high temperature !! , comparing this range of temperature and the operating range of components , we see that all components can't operate in this range (-81°C, 145°).

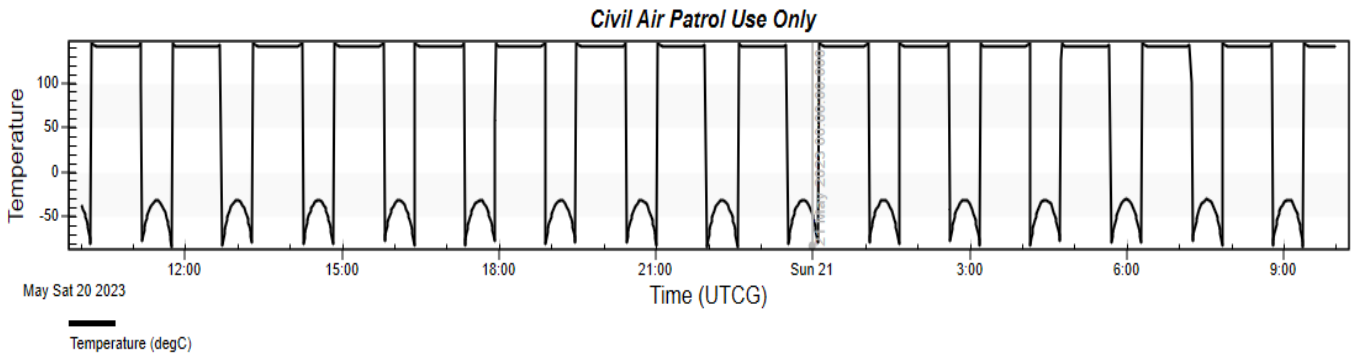


Figure 71 : temperature evolution along orbit of side1

Time (UTCG)	Temperature (degC)	Solar Flux (W/m ²)	Solar Intensity (%)
20 May 2023 10:00:00.000	-36.852	1333.310478	0.0000
20 May 2023 10:01:00.000	-38.780	1333.306252	0.0000
20 May 2023 10:02:00.000	-41.033	1333.302161	0.0000
20 May 2023 10:03:00.000	-43.630	1333.298223	0.0000
20 May 2023 10:04:00.000	-46.593	1333.294453	0.0000
20 May 2023 10:05:00.000	-49.949	1333.290868	0.0000
20 May 2023 10:06:00.000	-53.733	1333.287482	0.0000
20 May 2023 10:07:00.000	-57.990	1333.284309	0.0000
20 May 2023 10:08:00.000	-62.781	1333.281362	0.0000
20 May 2023 10:09:00.000	-68.184	1333.278653	0.0000
20 May 2023 10:10:00.000	-74.309	1333.276192	0.0000
20 May 2023 10:11:00.000	-81.312	1333.273990	0.0000
20 May 2023 10:12:00.000	145.073	1333.272055	100.0000
20 May 2023 10:08:00.000	-62.781	1333.281362	0.0000
20 May 2023 10:09:00.000	-68.184	1333.278653	0.0000
20 May 2023 10:10:00.000	-74.309	1333.276192	0.0000
20 May 2023 10:11:00.000	-81.312	1333.273990	0.0000
20 May 2023 10:12:00.000	145.073	1333.272055	100.0000
20 May 2023 10:13:00.000	144.318	1333.270393	100.0000
20 May 2023 10:14:00.000	143.545	1333.269011	100.0000
20 May 2023 10:15:00.000	142.756	1333.267914	100.0000
20 May 2023 10:16:00.000	141.955	1333.267104	100.0000
20 May 2023 10:17:00.000	141.146	1333.266584	100.0000
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Figure 72 : the maximum and minimum of temperature that side1 reach while orbiting

- **Temperature of side2**

Figure (73) represent the temperature changes of side 1, and the values of the maximum and minimum are in figure (74). from these figure we can see that that the minimum temperature is in the eclipse-

273°C/0°K and the maximum 63°C/336°K is when exposed to direct sunlight, in this side the min temperature is very low, comparing to the operating range of components , we see that all components can't operate in this range in this low temperature, except for infrared detectors.

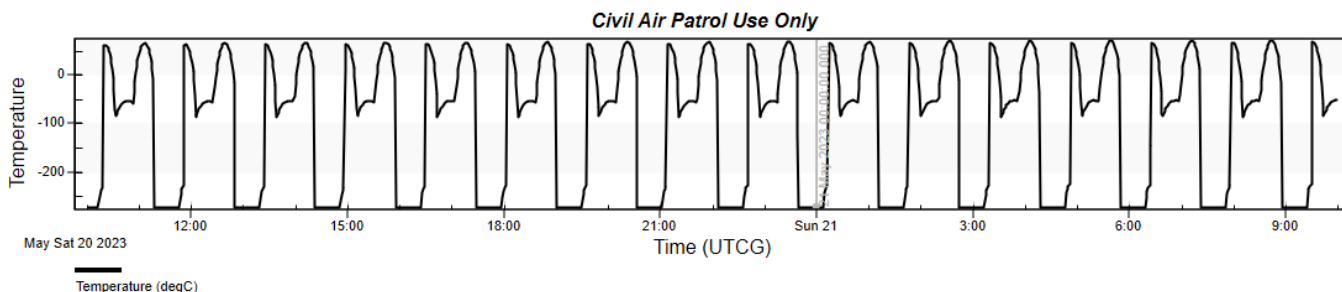


Figure 73 : temperature evolution of side2 while orbiting

20 May 2023 11:47:00.000	-273.150	1333.247904	0.0000
20 May 2023 11:48:00.000	-243.789	1333.244931	0.0000
20 May 2023 11:49:00.000	-235.685	1333.242195	0.0000
20 May 2023 11:50:00.000	-230.951	1333.239706	0.0000
20 May 2023 11:51:00.000	-227.532	1333.237476	0.0000
20 May 2023 11:52:00.000	62.527	1333.235512	100.0000
20 May 2023 11:53:00.000	63.408	1333.233821	100.0000
20 May 2023 11:54:00.000	63.154	1333.232410	100.0000
20 May 2023 11:55:00.000	61.714	1333.231284	100.0000
20 May 2023 11:56:00.000	59.016	1333.230445	100.0000
20 May 2023 11:57:00.000	55.023	1333.229897	100.0000
20 May 2023 11:58:00.000	49.851	1333.229639	100.0000

Figure 74 : the values of max of temperature and min temperature

- **Temperature of side3**

For side 3, figures (75) and (76), represent the limits of temperature for this side. These figures show that side 3 will get maximum heat in sunlight 133°C/406°K, and minimum -273°C/0°K. Also, comparing with the operating range of temperature of components, this is so dangerous for the cubesat on this side.

