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Orbital Study of MSU CubeSath

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By

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The project is an orbital design study of a proposed CubeSat at Mississippi State University. The launch date is not specified. As for the mission, it is defined as forest fire detection. CubeSats are small satellites that are 10 x 10 x 10 cm in dimension and has a mass no more than 1 kg. They are currently used in different applications in many countries as an easy access to space. The analyses of this project have been carried out using a commercial software package, System Tool Kit (STK), developed by Analytical Graphics, Incorporated. This software provides a tool for performing simulations required for determining the orbit or the trajectory for satellites. In addition, a perturbation orbital study has been conducted and different propagators have been tested.

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CHAPTER I

INTRODUCTION AND MOTIVATION

1.1 Introduction

Space has become a necessary place to discover and explore, giving researchers and scientists a wide research area with new goals to achieve, from ancient times, where men have tried to discover our seas and planets, and kept looking at the original shape of the earth, to reaching the air by starting to build flying machines, and later to space, by building spacecraft and expanding the research area of science to space. What at the beginning took days or months to arrive at, now takes only few hours.

Despite of the great revolution of science, and having a good experience in launching spacecraft and satellites, the access to space is still limited to a few. The cost is the most important reason. Trying to minimize the cost and expanding accessibility to space to everyone, are the main objectives of CubeSats. Despite the dimensions of the satellite, CubeSats need a payload to work with, depending on the type of the mission. Moreover the contact with the satellite is important for success of the mission, and this contact is set with respect to the location of the satellite in orbit so this position must be known and studied.

CubeSats are small satellites that have a mass of no more than 1kg, and is 10 x 10x 10 cm, the design has been simplified so almost anyone can build them, which has the purpose of making space available to small projects. CubeSats can be combined to make larger satellites for larger payloads. The generated power comes from deployable solar panels, and the cost to build a CubeSat is much less expensive than a normal satellite.

This project deals with some of the aspects of the mission analysis of the MSU CubeSat. In order to study the orbit, the theory behind orbit determination and predictions are necessary to understand the simulations performed. These simulations provide an extensive knowledge of the performance of the satellite and have been carried out using a commercial software called STK. Moreover, from these simulations, it is explained how the influence of the different parameters affect the satellite in its orbit apart from explaining how it is affected by the external perturbations. The project is structured into four chapters. The first chapter introduces the concept of CubeSats giving information about the project of the MSU. The second chapter is centered in the mission analysis, explaining the necessary theory of this project and the third chapter describe the simulation results including the setting of orbit and perturbation analysis. The final chapter offers conclusions and recommendations for further study.

1.2 Definition of a CubeSat

In 1999, Cubesats were first developed at the California Polytechnic State University (CalPoly) and Stanford University. CubeSat technology is giving a chance for access space for universities and high schools previously unable to have such an opportunity to use for educational purposes. Besides facilitating the access to space, the objective of such an innovation, is creating a new area of research opportunities.

A CubeSat is a small satellite in the shape of a 10cm cube with a mass up to 1 kg (see Fig 1.1). The design has been simplified so almost anyone can build them, they then can be combined to make larger satellites in case of bigger payloads. Deployable solar panels and antennas make Cubesats even more versatile [1].

The main reason why Cubesats are more popular with schools and governments are because they are comparatively inexpensive and relatively easy to build. Currently over 60 industries, universities and high schools are involved in the development of CubeSats. Their cost and scientific objectives make them an ideal concept to reach space for anyone.

All CubeSats are either 1U with dimensions 10 x 10 x 10 cm, or having their height multiplied by 1.5, 2 or 3, and called respectively, 1.5U, 2U or 3U. A 3U CubeSat has dimensions of 10 x 10 x 30 cm and has a mass up to 3 kg.

One of the most important parts of Cubesat's design is the method of deployment. The most common deployer is the P-POD (Poly Pico-Satellite Orbital Deployer) designed by CalPoly. The purpose of the P-POD is to protect the primary payload (The CubeSat) and insert it into the proper orbit. The P-POD is designed to carry either three 1U CubeSats, 1U and a 2U CubeSats, or one 3U CubeSat. It also represent an interface between the CubeSats and the launch vehicle.

The P-POD has a rectangular shape box and constructed mainly from aluminum with either an electrically activated spring-loaded door, or mechanical ejection spring. Figure 1.2 shows a P-POD structure first developed and designed by Cal Poly.

Figure 1.1 Typical 1U Cubesat Structure. [1]

Figure 1.2 P-POD structure. [17]

1.3 History of CubeSats

In 1999, Cubesats were first conceptualized by Cal Poly and Stanford as previously discussed. Jordi Puig-Suari and Bob Twiggs [3] were the first to turn this concept from an idea into reality. Puig-Suari moved to California Polytechnic University with a lot of issues relating to satellites development for student, especially the huge cost and time problems that everyone had been facing [3]. But with the help of Professor Twiggs of the Aeronautic and Astronautic Department of Stanford University, he had the ability to find solutions to these issues, bringing to life a new concept and idea about satellites, which they are called now pico-satellites as known as Cubesats.

The first Cubesats were launched in 2003, and just nine years later, the number increased to one hundred CubeSats that were developed and designed and had been put in orbit using the P-POD deployment system. This systems is the current role for Cal Poly to provide them as an interface, maintaining the payload, and coordinating launch opportunities and networking ground stations around the globe dedicated to CubeSat operations. [1].

By the 2012, 112 CubeSat missions were flown; designed and manufactured by more than 80 organizations from 24 countries including universities, national space and defense agencies, private companies, amateur organizations, and even high schools. Moreover, CubeSat missions are not guaranteed to be fully functional, at least not in the past decade. From the first 100 CubeSats flown, nine of the first thirteen failed to achieve their missions, and only 8 of the first 36 CubeSats survived launch and operated successfully. Additionally in 2006, 14 CubeSats were lost in a launch failure of DNEPR-1 which was named after the Dneipr River in Russia. It is important to notice that the number of CubeSat each year from 1999 till 2012 has not diminished. [2]

The applications of CubeSat Payloads varies from Earth remote sensing to Astro-biology. The first application included many CubeSat missions such as QuakeSat that was designed to help scientists improve the chance of earthquake detection, and Exo-

Cube which is a space weather satellite designed by Poly-Sat. As for Astro-Biology, GenSAT 1 is a CubeSat designed by NASA in order to carry genetics experiments performed with E.Coli bacteria. The following list gives some examples of CubeSat applications with their corresponding missions [4]:

Earth remote sensing:

- QuakeSat: Earthquake detection.
- SwissCube: air glow phenomena in air atmosphere.
- ExoCube: space weather satellite.

Space tether:

- MAST.
- STARS: Space Tethered Autonomous Robotic Satellite.
- Tempo 3: Tethered Experiment Mars inter-planetery operations 3, demonstration the generation of artificial gravity.

Biology:

- GeneSat 1.
- PharmaSat 1.

All these applications certify the use of CubeSat concept in different areas. It can be used not only for the previous applications but also in industrial and research. Table 1.1 gives an idea about some of success rate of the Cubesat missions that have already flown. [4]

Table 1.1 Some CubeSat missions launch history [4]

Batch #	Date	LaunchVehicle	No Contact	Semi-Contact	Full contact	Total
1	30 Jun 2003	Rokot	2(1U)	1(1u)	2(1U)+ 1(3U)	5(1U)+1(3U)
2	27 Oct 2005	Kosmos-3M	1(1U)	0	2(1U)	3(1U)
3	22 Feb 2006	M-V	0	0	1(2U)	1(2U)
4	26 Jul 2006	DNEPR (failure)	0	0	0	0
5	16 Dec 2006	Minotaur 1	0	0	1(3U)	1(3U)
6	17 Apr 2007	DNEPR	1(1U)	3(1U)+ 1(3U)	2(1U)	6(1U)+1(3U)
7	27 Apr 2008	PSLV	0	0	3(1U)+ 1(2U)+ 2(3U)	3(1U)+ 1(2U)+ 2(3U)
8	23 Jan 2009	H-IIA	2(1U)	0		2(1U)
9	19 May 2009	Minotaur 1	0	0	3(1U)+1(3U)	3(1U)+1(3U)

It can be seen from Table 1.1 that the success rate of CubeSat missions is really high without considering the failures due to launch vehicles. From 35 CubeSats launched, 29 of them had contact with ground station. And 24 out of 29 they had a full contact with their corresponding ground station. It can be deduced from these data that indeed these satellites are useful for many applications.

Table 1.2 gives some more example about recent CubeSat missions that were launched in 2015. [5]

Table 1.2 CubeSats launched in 2015 with their corresponding missions. [15]

Name	**LV**	**Launch Date**	**Type**	**Organization**	**Mission**
ExoCube	ELANA-X	31 Jan 2015	3U	Cal Poly Poly Sat	Space weather
FIREBIRD II	ELANA-X	31 Jan 2015	1.5U x 2	Montana State University University of New Hampshire Los Alamos National Laboratory Aerospace Corp	Space weather
GRIFEX	ELANA-X	31 Jan 2015	3U	University of Michigan NASA JPL	Atmospheric studies technology
OptiCube 3	Atlas V	20 May 2015	3U	Cal Poly, SLO	Targets for orbital debris studies (Active)
AeroCube 8B	Atlas V	20 May 2015	1.5U	Aerospace Corp.	Unknown (Active)
AeroCube 8A	Atlas V	20 May 2015	1.5U	Aerospace Corp.	Unknown (Active)
OptiCube 2	Atlas V	20 May 2015	3U	Cal Poly, SLO	Targets for orbital debris studies (Active)
GEARRS-2	Atlas V	20 May 2015	3U	NearSpace Launch Inc	Technology/Communications (Active)
OptiCube 1	Atlas V	20 May 2015	3U	Cal Poly, SLO	Targets for orbital debris studies (Active)

Table 1.2 (continued)

BRICSat-P	Atlas V	20 May 2015	1.5U	U.S. Naval Academy	Transponder experiment, electric propulsion technology (Active)
PSat A	Atlas V	20 May 2015	1.5U	U.S. Naval Academy	Unknown (Active)
USS Langley	Atlas V	20 May 2015	3U	U.S. Naval Academy	Unknown (Active)

It can be observed From Table 1.2 that most of the Cubesats Launched in 2015 are active and working successfully. Again the same conclusion can be said, these CubeSats are useful for different applications, including military, research and even space exploration.

1.4 The CubeSat of MSU

This effort represents an orbital study of a proposed MSU CubeSat. The mission of this proposed CubeSat is to use an infrared camera for detecting forest fires. This effort represents the orbital design and trades necessary to achieve the mission as well as the analyses of different scenarios in order to determine the orbit shape, altitude, and lifetime of the MSU CubeSat that would be accurate for the mission mentioned previously.

The comparison of the results obtained will give an idea about the lighting times and access of the MSU CubeSat required in order to complete its mission during its operational lifetime.

Forest fires are a major issue that causes tragic loss of valuable thousands of hectares of forest and houses. Wildfires are a great threat to the ecological grown forests and environment. Thousands of forest fires cause disasters across the world every year. This phenomena became a research interest for many years, the amount of solutions studied to limit this harm to nature and wildlife are in hundreds. However, it's still a problem to solve.

Forests are considered as the keepers of earth's ecological balance. The results of these forest fires are usually observed when they have covered a large surface area, causing irreparable damage and devastating loss to the ecology, environment and atmosphere. In addition, forest fires have a long term consequences such as destructive effects on weather, global warming and also the extinction of rare species.

The problem concerned with forest fires is because these areas are unmanaged and contains a huge amount of trees, dry leaves and woods, which react as combustible materials and thus contribute in the initial fire ignition caused either by humans such as smoking or nature such as higher temperature with the presence of broken glass, and act like fuel source for later stages of fire.

In this context, the MSU CubeSat will have a primary payload that consists of an infrared camera that can detect the initial stages of forest fire ignition. This idea is considered as one of many solutions to limit this issue that concerns and could happen in any portion of the globe. The type of the camera that will be used in the CubeSat is an Aurora 1000SK. This infrared camera is a space grade camera with high definition resolution and dimensions 45x50x80mm, acting in low earth orbit. The estimated Infrared range is between 1000 nm and 1700 nm and could occasionally reach 2000 nm.

