MASTER THESIS

Near Earth Objects Space Observatory

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ABSTRACT

In this Master Thesis we begin with an introduction about Near Earth Objects (NEOs). We start with the different kind of existing NEOs, and then we will focus more on which ones can represent the biggest hazard for Earth. Thus many studies suggest irrelevant number of meteorites hit the earth each year, but actually is very hard to know number of exact hit to Earth, but for introducing some meteors are caused by pea-sized of rock, for good estimated number of meteorites per year is necessary to carefully monitoring the meteorites per day in one area and finally extrapolate this data for all area of Earth, or find meteorites fall in to the dry regions and estimate for all area of Earth some valor.

However, is so hard to find exact value because of different size ranges and all procedures have errors, but the estimate value of the mass of material that falls on Earth each year rang from 37000-78000 tons [23]. Most of this mass would come from dust particles. A study done in 1996 calculated that for objects in the 10 grams to 1 kilograms size range 2900-7300 kilograms per year hit Earth, furthermore, between 36 and 166 meteorites larger than 10 grams fall to Earth per million square kilometers per year. Thus that translates to 18000 to 84000 meteorites bigger than 10 grams falls to Earth.

Nowadays different space agencies of several countries have their programs to detect hazardous NEOs, but in case of many of this agencies they need extra help from amateurs astronomers. Furthermore, all of this programs represent different disadvantages such as high cost of operation, no centralized data base and work with people that are amateurs and no depending to any agencies.

New systems will be proposed to detect on time, the hazardous NEOs. These new systems are an answer for the actual issues to detect NEOs on time, and issues of the main official agencies to resolve their problems with this kind of the space objects.

The system where is proposed here is a system based on the constellation of the satellites in the Low Earth Orbit (LEO), equipped with a Newtonian Telescope on board. Furthermore, this system had a ground stations and centralized database, thus that all information about NEOs compiled by satellites can be used for the space agencies to detect on time hazardous NEOs.

The Satellites use low cost components and they are respectable to the environment, the function of the satellites will be determined during this thesis, although the LEO present some conditions, like drag, and depending the mass of the satellites, the orbit can be free after several orbits, when the satellites burn because of contact with drag.

For design and simulation of the system we use required some specific tools like Solidworks for the 3D design and Moon2.0 for the orbital simulation, and finally we propose an alternative system to put our satellites in the orbit, with a system called QuickFast.

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ACRONYMS, ABBREVIATIONS AND DEFINITIONS

AB	Asteroid Belt
AU	Astronomical Unit
BST	Big Surface Telescope
CAD	Computer Aided Design
CCD	Charged Couple Devices
COTS	Commercial Off-The-Shelf
ESA	European Space Agency
IEOs	Inner Earth Objects
ISS	International Space Station
ITU	International Telecommunication Union
KB	Kuiper Belt
LEO	Low Earth Orbit
MOID	Minimum Orbit Intersection Distance
MPPT	Maximum Power Point Tracking
NASA	National Aeronautics and Space Administration
NEAs	Near Earth Asteroids
NEOs	Near Earth Objects
OC	Oort Cloud
OOP	Object Oriented Programming
PHAs	Potentially Hazardous Asteroids
RP	Rapid Prototyping
SD	Scattered Disk
TNO	Trans Neptunian Objects

INTRODUCTION

Near-Earth Objects (NEOs) are comets and asteroids that have been nudged by the gravitational attraction of nearby planets into orbits that allow them to enter the Earth's neighborhood. These objects are composed mostly of water ice with embedded dust particles. Comets originally formed in the cold outer planetary system while most of the rocky asteroids formed in the warmer inner solar system between the orbits of Mars and Jupiter. The scientific interest in comets and asteroids is due largely to their status as the relatively unchanged remnant debris from the solar system formation process some 4.6 billion years ago and their potential dangerously for life on the Earth.

Because of this interest, the main space agencies like NASA or ESA have programs to detect and register these NEOs for example the NASA program cost is ranging between $236 million and $397 million. But it is impossible for NASA or ESA cover 24 hours this observatory. Due to budget and personal limits these agencies need helps from amateur astronomers. Because the number of amateur astronomers is higher than the number of NASA or ESA personal, they can cover and register these functions instead the official space agencies.

In case of ESA, they have started a program to detect NEOs [24] under control of the NEO Coordination Centre (NEO-CC); it is operated by space dynamics services under a contract with Elecnor Deimos, Spain, on behalf of the agency's SSA program office. It serves as the central access point to a network of European NEO data sources and information providers being established under ESA's space situational awareness program [10].

On the other hand NASA has a program to detect and observe NEOs under Near-Earth Object Program; the purpose of them is to coordinate NASA-sponsored efforts to detect, track and characterize potentially hazardous asteroids and comets that could approach the Earth. With over 90% of the near-Earth objects larger than one kilometer already discovered, the NEO program is now focusing on finding 90% of the NEO population larger than 140 meters [9] before 2020.

But this reliance represents some problems for the official space agency, since amateur astronomers work as a hobby; it is impossible to represent a professional and regular service.

To avoid this reliance, we propose to make a new system to detect NEOs based on a satellite. In order to improve the NEOs observatory to the official and professional services, we need a constant and semiautomatic system working 24 hours.

In addition we are looking to reduce the cost of the program. We will propose low cost access to Space, using components and systems existing for the domestic market and COTS. The low cost part is very important for the implementation of this observatory system. Because of the requirements of the system, low cost access can help to make a faster and more feasible satellite. The objective of the low cost access is to avoid problems with budgets and financing for the production of the system. We know that space systems and subsystems nowadays tend to be very expensive and only few countries can finance these programs.

Finally in this final thesis master we will speak about mission design, drivers, requirements, system design, detailed subsystem design and independent space access and finally the conclusions.

In chapter one in the mission design we present some definitions and the objectives of this work, focusing on spacecraft orbital payload parameters. Furthermore we explain launch parameters used and preliminary system design and mission failures analysis and specific tools.

Chapter two is dedicated to explain the drivers and software tools. Moreover, we speak about mission requirements: what is a system requirement and the different high and low-level requirements.

Chapter three speaks about system design. In this part we introduce system design and we observe which one is the relation between high and low level requirements and system design. Moreover in this part we explain about link, power, thermal, and radiation protection, structural and mass budget and finally we explain fault tree analysis.

In chapter four we speak about detailed subsystem design parts, which are very important for satellites. We talk about power supply and communication systems and structures. For the good functionality of the satellite those systems should be able to use position and altitude determination and altitude control. Finally we speak about the optical subsystems, tracking and the onboard computer.

Chapter five is about independent space access. In this part we speak about the newest proposal for low cost space access, based on use of a low cost mini launcher and the space opportunities that come with the use of this method. Inside of this chapter we also introduce satellite operations and telecommand.

Chapter six is about implementation and validation of satellite, including process of 3D printing and manufacturing of the satellite, moreover this chapter include qualification and validation of satellite and issues during this process.

Finally chapter seven is dedicated to the general conclusions of this works, important aspects of this thesis and improvements of this work for the future. Moreover in this work we designed the satellite in 3D solid works and all detailed parts are in the annex. The idea is that they help to create new designs from our final proposal system.

Chapter 1

MISSION DESIGN

1.1. Mission definition

A long time ago scientist discussed what happened to the Jurassic era and the dinosaurs. Many ideas were proposed but one of them is getting better acceptance among the scientists community, and that is that dinosaurs disappeared because a very big meteorite impacted Earth, and due to this impact almost all of Earth's life extinguished.

This theory shows that an event of these characteristics is possible in the present or future, and that it would cause a massive extinction. Recent researches have shown that the impact of an asteroid caused the last Ice Age about 13,000 years ago [8].

In regard to this event and others, detection of meteorites that can impact Earth is essential. Not all meteorites are dangerous for continuity of life; they could be potentially dangerous for big cities and villages. It is mandatory to detect them on time and put the administrations in alert. Many hours and studies are invested to detect on time earthquakes, the detection of meteorites heading toward Earth is very similar, and deserves to have an observatory and register system.

The main objective of this thesis is to provide new systems based on a satellite for Space observatory with the main objective of detecting Near Earth Objects (NEOs). Furthermore this new system must be capable to detect, one day in advance, NEOs and alert to ground-based optical telescope for better observation. These are powerful and sensitive tools. But the have their limitations. Because ground-based optical telescope must operate at night, they are blind to objects that approach from regions of the sky near the sun. They are also limited by the need for good viewing conditions-a moonless and fairly turbulence-free sky, which might be on offer only a quarter of time.

Nowadays there exist some systems to detect NEOs, based in the observations of amateur astronomers around the world, without a professional network, collaborating with international administration like NASA or ESA or Sentinel program. This network provides to official agencies necessary data for their observatory sessions; the hazard of this network is that there are no professionals making observations, but rather hobbyist in their free time. In addition they cannot provide all time/all days observation.

"*NASA estimates that it has identified only about 10 per cent of all asteroids 140 meters and larger*," Martin wrote. *"Given its current pace and resources, (NASA) has stated that it will not meet the goal of identifying 90 per cent of such objects by 2020."* [7] Meanwhile this news represents a big issue for the existing systems, as there is no unified database with shared standards.

To make a better network that can observe space all time/all days, uses of satellites in Low Earth Orbit (LEO) are a good tool to use. In LEO it is possible to reduce effects of the atmosphere in the observations of space, and it can provide better resolution and image quality.

Is necessary especial mention about effects of atmosphere to space observatory such as direction and intensity of light when it through layers of atmosphere. In case of direction atmosphere affect in refraction and seeing. Moreover in case of intensity it affect to extinction and scintillation. But the most serious obstacle is water vapor in the atmosphere, which absorbs infrared light before it can reach the ground. The infrared part of the spectrum happens to be where NEOs are easiest to see. Asteroids may be dark, but they are warmed by sunlight and absorb and re-emit that light at infrared wavelength.

Focusing in the integrity of the network, all data collection most is sent to the ground station to analyze and to make a big database from all satellite data collections. This big database can help ground-based optical telescope to use it very fast and easily. The proposed constellation of LEO satellites should provide a very wide cover zone, extending 360º degrees.

All of the NEO discovery teams currently use so-called charged couple devices (CCDs) rather than photographic images. These CCD cameras are similar in design to those used in cell phones and they record images digitally in many electronic picture elements (pixels). The length and width of a CCD detector is usually given in terms of pixel elements. A fairly common astronomical CCD detector might have dimensions of about 2000 x 2000 pixels. While the CCD technology allows today, detectors to be more sensitive and accurate than the older photographic images, the modern discovery technique itself is rather similar.

Separated by several minutes, three or more CCD images are taken of the same region of the sky. These images are then compared to see if any NEOs have systematically moved to different positions from one image to the next. For a newly discovered NEO, the separation of the object's location from one image to the next, the direction it appears to be traveling and its brightness are helpful in identifying how close the object was to Earth, its size and general orbital characteristics. For example, an object that appears to be moving very rapidly from one image to the next is almost certainly very close to the Earth. Sophisticated computer-aided analyses of the CCD images have replaced the older, manual detection techniques but often times, a new NEO discovery is still verified using the human eye [9].

1.2. Mission objectives

The asteroids belts in the solar system represent unknown zones that a few years ago started to present an interest for scientists and physics. Currently in the solar system there exist different type of asteroids belts, the first belt from is the Asteroids Belts between the orbits of Mars and Jupiter. This asteroid belts was discovered using Titius-Bode law [1], according to this law if a numerical sequence began at 0, then included 3, 6, 12, 24, 48, etc., doubling each time and added 4 to each number and divided by 10, it produced a close approximation to the radii of the orbits of the known planets in solar system in astronomical units.

Titius-bode law shows to astronomers that between Mars and Jupiter should be another planet or space object. And it is real, as during XVIII and XIX century many astronomers find different objects to define this missed planet as Ceres [2]. But because of the size of all this objects and their situation they cannot be considered as a real planet. Nowadays we know this missed planet was in formation process but because of different reason it wasn't finished. Today between Mars and Jupiter exists an asteroid belt that is composed by objects of many different sizes and kind of asteroids orbiting around the Sun.

This *Asteroid Belt* (AB) it is the first hazard zone for the Earth. The state of the asteroids in AB represent very dangerous situation, as they move without any order and impact between them. Because of all this unpredictable situations, The Asteroid Belt can represent hazard to Earth, as some of the asteroid can move our direction impact.

On the other hand in the solar system there exist others hazardous zones packed with comets, asteroids and other objects. These zones are generally included inside the term *Trans-Neptunian object* (TNO). Insomuch these objects orbit the sun at greater average distance than Neptune orbit. Inside of TNO we can find the *Kuiper Belt* and *Scattered Disk and Centaurs* and finally the *Oort cloud* (OC).

The *Kuiper Belt* (KB) shown in Figure 1.1 is a new belt whose existence was proposed in relation to the discovery of Pluto in 1930; many speculated that the Pluto might not be alone. Until 1992 it was impossible to find evidence of its existence, but finally by the end of the eighties researchers from MIT got the evidence of the existence of the *Kuiper Belt*. This new asteroid belt is a region of the solar system beyond the known planets approximately 50 *Astronomical Unit* (AU) [3] from the Sun, the KB consist mainly of small bodies and remnants from the solar system's formation.

In case of the *Scattered Disk* (SD) it is a distant region of the solar system composed by icy minor planets, comets, asteroids, and a subset of the broader family of TNO. The SD zone begins, depending of different objects, from 30-35 AU, coinciding with the border limits of the KB, and up to 100 AU. This makes these scattered objects among the most distant and coldest objects in the Solar System [4]. Because of its unstable nature, astronomers now consider the scattered disc to be the place of origin for most periodic comets in the solar system, together with the centaurs, a population of icy bodies between Jupiter and Neptune, being the intermediate stage in an object's migration from the outter disc to the inner solar system [5].

On other side the *Oort cloud* is a spherical cloud of predominantly icy planetesimals believed to surround the sun at up to 50000 AU [6]. The outer limit of the *Oort cloud* defines the cosmographical boundary of the solar system and the end region of the Sun's gravitational dominances. In the following figure we can observe the composition of the *Kuiper Belt*, *scattered Disk* and *Oort cloud*.

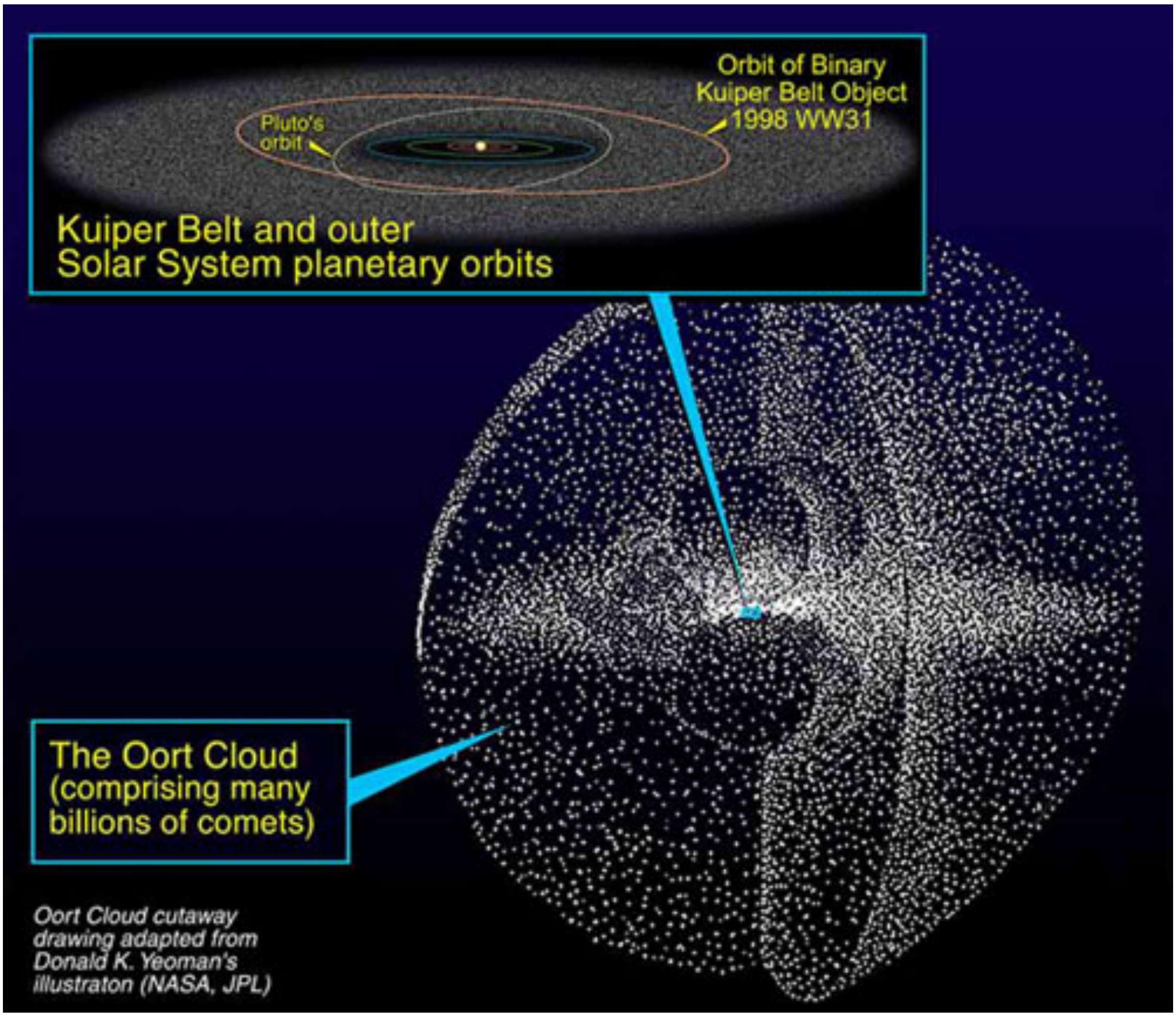


Figure 1.1 Kuiper belt and the Oort cloud

After localizing the regions that can be origin of the NEOs, information about their orbits becomes essential information to detect on time these objects. Not all asteroids or comets are hazardous for Earth though, and through their orbits we can differentiate between hazardous and non-hazardous comets or asteroids.

In terms of orbital elements, NEOs are asteroids or comets with perihelion distance (q) less than 1.3 AU. Near-Earth Comets (NECs) are further restricted to include only short-period comets. The vast majority of NEOs are asteroids, referred to as *Near-Earth Asteroids* (NEAs). NEAs are divided into groups like *Aten, Apollo, Amor*. See some examples in Figure 1.2 from NASA[1], According to their perihelion distance (q), aphelion distance (Q) and their semi-major axes (a), more details we can see in Table 1.1.

[1] Source: http://www.nasa.org

Table 1.1 Types of comets and asteroids that are in the interest in my thesis

Group	Description	Definition
NECs	*Near-Earth Comets*	*q<1.3 AU, P<200 years*
NEAs	*Near-Earth Asteroids*	*q<1.3 AU*
Atiras	*NEAs whose orbits are contained entirely within the Earth's orbit (named after asteroid 163693 Atira).*	*a<1.0 AU, Q<0.983 AU*
Atens	*Earth-crossing NEAs with semi-major axes smaller than Earth's (named after asteroid 2062 Aten).*	*a<1.0 AU, Q>0.983 AU*
Apollos	*Earth-crossing NEAs with semi-major axes larger than Earth's (named after asteroid 1862 Apollo).*	*a>1.0 AU, q<1.0167 AU*
Amors	*Earth-approaching NEAs with orbits exterior to Earth's but interior to Mars' (named after asteroid 1221 Amor).*	*a>1.0 AU, 1.02<q<1.3 AU*
PHAs	*Potentially Hazardous Asteroids: NEAs whose Minimum Orbit Intersection Distance (MOID) with the Earth is 0.05 AU or less and whose absolute magnitude (H) is 22.0 or brighter.*	*MOID<=0.05 AU, H<=22.0*

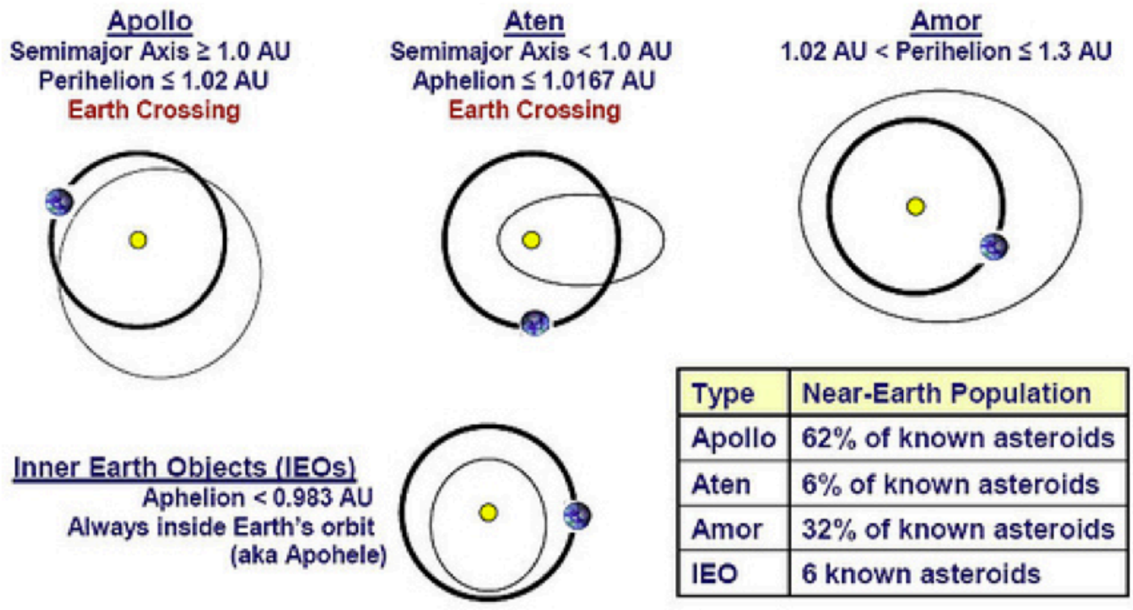


Figure 1.2 Near-Earth Asteroids (NEAs): Apollo, Aten and Amor. Inner Earth Objects[2] (IEOs)

Our proposed system should be able to detect all NEOs inside of the Potentially Hazardous Asteroids Group. Potentially Hazardous Asteroids (PHAs) are currently defined based on parameters that measure the asteroid's potential to make threateningly close approaches to Earth. Specifically, all asteroids within an Earth Minimum Orbit Intersection Distance (MOID) of 0.05 AU or less and an absolute magnitude (H) of 22.0 or less are considered PHAs. In other words, asteroids that can't get any closer to the Earth than 0.05 AU (roughly 7,480,000 km or 4,650,000 mi) or are smaller than about 150 m (500 ft) in diameter are not considered PHAs

[2] Source: http://www.nasa.org

Furthermore, detection systems need to make early warnings for the users, with the minimum delay possible and low or nonexistent shadows in the continuous observation. Mainly the European Space Agency (ESA) is going to be the final user of this system, as they have already a good network of ground stations in place. In addition ESA has invested on the last decades on how it can detect on time hazardous objects. This project attempts to help them achieve those objectives.

It is important to put emphasis that this system uses Low Earth (LEO) Orbits and that the payload used in the system after their short life, is reentered into the atmosphere, making the orbit free again, this system is very environmentally friendly and it causes a very low, or even nonexistent, environmental impact.

1.3. Spacecraft orbital payload parameters

The profile of the mission relies on the use of LEO as these orbits are very usable for space observation. When ground-based optical telescope on Earth’s surface want to search for space objects, they need to avoid atmospheric optical phenomenon. The atmospheric phenomena are often caused by the iteration of sunlight with atmosphere and particles like as drop waters, clouds or dust. For this reason we propose the use of satellites in LEO, as it is a very good solution for immediate or short time answer to NEO’s detection systems.

Furthermore, the use a satellite system in LEO’s is necessary, as we need to cover 100 per cent of spherical space around Earth, which is only possible using a constellation of satellites. These constellations should be able to provide all time cover around the Earth space, and send the data to the ground stations for its analysis and to provide alarms on time in case of emergency.

Thus to cover all sphere area of the Earth with telescopes on satellites orbiting, we propose configurations that get higher or lower coverage with higher or lower costs. Since the satellites are in low orbits, Earth will block big sections of the field of view. Because of this problem it is mandatory to put several satellites in several different orbital planes to achieve full, 360 degrees cover. The higher number of satellites per orbital plane is translating to the higher overall cost of the system. On the other fewer satellites can be put in each orbital plan, compromising then the response time as will be discussed in 1.6, where we will determined the limits of the system.

The minimum-working configuration of constellation is in a tetrahedron form, composed by four satellites where Earth is in the geometric center of tetrahedron. If we imagine a methane molecule CH_4, figure 1.3, the carbon atom represent the Earth and the other four hydrogen atoms represent the satellites. In all time satellites are looking towards space, with their backs towards Earth. When the angle between each satellite is 109.5º the four satellites are equidistant to each other.

To make this kind of configuration two orbital planes are needed, as a consequence we need to launch at least two times. In addition each launch should carry two satellites. Finally the cost of this constellation is low but it has very little redundancy if there is one satellite failure, it is impossible to guarantee visual coverage.

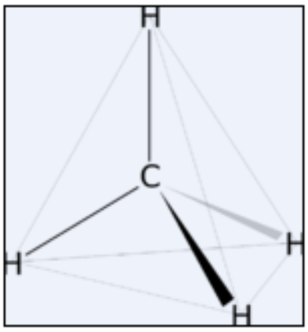


Figure 1.3 Tetrahedron satellite configuration. Methane model molecule.

Another configuration is the octahedron form, seen in figure 1.4is it also possible to achieve using two orbital plans; one of them at least should have four satellites and therefore other orbital planes must be composed by only two satellites. When the angle between each satellite is 90º the six satellites are equidistant to each other. In this configuration the opening angle of the camera cell sensor can be smaller, furthermore this design present high redundancy. Therefore at least two launching processes are necessary, where in each one of them the launcher carries up four satellites for each of the orbital plans.

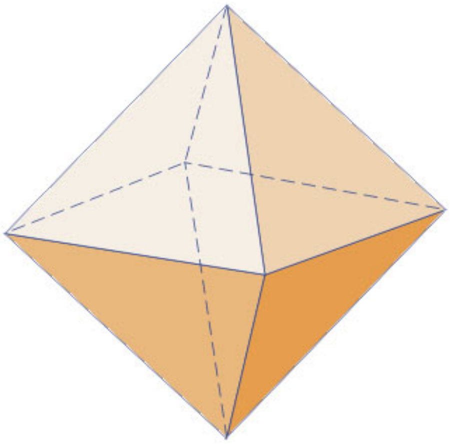


Figure 1.4 Octahedron satellite constellation

Table 1.2, is shows a summary of characteristics parameters for each constellation configuration, comparing between advantages and disadvantages.

Table 1.2 Summary of orbital strategies

	Tetrahedron	**Octahedron**
Number of plans	*2 (2 sat. each plan)*	*2 (4 each plan)*
Minimums satellites	*4*	*6*
Number of launches	*2*	*2*
Reserve satellites	*0*	*2*
Cost *Shuttle 50k€* *Satellite 10k€*	*2*50,000 €* *4*10,000 €* *140,000 €*	*2*50,000 €* *8*10,000 €* *180,000 €*
Advantages	*- Low cost constellation* *- Simple constellation*	*- It has redundancy in one orbital plane*
Disadvantages	*- Low overlap between images* *- It has not redundancy*	*- Expensive constellation*

1.4. Launch parameters

One of the main causes for the high cost of a launcher is its development cost. We can reduce this cost if the subsystem designs are reused, or if we use some of the satellite subsystems as part of the avionics of the rocket. For example, the IMU of the satellite can perform exactly the same task for the rocket, thus reducing its mass and complexity. In addition, the use of new technologies that are available in the market allows us to reduce the payload mass too.

As the total mass of the launcher is a function of the payload mass, a reduction in size means a reduction in the launch cost as well. The failure rate of rocket launchers have a clear impact on the cost of launch assurances; if the launch had a very low price, it would be possible to take higher risks even to repeat a launch many times, allowing a cycle of trial and error that would conduce to a more reliable system.

In consequence we can outline two types of the strategies and costs to launch satellites. First there is the possibility to do few launches with larger satellites and second we can do many launches with smaller satellites. Each one has advantages and disadvantages summarized in table 1.3 in such a way that they can be compared. A larger number of launches is a cheaper option because of the reduced cost of the satellites and launchers.

Table 1.3 Few and many launches

	Few Launch **Great Satellite**	**Many Launch** **Small satellite**
Advantages	• *Durability* • *Long life time* • *Capability to integrate more applications* • *Use of lower orbit altitudes* • *Capability to integrate bigger telescope / higher resolution* • *Use high frequency for communication* • *No big ground station network necessary* • *Low units of the satellites*	• *Small payloads* • *Low cost* • *Redundancy* • *Multi point of view*
Disadvantages	• *Greater payload* • *Expensive satellite* • *In case to failure long time to leave free orbit*	• *Weakness against radiation* • *Limited use of telecomm.*
Estimated cost	• *From 8 to 80 Million $*	• *From 50 to 500 k$*

1.5. Preliminary system design

Technological drivers are principles or laws which make feasible to accomplish a system need.

High power density batteries: It is possible to store a high amount of electrical power in a small volume and able to support high forces. An example of this is a Coin battery.

High integration level: It is possible to integrate a large number of transistors and components in a solid state. An example of this is the growing marked of SMD embedded components. Nevertheless, higher integration means higher sensitivity to radiation effects and shielding improvements are required.

Photovoltaic cell: It is possible to extract the energy from the sunlight and convert it in an efficient way into electrical power with a very low weight. Example: UTJ[3] solar cell with up to 28% of efficiency.

Photodiode sensor: Concentrating the light, it is possible to have a high quality image in a very integrated solid-state component. An example of this is a high definition HD camera.

[3] UTJ: Ultra Triple Junction http://www.spectrolab.com/DataSheets/cells/PV%20UTJ%20Cell%205-20-10.pdf

Accelerometer: It is possible to sense the motion of an object in a very integrated solid state and using a very light device. A good example of this is a solid accelerometer inside an IMU, also known as Inertial Measurement Unit.