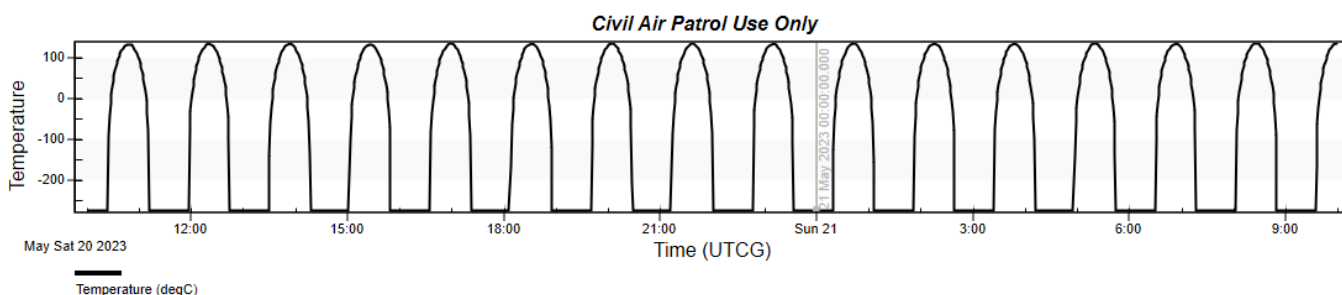


Figure 75 : temperature evolution of side 3 while orbiting

20 May 2023 10:03:00.000	-273.150	1333.327492	0.0000
20 May 2023 10:04:00.000	-273.150	1333.322890	0.0000
20 May 2023 10:05:00.000	-273.150	1333.318340	0.0000
20 May 2023 10:06:00.000	-273.150	1333.313863	0.0000
20 May 2023 10:07:00.000	-273.150	1333.309478	0.0000
20 May 2023 10:08:00.000	-273.150	1333.305203	0.0000
20 May 2023 10:09:00.000	-273.150	1333.301056	0.0000
20 May 2023 10:43:00.000	128.368	1333.300846	100.0000
20 May 2023 10:44:00.000	130.317	1333.304595	100.0000
20 May 2023 10:45:00.000	131.781	1333.308414	100.0000
20 May 2023 10:46:00.000	132.770	1333.312284	100.0000
20 May 2023 10:47:00.000	133.288	1333.316186	100.0000
20 May 2023 10:48:00.000	133.340	1333.320100	100.0000
20 May 2023 10:49:00.000	132.926	1333.324006	100.0000
20 May 2023 10:50:00.000	132.043	1333.327884	100.0000
20 May 2023 10:51:00.000	130.685	1333.331717	100.0000

Figure 76 : maximum and minimum temperature of side 3 , reported

- **Temperature of side4**

On this side, temperature ranges are from $-26^{\circ}\text{C}/247^{\circ}\text{K}$ at eclipse to $63^{\circ}\text{C}/336^{\circ}\text{K}$ when exposed to direct sunlight, and again comparing to the operating range of components, this range cause risks to the cubesat, and only solar panel can operate. Results are in figures (77 and 78).

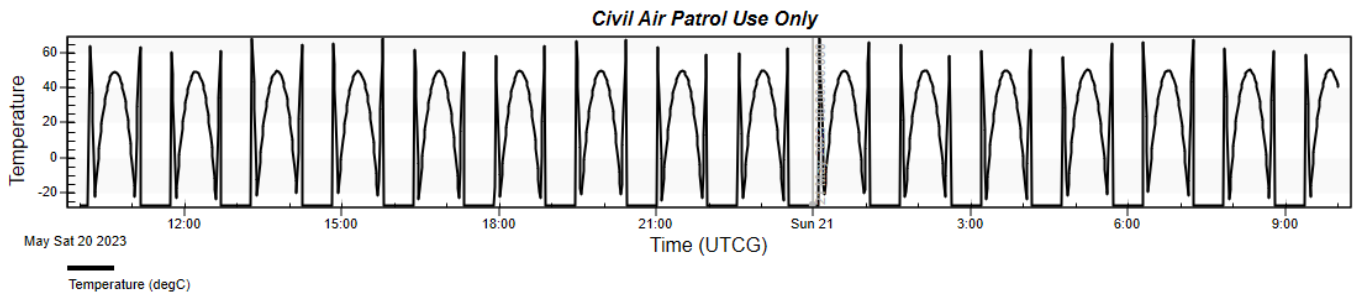


Figure 77 : temperature evolution of side 4 while orbiting

20 May 2023 10:02:00.000	-26.881	1333.302309	0.0000
20 May 2023 10:03:00.000	-26.882	1333.298366	0.0000
20 May 2023 10:04:00.000	-26.884	1333.294590	0.0000
20 May 2023 10:05:00.000	-26.885	1333.290998	0.0000
20 May 2023 10:06:00.000	-26.887	1333.287605	0.0000
20 May 2023 10:07:00.000	-26.889	1333.284424	0.0000
20 May 2023 10:08:00.000	-26.892	1333.281469	0.0000
20 May 2023 10:09:00.000	-26.895	1333.278751	0.0000
20 May 2023 10:10:00.000	-26.897	1333.276281	0.0000
20 May 2023 10:11:00.000	-26.901	1333.274069	0.0000
20 May 2023 10:12:00.000	63.652	1333.272124	100.0000
20 May 2023 10:13:00.000	51.476	1333.270452	100.0000
20 May 2023 10:14:00.000	37.481	1333.269059	100.0000
20 May 2023 10:15:00.000	21.000	1333.267854	100.0000

Figure 78 : maximum and minimum value of temperature for the side 4

- **Temperature of side5**

For the temperature limit we can see figures (79) and (80), these figures shows that the minimum temperature is about $-273^{\circ}\text{C}/0^{\circ}\text{K}$ in eclipse and maximum temperature when sunlight about $81^{\circ}\text{C}/354^{\circ}\text{K}$ and comparing with operating range of components, on this side temperature are restful.