This important range coverage of Aurora 1000SK is considered not only useful for detecting forest fires, but also can be deployed as a multi-task space camera, Such as thermal imaging, temperature sensing and Earthquake prediction.

Features related to Aurora 1000SK camera are cited as follows:

- 15 mm aperture with 0.15° field of view.
- Programmable operating parameters, including integration time and co-adding settings.
- Typical specifications: size 45 x 50 x 80mm, 280 g mass, 1 watt power requirement.
- Available in a range of focus formats.

CHAPTER II

MISSION ANALYSIS

2.1 Introduction to System Tool Kit (STK)

Systems Tool Kit, often referred to STK, is a software package from Analytical Graphics, Inc. (AGI) that allows engineers and scientists to design and develop complex dynamic simulations of real-world problems. Originally created to solve problems involving Earth orbiting satellites, it is now used in both the aerospace and defense communities to model complex system dynamics. [6]

In order to understand the use of STK, some basic definitions should be well defined:

Celestial mechanics: The field applies principles of physics, historically Newtonian mechanics, to astronomical objects such as stars and planets. Moreover, it is considered as a main part of astronomy that deals with the gravitational effects of celestial objects. It is distinguished from astrodynamics, which is the study of the creation of artificial satellite orbits.

*Astrodynamic*s: First determined from Isaac Newton's laws of motion and his law of universal gravitation. It is considered as the study of the motion of rockets, missiles, and space vehicles. It is a specific and distinct branch of celestial mechanics that focuses more broadly on Newtonian gravitation and includes the orbital motions of artificial and natural astronomical bodies such as planets, moons, and comets. Astrodynamics is principally concerned with spacecraft trajectories, from launch to atmospheric re-entry, including all orbital maneuvers, orbit plane changes, and interplanetary transfers.

Astronautics: is the branch of engineering that deals with machines designed to work outside of Earth's atmosphere, whether manned or unmanned. In other words, it is the science and technology of space flight. To perform the mission analysis, some kind of software is needed. The amount of data and calculations needed make it difficult and in some way impossible to achieve without specific software.

There exists different ways in order to create your own program for orbital calculations, this can be done using different software, some of them may be available for purchase, such as C++, Fortran, FreeMat, Numerit Pro or the most common one MATLAB, but the orbital analysis performed in this project has been done by the commercial software STK. [6]

Having more than 34,000 installations worldwide, and cooperating with different organizations such as NASA, ESA, CNES, JAXA, Boeing, Northrop Grumman, Lockheed Martin, DOD and EADS, STK is currently the state of the art software for orbital calculations and estimations. The product is currently used in various areas such as:

- Communications Analysis
- Space Exploration

- Missile Defense
- Spacecraft Mission Design
- Spacecraft Operations
- Geospatial Intelligence
- Unmanned Systems (UAVs)

In his early version, STK ran only on Silicon Graphics computers. However, as computers became more powerful, the software was converted to run on Windows. As the software expanded, more modules were added including the ability to perform calculations for different types of communication systems, interplanetary missions and radar.

After the addition of 3D viewing capabilities, military users adopted the tool for real time visualization of air, land, sea and space.

The analysis made in STK is called a scenario. In each scenario, a number of vehicles such as aircraft, missiles, satellites and ships must be created and introduced individually by adjusting the parameters and properties corresponding to each one of them including their constraints. In most of the cases, only a single scenario may exist containing different types of vehicles and different locations of ground stations needed for the analysis.

The results of the analysis made in STK could be generated in two ways, graphics or reports.

Each object could have multiple graphics and reports depending on its degree of importance in the analysis. In addition, through the use of the constellation and chains objects, multiple objects may be grouped together and the multipath interactions between them could be investigated.

2.2 Theoretical background of orbital mechanics

Orbital mechanics studies the motion of a spacecraft on a specific trajectory, called orbit, basing on the Newton's laws of motion and of universal gravitation. These laws are described as follows:

- 1st law: Every object in a state of uniform motion tends to remain in that state of motion unless an external force is applied to it.
- 2nd law: The relationship between an object's mass m, its acceleration a, and the applied force F is $F = ma$.
- 3rd law: For every action there is an equal and opposite reaction.

And from them, comes the three Kepler's law of planetary motion:

- 1st law: The orbit of every planet is an ellipse with the sun at one of the foci.
- 2nd law: The line joining a planet and the sun sweeps out equal areas during equal intervals of time as the planet travels around its orbit.
- 3rd law: The squares of the orbital period of planets are directly proportional to the cube of the semi-major axes of their orbit.

These laws can be applied to the motion of a satellite around a planet. In orbital mechanics, the spacecraft and the central body are considered as points with mass but without dimensions. As for describing the position and the speed of a point in a tridimensional space we need six parameters. In order to completely characterize the path of the satellite over an orbit we need the six so called Kepler's Orbital elements:

- The semi-major axis a.
- The eccentricity e.
- The true anomaly ϑ or ν.
- The inclination i.
- The longitude or right ascension of ascending node Ω or RAAN.
- The argument of perigee ω.

Sometimes the true anomaly is substituted by the time since perigee passage, introducing the position of the satellite on the orbit starting from the perigee: it's the only parameter that varies along the orbit as long as we maintain the hypothesis of ideal motion. These parameters are illustrated in Figure 2.1.

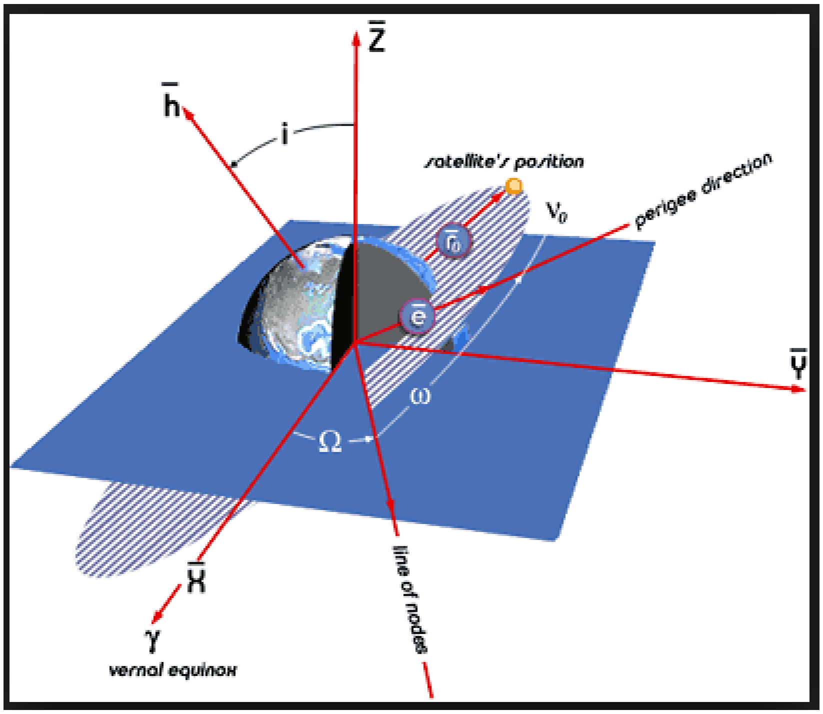

Figure 2.1 Orbital Parameters. [7]

The motion of the satellite is considered as ideal, and can be only determined by forces of gravity that links between the masses and the centrifugal force, neglecting any other perturbations such as aerodynamic drag or gravitational effects of other bodies.

From Newton's laws and the gravitational laws, we can define the orbital elements and other parameters that can be useful for the analysis. In order to proceed to the equations, we assume that r_p and r_a corresponds to the radius of perigee and apogee respectively, and μ is the earth gravitational constant. Thus, we can define the other parameters that remain constant:

- The angular momentum and its magnitude is:

$$\mathbf{h} = \mathbf{r} \times \mathbf{v} \quad ; \quad \boldsymbol{h} = |\boldsymbol{h}| = \boldsymbol{r}.\boldsymbol{v}\boldsymbol{cos}(\gamma) \qquad \text{(Eq. 2.1)}$$

Where r is the radius, v is the speed and γ is the flight angle

- The semi-major axis:

$$\boldsymbol{a} = \frac{\boldsymbol{r_p} + \boldsymbol{r_a}}{\boldsymbol{2}} \qquad \text{(Eq. 2.2)}$$

- The eccentricity:

$$e = \frac{r_a - r_p}{r_a + r_p} \qquad \text{(Eq. 2.3)}$$

- The orbit parameter which represents the radius of the circular orbit having the same angular momentum:

$$p = a(1 - e^2) = \frac{h^2}{\mu} = r_c \qquad \text{(Eq. 2.4)}$$

- Speed corresponding to the circular orbit with the same angular momentum:

$$v_c = \frac{\mu}{h} \qquad \text{(Eq. 2.5)}$$

- The energy:

$$E = -\frac{\mu}{2a} = \frac{v^2}{2} - \frac{\mu}{r} \qquad \text{(Eq. 2.6)}$$

V and r are the magnitude of speed and radius respectively.

- The Period:

$$T = 2\pi\sqrt{\frac{a^3}{\mu}} \qquad \text{(Eq. 2.7)}$$

Having the true anomaly ϑ, which represents the angle of the direction of the perigee and the current position of the satellite, we can calculate the radius in each point in the orbit:

$$r = \frac{p}{1 + e.\cos(\vartheta)} \qquad \text{(Eq. 2.8)}$$

And also, the perigee and apogee could be expressed as:

- The perigee:

$$r_p = r\ (\vartheta = 0) = \frac{p}{1+e} = a\ (1 - e) \qquad \text{(Eq. 2.9)}$$

- The apogee:

$$r_a = r\ (\vartheta = 0) = \frac{p}{1-e} = a\ (1 + e) \qquad \text{(Eq. 2.10)}$$

We could also define a link between time and true anomaly which gives an idea about the necessary time to go from one point to another. For a circular orbit, it is simple to define this quantity as the speed assuming it is always constant. However, for elliptical orbits it is a bit more complicated. In this context, Kepler suggested a solution to this issue by introducing a quantity M, called mean anomaly. This quantity represents the fraction of the orbit period which has elapsed since perigee. The quantity M is expressed as an angle, and defined as:

$$M - M_0 = n\ (t - t_0) \qquad \text{(Eq. 2.11)}$$

Where n is the average of angular velocity called mean motion.

The method of Kepler is not very accurate, it only calculates the average position and velocity. In order to have a more precise value, we need to introduce the eccentric anomaly E, which represents the angle between the perigee direction and the current position of the satellite projected onto the ellipse and perpendicular to the major axis as shown in Fig 2.2.

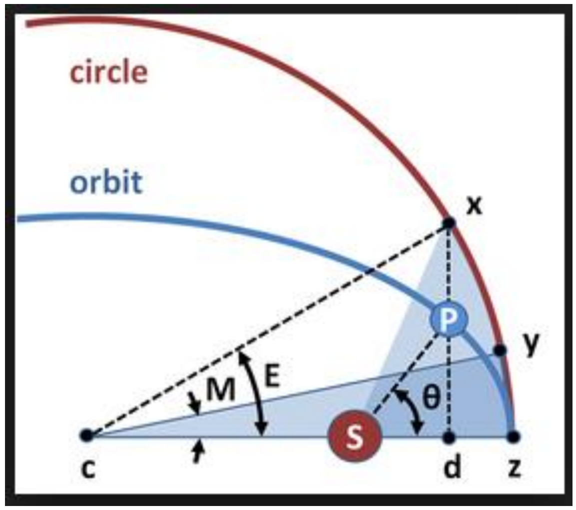

Figure 2.2 Eccentric anomaly E. [8]

We can relate the true anomaly with the following expression:

$$\tan\left(\frac{\vartheta}{2}\right) = \sqrt{\frac{1+e}{1-e}}.\tan\left(\frac{E}{2}\right) \qquad \text{(Eq. 2.12)}$$

And then we can figure out the time needed from the next expression:

$$t - t_0 = \sqrt{\frac{a^3}{\mu}}.(E - esin(E)) \qquad \text{(Eq. 2.13)}$$

2.3 Perturbation Techniques

Although the essential ideas of how objects move under gravitational force has been comprehended subsequent to the time of Isaac Newton, the determination of a body's position in orbit is significantly more intricate. With a specific end goal to have the capacity to choose the best orbital propagation, it is critical that we comprehend why this is so.