Gyroscope: It is possible to sense the turn rate of an object in a very integrated solid state and very light device. A good example of this is a solid angular rate sensor inside an Inertial Measurement Unit.

Synthetic Aperture Radar: It is possible to improve the range of a link concentrating and changing its direction without any mobile part. An example of this is a nano-SAR based on the synthetic aperture of the radiation lobe

1.6. Mission failure analysis

For failure analysis we consider few topics like Airspace, satellite, space debris and asteroids involved in the mission. In order to guarantee our mission, simulation in the *Moon2.0* is performed. Our constellation of the satellites cannot disturb others satellites, because of this we check possible conflicts with near satellites in the simulator. And finally we have to assure the integrity of our satellites; because of these conditions we have to test the risk of collision with space debris or other space objects.

Nowadays there exist different methods for failure analysis. We focused on two methods used by NASA and ESA, These are:

Most space debris detection processes are iterative and involve several risks criteria [11]. These risks depend on the quality of positional data available, but since we have no data, we cannot determine the risk. The ISS debris avoidance process is initialized by the USSTRATCOM screening on the entire catalog over a time window. In this case “all conjunction” within a $\pm 2 \times \pm 40 \times \pm 40$ km box, collision probability is assessed. On the other hands if exist “falls conjunction” within a $\pm 0.75 \times \pm 25 \times \pm 25$ km is necessary to considerer avoidance maneuvers. In addition ESA approach combines a $\pm 10 \times \pm 25 \times \pm 10$ km exclusion ellipsoid according to collision probability assessment.

When designing the criteria for the failure assessment two competing quantities have to be balanced. First the alarm frequency, and second a measure of the residual collision risk. Finally an efficient and effective risk criterion has to consider for a good trade-off between alarm frequency and collision risk reduction.

On the other hand there are risk assessment software tools; these are installed onboard the satellites. There are two NASA methods to compute the reentry survivability of spacecraft components. The Debris Assessment Software (DAS); Object Reentry Survival Analysis Tool (ORSAT). There are some differences between two methods, DAS is a conservative and easy to use tool, but ORSAT is more accurate, higher fidelity model requiring and training. Due to different concepts, NASA updates periodically the DAS tools, with new information of the space debris in Earth orbit.

The ORSAT tool is the primary NASA computer code for predicting the reentry survivability of satellite; this prediction is required in order to determine the risks. In this tool has predicted total debris casualty area, orbit inclination and time of reentry, showing that the risk should be less than 1:10,000.

Finally for detecting the NEOs in space by telescope satellites, position of the satellite and its pointing to the space is of the most importance for the project. In consequence the used technology for use in this system is an open source program named Elbrus. Elbrus is a free telescope-pointing program that it analyzes a sky image with a few stars, and calculates the coordinates of the image center.

The Elbrus program is also useful when the satellite is in space but we have no information on where it is pointing. It allows us to do a ring search of the space around the satellite to determine its attitude. Another important characteristic abut Elbrus, it is can also be used for refocusing the scope in between captures. Each image is analyzed and the movement of the references is computed. The offset is used by the mount-driving program to refocus telescope, so that we can capture images during long time in despite of the drift in the telescope tracking.

The analyzed area is 10x10 to 20x30 arc min, FITS and BMP image types are saved, the search process is about in a 5x5 and 30x30 degrees sky window, data base has 800 MB size and it is possible to communicate with other programs.

Here we need to introduce how the ELBRUS works, the process has different steps: the star extraction, searching in the database, building the polygons and finally the calculus of the image coordinates.

The ELBRUS start with a first image file containing stars from where satellite starts to work. The system reads pixels of the image and defines where the stars are based on pixel intensity. In Figures 1.5 and 1.6 we can see this proportion between dark and bright pixels.

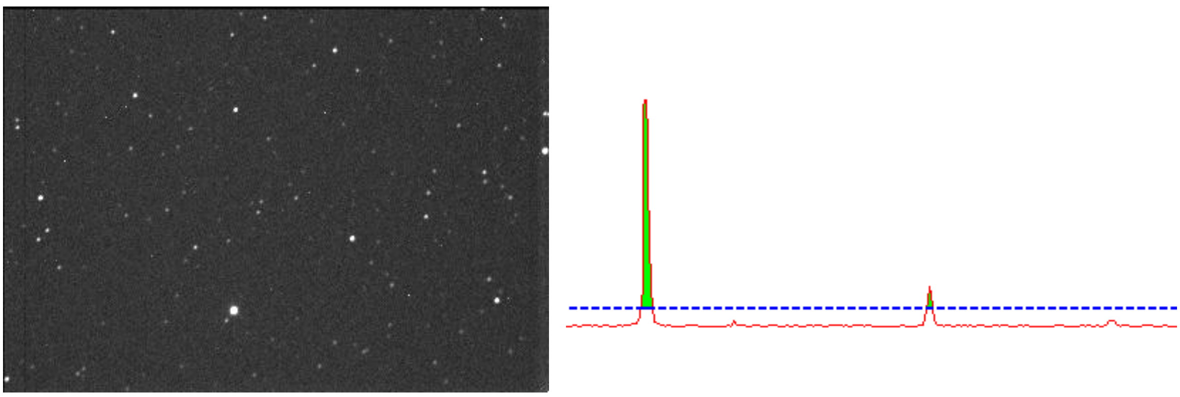


Figure 1.5 Pixel intensity from the optical sensor

The algorithms of ELBRUS use several thresholds until it finds an adequate number of stars to use. The program selects 10 strongest of them and numbered from 1 to 10 (“sample numbering”).

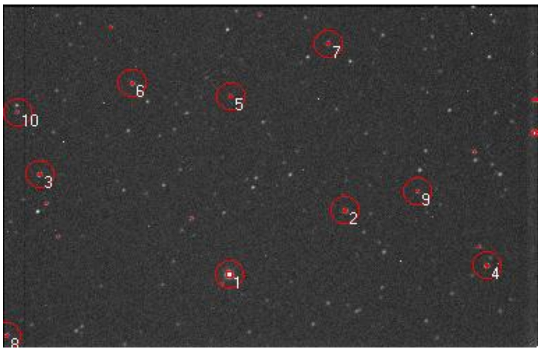


Figure 1.6 Reference points assigned by the ELBRUS program

Consequently ELBRUS is capable to calculate the distances between the 10 selected stars and compare their distances with the ones stored in the database.

Database has a large number of distances and comparing the obtained distances D1, D2…. Etc. It is possible to make all possible “V’s” (“V” is two distance from a common star captured in image) finally from “V” ELBRUs make polygons and missing sides of these polygons, so that it is capable to compare the new polygon with polygons from database, and if it coincide, all is ok, however if not, ELBRUs discard the polygon.

So, with ELBRUS it is possible to find the center of image and the direction of the telescope. From captured information and simulation the NEOs detection system is capable to find different space objects and compare them with a database and make new database with new and near Earth objects. That is the objective of this thesis.

1.7. Specific tools

For development of Near Earth Objects Space Observatory (NEOSO) the use of some specific tools is mandatory, for the simulation of the orbits, payloads, etc. The proposed tools is *Moon2.0*, this software is an open source toolkit that allows designing the full engineering-cycle of low cost of no-manned space missions. The following tools are also required for the mission design:

- Trajectory propagator
- Collision detection and satellite/debris database
- Atmospheric and Ionospheric model
- Ground station database and signal propagation model
- Image, optics and camera processing software like
- For 3D modeling a CAD software like Solid Works
- For data processing and statistics we can use Microsoft EXCEL
- For electronics design and simulator can be used a specific electronic tool like NI LabView and EaglePCB.
- Critical source tool for the satellite operation programming like Microsoft C and C++

- Subsystem development requiring hardware-in-the-loop can be developed by FPGA boards like XLING FPGA simulator

When satellites are orbiting, other tools will be required. For the download of scientist data, a telecommunications network is required like GENSO. For satellite telecommanding and control, a control center is required like Moon 2.0. Every launch requires a number of tools that will be owned by the launcher operator. These tools are out of this Master Thesis scope.

Chapter 2

TECHNICAL SOLUTION ARCHITECTURE

The architecture of the technical solution is divided in Drivers and requirements. Drivers are understood as principles or methods that permit the satellite functionality. These drivers are divided into Physical drivers, Technological drivers and Software drivers. Requirements are parameters or conditions to be fulfilled by the final satellite design and are divided into System Requirements, High Level requirements and Low Level requirements.

2.1. Physical drivers

The main physical drivers depicted in Figure 2.1 that are needed in this mission are pin-hole, reflection and photovoltaic drivers:

- Use of cameras based on the pin-hole phenomenon
- Use of magnification based on light reflection
- Use of the photovoltaic phenomenon

Figure 2.1 Physical drivers

2.2. Technological drivers

The technological drivers are technologies that allow us implement and fulfill the requirements and they are shown in Figure 2.2.

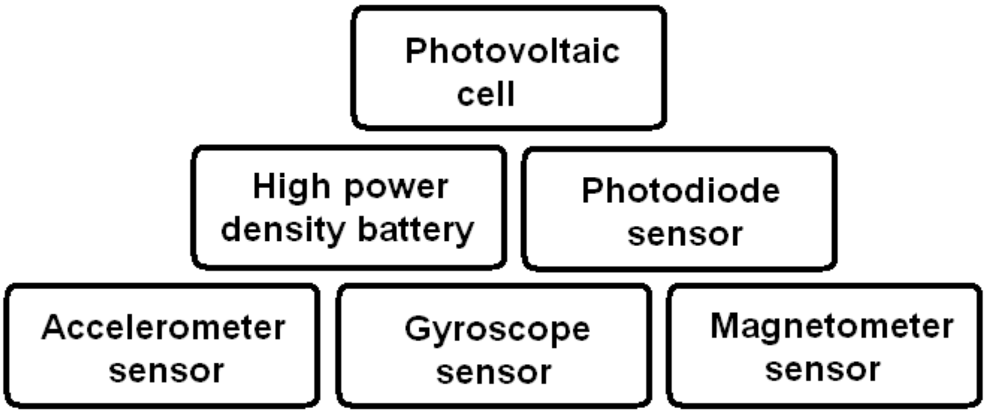


Figure 2.2 Technological drivers

These technological drivers are: photovoltaic cell, high power density battery, photodiode sensor, accelerometer, gyroscope and magnetometer.

- **Photovoltaic cell.** It is possible to extract the energy from the sunlight and convert it in an efficient way into electrical power with a very low weight. Example: UTJ solar cell with up to 28% of efficiency.
- **High power density battery.** It is possible to store a high amount of electrical power in a small volume and able to support high forces. An example of this is a Coin battery.
- **Photodiode sensor.** Concentrating the light, it is possible to have a high quality image in a very integrated solid-state component. An example of this is a high definition HD camera.
- **Accelerometer sensor.** It is possible to sense the motion of an object in a very integrated solid state and very light device. A good example of this is a solid accelerometer inside an IMU, also known as Inertial Measurement Unit.
- **Gyroscope sensor.** It is possible to sense the turn rate of an object in a very integrated solid state and very light device. A good example of this is a solid angular rate sensor inside an Inertial Measurement Unit.
- **Magnetometer sensor.** The measurement of the magnetic deviation of a coil embedded in a MEMS device also incorporated in some Inertial Measurement Unit.

2.3. Software drivers

The software drivers are depicted in Figure 2.3.

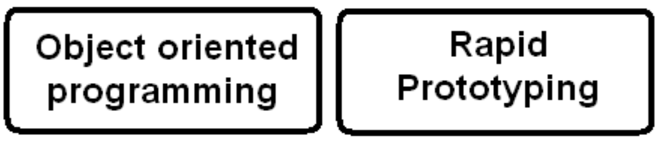


Figure 2.3 Software drivers

The software drivers are: Object oriented programming and Rapid prototyping.

- Object oriented programming (OOP). Historically, programming was oriented towards a logical procedure: The program obtains an input, it processes it and then it generates an output. OOP are programming languages based on data rather than logic and organized around objects rather than "statements".
- Rapid prototyping (RP). This software driver allows a quick fabrication of physical objects and devices using 3D computer aided design (CAD) data. These drivers are widely used in a large range of industries. The Rapid prototyping will allow us to convert our innovative satellite idea into a successful end product, quickly and efficiently.

2.4. Requirements

The list of requirements is presented in this chapter. The requirements are divided into groups from System Requirements (SR) there are a number of High Level (HL) requirements to be implemented. Inside each High Level requirement are a number of Low Level (LL) requirements that contains larger detail. This classification starts from the bigger part to the smallest part setting the atomization into a tree as shown in Figure 3.1.

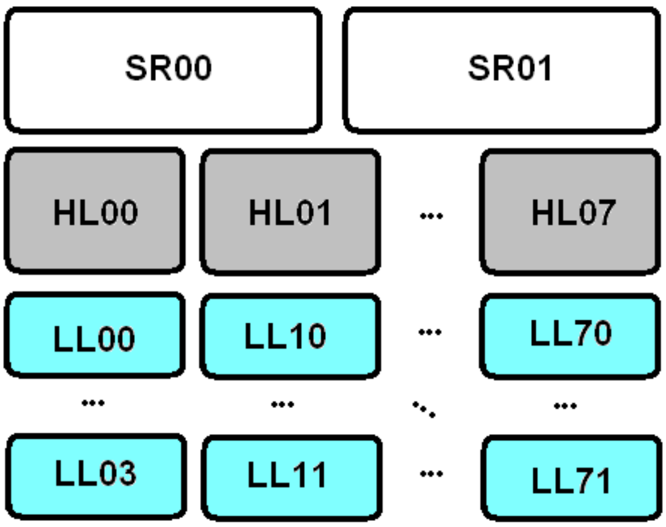


Figure 2.4 Requirements tree. System, High level and low level requirements

2.5. System requirements

Principal requirement of the NEOs Observatory is to work in redundancy, as the data collected is used to detect on time the hazardous NEOs. The environmental respect of this system is also important, especially to avoiding space debris.

Our satellite will use LEO orbit. This orbit is not desirable because of the presence of atmospheric drag, which slows down and damages the satellite, reducing the life san of the mission. However, this same drag creates a relatively free area of other satellites. Also it can be used to destroy the satellite when the mission is complete, freeing the orbit again, our requirements are:

SR00: The system should detect Near Earth Objects on time in order to identify the hazardous objects.

SR01: The system should not disturb others in case of a total failure.

2.6. High level requirements

HL00: The power supply subsystem should provide electrical power for the computing of the orbit and the tracking.

HL01: The communication subsystem should transmit and receive information

HL02: The structure subsystem should protect the satellite components and be used as a thermal path for thermal loads

HL03: The attitude determination subsystem should determine the attitude by inertial means and be helped by optic sensors

HL04: The position determination subsystem should determine the position in the orbit by inertial means and be helped by optic sensors

HL05: The attitude control subsystem should point the high gain antenna to the Earth in a passive way using the Earth’s magnetic field

HL06: The tracking subsystem should transmit its computed position to a ground station when passing over it

HL07: The telescope subsystem should be capable to make images between 0 to 90 degrees with respect to the Earth’s horizon

HL08: The telescope subsystem should be capable to work between -20º to 60ºC temperature

2.7. Low level requirement

LL00: The power supply subsystem shall charge the battery from the coming Sunlight
LL01: The power supply subsystem shall provide power during the Eclipse (Earth blocking the light from the Sun)
LL02: The power supply subsystem shall be protected against a short-circuit

LL03: The power supply subsystem shall resist Sun radiation during the satellite lifespan

LL10: The communication subsystem shall transmit the dangerous Near Earth Objects trajectories inside the link window
LL11: The communication subsystem shall receive new telecommand parameters from ground station

LL20: The structure subsystem shall support the satellite components
LL21: The structure subsystem shall absorb vibrations from the launching phase
LL22: The structure subsystem shall maintain in a passive way the electrical component temperature range by a correct surface emissivity design
LL23: The structure subsystem shall keep the dilatation below a limit in order to reduce the lens aberration

LL30: The attitude determination subsystem shall determine the large range attitude by inertial means
LL31: The attitude determination subsystem shall determine the accurate attitude by star tracker means
LL32: The attitude determination subsystem shall determine the magnetic field variations for each moment inside the orbit

LL40: The position determination subsystem shall determine the large scale position inside the orbit by the Kepplerian elements and an accurate clock
LL41: The position determination subsystem shall determine the accurate position in the orbit by a Global Positioning System device

LL50: The attitude control subsystem shall point telescope to the sky area to be scanned
LL51: The attitude control subsystem shall guarantee a minimum accurate maneuver to damp the oscillation

LL60: The tracking subsystem shall generate the computed position from the position determination subsystem
LL61: The tracking subsystem shall send this tracking information to the ground station to provide feedback to the operation center when they have to decide the new telecommand parameters

LL70: The telescope subsystem shall determine the hazardous Near Earth Objects by onboard processing
LL71: The telescope subsystem shall detect false alarm events from those objects that are not hazardous Near Earth Objects

Chapter 3

SYSTEM DESIGN

Chapter three focus on the system design, like definition of the systems. Then it will also include link, power and thermal budget and especial attention to the radiation budget. In addition study of the structural and mass budget is included in this chapter from preliminary studies about how it will work for the satellite. Finally the analysis of the system fault and fault tree is showed.

3.1. System definition

In system definition we speak about which technologies will be used in the process of the satellite evaluation and fabrication. One of these technologies is Micro-Electro Mechanical Systems (MEMS). MEMS are made up of components between 1 to 100 micrometers in size, and MEMS devices generally range in size from 20 micrometers to 1 millimeter. They usually consist of a central unit that processes data and several components that interact with the surroundings such as micro sensors; the Figure 3.1 shows some structures of MEMS.

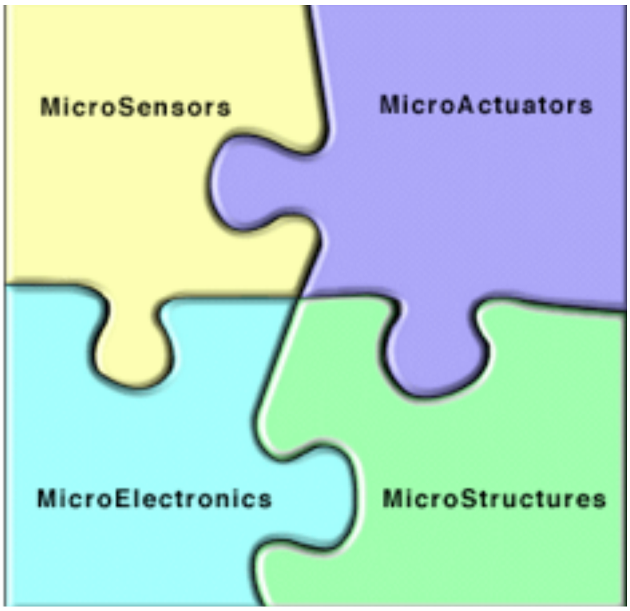


Figure 3.1 Micro-electro mechanical Systems structure

Other technology is Commercial off the Shelf (COTS), are usually components that are commercially available to the general public and which require no special modification or maintenance over its life cycle. COTS were used for software [13] but we are extending the concept to the hardware, they have a well-defined architecture, for example MEMS or SMD devices are COTS.

On the other hand there are the Surface mounted Device (SMD) which are electronic devices which comes from the concept of surface mount technology (SMT). This method allows creating electronic circuits where components are installed on a surface of a printed circuit directly. In addition SMD are usually smaller than classical circuits; they may have short pins, flat contacts or terminations on the body of the component.

3.2. Link Budget

The usual method for communications with LOE is called Store and Forward. Some other common strategies are presented in Figure 3.2. The idea behind it is to store all data recorded by the sensor until the satellite is in view of a ground station, when the satellite transmits (or forwards) the information to free up internal storing space.

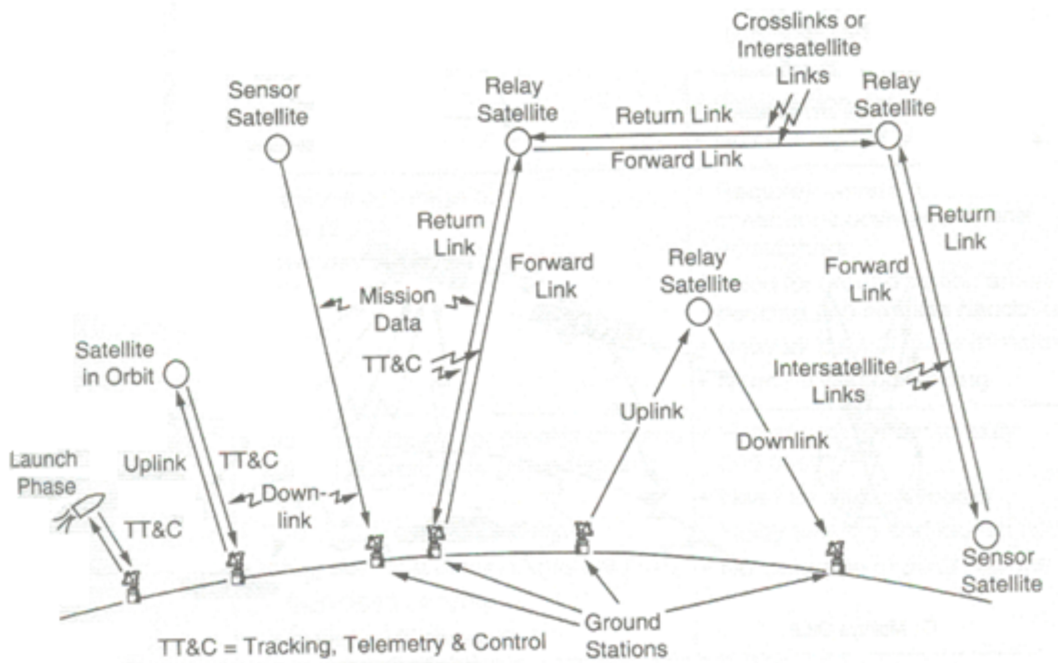


Figure 3.2 Common link strategies [26]

The advantages of this method are that large antennas are not needed, and communications are kept relatively simple. The disadvantage is that high access times are needed (in the order of hours) to download all the necessary data. Also data cannot be accessed in real time.

LEO and crosslink has few traffic jams and needs active control with multiple paths and some complexity as that the receiving station must identify packets coming from different downlinks.

In general, criteria for the design link budget are based on orbit characteristic like visibility, number of contacts per day, duration of the contacts, distance (emission power) and satellite attitude (data acquisition). In addition for the RF spectrums is necessary to determine: channel frequency, assigned bandwidth, antenna size, consumed power, emission permissions (ITU), maximum data rate and modulation/codification. Moreover a criteria for the link budget design are characteristics of the Data Rate, Duty Factor, link availability, access time and threats like solar storms, explosions, others.

For the transmission the proposed data rate is based in digital technology. Because of digital signals can be easily regenerated on ground, the noise dose note accumulate and it has smaller error rates, it can be multiplexed in a single signal. Generally it has less distortions and interferences and less transmission power is required.

For the design of the link budget we need to know the number and precision of the functions to be controlled. In case of the status of the satellite, we need to know the voltages, temperature, attitude data and the workload of the system. Design of the sampling rate and commands are necessary to make a simple link budget for the satellites. These options are executed following the sequence: transmission, storage, verification, execution and tracking.

For the link budget design it is necessary to define a formula that combines all aspects of satellite to ground station communications aforementioned. In this aspect it is necessary to define frequency selection, modulation and codification, bean size (antenna size, etc.) estimated losses, noise level, gain and power estimated. These aspects are all included in the formula 3.1, where it is possible to define a relationship between signal to noise density.

$$\frac{E_b}{N_0} = P + L_l + G_t + L_s + L_a + G_r + 228.6 - 10\log T_s - 10 \cdot \log R \tag{3.1}$$

$\frac{E_b}{N_0}$: Signal to noise density.
P : Emitted power
L_l : Fraction of energy lost in the transponder
G_t : Gain of the antenna
L_s : Spatial losses
L_a : Atmospheric losses
G_r : Gain of the receiving antenna
T_s : Noise temperature of the system
R : Data rate

Recommendation for the few transmission errors in signal to noise ratio, is about $\frac{E_b}{N_0}$ = 5 to 10 dB from knowing satellite orbit is possible to define S/N ratio defining the values of the variables P, G_t, G_r, R. It is very important to keep in mind the effects of meteorological conditions in satellite-ground communication, as that phenomenon can produce heavy absorption for frequencies larger than 10 GHz [26].

For transmitter antenna, the power flux density W_f (W/m^2), is calculate from formula 3.2:

$$W_f = \frac{PL_l}{4\pi S^2} G_t L_a \qquad \textbf{(3.2)}$$

S: Sphere radius
PL_l: Isotropic power
G_t: Antenna gain
L_a: Free space propagation losses

Finally we can redefine power flux density formula like Effective Isotropic Radiated Power (EIRP)

$$W_f = \frac{(EIRP)}{4\pi S^2} L_a \qquad \textbf{(3.3)}$$

For the received power (RP) the formula $C = W_f A_r$ be used where A_r is the effective aperture of the receiving antenna, and we can redefine RP formula like point 3.5.

$$A_r = \eta \frac{\pi D^2}{4} \qquad \textbf{(3.4)}$$

η : Efficiency
$\frac{\pi D^2}{4}$: Physical Area

$$C = \frac{PL_l G_t L_a D_r^2 \eta}{16 S^2} \qquad \textbf{(3.5)}$$

The formula of the gain is set in Equation 3.6:

$$G_{add} = P_{tx} + G_{tx} - e_{tx} - LFS - PLF + G_{rx} - e_{rx} - P_{\min} = 3\,dB \qquad \textbf{(3.6)}$$

Figure 3.3 shows an example of how the directivity of the transmitter beam should be pointed towards the receiver. Maintaining the EIRP, less power is required by increasing the directivity or reducing the area of the transmitter beam.

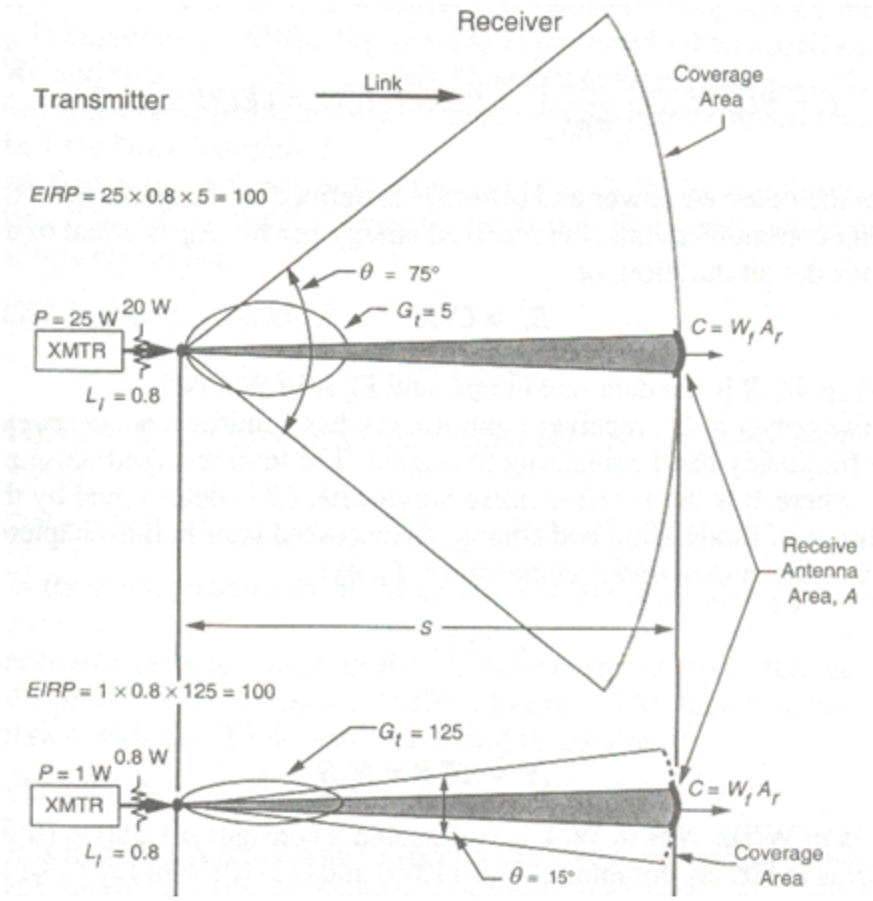


Figure 3.3 Directivity of the transmitter [26]

Figure 3.4 is an example of link budget divided in blocks: Transmitter, Power Amplifier, Transmitter antenna, Losses in Free Space, Extra losses, Receiver antenna, Low Noise Amplifier, Receiver Band Pass Filter, Second Low Noise Amplifier and the Receiver radio. In this case we use red of Iridium satellites.

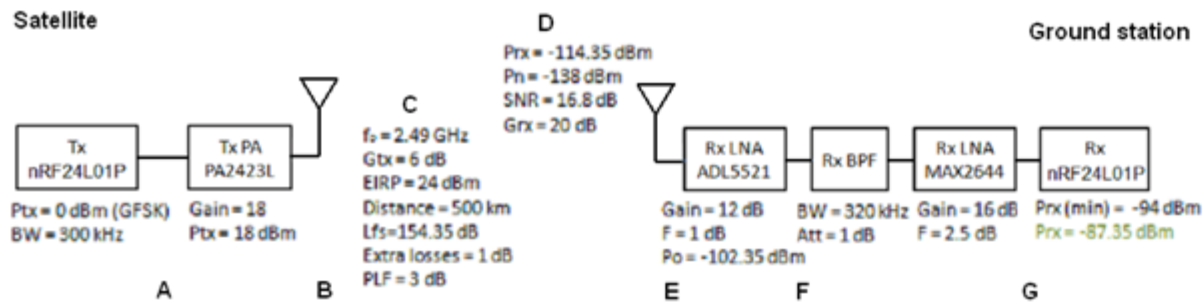


Figure 3.4 Radio-link system block diagram example

3.3. Power budget

First we need to determine the characteristics of Electric Power System (EPS) in power budget of satellite. The principal roll of the EPS is to condition, transfer and distribute power from solar cells to the different subsystems of the satellite and recharge the batteries. The EPS should comply with some functions as a minimum like, providing a continue electrical source during the lifetime of the mission, control and distribute electrical source, accomplish with requirements of electrical power (reduce power peaks) and protect the spacecraft utile charge against failures in electrical power subsystem.

