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Cubesat Constellation
Astrea

Report

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Chapter 1

Orbit Geometry

Throughout this chapter, the bases of orbital geometry will be explained in order to correctly understand the parameters that will later be exposed when dealing with the constellation orbits (or the position of the satellites in them). However, long theoretical explanations will be avoided so as not to distract the reader from the main objective of the project.

To understand the movement in space is enough to apply the Newton's laws. These, however, need an inertial non-rotating frame to be correctly described. When dealing with Earth-orbiting, one usually chooses a reference system called *geocentric-equatorial system* which is shown in the figure 1.0.1.a. As can be seen, the XY plane coincides with the plane Equatorial with the X axis pointing in the direction of the vernal equinox [1]. The Z axis correspond the axis of rotation of the earth and points to the north (following the right-hand rule).

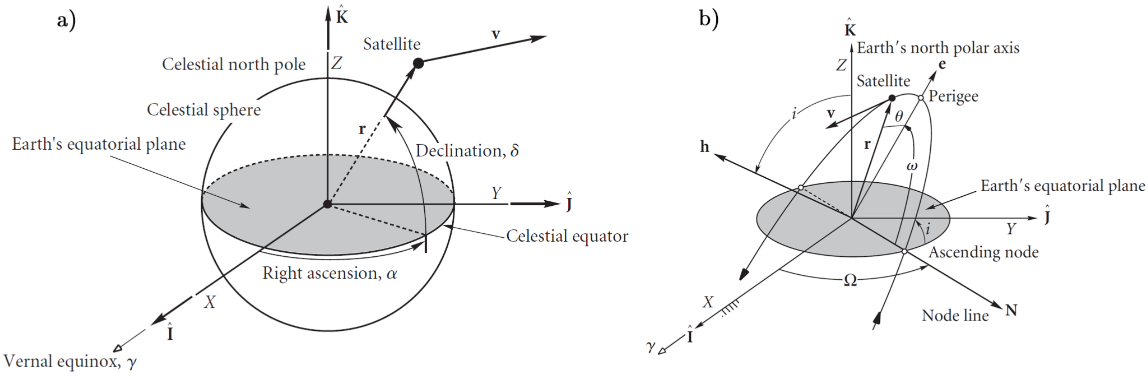


Figure 1.0.1: Geocentric-equatorial frame. Extracted from [4].

By defining this system, any point in the space can be depicted by its position vector r and we can study its movement by the velocity vector $\dot{r}$. These elements are useful especially for computational work but they nearly do not provide information about the orbit. For these reason, the orbital elements were developed.

[1]an imaginary line found by drawing a line from the Earth to the Sun on the first day of spring

1.1 Keplerian Geometry

The *Classical Orbital elements*, also known as the *Keplerian elements* as an attribution to Johannes Kepler, are six independent quantities which are sufficient to describe the size, shape and orientation of an orbit. This set of elements are shown in the figure 1.0.1.b and are defined as follows:

- **Semi-major axis** (a): It is related to the size of the orbit and its defined by the sum of the apogee (furthest point) and the perigee (closest point) divided by two.
- **Eccentricity** (e): It defines the shape of the orbit with respect to that of a circle. Thus, the eccentricity of a circular orbit is null while hyperbolic orbits have an eccentricity greater than one.
- **Inclination** (i): the inclination is the angle between the positive Z axis and the angular momentum vector ($\mathbf{h}$) which is perpendicular to the orbital plane. The inclination of the orbit can take a value from 0 deg to 180 deg. For $0\,\text{deg} \leq i \leq 90\,\text{deg}$ the motion *posigrade* and for $90\,\text{deg} \leq i \leq 180\,\text{deg}$ the motion is *retrogade*.
- **Right ascension of the ascending node - RAAN** (Ω): This parameter, along with the inclination define the orientation of the orbital plane. It is the angle between the positive X axis and the intersection of the orbital plane with the equatorial plane XY in counterclockwise direction. The intersection mentioned is called the node line and the point where the orbit passes through the node line (from south to north) is the ascension node ($0\,\text{deg} \leq \Omega \leq 360\,\text{deg}$).
- **Argument of perigee** (ω): Is defined as the angle between the ascending node and the perigee. It describes the orientation of the ellipse with respect to the frame ($0\,\text{deg} \leq \omega \leq 360\,\text{deg}$).
- **True Anomaly** (ϕ): This last quantity is used to describe the satellite's instantaneous position with respect to the perigee. Is the angle, measured clockwise, between the perigee and the satellite position. From all the orbital elements, the true anomaly is the only that changes continuously. Sometimes, true anomaly is substituted by the mean anomaly, which can be calculated using another auxiliary angle called the eccentric anomaly.

$$\begin{gathered} \cos E = \frac{e + \cos\theta}{1 + e\cos\theta} \\ M = E - e\sin E \end{gathered} \tag{1.1.1}$$

1.2 Dynamic equations

As aforementioned, the motion of an object in the space can be described using the Newton's laws. The basic idea developed by Newton is to study the Cubesat and the Earth as a spherical bodies in mutual gravitational attraction and neglect the gravitational forces caused by other objects (this is called the *two body* problem). The forces balance is simple since we only have the Earth gravitational attraction, which must compensate the centripetal acceleration of the satellite. Thus, using the law of universal gravitation,

$$-G\frac{M_E m_{sat}}{r^3}\vec{r} = m_{sat}\vec{a}_{sat} \tag{1.2.1}$$

Where G is the gravitational constant and r represents the distance between the satellite and the Earth. From the last equation, we only want to obtain the acceleration, therefore:

$$-G\frac{M_E}{r^3}\vec{r} = \vec{a}_{sat} = \frac{d^2\vec{r}}{dt^2} \tag{1.2.2}$$

For simplicity, it usual to denote $\mu = GM_{earth}$ resulting in the following equation:

$$-\frac{\mu}{r^3}\vec{r} = \frac{d^2\vec{r}}{dt^2} \tag{1.2.3}$$

This expression is a second order equation that models the motion of the Cubesat relative to the Earth and it can be analytically solved. The only problem is that several hypotheses have been applied that make the case different from reality. The formulation should be modified to take into account the effects due to:

- More bodies attracting the satellite (Sun, Moon, Venus, etc.)
- The existence of more forces like the drag, the solar radiation pressure, etc.
- The earth is not an spherical body.

The corrections for considering these things are called perturbations and they are explained in the Chapter 4 of this part of the report.

Chapter 2

Orbital Coverage

2.1 Satellite Footprint

2.1.1 Introduction

The first step to build a satellite network with global coverage is to compute a single satellite footprint.

The footprint of a satellite is defined as the region of Earth where a single satellite can be seen. This Earth coverage surface provided is spherical and depens on some orbital paramaeters such as:

- Height
  When increasing height the footprint of a satellite grows.
- Elevation angle
  When increasing the elevation angle, which is the angle between the satellite and the horizontal plane of an arbitrari point of the Eartg, the surface seen by the satellites descreases. (This parameter will be later studied in detail)

2.1.2 Footprint Computation

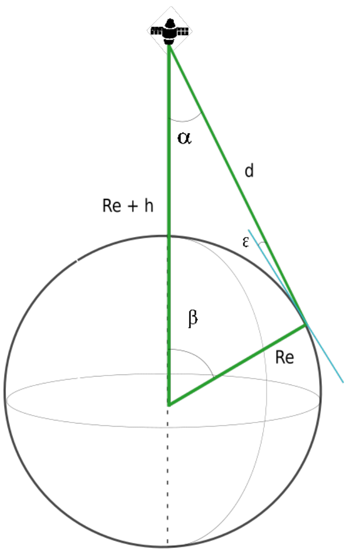


Figure 2.1.1: Single satellite coverage geometry

In order to compute the coverage area we must solve the triangle depicted in figure 2.1.1 where the basic geometry of a satellite footprint is shown.

The most needed parameters are the distance from a random point on Earth (where we can suppose our ground station to be) to the satellite denoted by d and the central angle, denoted with a β .

Applying cosines law to the triangle shown in figure 2.1.1, we obtain the following expression:

$$r^2 = R_{earth}^2 + d^2 - cos(90 + \epsilon) \tag{2.1.1}$$

Isolating d from the equation above and changing $r = R_{earth} + h$, where h is the actual height of the satellite regarding the Earth surface, we arrive at:

$$d = R_{earth}\left[\sqrt{\left(\frac{h + R_{earth}}{R_{earth}}\right)^2 - cos^2\epsilon} - sin\epsilon\right] \tag{2.1.2}$$

From the figure 2.1.1 we can also extract a relation between the central angle, the distance d and the elevation angle. This relation together with the equation 2.1.2 allow us to find β.

$$\begin{aligned} dcos\epsilon &= (R_{earth} + h)\, sin\beta \\ \beta &= \frac{1}{R_{earth} + h} \arcsin\left[d(\epsilon)cos\epsilon\right] \end{aligned} \tag{2.1.3}$$

Once the central angle β has been computed we are able to obtain the footprint satellite's are using the equetion below:

$$S = 2\pi R_{earth}^2(1 - cos\beta) \tag{2.1.4}$$

The size of the footprint will determine the level of coverage our constellation provides, therefore when deciding the value of the orbital parameters it has to be a factor to consider.

2.2 Elevation Angle

The angle of elevation is essential to calculate the geometry of our constellation. As discussed previously, our aim in this project report is to justify how global coverage will be fulfilled. First, we define for a given groundstation the angle between its beam pointing right to the satellite and the horizontal local plane as the elevation angle. Secondly, a study is conducted in order to relate the height of the satellite, the elevation angle and the coverage of the Earth. Finally, we complete our orbital design by configuring a constellation that will securely define a global coverage fulfillment. Next, we will be defining how these parameters are related.

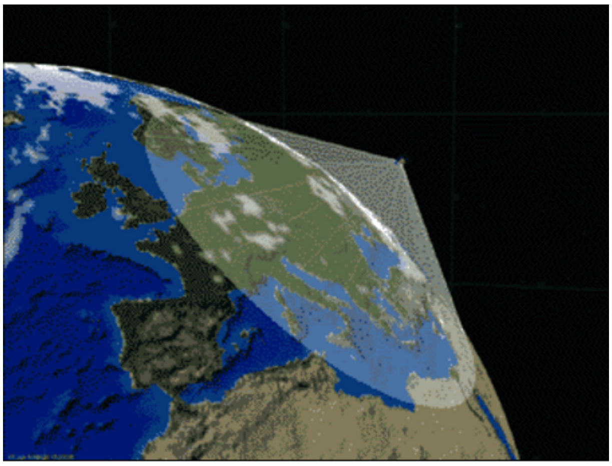


Figure 2.2.1: Elevation angle cone. Source: NOAA

2.2.1 Elevation angle cone

Global coverage will be discussed considering the elevation angle and its resulting footprint on Earth. The elevation angle is described by the angular orientation of the antennas in the ground station. However, this angle is also perceived by the satellite in a similar way - it will vary depending on the orientation of the satellite and the angle between horizontal local planes. In order to describe the footprints we must define a cone which vertex is set at the antennas of the satellite, pointing down to Earth, and which generatrix is given by the angle of elevation. This elevation angle based cone is the description of the paths that our communications can take place. In other words, the generatrix of this cone is setting the limits in which the antenna will operate as function of the elevation angle. This implies that our satellite will be able to communicate to all the points contained in the cone. Finally, this cone will be describing a circular surface on top of the Earth which we will call the footprint of the satellite. Additionally, this footprint is the coverage that a single satellite can generate, hence we will be distributing satellites all around the Earth in order to fulfill global coverage.

2.2.2 Atmospheric restrictive conditions

In order to obtain the final restrictive angle of elevation needed to contact the ground stations some considerations have to be made. Then, we will relate these to our bandwidth in order to analyse if they must be taken into account when communicating with ground stations [11]. The most important parameters are tge following:

- **Atmospheric gases:** water vapour and oxygen absorptions; important when frequencies are above 3 GHz. More information [12] and [13].

- **Precipitations and Clouds:** these conditions are relevant for signals above 10GHz.

By means of these physical phenomena we can substract the elevation angle as function of the latitude. However, we must take into account that these physical conditions give a value for the elevation angle which may not be the most restrictive. Global coverage conditions, bandwidths, inclination and the final distribution of our constellation will be considering this elevation angle and viceversa, iteratively.

The ASTREA CONSTELLATION was designed and optimized in order to fulfill global coverage for a constant elevation angle - respect to the latitude - of 20 degrees. This corresponds to a predefined model.

Our constellation will be operating at S-band for telemetry and X-band for data relay. Therefore, the satellites need to be operating up to 10 GHz. This directly implies that physical conditions such as atmospheric gases, precipitations and clouds must be studied when determining the elevation angle needed. We can obtain a more realistic model for the elevation angle comparing the the frequencies of our constellation to others that are currently operative. For this model we find the following:

– At low latitudes (between 0 and 30 degrees) the constellation fulfills global coverage generously.

– At ground station latitude (60 degrees) the constellation is covering the station succesfully. For the previous model coverage was well established with margin - as we set global coverage from the equator. Note: each orbit could be reduced to a lower number of satellites per plane, but this would endanger the correct and stationary behaviour of the constellation. In fact, in this case we would not be able to control possible incidencies such as unoperative satellites with enough margin.

– The ground stations are covered at all time for at least one satellite.

2.2.3 Elevation angle of other current constellations

Analysing the minimum elevation angle needed in order to fulfill global coverage requieres, as mentioned before, the understanding first of the restrictive conditions of the atmosphere and how these will alter it. As a consequence of the different physical conditions given before we will be able to determine a relation between latitude and elevation angle. All the same, the elevation angle depends on the bandwith in which the satellites operate, hence different distributions of this angle respect to the latitude will be described depending on the bandwidths used.

– Celestri: 18.8 to 20.2 GHz at 48 degree inclination.
– GlobalStar: 2.4 GHz at 52 degree inclination.
– Iridium: 20 to 30 GHz at 90 degree inclination - polar orbits.

Comparing our configuration to other present constellations some clarifications can be made:

– The minimum elevation angle peak is proportional to the bandwidth at which the satellite is communicating with Earth. For instance, Iridium's peak of elevation angle is the highest relative to the other configurations since it is also working with the highest frequency signals.

– The latitude position of the peaks is related to the inclination of the constellation. Iridium, - a polar orbit based configuration - describes a peak at 90 degrees of latitude whereas Celestri and GlobalStar are near 40 to 50 degrees.

With these tendencies our model can be likewise described. This model would be defined by a peak at about 70 degrees latitude, smaller than those of the Celestri and Iridium constellations, but higher compared to the Iridium constellation peak. Thus, the previous model of a constant 20 degree elevation angle fulfills these requirements.

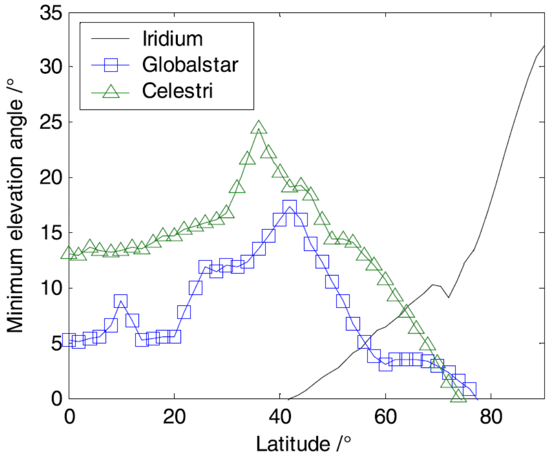


Figure 2.2.2: Minimum elevation angle as function of latitude. Source: [5]

2.3 Minimum Plane Inclination

As it has been pointed before, there are several factors to take into account in order to design a constellation that provides global coverage on Earth. In this section the minimum inclination to achieve that purpose is asessed. Using the theory previously developed, we can observe the following results:

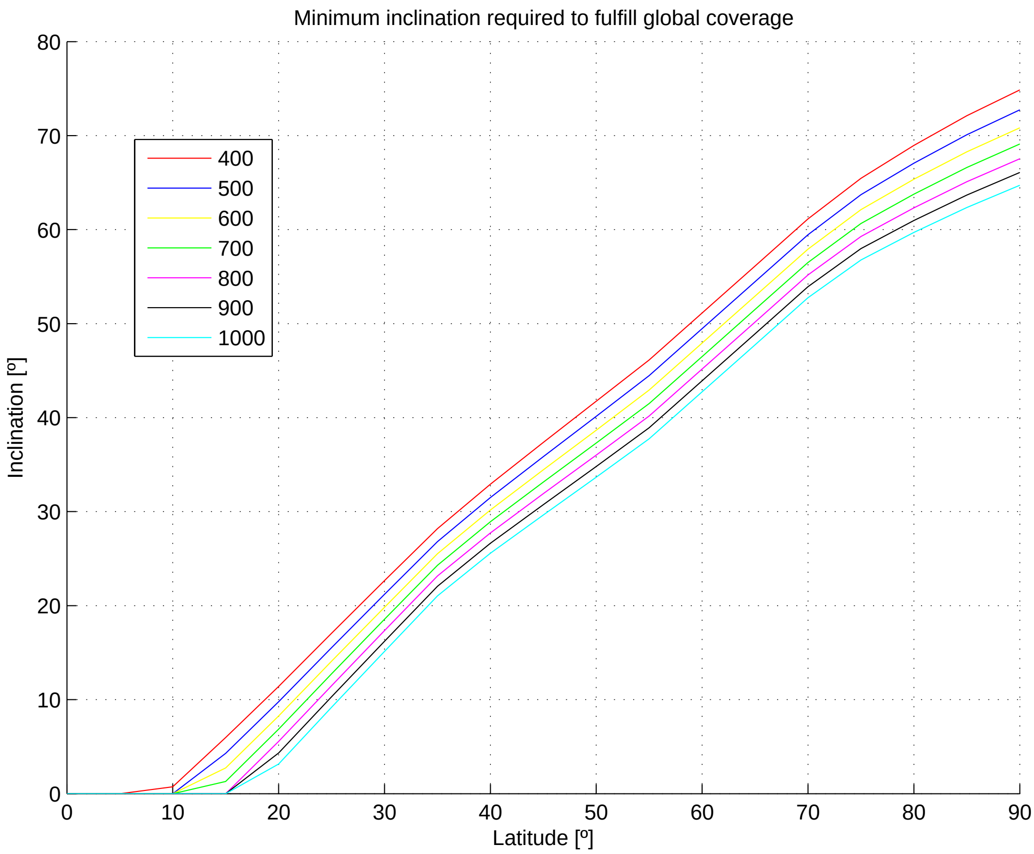


Figure 2.3.1: Minimum Inclination to provide coverage at different latitude for different orbit apogees.

As it can be observed, if the goal of the design is to provide full global coverage, the distribution of elevation angles with latitude is not significant, since the inclination is required to be higher than aproximately 75º. In the other cases, the change of minimum elevation angle distribution causes changes of tendency in the distribution of inclination required.

In conclusion

The main point is that there is a limit inclination for a Walker-Delta constellation configuration in order to provide global coverage at the desired latitude. With this study, this limits in the design algorithms can be set.

2.4 Satellite to Satellite Visibility

One of the restrictive conditions that we must take into account is the visibility between satellites. Communications among different satellites is they key point of our constellation. Therefore, this has to be guaranteed considering a model which will represent the conditions of the atmosphere for LEO communications.

In order to fulfill communications among satellites we must consider that a straight beam can be described between two consecutive satellites, which will then communicate with others. These two satellites will need to be at a distance such that the Earth itself doesn't interfere in this straight beam. Depending on the bandwidth of our constellation we will also have to consider that this communication beam will not interfere with a given element of the atmosphere such as the upper layers of the ionosphere. Thus, a model will be developed in order to limit the minimum altitude at which this beam is guaranteed to pass through safely.

This model is a restrictive condition that we need to satisfy when designing our constellation.The highest restrictive conditions are the upper layers of the ionosphere, specifically the E layers at 150 km above the surface of the Earth. Reflections and absorptions can occur for both E layers and sporadic E layers. E layers may reflect signals of frequencies below 10 MHz whereas Sporadic E layers can be a problem up to 225 MHz. Working for S bands and X bands implies that neither of these layers will alter the signals of our constellation.

Operating and computing with these conditions a maximum distance is a obtained which defines how far these satellites can be from each other. A simple equation is used to calculate this distance considering the height of the satellites and the height of the E layers in the atmosphere.

$$d = 2\sqrt{(R + h_{sat})^2 - (R + h_{atm})^2}$$

$$h_{sat} = 550\ km$$

$$h_{atm} = 150\ km$$

$$R = 6371\ km$$

The final expression for the distance between two satellites indicates that distance between two satellites has to be smaller than 4640 km approximately. For this result we conclude that this restrictive condition is actually less restrictive than the 9 planes needed for our constellation. Thus, satellite to satellite visibility is a parameter which will not affect the design of our constellation after all.

2.5 Market Study: Current Nanosatellites in Orbit

2.5.1 Criteria for the orbital height of the satellites

Satellites currently in Orbit

If only geometric considerations were to be applied in the design of a satellite constellation, it is clear that the higher the orbit the broader is the footprint in the surface leading to a smaller number of satellites. However, if the service of communications is to be offered, the satellites currently in orbit or in design phases need to be at higher orbit than the one of the constellation. The purpose of that requirement is to intersect the field of view of the satellites that nowadays point to Earth.

From source [?] we can study how the currently on orbit satellites are launched and specially, in which orbits. The results of the study of this source is presented below. All of them are in Low Earth Orbits, and half of them above 550km. In total, there are 203 operational satellites.