As indicated by Newton's Law of Universal Gravitation in Eq. 2.14, the gravitational attraction between two bodies is specifically relative to the result of their masses and inversely corresponding to the square of the separation between them. Newton concluded this law as the normal clarification for why an apple drops to the ground and moon is circling the earth.

$$F = G\frac{m_1 m_2}{r^2} \qquad \text{(Eq. 2.14)}$$

The satellite orbit's prediction is an important part of the mission analysis, it describes the most powerful aspect in the mission that affects mainly the power system and altitude control, and determines whether the mission is feasible. Computing the orbit of a satellite around a planet such as the Earth, more specifically in low earth orbit, has been analyzed in detail.

The computation of a satellite's orbit is not an easy task. The analysis of the orbit has to include many factors that may affect the trajectory of the CubeSat such as perturbations.

Starting with the gravitational perturbations, we can begin the simulations. In our straightforward model, we make an assumption that our two objects are point masses. The earth, in any case, is not a point mass. On the off chance that the earth were spherical and had a uniform thickness, be that as it may, it would at present be conceivable to regard it as a point mass [9] [10] The earth is not spherical (it swells at the equator) principally because of the outward compel of its own revolution. Neither does it have uniform thickness. It is enveloped with seas which swell under the tidal gravitation of the sun and the moon. The lighter material of the landmasses "floats" on the denser mantle.

With Newton's Law still satisfies, it is presently important to consider the gravitational impact of every detail of the earth on the satellite. Thus, our basic model has turned into significantly more complicated model.

Likewise we assumed that there were just have two gravitational masses in our unique model. We can't, be that as it may, overlook the gravitational impacts of the sun and the moon, however. There are non-gravitational forces to consider, too. The essential one, for some classes of earth orbits, is atmospheric drag. Overlooking this perturbation for a low earth orbit will introduce significant errors. Indeed, even tiny forces, such as sun radiation pressure, can significantly affect a satellite's orbit under the best possible conditions. [9][10]

The main factors that have a greater influence are perturbations such as:

- The non-spherical Earth.
- Atmospheric drag.
- Third-body perturbation.
- Solar radiation pressure.

Perturbative effects from the gravitational pull of the sun and other planets.

The primary force affecting the motion of our satellite is still the gravitational force. However, now we must take into consideration many additional perturbing forces in order to be able to accurately predict the position of the satellite in orbit. These additional forces will act in order to change our satellite's orbital trajectory from that of a true ellipse. Knowing which perturbations have the most significant effects and under which conditions will be extremely considerable.

Narrowing our analysis of the satellites in LEO, these satellites are mainly affected by the non-spherical nature of the earth and the atmospheric drag and even the thrusts in case of its implementation in the satellite. However, considering the satellites in the geosynchronous orbit, they are far away from the earth gravitational force but are still affected by lunar gravity, and even solar gravity. Effects such as atmospheric drag and radiation pressure are also very reliant upon the shape, size and mass of the satellite. [9]

2.4 Obit Prediction

2.4.1 Propagators

Orbits are not perfect shapes, this is the reason why orbits are simulated numerically for future time steps to predict where a satellite will be in the future, based on the types of perturbations cited before. The purpose of satellite orbit propagators is to provide high accuracy and precision in predicting the location of a satellite. There exists a wide variety of propagators, each propagator has a specific functionalities compared to others. These functionalities are used to develop precision that corresponds to the effects or forces taken into account for predicting the perturbations of the satellite's orbit. In STK, there are 2 basic types of propagators. The first type of propagator are analytical propagators, the analytical propagators use a closed-form solution of the time-dependent motion of a satellite to produce ephemeris or to provide directly the position and velocity of a satellite at a particular time. The second type of propagator are numerical propagators which performs numerical integration of the equations of motion for the satellite. However, the use of propagators in STK is limited, and so there exist only few propagators that could be implemented in the calculations of satellite's orbit perturbation. A list of the propagators available on STK is presented below. [11]

- Two-body
- J2
- J4
- HPOP (High Precision Orbit Propagator)

- SGP4 (Simplified General Perturbations)
- LOP (Long Term Propagator)
- Astrogator.

Figure 2.3 gives a representation about the propagators and the forces considered by each one for calculating the perturbation.

Table 2.1 Forces taking into consideration by each propagator

	Two-body	SGP4	J2	J4	HPOP	LOP	Astrogator
Earth gravity field	✓	✓	✓	✓	✓	✓	✓
J2		✓	✓	✓	✓	✓	✓
J4		✓		✓	✓	✓	✓
Sun Gravity		✓			✓	✓	✓
Lunar Gravity		✓			✓	✓	✓
Solar wind					✓	✓	✓
Atmospheric Drag		✓			✓	✓	✓
Other planets					✓		✓
Thrusts							✓

One issue related to the user version of STK is that some of the propagators cannot be used, which affects the output results, thus not all the propagators have the same accuracy. Table 2.1 provides definitions of some propagators used in STK. [16]

Table 2.2 Some examples of propagators used in STK [16]

Propagators	Description
Two-Body propagator	The Two-Body propagator take into consideration the force of gravity.
J2 propagator	The J2 perturbation is a first order propagator that considers for global variations in the elements of orbit due to oblateness. This propagator exclude some models in calculations such as drag or solar or lunar gravitational forces.
J4 propagator	The J4 perturbation is a second-order propagator that considers for global variations in the elements orbit due to oblateness. This propagator exclude some models in calculations such as atmospheric drag or solar or lunar gravitational forces. Note that The J4 propagator includes the first and second order effects of J2 and the first order effects of J4.

For further explanation of the general differences between those propagators, we extract the conclusion of Vallado [13] for examining the accuracy of propagation methods which uses two basic quantities, the accuracy of the state vector at some epoch which describes essentially the covariance, or uncertainty, and the accuracy of the state as it's propagated through time. Although the focus of this effort is not to discuss differential correction, it is necessary when defining the accuracy to use with a certain propagation technique within orbit determination. The propagation techniques can be classified in three categories, low, medium and high accuracy. Such a breakout distinguishes the state of the art (high), from the routine numerical operations (med), and the analytical (low). [13]

- Low - > 500m.

- Medium - 500m < 10m.
- High - < 10m.

The design accuracy of low routines could be limited as in a two-body propagator, or approximations for drag, and resonance, as in the SGP4 example. The low accuracy category includes analytical techniques for two-body, J2, SGP4 propagators. Medium routines accuracy effects uses the 4^{th} order Runge-Kutta, while high routines accuracy methods are the ones that uses numerical operations such as 8^{th} and 12^{th} orders Gauss-Jackson and Adams Bashforth.

Because the main propagators that takes into account more perturbation forces are SGP4 and HPOP. These two propagators will be included in the simulations of MSUSat orbit prediction.

2.4.2 SGP4 propagator

SGP4 is an acronym for Simplified General Perturbations No. 4. The SGP4 is a semi-analytical propagator that uses the two-line mean element (TLE) sets to propagate a satellite's orbit over time. The history of the SGP4 model goes back almost a half century. It was first introduced in 1980 in the Spacetrack Report Number 3[14]. The significance of the SPG4 model is that for the first time a model for orbit determination was proposed where the results were consistent with the data generated by North American Aerospace Defense Command (NORAD). The source code was originally FORTRAN arranged in five subroutines. The original SGP4 routine was used for Near Earth satellites and the SDP4 routine for the Deep Space objects. The most recent version of the SGP4 code by Vallado, however, combines these two subroutines into a single model. As mentioned in Spacetrack Report Number 3, the SGP4 model employs

Brouwer's gravitational and atmospheric drag models, and SDP4 in addition includes the third body effects of the moon and the sun, and certain sectorial and terrestrial spherical harmonics of the Earth. It must be noted that a lot of ambiguity exists around the development of the SGP4 code simply because no mathematical formulation has been published to back-up the source code. For example, Vallado refers to the Brouwer's and Clemence algorithm in his book and works through the gravitational effects up to J5, whereas in his published code he only uses the data up to J4.

While sensor sites and most military clients have switched to SGP4, the act of giving component set information that can be utilized as a part of either SGP or SGP4 has been protected. The essential distinction between the two component sets is the definition of mean motion and the atmospheric drag representation. While the SGP is a Kozai-based theory, SGP4 is a Brouwer-based theory.

The Brouwer-based theory is based on the Mean Element Theory. This theory began with the work of Lagrange himself, and has been developed further by many people over many years. It is a formal mathematical theory for approximating motion by separating the effects of fast motions and slow motion.[15]

2.4.3 HPOP propagator

HPOP is the High Precision Orbit Propagator. As its name implies, it uses a powerful propagation technique to incorporate sophisticated orbit perturbation models. HPOP deploys a variety of high-fidelity models including lunar and solar gravitational forces, and third-Body gravity, Atmospheric drag effects using either Jacchia or the Haris Priester model which takes into consideration daily variations in the height of the atmosphere due to solar heating among other parameters. In addition to radiation pressure which is a small force that originally propagate from the sun. [10][16]

The HPOP propagator has the ability to deliver accuracy on the order of 10 meters. This propagator uses numerical integration of the differential equations of motions in order to set up ephemeris. However, high precision is not without costs, it is the responsibility of user for picking force model settings that are relevant to the case being modeled and also ephemeris generation concede more computational time and effort than analytical propagation, which simply evaluates a formula. [10]

The study made for these propagators, provides differences and similarities between them. In addition to the forces taking into consideration for determining the different levels of accuracy of each of the propagators mentioned previously. This propagator is selected depending to the type of orbit where the satellite will be operating corresponding to its mission objectives.

In our analyses, the propagator that will be used in the simulations for orbit determination is the HPOP since it takes into consideration more forces applied on the trajectory of the satellite in the interest of maintaining high accurate results.

2.5 Lighting

At the point when the satellite is orbiting the Earth, its position relative to the Sun is consistently evolving. In general, the satellite is passing through three phases (see Fig 2.4 and 2.5) described below.

The position when the satellite is facing the sun, where the satellite spends most of its lifetime, provides full lighting of the satellite. The simulations performed in STK gives good evidence and proof of that.

The Umbra, this phenomena is described as the conical total shadow projected from the earth on the side opposite to the sun where the intensity of solar radiation is zero.

The Penumbra, which describes the transition zone between the two phenomena explained above. It is the partial shadow between the Umbra and the full light region. In this zone, the light of the sun is partially cut-off by the earth and the intensity of the sunlight is given between 0 and 1.

The position of the satellite with respect to the Sun influences the satellite in an imperative manner. In the event that the satellite is specifically uncovered to the daylight or it is under the shadow of the Earth will influence in the heat absorbed on the satellite and the solar radiation pressure perturbation force. These changes and variations affects the design and the lifetime of the satellite. In STK the user can perform calculations of duration, when the satellite is exposed directly to sunlight, and during the Umbra and Penumbra time.