The considered topology for design EPS has three parts: solar panels, batteries and DC-DC voltage regulators/convertors. Moreover, in general EPS has general specifications such as shown in figure 3.5 inside of these specifications they have considered energy efficiency, maximum power transference and capacity of circuit control.

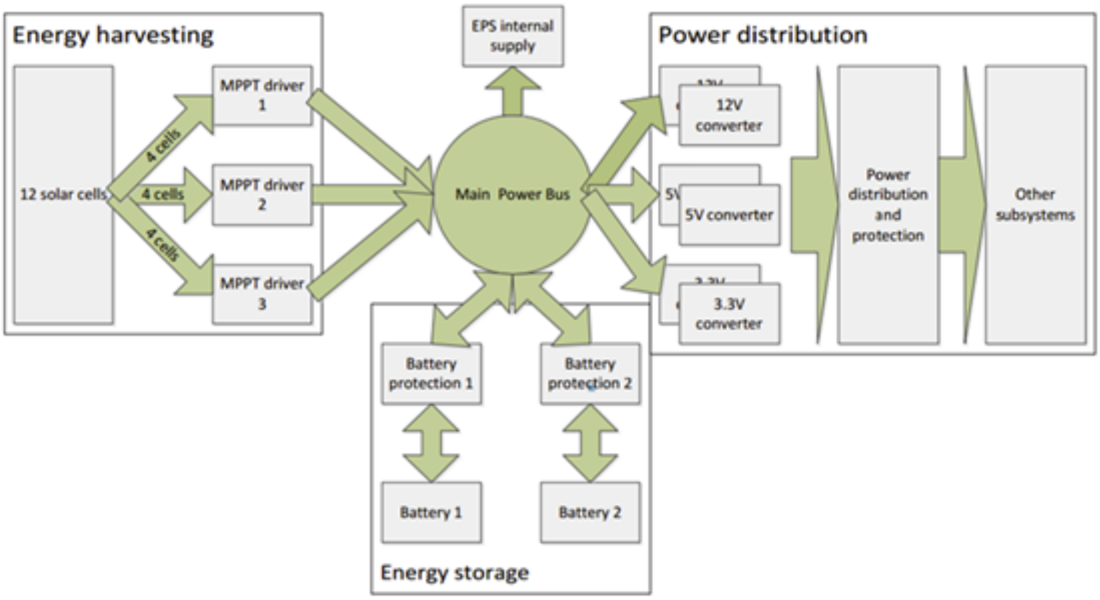


Figure 3.5 Genera design of the EPS

As a result to find the maximum range of EPS is mandatory to determine maximum peaks consumption of the different satellite subsystems. Table 3.1 contains values of the different subsystems of the AliNEO satellite. As explained above values of the table 3.1 contains maximum peak consumption of each subsystem, because EPS only needs to use their maximum capacity in short time, and if EPS is unable supply energy during these peaks, EPS fails in its mission.

Table 3.1 Types of components and their consumption in mA.

Components	Voltage (V)	Intensity (A)	Power (W)
Raspberry pi + camera	*5*	*900 mA*	*2,64 W*
GPS + Attitude control	*5*	*30 mA*	*41 mW*
Communication TX	*5*	*1.3 A*	*6.5 W*
Communication RX	*5*	*156 mA*	*0.78 W*
Magnetorquer (MT10-2-H)	*12*	*83.3 mA*	*1 W*
Magnetometer	*12*	*25 mA*	*< 0.3 W*
Total		*2.418 A*	*10.830 W*

From Table 3.1, we have accurate information from the datasheets of each component. Moreover, values are rounded up. The result of the power budget is $11W$ consumption in total, is special attention that this 11 W are in phase, because of existing different ways of add up power, we considered all in the same phase, and the change is minimum.

From safety regulations we need to add 20% to the real value of consumption obtaining a total value of $14W$. From real data we know that solar cells in space industry have $40W/kg$ and $300W/m^2$ [25], therefore, we can calculate solar cells weights and area such as shown in formula 3.7 and 3.8.

$$\frac{14W}{40W/kg} = 0.35kg \tag{3.7}$$

$$\frac{14W}{300W/m^2} = 0.04666m^2 \tag{3.8}$$

The most important for a good design of the EPS, is the power. All limitations and requirements of EPS, is marked by the maximum power and if we know the maximum power consumption (in this case $14W$), we can design EPS with all safety and condition requirements and avoiding its failure.

In addition the proposed cells to use in satellite there is a commercial cell for toys and it that has an area of $0.504\times10^{-3}m^2$. Thus we can calculate the number of solar cell needed from information such as one solar cell normally has $50mA$ and $0.6V$, which represent $30mW$ power output from each cell [25].

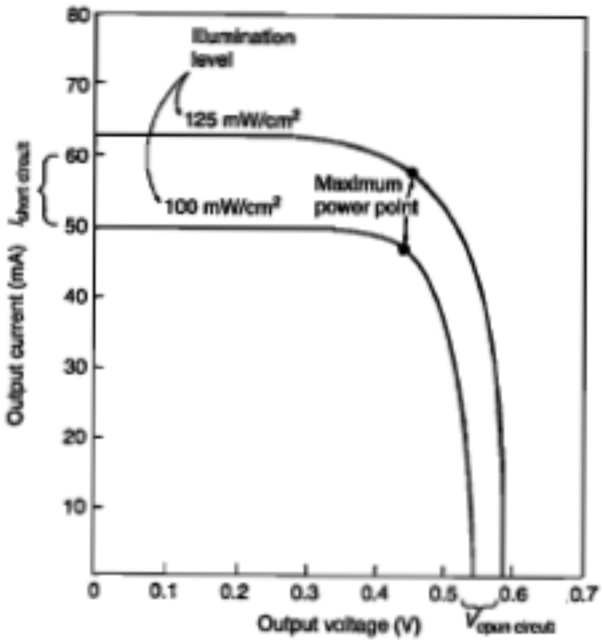


Figure 3.6 Solar cells characteristics

In figure 3.6 we can observe the characteristics of the solar cells technology that works to achieve necessary voltage from light radiation, moreover in case of the satellites in orbital 200 to 220 km of Earth, there are in ionosphere zone where is present the affects of the sun in different layers, in our case we are in layer F that it extends from about 150 km to more than 500 km above the surface of Earth, we can observe the structure of layers in ionosphere in figure 3.7 and the studies of NASA shows it is possible to work with characteristics solar cells inside of ionosphere layers.

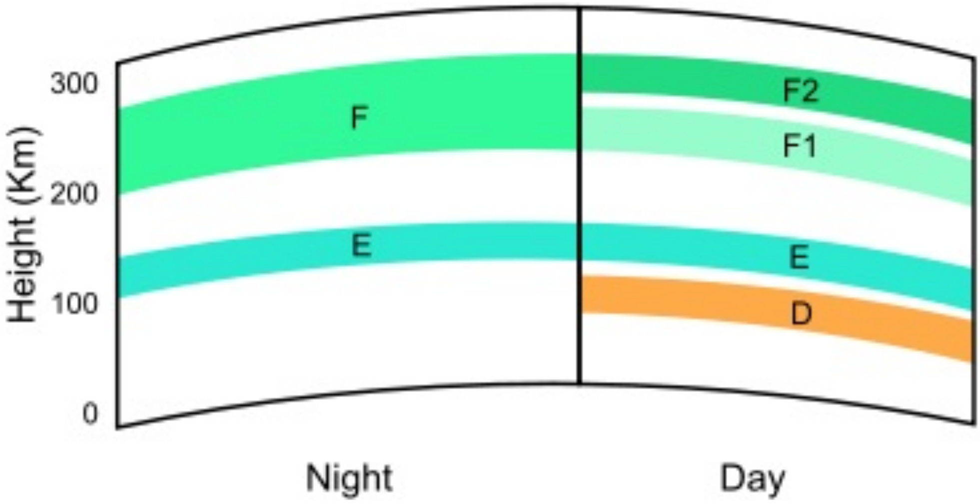


Figure 3.7 Ionosphere layers

Finally is necessary calculate how much radiation we received in each day, thus exist formula that in relation with each number of day of year we can calculate sun radiation and their quantity received on the Earth. In formula 3.9 we can appreciate formula it help us to know quantity of sun radiation. Moreover the LEOs are zones that no exist public information, as all of existent data are in use for military satellites, and we have no access to this material. So we can approximate to find solar illumination from formula 3.9.

$$E_{ext} = E_{sc} \cdot \left(1 + 0.033412 \cdot \cos\left(2\pi \frac{dn-3}{365}\right)\right) \quad \textbf{(3.9)}$$

E_{sc} : Solar illumination constant

dn : Number of day in year

Electrical power from renewable and clean sources, such as the sun, has always been the goal for alternative energy designers specifically for space applications. The application of Maximum Power Point Tracking (MPPT) algorithms to the solar energy filed can help us to find maximum point of power of solar cells and the possibility to improve consumption efficiency. In figure 3.8 we can appreciate the values of different consumption and maximum power.

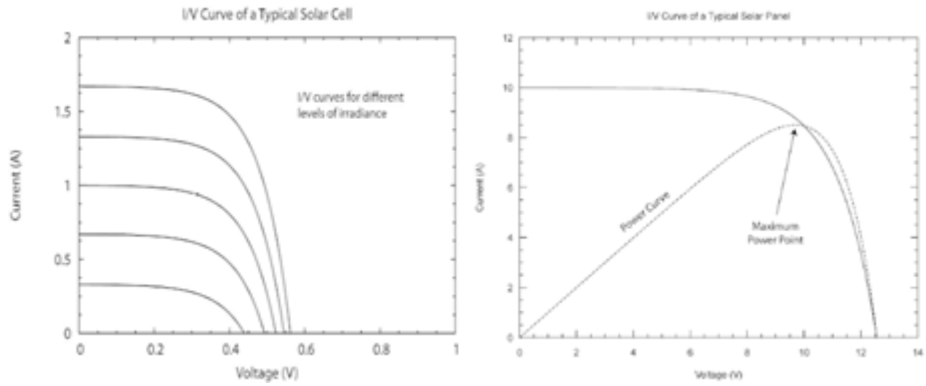


Figure 3.8 Maximum power point in solar cells

Now we can calculate the number of necessary cells from two designs: first design is based on a series connection and the second design on a parallel one. In general it is necessary $\frac{14W}{30mW} = 466,66 \approx 467$ coverage factor, furthermore, using parallel method we can reduce it until $\frac{34V}{0.6V} = 56,66 \approx 57$ cells, and on the other hand in series we need $\frac{467}{57} = 8.19 \approx 9$

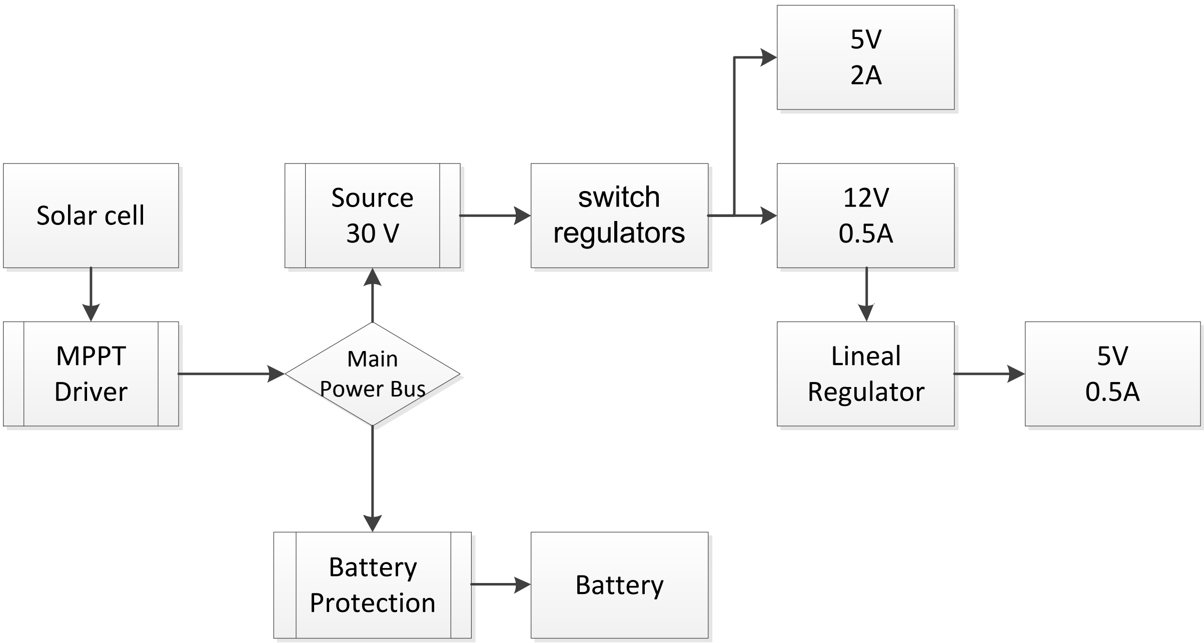


Figure 3.9 Sketch of ALINEO EPS

From figure 3.9 we can get some idea from general sketch of ALINEO EPS, and different parts and components for make it in good conditions. Moreover 57 are the required total amount of solar cell to achieve necessary power for the satellite and their EPS to work in safe conditions within the requested limits. On the other hand we need to calculate a number of necessary batteries for supply EPS, thus the satellite consumption diagram is showed in figure 3.10. Now using formula 3.7 we calculate medium intensity of EPS.

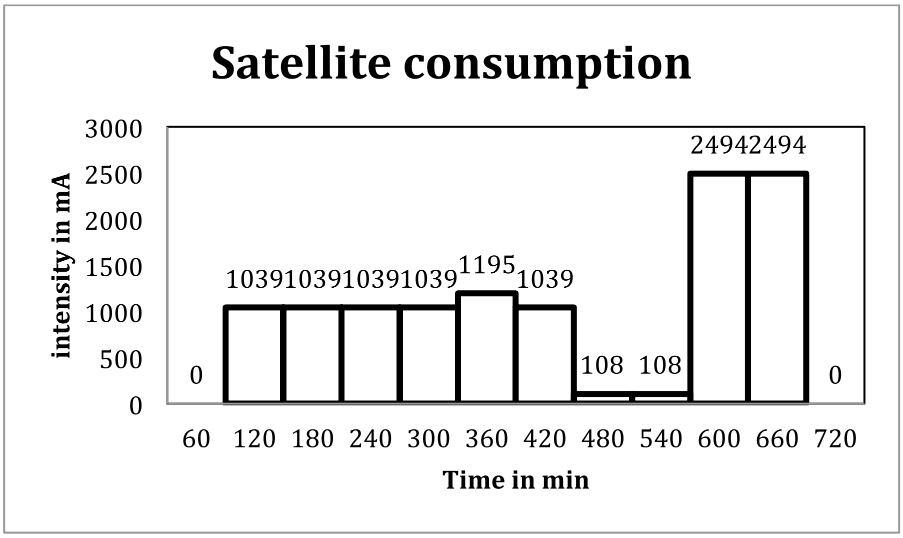


Figure 3.10 Satellite consumption diagram

As is possible to see in figure 3.10, the satellite is in standby first 60 min, after that the systems like motherboard, camera and attitude control start to work during 4 hours, after that is starts to work reception system of communication and turn off after one hour, moreover all systems turn off for two hours, only work attitude control, and finally all systems from attitude control to mother board, communication systems turn on to work for two hours, and after that all systems came back to turn off.

This diagram is for 12 hours periodic time works for satellite, and for this diagram it was considered two hours for refrigerate satellite in case of arrive to work limit temperature.

$$\mathrm{I}medium = \frac{\sum_i (T_i \cdot Pick_i)}{T_{total}} \quad \textbf{(3.10)}$$

$$I_{medium} = \frac{[(1.04 \times 60) \times 5] + (1.20 \times 60) + [(0.11 \times 60) \times 2] + [(2.42 \times 60) \times 2]}{720} = 0.95A$$

Designing the satellite we consider time of shadow orbital of satellite is about 12 hour and capacity of satellite batteries supplying shadow time is $12\ \mathrm{h} \times 0.95\ \mathrm{A} = 11.4\ \mathrm{Ah}$

In accordance with NASA studies, the range of voltage used in satellite is between 25 to 30 volts [25], and the batteries we use for satellite have 3.7 nominal voltages with 900 mAh. Finally to find the number of batteries cells, we need to make different design. As we know if we want high voltage we add batteries in series, and if we want high intensity we add batteries in parallel. Now we know $11.4\ \mathrm{Ah} \times 900\ \mathrm{mA} = 12.6\ \mathrm{Cells}$ and finally there is 13 batteries cells connected in parallel for supply our 11,4 Ah required with total weight about 300 grams.

Designing EPS we appreciated that we need two different kinds of regulators, one of them in lineal and on the other hand other use switches. The function of a regulator is to provide a constant output voltage to a load connected in parallel with it in spite of the ripples in the supply voltage or the variation in the load current.

The linear regulator use Zener diode technology, this kind of diodes use their breakdown region with forward bias region, figure 3.11, to control itself, and it very good for use in regulate or stabilize a voltage source against supply or load variations. The Zener diode will continue to regulate the voltage until the diodes current falls below the minimum I value in the reverse breakdown region.

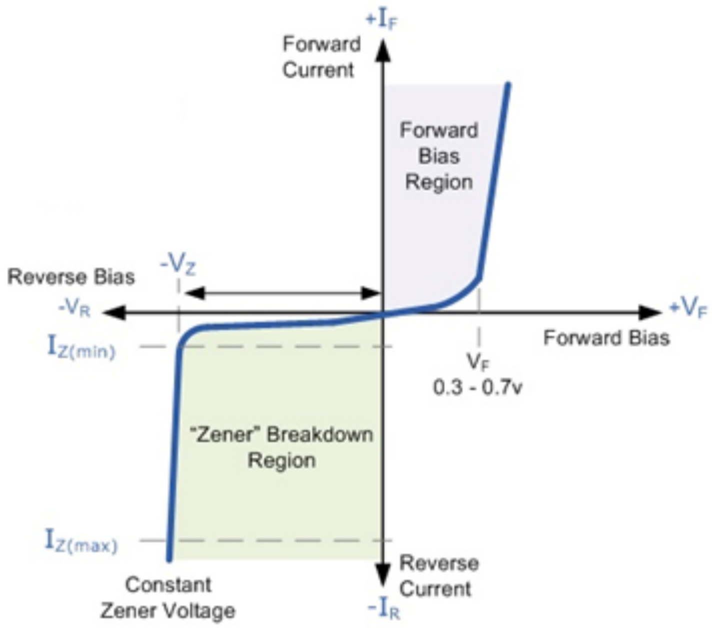


Figure 3.11 Zener diode work region

Moreover we know how is worked lineal regulator we design the necessary lineal regulator for our EPS. In figure 3.12 that we appreciate sketch of lineal regulator we calculate maximum current, R_s, I_L and I_S.

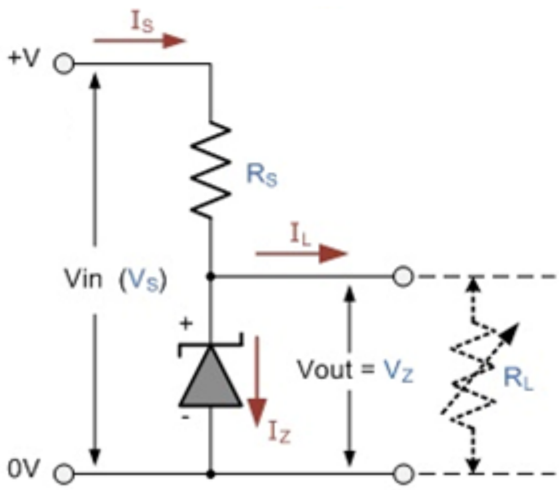


Figure 3.12 Sketch of linear regulator

Finally we find Zener current at full load, which we will use to supply our subsystems.

$$Current_{\max} = \frac{Watts}{Voltage} = \frac{2.64W}{5V} = 528mA \quad \textbf{(3.11)}$$

$$R_S = \frac{V_S - V_Z}{I_Z} = \frac{12-5}{528mA} = 5\Omega \quad \textbf{(3.12)}$$

$$I_L = \frac{V_Z}{R_L} = \frac{5}{1000\Omega} = 5mA$$
$$I_Z = I_S - I_L = 528mA - 5mA = 523mA \quad \textbf{(3.13)}$$

Especial attention is necessary for Zener diodes and lineal regulators; the efficiency of these regulators depends on $\eta = \frac{V_{out}}{V_{in}} 100\%$ and in this case is better to use Lineal regulator after switch regulator with 12 V_{output} this form efficiency of the lineal regulator stays in 42% from 12 v to 5 v. an example of lineal regulator is chip LM7805, and this kind of regulators incorporate transistors further the Zener diodes [29].

Now we need design switch regulators but it is very hard to design and improve them, in consequence we use Texas instrument[4] webpages to design our necessary regulator.
In these cases we need two kind of the regulator:

First case 30 V to 5.5 V design showed in figure 3.13 with their Thermal analysis figure 3.14.

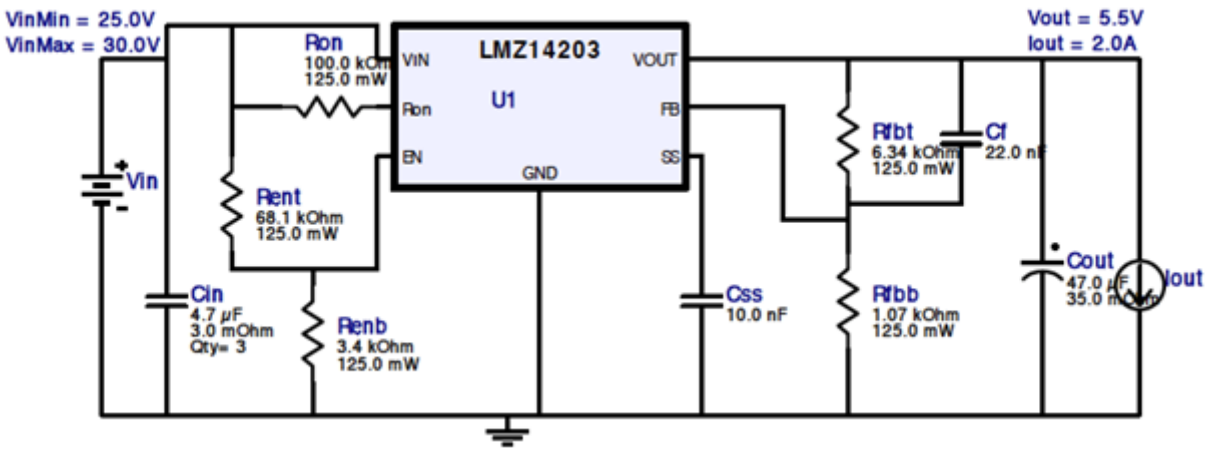


Figure 3.13 Switch regulator Vout: 5.5 V.

4 www.ti.com

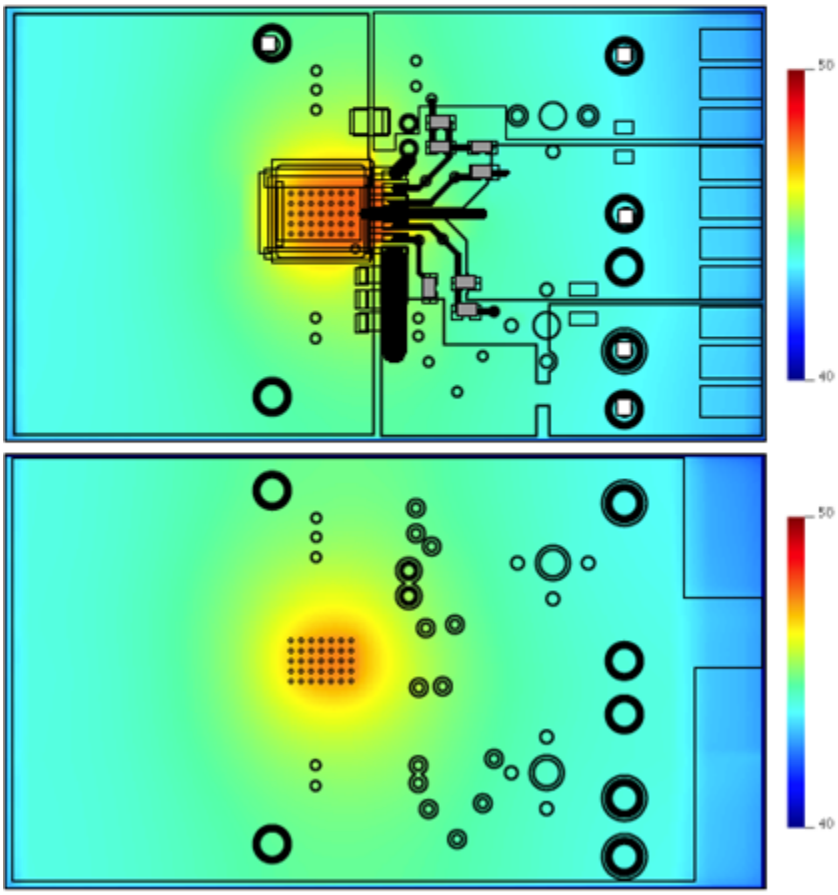


Figure 3.14 Thermal analysis Vout: 5.5 V.

Second case 30 V to 12 V design showed in figure 3.15 with their Thermal analysis, figure 3.16.

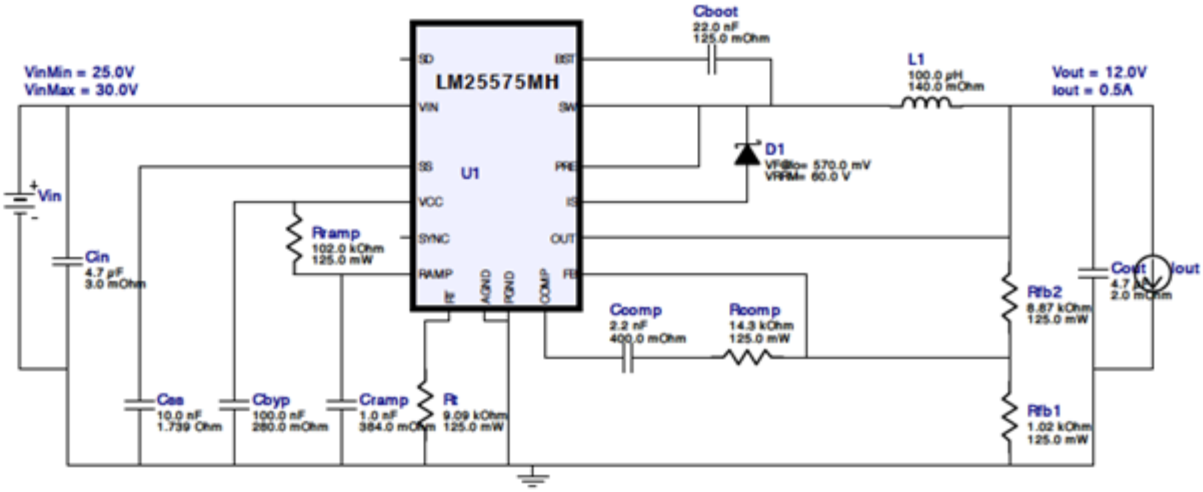


Figure 3.15 Switch regulator 12 V

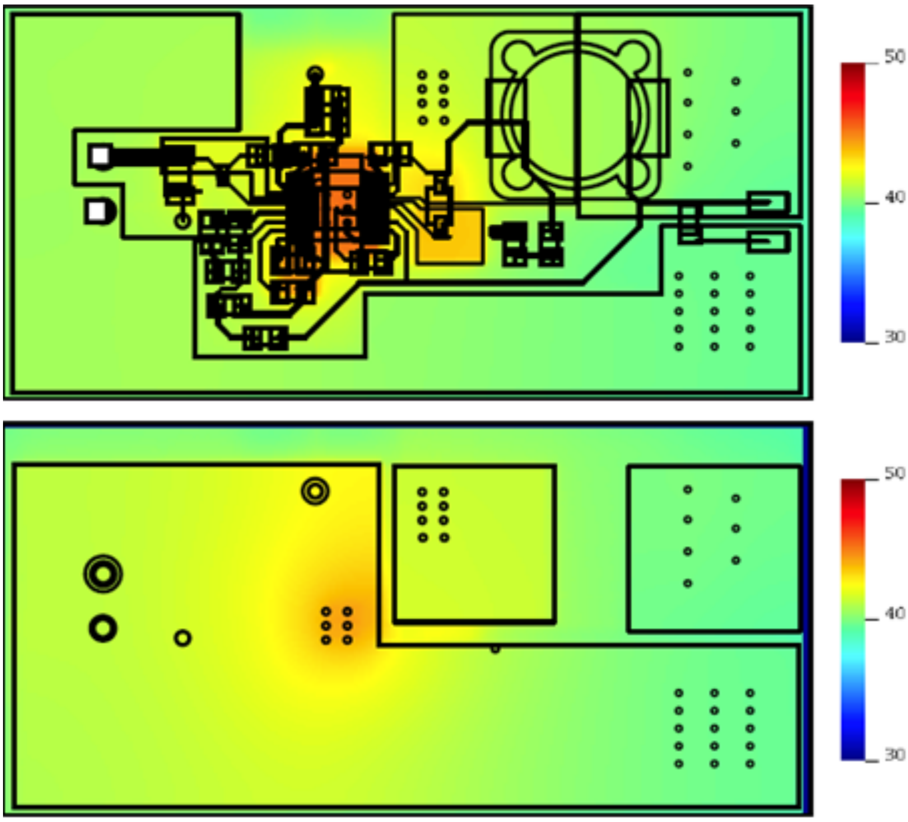


Figure 3.16 Thermal analysis Vout: 12 V.