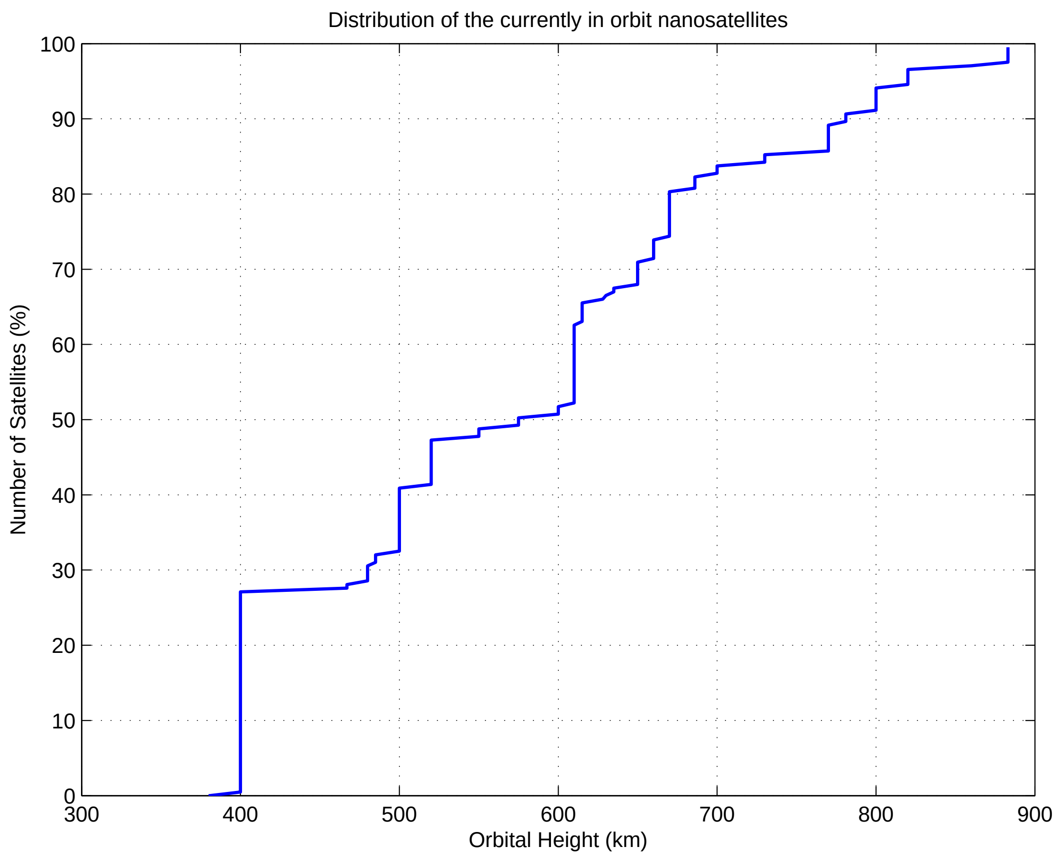


Figure 2.5.1: Distribution of the currently in orbit nanosatellites.

The most interesting potential clients

Lots of satellites are orbiting at heights lower than 500km, mainly because one of the most feasible way of launching a small satellite is from the International Space Station. However, this very low LEOs are related to very high speeds and specially to low lifetimes, since drag affects them in a more significant way. To the interest of the constellation, the satellites at higher altitudes are a better commercial target, since they are going to be in orbit for longer missions. In addition, the same orbit decay problems are avoided for the constellation satellites.

2.5.2 New Space: Adapting to new society needs

Nowadays new satellites willing to provide services to Earth are being positioned closer than ever. Where closer can be applied in many points of view. Physically, the satellites are placed every time at lower orbits, since the energetic requirement is lower. Technically, the space certified materials and hardware are becoming more feasible, and new launchers are smaller. In the end, everything comes down to an economic approach, launching satellites is becoming cheaper every time and this means closer to the private pocket.

In the future, the possibility of using the Astrea constellation to contact Earth can reduce the requirements for the antennas and AOCSs to communicate with ground, leading to a whole new level of resources for the satellite payload. For instance, by communicating to the constellation pointing to outter space instead of pointing down to Earth. That is just a way in which Astrea is in the New Space Generation.

In conclusion, In the decision process one of the statistics considered with certain weight will be the following: the ratio of satellites at which the constellation will be able to provide service considering that nowadays all of them point down to Earth.

Chapter 3

Constellation Configuration

"Our two greatest problems are gravity and paperwork. We can lick gravity, but sometimes the paperwork is overwhelming."

Werner von Braun, 1958

3.1 Introduction: The Global Positioning System Example

Depending on the application the Space Segment of a mission can vary in an infinite number of ways. Probably the most famous and widely used satellite constellation is the the Global Positioning System satellite network. In this case, it uses an irregular geometry.

The GPS Constellation: An example of irregular distributed orbits [?]
The GPS is a constellation property of the U.S. It provides positioning, navigation and timing. The constellation was designed with a 24-slot arrangement to ensure a visibility of at least four satellites from any point on the planet. Nowadays the constellation has expanded to a total operative number of 27-slot since June 2011. Some characteristic parameters of the satellites are the following:

- Orbit: Almost Circular
- Height = 20,200 km (MEO);
- Lifetime = 12.5 years;
- Satellite Cost = 166 million USD;
- Inclination = 55º;
- Number of planes = 6;
- Phasing: 30º-105º-120º-105º;

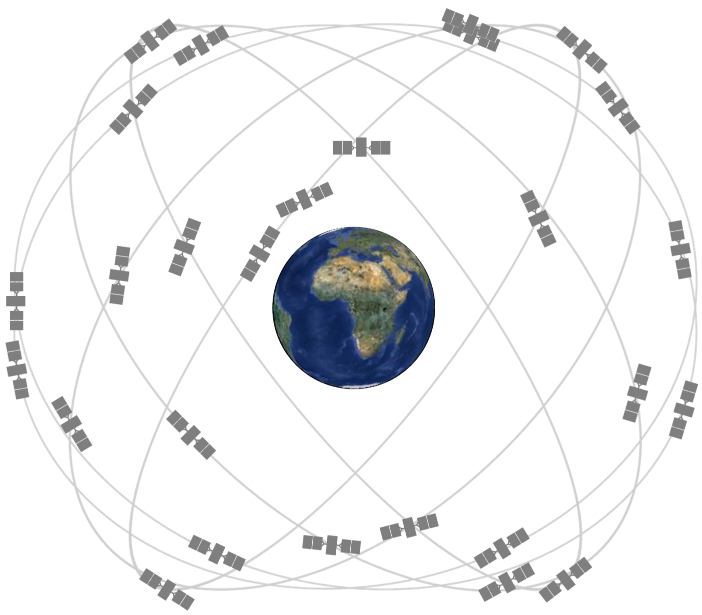


Figure 3.1.1: Distribution of the expanded 24-slot GPS constellation. [?]

3.2 Polar Orbit Constellation

3.2.1 Introduction

Polar Orbits are probably the simplest way to configure an evenly spaced constellation. As we will see in the section **Orbit Perturbations** when the inclination is the same for all the planes, the deviations tend to be the same for all the satellites. In addition, the computation of the number of satellites required is also easier.

The Iridium Constellation: An example of near polar orbits [?]
The Iridium constellation is a private constellation. It provides voice and data coverage to satellite phones among other services. The constellation was designed with 77 satellites, giving name to the constellation by the chemical element. The constellation was reduced to a number of 66. Sadly, Dysprosium is not such a good commercial name. Some characteristic parameters of the satellites are the following:

- Orbit: Almost Circular
- Height = 781 km (LEO);
- Satellite Cost = 5 million USD;
- Inclination = 86.4º;
- Number of planes = 11;
- Phasing: Regular;

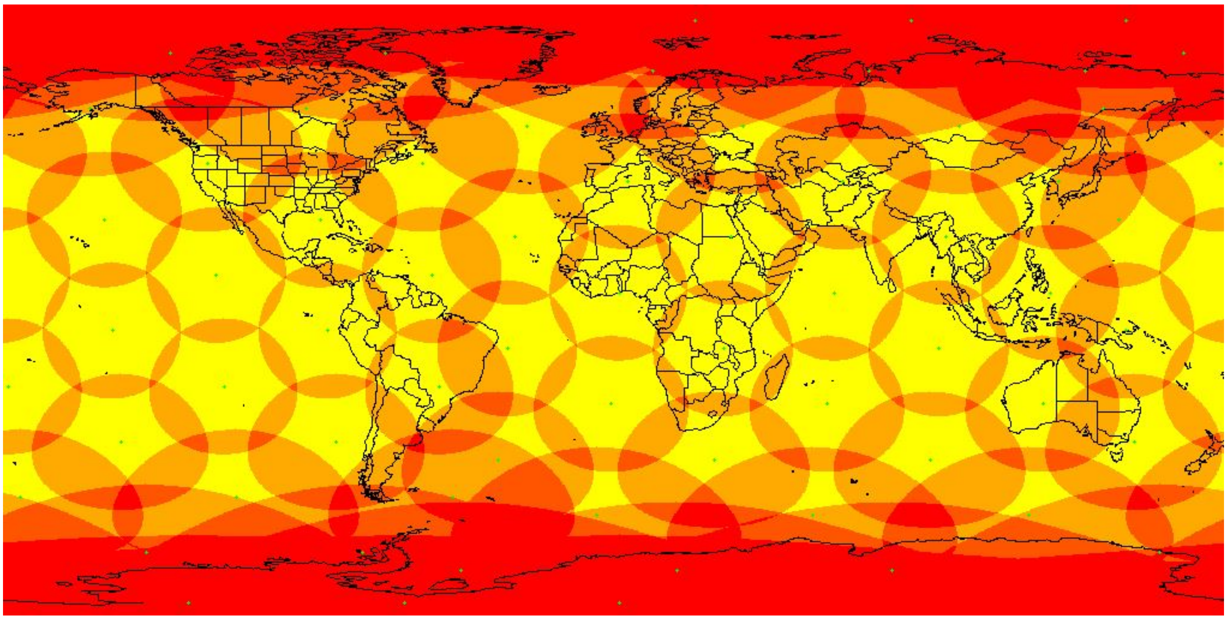


Figure 3.2.1: Distribution of the 66 Iridium constellation satellites. Generated using [?]

3.2.2 General Configuration

The Polar Orbits configuration consists in the distribution of plains with inclination equal to 90 degrees. Note that the satellites will be travelling parallel to the satellites of the next plain except for the communications between the first and the last plane.

The communications between satellites in antiparallel directions require less space between plains to be fulfilled. In order to solve this isconvenience the separation between the first and the last plain is reduced.
The plains are splitted in the following pattern:

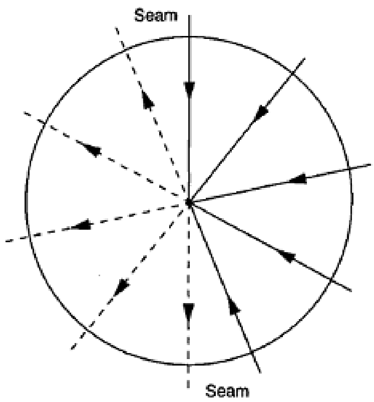


Figure 3.2.2: Distribution of the planes for Polar Orbits design.

3.2.3 The Streets of Coverage Method

This Street of Coverage Method is obtained from [8]. As you can see in the figure below, the relations between angles seen from different satellites can be easily computed. The main variables are the following:

Streets of Coverage Method Variables	
N	Number of Satellites
n_p	Number of Planes
N_{pp}	Number of Satellites per plane
S	Separation between satellites of the same plane
D	General space between planes [º]
D_0	Space between antiparallel planes [º]
ε	Elevation angle [º]
λ_{street}	Street of coverage Width [º]
λ_{max}	Maximum footprint Radius [º]

Table 3.2.1: Streets of Coverage Method main variables

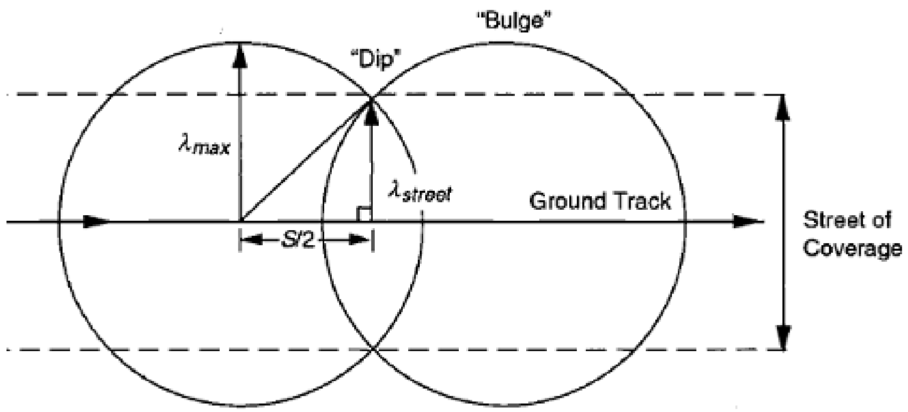


Figure 3.2.3: Single plain street of coverage. The footprints of the satellites superpose leading to a street. [?]

From the figure it can be inferred:

$$S < 2\lambda_{max}$$

$$cos(\lambda_{street}) = cos(\lambda_{street})/cos(S/2)$$

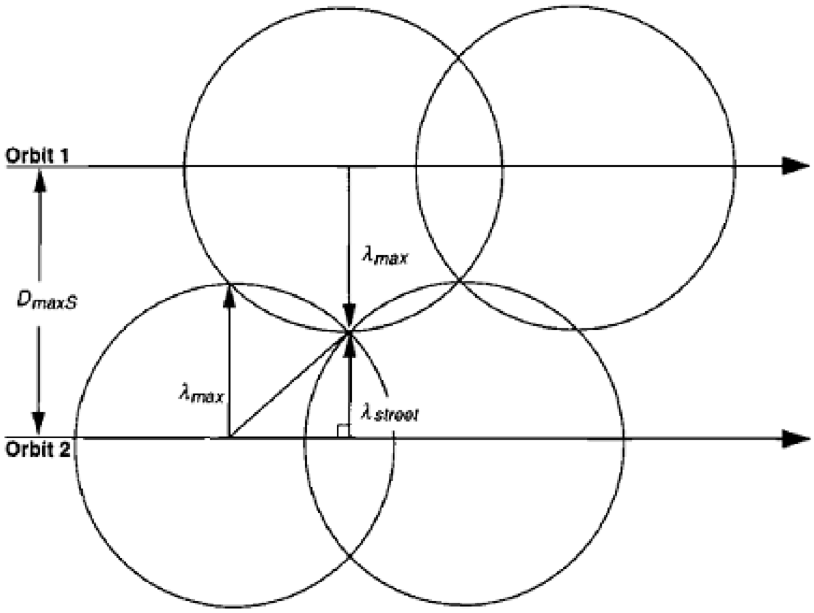


Figure 3.2.4: Two plains streets of coverage. An optimum phasing needs to be obtained. [?]

From this point of view, in general:

$$D = \lambda_{street} + \lambda_{max}$$

n For the antiparallel planes:

$$D_0 = 2\lambda_{street}$$

And the overall relationship between planes sums:

$$180 = (n_p - 1)D + D_0$$

The algorithm for computing the Streets of Coverage Results is defined in the following way:

$$\text{Inputs: Height, elevation, inclination...} \rightarrow \lambda_{max} \rightarrow N_{pp} = \left\lceil \frac{360}{2\lambda_{max}} \right\rceil \rightarrow$$
$$S = 360/N_{pp} \rightarrow \lambda_{street} \rightarrow n_p \rightarrow N = N_{pp} * n_p$$

3.2.4 Results of Streets of Coverage

A MATLAB routine has been designed to compute the previously described algorithm. In this conceptual design phase, different heights are computed in order to see the evolution of the number of satellites.

General Solution

The program in runned in a broad range of parameters to see the evolution of the number of satellites. As it can be predicted, as the height increases the number of satellites is reduced. The reason is that the footprint of the satellites increases with the height. In addition, as the minimum elevation over the horizon to contact the satellites is reduced, the number of satellites is also reduced for the same reason.

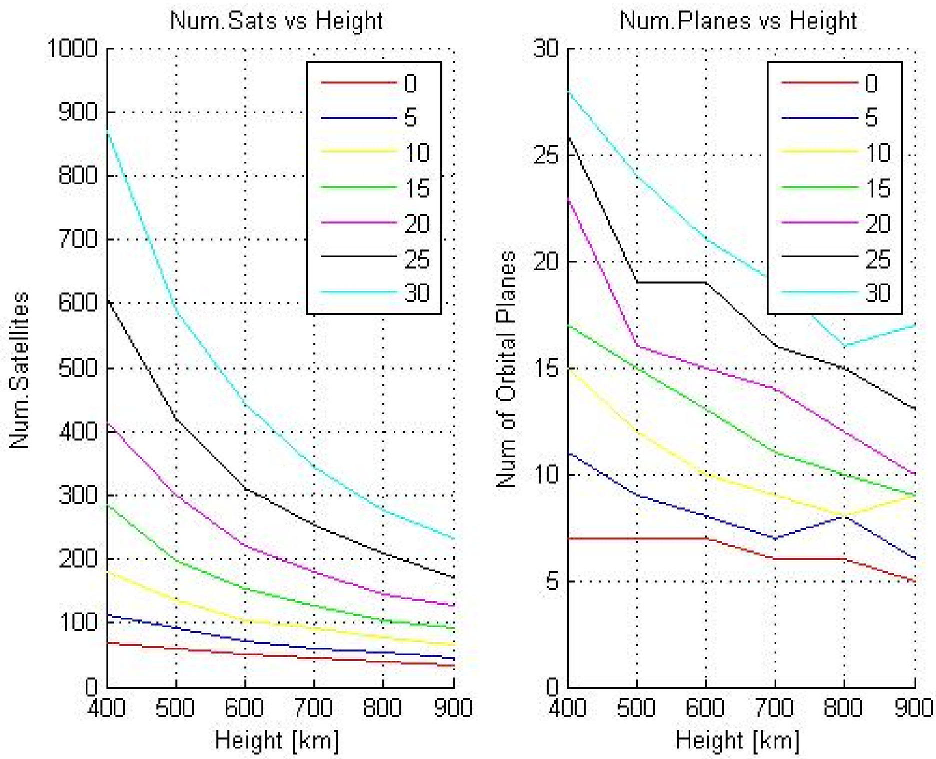


Figure 3.2.5: Variation of number of satellites for different heights and elevation angles

Detailed Solution

Given the previously justified assumptions, the same simulation is computed for a more reasonable range of results. In this case, the elevation is set as:

$$\varepsilon = 20^{\mathrm{o}}$$

.

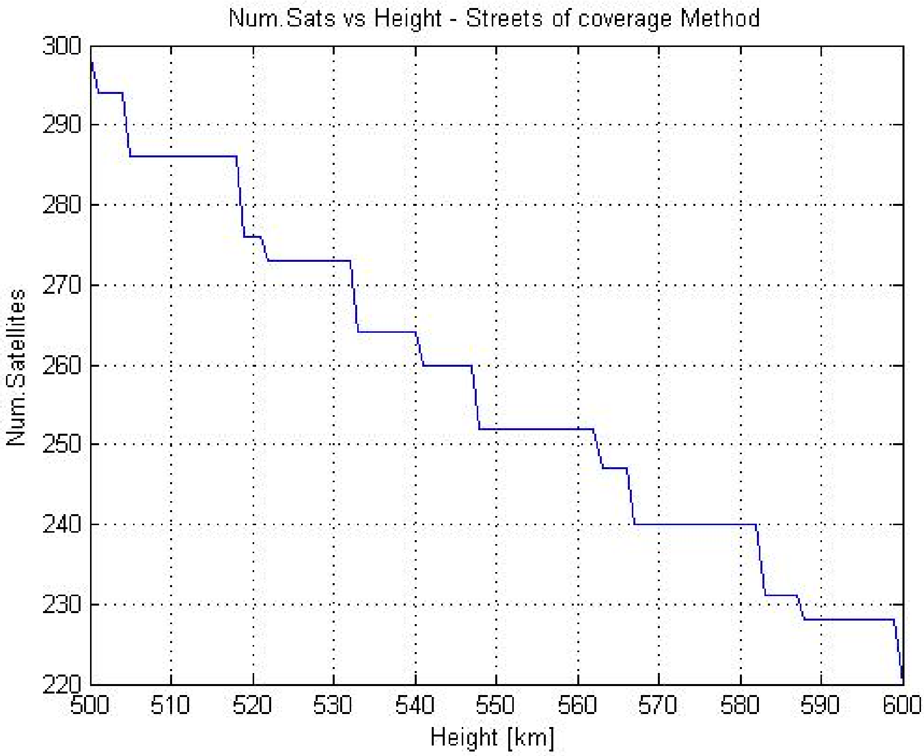


Figure 3.2.6: Variation of number of satellites for different heights between 500 and 600km.

Conclusion

The computation and the design of this constellation requires small computational and conceptual effort. However, the number of satellites and planes is greater than expected. Even though the technical complexity can be reduced, the availability of small launchers to reach this particularly inclined orbit is also small. In conclusion, more constellation configurations need to be assessed to compare and select the most feasible one.

3.3 Walker-Delta Constellation

Walker Delta Pattern constellations are a type of symmetric, inclined constellation made of equal-radius circular orbits, with an equal number of satellites each one. There are several ways to construct a Walker-Delta Constellation:

- Full Walker-Delta Configuration
- Semi Walker-Delta Configuration
- Custom Walker-Delta Configuration

3.3.1 Full Walker-Delta Constellation

3.3.1.1 Characteristics

A typical delta pattern has the following characteristics:

- The constellation contains a total of T satellites evenly spaced in each of the P orbital planes. All planes have the same number of satellites, defined as S, equally distributed. Thus:

$$T = SP \tag{3.3.1}$$

$$\Delta\varphi = \frac{2\pi}{S} \tag{3.3.2}$$

  Where $\Delta\varphi$ is the angle between satellites in the same plane.

- All orbits have equal inclinations δ to a reference plane. If this plane is the Equator (it usually is), then the inclination δ equals the orbital parameter inclination i [6].

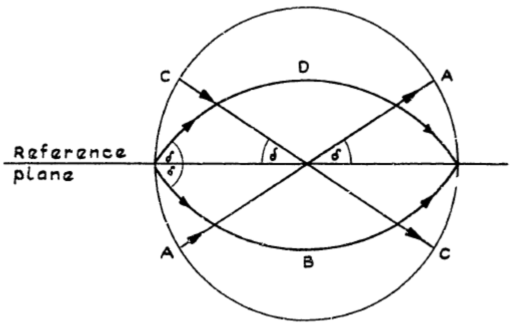


Figure 3.3.1: Definition of the inclination δ. Extracted from [6]

- The ascending nodes of the orbits are equally spaced across the full 2π (360° of longitude) at intervals of:
$$\Delta\Omega = \frac{2\pi}{P} \tag{3.3.3}$$
- The position of the satellites in different orbital planes is measured through the factor F. When a satellite is at its ascending node, a satellite in the most easterly adjacent plane has covered a relative phase difference F. The real phase difference is defined as:
$$\Delta\Phi = F\frac{2\pi}{P} \tag{3.3.4}$$
In order to have the same phase difference between all orbital planes, F is defined as an integer, which may have any value from 0 to (P-1).