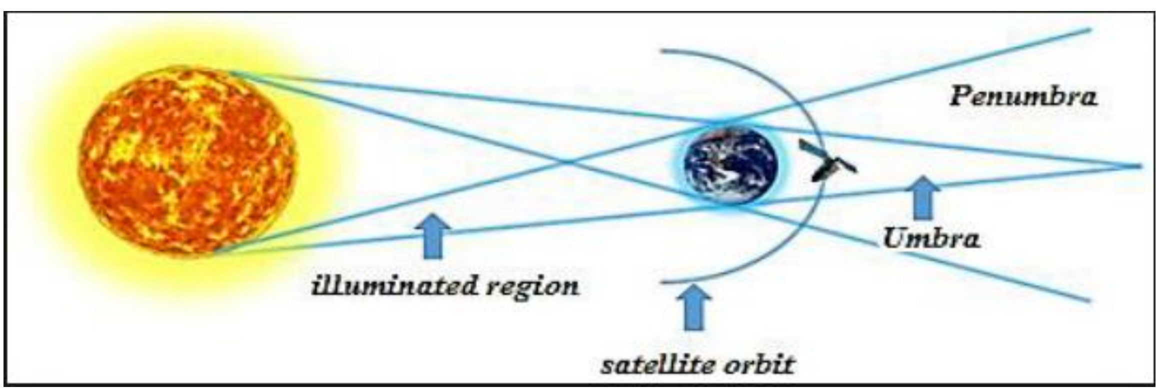

Figure 2.3 Lighting of satellite while orbiting the earth [18]

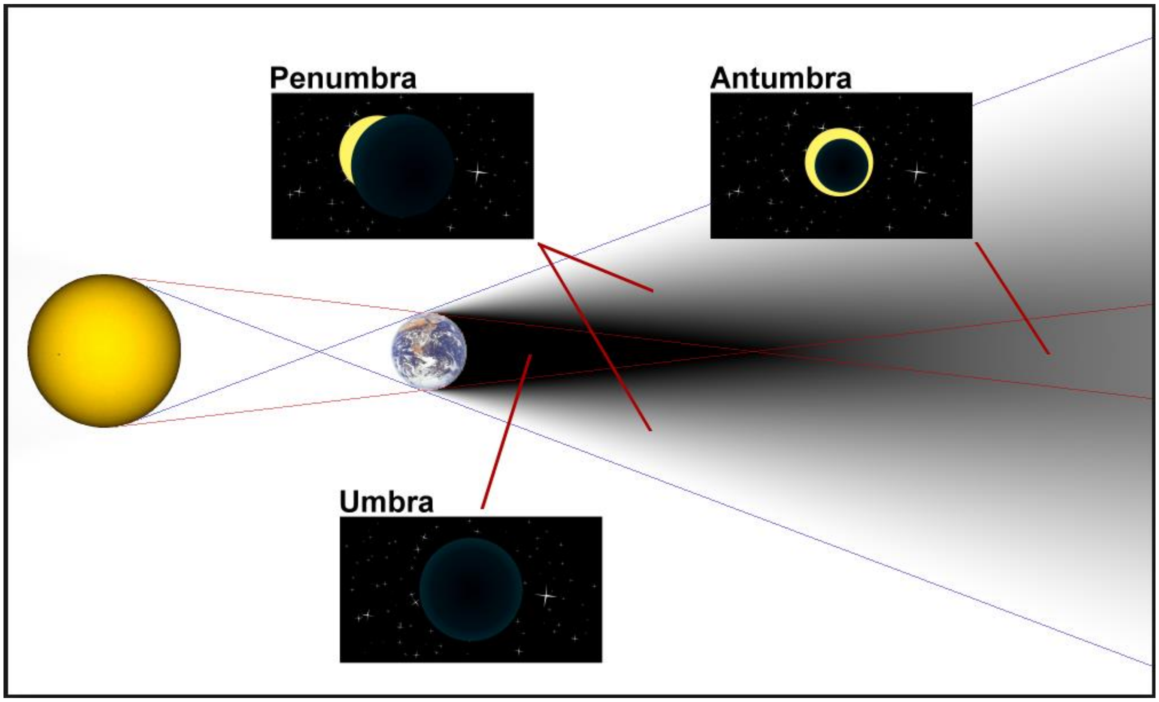

Figure 2.4 The effects of the Umbra and Penumbra [18]

CHAPTER III

SIMULATION RESULTS

3.1 Input Data

In this chapter, the results of the simulations performed in STK are given in a clear and precise manner, using tables, graphs and figures, and accompanied by explanations of the obtained results. A large amount of data have been analyzed and different simulations have been carried out using STK. Different cases have been studied and analyzed in order to determine the inclination and shape of the orbit. However, those last two parameters that describes the orbit, besides the way that MSU CubeSat will be launched, are still not yet decided. Thus, we set four scenarios for the MSU CubeSat in order to get accurate results and analyze them to make conclusions and determine the best shape and inclination desired for the MSU CubeSat in low earth orbit.

Table 3.1 describes the different four scenarios set for launching the MSU CubeSat. Recall that LEO is defined as Low Earth Orbit.

Table 3.1 Data used in the four scenarios set for MSU CubeSat orbit

	Launching Site	Operating Orbit	Shape of Orbit	Inclination	Apogee	Perigee
Scenario 1	International space station	LEO	Circular	51.64 °	404 Km	402 Km
Scenario 2	Cape Canaveral (FL)	LEO	Elliptical	28°	1250 Km	350 Km
Scenario 3	French Guyanna	LEO	Circular	Equatorial (0 °)	750 Km	750 Km
Scenario 4	Vandenburg Air Force Base	LEO	Circular	Sun-Synchronous ~100°	750 Km	750 Km

In addition, two types of propagators have been used for simulations in STK. The HPOP is the most reliable one when compared to the other propagators, and the SGP4, has similar reliability. The SGP4 propagator was used for comparison of the results obtained with the HPOP propagator.

Finally, simulations have been carried out during one year period in order to obtain accurate results of the lifetime of the satellite, starting from 1 September 2016 until 1 September 2017 with an elevation angle of 15°.

3.2 Lifetime comparison

The lifetime of a satellite is one of the major issues in the mission design analysis. This is the time when the satellite will be operating in the earth orbit, before falling into the atmosphere or continue to orbit as a debris.

The decay of a satellite is strongly influenced by forces acting on it. In LEO, atmospheric drag is one of the forces that has the greatest influence on the satellite because of its lower altitude and its closer location relative to the earth. In other words, if you want your satellite to have a lifetime counted in years you need a propulsion system to change the satellite's altitude when approaching dangerous low altitudes. This type of propulsion system is considered heavy for a CubeSat to carry, so no propulsion is added to the payload.

There exists certain other forces such as solar radiation pressure and other perturbations from other planets that acts on the satellite trajectory but their effect is relatively small.

In order to simulate this parameter, the Lifetime module from the STK professional edition is needed so it has been simulated using the evaluation license provided by STK. Propagators used in this simulation are the HPOP and the SGP4, previously explained. Additionally, simulations have been performed for the four scenario cases shown in Table 3.1. The atmospheric density model and the radiation coefficient used is the same as for the HPOP simulations. Taking reflection coefficient as 1.5 and drag coefficient as 2.2. Other input parameters are shown in Figure 3.1.

3.2.1 Input data

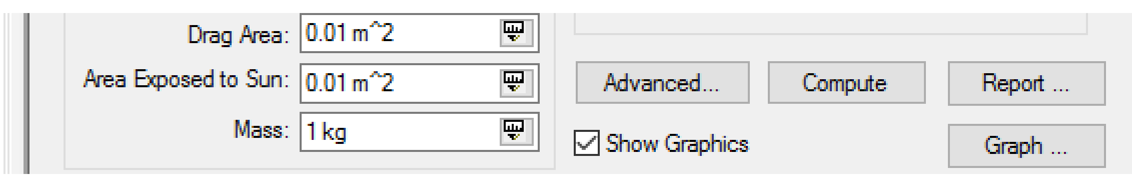

Figure 3.1 Input data for computing the lifetime of the MSU CubeSat

The main reason for taking drag area and the area exposed to sun equal to 0.01 m^2 is that our CubeSat is a 1U CubeSat. Thus, we estimated that only one face will be exposed to the sun. Tables 3.2 and 3.3 shows the results for the lifetime of MSU CubeSat obtained using HPOP and SGP4 propagator respectively.

Table 3.2 Results obtained for lifetime corresponding to the four scenarios using HPOP propagator.

Scenario cases	Lifetime
Scenario 1	362 days
Scenario 2	5.3 years
Scenario 3	190.6 years
Scenario 4	238.9 years

Table 3.3 Results obtained for lifetime corresponding to the four scenarios using SGP4 propagator.

Scenario cases	Lifetime
Scenario 1	1.2 years
Scenario 2	9.6 years
Scenario 3	412.4 years
Scenario 4	435.6 years

As shown, the lifetime of MSU CubeSat is estimated to be more than 400 years in some cases. However, let's not forget that this is just an estimation based on some parameters in STK. In addition, there exist no reference that prove this number is wrong because no satellite have flown this time range.

The difference in Lifespan of MSU CubeSat can be explained first from the different orbital shapes simulated for MSU CubeSat. Second, the difference of years between all the scenario cases is observed and can be only explained from the effects of the atmospheric drag on MSU CubeSat in different altitudes.

Finally, it can be also observed from the results obtained that the lifetime simulated using HPOP is lower than the lifetime simulated using SGP4. This can be explained because of the forces that each one takes into consideration. The HPOP for example takes into account solar wind, while SGP4 doesn't. In case of doubt, it is advised to take the value of HPOP which has a more reliable atmospheric drag model. Figure 3.2 represent the decay of MSU CubeSat in scenario 1 using HPOP.

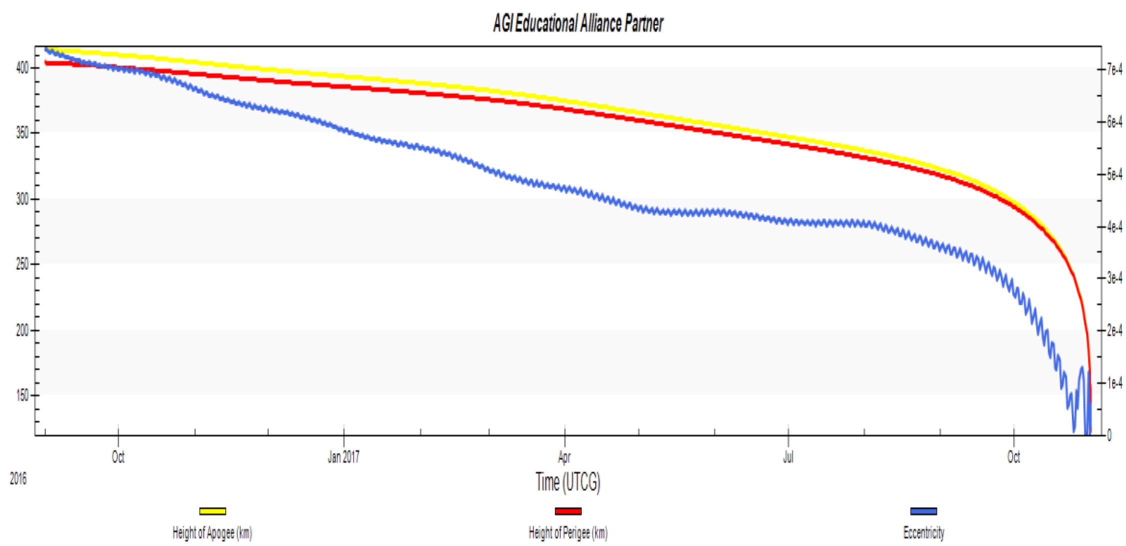

Figure 3.2 Decay of lifetime of the MSU CubeSat for scenario 1 in STK using HPOP propagator.

3.3 Lighting comparative

In this section, the simulations performed in STK for lighting of MSU CubeSat using the HPOP propagator for the different four scenarios are presented.

The results obtained in this section are important for determining which of the orbits will provide a sufficient lighting for MSU CubeSat that will be charging the power needed for the battery. In our case, the MSU CubeSat would orbit in low earth orbit so at some point it would face directly the sun but in other case, it would be at the shadow of the earth.

Depending on which of these situations the satellite is, in tables and graphics about the time the satellite is at lighting, penumbra, and umbra are shown for the four different scenarios in the following Table 3.4, 3.5, 3.6, 3.7 and Figures 3.3, 3.4, 3.5 and 3.6.

Table 3.4 Lighting, penumbra, umbra times of MSU CubeSat for scenario 1.