Therefore, temperature radiation from EPS and dissipation of heat in space convert in substantial task. Thus we calculate the necessary temperature to work in safety rang in formulas 3.14. In this formula the R_T mean Thermal resistance $R_{T_{je}}$ T.Re Junction environment, $R_{T_{jc}}$ T.Re Junction capsule, $R_{T_{ce}}$ T.Re capsule environment.

$$T_j = T_e + \left(P \cdot R_{T_{je}}\right) \quad \textbf{(3.14)}$$

In case of first switch regulator it has P=0.84 W, if we have $R_{T_{je}}$ =50 ºC/W and the environment temperature inside of satellite is about 30º C, we obtain 72º C for junction temperature. Exactly the same operation for second switch regulator with P=0.33 W we obtain 46.5º C.

Now we calculate values for maximum environment temperature knowing the maximum temperature of junction that transistors endure is 200º C, and in two cases we to accomplish next relation:

$$T_e + \left(P \cdot R_{Tje}\right) \le T_{j\ Max} \quad \textbf{(3.15)}$$

In first case we have 139.52º C and 184.65º C, observing these two values we see that have a large margin of safety.

3.4. Thermal budget

There is a heat flow of income heat and outcome heat. The heat flow received by the satellite is determined by equation 3.16 and the heat emitted by the satellite is determined by equation 3.17.

$$Q_{in} = \alpha \cdot I \cdot F \cdot A \tag{3.16}$$

α : Intensity
I : Absorbency
F : Geometric factor
A : Effective area

$$\dot{Q}_{out} = \sigma \cdot T^4 \cdot \sum (\varepsilon \cdot A) \tag{3.17}$$

σ : $5.67 \cdot 10^{-8} w/(m^2 \cdot K^4)$ Boltzmann constant
T : Effective temperature
ε : Emissivity

First we propose a lightweight design without sophisticated systems, in which passive thermal control subsystem represents a good option. Using composite materials, paint with good albedo, very good reflective areas, etc. so these concepts of design can help to achieve good passive thermal control subsystems

In addition we use CAD model of satellite for thermal propagation study. The proposal alloy for this study was 7000 Series Aluminum Alloy (AA), and more specifically the 7075-T6 AA. This kind of AA has a specific characteristics showed in the Table 3.2 for more information.

Table 3.2 Aluminum Alloy properties

Properties	**Metric**	**Comments**
Density	*2.81 g/cc*	*AA; Typical*
Ultimate Tensile Strength	*572 MPa*	*AA; Typical*
Tensile Yield Strength	*503 MPa*	*AA; Typical*
Modulus of Elasticity	*71.7 GPa*	*AA; Typical; Average of tension and compression. Compression modulus is about 2% greater than tensile modulus.*
Poisson's Ratio	*0.33*	
CTE, linear 250°C	*25.2 µm/m-°C*	*Average over the range 20-300ºC*
Specific Heat Capacity	*0.96 J/g-°C*	
Thermal Conductivity	*130 W/m-K*	
Melting Point	*477 - 635 °C*	
Aging Temperature	*121 °C*	

After application of Aluminum Alloy over CAD by SolidWorks, we process the thermal propagation in the satellite model, and the final result is showed in figure 3.17. For the propagation test we consider the emissivity of the AA about 0.15, 0.5 of view factor and in this case the thermal flow is about 230 W/m^2 and then we studied what would happen if the satellite received 100 degree of Kelvin.

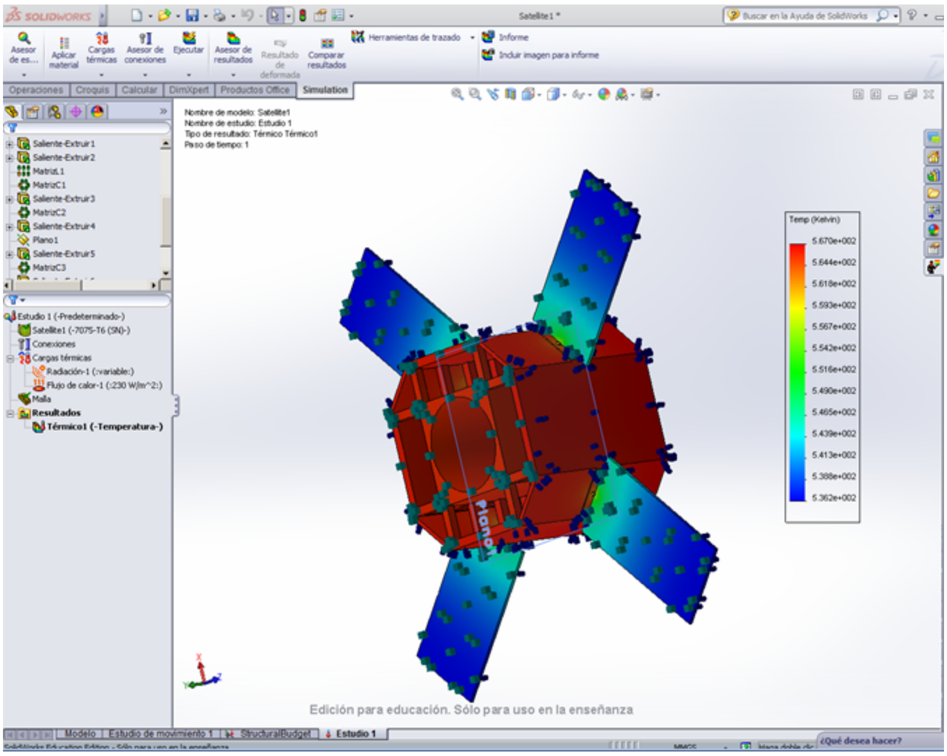


Figure 3.17 SolidWorks Thermal Study

On the other hands we know about use of aluminum alloy require more protection for avoiding problems of radiation, because of no existing protection layers. Due to this point, is mandatory to use composite material layer for protection of structure, so add more mass to structure for protection from radiation is one solution but other solution exist in replace all of structure from aluminum to composite material, in this way is helpful to avoid add more material and excess mass.

Now the question is how is possible to make structure that including radiation shield, and answer is use material that has Hydrogen in his formula [22]. According to NASA studies radiation-shielding materials containing Hydrogen, Boron, and Nitrogen.

Thus we propose to make principal structure of satellite by composite material, PUR. Polyurethane is a polymer composed of a chain of organic units by carbonates links. In the figure 3.18 is shown chain of Polyurethane synthesis in formula

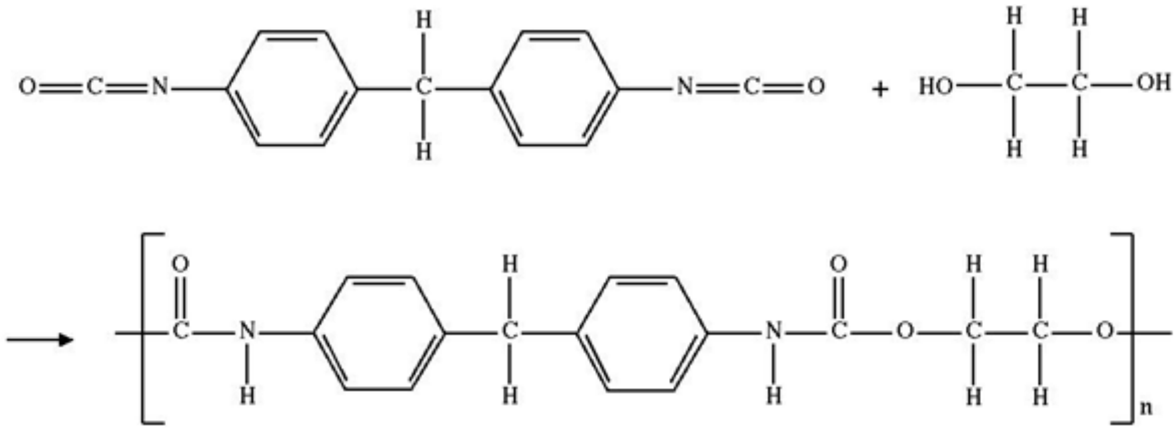


Figure 3.18 Polyurethane synthesis formula

Polyurethane synthesis has a specific characteristics showed in the Table 3.3 for more information.

Table 3.3 PUR properties

Properties	Values
Elasticity module	2409999872 N/m^2
Poisson	0.3897
Cutting module	862200000 N/m^2
Mass density	1260 kg/m^3
Traction limit	40000000 N/m^2
Thermal conductivity	0.2618 W/(m.k)
Specific heat	1900 J/(kg.K)
Emissivity	0.01

To compute for the total heat flow we need to consider concepts such as Sun, albedo, irradiation and dissipated heat. Moreover in this part it is mandatory the fabrication of the satellite and measure all positions and situations for calculation the heat flow. Consequently in this work it is impossible to calculate. Thus we discuss about thermal design.

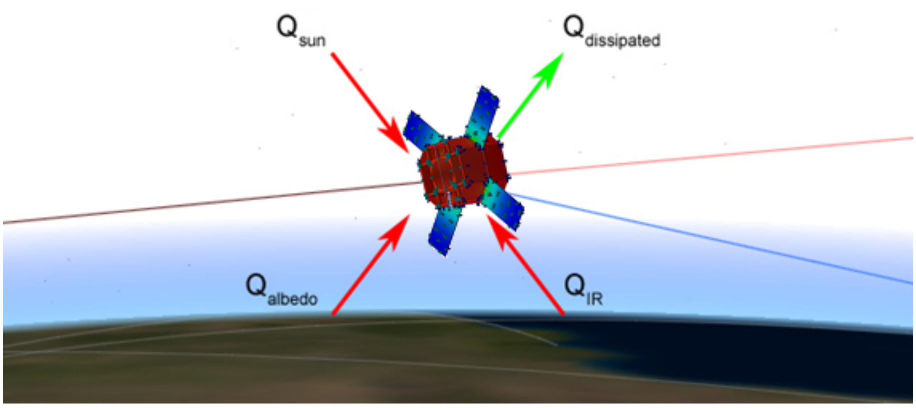


Figure 3.19 SolidWorks Thermal Study

The AliNEOs satellite has a passive thermal design based on the correct choose of face material selection. The satellite is a box of 180x180x120 mm (Face A) where one face is oriented to the Sun including the four solar panels of 120x50 mm (Faces G, H, I and J). Opposite faces are exposed to the Albedo and the Earth IR are B, C, D, E, F and for the solar panels are K, L, M and N.

When the satellite is exposed to the Sun, large amount of heat is incoming through but when Eclipse, only the Earth exposed surfaces are warming the satellite. The satellite is made of polished copper (Faces *A*, *C*, *D*, *E*, *F* and *G*), which sets the minimum temperature range of 74 degrees between the minimum during the Eclipse and the maximum exposed to the Sun.

The average satellite emissivity is $\bar{\varepsilon}_{Sat}$=0.8089 when the *A* face defined by the *A* and 4 solar panels face, with different areas.

$$\begin{aligned} A\,face &= 0.18*0.18 = 0.0324\,m^2 \\ B\,face &= 0.12*0.18 = 0.0216\,m^2 \\ C\,face &= 0.12*0.18 = 0.0216\,m^2 \\ D\,face &= 0.12*0.18 = 0.0216\,m^2 \\ E\,face &= 0.12*0.18 = 0.0216\,m^2 \\ F\,face &= 0.18*0.18 = 0.0324\,m^2 \end{aligned}$$

$$\begin{aligned} A_\varepsilon &= A*0.485 = 0.0157 \\ B_\varepsilon &= B*0.99 = 0.0214 \\ C_\varepsilon &= C*0.99 = 0.0214 \\ D_\varepsilon &= D*0.99 = 0.0214 \\ E_\varepsilon &= E*0.99 = 0.0214 \\ F_\varepsilon &= F*0.485 = 0.0157 \end{aligned}$$

$$G\,to\,N\,face = 0.12*0.05 = 0.006\,m^2$$
$$8*0.006 = 0.048\,m^2$$

$$GtoJ = G*0.85 = 0.0051$$
$$KtoN = K*0.99 = 0.0059$$

$$\begin{aligned} Sun\,face &= A+G+H+I+J = 0.0564\,m^2 \\ Earth\,face &= B+C+D+E+F+ \\ &+K+L+M+N = 0.1428\,m^2 \\ S &= A+B+\ldots+N = 0.1992\,m^2 \end{aligned}$$

$$\sum A\cdot\varepsilon = 0.1611$$

$$\bar{\varepsilon}_{Sat} = \frac{\sum A_n\varepsilon_n}{S} = \frac{0.1611}{0.1992} = 0.08089 \quad \textbf{(3.18)}$$

In case of AliNEO satellite the maximum cooling heat flow in hotter limit work zone of satellite is about 60ºC (333.15 K) and it is capable to dissipate near 112.59 W

$$Q_{Cooling} = \sigma T^4 \sum A\cdot\varepsilon = 5.67\cdot10^{-8}\cdot(333.2)^4\cdot0.1611 = 112.59\,W \quad \textbf{(3.19)}$$

Furthermore, when the satellite is exposed to the Sun $E_{Sun} = 1{,}420\,W$, faces A and four solar panels (G, H, I and J) are heated, but faces B, C, D, E, F and other four solar panel (K, L, M and N) are in the shadow towards the Earth. Since Sunrays are somehow parallel, the form factor is $k_{Sun} = 1$.

The faces opposite of sun receive heat from Albedo and IR sources. The infrared energy is $E_{IR} = 244W$ and the form factor is $k_{IR} = 0.75$ because rays are coming from the near Earth, rays are not parallel. The heat from the Albedo is the Sun heat reflected by the Earth while the infrared heat is due to the Earth temperature. We assumed Albedo as $E_{Albedo} = 0.32 \cdot E_{Sun} = 454.4W$ and the form factor is $k_{Albedo} = 0.5$ because rays coming from the Earth reflection are very dispersive. For the Sun case, there are two conditions when Idle and when Transmitting:

$$\begin{aligned}
&Q_{Sun} = \alpha \cdot E_{Sun} \cdot k_{Sun} \cdot A_{Sun} = 0.9 * 1{,}420 * 1 * 0.0564 = 72W \\
&Q_{Albedo} = \alpha \cdot E_{Albedo} \cdot k_{Albedo} \cdot A_{Albedo} = 0.9 * 454.4 * 0.5 * 0.1428 = 29.2W \\
&Q_{IR} = \alpha \cdot E_{IR} \cdot k_{IR} \cdot A_{IR} = 0.9 * 244 * 0.75 * 0.1428 = 23.5W \\
&Q_{Idle} = 0.055W \\
&Q_{TX} = 14W
\end{aligned} \tag{3.20}$$

$$\begin{aligned}
&Q_{SunIdle} = 72 + 29.2 + 23.5 + 0.055 = 124.8531W \\
&T_{SunIdle} = \sqrt[4]{\frac{Q_{Idle}}{\sigma \sum A \cdot \varepsilon}} = \sqrt[4]{\frac{124.853}{5.67 \cdot 10^{-8} \cdot 0.8089}} = 341.91K = 68.76\,^{\circ}C \\
&Q_{SunTX} = 72 + 29.2 + 23.5 + 14 = 138.8W \\
&T_{SunTX} = \sqrt[4]{\frac{138.8}{5.67 \cdot 10^{-8} \cdot 0.8089}} = 351.08K = 77.93\,^{\circ}C
\end{aligned} \tag{3.21}$$

On the other hand when the satellite enters in shadow zone, satellite only receives heat from IR and it is necessary to recalculate two conditions Idle and transmitting.

$$\begin{aligned}
&Q_{EclipseIdle} = 23.5 + 0.055 = 23.6W \\
&T_{EclipseIdle} = \sqrt[4]{\frac{Q_{Idle}}{\sigma \sum A \cdot \varepsilon}} = \sqrt[4]{\frac{23.6}{5.67 \cdot 10^{-8} \cdot 0.8089}} = 225.38K = -47.77\,^{\circ}C \\
&Q_{EclipseTX} = 23.5 + 14 = 37.5W \\
&T_{EclipseTX} = \sqrt[4]{\frac{37.5}{5.67 \cdot 10^{-8} \cdot 0.8089}} = 253.15K = -20\,^{\circ}C
\end{aligned} \tag{3.22}$$

Moreover should take into account the satellite never will be hotter than the maximum temperature established by design (-20ºC to 70ºC) and in this case we can observe some values are out of range, and it is not in accordance of design, but this values has safety margin and we can tolerate this results. But is necessary to remember the satellite emissivity should be adjusted by painting it with white color.

Table 3.4 summarizes the heat flow through the AliNEO satellite, exposing maximum cooling heat and in comparing between sun case and eclipse case. But the important conclusion is if we are in sun case and the satellite is out of range temperature, the satellite turn off the transmission system to cooling satellite, and if the satellite is in eclipse case and out of range the satellite turn on the transmission system to heating satellite. Moreover the range of work is between -20º to 77ºC and al components will be able to operate in the range of temperature that satellite will be exposed.

Table 3.4 Thermal budget of the AliNEOs satellite

Maximum cooling heat flow						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Cooling	-	-	-	-	112.59 W	333.15 (60ºC)
Sun case						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Idle case	72 W	29.2 W	23.5 W	0.055 W	124.85 W	341.91 (68.76)
TX case	72 W	29.2 W	23.5 W	14 W	138.8 W	351.08 (77.93)
Eclipse case						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Idle case	0.000 W	0.000 W	23.5 W	0.055 W	23.6 W	225.38 (-47.77)
TX case	0.000 W	0.000 W	23.5 W	14 W	37.5 W	253.15 (-20)

3.5. Radiation budget

Radiation in space can classify between many types, for example according to type of radiation provenance, nature and effects. For our radiation budget it is convenient to consider two kinds of radiation: cosmic radiation (X and gamma rays) and trapped radiation. Cosmic radiation corresponds to electromagnetic radiation issued from cosmic phenomenon, for example supernovas, gamma-ray bursts, etc. Because of the high energy of this type of rays, it creates ionization radiation, capable to penetrate in matter more than other type of rays.

Trapped radiation consists in emitted radiation from Sun, like energetic particles in Omni directions sun storms. These particles have different intensities. Moreover these particles are composed from protons, electrons, helium core, ions and other types of heavy elements (C, O, N, etc.). However, around the Earth there are magnetic fields named the Van Allen belts, represented in the figure 3.20. The magnetic field helps to decrease effects of the cosmic and trapped radiation. Moreover, the flux of particles increases with the altitude. In LEO orbit the radiation is less harmful to the electronic components than in the higher orbits.

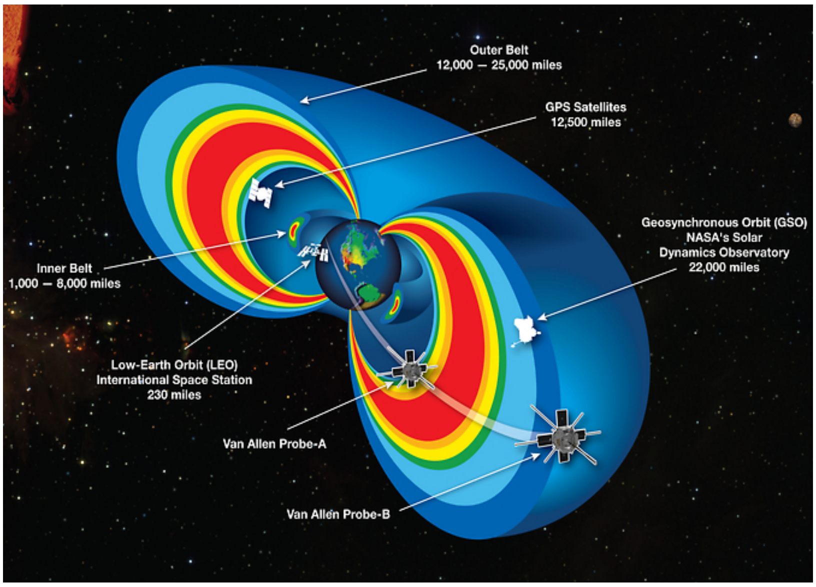
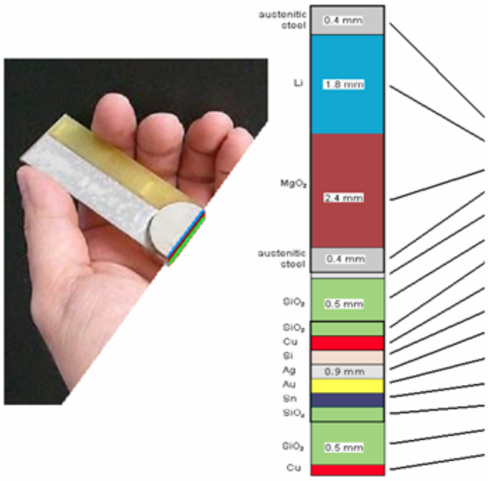


Figure 3.20 Van Allen belts

Finally, for protection of the satellite from different kind of radiation, use of the layers protection in making structure of the satellite is mandatory. For example in case of the WikiSat satellite radiation environment was studied by Molas and Bermejo [14, 15].

Figure 3.21 is a graphic explanation of the different layers of the WikiSat satellite, depending on the material and the thickness of layer; the particles will penetrate easier or not. The first 4 layers correspond to the battery, 2 correspond to thermal shield, 7 layers correspond to integrated circuits and 2 correspond to satellite boards.



Layer	Material	Thickness (mm)
1	Austenitic steel	0.4
2	Li	1.8
3	MgO_2	2.4
4	Austenitic steel	0.4
5	Al	0.02
6	SiO_2	0.5
7	SiO_2	0.13
8	Cu	0.13
9	Si	0.13
10	Ag	0.13
11	Au	0.13
12	Sn	0.12
13	SiO_2	0.13
14	SiO_2	0.5
15	Cu	0.1

Figure 3.21 WikiSat radiation layer protection

However WikiSat design correspond to femtosatellites, and in case of the NEOs detection satellites it is based about less or mores 1 Kg. also we need to order components of the satellite in the way that protect circuits of the satellites as much as possible.

3.6. Structural budget

To begin with structural budget, we consider some characteristics for the final design of the satellite. First for structural study we only consider the part of satellite that we design, avoiding COTS components, like the telescope. After that we apply the Aluminum Alloy to the structure and from it, we obtain final result. All computed data is from SolidWorks program.

First the Aluminum alloy choose was 7075-T6 (SN), this alloy has characteristics showed in table 3.2, after that the total mass of the structure is about 7717.53 grams, approximately 7.72 Kg. Moreover, the principal axes of inertia and the principal momentum of inertia measured from center of mass are:

$$I_x(grams \cdot mm^2) = \begin{pmatrix} 0.00 & 0.00 & 1.00 \end{pmatrix}$$
$$I_y(grams \cdot mm^2) = \begin{pmatrix} 1.00 & 0.00 & 0.00 \end{pmatrix} \qquad \textbf{(3.23)}$$
$$I_z(grams \cdot mm^2) = \begin{pmatrix} 0.00 & 1.00 & 0.00 \end{pmatrix}$$

$$L_{xyz}(grams \cdot mm^2) = \begin{pmatrix} 77578392.86 & 0 & 0 \\ 0 & 109767583.29 & 0 \\ 0 & 0 & 77578392.86 \end{pmatrix} \qquad \textbf{(3.24)}$$

Furthermore, for the behavior of the satellite during reentry of the satellite in their final lifetime, we apply a study for relation of the force among the one face and four solar panels. The final result was computed by SolidWorks program and it showed in the figure 3.22. Nevertheless the applied condition was 7.8 km/s^2 of velocity in LEO, with 0.000628 kg/m^3.

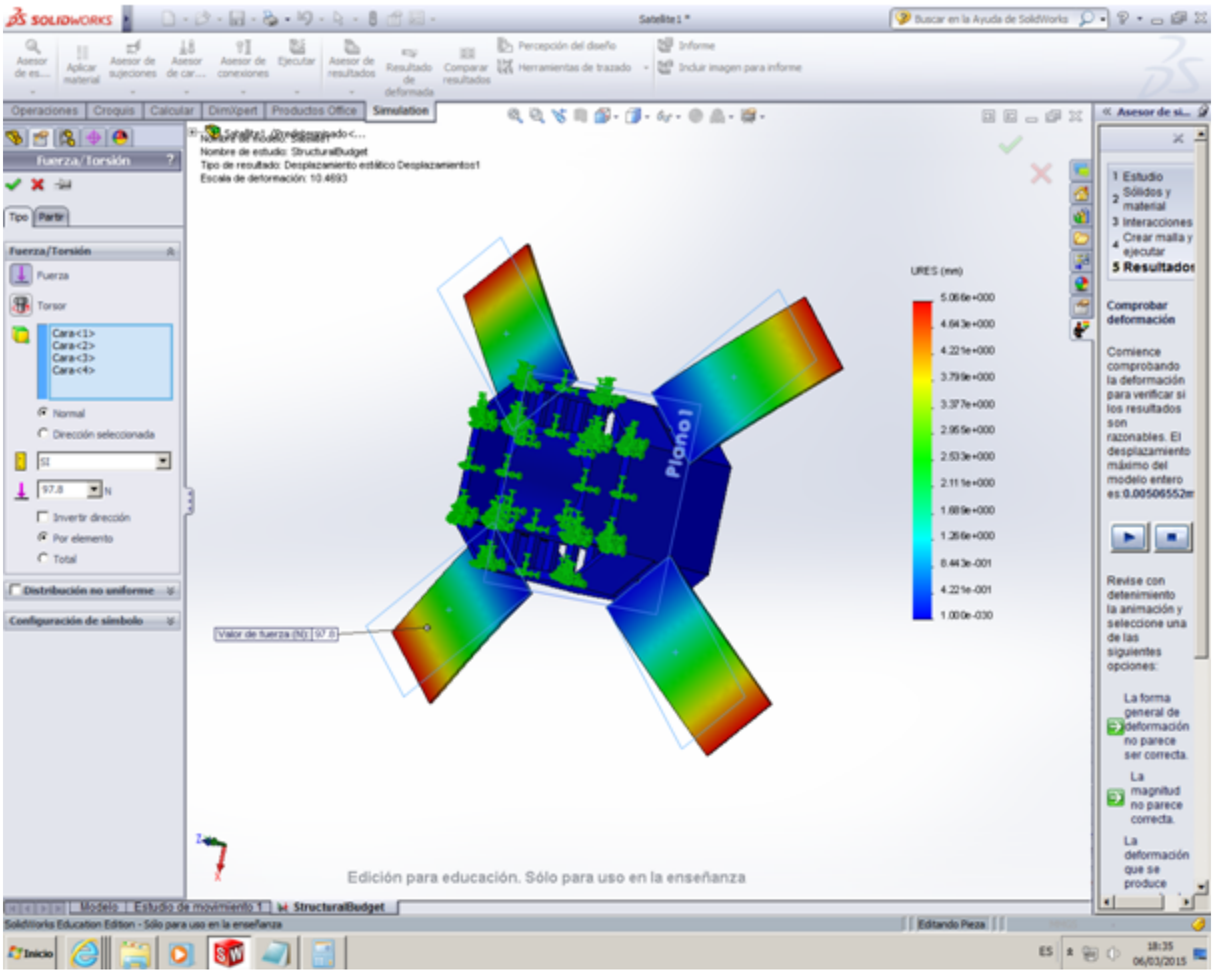


Figure 3.22 SolidWorks Structural Study

3.7. Mass Budget

We start to add up all mass of each component to give the total final mass of the satellite. It is very important in order to calculate the payload of launcher and see in which category it fits.

In table 3.5 we have all mandatory components to achieve functionality and their mass. The total is 2.2 kg; rounding to the top it will be 2.5 kg.

Table 3.5 Types of components and their mass and limitations

Components	Mass (grams)	Commentary
Raspberry pi + camera	*20*	*10000g*
GPS	*5*	*UBX-G7020*
Communication system	*10*	*Iridium*
Telescope	*1000*	*NG compact telescope*
Magnetorquer	*350*	*Model MT10-2-H*
Magnetometer	*60*	
Solar cells	*116*	*58 Toy solar cell*
Battery	*23,17*	*13 units of 3,7 V battery*
Structure	*682.87*	
Total	1862.21	

In addition for some orientation about how much will it cost of send 2 kg (4.40925 pounds) to LEO, in the table 3.6 we can see some total prices for different launch systems.

Table 3.6 Total cost of launching 2 kgs of payload

Launch Systems	***Cost per pound***	***Total Payload (pounds)***	***Final cost $***	***4 satellites***
Atlas V	*$13182*	*4.41*	*58132.62*	*232530.48*
Atlas IV	*$13072*		*57647.52*	*230590.08*
Ariane 5	*$10476*		*46199.16*	*184796.64*
Falcon 9 v1.1	*$4109*		*18120.69*	*72482.76*
DNEPR	*$3784*		*16687.44*	*66749.76*

As we can see in table 3.6 launching cost of 4 satellites in three cases is very expensive and the other two options are reasonably expensive, other aspect that we have not taken into account is lead time for each launch, in some case this time can be two years. These concepts encourage us to seek cheaper alternatives with shorter lead times.

3.8. Optical Budget

For the optical part of the satellite we need to take into account that the raspberry pi board, has a camera in it that can be connected directly on the board and accomplish with all characteristics that represent SMD. This camera is capable to record and make images from their sensors, and combined by COTS telescopes, we can record all necessary data for processing and transmit it to the ground station.

Figure 3.23 Low-cost infrared camera

On the other hand we propose to use the most popular telescope design, a Newtonian reflector telescope. This kind of telescope consists in single concave primary mirror and a reflecting flat. It has a single optical surface of the primary mirror that collects light coming from the image, focusing the lights to flat, and it directs light out to the side where observation of the light is accessible.

Exactly in this accessible place is where we link together telescope and raspberry camera board. In case of telescope, we decide to buy a compact telescope existing in the market (COTS) to reduce the costs and time of development [21].