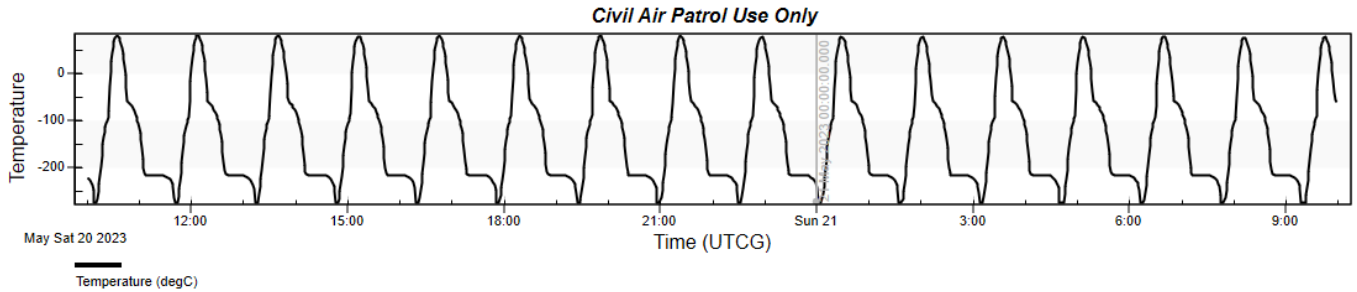


Figure 79 : temperature evolution of side 5 while orbiting

20 May 2023 10:01:00.000	-221.271	1333.306405	0.0000
20 May 2023 10:02:00.000	-223.109	1333.302309	0.0000
20 May 2023 10:03:00.000	-225.354	1333.298366	0.0000
20 May 2023 10:04:00.000	-228.152	1333.294590	0.0000
20 May 2023 10:05:00.000	-231.785	1333.290998	0.0000
20 May 2023 10:06:00.000	-236.949	1333.287605	0.0000
20 May 2023 10:07:00.000	-246.767	1333.284424	0.0000
20 May 2023 10:08:00.000	-273.150	1333.281469	0.0000
20 May 2023 10:09:00.000	-273.150	1333.278751	0.0000
20 May 2023 10:10:00.000	-273.150	1333.276281	0.0000
20 May 2023 10:11:00.000	-273.150	1333.274069	0.0000
20 May 2023 10:12:00.000	-263.756	1333.272124	100.0000
20 May 2023 10:13:00.000	-248.514	1333.270452	100.0000
-			
20 May 2023 10:28:00.000	27.100	1333.279134	100.0000
20 May 2023 10:29:00.000	44.505	1333.281762	100.0000
20 May 2023 10:30:00.000	57.891	1333.284590	100.0000
20 May 2023 10:31:00.000	67.939	1333.287603	100.0000
20 May 2023 10:32:00.000	75.060	1333.290786	100.0000
20 May 2023 10:33:00.000	79.551	1333.294123	100.0000
20 May 2023 10:34:00.000	81.656	1333.297596	100.0000
20 May 2023 10:35:00.000	81.590	1333.301188	100.0000
20 May 2023 10:36:00.000	79.538	1333.304881	100.0000
20 May 2023 10:37:00.000	75.652	1333.308656	100.0000
20 May 2023 10:38:00.000	70.038	1333.312494	100.0000

Figure 80 : the temperature limits of side 5

- **Temperature of side6**

Figures (81) and (82) are showing that temperature on side 6; change from a minimum of about $-26^{\circ}\text{C}/247^{\circ}\text{K}$ in eclipse to a maximum of about $64^{\circ}\text{C}/337^{\circ}\text{K}$ in sunlight. These values are also cause risks to the cubesat , comparing to the operating range of components.

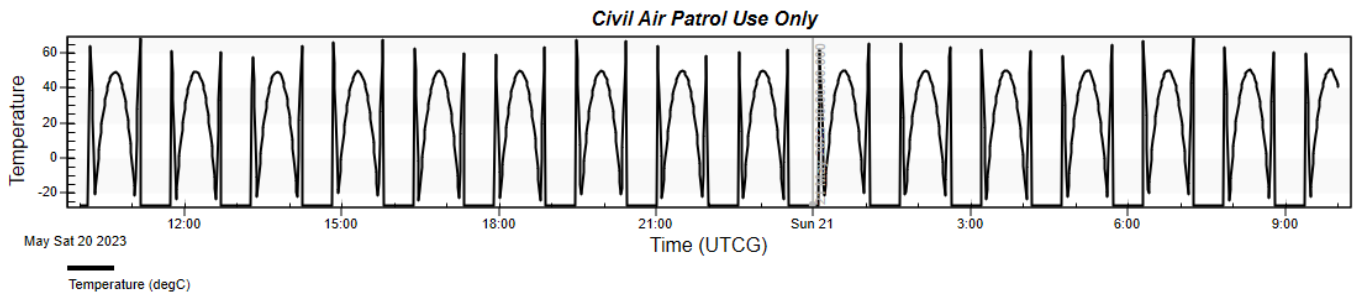


Figure 81 : temperature evolution of side 6 while orbiting

20 May 2023 10:10:00.000	-26.897	1333.276406	0.0000
20 May 2023 10:11:00.000	-26.901	1333.274178	0.0000
20 May 2023 10:12:00.000	64.372	1333.272215	100.0000
20 May 2023 10:13:00.000	52.297	1333.270526	100.0000
20 May 2023 10:14:00.000	38.432	1333.269116	100.0000
20 May 2023 10:15:00.000	22.221	1333.267989	100.0000
20 May 2023 10:16:00.000	2.822	1333.267150	100.0000

Figure 82 : the maximum and minimum temperature for side 6

4.4.7 Discussion of thermal analysis with STK

From the results obtained previously we can resume the temperature changes for all the sides in the table (15).

Table 15: summary of temperatures

Side 1	Max temperature	145°C/418°K
	Min temperature	-81°C/192°K
Side 2	Max temperature	63°C/336°K
	Min temperature	-273°C/0°K
Side 3	Max temperature	133°C/406°K
	Min temperature	-273°C/0°K
Side 4	Max temperature	63°C/336°K
	Min temperature	-26°C/247°K
Side 5	Max temperature	81°C/354°K
	Min temperature	-273°C/0°K
Side 6	Max temperature	64°C/337°K
	Min temperature	26°C/247°K

Comparing these results with the operating range of components we see that none of the components can operate in these results, and this confirms the need of using TCS to protect the cubesat from the heat in space while orbiting, also we see that each side has different temperature limits which mean that the components inside the satellite must be installed after doing this thermal analysis so that the component be in its operating range of temperature, also we see that so of the sides need to be heating up, and others need to be cooling down, and by using the appropriate thermal method we can achieve that easily.

4.4.8 thermal analysis with different coating

This table (table 16) show different types of coating with their absorptivity and emissivity , and from these results of the maximum temperature that the surface coatings will help control the temperature when exposed to direct sunlight. When compared theses results with the temperature limits of components , it can be seen that the components are more likely to survive the operational and survival limits than with no coating. However, the coatings decreased the temperatures even further than if the CubeSat had no coating, and every type of coating affect differently on the cubesat.

Table 16:maximum temperature when different coating are used

Teflon, silvered, 10 mm		Teflon, silvered, 2 mm		Teflon, Aluminized, 1 mm	
$\alpha = 0,1626$	$\varepsilon = 0,5038$	$\alpha = 0,157$	$\varepsilon = 0,3918$	$\alpha = 0,1906$	$\varepsilon = 0,347$
T max	290,98°K	T max	310,65°K	T max	325,71°K
	17,98°C		37,65°C		52,71°C

For the cold case the satellite is losing heat when it moves into eclipse and that a thermal control system must be designed to retain the heat that is accumulated when the CubeSat is in direct sunlight. This can be done with an MLI blanket. An MLI blanket will not only limit the amount of external thermal radiation from entering the CubeSat, but will also limit the amount of internal thermal radiation from leaving the CubeSat, which will potentially keep the internal components within their required temperature limits, alsop different divices can heat up the cubesat when needed.