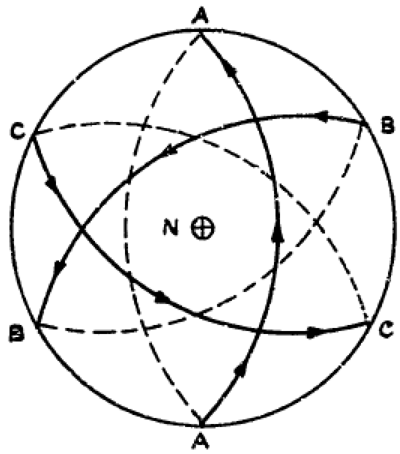


Figure 3.3.2: Delta pattern as seen from the North Pole. Extracted from [7]

With these characteristics, delta constellations are more complex than polar constellations. Because of the inclination of the orbits, the ascending and descending planes and the coverage of the satellites continuously overlap. This characteristic is a constraint on intersatellite networking because the relative velocities between satellites in different orbital planes are larger than in a polar constellation. Consequently, tracking requirements and Doppler shift are increased [?].

3.3.1.2 Notation

J.G. Walker developed a notation to define this constellations with only 4 parameters [7]:

$$i : T/P/F$$

Since all satellites are placed at the same altitude, with these notation the shape of the pattern is completely determined. However, to determine all the orbital parameters it is necessary to know the radius of the orbits.

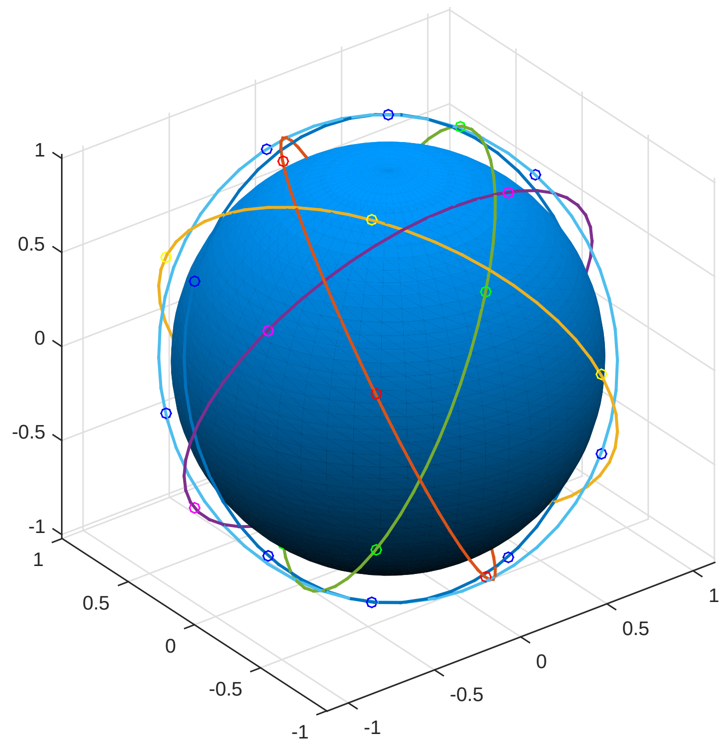


Figure 3.3.3: Delta pattern 65°: 30/6/1

3.3.1.3 Coverage

The previous section has shown that in polar orbits the coverage of the constellation could be determined with the streets of coverage method. On the other hand, in delta patterns it is necessary to study each configuration to verify its coverage. J.G. Walker determined that delta patterns gave better coverage than polar orbits, but not substantially better in the case of single coverage. This kind of patterns are more useful for double or triple coverage constellations, as it can be seen in Figure 3.3.4. However, his calculations were for a low number of satellites, so it is necessary to compute new results for the number of satellites of the Astrea constellation.

3.3.2 Semi Walker Delta Configuration

In order to reduce the necessary costs to design this satellite-based constellation some other configurations will be discussed. The Walker Delta Configuration (WDC) represents the most general constellation for a given inclination different to 90 degrees, i.e. 75 degrees. The WDC is a uniform based 360 degree generated configuration with equidistant orbits, which implies a certain redundant Earth coverage as described in the previous chapter.However, this can and will be solved by generating a 180 degree constellation - Semi Walker Delta Configuration (SWDC) - which will also fulfill global coverage although

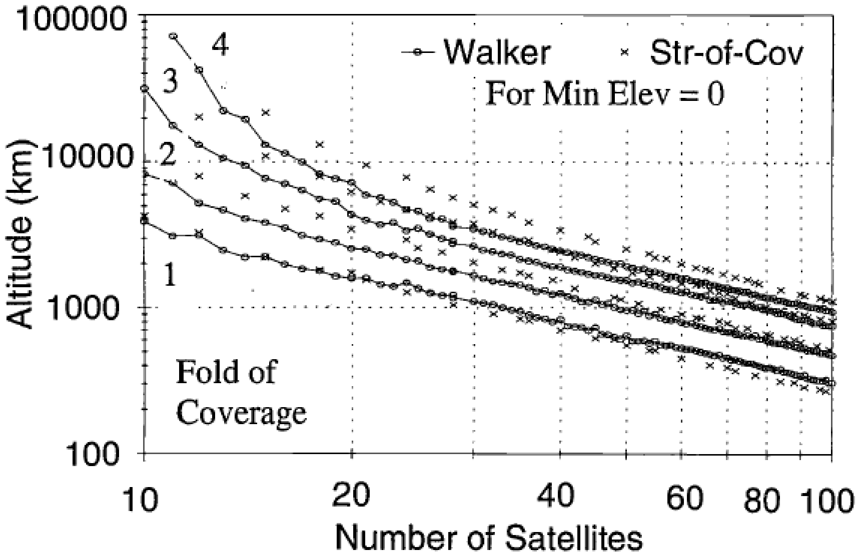


Figure 3.3.4: Minimum altitude for continuous global coverage. Comparison between polar patterns and Walker delta patterns. Extracted from [8]

having some inconvenients.

3.3.2.1 Advantages

- **Distance between planes reduced.** With the SWDC constellation the redundant orbits are directly corrected, thus the distance between planes is reduced to half, as results from the geometry itself.

- **Less number of planes needed.** This means that in order to approach global coverage fewer planes will be requiered due to the decrease in distance between planes.

- **Satellites following the same direction - sense** With the SWDC constellation the orbits have no interaction with each other, thus the satellites for each orbit can be set following the same direction. This will significantly improve the communications among satellites from different planes; also, we will be avoiding the Doppler Effect.

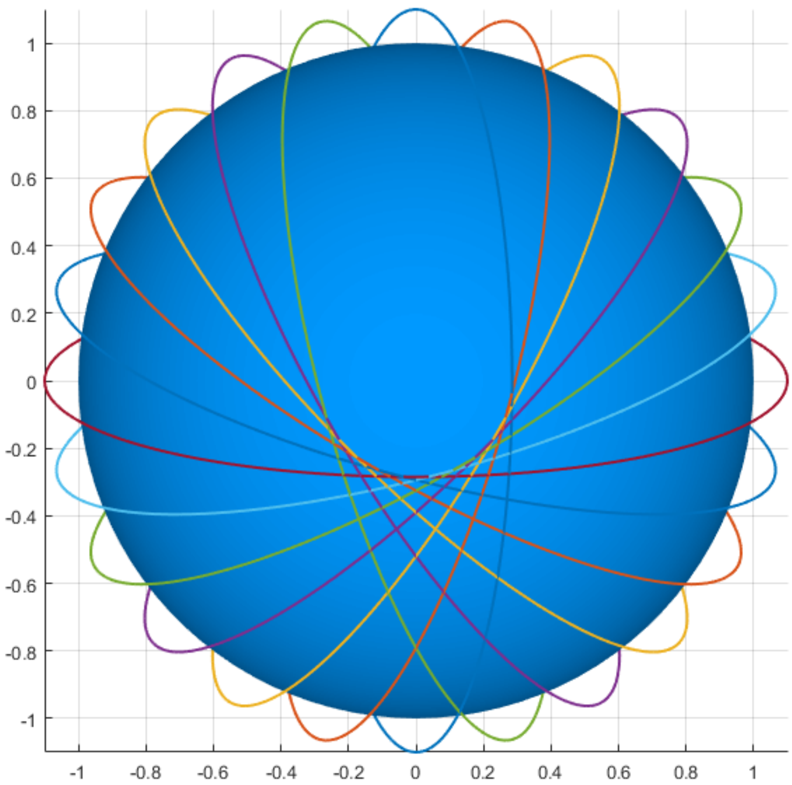


Figure 3.3.5: 12 plane SWDC. Note the gap and the equidistant planes

3.3.2.2 Disadvantages

- **Gap configuration.** With the SWDC constellation the main problem is the gap that results from configuring the constellation at a given inclination and describing equidistant orbits. In order to fulfull global coverage this gap will have to be covered by means of auxiliar orbits.

3.3.3 Other Walker Delta Configurations

As we have discussed for the SWDC, the main disadvantage respect to the Walker Delta Configuration is the fact that a gap is obtained, thus a global coverage network cannot be described. In order to cover the entire Earth we have analysed some ways of covering the gap with auxiliar orbits.

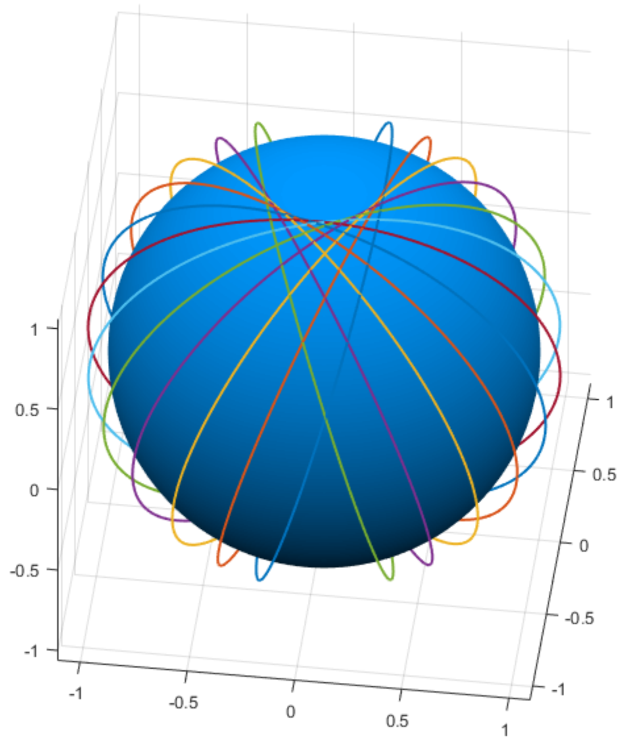


Figure 3.3.6: This geometry distribution induces a large anti-symmetric gap

3.3.3.1 SWDC including an additional polar orbit.

This polar orbit would be set directly on top of the gap described by the SWDC. The main issue with polar orbits, as discussed before in this report, is the complex reorientation and decay in inclination that takes place. We must take into account these considerations when covering the entire Earth, especially if we only have one polar orbit in our constellation.

3.3.3.2 Mixed Walker Delta.

In order to avoid using polar orbits and their complex reorientations, we can contemplate adding planes to the SWDC. In result, different configurations distributed around the Earth can be described and set in order to fulfill global coverage. As discussed before, the SWDC constellation is generated around 180 degrees whereas the Walker Delta Constellation is a 360 degree generated configuration. This Mixed Walker Delta (MWDC) is the result of adding some planes to the SWDC, thus a constellation can be generated for different degree values, such as 200, 225, 240, etc.

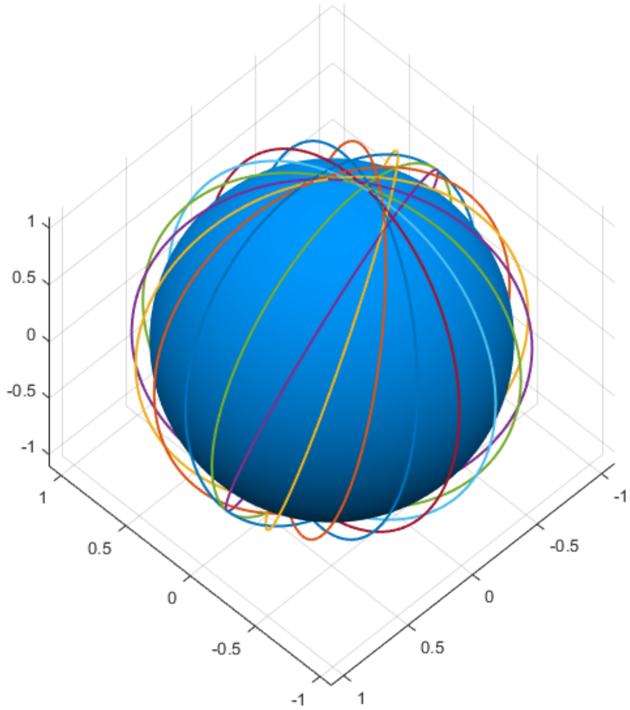


Figure 3.3.7: Added polar orbit to the 11 plane based SWDC

After different mathematical approaches and optimal solutions, the department of Orbital Design considered that the best option in order to have a global coverage constellation with the least economic and strategic issues - exposed and discussed in previous chapters - would be that of a 225 degree generated MWDC, defined by 9 planes and 21 satellites per plane. This configuration was found optimizing the whole Earth in order to have full coverage without gaps (except for the limitations of this model at high latitudes). An important consideration is that we also analysed other Mixed Walker Delta Configurations for 210 and 240 degrees, but these resulted in a more expensive distribution of satellites.

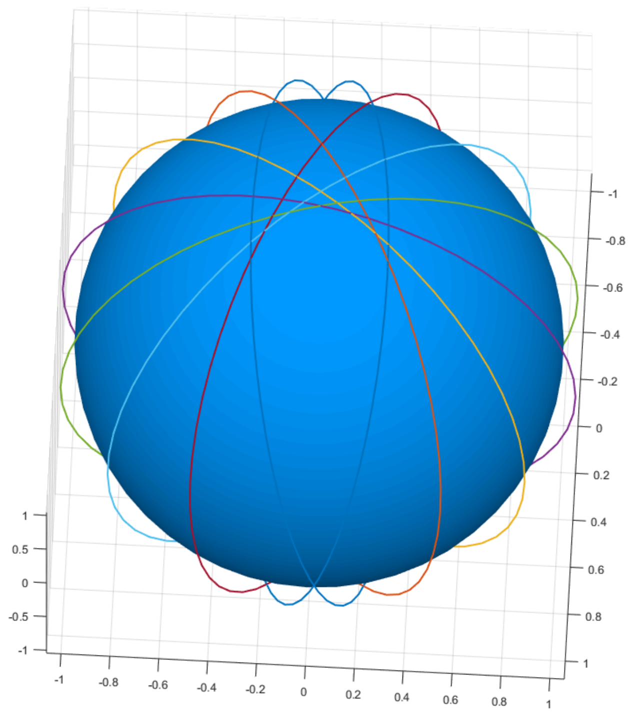


Figure 3.3.8: 8 plane based MWDC generated for 210 degrees

3.4 Testing Method

3.4.1 Introduction

To design Astrea constellation the orbit parameters must be decided following the stablished requirments. As seen in the previous sections, there are different types of constellation that must be considered when selecting those parametrs.

The main requirement in the bases of this chapter is to fulfill global coverage of the Earth. Therefore all the possibles solutions have to be tested to ensure they pass this specification.

3.4.2 Method Bases

The testing method is deigned to evaluate the achievment of global coverage. The main variables needed for the development of it are the following:

Coverage Testing Method Variables	
typeC	Type of constellation
ε	Elevation angle [°]
h	Height [km]
in	Inclination angle [°]
n_p	Number of Planes
N_{pp}	Number of Satellites per plane

Table 3.4.1: Coverage Testing Method main Variables

It consist in evaluating all the possible variables combinations within established margins and testing them to know if they fulfill the determined conditions than ensures global coverage.

3.4.2.1 Global Coverage Conditions

Same plane condition

In order to fulfill the desired coverage, the distance between two satellites on the same plane must not be more than two times the central angle β. This condition is visually represented in Figure 3.4.1 .

Different plane condition

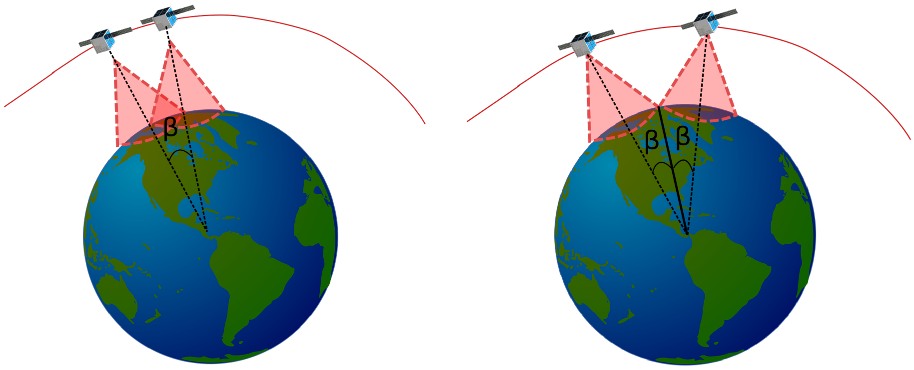


Figure 3.4.1: Geometrical conditions needed to fulfill global coverage.
On the left: Condition between satellites of different planes.
On the right: Condition between satellites of the same plane

To accomplish the coverage requirments, the distance between two satellites on different planes must not be more than the central anglep β. This condition is visually represented in Figure 3.4.1 .

3.4.2.2 Results of Testing Method

A MATLAB routine has been designed to compute the describe algorithm. In this phase different values of all the variables have been computed in order to found the most suitable solution. The values tested are the following:

Coverage Testing Method Variables	
typeC	[180 210 225 240 360] [º]
ε	[20] [º]
h	[540-550] [km]
in	[70-80] [º]
n_p	[5-12]
N_{pp}	[10-24]

Table 3.4.2: Testing Values for the Coverage Testing Method

General Solution

The program has been runned for all the range specified above to see the evolution of a satellite network configuration regarding the variation of the orbital parameters in order to find the best constellations options.

As it can be deduced both the number of planes and satellites decreases when increasing

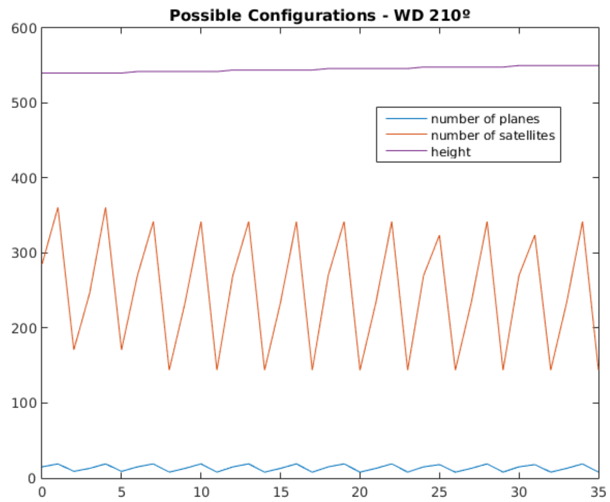


Figure 3.4.2: Possible satellite configurations for a 210° Walker Delta configuration

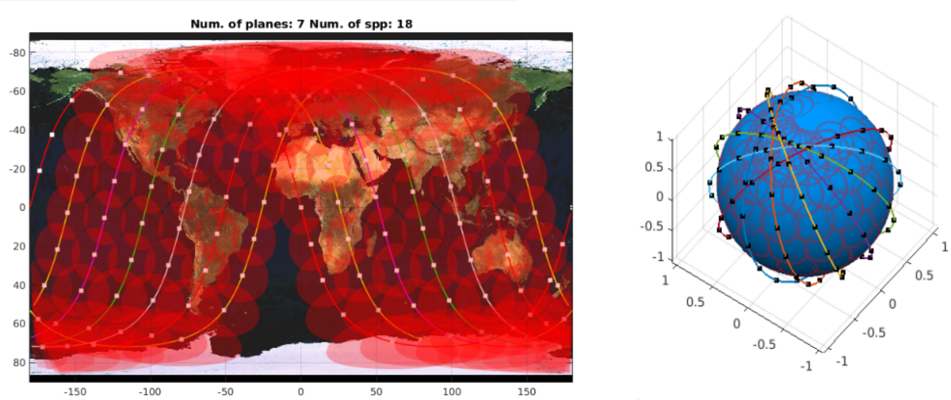


Figure 3.4.3: Ground track and spherical representation for a 180° Walker Delta configuration

height because as explained before the footprint of the satellites gets incremented with height. If height is left as a constant, a less intuitive results are obtain. We have now different configurations in terms of number of satellites an planes due to the variation of the inclination angle of the planes. In the Figure 3.4.5 is shown the results obtained for one of the analysed configurations.

Once all the possible configurations have been computed, the ground track of three of them has been ploted to visualy check the coverage obtained.

Conclusions

From the developed code that runs all the parameters needed to define a Walker Delta configuration it is possible to obtain for a chosen requirment which are the optimum

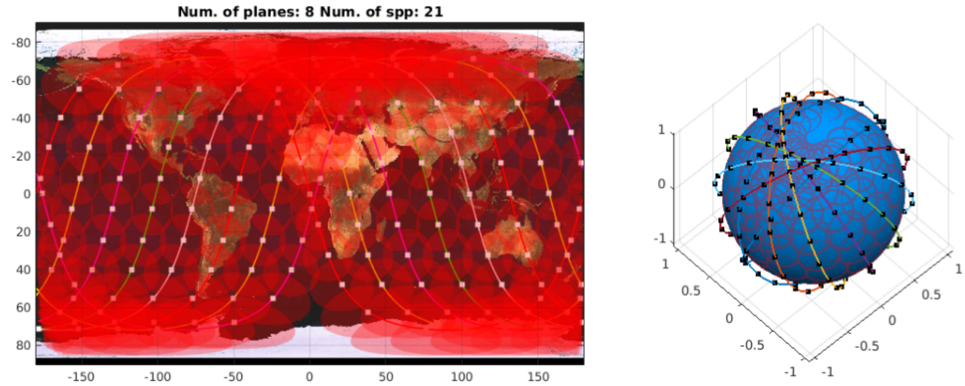


Figure 3.4.4: Ground track and spherical representation for a 210° Walker Delta configuration

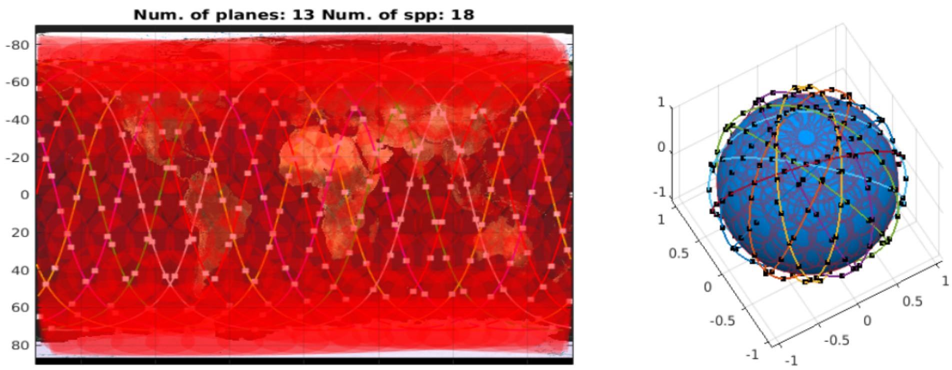


Figure 3.4.5: Ground track and spherical representation for a 360° Walker Delta configuration

configuration. Therefore defining the criteria in function of the constellation needs it will be possible to optimize the design. The configurations that will be later considered to perform an analysis of weighted weights are extracted from this routine.