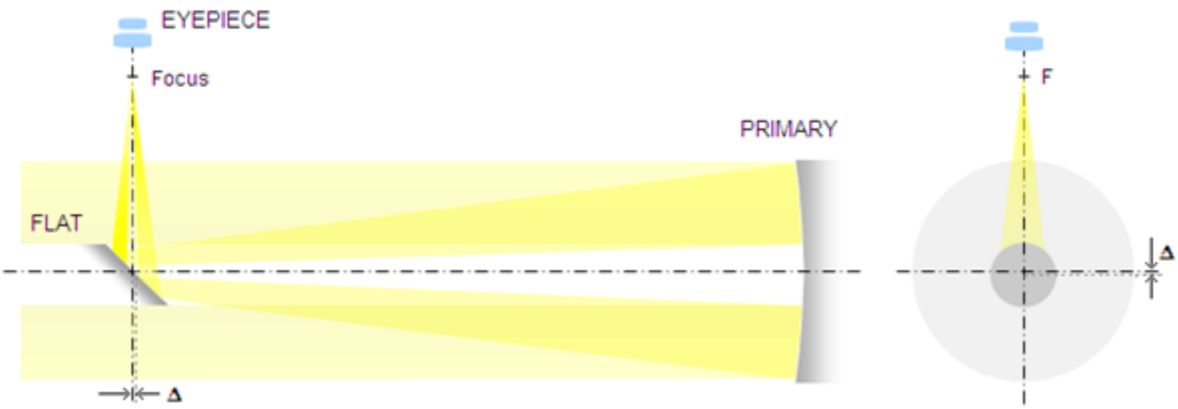


Figure 3.24 Sketch of Newtonian telescope

3.9. Fault tree analysis

A system is often considered in terms of probability of failure condition, considering at the same time the severity effects of failure. Figure 3.25 shows what is considered as acceptable in a general way.

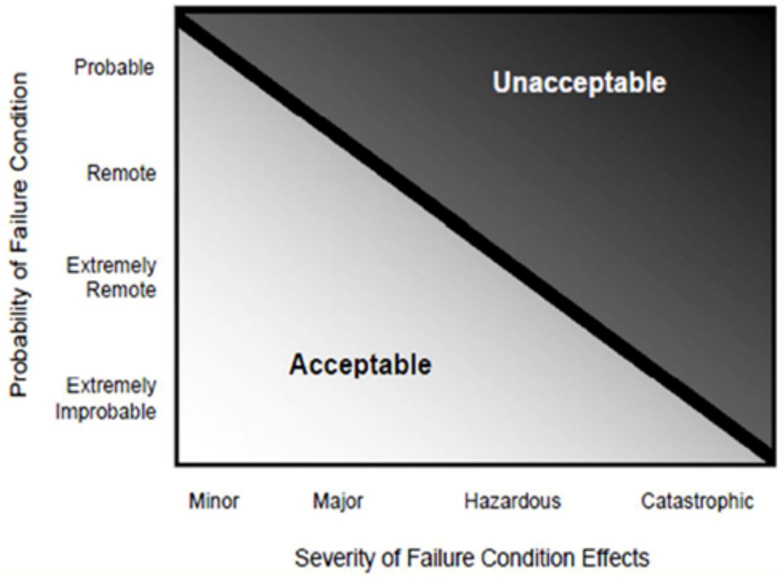


Figure 3.25 Probability versus Severity of Failure of the satellite mission

The fault tree analysis studies the probability of a Mission Failure. Figure 3.26 shows an example of a fault tree where the mission failure probability is about $4 \cdot 10^{-6}$, which is an extremely remote probability of failure condition based on standards like *ARP4761*. For demonstrate our fault tree we propose to use simulators to achieve quickly the final results, for example the Moon2.0 [28] is simulator used in WikiSat program, and it have very good data base for simulation.

In the following example it is considered three main components that can fail in the satellite: The Onboard Computer, the Camera Sensor and the Transmitter Radio. The probability of failure of each component and the reasons are detailed here:

- $P_{missionFailure}$= 10^{-6}. For mission failure should be fail electronic systems or transmission systems of Radio.
- $P_{TXfailure}$= 10^{-9}. For radio failure it most stop working.
- $P_{ElectronicFailure}$= 10^{-6}. For that exist an electronic failure, its necessary to fail camera or onboard computer.
- $P_{OCFailure}$= $2 \cdot 10^{-3}$. A failure of onboard computer it can happen in consequence of a Cosmic Ray or a Single Event.
- $P_{CameraFailurte}$= $2 \cdot 10^{-3}$. A failure of the camera it is possible in relation with Cosmic Ray or a Single Event.

Camera Sensor Failures

- $P_{cameraSingelEvent}$= 10^{-6}. As a consequence of radiation some transistors of the electronic circuits locked and broken, in addition the mission becomes unusable.

- $P_{cameraCosmicRays}$= 10^{-4}. Due to Cosmic Ray or radiation, some pixel brokes and the mission is compromised.

Onboard Computer Failures

- $P_{OCSingelEvent}$= 10^{-3}. As a consequence of radiation some transistor of electronic circuits is loocked or broken and as a result the mission becomes unusable.
- $P_{OCCosmicRays}$= 10^{-3}. Due to cosmic rays or radiation, some onboard compotator doors it broken and the mission is compromised.

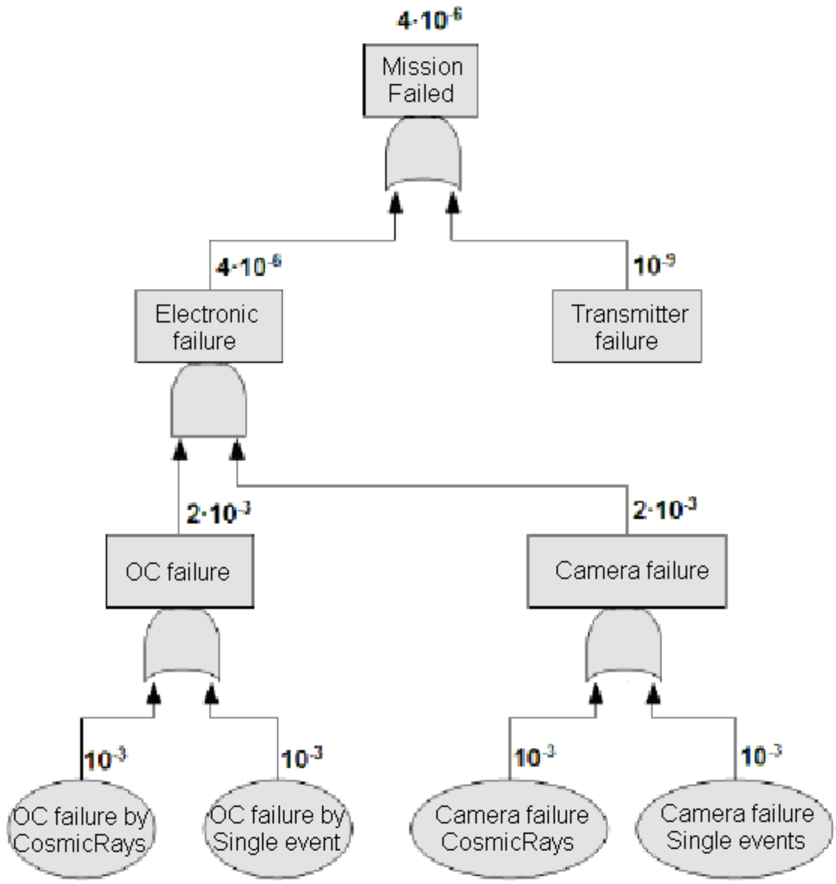


Figure 3.26 Fault Tree

Chapter 4

DETAILED SUBSYSTEM DESIGN

The Near Earth Objects Space Observatory has as a default configuration the following subsystems:

- Power Supply subsystem
- Communication subsystem
- Structure subsystem
- Position determination subsystem
- Attitude determination subsystem
- Attitude control subsystem
- Optical subsystem
- Tracking subsystem
- Onboard computer subsystem

4.1. Power supply subsystem

The power supply subsystem shall provide electrical power to the satellite, mainly for the computing of the orbit and the tracking [HL01].

These solar cell subsystems are based on the 2.4x2.1 centimeters; Solar cells use light energy from Sun to generate electricity, using photovoltaic effects [LL002]. In the general case of the photovoltaic effect cells these use very thin film wafer-based crystalline silicon or other type of materials. The photovoltaic cells proposed for satellites consist in very close cell used in toy car showed in the Figure 4.1. The principal reason to choose this solar cell is their low cost and little size. Many others kind of cells exist in the domestic market, but after some experiments with this solar cell, stiffness and durability represent good qualification to finally choose.

Figure 4.1 Example of low-cost solar panel in COTS

4.2. Communication subsystem

Communication links allow a satellite system to function by carrying tracking, telemetry, and command data and mission data between its elements.

In this case the proposed communication subsystem is based on the Iridium 9603 module. This transceiver combines the global coverage of the Iridium constellation, low latency and highly reliable satellite communications. The model number 9603 [18] is ideal for space applications including monitoring, tracking and alarm system.

With Iridium it is possible to observe environmental condition like Operating temperature range -30ºC to +70ºC also acceptable for this mission. In addition telecommunications frequencies are in the range of 1616 MHz to 1626.5 MHz, Duplexing method TDD (Time Domain Duplex), Multiplexing method TDMA/FDMA.

Finally power input of iridium is about voltage range 5.0V +/- .2V DC, an average Idle current about 45mA, with a peak 195mA. Peak and peak of transmit and receive current is about 1,5A to 195mA.

Figure 4.2 Low-cost satellite based modem

4.3. Structure subsystem

The structure of the satellite is based on commercial telescopes that exist in the market, helping to save costs of the system design. The structure of the satellite, involves all electronic, bus communication, processor and transmission and receiver data, magnet torques and etc. This second part is designed by 3D design program SolidWorks. First we have modeled in Solidworks a commercial telescope part, and second we also taken into account the size of the selected parts.

First part is shows in the Figure 4.3, second part in Figure 4.4 and the final result of the assembly of the two is shown in Figure 4.5. In addition it is necessary to take into account that the second part of the satellite will be manufactured using the newest technology of 3D printing by the inclusion of composite materials. Finally we propose to use name ALINEO for this satellite, the name is from combination of ALI (Author name) and NEO from Near Earth Object. And ALINEO in Spanish language means "Line up". This name we believe can represent the sprit of this project.

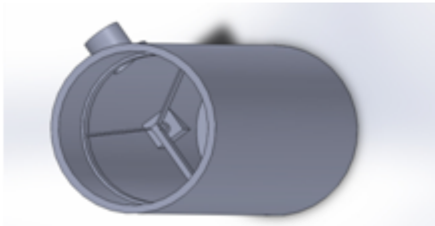


Figure 4.3 COTS telescope 3D model

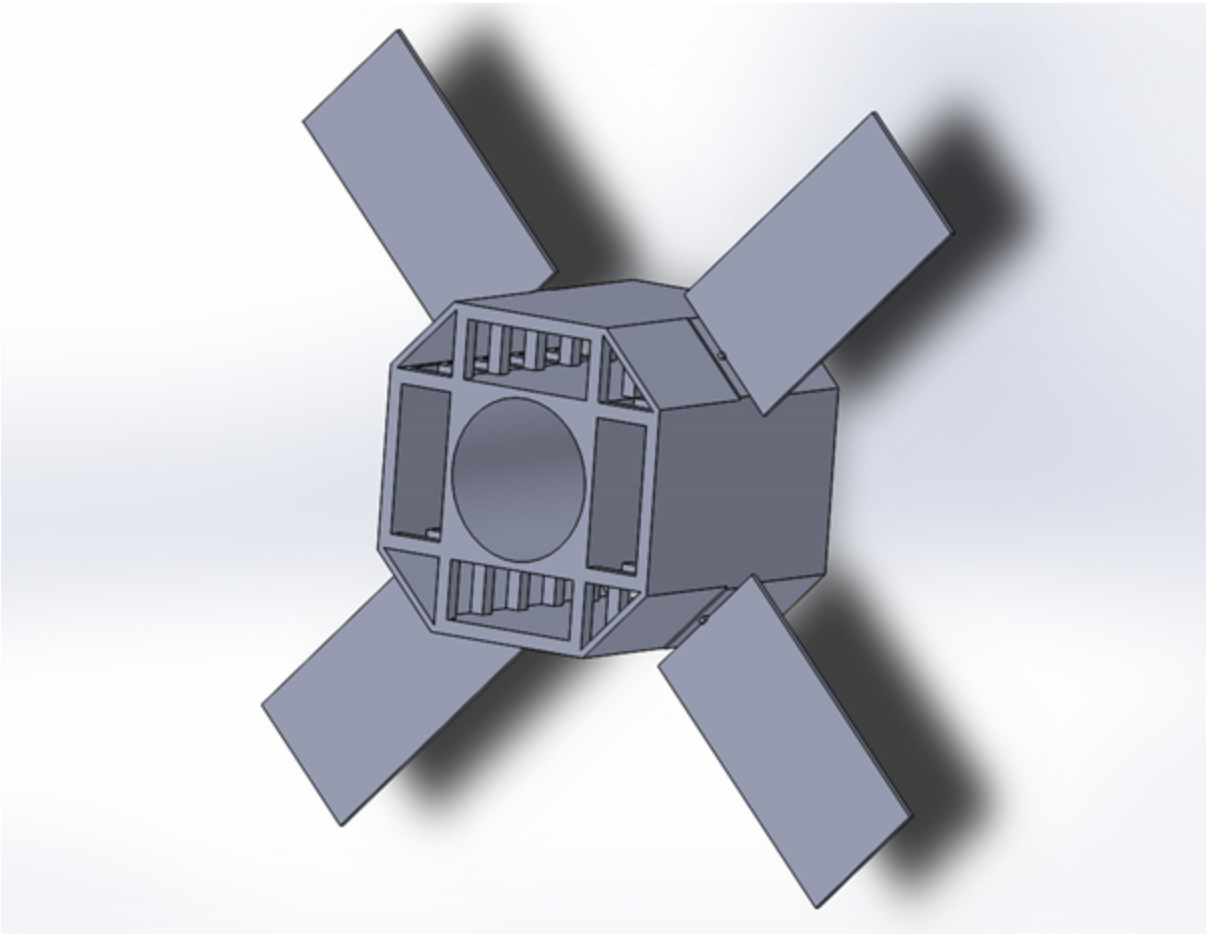


Figure 4.4 Satellite structure 3D model

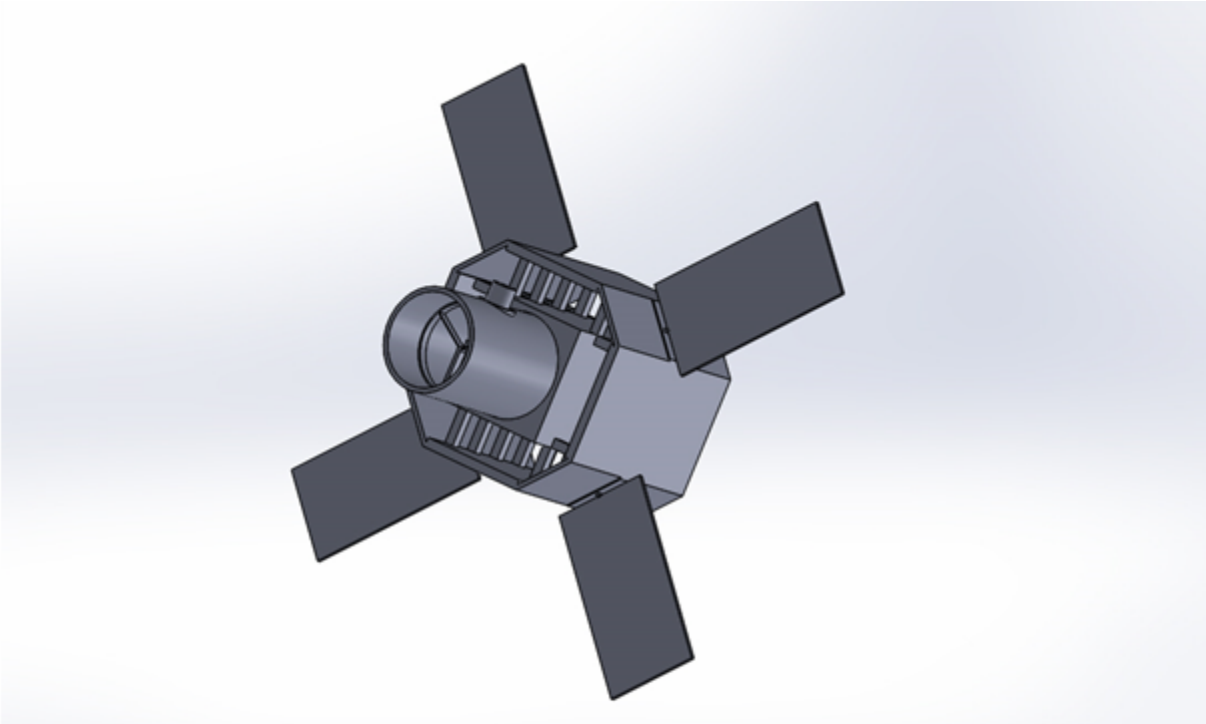


Figure 4.5 Final satellite assembly 3D model

4.4. Position determination subsystem

The position determination subsystem shall determine the position inside the orbit by use multi-GNSS standalone positioning chips; the proposal is the newest family of u-blox, the UBX-G7020 [17]. This model shown in Figure 5.6 is capable to use signals from GPS, GLONASS, QZSS, SBAS - Galileo/BeiDou. High performance 7 multi-GNSS position engine delivers exceptional sensitivity and acquisition times. The voltage supply range from 1.8V to 3.0V compliance supports a wide variety of applications.

In case of the position determination subsystem receiver type, accuracy and sensitivity are important for GNSS positioning the satellites.

Receiver type
- 56-channel u-blox 7 engine
- GPS & QZSS L1 C/A, GLONASS L1OF
- SBAS: WAAS, EGNOS, MSAS
- Navigation update rate up to 10 Hz

Accuracy
- GPS 2.5 m CEP, SBAS 2.0 m CEP
- GLONASS <4 m CEP

Sensitivity
- GPS: Tracking -162 dBm, Coldstarts -148 dBm,

- Reacquisition -160 dBm
- GLONASS: Tracking -158 dBm, Coldstarts -140 dBm,
- Reacquisition -156 dBm

Figure 4.6 Low-cost GPS chip in SMD format

4.5. Attitude determination subsystem

The attitude control subsystem shall determine the attitude of the satellites. Controlling the attitude of the spacecraft and its appendages is done autonomously on board for nearly all satellites. Controlling the attitude from the ground is too expensive and too risky. The attitude control system on board most spacecraft provides various attitude control modes and can work over extended periods with little or no intervention from the ground. The proposal attitude control is use the 11DOF IMU board (55x16mm) [16] with GPS receiver and 5V logic level. Moreover these boards integrate:

- HMC5883L Triple Axis Magnetometer
- BMA180 Triple Axis Accelerometer
- BMP085 Barometric Pressure Sensor
- ITG3200 Triple-Axis Digital-Output Gyro
- NEO-6Q GPS receiver (Avail. UART and I2C interface)

Figure 4.7 Low-cost 11 DoF Inertial Measurement Unit and GPS board

4.6. Attitude control subsystem

The main propose of the Attitude Control Subsystem (ACS) is to orient and control of an object to an inertial frame of reference. Controlling space vehicle attitude requires sensors to measure orientation, and supply necessary torques needed to reorient the space vehicle. Therefore ACS needs sensors and actuators.

Therefore, ACS divide between two important parts, Sensor and Actuators, as for the AliNEOs satellite we propose to use Magnetometer for sensors and Magnetic torquers as actuators.

The proposed model of magnetometer is AMR-RS422-LV, figure 4.8, from ZARM-Technik Company. This model is a microcontroller designed to measure the external magnetic field vector for satellite attitude determination and control. Because of integrated set of orthogonally arranged Anisotropic-Magneto resistive (AMR) sensors, the AMR-RS422-LV is capable to measure the magnetic field in all three directions, X, Y, Z.

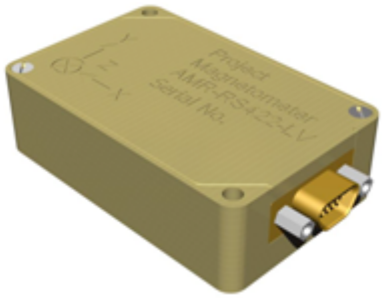


Figure 4.8 Magnetometer AMR-RS422-LV

On the other hand for ACS, the proposed actuators are the Magnetic Torquers, MTO families from ZARM-Technik Company, figure 4.9. These actuators interact with Earth's magnetic field and create control torque, which can be adjusted to the required value. The typical value of the Earth magnetic field intensity[5] is 20 uA/m^2 at 250 km of altitude.

They provide all the power needed to maintain the spacecraft attitude. Moreover, they were designed for cubesat applications where mounting space and mass is limited.

Figure 4.9 Magnetic Torquers MTO families

[5] Source: NOAA orld Magnetic Model http://www.ngdc.noaa.gov/geomag/WMM

4.7. Optical subsystem

As we can see in figure 4.10 the raspberry Pi camera board plugs directly into the CSI connector on the board. The raspberry Pi camera board features a 5MP (2592x1944 pixels) Omnivision 5647 sensor in a fixed focus module. The module attaches to Raspberry Pi, by way of a 15 Pin Ribbon Cable, to the dedicated 15-pin MIPI camera Serial Interface (CSI), which was designed especially for interfacing to cameras. The size of camera board is tiny, around 25mm x 20mm x 9mm and near 3 grams of the weight.

Figure 4.10 Low-cost camera sensor

It is very compact and usable Newtonian telescope, from national geographic where it consist in focus wheel, telescope tube, compass, Alt-azimuth mount, azimuth scale, scale with 90º steps and height adjustment wheel. Moreover for calculating the magnification exist follow formula “Focal width of the telescope tube / Focal width of the eyepiece = Magnification”.

The magnification is also depends on the focal width of the telescope tube. The telescope has a focal length of 350 mm, from this formula; we see that if we use an eyepiece with a focal width of 20 mm, we will get the following magnification.

$$\frac{350mm}{20mm} = 18X \cdot Magnification \tag{4.1}$$

Figure 4.11 National Geographic compact telescope

4.8. Tracking subsystem

The tracking subsystem shall transmit its computed position to a ground station only when it is passing over the ground station. The tracking massage contain the satellite ID, the position, the time and the battery voltage, the proposed tracking chip is a SMD Low noise amplifier type SMA661AS.

Figure 4.12 Tracking chip SMA661AS[6]

4.9. Onboard computer subsystem

We must include the capability to store commands if we require controlling the spacecraft when it is not in view of its ground stations, or as a means of recovery if the communication link is lost. These commands may be controlled by matching a time-tag or by a simple delay counter from controlled timing event. Stored commands of this type may be easily implemented without a general-purpose processor.

We must add an onboard computer if we require a decision-making element on the many functions, including the stored command capability, attitude control, and data processing and storage. Integrating attitude control with the command system will typically add some special interface requirements for driving control elements.

The proposal onboard computer subsystem presented in Figure 4.13 is Raspberry Pi, the Raspberry Pi is a Broadcom BCM2835 system on a chip, where in the same main board exist different unites to use, such as 700 MHz ARM1176JZF-S CPU [20] with a Broadcom VideoCore IV GPU, and 256 MB of SDRAM. The Raspberry Pi is a platform designed for education, for technological enlightenment, but it meets all characteristics that system needs in the same main board.

6 http://www.qic.com.cn/qicfileserver/readFile.action?filePath=/pdf/STMICROELECTRONICS/SMA661AS.pdf

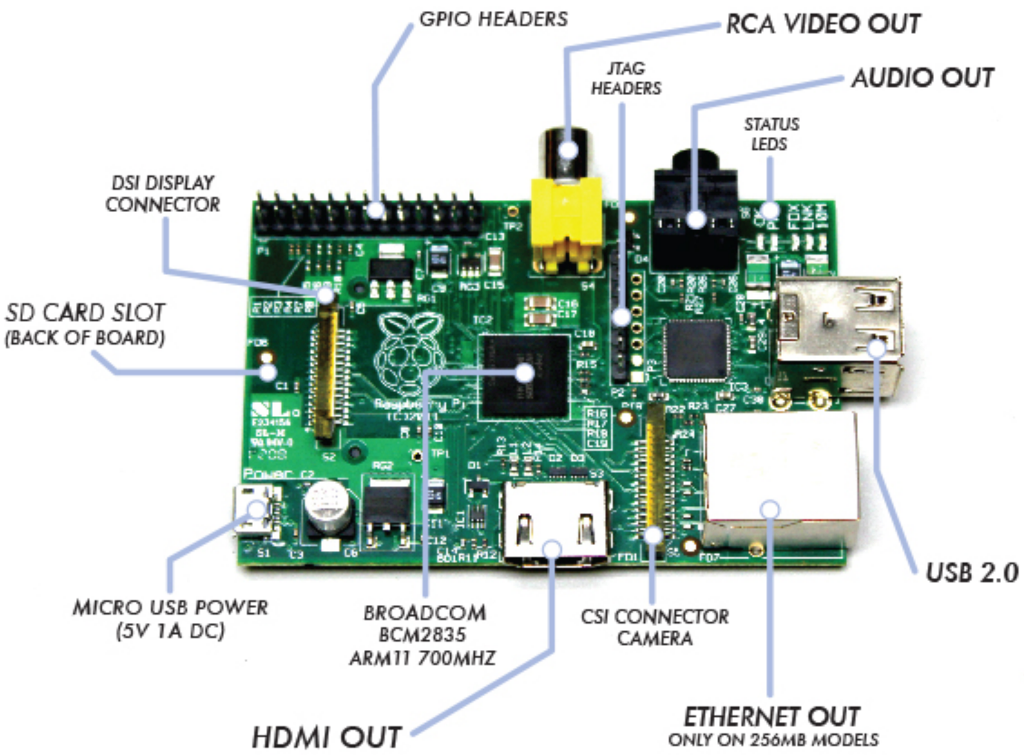


Figure 4.13 Raspberry PI board. Source: PC MAG

Finally we intent to install software we use for control of satellite in raspberry Pi, but it was impossible to many different problems of compatibility the programs with raspberry, and final we find the solution in use the model from Intel company, Intel Galileo, figure 4.14.

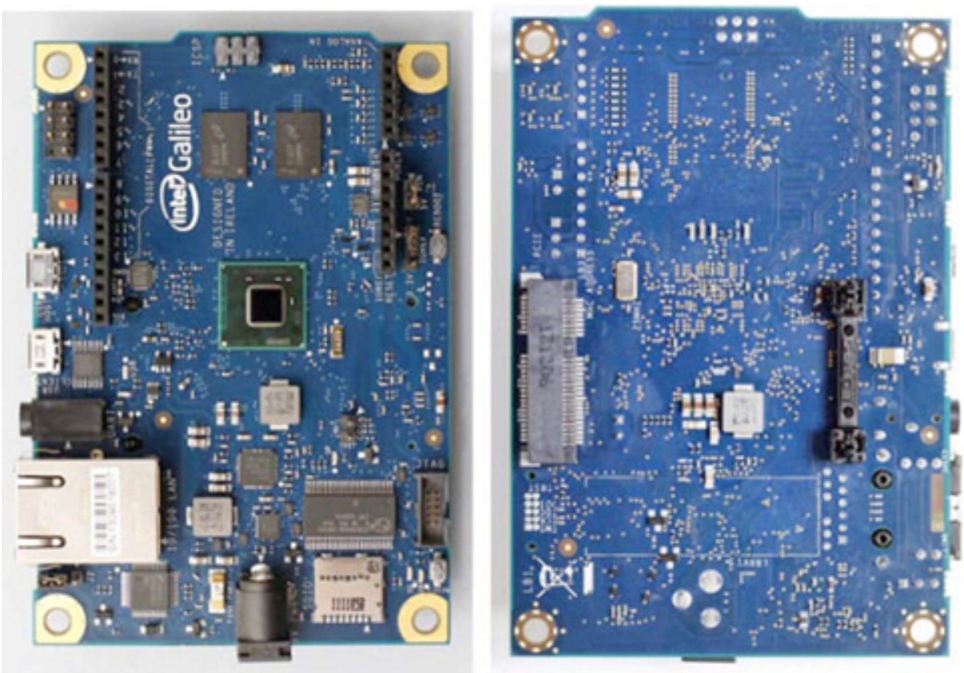


Figure 4.14 Intel Galileo microcontroller

Galileo is a microcontroller board based on the Intel® Quark SoC X1000 Application Processor, a 32-bit Intel Pentium-class system on a chip. It's the first board based on Intel® architecture designed to be hardware and software pin-compatible with Arduino shields designed for the Uno R3. Digital pins 0 to 13 (and the adjacent AREF and GND pins), Analog inputs 0 to 5, the power header, ICSP header, and the UART port pins (0 and 1), are all in the same locations as on the Arduino Uno R3. This is also known as the Arduino 1.0 pin out.

Thereby we achieve solve problems of software installation in microcontroller, is important to remake Intel Galileo is new microcontroller in market that support programs for windows and it help to use of many of this programs, on the other hand the cost of this microcontroller is slightly higher than raspberry pi, but all of uses we can do with this model is rewarded.

Chapter 5

INDEPENDENT SPACE ACCESS

In this chapter we have two main objectives, the first of one to provide an available, reliable and competitive independent launch service at the end of this action. And to enable a key advance in low-cost disruptive technologies to achieve a quick and frequent access to Space adapted to market demands that are not covered by current launcher platforms.

This action refers Independent access to Space, a consistent approach that will fulfill the market demands over classical systems, demonstrating that this is an affordable and reliable launcher trained to benefit from the wide spectrum of European research and technological development (RTD) community needs, from sub-orbital to orbital injection.

We are in line with disruptive advances over current technologies and functionalities that must be proven and assessed in terms economic end-to-end viability. Key advances to achieve a quick and frequent access to space will be prioritized.

5.1. Low-cost mini-launchers

The use of low-cost mini-launchers will enable a new space access market through a massive use of rockoons (balloon and rocket) that injects small satellites into a low Earth orbit. These satellites should have less than 100 grams to ensure they last for few weeks before being burned in the re-entry, reducing the space junk problem.

Using balloon-based launches saves 80% of propellant mass [12] and makes the launch phase much safer; overcoming the barrier that ground-based launch regulations impose. To allow a smart, green and integrated transport, it is essential to reduce this environmental impact through targeted technological improvement, bearing in mind that each way of transport faces variable challenges.

5.2. Space access opportunities

On the other hand this method helps to launch from any part of the Earth and in any time that is necessary, also with this method it is easy to repeat the launching process. This capacity of the launch repeatability makes it free from limitations of the classic launch systems, such a minimum of two years waiting time between launches.