4.4.9 Spacecraft thermal testing

The experimental test is the ultimate validation of real behavior of all of this analysis. But these tests are expensive, not only on the financial budget but on time demanded and other precious resources as qualified personnel.

Thermal tests can be performed at component, subsystem, or system level. Although tests should be carried out with maximum fidelity to expected operational procedures and realistic space environment (notably vacuum), sometimes, [48] preliminary tests are performed at room pressure, like temperature cycling and operational sequences (e.g. turn-on and turn-off active components), but thermal vacuum tests are needed at the end. Main parameters of thermal cycling are temperature extremes, number of cycles, and rate of temperature change.

Qualification tests are performed to validate all the steps in the design and manufacturing process, from requirements to testing techniques (tooling, handling procedures, personnel expertise...). Hardware for thermal qualification is subjected to more severe tests than flight hardware (i.e. to cold and hot thermal environments beyond those expected in flight), to demonstrate that a safety margin exists), although both are to be manufactured to the same standards. Final testing of a large spacecraft for acceptance as a delivered product is at the present limit of technology, since very large vacuum chambers, with a powerful collimated solar-like beam, and walls kept at cryogenic temperatures, must be provided (and the spacecraft able to be deployed, and rotated in all directions, while measuring). Typical temperature discrepancy between the most advanced numerical simulation and the most expensive experimental tests may be some 2 K for most delicate components in integrated spacecraft (much lower when components are separately tested).

CONCLUSION

From the work of this document, and the results obtained in the thermal analysis of the 3U cubesat launched in the ISS orbit, it is confirmed that the cubesats must be maintained within their acceptable temperature limits in order to remain operational, to do that TCS is required.

The thermal analysis showed that cubesat while orbiting has two cases, the first one is when the cubesat is exposed to direct sunlight and it is the case where the cubesat absorb the maximum of heat, it reach the highest temperature around $120^{\circ}\text{C}/393^{\circ}\text{K}$ in this analysis and it considered as a hot case, and the second one is when the cubesat passes by the eclipse (total shadow) which the only heat absorbed is the infrared emitted from earth, where the satellite reach the lowest temperature about $-40^{\circ}\text{C}/233^{\circ}\text{K}$ in this analysis and it is the cold case.

Appropriate thermal analysis should be done to ensure the thermal control before the mission start. Satellite thermal analysis usually involves the use of software-assisted analytical processes to determine node temperatures in the model using numerical method approximation, such as Euler or Runge-Kutta method. Thermal analysis outcome is dependent to solar factor, albedo factor, eclipse times, orbital position, material selections and material properties etc.

About softwares used in this study are : for the orbital and thermal analysis: System Tool Kit software (STK.11), and MATLAB R2015a, other needed calculations with Excel.

This thesis showed the importance of using TCS, and that there are two types of TCS ,passive thermal control methods like :MLI and thermal straps , and active thermal control methods such as :heaters and louvers. This method work together to maintain the cubesat in its allowable range of temperature. Ideally, the thermal system would be as passive as possible to minimize mass and power requirements.

In order to get more and more details about the thermal control and to create a more complex models, additional model revisions can be made. Structures, altitudes, surface properties, structural properties etc. can be changed to make different models. It is possible to model electrical and mechanical components more detailed and improve nodal resolution. Also it can be author simulation when different coating is used. In this study a lot of values of some parameters and properties, and material properties, are estimated, and this estimation may lead to model errors. Also angles can affect the analysis, however additional complexity can also introduce additional errors to the model.

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[45]”system tool kit” , page last update 2023 from Wikipedia on the following link:
https://en.m.wikipedia.org/wiki/system_tool_kit

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[47] “ what’s the typical temperature of a satellite orbiting the earth” , 2015, from the following link: <https://www.space.stackexchange.com/questions/7827/whats-the-typical-temperature--of-a-satellite-orbiting-the-earth>

[48] article pdf “ spacecraft thermal modeling and testing” published from the university polytechnique of Madrid , downloaded 20/march/2023 , it is available online on the following link:
<http://imartinez.etsiae.upm.es/isidoro/tc3/Spacecraft%20thermal%20modelling%20and%20testing.pdf>

Annex

1. Start STK from MATLAB (MATLAB code):

```
app = actxserver('STK11.application');
root = app.Personality2;
scenario = root.Children.New('eScenario','leosat');
scenario.SetTimePeriod('Today','+24hr')
root.ExecuteCommand('Animate * Reset')
satellite = scenario.Children.New('eSatellite','LeoSat');
root.ExecuteCommand(['SetState */Satellite/LeoSat Classical TwoBody "',scenario.StartTime,'"
"',scenario.StopTime,'" 60 ICRF "',scenario.StartTime,'"6784500.0 0.0005 51.6 331.3 290.4 110.85
']);
% IAgSatellite satellite: Satellite object
lighting = satellite.Graphics.Lighting;
%Settings for vehicle in sunlight
sunlight = lighting.Sunlight;
sunlight.Visible = true;
sunlight.Color = 65535; %yellow
sunlight.LineWidth = 3;
%Settings for vehicle in penumbra
penumbra = lighting.Penumbra;
penumbra.Visible = true;
penumbra.Color = 42495; %orange
penumbra.LineWidth = 2;
%Settings for vehicle in umbra
umbra = lighting.Umbra;
umbra.Visible = true;
umbra.Color = 255; %red
umbra.LineWidth = 1;
```

root.CloseScenario.

2. Matlab code for the temperature calculation :

```
clear all
```

```
close all
```

```
clc
```

```
% Define known parameters
```

```
H = 400; % altitude [km]
```

```
R = 6370; % Earth radius [km]
```

```
G=6.67*10(-11) % Earth's gravitational constant [N*m2/kg2]
```

```
M=5.97*10(24) % Earth's mass [kg]
```

```
T0=5561; % period of orbit [s]
```

```
alpha = 0.672; % absorptivity of cubesat
```

```
Across = 0.03; % cross section area [m2]
```

```
S = 1414 ; % solar irradiance for hot case [W]
```

```
A = 0.14; % total area of spacecraft [m2]
```

```
Fbp = 0.8853; % view factor
```

```
alb = 0.35; % albedo
```

```
epsilon = 0.459; % emissivity of cubesat
```

```
ep = 0.6; % emissivity of planet
```

```
sig = 5.67e-8; % Stefan-Boltzmann constant
```

```
Tearth = 288; % temperature of earth [K]
```

```
m = 3; % mass of spacecraft[kg]
```

```
cp = 879; % heat capacity of Al 6061-t6 [J/kg.K]
```

```
Fb0 = 1; % view factor of unity
```

```
earthem=261 % IR constant
```

```
i = 51.6; % inclination [deg]
```

```
h = (H+R)/R; % function of relative orbit radius
```

```

beta = i*pi/180; % beta angle [rad]
beta_max = pi/2 - acos(1/h);
% Define eclipse duration
Te=T0*((1/pi)* acos(sqrt(h^2 - 1)/(h*cos(beta)))); % [s]
% Define orbital angle range and intervals
phi = 0:pi/48:2*pi;
% angular position eclipse starts
phi_s = pi - acos(sqrt(h^2 - 1)/(h*cos(beta))); % [rad]
% angular position eclipse ends
phi_e = pi + acos(sqrt(h^2 - 1)/(h*cos(beta))); % [rad]
% Define heat fluxes
Qs0 = alpha*Across*S; % solar input
Qa0 = alpha*A*Fbp*alb*S; % abledo input
QiR = epsilon*A*Fbp*earthem; % infrared input
% step function for solar input
for i = 1:97
    if phi_s < phi(i) && phi_e > phi(i)
        Fs(i) = 0;
    else
        Fs(i) = 1;
    end
end
% albedo view factor from 0 to 180 deg
for i = 1:49
    if phi(i) > phi_s && phi(i) < pi
        Fe(i) = 0;
    else