Chapter 4

Orbit Perturbations

4.1 Sources of Perturbation

4.1.1 Introduction to Orbit Perturbations [?]

In this chapter it is seen how the designed orbit configuration varies in time due to external perturbation sources. While some of them can be neglected, there are other of major importance to the future of the constellation. For instance, atmospheric drag determines in plenty of cases the lifetime of the constellation. A first classification of perturbations depending on the time in which their effects are present is the following:

- Secular terms (Sec): They depend on the semimajor axis, the excentricity and the inclination.
- Short Period terms (SP): They depend on the anomalies, this leads to a strong variation in each period.
- Long Period terms (LP): They depend on the argument of the periapsis or the ascendent node.

Even though most of the outter space is vacuum, there ideal models need to consider some factors that escape the typical two body problem. For instance, we can no longer consider Earth as a punctual mass, neither the atmospheric density equal to 0. To enumerate, here is a typical list of the main perturbation sources:

Sources of perturbation:

- Gravity Field of the Central Body
- Atmospheric Drag
- Third Body perturbations
- Solar-Radiation Pressure
- Other Perturbations

All the perturbations can be deeply studied. Consequently, analytical solutions are very hard to find, and even they were found, they do not show clealy a meaning or are not really useful. Instead, there are two mainly used approaches:

- Special Perturbacion: Step-by-step numerical integration of the motion equations with perturbation.

- General Perturbation: Through analytical expansion and integration of the equations of variation of orbit parameters.

The Approach of the Perturbations Study For the purposes of these study the different approaches will be assessed. The first analysis will discuss which of the peturbations are the most significant to the study. This analysis will be done considering General Perturbation Techniques. In a deeper second analysis, the two approaches for the perturbations will be assessed and compared considering only the most significant perturbation sources.

4.1.2 Gravity Potential of Earth

Earth's aspherical shape can be modelled as a sum of terms corresponding to the Legendre polynomials. These polynomials can be empirically measured and consider radial symmetry. If one would like to compute also variations in longitude, then should use associated Legendre polynomials.

$$V(r,\delta,\lambda) = -\frac{\mu}{r}\left[\sum_{n=1}^{\infty}\left(\frac{R_e}{r}\right)^n \sum_{m=0}^{n} P_{nm}cos(\delta)(C_{nm}cosm\lambda + S_{nm}sinm\lambda)\right] \tag{4.1.1}$$

General Legendre associated polynomials developed Gravitational Potential

$$V(r,\delta) = -\frac{\mu}{r}\left[1 - \sum_{n=2}^{\infty} J_n \left(\frac{R_e}{r}\right)^n P_n(sin\delta)\right] \tag{4.1.2}$$

General Legendre polynomials developed Gravitational Potential

For Earth, the J_n coefficients are the following:

$J_2 = 0.00108263$ $J_3 = -0.00000254$ $J_4 = -0.00000161$

Given this distribution, the only sognificant term J_2.

$$V(r,\delta) = -\frac{\mu}{r}\left[1 - \frac{1}{2}J_2\left(\frac{R_e}{r}\right)^2 (1 - 3sin^2\delta\right] \tag{4.1.3}$$

Aproximated Gravitational Potential

If we integrate the force that derives from this potential we can afterwards compute the effect of J_2 On the different orbtial elements:

- $\Delta a = 0$

- $\Delta e = 0$
- $\Delta i = 0$
- $$\Delta\Omega = -3\pi\frac{J_2 R_e^2}{p^2} cos\, i\ [rad/orbit] \tag{4.1.4}$$
- $$\Delta\omega = \frac{3}{2}\pi\frac{J_2 R_e^2}{p^2}(4 - 5sin^2\, i)\ [rad/orbit] \tag{4.1.5}$$

4.1.3 Atmospheric Drag

In order to compute the effect of the remaining atmosphere we use the typical definition of atmospheric drag knowing a drag coefficient:

$$\vec{a}_{drag} = \frac{1}{2}\frac{C_d A}{m}\rho v_{rel}^2 \frac{\vec{v}_{rel}}{|\vec{v}_{rel}|} \tag{4.1.6}$$

The **ballistic coefficient** B_c is defined as $\frac{m}{C_d A}$, characterizing the behaviour of the satellite against atmospheric drag.

Modelling the Atmosphere
There are several modells for the atmosphere. For instance, the most commonly used, the exponential model:

$$\rho = \rho_0 e^{-\frac{h-h_0}{H}} \tag{4.1.7}$$

$$H = \frac{kT}{Mg} \tag{4.1.8}$$

Where:

Exponential Atmosphere Variables	
ρ	Density at given height
ρ_0	Density at a reference height
h	Height over the ellipsoid
h_0	Reference height
H	Scale Height
k	Boltzmann Constant
T	Temperature
M	Molecular Weight
g	Gravity

Table 4.1.1: Exponential Atmosphere Model main Variables

In addition, other models for the exospheric temperature and the molecular weight need to be used. For this study the ones proposed by The Australian Weather Space Agency are used.

In addition, it is important to note that the following phenomena interfere with the previsions:

- Diurnal Variations
- 27-day solar-rotation cycle
- 11-year cycle of Sun spots
- Semi-annual/Seasonal variations
- Rotating atmosphere
- Winds
- Magnetic Storm Variations
- Others: Tides, Winds,...

Again, if we integrate this force in a period of time, considering the orbit nearly circular, we obtain:

$$\Delta r = -2\pi \rho r^2 / B \ [/orbit] \tag{4.1.9}$$

4.1.4 3rd Body Perturbations

The effects of this extra bodies in the system can be computed considering the motion equations. However, some aproximations can be found in the reference as:

$$\dot{\Omega} = \frac{A_m + A_s}{n} cos\, i \; [^o/day] \tag{4.1.10}$$

$$\dot{\omega} = \frac{B_m + B_s}{n} (4 - 5sin^2\, i) \; [^o/day] \tag{4.1.11}$$

Where n stands for the rate of rotation in orbits/day. In that case, the A_m, A_s, B_m and B_s coefficients take as values:

	$A_m + A_s$	$B_m + B_s$
Moon	-0.00338	0.00169
Sun	-0.00154	0.00077

Table 4.1.2: Third Body Perturbations Coefficients

4.1.5 Other Perturbations

In this bag the following low-intensity can be classified:

- Solar Radiation Pressure
- Solid-Earth and Ocean Tides
- Magnetic Field
- South Atlantic Anomaly

4.2 Significant Perturbations

Propagation Algorithm

Given the definitions and aproxmations to compute perturbations described in the previous section, a propagation in time for the change in orbital parameters is solved. The results are plotted in the graph below:

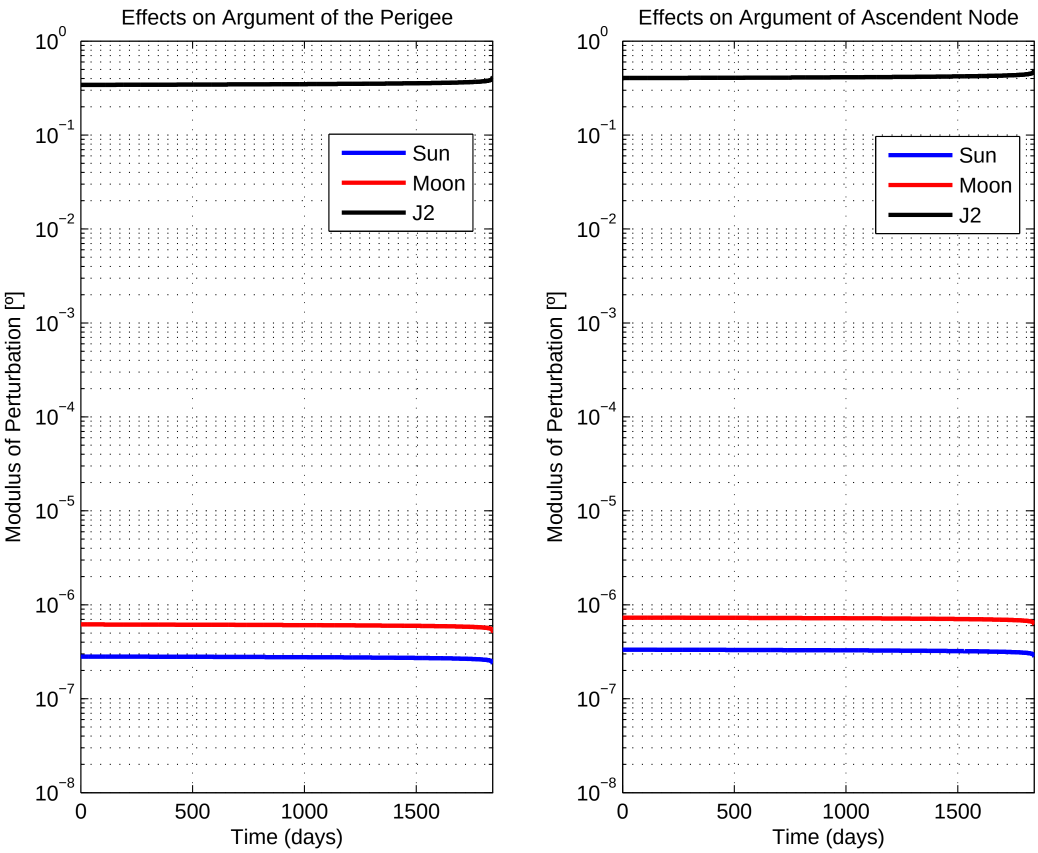


Figure 4.2.1: Logaritmic plot of the modulus of the increases in Angular Arguments of the orbit

As it can be seen, the perturbations caused by 3rd bodies are several orders of magnitude below the order of magnitude of the variation caused by Earth's oblateness. It is also remarkable that the moon has a higher effect than the sun given the relative distance to Earth, even if the sun is way more massive.

Another important obsevation is that given the very low eccentricity we are considering, the deviation of the argument of the perigee does not affect the performance of the constellation. In other words, since the orbits are considered almost circular there is not a defined Perigee for the orbit.

In conclusion

The effects of the Moon and the Sun are neglected in comparison with the effects of J2 for the Argument of the ascendent node as well as for the argument of the Perigee.

4.3 Orbit Decay

In this chapter the effects of the main perturbations are deeply studied. Firstly, an introduction on the effects of Earth's oblateness on the orbital parameters. Secondly and in more detail, the effects of Atmospheric drag. This is significant because it deviates the power and mass budget to engines and propellant.

4.3.1 Effects on the Ascention Node

4.3.1.1 Introduction

Due to the non sphericity of the Earth, two deviations exist in terms of perigee and ascendent node. This perturtions are related to the J2 effect descrived before. Both effects are related to the orbital planes inclination angle, so depending in which inclination they are positionated, the perturbation will be more or less significant.

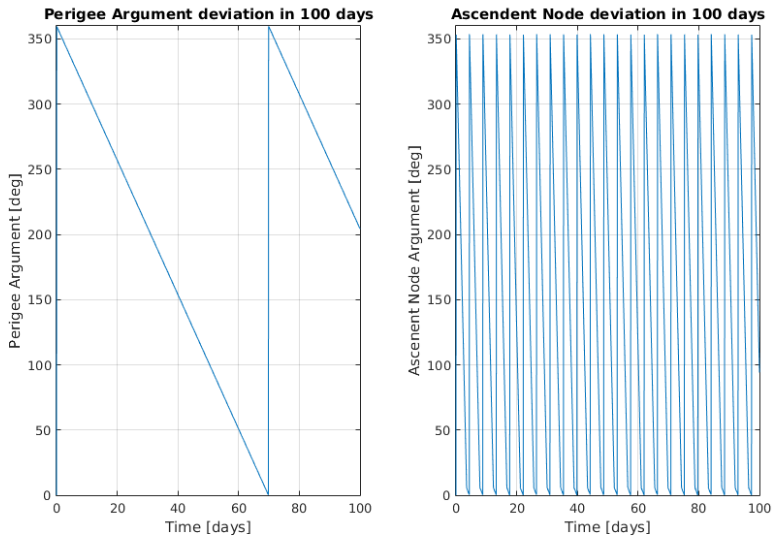


Figure 4.3.1: Ascention node perturbation
On the left: Perigee deviation in terms of time.
On the right: Ascending node deviation in terms of time

4.3.1.2 Perigee Effect

The Perigee effect is the responsible of the rotation of the orbit regarding the Earth and is found inside the orbital plane itself. Therefore the perigee of an eliptical orbit is not static in an Earth's point but moves around it.

This effect is noticed when having eliptical orbits. Consequently Astrea constellation will not be affected because the satellites describe almost circular orbits.

4.3.1.3 Ascention Node

In this case the perturbation affects the rotation of the orbital plane. So the plan longitude variates with time. That means, that if we had just one orbital plane it would not cover always the same fraction of Earth.

This effect is noticed when having planes with different inclinitations. That is not Astrea's constellation case since all its planes are positioned in the same inclination angle.

4.3.1.4 Conclusion

As explained, both perturbations do not affect Astrea's constellation so they will not be considered as atctive agents on the orbit decay proces

The Figure 4.3.1 shows the propagation in time of both effects which are periodic due to the constant velocity of orbits.

4.3.2 Effects of the Solar Cicle

It is important to consider many parameters when calculating the orbital decay of a satellite. The most important of these parameters for LEO based constellations is drag. As discussed in other chapters, the drag of a satellite depends on the coefficient of drag, its surface, the density of the air and the velocity at which operates. Solar cycles will directly affect the density of the upper atmosphere. This phenomena is relevant when calculating the drag of the satellite and therefore is essential to compute the orbital decay.

Solar cycles are periodic changes in the Sun's activity of approximately 11 years. In each period a solar maximum and minimum can be determined, referring to the amount of periods of sunspot counts. The intensities for these periods vary from cycle to cycle.

Different studies have been made throughout the 20th century cycles. In order to understand the change density of the air changes as consequence of these solar cycles we considered the result data of an old study regarding the 19th solar cycle, which had a duration of 10.5 years between 1958 and 1968. This solar cycle had the highest maximum smoothed sunspot number ever recorded (since 1755), which was of 201.3. This maximum value was recorded in March 1958. This value is high in comparison to other cycles, especially when comparing it to the current 24th solar cycle. In this chapter an analysis will be developed in order to study the influence of the solar cycles on the drag of our satellites. Data obtained of [?].

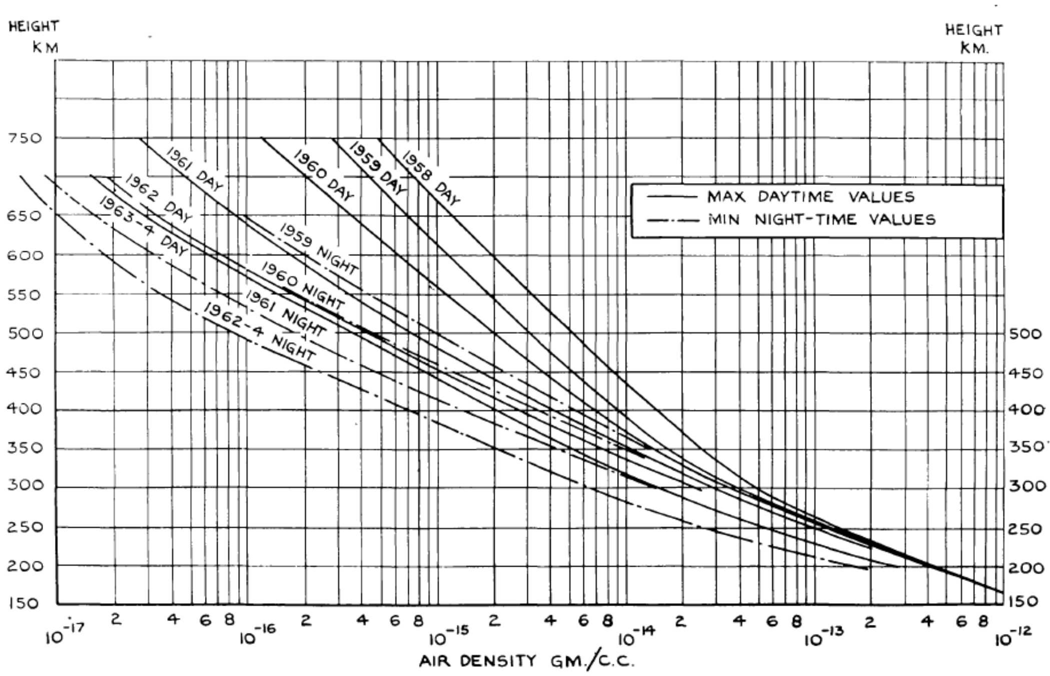


Figure 4.3.2: Deviation of densities in the upper atmosphere due to the 19th solar cycle. Source: [?]

At 550 km:

Year	D/N	Density at 550km [g/cc]
1958	Day	3.2E-14
1958	Night	5.0E-15
1964	Day	1.35E-15
1964	Night	3.35E-16

These values referring to day and night are the densities of the upper atmosphere at 550 km of altitude respect to the surface of the Earth. The upper atmosphere densities rise during the day following the increase of temperature caused by the radiation of the Sun whereas these values are reduced at night. The orbital decay is on the order of several years whereas these deviations appear every few hours. Thus, in order to compute the orbital decay we will not be taking into account these daily deviations but rather a main value. Therefore the mean density for 1958 will be of 1.85E-14 g/cc and the solar minimum's density of 1964 will be of 8.4E-16 g/cc.

In order to analyse how these values may apply to our constellation we first must adjust these - which belong to the 19th solar cycle - to those of the current 24th cycle, which is noticeable less intense. A way of operating this adjustment is comparing the mean solar maximum achieved by each cycle. The maximum monthly smoothed sunspot number of the 19th cycle had a value of 201.3 and a minimum of 9.6 whereas the current 24th ranges between 11.7 and 81.9 approximately. This means that for the 19th cycle a total deviation of 191.7 was measured whilst for the 24th cycle this deviation was only of 70.2. This is crucial if we want to analyse the solar maximum densities.

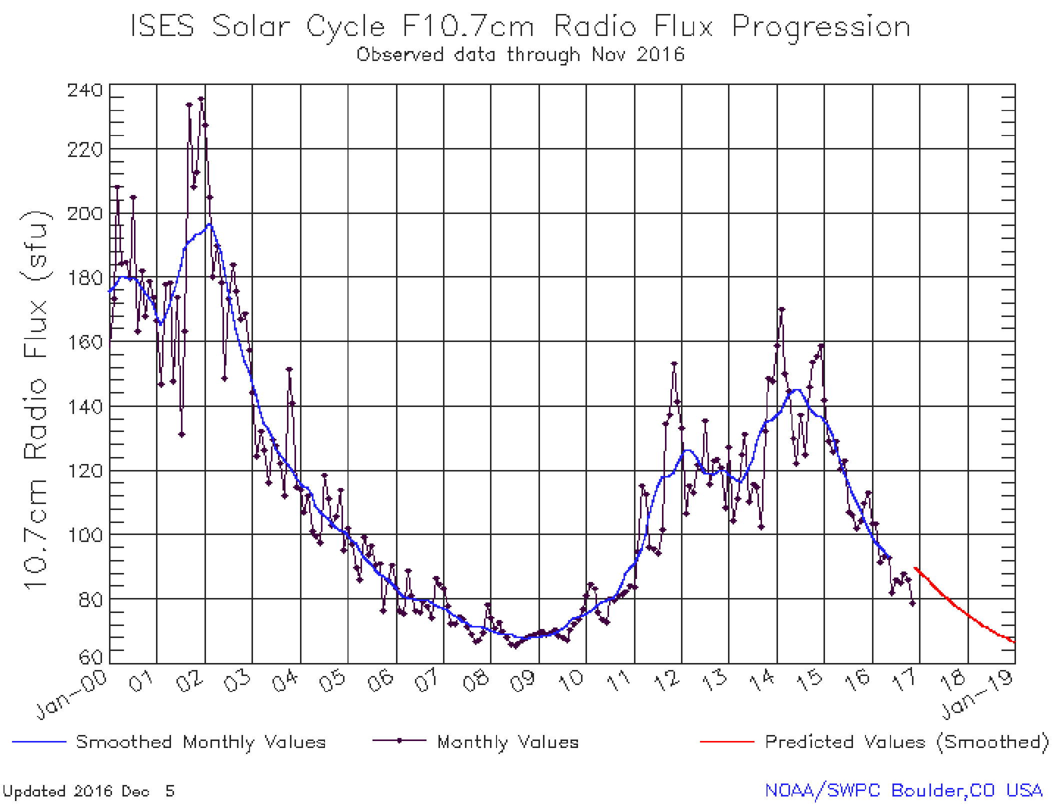


Figure 4.3.3: Measured intensities of the 23rd and 24th solar cycles. Source: NOAA

We must now adjust the mean constant density defined initially to the conditions that this 24th cycle imposes. It is important to note that our satellites will be launched in 2017, and that the 24th cycle is currently decreasing its intensity. Thus, our calculations will be near the conditions of solar minimum, meaning that the drag of our satellite will be smaller than first considered.

Our new approach to the density of the atmosphere at 550 km is near the first approximation, but will consider that we are now entering the solar minimum which will remain more or less constant until 2022. As discussed before, the solar minimum

Table 4.3.1: Selected data to compute orbit decay extracted from figure ??

Selected Values	
Year	F10 Radio Flux
2002	195
2004	115
2009	70
2013	120
2016	100

represents a singularity with a minimum density of 8.4E-16 g/cc. The approximation taken will be the resulting constant value which represents the mean smoothed densities between 2017 and 2022.

The final density at 550 km considering the solar minimum during 2017 to 2022 will be of 2.0E-15 g/cc.

4.3.3 Orbital Decay Propagation Results

4.3.3.1 Introduction

In this section a first aproach of the drag computation have been done in order to determine the orbit decay and consequently compute how much time a satellite last untill it reenters the Earth atmosphere.