	Min Dur.	**Max Dur.**	**Mean Dur.**	**Total Dur.**
Lighting times	31min 15s	3d 16h 11min	1h	3M 9d 22h 22min
Umbra	1min 51sec	37min 19sec	32min 47sec	1M 24d 15h
Penumbra	8 sec	6min 44sec	12 sec	16h 33min

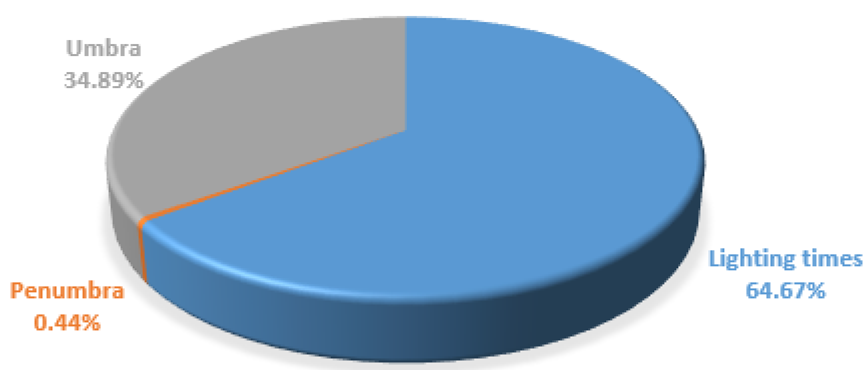

Figure 3.3 Total duration lighting percentage for Scenario 1.

The results obtained for scenario 1 show that more than 64% of the time the satellite is orbiting the earth it is directly facing the sun. We can notice also that the maximum continuous time MSU CubeSat would be facing the sun is 3.6 days. Note that the sum of the total duration of lighting, penumbra and umbra times is calculated such that it is not equal to one year but only 150 days. However, in the lifetime section we found a value of 362 days and this proves that the lifetime calculated in STK is not time accurate and its considered only as an estimation of the satellites lifetime.

Table 3.5 Lighting, penumbra, umbra times of MSU CubeSat for scenario 2.

	Min Dur.	Max Dur.	Mean Dur.	Total Dur.
Lighting times	4min	1h 17min 51sec	1h 6min 23sec	7M 29d 1h 52min
Umbra	20min 54sec	36min 37sec	33min 14sec	3M 29d 16h
Penumbra	7 sec	26 sec	10 sec	1d 6h 40min

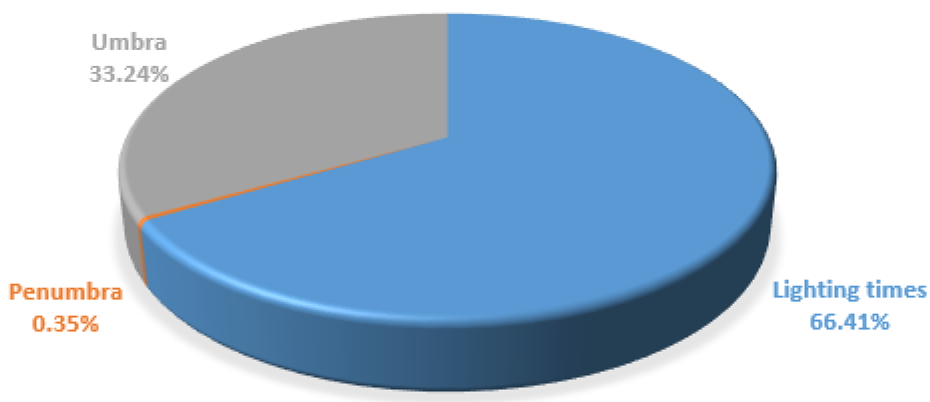

Figure 3.4 Total duration lighting percentage for Scenario 2.

The results obtained from scenario 2 have similar values to the satellites orbiting in the elliptical shape. Here we notice that more than 66% of the time, the satellite will be facing the sun and the maximum duration is only 1 h 17min compared to the previous case which is 3.4 days. However the average is approximately similar.

Table 3.6 Lighting, penumbra, umbra times of MSU CubeSat for scenario 3.

	Min Dur.	Max Dur.	Mean Dur.	Total Dur.
Lighting times	30min 58sec	1h 5min 39sec	1h 3min	7M 27d 21h 36min
Umbra	33min 34sec	35min	34min	4M 6d
Penumbra	8sec	10sec	9sec	27h 22min

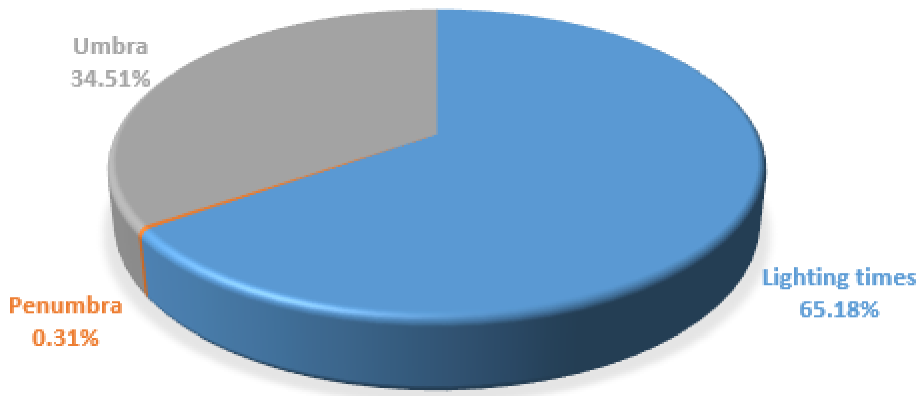

Figure 3.5 Total duration lighting percentage for Scenario 3.

The scenario 3 gives similar results obtained in the previous scenario cases, with more than 65% of the time the satellite would face the sun and a maximum duration around 1 h 5 min, and also an average of 1 h 3 min which is about the same as the averages in the previous lighting results.

Table 3.7 Lighting, penumbra, umbra times of MSU CubeSat for scenario 4.

	Min Dur.	Max Dur.	Mean Dur.	Total Dur.
Lighting times	1h 17min 51sec	6M 22d 3h	5h26min	11M
Umbra	33sec	21min 12sec	16min 52sec	17d 3h
Penumbra	22sec	6min	37sec	1d 7h

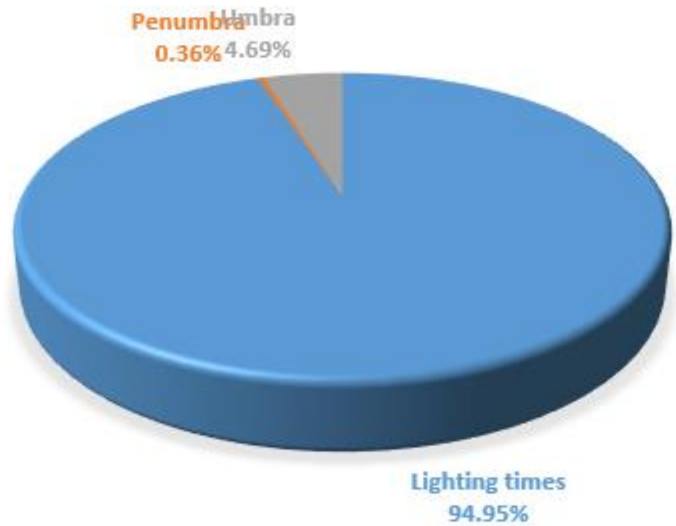

Figure 3.6 Total duration lighting percentage for Scenario 4.

In this case, the results show us that more than 94% of the time, the satellite will be directly facing the sun. In addition, the maximum duration obtained is considered to be more than half a year so during this time the satellite will be continuously facing the sun for a long period of time. This is due to the fact that most of the time the orbit would have an orientation in the sense that it would continuously face the sun in some subsequent revolutions.

Figure 3.7 gives an idea about the lighting, penumbra and umbra times of MSU CubeSat for the different four scenarios described above.

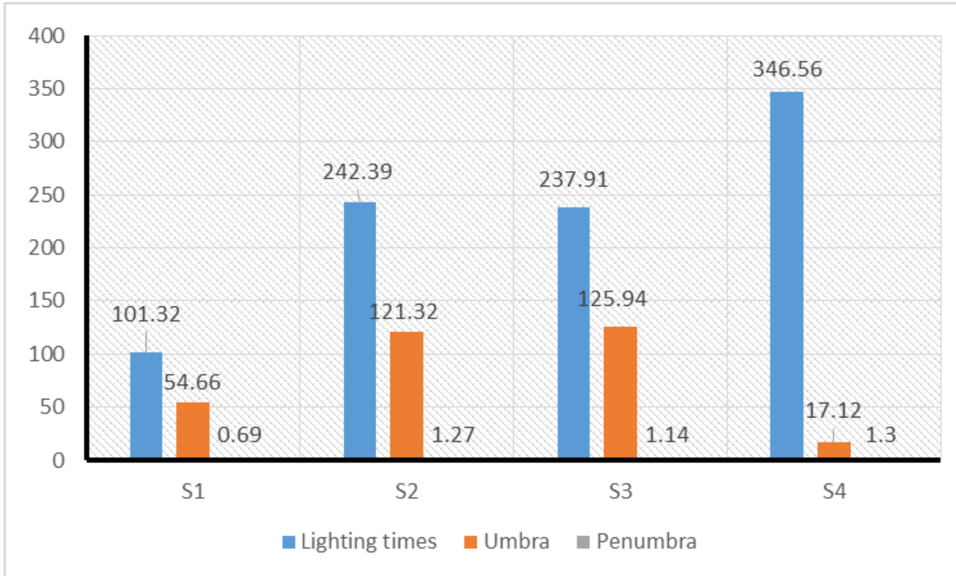

Figure 3.7 Lighting comparative for the four different scenarios in days.

Based on Figure 3.7, the previous comments are satisfied and the lighting times in all four scenarios have more than 60% of the total time. This graph also show us that the scenario 4 is the best case when MSU CubeSat will have a continuous time facing the sun. Figure 3.8 represents the data obtained for lighting, penumbra and umbra times for scenario 4 during 1 year simulation.

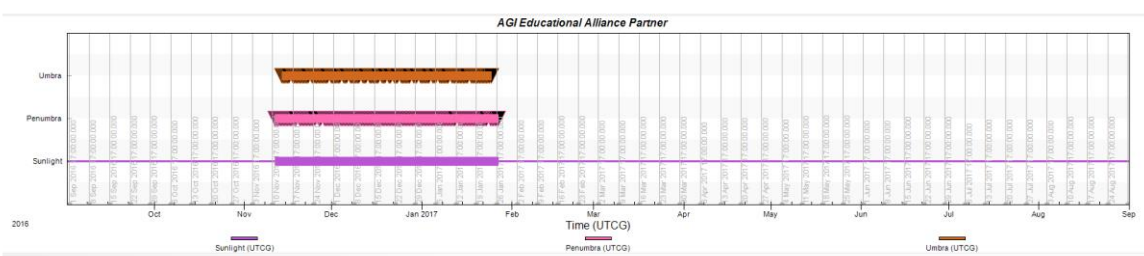

Figure 3.8 Lighting, penumbra and umbra times for scenario 4.

Note that the lower graph represents the lighting times, the middle graph represents the Penumbra and the upper graph represents the Umbra.

In the next experiment, we reduced the simulation time from 1 year to 1 day simulation. This change was done in order to get a clear graph besides having a good idea about what is happening to the lighting of MSU CubeSat during 1 day. Figure 3.9, 3.10, 3.11 and 3.12 and tables 3.8, 3.9, 3.10 and 3.11 shows the lighting times for MSU CubeSat in different four scenarios during 1 day simulation.

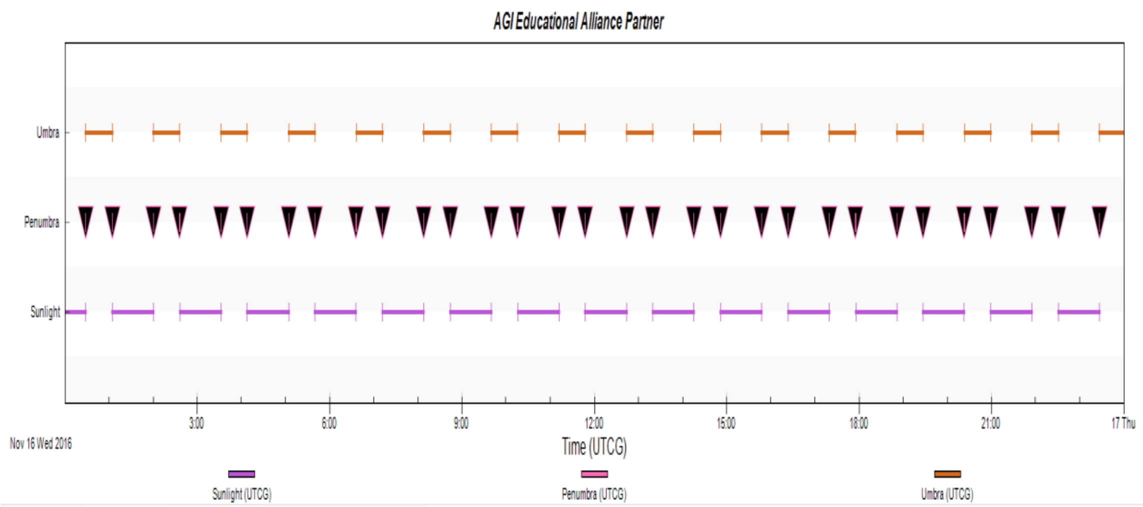

Figure 3.9 Lighting, penumbra and umbra times for scenario 1 during 1 day simulation.