```



```

Fe(i) = 1;
end
end
%albedo view factor from 180 to 360 deg
for i = 50:97
if phi(i) < phi_e && phi(i) > pi
Fe(i) = 0;
else
Fe(i) = 1;
end
Fa=-0.5+0.5*cos(phi(i))^2*(1-(phi(i)/phi_e)^2)*Fe(i);
end
Qs = Qs0*Fs;
Qa = Qa0*Fa;
QiR = 0*phi + QiR;
% initial temperature. Taken from hot case calculations
T = 342;
% Determine change in temperature for each change in angle. Add that
% change to previous temperature to obtain new temperature. Repeat
% process.
for i = 1:96
dT_dphi(i) = (T0/(m*cp*2*pi))*(Qs0*Fs(i) + Qa0*Fa(i) + QiR(i) ...
- epsilon*A*Fb0*sig*T(i)^4);
T(i+1) = T(i) + dT_dphi(i);
end
dT_dphi(97) = (T0/(m*cp*2*pi))*(Qs0*Fs(97) + Qa0*Fa(97) + QiR(97) ...
- epsilon*A*Fb0*sig*T(97)^4);

```

```

% convert from [rad] to [deg]
phi_s = phi_s*180/pi; % [deg]
phi_e = phi_e*180/pi; % [deg]
phi = phi*180/pi;
beta=beta*180/pi; % [deg]
beta_max =beta_max *180/pi; % [deg]
% Convert angle to time
t = phi*T0/(360*60);
% Convert time from [s] to [min]
Te=Te/60 ; % [min]
% Plot solar input
figure(1),
plot(phi,Qs,'linewidth',2)
title('Solar Heat Flux Along Orbit'),
xlabel('Orbital Angle, \phi [deg]'), ylabel('Solar Heat Flux [W]')
% Plot albedo input
figure(2),
plot(phi,Qa,'linewidth',2)
title('Albedo Heat Flux Along Orbit'),
xlabel('Orbital Angle, \phi [deg]'), ylabel('Albedo Heat Flux [W]')
% Plot infrared input
figure(3),
plot(phi,QiR,'linewidth',2)
title('infrared Heat Flux Along Orbit'),
xlabel('orbital Angle, \phi [deg]'), ylabel('infrared [W]')
% Convert angle to time
t = phi*T0/(360*60);

```

% Convert angle to time

t = phi*T0/(360*60);

T = T - 273; % convert from Kelvin to Celsius

% Plot time evolution of temperature

figure(4),

plot(t,T,'linewidth',2), grid on

xlabel('time [min]'), ylabel('temperature [C]')

title('Temperature Evolution Along Orbit the hot case')

3. Optical Properties of Common Surface Coatings and Finishes [31]

Material	Absorptivity	Emissivity
<u>Optical Solar Reflectors</u>		
Teflon, Aluminized, 0.5 mm	0.14	0.4
Teflon, Aluminized, 1 mm	0.14	0.6
Teflon, silvered, 2 mm	0.08	0.68
Teflon, silvered, 10 mm	0.09	0.88
<u>Black Coating</u>		
Chemglaze Z306 Black Paint	0.96	0.91
Black Z306 polyurethane paint, 3 mm	0.95	0.87
Ebanol C Black	0.97	0.73
Rough black matte, black paint	0.9	0.9
<u>Films and Tapes</u>		
Kapton, aluminized, 0.25 mm	0.31	0.45
Kapton, black (carbon loaded), 1 mm	0.92	0.88
Tape, 235-3M, black	0.95	0.9
Tape, aluminum	0.1	0.04
<u>White Coatings</u>		
Chemglaze A276 white paint	0.24	0.9
Hughson A-276 white paint	0.26	0.88
Magnesium oxide white paint	0.09	0.9
Polyurethane white paint	0.27	0.84
<u>Other Paints</u>		
Aluminum Paint	0.3	0.31
Chromacoat aluminum paint	0.28	0.05
Silicone aluminum paint	0.29	0.3
<u>Metals</u>		
Aluminum, buffed	0.16	0.03
Aluminum, polished	0.15	0.05

4. Start up



**MINISTRE DE L'ENSEIGNEMENT SUPERIEUR ET DE
LA RECHERCHE SCIENTIFIQUE**

Université Blida 1

Institut Aéronautique et des Etudes Spatiales IAES



Mémoire de fin d'études

En vue de l'obtention du diplôme « une startup » / un diplôme «
un brevet » dans le cadre de l'arrête ministériel N° 1275

Domaine : Sciences et Technologies

Département : les études spatiales

Spécialité : propulsion spatiale

Intitulé du Projet :

« thermal control of a low earth orbit satellite »

Projet **Startup** présenté dans le cadre de l'arrête ministériel "**1275**",
assurée par **Institut d'Aéronautique et des Etudes Spatiales IAES**

Présenté par :
1BELGADER SOFIA

Encadré par :
Encadreur : Dr.KHALI ELHADI

Formateurs :
1- DR . KRIM M
2- DR. LEBSIR A.