4.3.3.2 Drag Computation Algorithm

Given the definitions to calculate orbital perturbations described in **??** a computation of the atmosphere drag has been done together with the computation of the other main perturbations that have been discussed in previous sections.

As explained in the last section the atmospheric drag depens on the drag's coefficient and it surface, that are constant values, on the velocity at which the satellite operates and on the air density.

So in orther to see the effects of variations in air density the orbit decay has been estimated and plotted for several F10 Radio Flux values corresponding to different moments of a solar cycle. (This data has been extracted from the figure **??**).

The data selected and the results obtained are shown in 4.3.1 and 4.3.4 respectively.

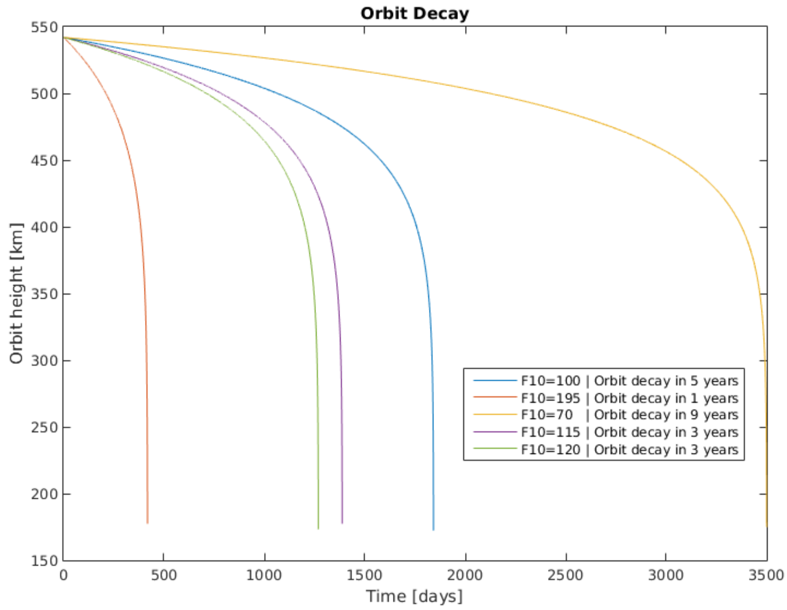


Figure 4.3.4: Orbit Decay computed for several values of

As it can be seen, the orbit decay strongly depens on the positioning in time of a solar cycle. (In 7 years the difference in lasting time of the satellite is reduced in 4 years).

In conclusion The lasting time in orbit of satellites is affected by period of the the solar cycle we are in. According to the data then Astrea's satellites will have an aproximated orbit decay of 5 years.

4.3.4 Dynamic Orbit Decay Computation

4.3.4.1 Introduction

In this part of the chapter the orbital is studied using the model of special perturbations, which as previously defined, is the one that uses a numerical step-by-step integration. There are three manly used methods to study the dynamic propagation of an orbit, which are:

Cowell's method: This is the simplest method since it does not require any assumption or approximation. It is based on quantifying the accelerations produced by the perturbations and adding them to the dynamic equation of a Keplerian orbit 1.2.3 leading to:

$$\frac{d^2\vec{r}}{dt^2} = -\frac{\mu}{r^3}\vec{r} + \vec{a}_p \tag{4.3.1}$$

Where $\vec{a}_p$ is the acceleration produced by the perturbations. This second order differential equation is the one that must be integrated in order to propagate the orbit. Although the formulas and application of this method are simple, this does not imply that it lacks robustness or precision. Its results are as good as any of the following two methods but the major drawback of *Cowell's method* is that it requires smaller time-steps being therefore slower (in terms of computation speed).

Encke's Method: This method is based on correcting the defects of the previous method. Encke uses a schema based on what is called *predictor-corrector*. First, it evaluates the orbit as if it were a Keplerian orbit (i.e. without perturbations) and then it integrates only the perturbations to correct the deviation caused by considering the unperturbed orbit. Its advantage over Cowell's method is clear, since it only integrates perturbations, and since these vary less over time than the position itself, we can relax the integration by increasing the time step. In short, this scheme is faster but also more complex to program than the one proposed by Cowell.

Variation of the parameters: This method, developed by Lagrange, is based on considering the orbit as a succession of Keplerian orbits, each of them being tangent to the satellite orbit at a certain point. Thus we can obtain differential equations that model the variation of the orbital parameters as a function of time.

The formulations and schemes followed by each of these methods can be found in any reference dealing with orbital mechanics. For example, the reader can refer to [**?**] or the chapter 20 of [**?**] to obtain more detailed information about these methods.

For the purposes of this study, implementing the simplest method is enough. As it has already mentioned, it is based on adding the perturbations (discussed at the beginning of this chapter) to the dynamics equation. A *Matlab* routine has been developed that follows the next scheme:

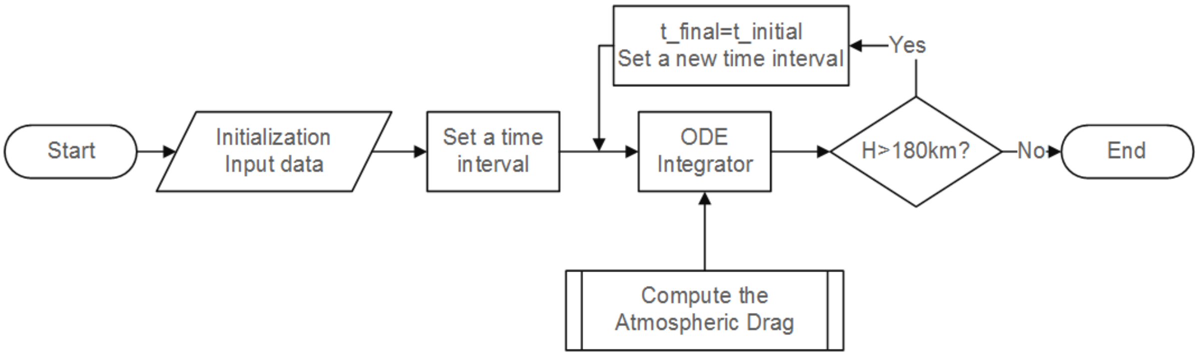


Figure 4.3.5: Algorithm of resolution used to solve the orbital propagation.

The disturbances that the routine includes are:

- The potential field of the earth.
- The Atmospheric drag.
- The influences of 3 bodies.
- Solar Radiation Pressure.

As it has been seen in previous sections, the only truly significant perturbation for the orbital decay at the altitude in which the constellation is located is the one caused by the atmospheric drag. Thus, other contributions have been deactivated to speed up the calculation. Therefore, explaining the formulation used to obtain the accelerations caused by these perturbations is not of interest for the development of the study. However, the following are the sources from which they were obtained:

- The calculation of the Earth gravity Potential uses the equation 4.1.1. Following the indications of [**?**] both the Legendre polynomials and the parameters C_{nm} and S_{nm} can be obtained.
- the equations present in **??** have been used to compute the perturbations due to other bodies,
- For Solar Radiation pressure the formulation used is the one presented in [**?**] including a 'shadow factor' (if the earth is between the sun and the satellite, the latter will not receive direct radiation from the Sun) modeled by a normal statistical distribution.
- For the calculation of Drag, the equation 4.1.6 and the atmosphere model presented in the same section have been used.

To be able to integrate the system we must take into account that, in fact, as we work in Cartesian coordinates, it is a system of three equations. Moreover, since it is a second-order equation we must rewrite it as a first-order system. Let $x_1 = r = (x, y, z)$ and $x_2 = \dot{x}_1 = \dot{r} = (vx, vy, vz)$. Therefore:

$$X = \begin{pmatrix} x \\ y \\ z \\ v_x \\ v_y \\ v_z \end{pmatrix} \Rightarrow \dot{X} = \begin{pmatrix} v_x \\ v_y \\ v_z \\ \ddot{x} \\ \ddot{y} \\ \ddot{z} \end{pmatrix} = \begin{pmatrix} v_x \\ v_y \\ v_z \\ a_{p,x} - \frac{\mu}{r^3}x \\ a_{p,y} - \frac{\mu}{r^3}y \\ a_{p,z} - \frac{\mu}{r^3}z \end{pmatrix} \tag{4.3.2}$$

To integrate this system, you can use the *Matlab* built-in function **ode45**, which is a runge-kutta 4-5 with a variable step control that basically modifies the time step if the error is too large. Also, the **juliandate.m** function (included in the Matlab Aerospace module) have been used. It calculates the Julian Date,that is the number of days since noon Universal Time on January 1, 4713 ECB (On the Julian calendar).

4.3.4.2 Results

A simulation has been executed with the same parameters as in the previous section. After 932 seconds of computation, the results obtained are shown below:

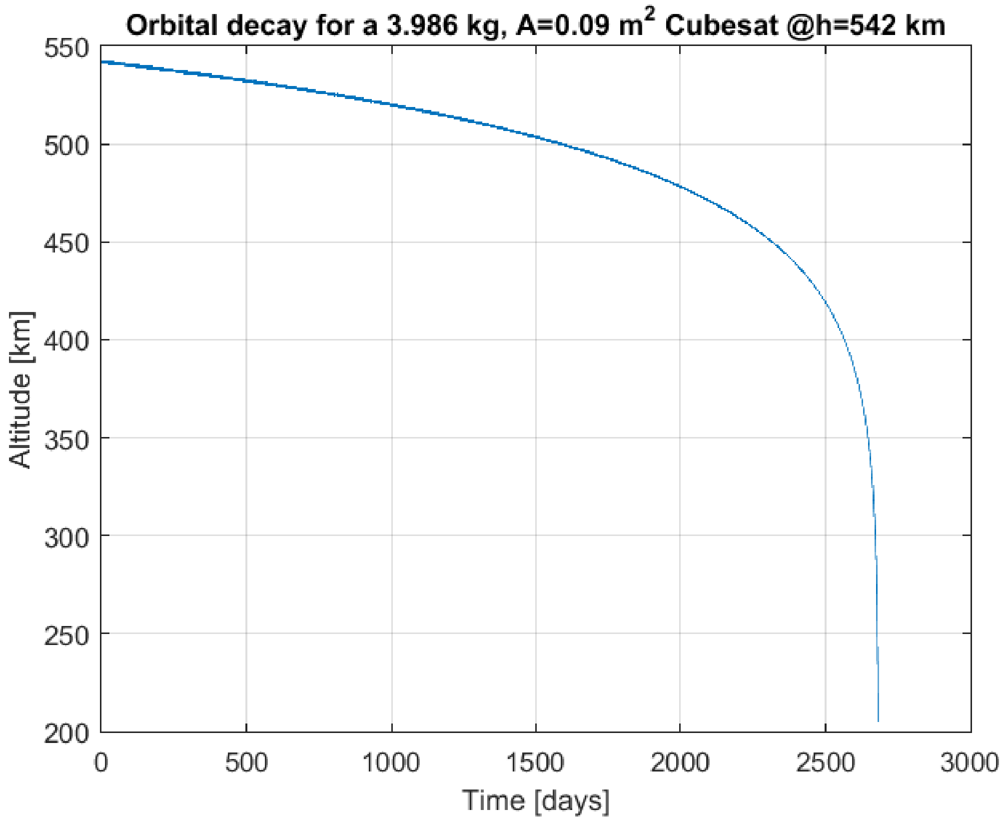


Figure 4.3.6: Orbital decay of the satellite.

As it can be seen, the estimated orbital decay for a satellite like Astrea's Cubesat is about 2700 days or, what is the same, 7.4 years. This estimation and the temporal evolution of the altitude is in agreement with the results obtained by the semi-analytic method. It is therefore verified that for a preliminary analysis and the respective modifications that it can present (i.e. changes in weight, changes in area, initial height, geometry of the orbit) it is enough with the results obtained by the semi-analytic study, which do not require almost computation time (only a few seconds), avoiding the expense of computing resources that would produce a dynamic simulation for every modification.

4.4 Orbital Station-Keeping

We will study:

- Increased height
- Thrusters

4.4.1 Raising the orbit height to increase Lifetime

The key to understand this solution is to see from another point of view the atmospheric drag phenomena. Once we have designed the constellation to provide certain coverage to specific points of the globe, the action of increasing the height of the orbit has the effect of increasing the footprint area on the surface of the earth. As the constellation is set, the time that take the satellites to reach the design height is extra lifetime.

From this point of view, the atmospheric drag phenomena can be recomputed and plotted it in this new way:

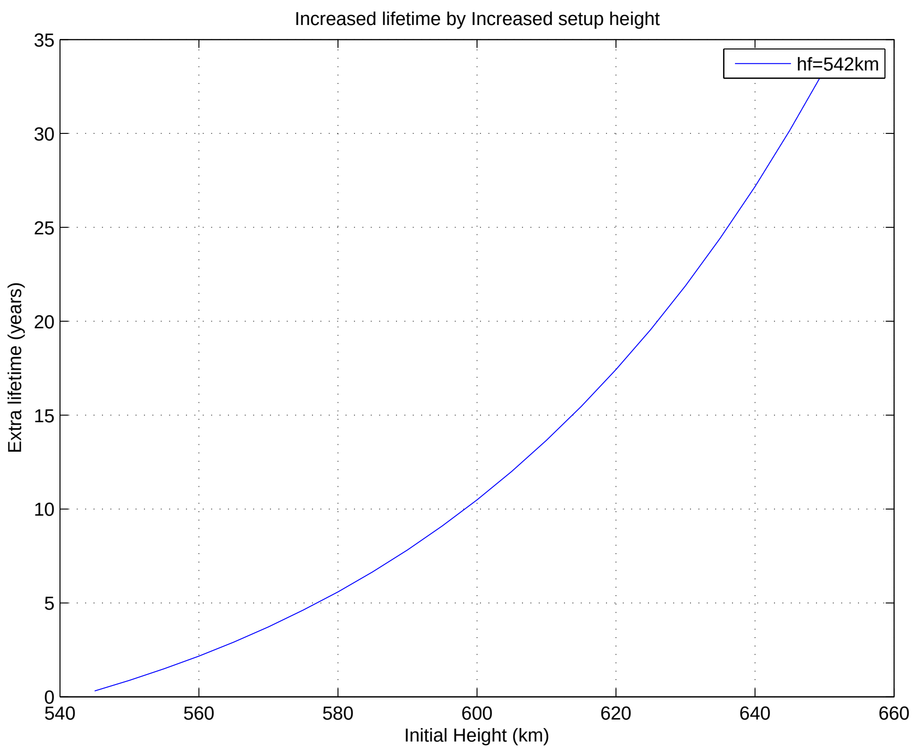


Figure 4.4.1: Increase in the Lifetime obtained by setting the constellation in a higher orbit

As it can be seen, the lifetime increases radically with time. However, this is a dangerous solution, since the coupling with another design parameters is compromised. To list the complications that can lead to:

- **Clients:** With the current technology, the satellites currently in orbit are set to point towards Earth. This means, if the contellation's satellites are at a higher orbit, the contact is impossible. As the market study reveals, it is important to place the satellites as low as possible.
- **Spacecraft Subsystems:** A higher orbit means a higher gain for the antennas and therefore an increase in the required power.
- **Constellation Reconfiguration:** The overall time to reconfigurate the constellation increases with height, since the period of the transition orbits is higher.

In conclusion

This tool is a very powerful option to deal with the orbit decay, even though it is not exactly an operation of Station Keeping itself. Given the high correlation it shows with another subsystems, the possibility of using it needs to be considered while the other design decisions are taken.

4.4.2 Using Thrusters to increase Lifetime

In order to maintain the configuration of the constellation for a longer time, a thruster is installed in each satellite to correct the decrease in altitude due to the orbit decay. The most optimal way to maintain the altitude is through a low-thrust maneuver. However, since this is a preliminar study, the calculations will be computed for a Hohmann transfer maneuver, which is simpler and more effective, but requires more propellant and greater increases of velocity. That is, by computing the velocity and propellant needed for a Hohmann maneuver, the results will be safe for a low-trust maneuver, because the late one requires less energy.

4.4.2.1 Energy equation

The deduction of the equations needed to solve the Hohmann maneuver begins with the energy equation:

$$\frac{V^2}{2} - \frac{\mu}{r} = -\frac{\mu}{2a} \tag{4.4.1}$$

where V is the orbital velocity of the satellite, r is the distance from the focus, a the semimajor axis of the orbit and μ the gravitational constant of the attracting body, in this case, the Earth. This expression shows that the total energy of the satellite equals the sum of its kinetic and potential energy (per mass unit).

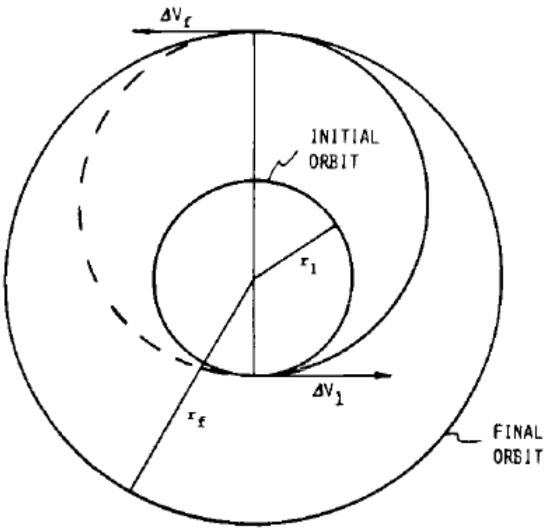


Figure 4.4.2: Hohmann transfer. Extracted from [8]

This equation can be arranged to obtain the velocity of the satellite. In the case of a circular orbit, the radius is constant, and equal to the semimajor axis. Replacing $a = r$ in the energy equation and after some operations, the expression of the velocity of a circular orbit is obtained:

$$V_c = \sqrt{\frac{\mu}{r}} \tag{4.4.2}$$

As it can be deduced from the energy equation, a change in orbital velocity leads to a change in the value of the semimajor axis. This property is used in satellites to change their orbit through a velocity increment ΔV. This process is called an orbital maneuver.

4.4.2.2 Delta-V

If the velocity increment ΔV is done instantaneously, the maneuver is called an impulsive maneuver. The Hohmann transfer is a two-impulse transfer between coplanar circular orbits. From an inicial circular orbit, a tangential velocity increment ΔV_1 is applied to change the orbit to an ellipse. This ellipse is the transfer orbit, in which the perigee radius is the radius of the initial circular orbit and the apogee radius equals the radius of the final circular orbit. When the satellite reaches the apogee, a second velocity increment ΔV_2 is applied, so that the satellite reaches the final circular orbit with the apogee radius. If this second velocity is not applied, the satellite will remain in the elliptic orbit.

With the energy equation defined above, it is easy to determine the velocity of the satellite in each orbit. The first orbit and the final ones are circular:

$$V_1 = \sqrt{\frac{\mu}{r_1}} \tag{4.4.3}$$

$$V_f = \sqrt{\frac{\mu}{r_f}} \tag{4.4.4}$$

The velocity in the transfer orbit can be easily calculated with the energy equation applying the definition of the semimajor axis of an ellipse:

$$a = \frac{r_1 + r_f}{2} \tag{4.4.5}$$

The velocities in the perigee and apogee are:

$$V_p = \sqrt{\frac{2\mu r_f}{r_1(r_1 + r_f)}} \tag{4.4.6}$$

$$V_a = \sqrt{\frac{2\mu r_1}{r_f(r_1 + r_f)}} \tag{4.4.7}$$

Therefore the velocity increments are:

$$\Delta V_1 = V_p - V_1 = \sqrt{\frac{2\mu r_f}{r_1(r_1 + r_f)}} - \sqrt{\frac{\mu}{r_1}} \tag{4.4.8}$$

$$\Delta V_2 = V_f - V_a = \sqrt{\frac{\mu}{r_f}} - \sqrt{\frac{2\mu r_1}{r_f(r_1 + r_f)}} \tag{4.4.9}$$

4.4.2.3 Time

It is also necessary to know the time needed to do the maneuver. This time is equal to half of the period of the transfer ellipse:

$$t = \frac{T}{2} = \frac{1}{2}\sqrt{\frac{4\pi^2 a^3}{\mu}} \tag{4.4.10}$$

4.4.2.4 Propellant

In order to know the mass of propellant needed in the maneuver, the Tsiolkovsky rocket equation is applied:

$$\Delta V = g_0 I_{sp} \ln \frac{m_1}{m_f} = g_0 I_{sp} \ln \frac{m_1}{m_1 - m_{prop}} \tag{4.4.11}$$

where $\Delta V = \Delta V_1 + \Delta V_2$ is the total velocity increment of the maneuver, g_0 is the Earth's gravity, I_{sp} is the specific impulse of the thruster used, m_1 is the initial mass of the satellite, m_f is its final mass and m_{prop} is the mass of propellant used in the maneuver.

$$m_{prop} = m_1\left(1 - \exp\left(-\frac{\Delta V}{g_0 I_{sp}}\right)\right) \tag{4.4.12}$$

Thrust	100 μN
Specific Impulse	2150 s

Table 4.4.1: Simulation Thruster Parameters

4.4.2.5 Orbit maintenance

As explained at the beggining of the section, the orbital maneuvers exposed are intented to maintain the altitude of the satellite for a longer time and, consequently, lengthen its life. The method proposed begins when the satellite is deployed at a given height. This height will decrease due to the orbit decay, reaching a critical value, the limit altitude in which the constellation provides global coverage or another given height. Once this critical altitude is achieved, the satellite is put once again at its initial height through a Hohmann maneuver. The process is repeated several times until the satellite runs out of propellant or until it reaches its desired lifetime.
In reality the satellite will perform a low-thrust maneuver, which is more practical for an electric thruster. In this non-impulsive maneuvers, the thruster is constantly providing a velocity increment to the satellite, but it is so small that the whole transfer maneuver requires a lot of time. This means that it is not necessary to wait until the satellite reaches the critical altitude. The maneuver will start when the satellite is deployed or when it reaches a given altitude (higher than the critical altitude) so that it counteracts the effect of the orbital decay.

4.4.2.6 Results

The results are computed for a 3U CubeSat with an ion thruster. The characteristics of the thruster are the ones shown on table 4.4.1.

The first parameters to be defined are the maximum and minimum height of the orbit, mesured from the surface of the Earth. The maximum height is the altitude at which the satellite is deployed, and minimum height is the altitude at which the Hohmann transfer maneuver is applied. The satellite has to be above the minimum height to be functional.