Table 3.8 Lighting times for scenario 1 during 1 day simulation.

	Min. Dur	**Max Dur.**	**Avg Dur.**	**Total Duration**
Lighting times	29min	56min	54min	14h28min

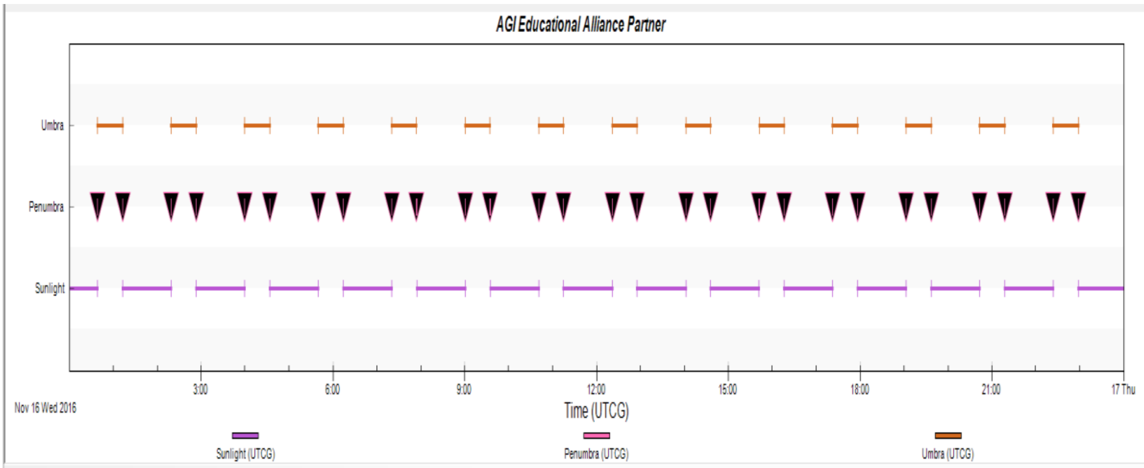

Figure 3.10 Lighting, penumbra and umbra times for scenario 2 during 1 day simulation.

Table 3.9 Lighting times for scenario 2 during 1 day simulation.

	Min Dur.	**Max Dur.**	**Avg Dur.**	**Total Dur.**
Lighting times	39min	1h06min	1h04min	16h10min

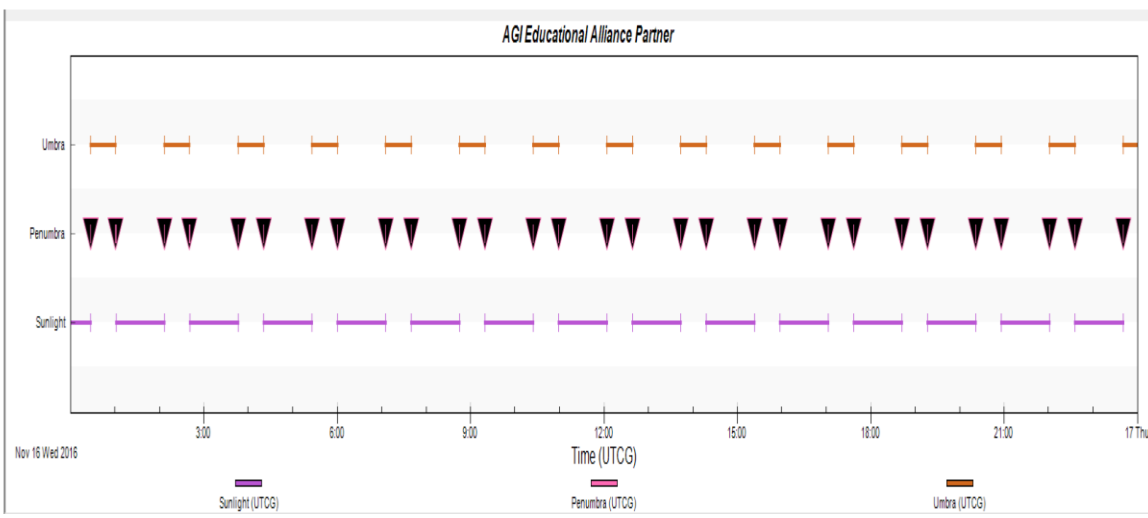

Figure 3.11 Lighting, penumbra and umbra times for scenario 3 during 1 day simulation.

Table 3.10 Lighting times for scenario 3 during 1 day simulation.

	Min Dur.	Max Dur.	Avg Dur.	Total Dur.
Lighting times	27 min	1h05min	1h03min	15h39min

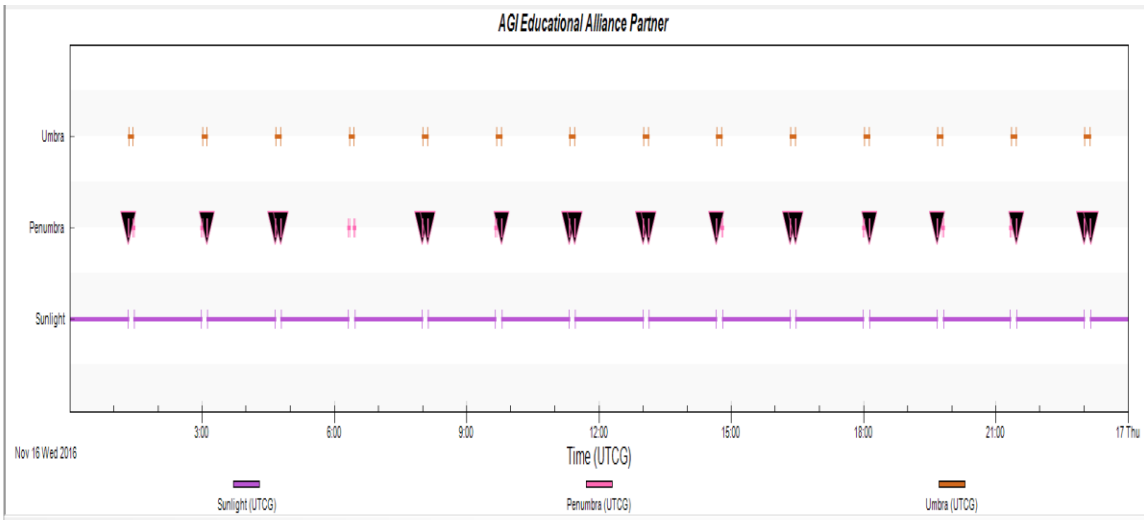

Figure 3.12 Lighting, penumbra and umbra times for scenario 4 during 1 day simulation.

Table 3.11 Lighting times for scenario 4 during 1 day simulation.

	Min Dur.	Max Dur.	Avg Dur.	Total duration
Lighting times	51min	1h36min	1h28min	22h

From previous graphs and tables, the difference of lighting times between all the scenarios set for this project have been simulated during1 day. The results obtained in scenario 1, 2 and 3 have a slightly similar values of lighting of MSU CubeSat which is around 15h. Compared to the scenario 4 the amount of time corresponding to the lighting of MSU CubeSat is found to be 22h. So scenario 4 has the highest amount of lighting times compared to the other cases. This remark satisfies the previous results obtained in one year simulation.

3.4 Accessibility comparative

Depending on the time you can access the satellite, the operators would be able to give more commands to the satellite, download more data or just trying to have more time to control it. This time is affected by the shape of the orbit independently of its inclination. Simulation results obtained are presented in Table 3.12 that represent number of access per year for MSU ground station in the four scenarios.

Table 3.12 Accessibility to MSU facility in all scenarios during 1 year simulation.

	# AY	**# AD**	**Min Dur. (sec)**	**Max Dur. (sec)**	**TDY (sec)**	**MAD (sec)**
Scenario 1	422	1.3	5.8	318	1d 1h 40min	220
Scenario 2	1465	5.2	20	805	7d 12h 37min	443
Scenario 3	No Access	No Access	No Access	No Access	No Access	No Access
Scenario 4	1107	2.8	26	510	5d 3h 7min	401

AY: Number of access per year
AD: Number of access per day
TDY: Total duration per year
MAD: Mean Access duration

Results show that indeed, a higher altitude would benefit the access of the satellite. However, a larger distance means more power must be transmitted and which nearly all the times translates to a bigger antenna which also means more weight for the CubeSat. It is true that this increase in distance is not a major problem to deal with but it is important to take it into account.

For scenario 3, which is equatorial orbit shape (0 degrees), there is no line of sight between the MSU CubeSat and the facility located at Latitude : 33.454° and a Longitude : -89.2° corresponding to Starkville, Mississippi, United States, due to the low altitude taken which in this case 750x750 Km .

Figure 3.13 and 3.14 represents Access to MSU facility in one year simulation for scenario 1 and 2, respectively.

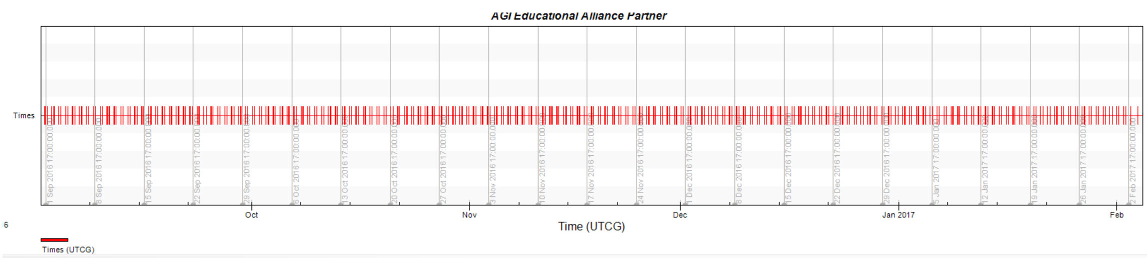

Figure 3.13 Scenario 1 access graph during 1 year simulation in STK.

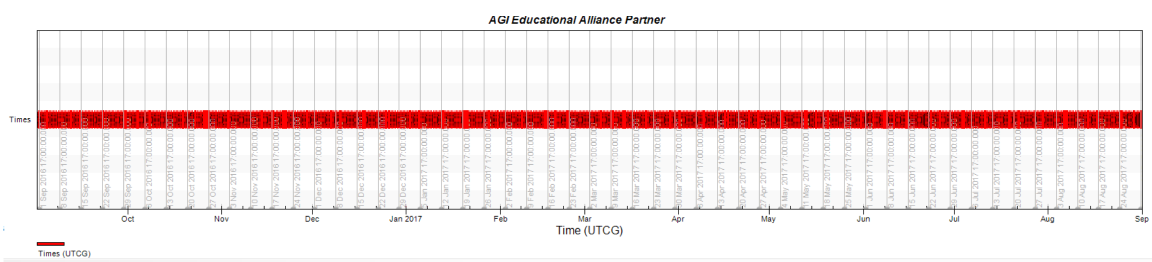

Figure 3.14 Scenario 2 access graph during 1 year simulation in STK.

In order to get a clear and clean graph of accessibility, we reduced the timeframe from a 1 year to a 3 days simulation. This has been done to have visibility for access results and get an approximate estimation for the number of access for 3 days. Table 3.13 and Figures 3.15, 3.16 and 3.17 shows the results obtained for this simulation.