Furthermore, cost of the each classical launching methods is about 200 millions € (Arian 5). This price is very high and out of range for the project. This low cost method helps to deploy very fast and easily the constellation and replace each on at any given time from anywhere in the world.

5.3. Satellite operation and telecommand

The Low Earth Orbits (LEO) and low cost launch system permit an easy and very automated operation in relation with the NEOs detection systems. In consequence control center dependency is greatly reduced. One of the characteristics of the system is that it needs very low bandwidth for communication. It can use commercial bandwidth as that receiver and transmitted information does not need to be encrypted.

This system is like a service of common interest, and it can be open source, such as weather forecasts. Nevertheless although satellites are autonomous, they permit to receive new orders from Earth and these ones should be encrypted. Finally operation and telecommand of the satellites helps NASA or ESA, for detect early presence of new objects and determine their coordinates, this coordinate is useful for space observatories around the world to focus with their telescope, as this telescopes has very good resolution and computing system to calculate orbit and velocity of the NEO detected by AliNEO satellites.

Chapter 6

IMPLEMENTATION AND VALIDATION

In this chapter we try to implement as much as possible subsystems of the satellite between the limitations in time and budget. We will use the 3D printer technology for making structure of satellite and will try to capture images from celestial sphere objects. Also we validate these subsystems in realistic conditions.

6.1. 3D Printer technology

The 3D printer is newest technology in market at present. This technology permit to everybody brings to reality something designed from drawings. The 3D printer technology starts from CAD models rendering in computer programs and make this objects in 3 dimension using plastics, polymers, composite materials, ceramics, titanium alloy and etc. principal concept for 3D printing consist in apply additive process and in which successive, layers of material are laid down, over base and controlled by computer programs.

So that nowadays is very easy to make an idea, with 3D printing, thus we will use this technology to make structure of satellite designed by CAD, more specifically part of satellite showed in Figure 6.1 that contain different subsystems of satellite.

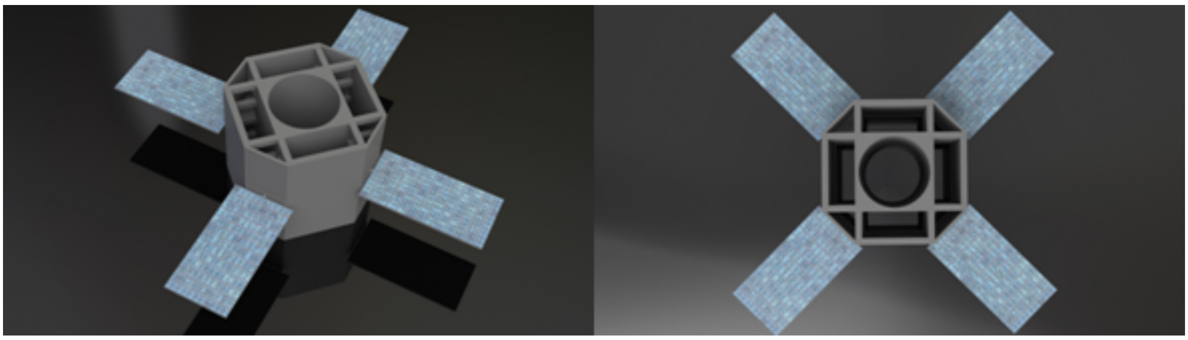


Figure 6.1 Satellite prototype results

The 3D Figure 6.2 printer we will try to make structure of satellite is homemade printer by “AITOR” he collaborated with different projects of UPC-EETAC, and during this project I proposed to him, to make principal structure of satellite with 3D printer. Thus the 3D printer used in this Project has been designed and built from scratch.

This project began in 2011 when the only printers available in the market, at a reasonable price, were the *RepRap*® and *Makerbot*® printers. The idea was using only the extruding concept from the available printers, in order to take advantage of the commercial parts such as the nozzles, heaters, filament, etc.

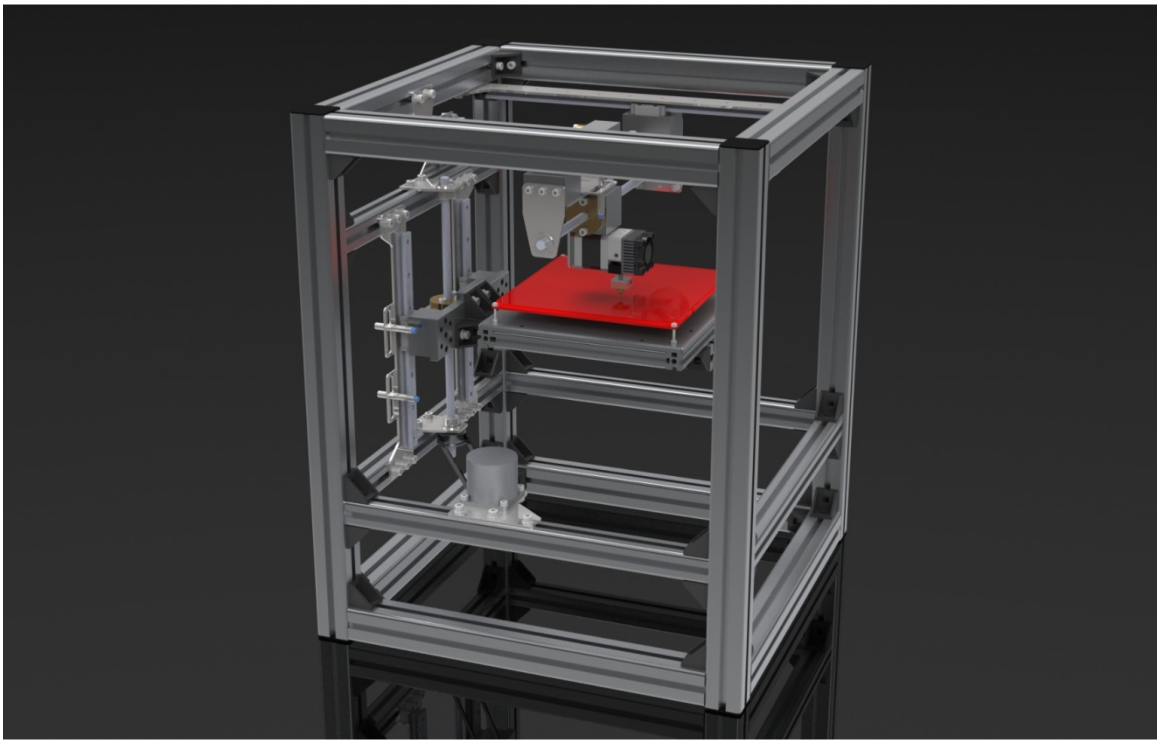


Figure 6.2 Homemade 3D printer

The mechanical design was based in the *CNC* router concept. All axes, except for the extruder, would be driven by screws in order to achieve better accuracy and torque (in comparison with the belt driven 3D printers). High precision ball-screws were considered as an option, although the prices were quite high at that moment, reason why it was decided to use metric screws. Linear bearings were used in order to avoid play between the axes and achieve good tolerances.

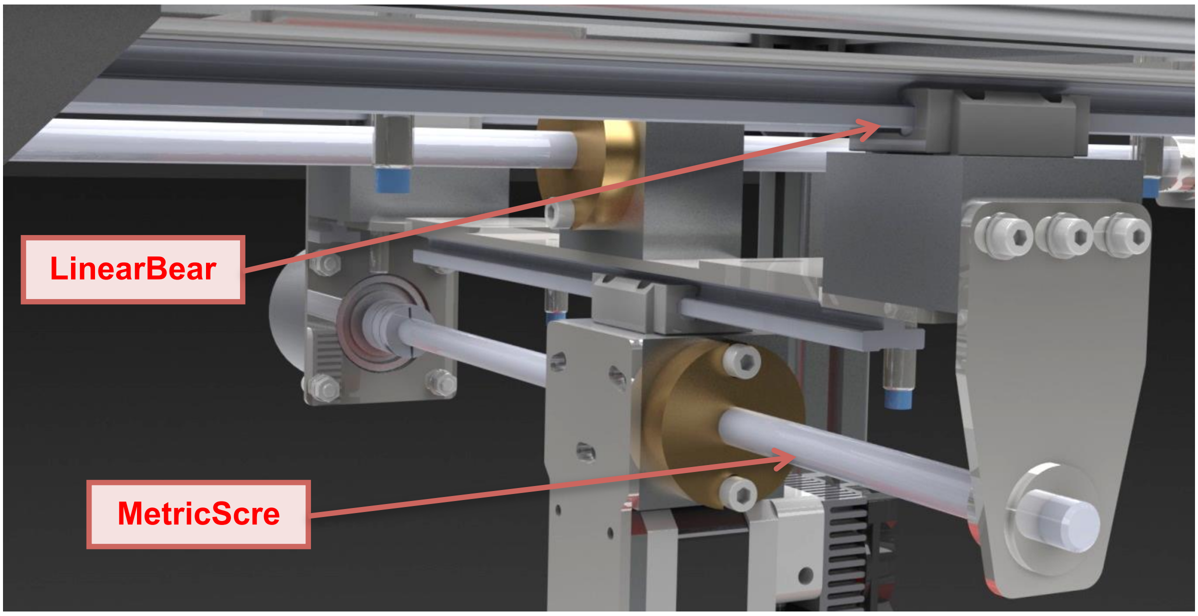


Figure 6.3 Details of 3D printer

In order to make the printer easy to set-up and upgrade, the frame was built using extruded aluminum profiles. It would give a high degree of flexibility in future modifications.
The whole project was designed with *Solidworks*. This allowed us to identify mistakes and interferences before manufacturing the parts, which resulted in time and money savings. The manufacturing of all the parts was done at local workshops. Manufacturing drawings were created for the aluminum parts to be milled/machined. All *linear rail holders* were made of laser cut stainless steel. For that purpose, *Autocad* drawings were created, in order to be loaded into the laser cutter equipment.

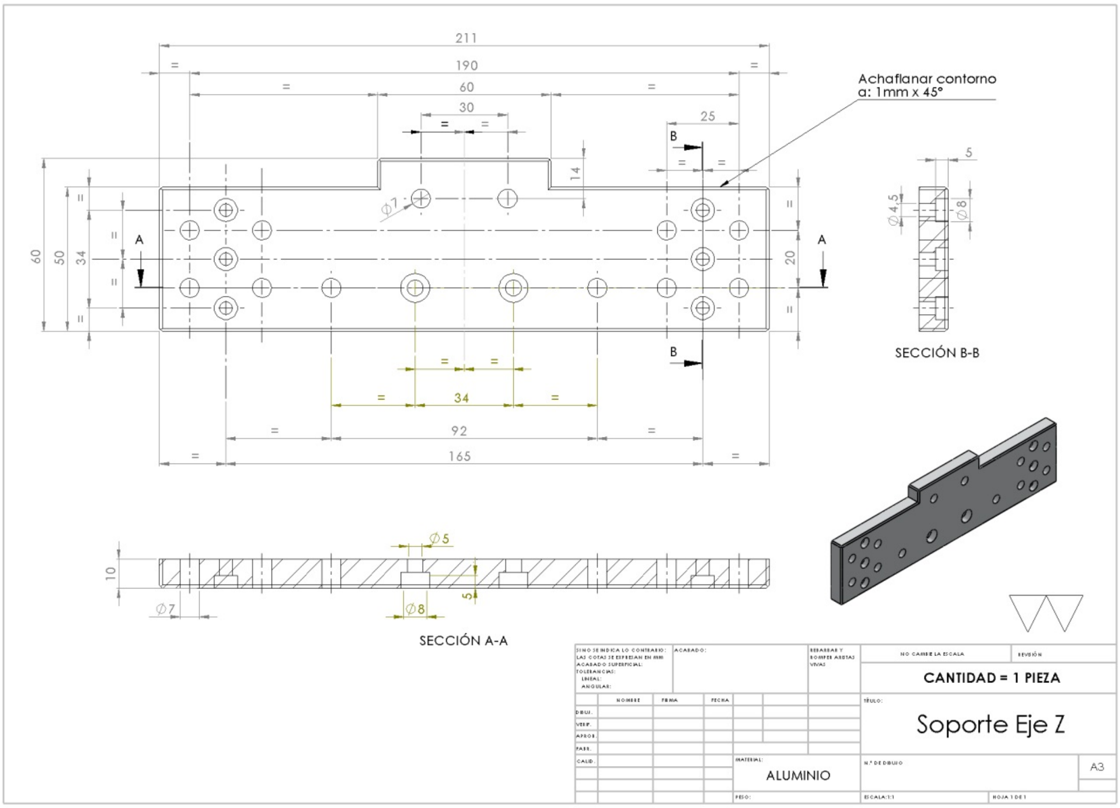


Figure 6.4 Example of a manufacturing drawing used in this project

The control of the printer is based in *Arduino*. From the different options available in the market, the most popular one was The Ramps V1.4 electronic board (so called: "*mother board*"). This board is assembled on top of the *Arduino Mega* and includes connecting ports for all peripherals.

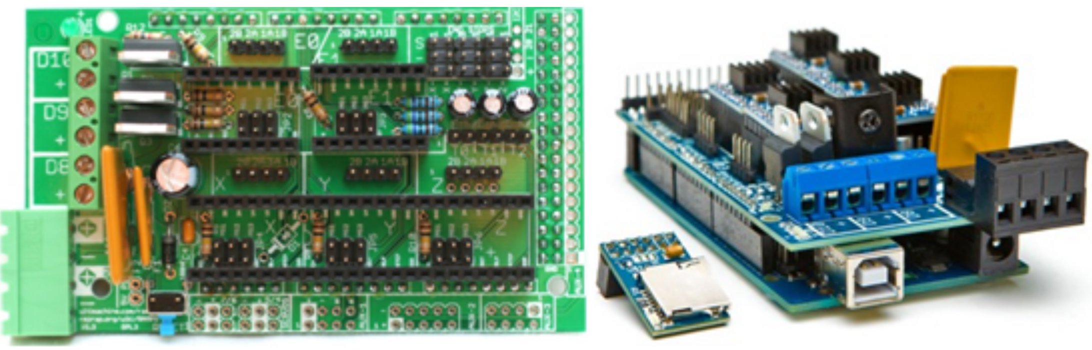


Figure 6.5 Electronic motherboard components

The Ramps V1.4 board is able to feed the necessary power (up to 10A) to the heating elements, such us the extruder hot end and the heated bed. An additional 12V power supply is required to supply the elements.
The *Arduino* board requires drivers to run the stepper motors. In this project, the selected stepper motor drivers were the *Pololu* drivers. One driver for each axis is required (4 drivers in total).

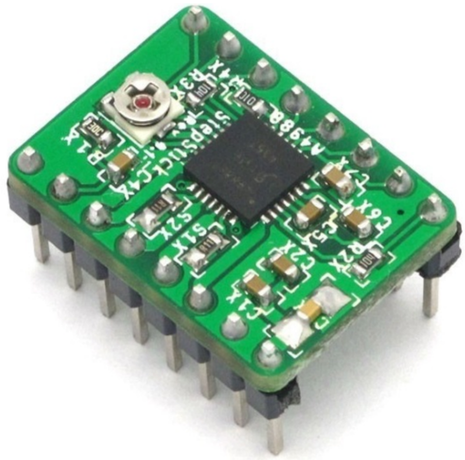


Figure 6.6 Electronic circuits components

The positioning of all axes is done by means of counting the steps (open-loop control). There are no encoders to close the control loop. This requires a good set-up of the firmware and printing parameters, otherwise it is easy to miss/skip steps that result in low accuracy printing.
The rest of the controls involve: temperature control of the extruder hot end and the heated bed. The hot end is responsible of melting the filament and the heated bed is responsible of sticking the printing on the bed surface. The temperature control use thermistor to check the temperature, therefore this is a closed loop control.

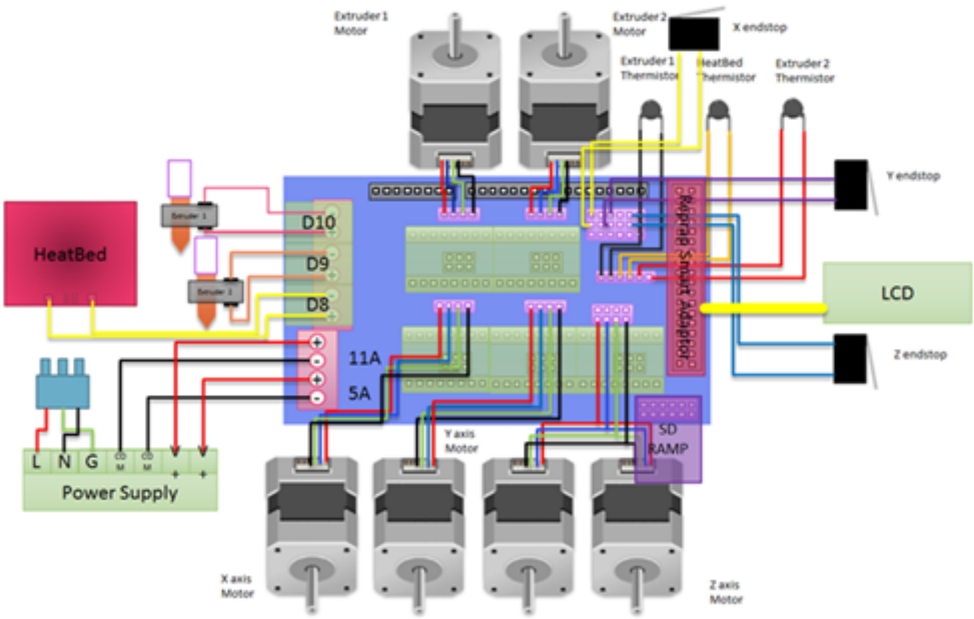


Figure 6.7 Sketches of the closed loop control

It is necessary to install a firmware into the *Arduino* board, in order to make it work. There are two main options available for free. The most used one is called “Sprinter”. The firmware is the core of the printer. It allows the user to configure the printer, to communicate the printer with a PC, to understand the necessary information to print, etc.
In order to operate the printer it is necessary to install software for that purpose. There are many options on internet, but the most popular ones are: *Repetier* and *Pronterface*. In our case we will use *Pronterface*.

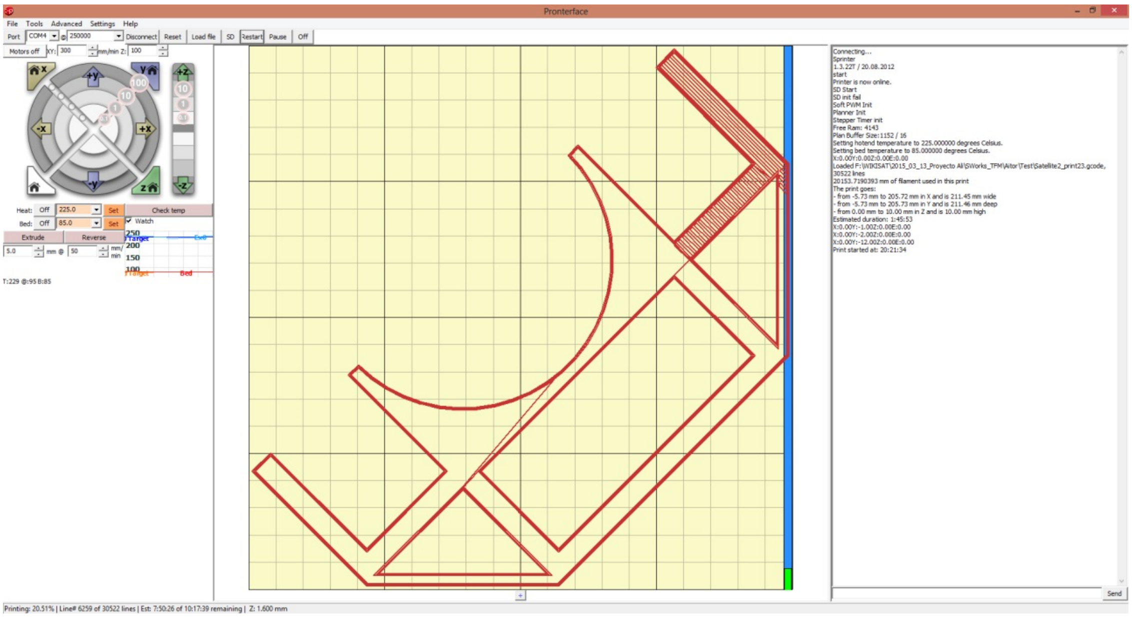


Figure 6.8 Information panel of 3D printer

A “slicer tool” is required to convert a STL file into a sequence of instructions (G-Code). This program is where all the printing parameters such us speed, temperature, type of layering, etc. is configured. This parameters are embedded into the G-Code and generate the printing.

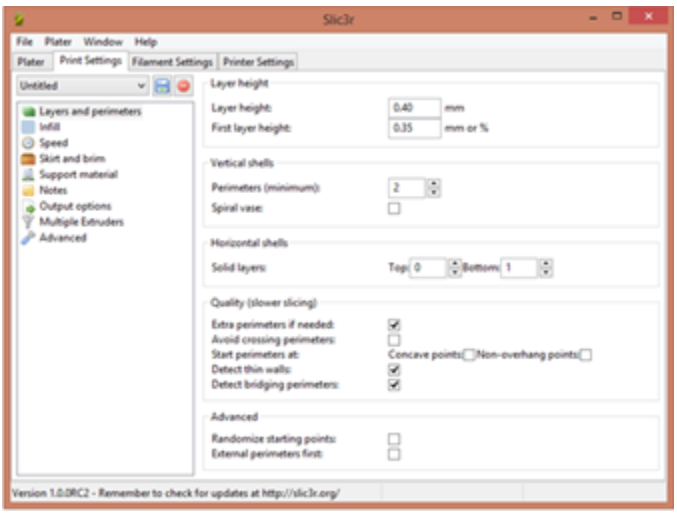


Figure 6.9 Print setting of 3D printer

The working flow diagram is as follows:

- Creating a 3d model using any of the available 3D modelling software (in our case Solidworks).
- Once the model is created, it is required to save it in STL format.
- Open a "Slicer" software and load the STL file.
- Generate the G-Code from the STL format.
- Open the Printing software and load the G-Code and push print.

Finally we start to print the principal structure of satellite in accordance of CAD model designed by SolidWorks.

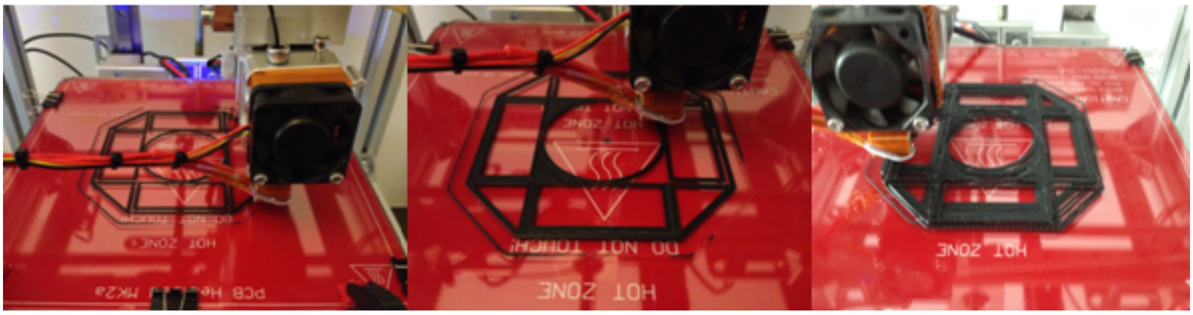


Figure 6.10 Satellite prototype printing process with error

For satellite structure manufacturing we use ABS material and the 3D printer starts printing different parts of satellite from CAD design. First results of printing were erroneous because of jumping steps in each layer by motors of "x" and "y" axis. The result of error it was that printed objects its go to stagger in "z" axis, the Figure 6.11 show the final result of printing with error. Thus looking for source of error, we found that the error it was caused by sum of different errors such as motor drivers is not adjusted good, the motors turned back very hot and it causes to restart it by drivers.

On the other hand some components such as pulley and elastic coupling there are floppy and causes slide in spindle of the printer. So after 8 and 9 attempts and corrections we achieve good and precise printing method. Final result is showed in Figure 6.11.

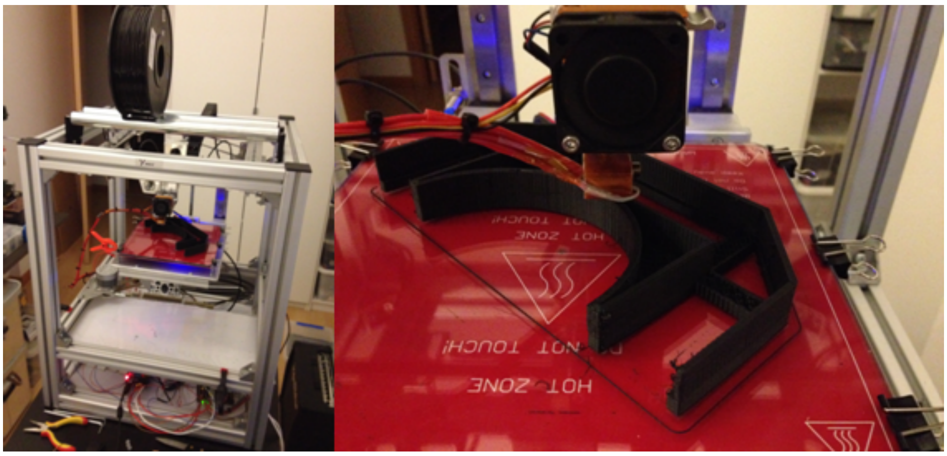


Figure 6.11 Satellite shape 3D printing final result

6.2. Satellite casting matrix process

After different intents for printing the shapes of the satellite, and take into account time of production we arrive to conclusion is better and easier to create shapes of satellite use casting matrix process.

Figure 6.12 Material used for casting satellite shapes

In consequence of lack of time we decided to use one of the good printed shapes for casting matrix showed in figure 6.12, this method help us to make faster all of necessary shapes for the satellite and during this method we use Silcast 517, and other products showed en figure 6.13 for making the mold.

Figure 6.13 Mold making process

How is possible to watch in figure 6.14 after mix different product and throw silicon over the original shape, we stop to harden and finally we have our casting matrix.

Figure 6.14 Casting of satellite shapes

Finally we produce necessary shapes for satellite structure by means of casting process, figure 6.15, this process help to reducing time of production and help to use Polyurethane such as principal material of the shapes. Polyurethane as mentioned in section four, has very good characteristics to protect satellite, furthermore, this process gives us freedom to try with different in future works.

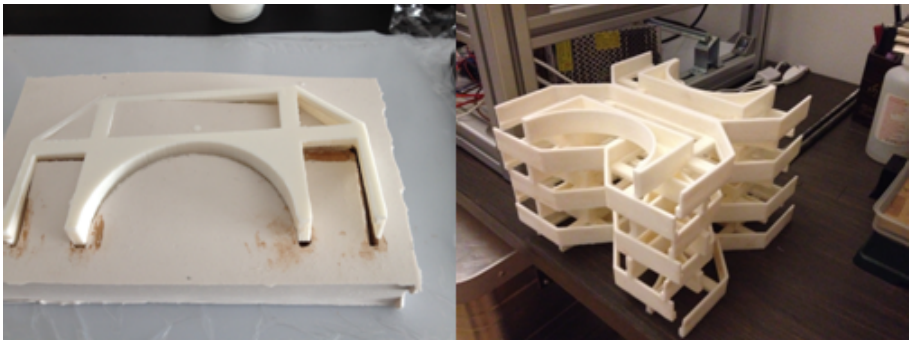


Figure 6.15 Serial production of shapes

6.3. Satellite manufacturing

Satellite manufacturing starts with pasting different shapes of satellite according to CAD design, and showed in annexes. The satellite is composed by two kind of shapes each one consist in keep the original design and other one with some eliminated part, in figure 6.16 we can see the differences of each one, and total mass.

Figure 6.16 Mass details of different shape structure

Thus when we have necessary prepared shapes for satellite the process of pasting starts, and in laboratory we achieve necessary behavior and tools for finishing the pasting process showed in figure 6.17.

Figure 6.17 Pasting process of different shapes

During this process we find some problems with attachment of structure body and it was necessary some reorientation and redesign of structure of satellite, but finally we achieve correct attachment of satellite structure with telescope, finally we solve this setbacks and other details. In figure 6.18 is possible to observe this process and final result, model of satellite is finished.