Figure 4.4.3 is an example of the height variation of the satellite using the Hohmann maneuver to reach the maximum height once the satellite is in the minimum height. The results of this maneuver are:

Since the thruster used is an ion thruster, the specific impulse is big, and the mass propellant is very low. In this case, the variation of height due to the orbit decay is approximately 3 km per year, so the thruster needs to do a Hohmann maneuver per year. With only 10 g of propellant, the lifetime of the satellite is over 30 years.

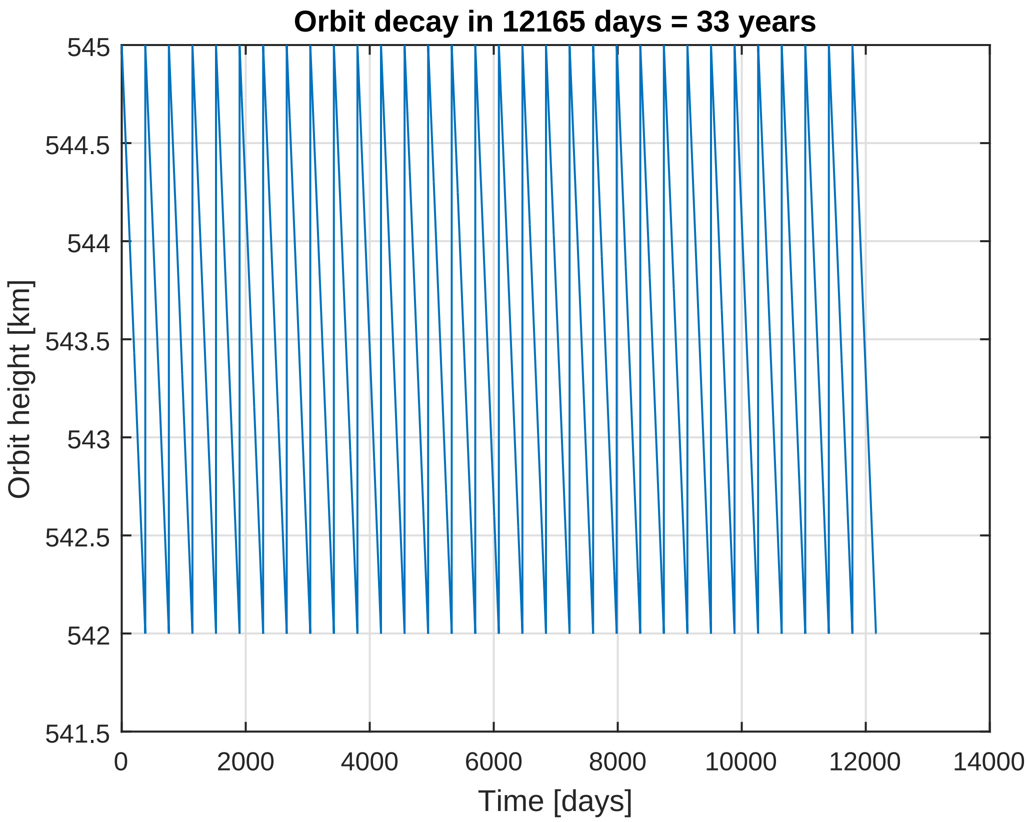


Figure 4.4.3: Height variation of the satellite

Figure 4.4.4 is another example of the Hohmann maneuver with the same amount of propellant but with a more restrictive range of operational heights, only 80 m. It should have the same shape as Figure 4.4.3, but since a lot of maneuvers are applied, the lines have overlapped. The characteristics of this maneuver are:

Comparing these results with the previous ones, it can be seen that with a more restrictive range of heights, the lifetime of the satellite is practically the same. The velocity increments are lower because the difference in the heights is extremely low, but at the same time, the satellite reaches before the minimum height and the maneuvers needed to maintain the satellite in this range are many more than on the other case. Since the ΔV budget is practically the same in both cases, it can be assured that the only difference between them is the number of maneuvers computed.

As mentioned earlier, the results obtained are for a Hohmann maneuver when in reality the satellite will compute a low-thrust maneuver, that requires less velocity increments and less propellant. In conclusion, taking into account these results, it can be stated that the lifetime of the satellite will not be determined by its orbit decay but for the failure of its systems or other external causes. It can also be assured that the satellite is capable

Maximum height	545 km
Minimum height	542 km
Number of Hohmann Maneuvers	32
Maximum ΔV_1	0,8237 m/s
Maximum ΔV_2	0,8236 m/s
Total ΔV Budget	52,7116 m/s
Propellant mass	10 g
Lifetime of the satellite	33,3288 years

Table 4.4.2: Station-Keeping with Thrusters Simulation 1 Results

Maximum height	545 km
Minimum height	544,92 km
Number of Hohmann Maneuvers	1200
Maximum ΔV_1	0,0221 m/s
Maximum ΔV_2	0,0221 m/s
Total ΔV Budget	52,7570 m/s
Propellant mass	10 g
Lifetime of the satellite	34,5726 years

Table 4.4.3: Station-Keeping with Thrusters Simulation 2 Results

of carrying enough propellant to maintain its altitude and to compute other maneuvers if necessary.

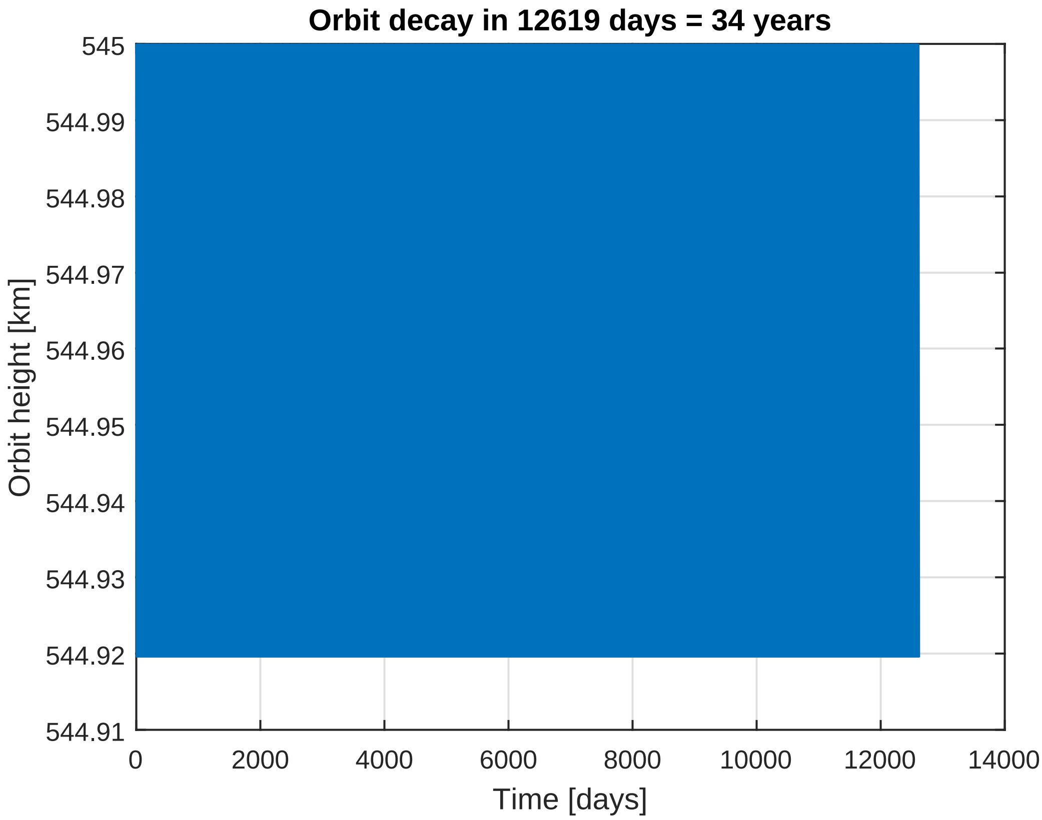


Figure 4.4.4: Height variation of the satellite with a more restrictive minimum height

Chapter 5

Constellation Design Decision

"Aerospace Engineering is the way to the universe."

Marc Cortés Fargas, 2012

5.1 Considered Designs

5.1.1 Introduction

In this chapter it is seen how the final constellation decision is made. To do that an analysis of weighted weights will be performed.

The constellations candidates selected to their later evaluation are the following:

5.1.2 Candidate 1: Polar - Global Coverage

This polar constellation (Figure 5.1.1) came from the street coverage method explained in **??**. It is a network of polar orbits that provides global coverage. Its characteristics orbit parameters are the following:

- Height: 560 km
- Inclination of the planes: 90 º
- Number of planes: 20
- Number of satellites per plane: 21
- Total number of satellites: 420
- Range of argument of ascending node: 360 º

5.1.3 Candidate 2: Polar - GS Coverage

The second candidate that will be compared is a polar orbit extracted from the coverage method explained in **??**(Figure 5.1.2). This constellation provides total coverage to the Astrea's team ground stations. The network orbits parameters are:

- Height: 550 km
- Inclination of the planes: 90 º
- Number of planes: 18
- Number of satellites per plane: 16

- Total number of satellites: 288
- Range of argument of ascending node: 360 º

5.1.4 Candidate 3 and 4: Walker-Delta GS Coverage

Two Walker-Delta constellation configurations have been also chosen due to their reduced number of planes and satellites while being able of providing total coverage on the lattitudes where the ground stations are located.(Figures 5.1.3 and 5.1.4). This constellations have been optained with the algorith explained in **??**

Candidate 3

- Height: 542 km
- Inclination of the planes: 72 º
- Number of planes: 8
- Number of satellites per plane: 21
- Total number of satellites: 168
- Range of argument of ascending node: 210 º

Candidate 4

- Height: 542 km
- Inclination of the planes: 72 º
- Number of planes: 9
- Number of satellites per plane: 17
- Total number of satellites: 153
- Range of argument of ascending node: 225 º

5.1.5 Candidate 5: Walker-Delta Lat: 0-58

Another Walker-Delta constellation has been selected with the criteria of total coverage of a range of lattitudes going from 0 to 58 (Figure 5.1.5). Therefore the parameters needed to fulfill this particular condition of the constellation obtain from **??** are the following:

- Height: 560 km
- Inclination of the planes: 72 º
- Number of planes: 14
- Number of satellites per plane: 19
- Total number of satellites: 226
- Range of argument of ascending node: 210 º

5.1.6 Candidate 6: Polar - Walker-Delta J2 + Rotació

With the goal of providing constant coverage at the Ground Stations we can design a constellation that takes profit of the rotation of the Earth. If we also consider Earth's oblateness that causes another Ω derivative with time, we can exacty compute the longitudinal position of a plane after an orbit has passed. Now, if we design the constellation in a way that this deviation after an orbit matches the separation between planes, a line of satellites will always be on the GS. (Figure 5.1.6)

- Height: 560 km
- Inclination of the planes: 72 º
- Number of planes: 14
- Number of satellites per plane: 19
- Total number of satellites: 226
- Range of argument of ascending node: 210 º

5.1.7 Candidate 7: Walker-Delta GS Coverage 3

The last configuration to be studied is a Walker-Delta constellation configuration designed to provide total coverage to the ground stations (Figure 5.1.7). It came up from candidate

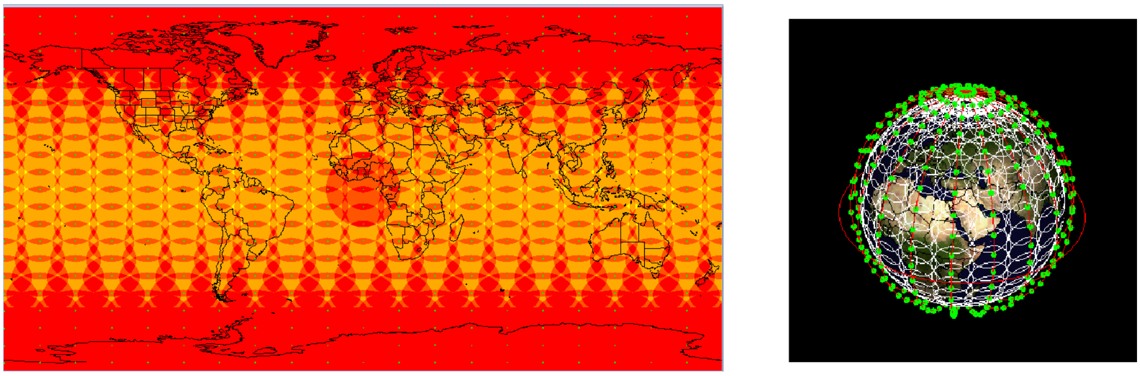


Figure 5.1.1: Candidate 1. Full Polar constellation with global coverage. h= 560km; Np=20; Npp=21; Tsat=420

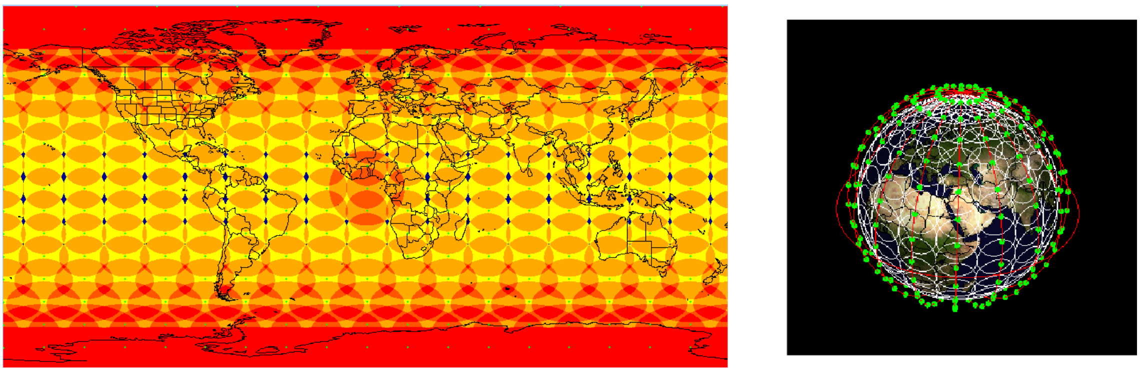


Figure 5.1.2: Candidate 2. Full Polar constellation with total ground station coverage. h= 550km; Np=18; Npp=20; Tsat=288

3 constellation adding one more plane in order to increase its global coverage and minimize the gaps. As can be seen below, its parameters are the same as candidate 3 adding a single plane.

- Height: 542 km
- Inclination of the planes: 72 º
- Number of planes: 9
- Number of satellites per plane: 21
- Total number of satellites: 189
- Range of argument of ascending node: 225 º

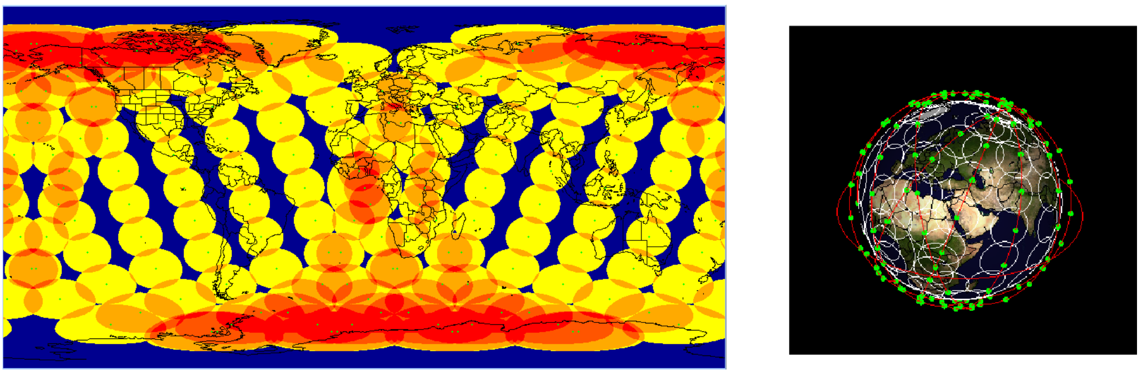


Figure 5.1.3: Candidate 3. 210^o Walker-Delta constellation configuration. h= 542km; in=72; Np=8; Npp=21; Tsat=168

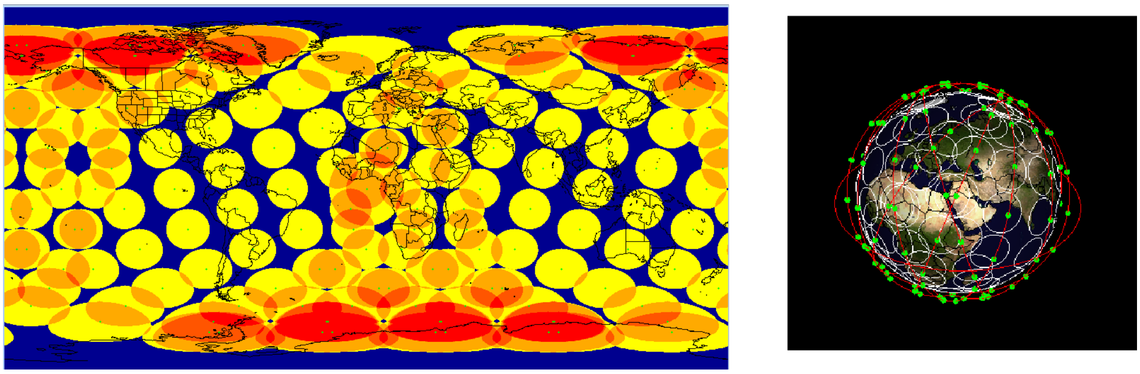


Figure 5.1.4: Candidate 4. 225^o Walker-Delta constellation configuration. h= 542km; in=72; Np=9; Npp=17; Tsat= 153

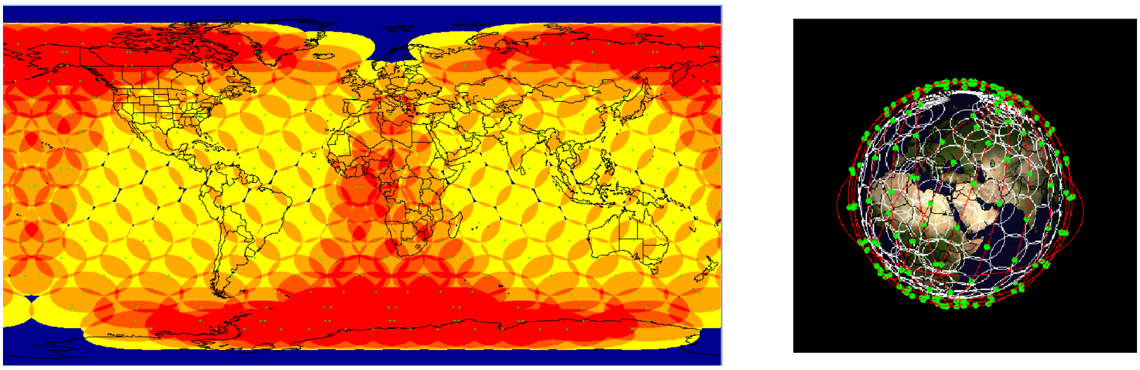


Figure 5.1.5: Candidate 5. 210^o Walker-Delta constellation configuration with total coverage of the lattitudes from 0 to 52 degrees. h= 560km; in=72; Np=9; Npp=17; Tsat= 153

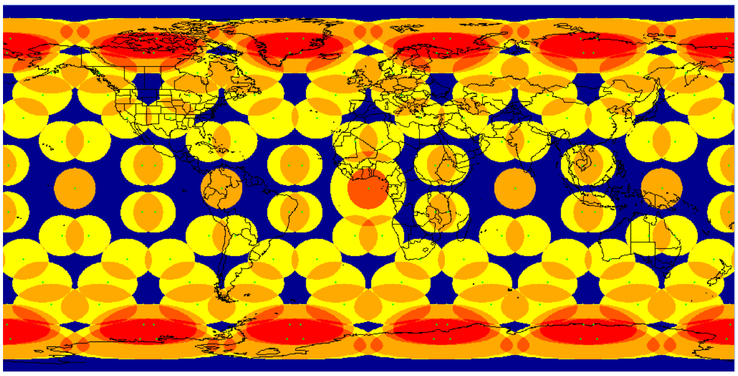
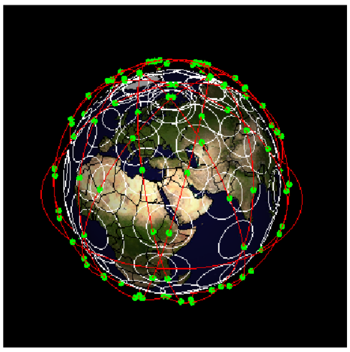



Figure 5.1.6: Candidate 6. 225º Walker-Delta constellation configuration.
h= 542km; in=72; Np=9; Npp=21; Tsat= 189

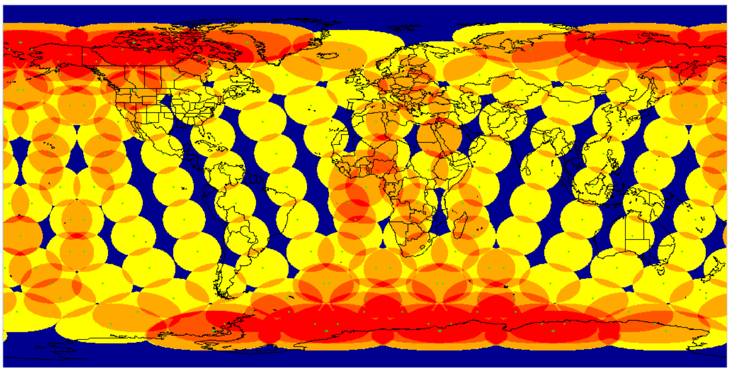
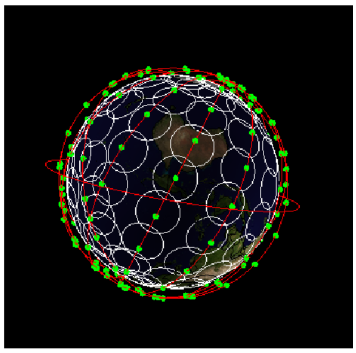



Figure 5.1.7: Candidate 7. Full Walker-Delta constellation configuration.

5.2 Constellation Performance Analysis

Even though the design requirements are included in the computation of the different configurations, it is necessary to evaluate how does the constellation perform when deployed. With this purpose, another MATLAB routine was developed.

Time factor

It is important to remark that the design methods used so far did not consider coverage in a certain period of time, but the coverage at a given instant. This section summarizes a method to compute this variation.

Quality Time

Another factor that was not considered in the design process was the pass times of the satellites. If a pass is too short the contact with the satellite cannot be produced.

5.2.1 Performance Evaluation

In order to determine if the performance of the Constellation is good enough and to compare different constellations, we define the following parameters that are to be used in the weighted ordered average decision5.3.1.