Table 3.13 Accessibility to MSU facility in all scenarios in 3 days simulation.

	# Access	# AD	Min Dur. (sec)	Max Dur. (sec)	TD (sec)	MAD (sec)
Scenario 1	10	3.33	90	318	38min 21s	230
Scenario 2	12	4	253	560	1h19min	399
Scenario 3	No Access	No Access	No Access	No Access	No Access	No Access
Scenario 4	10	4.1	113	508	1h 2min	367

\# Access: Number of Accesses
\# AD: Number of access per day
TD: Total duration
MAD: Mean Access duration

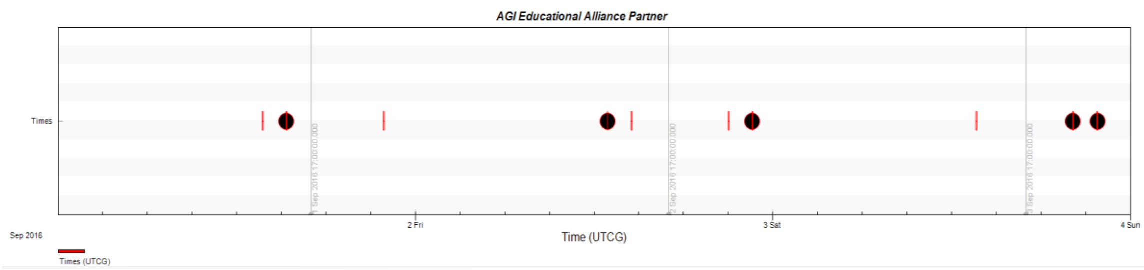

Figure 3.15 Access graph for scenario 1 during 3 days simulation.

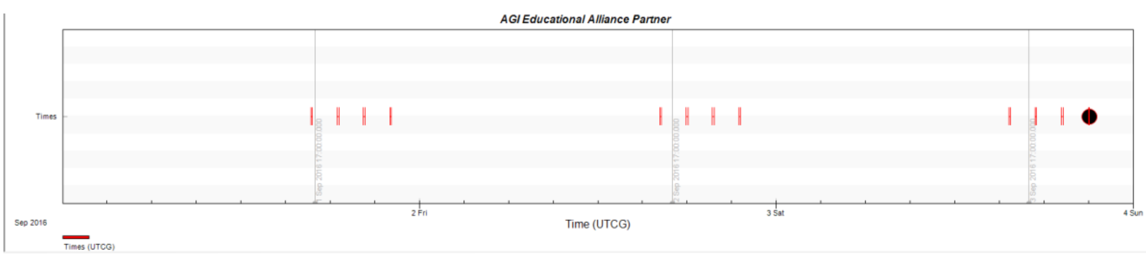

Figure 3.16 Access graph for scenario 2 during 3 days simulation.

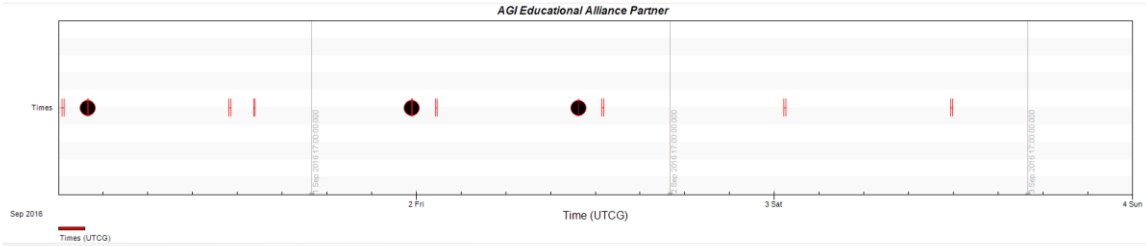

Figure 3.17 Access graph for scenario 4 during 3 days simulation.

From these graphs, the access time of MSU CubeSat to the facility is counted in minutes and the number of access during this period for scenario 1 and 4 is found to be 10 accesses. As for the scenario 2 the number of access is 12. The difference number of access between the cases studied is explained by the change in altitude, so the higher the altitude, the more access you get to the facility.

In addition, The MSU CubeSat would be able to be seen more time during the day, which is translated to more security to the Cubesat. The main drawback of increasing the orbit apogee to 1250 km (scenario 2) would be the difficulty in communications.

As previously stated, the time to communicate with the CubeSat is found to be a few minutes in each access during this simulation and in order to study the consistency of access time during one year, we simulate each 2 days separately. Figure 3.18, 3.19 and 3.20 shows the results obtained from these simulations.

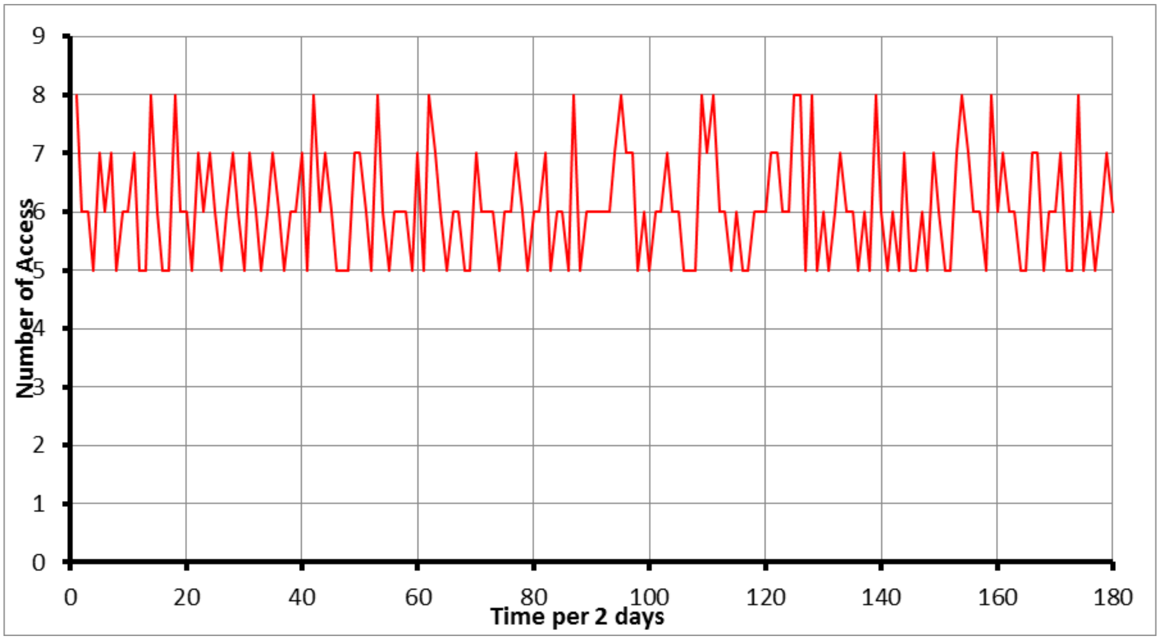

Figure 3.18 Number of Access to MSUS facility every 2 days simulation for 1 year.

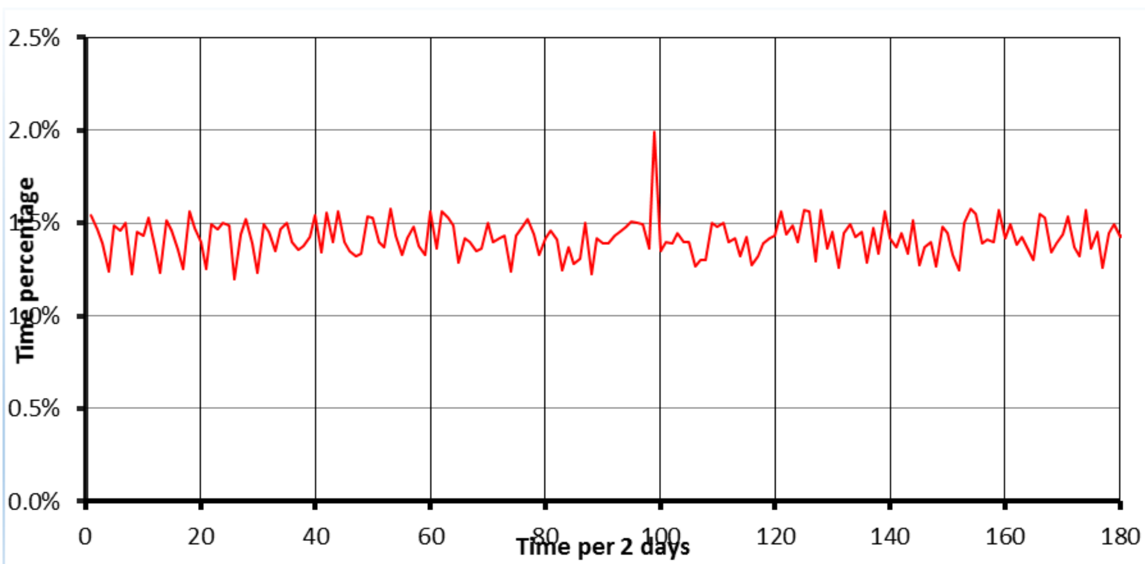

Figure 3.19 Access time percentage every 2 days simulation for 1 year.

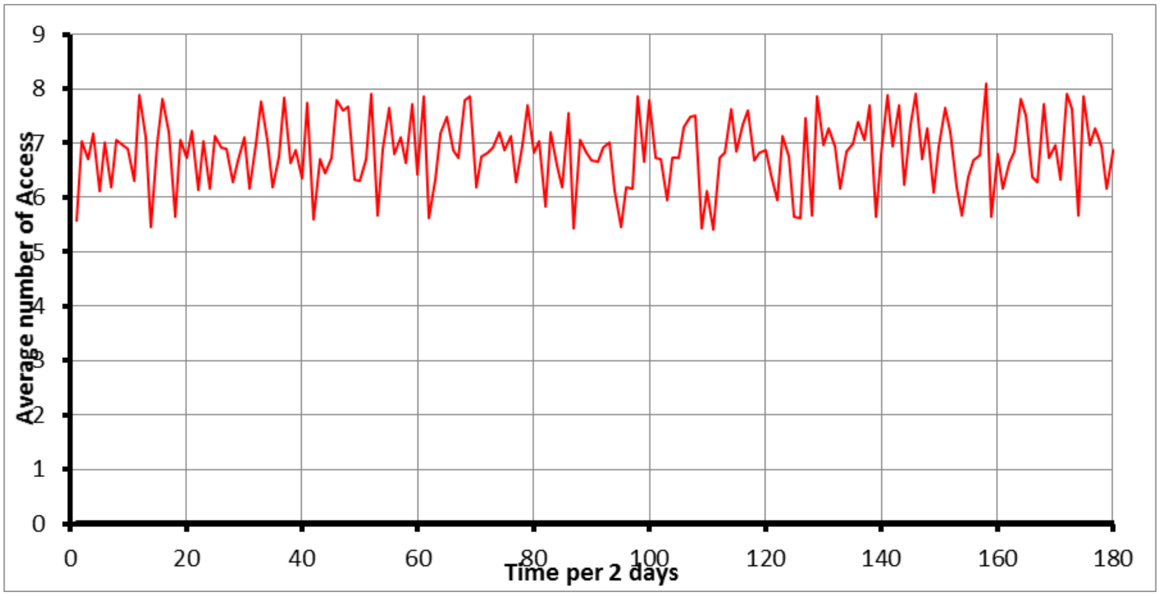

Figure 3.20 Average time per access in minutes for every 2 days simulation during 1.

The results obtained in Figure 3.18 show us that there is fluctuations between 5 and 8 access per two days, in Figure 3.19 we've noticed that the CubeSat has an average access time of 1.5% which is around 45 minutes per 2 days during 1 year. As for Figure 3.20 the average time per access can go from 5.8 to 7.9 minutes.

These results indicate that the time of access is consistent within 1 year and it is seen to be 1.5 %. However this time range is considered to be low in order to communicate and download data from the CubeSat.

There exist a global network of ground stations located worldwide in facilities and universities where the satellites could have access to them independently while orbiting the earth. This characteristic could be useful for the CubeSat and so he can have access to different ground stations in different part of the world which will improve the exchange of data with the satellite and also the total number of access to the facilities in each orbit.