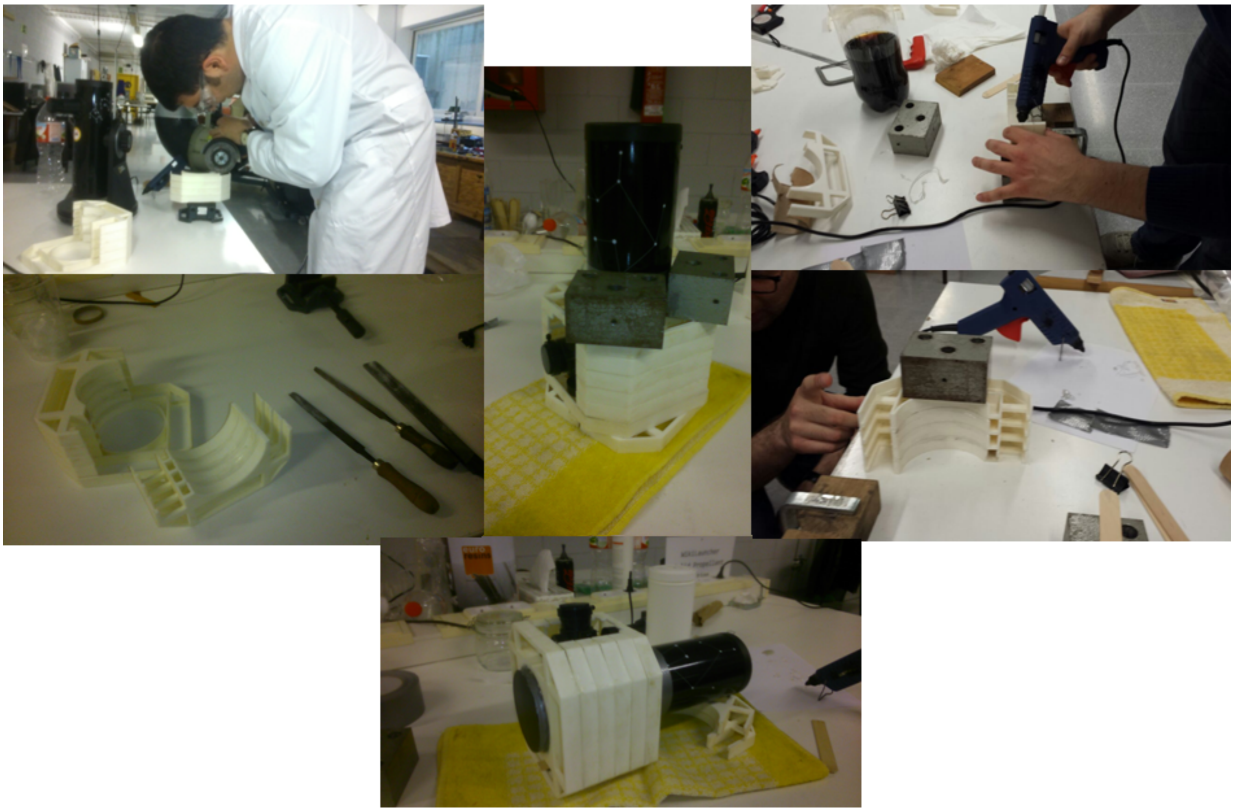


Figure 6.18 process and final results of pasting

6.4. Satellite space qualification

For space qualification we need to prove the satellite is capable to detect and record images in space, but firstly we do filed test for sensors and subsystems recording images from low distance like figure 6.19 in this case record distance was 1.5 km in cloudy day.

Figure 6.19 First field test, 1.5km

Thus in second field test we increase distance of test to 5 km, showed in figure 6.20 and we observe the capacity of record images of satellite it keeps in use condition, between first and second filed test we observe the limited distance result very good for performance of satellite, but it is necessary to improve distance to very large distance like AU, to test subsystems.

Figure 6.20 Second field test 5km

Because of the lack of time inside this thesis we have got not enough neither time nor resources to qualify one of our satellite prototypes in space environment. For this reason we decided to simulate the process of recording a NEOs from the Earth surface to validate the subsystem. For validation of system, we propose to take a image using telescope from Earth surface focusing to Jupiter, and observe the result knowing the distance from Earth to Jupiter is 4,5AU this distance helps us to gauge our systems.

Thus with this prove showed in figure 6.21 we can prove our system is capable to record and detect space objects from very far distance.

Figure 6.21 Jupiter detection by the satellite camera

For recording images we use QHY5CMOS camera, this camera is designed on 2006. By successes using the MT9M001 1.3 mega pixel ½ inch CMOS sensor it made an easy and low cost way for the auto guide working. The model QHY5 mono color has maximum resolution of 1280x1024.

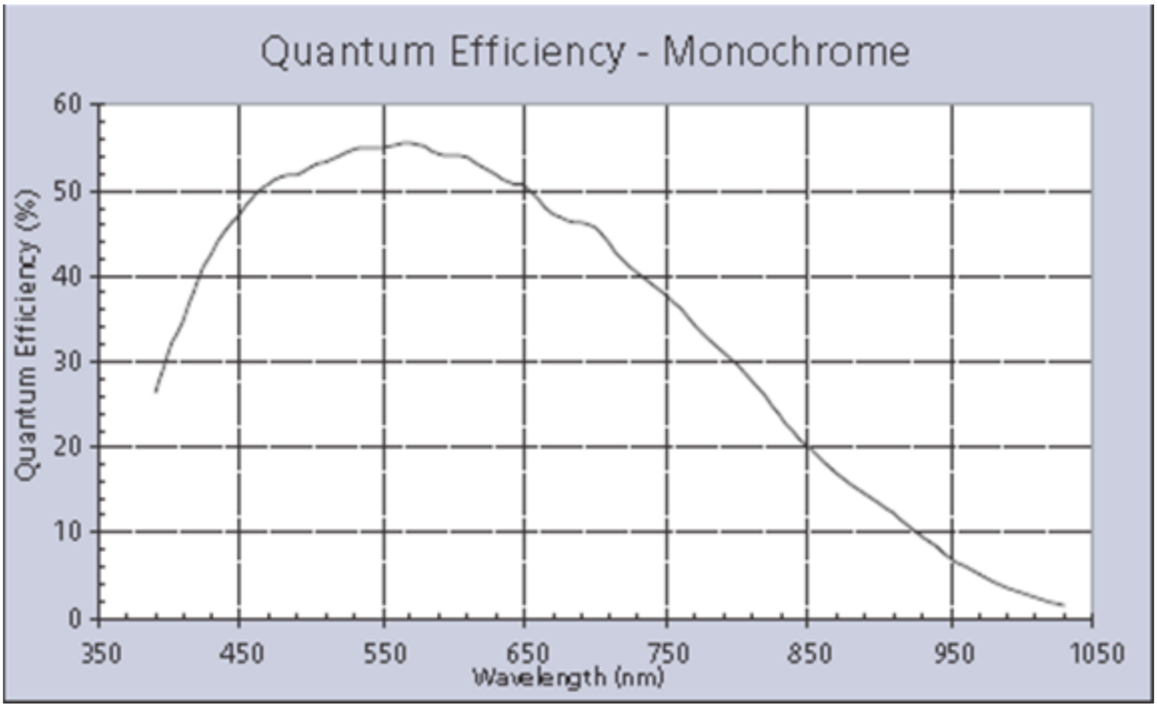


Figure 6.22 Graphic of Wavelength

In figure 6.22 we observe quantum efficiency of QHY5 and wavelength of different lights, like green (550 nm) and infrared (750 nm) enter in there rang of capacity of work.
On the other hand we observe in figure 6.23 an example of this camera and final result of recording images from Jupiter.

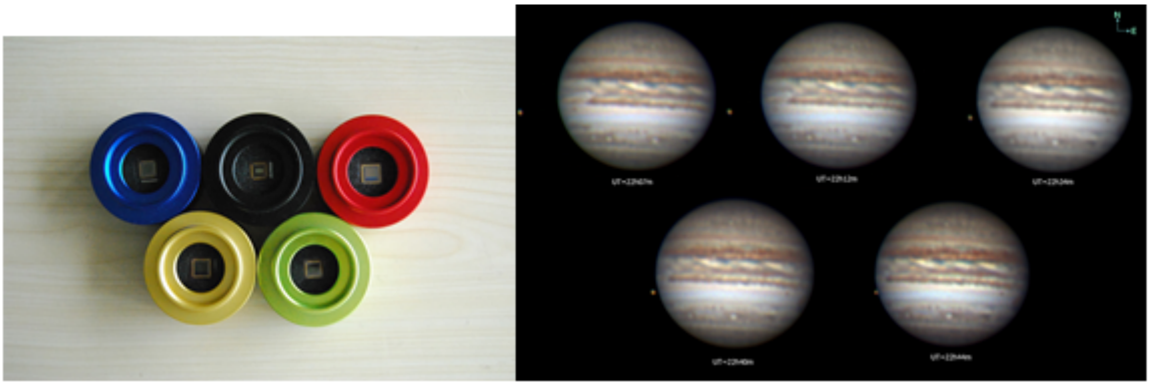


Figure 6.23 HQY5 Camera with Jupiter images

During field test we observe the images are not clear, depending of the focusing of telescope we observe problem with collimating of telescope. Before any observation of space all telescopes needs to put in accurate line of mirrors. If the telescope is out of collimating, it means that the mirror are not aligned to the optical axis and this produce deformed images. One way to see if the telescope is out, we need to pointing to a star and defocus the telescope, to blur the image, the star becomes a “donut”. When the donut is perfectly concentric, it means that the telescope is perfectly collimated. This should occur whether sub-focus as if on-focus images.

In figure 6.21 we can observe light out of collimate in telescope, for this we act to collimate the telescope as we can observe in figure 6.24.

Figure 6.24 Process of collimating

Even so we observed in figure 6.25 the problem of collimating endure in telescope, only we are capable to achieve some good improvement recording images.

Figure 6.25 Jupiter from Earth with some errors

The software used in recording images with HQY5 camera is Astronomical Image Capture and real time Preprocessing [27] it is possible to install in Galileo board, and easy to use with open source background.

To sum up, we prove our satellite set systems is capable to detect and record space objects from 1.5 km to Jupiter 4,5AU distance. But we need to detect objects with different diameter from 140 meter to 1000 meter, and is important to prove how many pixels are necessary to detect objects and determine how far it can be, to detect it. Thus we need to use information from HQY5 Camera and find this limitation of detection. So for this process we use formula 6.1 from real data of datasheet of the camera and telescope, we know pixel size is 5,2 microns, focal length of telescope is 350 mm, and number of pixels in X axis is 1284 and in Y axis is 1024.

$$\left(\frac{Pixel_{size}}{Focal_{length}}\right) \times 206.265 = Arc\sec onds \,/\, Pixel \tag{6.1}$$

The result of formula 6.1 is 3.064508571 (is necessary to use maximum of decimals because of next values to use it). Now we are capable to calculate sensor filed of axis X and Y using formula 6.2. Final result is in degree.

$$\left(Arc\sec onds \times N^{o}\ \ Pixel_{x,y}\right)/3600 = Field\ Sensor\ Axis_{x,y} \tag{6.2}$$

Moreover using values of datasheet and final result of formula 6.1 we have values for X and Y axis. These values are 1,089603048º for X axis and 0,871682438º for Y axis.

$$Field\ Sensor\ Axis_{x}/Field\ Sensor\ Axis_{y} = Degree \,/\, Pixel \tag{6.3}$$

To sum up, using formula 6.3 we find the minimum grades/pixel relation for detect in camera pixels, this values is 0,000851252. This value indicates our limit to detect objects in space.

In figure 6.26 and 6.27 we can see from each distance and NEOs size our system is capable to see it and register their movement, and all necessary information.

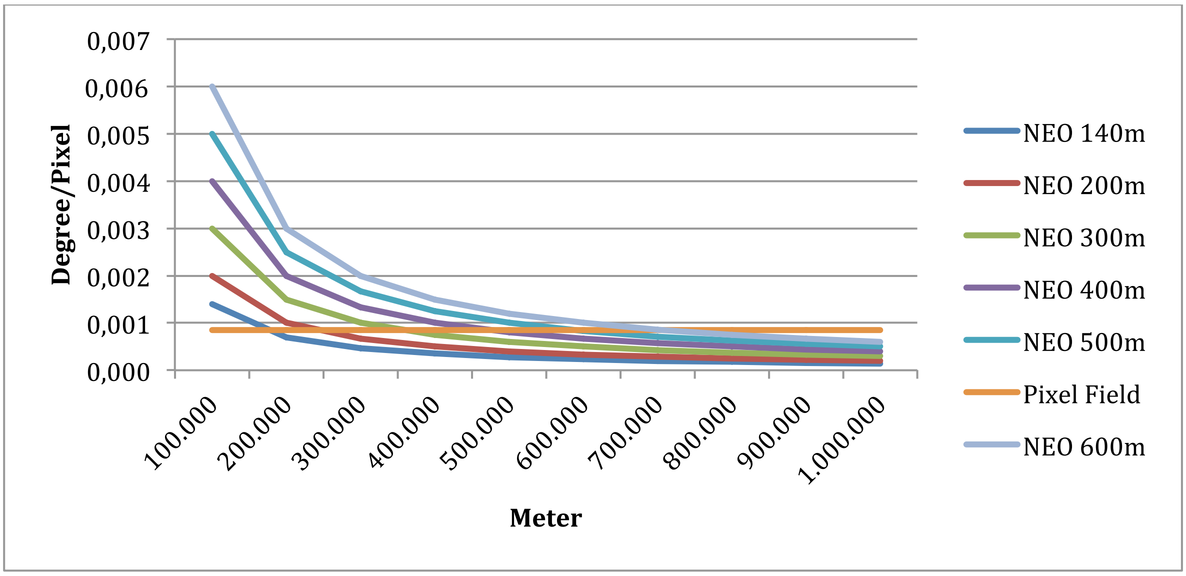


Figure 6.26 Detection rang 1

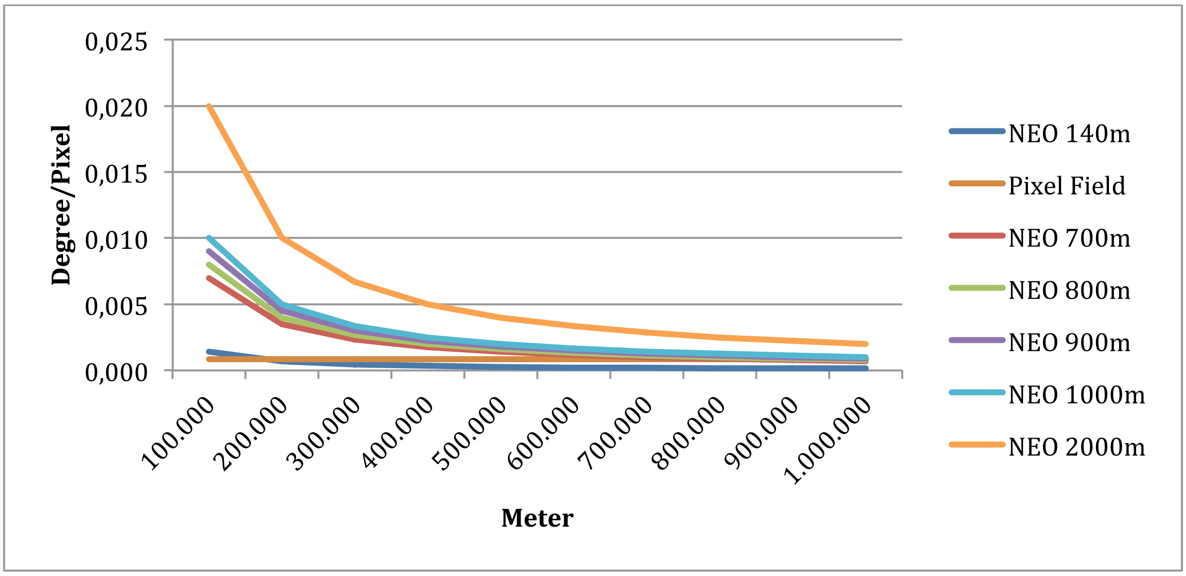


Figure 6.27 Detection rang 2

Focusing to size of NEOs and hazardously of each one for Earth, in the past few decades, astronomers have spotted more than 12,000 of NEOs. But based on the low rate of discovery (how infrequently we see the same objects over and over again), we know there are many more out there. If there were no NEOs left to find, we would see only ones we'd already found; instead, our surveys keep turning up new ones. Relatively small objects dominate the NEO population. The number are a bit uncertain given the incompleteness of our current observations, but it is reasonable to expect that there are millions bigger than about 20 meters, more or less the length of a train car [30].

Even a smaller NEO could create devastating loss of life and property if it were to strike a populated area. A 30-meter-wide asteroid hurtling through space can carry the kinetic equivalent of megatons of TNT, on the order of a hundred or more time the energy contained in the bomb dropped on Hiroshima.

Figure 6.28 ALINEO in action

6.5. Satellite in-flight validation

By a space qualification test is possible to gain a technological readiness level of about TRL6.

In order to validate the satellite inside a real environment, it is mandatory to test one of these new satellites in a real flight in example through a piggy-back opportunity available by a real ESA launcher. In this case the cost will be about 300,000 Euros and two years of development. There are other options to achieve a real test like the low-cost launchers that are able to send a payload of 50 kg to a Low Earth Orbit of about 250 km of altitude. The problem with this other case is that those launching platforms are still in development and currently are not an option.

By a real mission such this; it will be possible to increase the technological readiness level at least up to TRL7.

Chapter 7

CONCLUSIONS

This section explains about general conclusions of this thesis and different points of view about how apply technologies to make a system to detect NEOs and expected result of detection on time hazardous NEOs. Moreover we saw the possibility of the future works from this thesis for next users and possibility of technological concept improve and finally explain about the environmental impact of the NEOs detection system.

7.1. General conclusions

Firstly during this Master thesis I achieve to complete my first objective, it was, propose a system capable to detect NEOs, with autonomy and free status from actual structures. As a consequence I needed much information from my bachelor degree and Master, and it was a pleasure to use it to resolve my dudes and other problems I found during this project. Furthermore, in this thesis we propose a principal sketch about a system based on satellite, for detection on time Near Earth Objects, in Earth impact direction.

Secondly we saw and defined high and low level requirements for asteroids and comets detection by satellites in LEO then we designed the subsystems as a result of these requirements. We designed a satellite prototype and make it for qualification, and we observed that is important use of composite materials to make good radiation protection. On the other hand we observed our EPS work near limit of temperature requirements, but inside of acceptable values. During this master thesis we find some problem with programs to install it inside of onboard computer, but finally we find solution.

The NEOs Space Observatory proposed in this thesis need more focusing to resolve some problems funded during process of qualification, such as collimator problems of Newtonian telescopes. We implemented a model of the satellite where COTS components were assembled by manufacturing 3D printer technology and validating the proposed satellite platform.

Finally we performed a simulation of NEOs detection by mounting the satellite platform in a TRIPODE and recording a picture from Jupiter. This test validated the satellite subsystem achieving the main goal of this thesis.

During this final thesis use of the tools such as SolidWorks and 3D printer has accelerated the process of design, implementation and validation of the subsystems. Moreover, it helped to prove these subsystems to detect hazardous NEOs for Earth, and permit Earth to endow a system for early warning about NEOs. Thus Earth achieves a system to alarm about asteroids and comets that actually any agency has a system with reliability, accuracy and safety to detect hazardous Near Earth Objects.

7.2. Future work

About possible future works and secondary objectives, as it were impossible to achieve during this thesis, due to lack of time and budget. Some of these objectives are different subsystems that are usable in little satellites, and real prove in real flight of rockets to achieve information about real comportment of the satellite and their subsystems in the space. Real work of the thermal and electric and link budget is necessary to considerer satellite workable.

Use of this system with different model of telescopes to improve distance of detection, and it is usable for space debris detection and finally to use it for Earth observatory. For real launch of satellite is necessary to considerer some existing lunch system like Piggyback payload in different space agency launch systems or equivalent systems. And it is necessary to fundraiser necessary budget for real launch.

7.3. Environmental impact

In the first place the System of detection of Near Earth Objects Based on satellite, represent good advantages respects other systems, and more important of them is avoiding light contamination for space observation and in the same time accomplish reduce a lot of the environmental impact comparing with other methods, such as politics of zero space debris, low pollution and very low risk to the pollution.

Using this method we will save energy and pollution in front of the use large satellites, this system is based in constellation and when is necessary it is lunched to replace the failure or lost satellites. The failed satellites are burned in the atmosphere in short time so there is no space debris caused by these satellites. The MEMS components are typically manufactured without lead to not contaminate and the factories follow processes that respect the environment.

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Annexes. EPS Sketches

TEXAS INSTRUMENTS

WEBENCH ® Design Report

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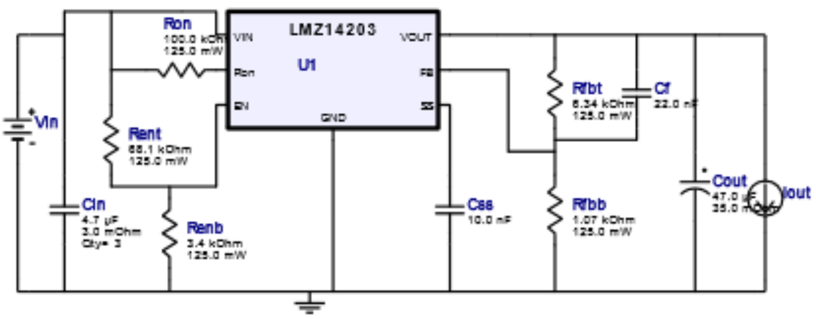
VinMax = 30.0V
Vout = 5.5V
Iout = 2.0A

Device = LMZ14203TZ-ADJ/NOPB
Topology = Buck
Created = 5/16/15 11:42:08 AM
BOM Cost = $10.45
Footprint = 296.0 mm²
BOM Count = 12
Total Pd = 0.84W

Design : 3705031/19 LMZ14203TZ-ADJ/NOPB
LMZ14203TZ-ADJ/NOPB 25.0V-30.0V to 5.50V @ 2.0A

VinMin = 25.0V
VinMax = 30.0V

Vout = 5.5V
Iout = 2.0A



Electrical BOM

#	Name	Manufacturer	Part Number	Properties	Qty	Price	Footprint
1.	Cf	Yageo America	CC0805KRX7R9BB223 Series= X7R	Cap= 22.0 nF VDC= 50.0 V IRMS= 0.0 A	1	$0.01	0805 7 mm²
2.	Cin	MuRata	GRM31CR71H475KA12L Series= X7R	Cap= 4.7 uF ESR= 3.0 mOhm VDC= 50.0 V IRMS= 4.98 A	3	$0.07	1206 11 mm²
3.	Cout	Panasonic	10TPE47MAZB Series= 1281	Cap= 47.0 uF ESR= 35.0 mOhm VDC= 10.0 V IRMS= 1.4 A	1	$0.39	3528-21 17 mm²
4.	Css	MuRata	GRM216R71H103KA01D Series= X7R	Cap= 10.0 nF VDC= 50.0 V IRMS= 0.0 A	1	$0.01	0805 7 mm²
5.	Renb	Panasonic	ERJ-6ENF3401V Series= 225	Res= 3.4 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
6.	Rent	Panasonic	ERJ-6ENF6812V Series= 225	Res= 68.1 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
7.	Rfbb	Panasonic	ERJ-6ENF1071V Series= 225	Res= 1.07 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
8.	Rfbt	Panasonic	ERJ-6ENF6341V Series= 225	Res= 6.34 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
9.	Ron	Panasonic	ERJ-6ENF1003V Series= 225	Res= 100.0 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²

 1 ti.com/webench

WEBENCH® Design Report LMZ14203TZ-ADJ/NOPB : LMZ14203TZ-ADJ/NOPB 25.0V-30.0V to 5.50V @ 2.0A May 16, 2015 11:44:11 GMT-07:00

WEBENCH® Design

#	Name	Manufacturer	Part Number	Properties	Qty	Price	Footprint
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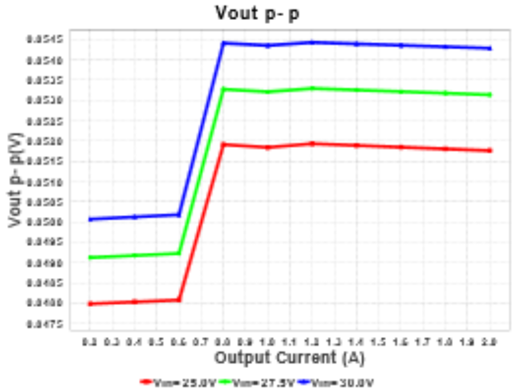

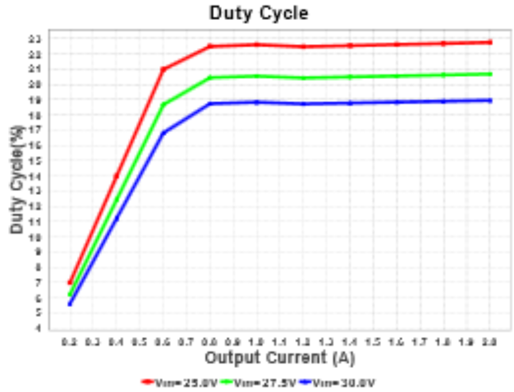

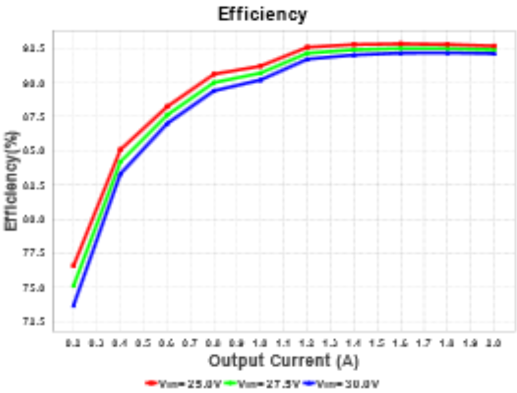

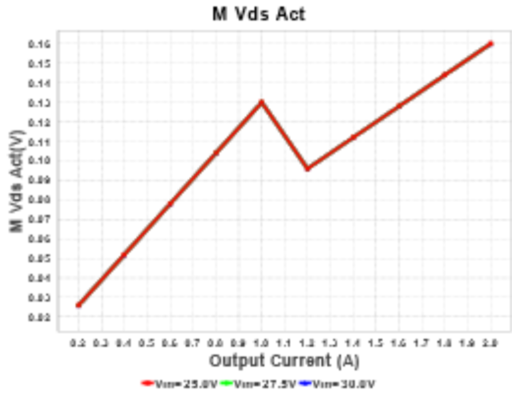

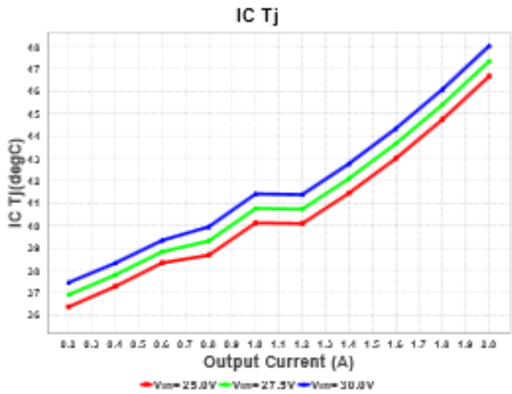

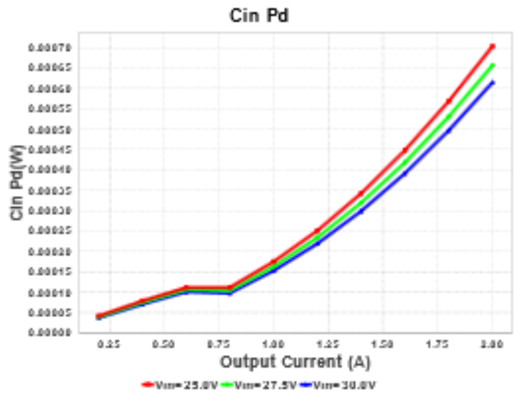

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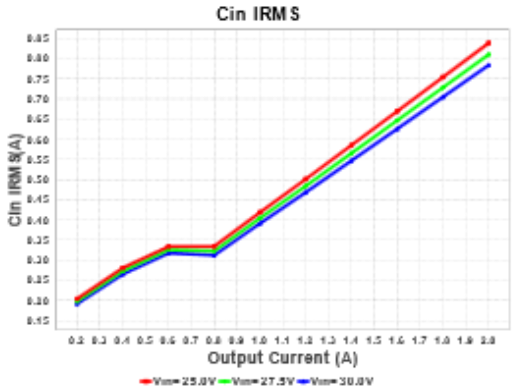
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Cin IRMS(A)
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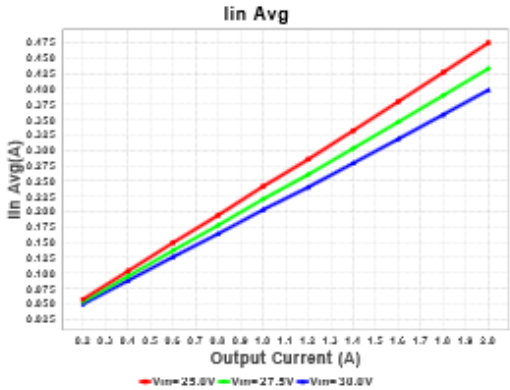
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Iin Avg(A)
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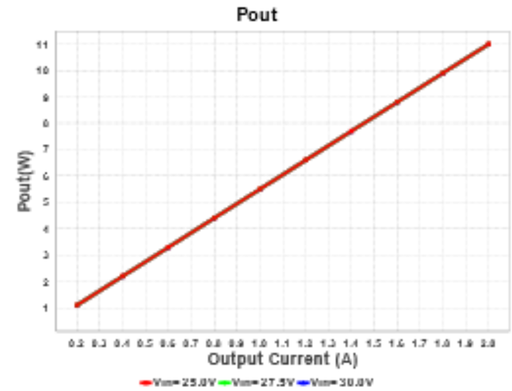
Pout
Pout(W)
Output Current (A)
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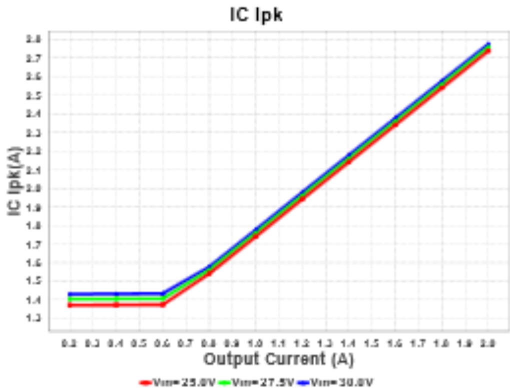
IC Ipk
IC Ipk(A)
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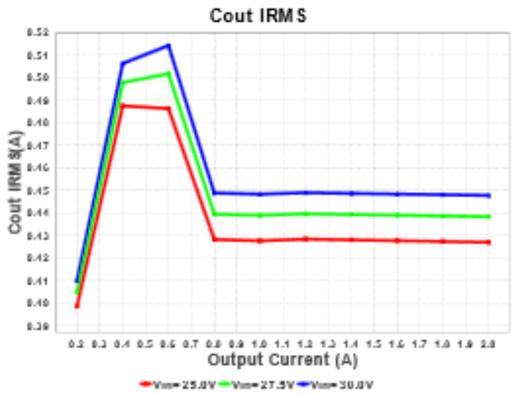
Cout IRMS
Cout IRMS(A)
Output Current (A)
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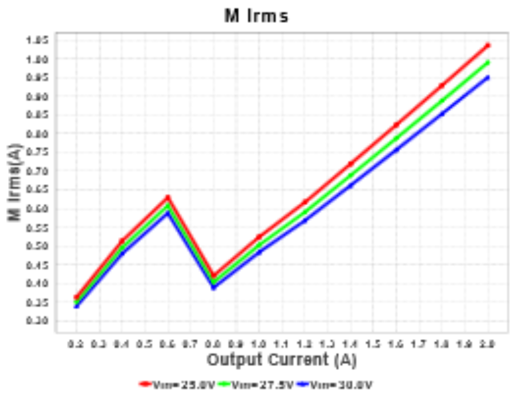
M Irms
M Irms(A)
Output Current (A)
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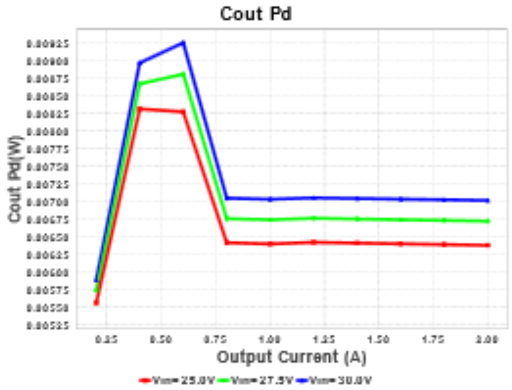