Simulation parameters important to clarify:

- Simulation time: 25h. This time is enough to observe the motion of the whole constellation on Earth considering its rotation and the rotation of the plains due to the Earth's oblateness.
- Minimum contact time: 3 minutes. Time enough to download data, tracking and Telecommanding the satellite.
- Time precision: 10 seconds. It is empirically observed to be precise enough.

The computed parameters:

- Fraction of time with flybys on the GS: Ratio between the time in which there is any satellite in the field of view of the Ground Station and the total simulation time. (Referred in table 5.3.1 as % Coverage)
- Mean number of links with the satellite

- Fraction of time with flybys longer than 3 minutes: In this case the ratio is with the time in which there is a satellite doing a useful pass, since a full contact can be done. (Referred in table 5.3.1 as %Quality Time)
- Mean pass time: This parameter is used to guarantee a minimum of quality and to compare different configurations. (Referred in table 5.3.1 as Average Pass Time)
- Number of gaps: Gaps are in this chapter defined as periods of time without a pass that is lasting/will last more than 3 minutes. (Referred in table 5.3.1 as Num Gaps)
- Maximum gap time: At high latitudes all the Walker-Delta configurations show a characteristic gap that can last even for hours, which is not admissible. This parameter will tell us if we exceed a maximum defined as 3 minutes for this study. (Referred in table 5.3.1 as Max Gap Time)
- Mean gap time: As it is obvious, a minimum or a 0 is desired.

You can find below an example of the analysis, for a constellation in a Semi Walker-Delta configuration.

Constellation	Full WD
Number of Planes	$p = 8$
Satellites per plane	$spp = 18$
Inclination	$i = 75^{o}$
GS Latitude	$\lambda = 80^{o}$
GS Longitude	$\phi = 0^{o}$

Table 5.2.1: Constellation parameters for the Example Constellation

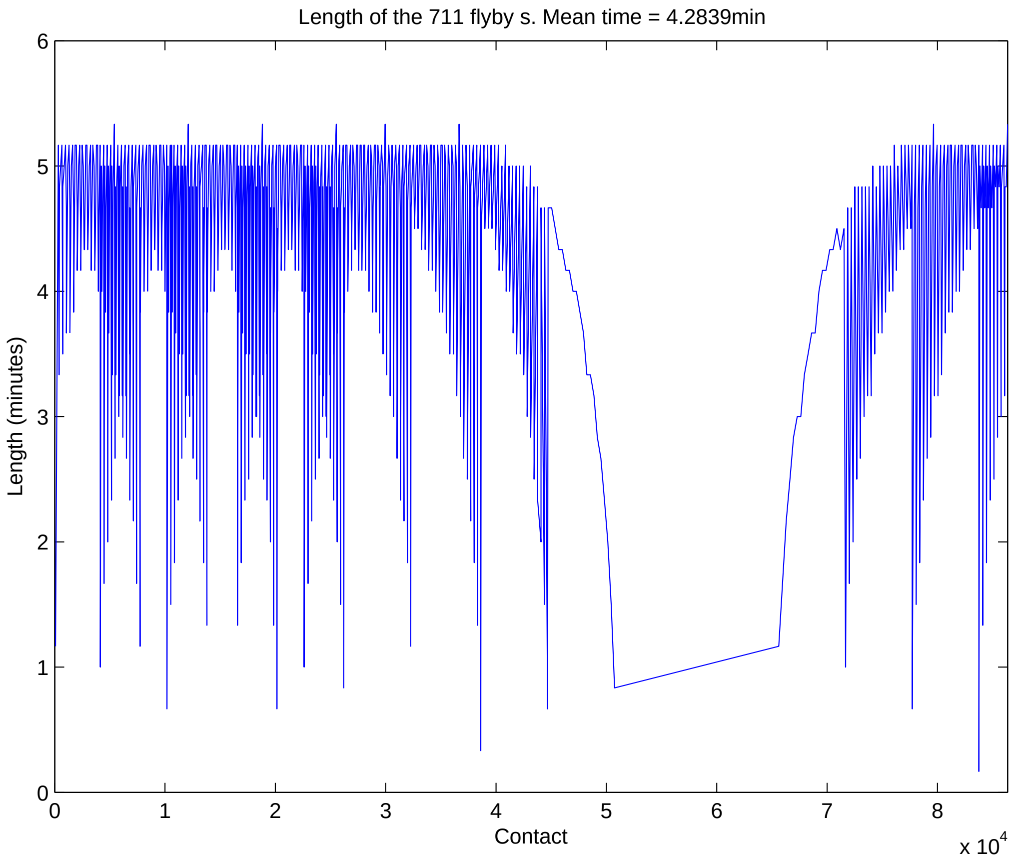


Figure 5.2.1: Length of the passes on the example GS.

Pass Time Ratio	77.53%
Quality Time Ratio	75.77%
Mean Pass Time	$4.28min$
Number of gaps	37
Maximum Gap Time	$314.33min$

Table 5.2.2: Performance Parameters for the Example Constellation

Given the high latitude of the Ground Station plus the Semi Walker-Delta Configuration there is an enormous gap. In addition, between planes some gaps are also observed.

5.3 Ordered Weighting Average based Decision

The Described Constellations are weighted and averaged in the table below. The detailed explanation of the parameters can be found in 5.2.1:

		Candidates						
Criteria	W	1	2	3	4	5	6	7
Price	15	1	2.35	5	4.94	3.21	3.92	4.67
% Coverage	4	5	4.77	2.94	2.14	4.43	1	3.86
Max Gap Time	3	3.12	3.62	1	2.88	3.51	5	4.75
%Quality time	5	4.91	4.49	4.05	1	3.19	5	4.98
Average Pass Time	5	1.21	1.14	1.14	1	1.90	5	4.72
Num Gaps	2	4.73	4.44	4.23	1	3.03	4.99	5
% Sats above	6	1	1	5	5	1	5	5
SUM (p*g)	40	90.42	108.17	154.19	133.29	113.94	167.71	188.21
OWA		0.452	0.541	0.771	0.666	0.570	0.838	0.941

Table 5.3.1: Constellation Configuration OWA Decision

With this comparison table, the optimum Constellation is option number 7:

The Astrea Constellation

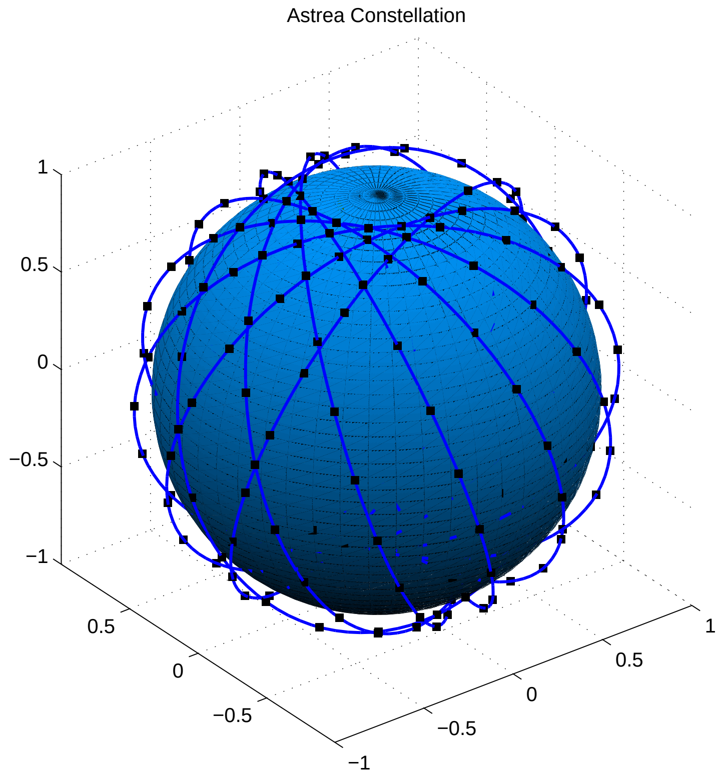


Figure 5.3.1: Astrea Constellation Final Configuration.

Part II

Constellation Deployment

Chapter 6

Constellation Deployment

6.1 Constellation Deployment Department

This introductory section aims to present the Constellation Deployment Department and its duty.
The mentioned department is composed of four members of the Astrea Constellation. It is created in order to asses all the issues related to the launcher which will put the constellation into orbit and the different strategies to be followed to ensure the adequate set up and posterior maintenance of the constellation. In the Project Charter document, four tasks are assigned to the department:

- A comparison among the existing launching platforms to find one that fulfills the requirements of the constellation at a reasonable economic conditions.
- The book of a launching date if the selected launcher requires it.
- The follow of the current legislation referred to launchers and deployers.
- The design of an End of Life and a Replacement strategy.

In order to simplify the accomplishment of this assignments, the department decides to divide its tasks into six parts each of one is a section of this report:

- Launching System
- Deployer
- First Placement
- Replacement Strategy
- Spare Strategy
- End of Life Strategy

6.2 Launching System

The aim of this section is the selection of a launching platform. First of all, a review of the available ones on the market is carried out, secondly a small group of launchers is chosen and finally, an optimization is developed in order to find the most suitable system.

6.2.1 Launch site and vehicle analysis

Now a days there is such a great amount of launchers available over the world. Nevertheless, most of them are designed for very specific missions. In addition, the space career of a country is usually highly attached to the government, for both economic and political reasons. When searching for a launching system, some parameters have to be taken into an account like payload mass, possible inclination angles, launching site, etc. This analysis only considers those rockets which parameters seem adequate for the Astrea constellation launching.

A general research is done in order to filter all the launchers that can be discarded without any study. The result of this research is that there are seven potential rockets in the market capable of deploying the constellation as well as carrying out the replacement needs. The launchers can be divided in two categories: the powerful ones and the small ones. The first ones are capable of carrying heavy payloads, however they present high operation costs whereas the second ones are way more economic due to the reduced size. In addition, the small rockets are more focused on commercial flights without having to attend governmental issues.

The following table displays the first seven candidates.

ENTERPRISE	ROCKET	LAUNCHING SITE	TYPE
Rocket Labs	Electron	North Island (New Zealand)	Light
Kosmostras	Dpner	Baikonur Cosmodrome (Kazakhstan)	Light
Arianespace	Ariane V	Guiana Space Center (French Guiana)	Heavy
Arianespace	Vega	Guiana Space Center (French Guiana)	Light
SapceX	Falcon 9	USA	Heavy
PLDSpace	ARION-2	Huelva and Cape Canaveral	Light
LEO Launch and Logistics	-	USA	Light

Table 6.2.1: List of Launchers

6.2.2 Last candidates and selection

Once this first selection is done, more accurate information is needed so as to reach a reliable conclusion. However, none of these enterprises shows its information on the Internet or any similar divulgation channel with the exception of Arianespace. Thus, all of them must be contacted to get the needed data. The same email is sent to all seven enterprises and several days later, three of them show interest in the Astrea constellation: Rocket Labs, PLDSpace and LEO Launch & Logistics. Since the other enterprises do not answer the requests and, as a consequence, will not provide the necessary information, they can be directly discarded. Hence, the candidates list is reduced to those three who responded the enquire plus Vega, given that its information is available online. Although the needed data of Ariane V is also known, it is discarded by the fact that it presents high operation costs and it is capable of carrying about 5,000 cubesats 3U when the Astrea constellation will have 189 sats. Therefore, the four remaining candidates are studied in more detail and are subjected to an optimization.

In order to find the most suitable option achieving the project objectives, it is thought to do an evaluation process following the Ordered Weighted Average (OWA) method . First of all, the required parameters for the decision have to be determined. According to the orbit design, the range of inclinations, the number of orbital planes and the range of heights must be taken into an account. Nevertheless, more parameters are needed in order to ensure a reliable result: cost per satellite, frequency of launchings per year and number of satellites deployed per launch. Both range of inclinations and number of satellites per launch act as a restriction due to the following two reasons. First, since orbital plane changes are very expensive and are out of consideration, the minimum number of launchings must equal the number of orbital planes. In addition, being capable of deploying the constellation with the minimum number of launchings is an adequate solution. This turns the number of CubeSats per launch into a restriction: the chosen launcher must be capable of launching at least the number of satellites in an orbital plane. Secondly, the inclination is considered a restriction by the fact that if a rocket is not capable of deploying a satellite in the desired inclination, it makes no sense to use it.

Since the number of orbital planes is 9 and the inclination is 72°, any launcher which doesn't fulfills one of this restrictions can be automatically rejected.

Moreover, the following table contains all the information mentioned above which is helpful to compare the different launchers and see if they accomplish the basic features.

Parameters	**Rocket Lab**	**PLD**	**LEO L&L**	**Vega**
Satellites/Launch	24	34	150	325
Inclination(°)	39.2 to 99	116 or 140	any	any
Cost/Satellite (US dollars)	240,000	-	266,667	100,000
Orbital planes	1	1	1	1
Frequency/year	9	8	8	2
Range of heights (km)	LEO	LEO	LEO	LEO

Table 6.2.2: Criteria

It is important to point out that all the rockets available in the market can achieve the necessary amount of satellites per launch. Although all of them reach the height the CubeSats need, PLD does not attempt the inclination needed which is 72°. As a result, this launcher is not appropriate for the project purpose and it is rejected. According to the remaining 3 candidates, all of them are adequate candidates, nevertheless there is a characteristic that may interfere with the mission goals. At first instance, the frequency per year has not been considered a critical parameter. Those have been chosen regarding orbital parameters only, however, although the frequency does not influence de capability of the rocket of deploying a CubeSat in the desired orbit, it can compromise the set up of the constellation and the posterior replacements. The lower the frequency is, the slower the deployment will be. Therefore, the frequency of the three remaining candidates must be analyzed. As seen in the table, Vega presents the lowest frequency (two launchings per year). This value is not acceptable due to the intention of deploying one single orbital plane per launch. The placement of the whole constellation would last four years, this mean that de first planes would be near their replacement time while the last ones would only have been nearly a year in orbit. Thus, Vega can also be discarded.
This leaves the selection with only two options: Rocket Lab and LEO Launch&Logistics. An Ordered Weighted Average can be made between those two candidates taking the cost/satellite, the number of orbital planes, the frequency and the range of heights into account. Yet, they both present the same number of planes and range of heights, consequently the OWA can be done regarding only the two cost and frequency. The first

has to be minimized and the second maximized. Since Rocket Lab presents best values in one parameter and the other (240,000 US dollars vs 266,667 and 9 launchings/year vs 8) there is no need to develop an OWA. In addition, an e-mail from Rocket Lab is received stating that a launch per week is achievable. Thus, the chosen rocket is Electron, from Rocket Lab enterprise. This rocket fulfills all the requirements of the constellation.

Figure 6.2.1: Electron Rocket

6.2.3 Launcher overview

Following, a brief description of Electron is provided.

Figure 6.2.2: Second Stage

Shown in 6.2.1, Electron is a two stage light rocket constructed from carbon fiber composite. It is powered by ten Rutherford engines, all of them use liquid oxygen (LOX) and rocket kerosene. The first stage has nine out of the ten engines which generate 152 kN of thrust. The second one, displayed in 6.2.2, has the remaining engine which produces 22 kN. The second stage contains the fairing where the payload is placed. Electron is 17 m long and its diameter is 1.2 m. It is capable of launching 24 3U CubeSats every week at a LEO orbit with a range of inclinations from 39.2 to 99 degrees.

Figure 6.2.3: Electron Rocket Fairing

The injection maneuver is carried out following the flight profile shown in the table 2.3 . The accuracy of the injection is mission dependent, however a typical value would be ±15 Km. According to the CubeSat/Fairing interface, Electron is compatible with the standard CubeSat deployers like ISIS or P-POD, in addition, if those deployers are used, Rocked Lab is able to situate the satellites inside the rocket in a more efficient disposition.

Event	Time (s)	Altitude (km)
Lift-off	0	0
Max Q	79	11
MECO/S1 Separation	152	69
Stage 2 Igintion	159	69
Farinig Separation	183	110
SECO	457	284
Satege 2 Apogee Kick	3157	499
Payload Separation	3200	500

Table 6.2.3: Flight Profile

Rocket Lab facilities are located in New Zeland. The test laboratories are placed near the airport of Auckland and the launch site is in Mahia (6.2.4).
Finally, the cost per satellite is 240.000 US dollars or if the rocket is totally filled, 5.760.000 US dollars the entire launch.

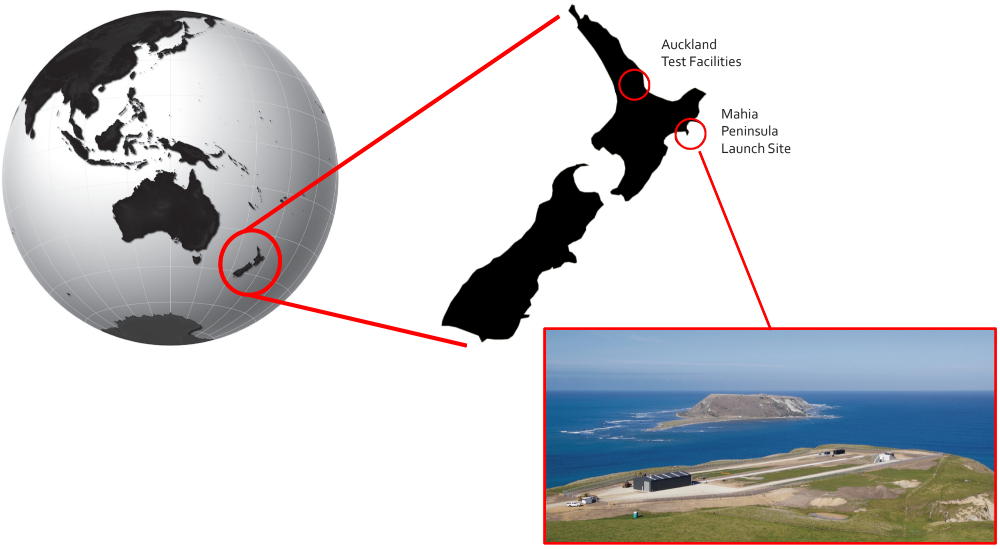


Figure 6.2.4: Rocket Lab Facilities

6.3 Deployer

The objective of this section is to give a brief explanation of what is a deployer and how it works. Additionally, some examples of available ones in the market are shown and so is the selected one.
As introduced above, there must be an adaptor between the rocket and the satellite in order to ensure subjection during the flight, efficient organization of the space in the fairing and a correct separation during the injection maneuver. This duty falls on the deployer. It consists on a prismatic structure prepared to carry the CubeSat inside. When the desired orbit is reached, the deployer uncovers one of its faces so as to let the satellite leave. There is a spring in the bottom that provides a little push to ensure that the CubeSat separates from the rocket. There are many types of deployers, some of them are designed for an specific type of mission. As stated before, Electron is compatible with the standard CubeSat deployers, hence, only this type is considered. Similar to the case of the launcher selection, almost all the enterprises don't show enough information on the internet to reach a reliable conclusion, thus, some of them are contacted. Only two answers are obtained, one from ISIS (ISIPOD Deployer) and GAUSS (GPOD deployer). POD stands for Pico-satellite Orbital Deployer.
They both present similar characteristics, however there are some differences. First, the main features that both offer are outlined, secondly, the small differences between them are pointed out.

- Main features
  - Provide deployment status signal.
  - No battery needed nor external power source
  - No pyrotechnics
  - Protect the CubeSat from external environment
  - Mechanically interfaces with the CubeSats by means of guidelines
  - Mechanically interfaces with the launch vehicle by means of standard fasteners
  - Qualified for multiple of launch vehicles
- ISIPOD
  - The satellites are fully enclosed inside the deployer, once the CubeSat is fit in, there is no access to it (see image 6.3.1)
  - Electrically interfaces with launch vehicle for telemetry
- GPOD

- Accessible panels: all the side panels allow the access to the integrated CubSat (see image 6.3.2). This means that the entire area between the guide rails over the entire CubeSat length may be freely accessed.
- The price for a single deployer 3U is 16000 euros.

In order to reach a reliable conclusion, two issues must be taken into consideration. First, the CubeSats of the Astrea Constellation are equipped with thrusters which increase the length of the satellite, thus, the deployer chosen cannot be fully closed. This condition automatically rejects the ISIPOD, nevertheless, there is a second reason for choosing the GPOD, the enterprise ISIS does not show the prices of their deployers even when a request is sent. Without this information it is decided that it cannot be taken into account.

Figure 6.3.1: ISIPOD

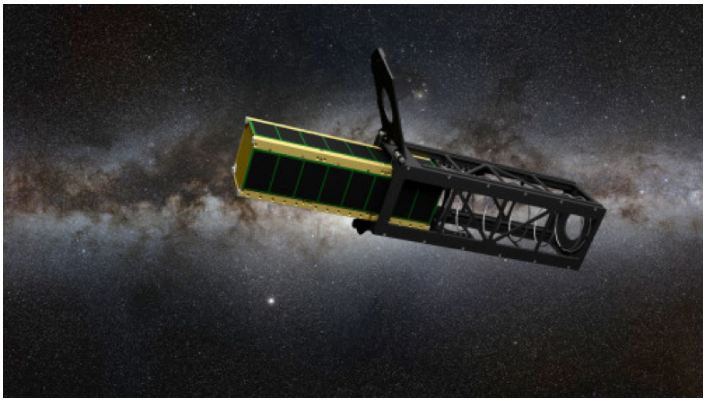


Figure 6.3.2: GPOD

6.4 First Placement

The aim of this part is to explain the first placement of the constellation. It is divided in two parts, the first one is intended to give a first approach to the logistics involved in the first placement. The second one is focused on the maneuver required so as to deploy the satellites into orbit.