Année académique : 2022/2023



MINISTÈRE DE L'ENSEIGNEMENT SUPÉRIEUR
ET DE LA RECHERCHE SCIENTIFIQUE
Université Blida 1



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ministériel N° 1275

Domaine : Sciences et Technologies

Département : les études spatiales

Spécialité : propulsion spatiale

Intitulé du Projet :

Structuration et planification du projet de :

« thermal control of a low earth orbit satellite »

Projet de **Startup** présenté dans le cadre de l'arrête ministériel "1275", assurée par
Institut d'Aéronautique et des Etudes Spatiales. IAES

Devant le jury composé de :

Membres des jurys	Nom et prénom	Grade	Etablissement	Signature
Président		Professeur		
Encadreur		MCB		
Co-encadreur		MCB		
Examineur		MCB		
Représentant incubateur/ Maison de l'entrepreneuriat		Directeur		
Représentant du partenaire économique		Directeur		
Responsable du Centre d'Appui à la Technologie et à l'Innovation		Directeur		

Année académique : 2022/2023

Carte d'information



A propose de l'équipe d'encadrement
du groupe de travail





Carte d'information

A propose de l'équipe d'encadrement du groupe de travail

Equipe de l'Encadrement du Groupe principale (a titre indicatif)	SPECIALITE	FACULTE	ETABLISMENT
Encadrant 01	DR KHALI ELHADI	Institut d'aéronautique et les études spatiale	Saad dahleb blida 1

A propose de l'équipe du projet du groupe de travail

Equipe du projet du Groupe principale (a titre indicatif)	SPECIALITE	FACULTE	ETABLISMENT
BELGADER SOFIA	Propulsion spatiale	Institut d'aéronautique et les études spatiale	Université de SAAd DAHLEB blida 1

Business Model Canvas

Porteurs de projet :

Promoteurs :

Code de projet :

Business Model Canvas - BMC

1- BELGADER SOFIA

P-DR.KHALI ELHADI

05-17- 3232

CO-P-

Projet Startup :

Key partners :	Key activités :	Value proposition:	Customer relation-ship :	Customers :
<ul style="list-style-type: none"> - Space agencies - Hardware companies - Airline companies - Algerian air-force. 	<ul style="list-style-type: none"> - Study - Design - Produce - Distrubution 	<ul style="list-style-type: none"> - Minimise absorbed solar energy + IR emission - Solve problem of in-space stability - Solve problem of outgassing - Avoid mechanical adhesion - Thermal controlling for the spacecraft - Globaly recognized for the International standers products 	<ul style="list-style-type: none"> - Cost-saving - Long term contrat - Economic support - Ensure succesful missions. 	<ul style="list-style-type: none"> - Space agencies - Airlines companies - Military market - Universities (university scientific clubs, studies projects) - Satellite manifacters - Private companies interested in space exploration and development.
	<p>Key ressources :</p>		<p>Chanel :</p>	
	<ul style="list-style-type: none"> - Engineers - Physiciens - Chimistes - Employees - E-commerse devlopers - Test chambers - Large space 		<ul style="list-style-type: none"> - Website - App store - Official social media accounts. <p>Office +Direct store</p>	
<p>Costs :</p>		<p>Revenue :</p>		
<ul style="list-style-type: none"> - Salery of workers - Cost of primery materiales - Hardware and software - Transportation - Test chambers 		<p>Sales of local products (thermal surface finishes , solar reflector, MLI)</p>		

Annexe



La fiche technique de projet startup 1275



La fiche technique de projet

Carte d'information	
بلقادر صوفية	الاسم و اللقب
BELGADER SOFIA	Your first and last Name
BELGADER SOFIA	Votre prénom et nom
التحكم الحراري في الأقمار الصناعية التي تدور في مدار منخفض	الاسم التجاري للمشروع
Le contrôle thermique d'un satellite dans les orbites basse.	Intitulé de votre projet
Thermal control of low earth orbit satellite	Title of your Project
ش ذ م م	الصفة القانونية للمشروع
SARL	Votre statut juridique
SARL	Your legal status
06.57.54.33.42	رقم الهاتف
	Votre numéro de téléphone
	Your phone number
sofiabelgader@gmail.com	البريد الالكتروني
	Votre adresse e-mail
	Your email address
جامعة سعد دحلب البلدية 1	مقر مزاولة النشاط (الولاية- البلدية)
Universite de Saad Dahleb blida 1	Votre ville ou commune d'activité
University of saad dahleb blida1	Your city or municipality of activity

طبيعة المشروع (طبيعة الابتكار)

La nature de projet

<p>النتوج ذو طابع انتاجي و خدماتي معا Vente de marchandise et de service aussi Sales of goods and seviles together</p>	<p>المنتوج ذو طابع إنتاجي أو خدماتي Vente de marchandises ou de services Sale of goods or services</p>
---	---

La proposition de valeur ou l'offre faite

<p>تحديد المشكل الذي يواجهه الزبون</p> <p>Déterminer le problème rencontré par le client</p>	
<p>The problem of radiation and absorbed heat and infrared emitted from the earth along low earth orbit.</p>	<p>ما هي المشكلة التي تريد حلها؟</p>
<p>The thermal analysis of 3U cubesat in low earth orbit without thermal control system , showed that the cubesat will experience very dangerous changes in the temperature while exposing to sunlight or in the eclipse .</p>	<p>ما هي البيانات المتوفرة لديك التي تدل على وجود المشكلة المحددة؟</p>
<p>In the space agencies have done proven thermal analysis and applied thermal control system to their cubesats and satellites in general, and there different methods to do , coating and thermal staps are solutions for this problem.</p>	<p>ما هي المشاريع الأخرى التي استهدفت نفس المشكلة والتي جرى تنفيذها؟</p>
<ul style="list-style-type: none"> - The purpose is to give solution of the thermal problem. - The solution is to use a thermal control system . - Thermal control system is methods for maintaining the satellite in an operational range of temperature, so that the satellite can cool up in the sunlight and heat up in the shadow. - The are two types of thes methods : active and passive methods. Both are for the thermal control of the satellite from the inside and from the outside too. - From the out side we use insulation like MLI, solar reflector and thermal finishes . 	<p>ماهي أهداف مشروعك و/أو نتائجه المتوقعة؟</p>

We can use thermal finishes in the inside too, to control heat inside the satellite and between components.	
---	--

Customer Relationship مع العملاء	
Contrat annuelle – des promotions pour les grandes quantites	كيف تدير علاقاتك مع العملاء؟
MICROSOFT DYNAMICS MONDAY CRM ZOHO CRM	ماهية أهم البرامج التي ستعتمد عليها في ادارة العلاقة مع الزبون Microsoft Dynamics Monday CRM Zoho CRMالخ

structure Costs هيكل التكاليف	
En milliers DZD	structure Costs هيكل التكاليف

	Frais d'établissement تكاليف التعريف بالمنتج أو المؤسسة
500000 da	Frais d'ouverture de compteurs (eaux-gaz-....) تكاليف الحصول على العدادات (الماء- الكهرباء)
10000000 da	Logiciels, formations تكاليف (التكوين- برامج الاعلام الالي المختصة)
	Dépôt marque, brevet, modèle تكاليف براءة الاختراع و الحماية الصناعية و التجارية
20000000 da	Droits d'entrée تكاليف الحصول على تكنولوجيا او ترخيص استعمالها
	Achat fonds de commerce ou parts شراء الأصول التجارية أو الأسهم
/6000000 da par an) engare minimum 300m ² (500000 da par moi	Droit au bail الحق في الإيجار
20000 da	Caution ou dépôt de garantie وديعة أو وديعة تأمين
40000 da	Frais de dossier رسوم إيداع الملفات
300000 da pour noteur / 300000 da avocat	Frais de notaire ou d'avocat تكاليف الموثق-المحامي-
60000 da	Enseigne et éléments de communication تكاليف التعريف بالعلامة و تكاليف قنوات الاتصال
50000000 da	Achat immobilier شراء العقارات
10000000 da	Travaux et aménagements الأعمال والتحسينات الاماكن
20000000 da	Matériel الآلات- المركبات- الاجهزة
2000000 da	Matériel de bureau تجهيزات المكتب
30 00000 da	Stock de matières et produits تكاليف التخزين
10 0000000 da	Trésorerie de départ التدفق النقدي (الصندوق) الذي تحتاجه في بداية المشروع.
200000000 da	GLOBAL = المجموع