In addition to MSU facility, we set two more ground stations, The International university of Rabat (UIR) located in Morocco, North Africa with a Latitude 33.98° and a longitude -6.72° and North Norcia Deep Station (NNDS) located in Australia with a latitude -31.04° and a longitude 116.19° are candidate ground stations.

Table 3.14 and Figure 3.21 shows results obtained for the four scenarios in the three different ground stations during one year simulation.

Table 3.14 Total number of accesses to the ground stations in 1 year simulation.

	MSU	**UIR**	**NNDS**
Scenario 1	422	977	795
Scenario 2	1465	2358	2404
Scenario 3	No Access	No Access	No Access
Scenario 4	1107	2104	1903

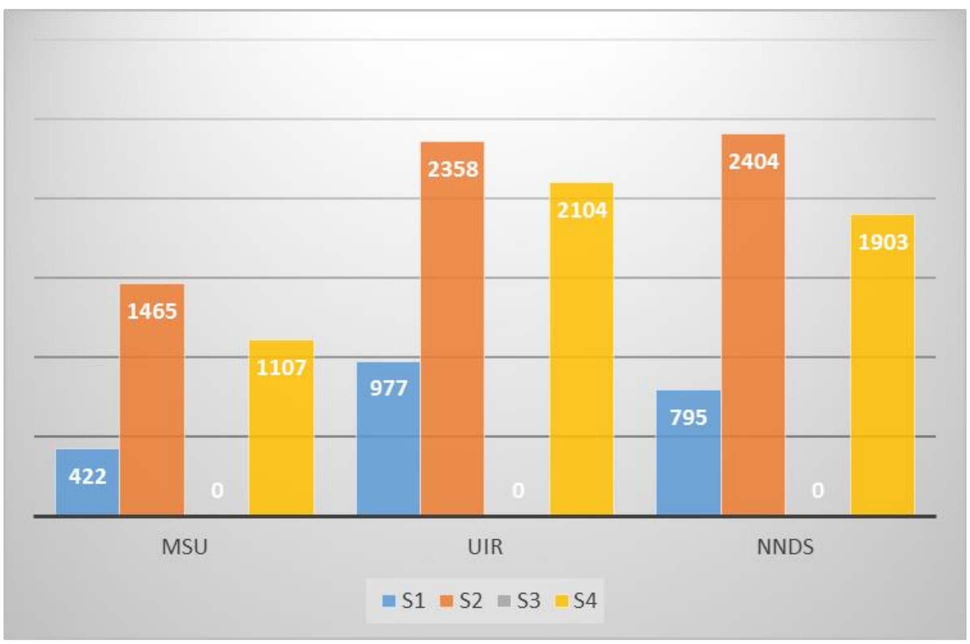

Figure 3.21 Accessibility for each ground station for different scenarios.

The comparison observed in Table 3.14 and Figure 3.21 indicates that the MSU ground station has the lowest number of access compared to the number of access corresponding to UIR and NNDS. This is primarily due to the different locations of ground stations and their correspondence to the trajectory of the CubeSat. Moreover, the scenario that has the largest number of access is found to be scenario 2 due to its high altitude which is 1250x350 Km. Note that results shows zero access for scenario 3 as we already got the same value in the previous simulations for this specific scenario.

3.5 Propagator comparative

3.5.1 Mean Access Duration

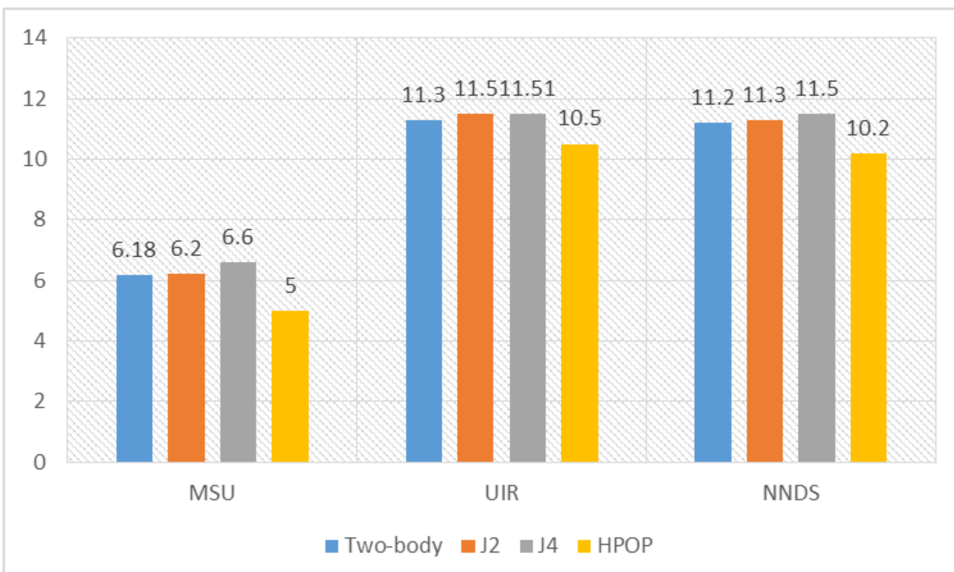

Figure 3.22 Mean access duration comparative for three ground stations using different propagators.

From results obtained in Fig 3.22 the HPOP propagator gives considerably different results than the other three propagators. This propagator is the most accurate of the four because it takes into account not just the gravitational force from the Earth, the J2 and the J4 terms but also the atmospheric drag and the gravity coming from the Sun and the Moon. Because of all these effects, the duration a ground station is accessing the satellite is smaller than that of the other propagators. In each revolution, the satellite is closer to the Earth because of the atmospheric drag so the time passing through the ground stations would be smaller each time.

3.5.2 Lighting:

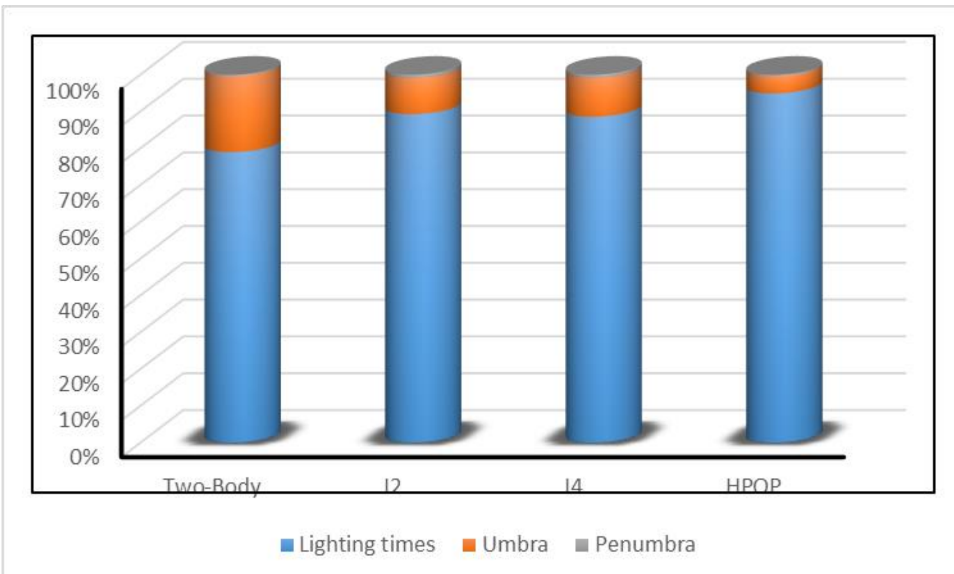

Figure 3.23 Lighting properties comparative for different propagators.

When simulating the effects of propagators in lighting seen in Fig. 3.23, similarities arise from those regarding accessibility. J2 and J4 propagators have nearly exact results while the other ones, the two-body and the HPOP propagators are significantly different. We can extract from Fig. 3.23 that when using the HPOP propagator more lighting is obtained. However, using the two-body propagator, we obtain less lighting probably because of the inconsistency of the data thus just the gravitational force of the Earth is taken into account as a perturb acceleration. Note that these comparative was made for scenario 4 which is sun-synchronous.

CHAPTER IV

CONCLUSIONS AND RECOMMENDATIONS

The objective of this effort has been to analyze some aspects of the orbit determination and mission analyses of the Cubesat that MSU proposing to design and launch. In order to perform this analyses, interpreted study of orbital mechanics has been undertaken and a series of investigations concerning propagators and perturbation effects have been analyzed. Also, simulations have been performed using the commercial software STK which provides a great variety of parameters. In addition, it is important to note that this project is one of the first ones regarding the proposed Cubesat at Mississippi State University. The data contained does not represent a final design and changes may appear during the project in the upcoming months.

It is recommended to use the HPOP propagator in other projects related to orbital determination analyses as being done in this effort. This propagator provide a few differences comparing to SGP4, these differences can change the orbital elements during a large amount of time. Perturbations affecting the satellite have been studied independently and all together concluding that the atmospheric drag and the non-spherical shape of the Earth are the ones that affect more the satellite no matter which orbit is being used. Indeed, the shape of the orbit is really important in the mission analysis. Among other differences, using the four different shapes of orbit can produce a variation in lifetime.

Moreover, the results obtained for lighting of MSU CubeSat are satisfactory for all the scenarios studied. It could be recommended to make an additional study and analyses of satellite resistance for long time lighting. In the other hand, the accessibility results obtained are relatively short in order for the CubeSat to communicate with the MSU ground stations.

The matter of choosing which orbit MSU CubeSat will operate depends mainly on the type of the mission. In this case, detecting forest fires using IR camera requires the satellite to orbit the earth in such a way it passes over the same location at the same local solar time. Such an orbit can a place a satellite in direct sunlight. Thus, the orbit suggested is the Sun-Synchronous orbit.

Finally, this project and its results could be a good help and a useful information for other studies. Further work should be done such as knowing the specific requirements of the battery, payload details and launch vehicle in matter of orbit in order to obtain more accurate results.

REFERENCES

[1] CubeSat community website, URL: http://www.cubesat.org/, Accessed December, 2015.

[2] Swartwout, M. (2013): JoSS, The First one hundred Cubesats: First look. Vol. 2, No. 2, pp. 213-233

[3] Interview with Jordi Puig-Suari, URL: http://www.publico.es/ciencias/181834/cubesat/satelite/elevado/cubo, Accessed March, 2016.

[4] List of CubeSat satellite missions, URL: http://mtech.dk/thomsen/space/cubesat.php, Accessed December, 2016.

[5] List of CubeSats, //en.wikipedia.org/wiki/List_of_CubeSats, Accessed, May, 2016

[6] Analytical Graphics, Inc. website, URL:http://www.stk.com/, Accessed January, 2016.

[7] NASA website, URL:http://spaceflight.nasa.gov/realdata/elements/graphs.html Accessed February, 2016.

[8] Mathematics, URL: http://math.stackexchange.com/questions/310306/is-it-fair-to-say-that-keplers-equation-involves-squaring-the-circle. Accessed June, 2016

[9] Bate, R. R., Mueller, D. D. & White, J. E., Fundamentals of astrodynamics, New York Dover, 1971.

[10] Farahmand, M., Orbital propagators for horizon simulation framework, California Polytechnic State University, 2009.

[11] Analytical Graphics Inc, https://www.agi.com/resources/educational-alliance-program/astro-primer/primer102.htm Accessed February, 2016.

[13] Vallado, D. A., Crawford, P., Hujsak, R., & Kelso, T. S.," Revisiting Spacetrack Report #3," AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, Co, 2006.

[14] Vallado, D. A., SGP4 Orbit Determination, American Institute of Aeronautics and Astronautics, 2007.

[15] Astronautics Primer, Analytical Graphics Inc., URL: http://www.insight3d.com/resources/educational-alliance-program/astro-primer/primer106.htm. Accessed March 2016.

[16] Analytical Graphics, Inc. website, URL: http://www.stk.com/. Accessed January 2016.

[17] Analysis and Mitigation of the cubesats dynamics, http://Digital commons.CalPoly.edu. Accessed June, 2016

[18] Umbra, penumbra and antumbra, Wikipedia, The Free Encyclopedia, http://en.wikipedia.org/wiki/Umbra. Accessed April 2016.