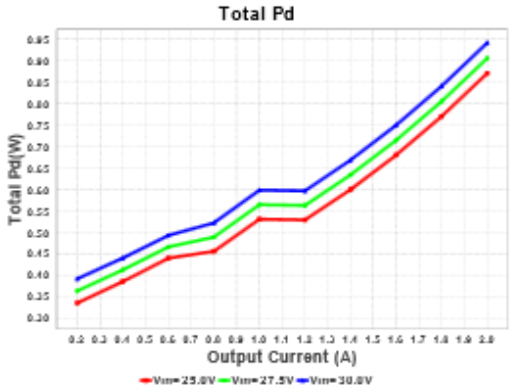

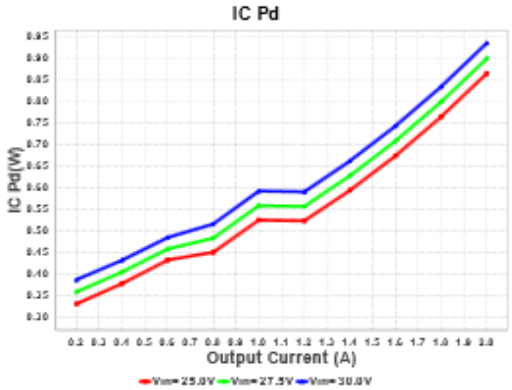

Operating Values

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2.	Cout IRMS	447.758 mA	Current	Output capacitor RMS ripple current
3.	IC Ipk	2.776 A	Current	Peak switch current in IC
4.	Iin Avg	394.55 mA	Current	Average input current
5.	M1 Irms	942.522 mA	Current	Q Iavg
6.	BOM Count	12	General	Total Design BOM count
7.	FootPrint	296.0 mm²	General	Total Foot Print Area of BOM components
8.	Frequency	437.428 kHz	General	Switching frequency
9.	IC Tolerance	20.0 mV	General	IC Feedback Tolerance
10.	M Vds Act	160.0 mV	General	Voltage drop across the MosFET
11.	Pout	11.0 W	General	Total output power
12.	Total BOM	$10.45	General	Total BOM Cost
13.	Vout OP	5.5 V	Op_Point	Operational Output Voltage
14.	Duty Cycle	18.955 %	Op_point	Duty cycle
15.	Efficiency	92.932 %	Op_point	Steady state efficiency
16.	IC Tj	46.0 degC	Op_point	IC junction temperature
17.	ICThetaJA	19.3 degC/W	Op_point	IC junction-to-ambient thermal resistance
18.	IOUT_OP	2.0 A	Op_point	Iout operating point
19.	VIN_OP	30.0 V	Op_point	Vin operating point
20.	Vout p-p	54.288 mV	Op_point	Peak-to-peak output ripple voltage
21.	Cin Pd	614.489 µW	Power	Input capacitor power dissipation
22.	Cout Pd	7.017 mW	Power	Output capacitor power dissipation
23.	IC Pd	829.007 mW	Power	IC power dissipation
24.	Total Pd	836.604 mW	Power	Total Power Dissipation

Design Inputs

#	Name	Value	Description
1.	Iout	2.0	Maximum Output Current
2.	Iout1	2.0	Output Current #1
3.	VinMax	30.0	Maximum input voltage

#	Name	Value	Description
4.	VinMin	25.0	Minimum input voltage
5.	Vout	5.5	Output Voltage
6.	Vout1	5.5	Output Voltage #1
7.	base_pn	LMZ14203	Base Product Number
8.	source	DC	Input Source Type
9.	Ta	30.0	Ambient temperature

Design Assistance

1. **LMZ14203** Product Folder : http://www.ti.com/product/lmz14203 : contains the data sheet and other resources.

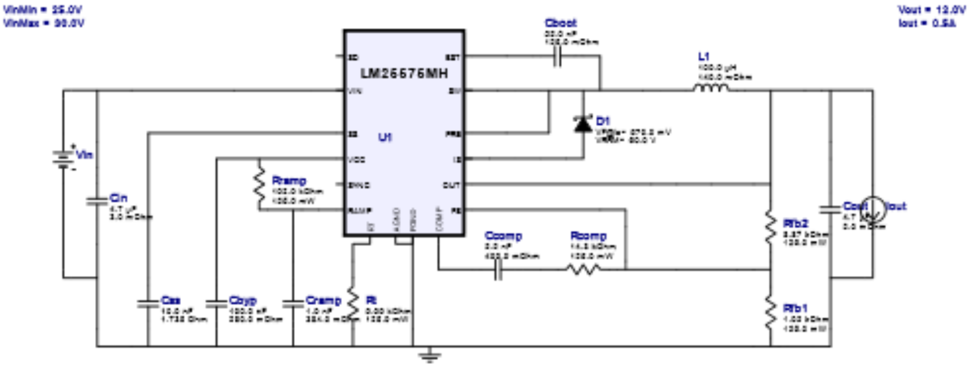
TEXAS INSTRUMENTS

WEBENCH ® Design Report

Design : 3705031/21 LM25575MHX/NOPB
LM25575MHX/NOPB 25.0V-30.0V to 12.00V @ 0.5A

VinMin = 25.0V
VinMax = 30.0V
Vout = 12.0V
Iout = 0.5A

Device = LM25575MHX/NOPB
Topology = Buck
Created = 5/16/15 11:59:53 AM
BOM Cost = $2.38
Footprint = 352.0 mm²
BOM Count = 15
Total Pd = 0.33W



Electrical BOM

#	Name	Manufacturer	Part Number	Properties	Qty	Price	Footprint
1.	Cboot	Kemet	C0805C223K5RACTU Series= X7R	Cap= 22.0 nF ESR= 125.0 mOhm VDC= 50.0 V IRMS= 645.0 mA	1	$0.01	0805 7 mm²
2.	Cbyp	AVX	08053C104KAT2A Series= X7R	Cap= 100.0 nF ESR= 280.0 mOhm VDC= 25.0 V IRMS= 0.0 A	1	$0.01	0805 7 mm²
3.	Ccomp	Kemet	C0805C222K5RACTU Series= X7R	Cap= 2.2 nF ESR= 400.0 mOhm VDC= 50.0 V IRMS= 251.0 mA	1	$0.01	0805 7 mm²
4.	Cin	MuRata	GRM31CR71H475KA12L Series= X7R	Cap= 4.7 uF ESR= 3.0 mOhm VDC= 50.0 V IRMS= 4.98 A	1	$0.07	1206 11 mm²
5.	Cout	MuRata	GRM21BR61E475MA12L Series= X5R	Cap= 4.7 uF ESR= 2.0 mOhm VDC= 25.0 V IRMS= 7.29 A	1	$0.06	0805 7 mm²
6.	Cramp	Kemet	C0805C102K5RACTU Series= X7R	Cap= 1.0 nF ESR= 384.0 mOhm VDC= 50.0 V IRMS= 214.0 mA	1	$0.01	0805 7 mm²
7.	Css	Kemet	C0805C103K5RACTU Series= X7R	Cap= 10.0 nF ESR= 1.739 Ohm VDC= 50.0 V IRMS= 411.0 mA	1	$0.01	0805 7 mm²
8.	D1	NXP Semiconductor	PMEG6010CEH,115	VF@Io= 570.0 mV VRRM= 60.0 V	1	$0.11	SOD-123F 12 mm²

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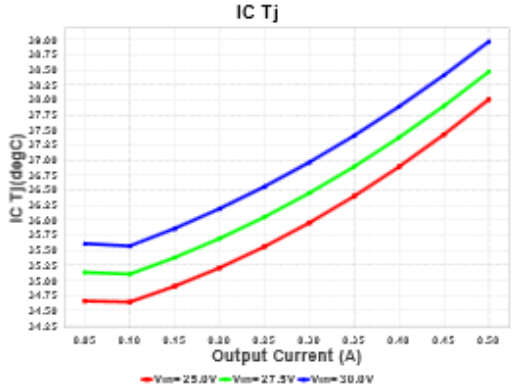
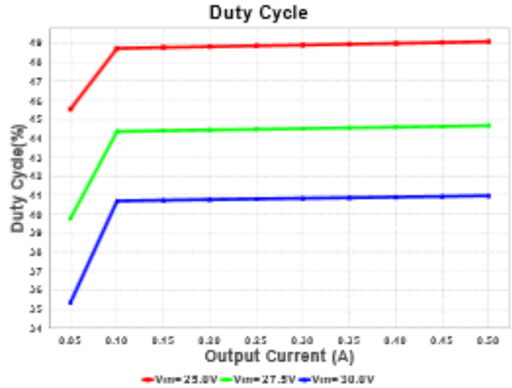
#	Name	Manufacturer	Part Number	Properties	Qty	Price	Footprint
9.	L1	Bourns	SRR1210-101M	L= 100.0 µH DCR= 140.0 mOhm	1	$0.44	 SRR1210 196 mm²
10.	Rcomp	Panasonic	ERJ-6ENF1432V Series= 225	Res= 14.3 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
11.	Rfb1	Panasonic	ERJ-6ENF1021V Series= 225	Res= 1.02 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
12.	Rfb2	Panasonic	ERJ-6ENF8871V Series= 225	Res= 8.87 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
13.	Rramp	Panasonic	ERJ-6ENF1023V Series= 225	Res= 102.0 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
14.	Rt	Panasonic	ERJ-6ENF9091V Series= 225	Res= 9.09 kOhm Power= 125.0 mW Tolerance= 1.0%	1	$0.01	0805 7 mm²
15.	U1	Texas Instruments	LM25575MHX/NOPB	Switcher	1	$1.60	 MXA16A 59 mm²

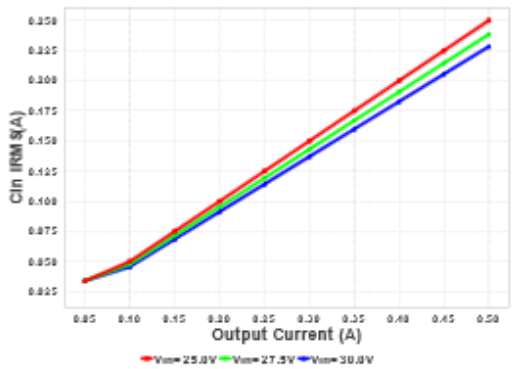
IC Tj



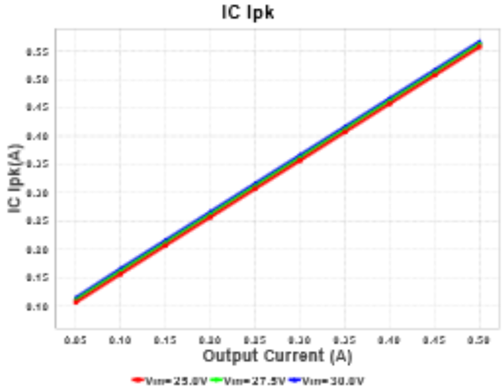
Duty Cycle



Cin IRMS

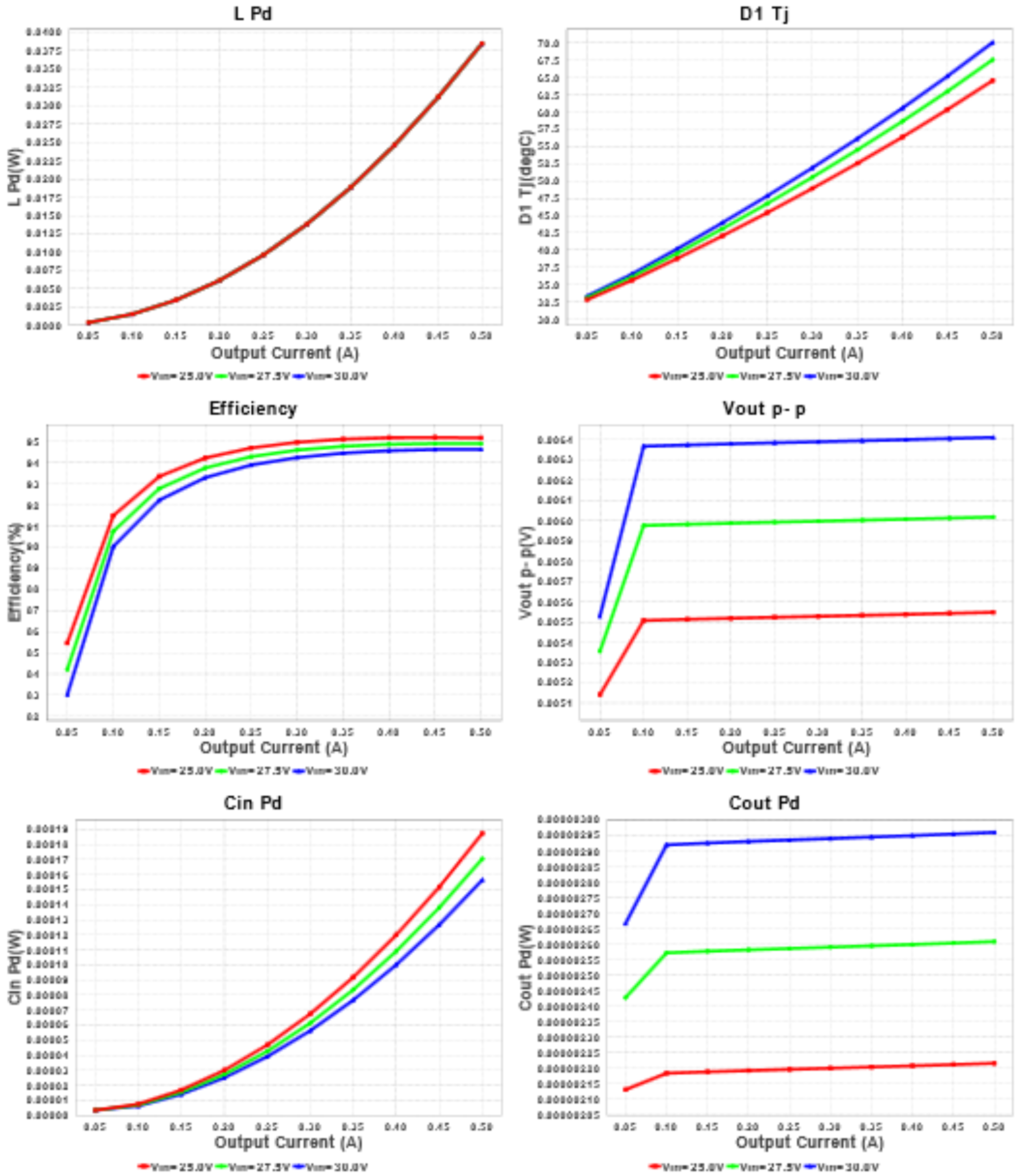


IC Ipk



WEBENCH® Design Report LM25575MHX/NOPB : LM25575MHX/NOPB 25.0V-30.0V to 12.00V @ 0.5A May 16, 2015 12:00:24 GMT-07:00

WEBENCH® Design



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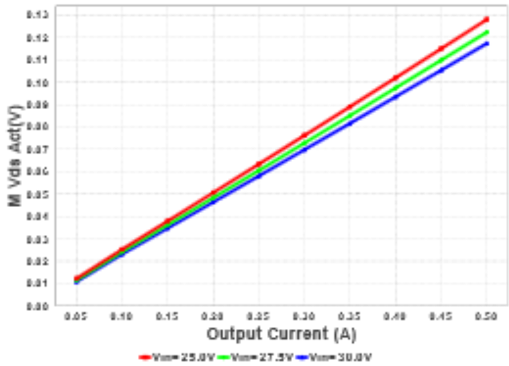
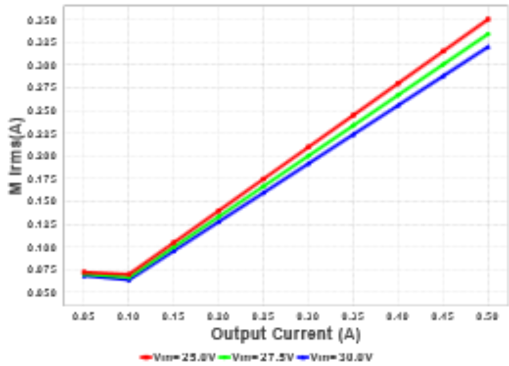
WEBENCH® Design Report LM25575MHX/NOPB : LM25575MHX/NOPB 25.0V-30.0V to 12.00V @ 0.5A May 16, 2015 12:00:24 GMT-07:00

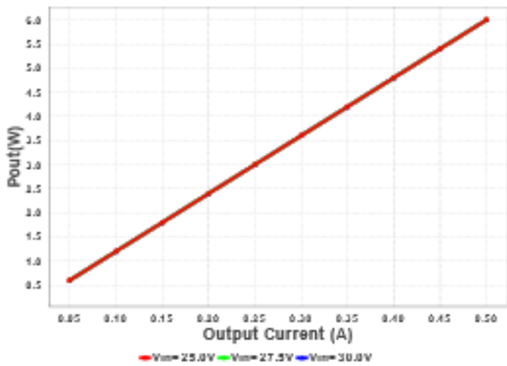
M Vds Act



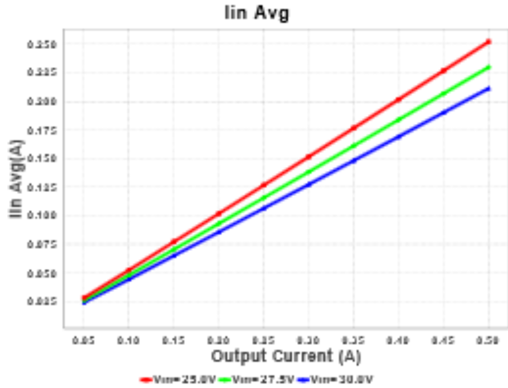
M Irms



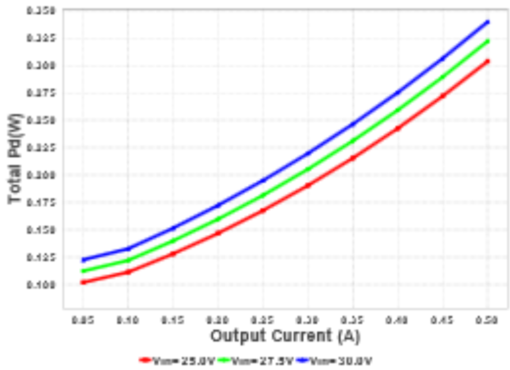
Pout



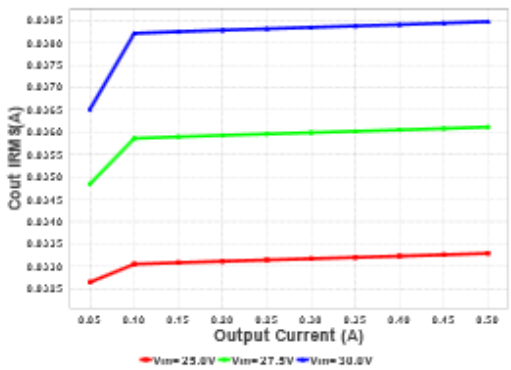
Iin Avg



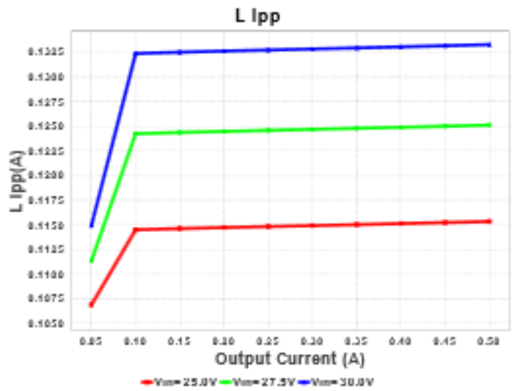
Total Pd

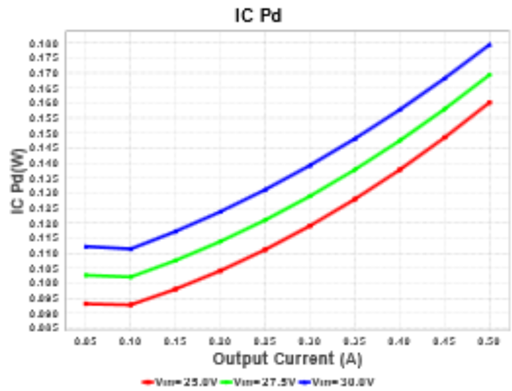


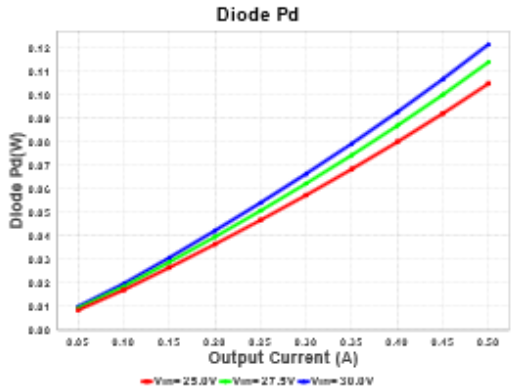
Cout IRMS



WEBENCH® Design







Operating Values

#	Name	Value	Category	Description
1.	Cin IRMS	228.382 mA	Current	Input capacitor RMS ripple current
2.	Cout IRMS	38.436 mA	Current	Output capacitor RMS ripple current
3.	IC Ipk	566.376 mA	Current	Peak switch current in IC
4.	Iin Avg	211.13 mA	Current	Average input current
5.	L Ipp	133.15 mA	Current	Peak-to-peak inductor ripple current
6.	M1 Irms	319.89 mA	Current	Q Iavg
7.	BOM Count	15	General	Total Design BOM count
8.	FootPrint	352.0 mm²	General	Total Foot Print Area of BOM components
9.	Frequency	553.357 kHz	General	Switching frequency
10.	IC Tolerance	18.0 mV	General	IC Feedback Tolerance
11.	M Vds Act	117.324 mV	General	Voltage drop across the MosFET
12.	Pout	6.0 W	General	Total output power
13.	Total BOM	$2.38	General	Total BOM Cost
14.	D1 Tj	68.209 degC	Op_Point	D1 junction temperature
15.	Vout OP	12.0 V	Op_Point	Operational Output Voltage
16.	Duty Cycle	40.932 %	Op_point	Duty cycle
17.	Efficiency	94.73 %	Op_point	Steady state efficiency
18.	IC Tj	38.968 degC	Op_point	IC junction temperature
19.	ICThetaJA	50.0 degC/W	Op_point	IC junction-to-ambient thermal resistance
20.	IOUT_OP	500.0 mA	Op_point	Iout operating point
21.	VIN_OP	30.0 V	Op_point	Vin operating point
22.	Vout p-p	6.405 mV	Op_point	Peak-to-peak output ripple voltage
23.	Cin Pd	156.475 µW	Power	Input capacitor power dissipation
24.	Cout Pd	2.955 µW	Power	Output capacitor power dissipation
25.	Diode Pd	115.784 mW	Power	Diode power dissipation
26.	IC Pd	179.359 mW	Power	IC power dissipation
27.	L Pd	38.5 mW	Power	Inductor power dissipation
28.	Total Pd	333.797 mW	Power	Total Power Dissipation

Design Inputs

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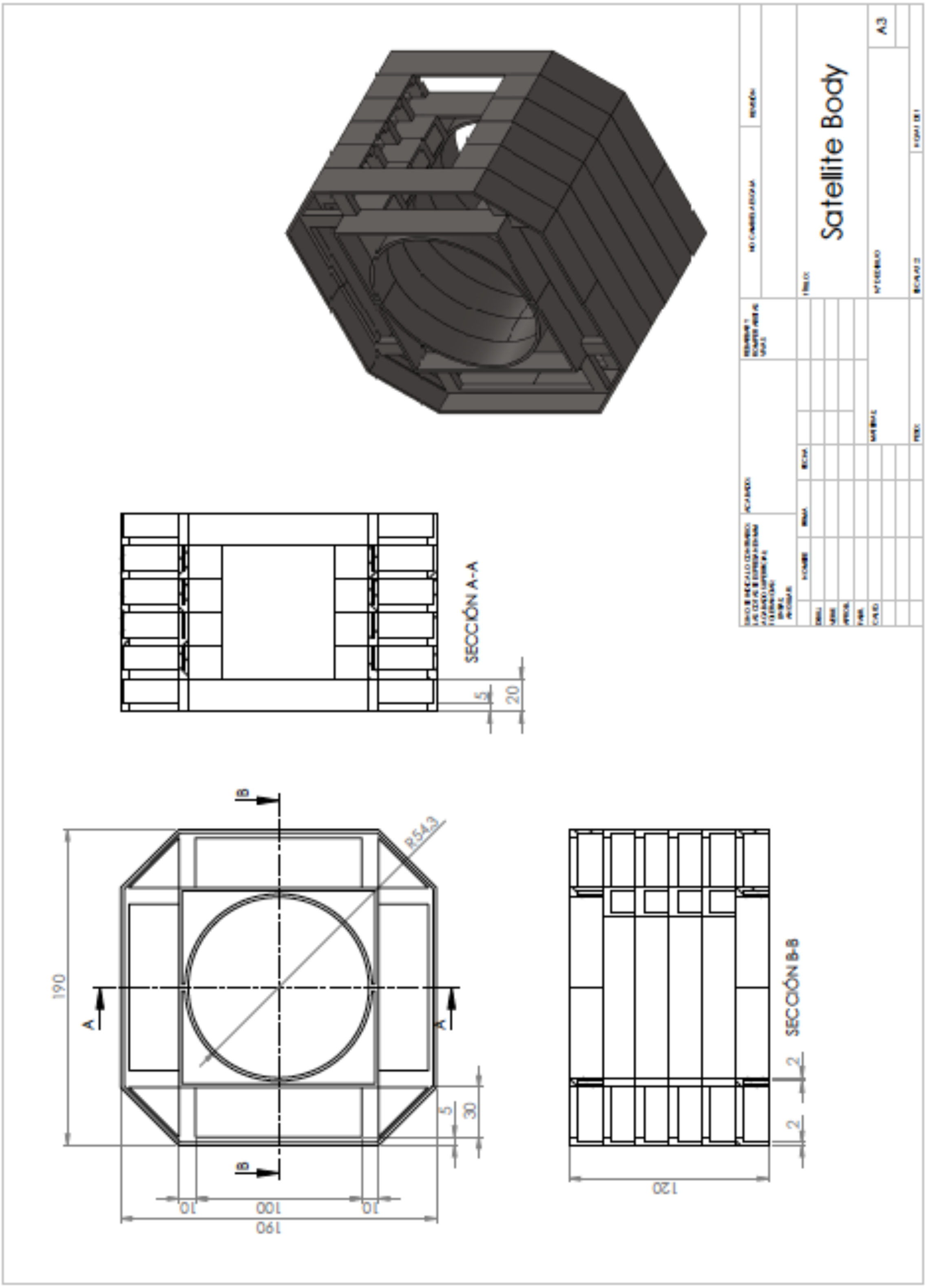
WEBENCH® Design Report LM25575MHX/NOPB : LM25575MHX/NOPB 25.0V-30.0V to 12.00V @ 0.5A May 16, 2015 12:00:24 GMT-07:00

#	Name	Value	Description
1.	Iout	500.0 m	Maximum Output Current
2.	Iout1	500.0 m	Output Current #1
3.	VinMax	30.0	Maximum input voltage
4.	VinMin	25.0	Minimum input voltage
5.	Vout	12.0	Output Voltage
6.	Vout1	12.0	Output Voltage #1
7.	base_pn	LM25575	Base Product Number
8.	source	DC	Input Source Type
9.	Ta	30.0	Ambient temperature

Design Assistance

1. **LM25575** Product Folder : http://www.ti.com/product/lm25575 : contains the data sheet and other resources.

Annexes. The sketches of 3D design



Annexes. The cost of 3D printing

impresoras3d.com

I3D DIGITAL MEDIA S.L.
NIF : B04771499
C/ Arapiles, 15 1" Derecha
04001 Almería

Aitor Martinez
NIF:

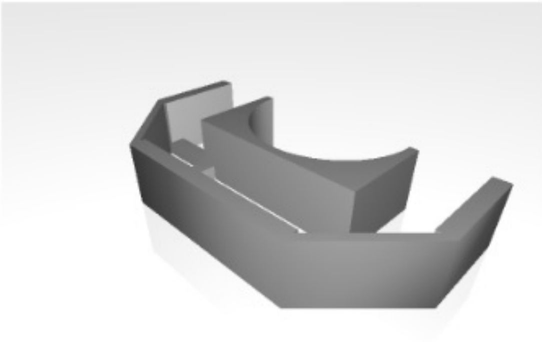
()

Fecha: 23 / 03 / 2015

nº Presupuesto: #984

Página: 1 de 1

Cantidad	Descripción	Material	Total
1	Impresión 3D del modelo "satellite2.stl"	ABS/PLA Low Cost	49,68 €



Largo: 220.00 mm
Ancho: 110.00 mm
Alto: 40.00 mm
Volumen: 240,88 cm³
Densidad: 24,88 %

Total Presupuesto: 49,68 €

impresoras 3D

Página 1 de 1