2. نفقاتك أو التكاليف الثابتة الخاصة بمشروعك

En milliers DZD						نفقاتك أو التكاليف الثابتة الخاصة بمشروعك
PREVISION			REALISATION			
N	N +1	N+2	N	N -1	N -2	

						Assurances التأمينات
10000 da par moi						Téléphone, internet الهاتف و الانترنت
						Autres abonnements اشتراكات أخرى
250000 da par moi						Carburant, transports الوقود و تكاليف النقل
300000 da par moi						Frais de déplacement et hébergement تكاليف التنقل و المبيت
1600000 da chaque trimestre						Eau, électricité, gaz فواتير الماء - الكهرباء- الغاز
						Mutuelle التعاضدية الاجتماعية
500000 da						Fournitures diverses لوازم متنوعة
500000 da chaue 6 mois						Entretien matériel et vêtements صيانة المعدات والملابس
200000 par moi						Nettoyage des locaux تنظيف المباني
100000 da chaue moi						Budget publicité et communication ميزانية الإعلان والاتصالات
30000000 da						= المجموع GLOBAL

3. مصادر الإيرادات Revenue Stream

مصادر الإيرادات Revenue Stream	
No	المساهمة الشخصية أو العائلية Apport personnel ou familial
No	التبرعات العينية Apports en nature (en valeur)
No	قرض رقم 1 اسم البنك Prêt n°1 (nom de la banque)
No	قرض رقم 2 اسم البنك Prêt n°2 (nom de la banque)
No	قرض رقم 3 اسم البنك Prêt n°3 (nom de la banque)
No	منحة 1 Subvention n°1 (libellé)
No	منحة 2 Subvention n°2 (libellé)
No	تمويل آخر Autre financement (libellé)
0 da	GLOBAL المجموع =

بيع المنتج في السنة الأولى

Votre Chiffre d'affaires de la Première Année

متوسط أيام العمل في الشهر	بيع المنتج في السنة الأولى
20	الشهر 1Mois (N)
20	الشهر 2Mois (N+1)
20	الشهر 3Mois (N+2)
20	الشهر 4Mois (N+3)
20	الشهر 5Mois (N+4)
20	الشهر 6Mois (N+5)
20	الشهر 7Mois (N+6)
20	الشهر 8Mois (N+7)
20	الشهر 9Mois (N+8)
20	الشهر 10Mois (N+9)
20	الشهر 11Mois (N+10)
20	الشهر 12Mois (N+11)
240	GLOBAL المجموع =
365	النسبة المئوية للزيادة في حجم الأعمال بين كل شهر لسنة الأولى؟

بيع المنتج في السنة الثانية

Votre Chiffre d'affaires de la Deuxième Année

متوسط أيام العمل في الشهر	بيع المنتج في السنة الثانية
20	الشهر (N) 1Mois
20	الشهر (N+1) 2Mois
20	الشهر (N+2) 3Mois
20	الشهر (N+3) 4Mois
20	الشهر (N+4) 5Mois
20	الشهر (N+5) 6Mois
20	الشهر (N+6) 7Mois
20	الشهر (N+7) 8Mois
20	الشهر (N+8) 9Mois
20	الشهر (N+9) 10Mois
20	الشهر (N+10) 11Mois
20	الشهر (N+11) 12Mois
240 jour	= GLOBAL المجموع
365 jour	النسبة السنوية للزيادة في حجم الأعمال بين كل شهر لسنة الثانية؟

بيع المنتج في السنة الثالثة

Votre Chiffre d'affaires de la Troisième Année

متوسط أيام العمل في الشهر	بيع المنتج في السنة الثانية
20	الشهر 1Mois (N)
20	الشهر 2Mois (N+1)
20	الشهر 3Mois (N+2)
20	الشهر 4Mois (N+3)
20	الشهر 5Mois (N+4)
20	الشهر 6Mois (N+5)
20	الشهر 7Mois (N+6)
20	الشهر 8Mois (N+7)
20	الشهر 9Mois (N+8)
20	الشهر 10Mois (N+9)
20	الشهر 11Mois (N+10)
20	الشهر 12Mois (N+11)
240 jour	المجموع GLOBAL =
365 jour	النسبة المئوية للزيادة في حجم الأعمال بين كل شهر لسنة الثالثة ؟

5. تطور حجم رقم الأعمال في السنة

النسبة المئوية %	الزيادة في حجم الأعمال
no	بين السنة 1 والسنة 2
no	بين السنة 2 والسنة 3
No	GLOBAL المجموع =

6. رواتب الموظفين و مسؤولين الشركة

صافي أجور المسؤولين Rémunération nette dirigeant	رواتب الموظفين Salaires employés	رواتب الموظفين و مسؤولين الشركة
1 800 000 da	100000 da par moi / 12 000 000 da	CHIFFRE D'AFFAIRES
	14 000000 da	GLOBAL المجموع =

Title :

Thermal control of low earth orbit satellite

(thermal surface finishes and insulation)

Domaine :

This project is about the thermal control of cubesats in low earth orbit, and it concerne the surface finishes and insulation that maintain the cubesat in an allowable temperature.

introduction :

once the satellite is launched to the orbit , it is exposed to diffrent heating fluxes, that are different from the ground environment , it is the space environment !! , direct solar heat flux, the heat reflected from the earth (albedo), heat emitted from earth (infrared).

And the satellite will experience two dangerous states; exposing to direct sunlight wich can lead to over heating or it will be in the shadow of the earth and that lead to under cooling, this two phenomens can cause failure of the desired mission , or even explosion of the satellite.

Purpose of the project :

- The purpose is to give solution of the thermal problem.
- The solution is to use a thermal control system : thermal surface finishes and insulation .
- Thermal control system is a method for maintaining the satellite in an operational range of temperature, so that the satellite can cool up in the sunlight and heat up in the shadow.
- The surface finishes and the insulation is for the thermal control of the cubesat from the outside and inside too.

Avantages of the project :

- Minimise absorbed solar energy + IR emission
- Solve problem of outgassing
- Avoid mechanical adhesion
- Thermal controlling for the spacecraft
- Globaly recognized for the International standers products.
- Algerian space products.
- Do the thermal analysis for satellites with advanced softwares.
- Help space agencies for the success of the mission by avoiding radiation in space.
- Help university clubs and student to realise realastic satellites.
- Products appliable in aeronautic sector too.

About the project :

- Thermal insulation :MLI , solar reflectors.
- Thermal coating and paints.
- Thermal and orbital analysis.

Clients :

- Space agencies .
- Airline companies.
- Universities.

- Military.

Cost :

- Salery of workers
- Cost of primery materiales
- Hardware.
- Software .
- Large space

Conclusion :

The table below (table 4.7) is taken from the thermal analysis that we did in the study for 3U cubesat with no thermal control system , it shows the minimum and maximum temmperture in the hot case and the cold case , comparing these temperatures with the operationg ranges of temperatures of the components of the satellite (table 4.5) , we see the importance of using the thermal control system so the satellite will be working in the limit of temperature during exposing to direct sunlight or even the eclipse phase.

Transient state : one node model results			
Hot case		Cold case	
highest temperature	lowest temperature	highest temperature	lowest temperature
120°C/393°K	-22°C/251°K	85°C/358°K	-40°C/233°K

Table 4.7 : transient state summary

Component or Subsystem	Operating Temperature (°C)	Survival Temperature (°C)
General electronics	-10 to 45	-30 to 60
Batteries	0 to 10	-5 to 20
Infrared detectors	-269 to -173	-269 to 35
Solid-state particle detectors	-35 to 0	-35 to 35
Motors	0 to 50	-20 to 70
Solar panels	-100 to 125	-100 to 125

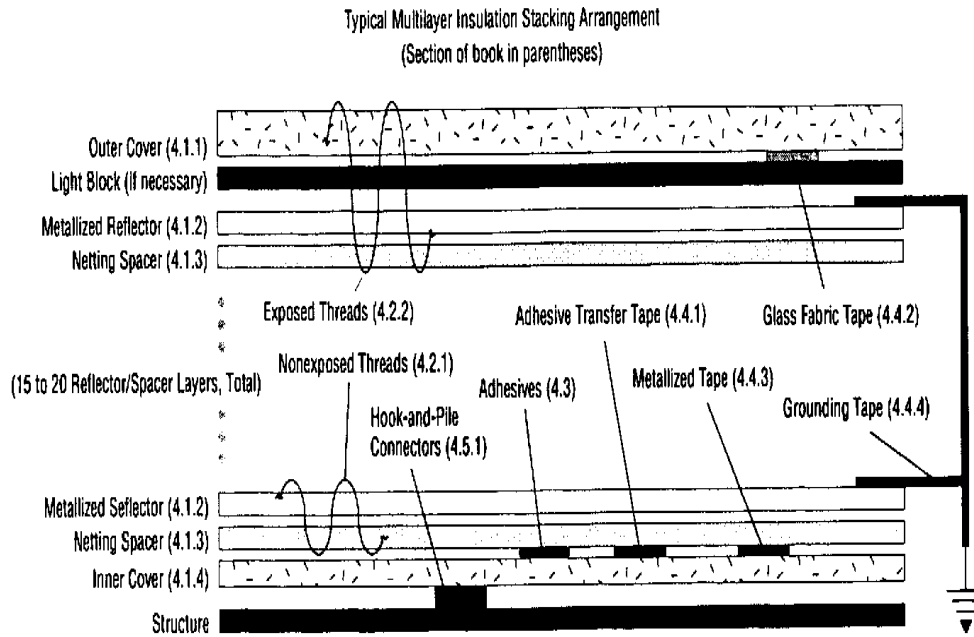
Table 4.5 : temerature limits for some spacecraft components[47]

Our project is to give this service of controlling the temperature of the satellite by protecting the external surfaces of the satellite, also from the inside too.

The surface properties of a spacecraft can be modified by adding specialized paints, coatings, surface finishes, or adhesive tapes, depending on the needs of the spacecraft

Figures:

MLI is composed of multiple layers of low-emittance films, as shown in Fig.



Note: Details and features are shown for illustration and will vary with actual design and installation.

Figure 83 : Typical MLI blanket structure [41]

The simplest MLI construction is a layered blanket assembled from thin (1/4- mil thick) embossed Mylar sheets, each with a vacuum-deposited aluminum finish on one side.

As a result of the embossing, the sheets touch at only a few points, and conductive heat paths between layers are thus minimized.

The layers are aluminized on one side only so that the Mylar can act somewhat as a low-conductivity spacer.

Higher-performance construction is composed of Mylar film metalized (with aluminum or gold) on both surfaces with silk or Dacron net as the low-conductance spacers.

replacing aluminized Mylar/Dacron spacers with embossed, aluminized Kapton and no spacers increased the effective emittance of blankets by 19%.

Blankets were originally used only for limiting the heat flow to and from a spacecraft. Today they may also protect against micrometeoroids, atomic oxygen (AO), electron charge accumulation, and rocket-engine plume impingement.

In addition, blanket design must accommodate requirements for durability, flammability, contamination control, launch loads, pressure decay, spacecraft venting, glint minimization, and restrictions on magnetic materials. Because most launch sites are near beaches (or even in the middle of the ocean, as in the case of SeaLaunch), exposure to salt spray and other corrosive agents is possible, so blanket design must take that exposure into account. All of these functions and design requirements must be addressed by blanket developers, who are also striving to minimize mass, cost, risk, and development time.

Thermal surface finishes :

Material	Absorptivity	Emissivity
<u>Optical Solar Reflectors</u>		
Teflon, Aluminized, 0.5 mm	0.14	0.4
Teflon, Aluminized, 1 mm	0.14	0.6
Teflon, silvered, 2 mm	0.08	0.68
Teflon, silvered, 10 mm	0.09	0.88
<u>Black Coating</u>		
Chemglaze Z306 Black Paint	0.96	0.91
Black Z306 polyurethane paint, 3 mm	0.95	0.87
Ebanol C Black	0.97	0.73
Rough black matte, black paint	0.9	0.9
<u>Films and Tapes</u>		
Kapton, aluminized, 0.25 mm	0.31	0.45
Kapton, black (carbon loaded), 1 mm	0.92	0.88
Tape, 235-3M, black	0.95	0.9
Tape, aluminum	0.1	0.04
<u>White Coatings</u>		
Chemglaze A276 white paint	0.24	0.9
Hughson A-276 white paint	0.26	0.88
Magnesium oxide white paint	0.09	0.9
Polyurethane white paint	0.27	0.84
<u>Other Paints</u>		
Aluminum Paint	0.3	0.31
Chromacoat aluminum paint	0.28	0.05
Silicone aluminum paint	0.29	0.3
<u>Metals</u>		
Aluminum, buffed	0.16	0.03
Aluminum, polished	0.15	0.05

Tests :

are high cost , in some specific conditions , vaccum chamber and others .