6.4.1 First Placement logistics

The objective of this section is to give a general idea of the first placement logistics. Although some temporal data is provided, it is a qualitative explanation, only to clarify the order in which the different elements must be purchased, assembled, transported, etc. Rocket Lab provides two gantt diagrams on which their launching procedure is explained (images 6.4.1 and 6.4.2)

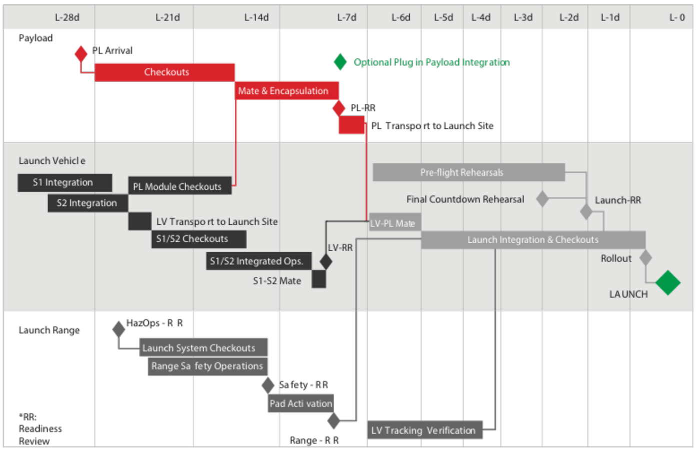


Figure 6.4.1: Launch Range Operations Flow/Schedule

The constellation has 189 3U CubeSats distributed in 9 orbital planes. One of the conclusions stated in the Launching System section is that the quickest way to deploy the whole constellation is by carrying out one launching per orbital plane, consequently, the first placement consists on 9 launchings and all the logistics around them. Rocket Lab is capable of launching once a week, therefore, the first placement takes 9 weeks. Due to the magnitude of the mission, the whole rocket is filled with Astrea satellites, hence, there is no need to share it with other missions. Also, Rocket Lab offers an online booking procedure to reserve a date, however, The Payload User's Guide (provided by Rocket Lab) recommends contacting directly with them in case of filling several rockets with a mission instead of booking online.
Since the schedule of Rocket Lab is fixed, the logistics needed in order to deliver the payload on time are going to be explained starting from the launching day, going back

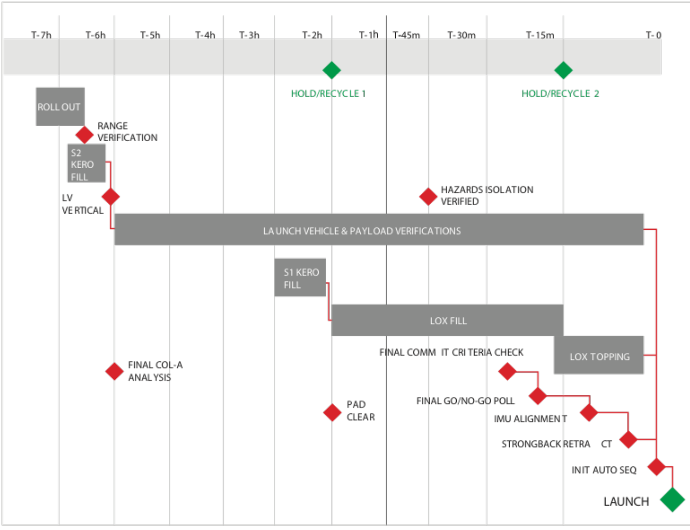


Figure 6.4.2: Countdown Operations Flow

in time until the first movements in Terrassa, where the satellites are assembled. The launching day is designed L henceforth, and all the other ones are referred to this one (eg. L-30d means 30 days before launching).

As seen in figure 6.4.1, Rocket Lab needs 28 days to prepare the payload, place it into the rocket and prepare the rocket itself. Thus, the CubeSats have to arrive at the Rocket Lab launching facilities the L-28d. The satellites are assembled in Terrassa, hence, they have to be brought to New Zealand. Due to the large amount of CubeSats, the chosen transport is sea transportation. The estimated time from Terrassa to New Zealand is 30 days, so the CubeSats have to leave Terrassa the L-58d. At this point, there is two options. First, the 189 satellites can be divided in groups of 21 (number of sats in an orbital planes) and sent separately to New Zealand so that every group arrives 28 days before its departure. The other option is to send all 189 CubeSats at the same time so that they arrive 28 days before the first launching. Each option has its pros and its drawbacks. Option one does not need to store the satellites in Rocket Lab facilites, conversely, the logistics of carrying each group of satellites separately is complicated. Option two allows to assemble all the satellites and send them in one ship, however, once they arrive to their destination, they have to be stored somewhere until their departure day arrives. Option two is selected because it is simpler and it is more likely to not cause delays delivering the payload to Rocket Lab, in addition, it is concluded that sending 9 ships with one week separation is not as efficient as sending a single one.
The estimated time of assembling the satellites is twelve months, consequently, they have to be ordered the L-423d.
As clarified above, it is important to remember that the stated times are an approximation and the goal of this section is to give a first idea of the order of the different actions.

6.4.2 1st Placement Maneuver

Once the Constellation is designed, it is essential to plan a proper procedure to put it in orbit. The Constellation is configured in several planes and satellites in each plane which work and communicate together in order to give signal coverage around the globe to finally accomplish their final purpose: intercommunicate other satellites form our customers.

One of the purposes of the project is to ensure the system is able to provide partial service right from the very beginning of its life, that is since the first orbital plane is put into orbit. Therefore, along with the maneuvers required to separate satellites in a certain orbital plane, the order in which the planes are put into orbit will also be assessed in this section. This particular section is crucial as it describes how the constellation is born.

6.4.3 In-Orbit Injection

It wouldn't be fair to start without mentioning the spaceship that will bring the whole system to life, and this is no more and no less than the Electron, from Rocketlab USA in New Zealand. The Electron is able to carry 24 3U CubeSats at once. Since 21 is the number of satellites needed in 1 orbital plane, it will be able to put one orbital plane into orbit in just one launch using the procedure described in the upcoming paragraphs.

Before starting any procedure description, it is important to set a start point. The first consideration is that there are still no Astrea satellites orbiting the earth. Therefore it is the first orbital plane that will be put into orbit. It is also considered that the rocket loaded with the 21 satellites has already accomplished all necessary maneuvers after lift-off and has just been able to arrive at the satellite's orbit, that is, proper altitude above Earth and proper tangential velocity. Of course at this point only the 2nd stage of the initial Electron rocket remains. Moreover, this stage is the one responsible of carrying the payload along with every single deployer. Once the start point is set, it is possible to thoroughly describe the procedure.

At the very described moment the first CubeSat is deployed into its final orbit around the Earth, which is a circular orbit at $542\,km$ above Earth's surface. In order to deploy the second satellite at a given phase separation from the first one, the rocket must enter into an elliptical orbit with a slower period. Adopting this procedure will allow the needed phase separation between satellites given the fact that after one revolution of the rocket around the Earth, the first satellite will have gone through one revolution and a fraction more. In other words, at the very moment the rocket passes through the initial point which is tangential to the satellite's orbit, the first deployed satellite will be phase-wise ahead of the rocket. Obviously, the elliptical orbit mentioned must be accurately computed in terms of the increments in speed required to enter into it.

In a more schematic way, the procedure goes as follows:

1. The rocket goes through the procedure designed by Rocketlab USA to get to the destination orbit. The approximate trajectory during this stage is represented in 6.4.3.Right after entering into the destination orbit, the first satellite is deployed into it as seen in 6.4.3 represented with a red dot.

2. Once the latter is completed, the rocket's engine gives it the necessary ΔV in order to get to the elliptical spacing orbit. In 6.4.4 half a revolution of the rocket is represented along with the orbit of the first deployed satellite at the same point in time.

3. After one full revolution of the rocket in the elliptical orbit, the first satellite will have left the right phase spacing with respect to the rocket. At this point the rocket's engine gives the same ΔV as in 2 but negative. This will cause it to enter again into the circular orbit of the satellites. At this point the rocket deploys the second satellite as shown in 6.4.5. Right after this deployment the rocket enters into the elliptical orbit again.

4. 6.4.6 represents again half a revolution of the rocket in the elliptical orbit along with the deployed satellites so far.

5. Finally, the rocket reduces its velocity again to enter into the circular destination orbit in order to deploy the third satellite (6.4.7).

6. The procedure is iterated until the orbital plane is full.

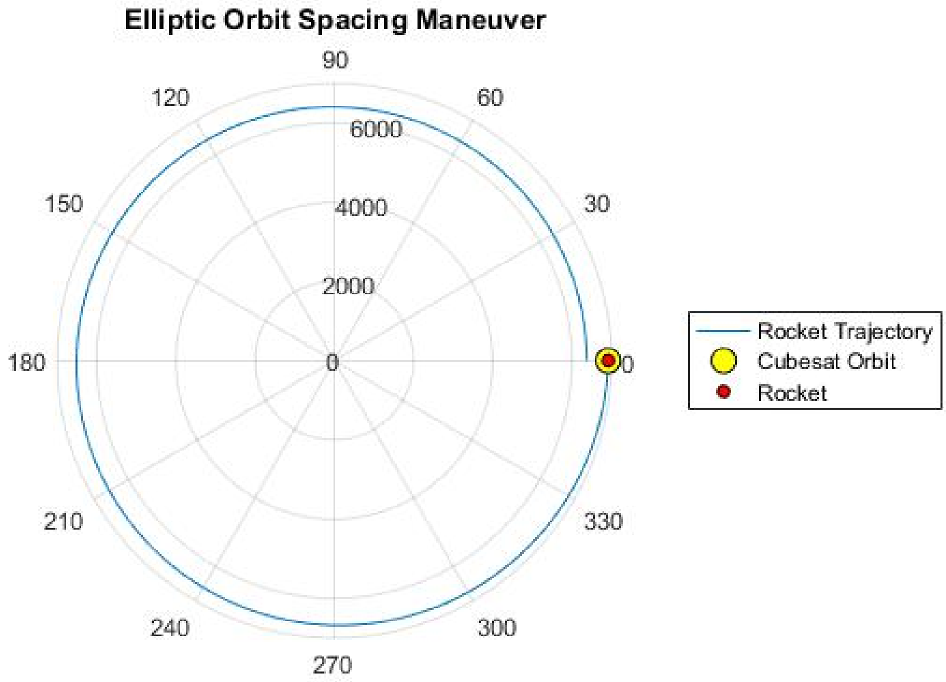


Figure 6.4.3: Rocket's trajectory from lift-off to final orbit.

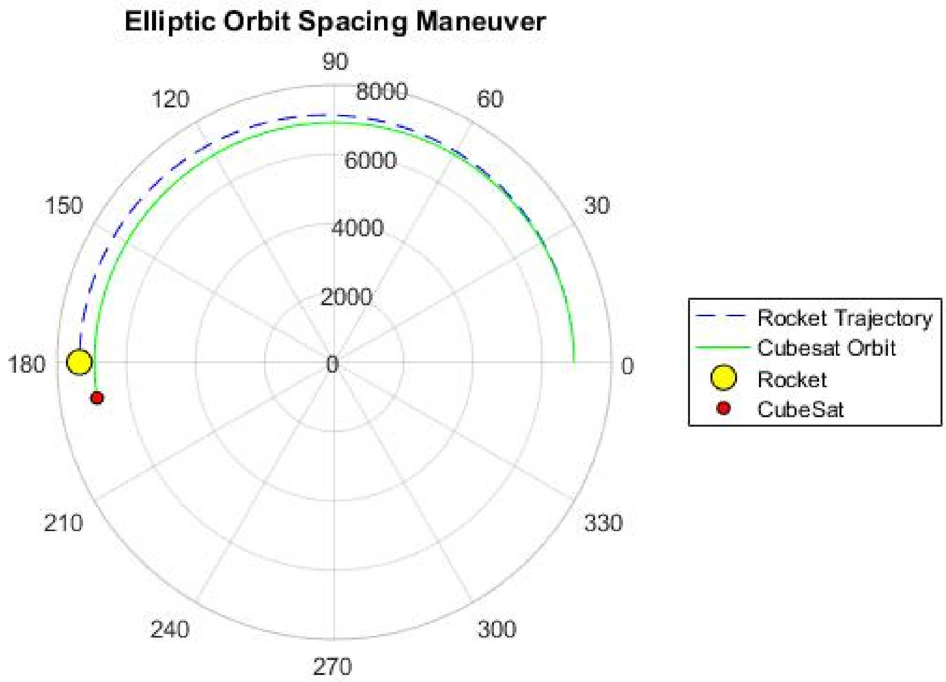


Figure 6.4.4: Half of a revolution of the rocket in the elliptical spacing orbit.

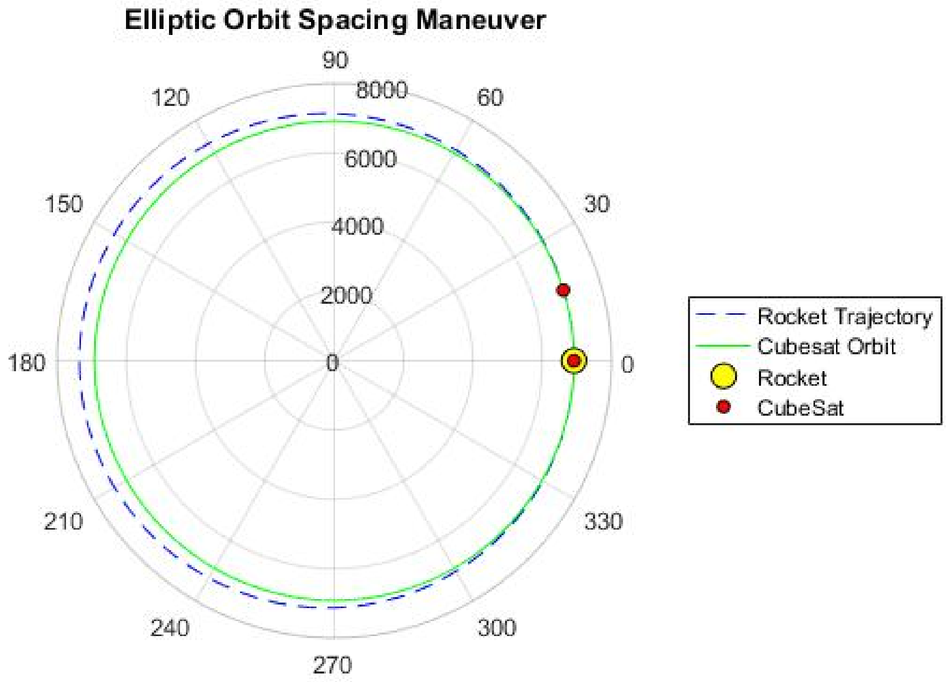


Figure 6.4.5: Deployment of the second satellite.

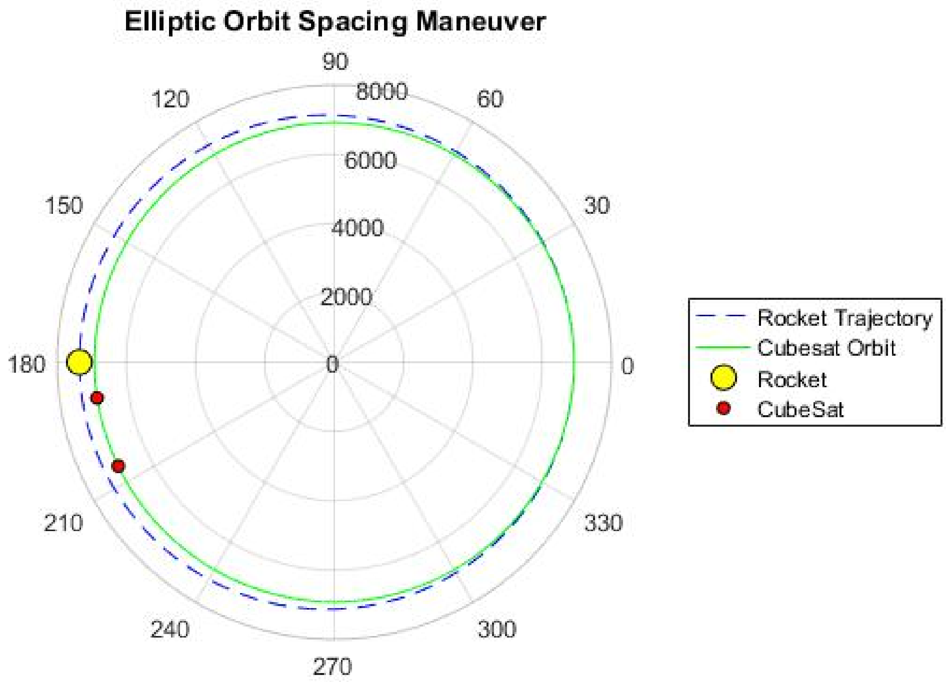


Figure 6.4.6: Half of a revolution of the rocket after the deployment of the second satellite.

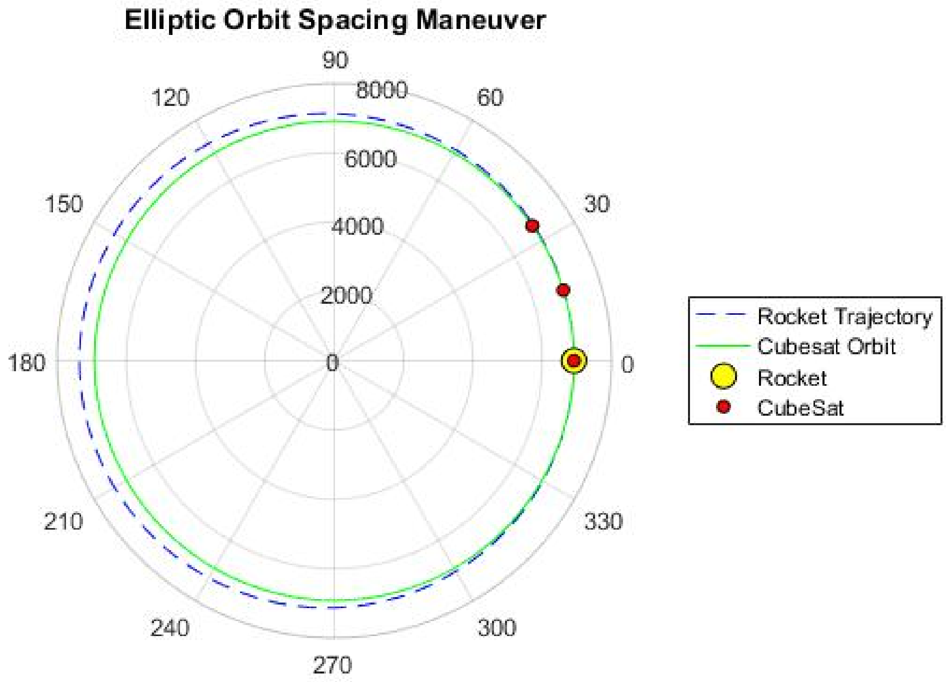


Figure 6.4.7: Deployment of the third satellite.

Having pointed all of the above, it would make no sense to proceed without thoroughly going through the calculations of every single one of the required parameters to perform the manoeuvre. The first thing to take into account is the number of satellites for orbital plane. A number of 21 satellites per plane has been established, thus, a separation of $360°/21 = 17.14°$ between satellites will have to be accomplished. The velocity of the satellites and the period of their orbit is now computed:

$$V_s = \sqrt{\frac{GM_t}{R_t + h}}$$

$$T_s = \frac{2\pi * (R_t + h)}{V_s}$$

Where R_t and h are Earth's radius and height above Earth's surface respectively. For $h = 542\,km$, the values obtained are $V_s = 7,589.6\,m/s$ and $T_s = 5,723.1\,s$. Let's call the spacing between satellites $\theta = \frac{360°}{21} = 17.14°$ and $R = R_t + h$. Using these values it is possible to compute the period of the elliptical orbit, T_r, along with the rest of the parameters:

$$T_r = T_s + \frac{\theta R}{V_s} = 5,995.6\, s$$

$$a = (\frac{T_r}{2\pi}^2 GM_t)^{\frac{1}{3}} = 7,130.8\, km$$

$$R_1 = R; \quad R_2 = 2a - R_1$$

$$c = a - R_1; \quad b = \sqrt{a^2 - c^2}$$

$$\epsilon = \sqrt{1 - \frac{b^2}{a^2}} = 0.0305$$

$$\Delta V = \sqrt{\frac{GM_t}{R1}} (\sqrt{\frac{2R_2}{R1 + R2}} - 1) = 115.01\, m/s$$

Astrea's main purpose when it comes to 1st placement is to provide service as quickly as possible. This means that the time it takes to put a plane into orbit is crucial. This time will be determined by the period of the elliptical separation orbit that the rocket uses between deployments and of course by the number of satellites in each plane. Since 21 are the satellites that need to be put in orbit, 21 elliptical orbits will be needed. Therefore the time needed for one orbital plane is $3200\, s + 21 * T_r = 129,191.6\, s$ which means 35.9 hours.

6.4.3.1 Plane Order

Having described the procedure used to put one orbital plane in orbit, it is now time to describe the order in which all of the 9 planes are put into orbit. The fact that establishes one path or another is the fact that satellites can only communicate with neighbours, that is, one satellite can only communicate with its neighbours from the same plane and the neighbours from the neighbour planes.

When it comes to the order in which the planes are put into orbit, there are two main ways that come to mind. The first one is putting the planes consecutively into orbit. The second one is to put the planes into orbit leaving space between them for future planes. For example plane number one is put into orbit. The second plane to be put into orbit leaves space for one plane in between them. Then the third leaves space for one plane from the second, and so on. Leaving more space than for one plane could also be an option.

On the one hand, when using the first way the satellites from each plane could communicate with the ones from their neighbourhood. Therefore the range of communication would start being narrower but as new planes are put into orbit, the range would become wider. For instance, when three planes are already working, a given satellite form a customer could communicate with satellites that are at the other side of the planet in a determined range given by the width of signal that those three orbital planes could cover. When new planes are put into orbit this width becomes bigger up until the full globe is covered. Of course the main drawback of using this consecutive way of putting planes into orbit would be the long time of inactivity right at the beginning when few planes are working.

On the other hand, when using the second described way, the satellites can't communicate with other satellites from neighbour planes but the time of inactivity for customer's satellites would be less as a gap between planes is left for future ones. Nevertheless, this kind of configuration has a huge drawback and it's that when a satellite communicates with one given plane, this one can only communicate with other satellites that are in the range of signal emission of that given plane. This is due to the fact that as neighbour planes are further apart they can't communicate with each other and therefore the range of communication is affected.

. Having pointed out all of the advantages and drawbacks of each configuration it is time to choose and it all comes down to Astrea's preferences. The configuration that fulfils these preferences for the most part is the consecutive .It allows the satellites to communicate in a broader range as the constellation grows and progressively conquer the sky.

6.5 Replacement Strategy

Due to the lifespan of the CubeSats, the whole constellation is replaced every five years, hence, a replacement strategy has to be designed. As stated in the First Placement section, the orbital planes are deployed consecutively, thus, the replacement has to be so also. One simple solution could be waiting for a plane to de-orbit and then place a new one into the same position, however, this procedure would spend to much time by the fact that the satellites approach the atmosphere in a very slow rate. Additionally, the replacement of different planes would probably overlap. Since the first placement has been carefully designed, it is thought to adapt the same procedure to the replacement process, that means, to consider the replacements as a first placement. Obviously, some differences have to be taken into account given that at this point there is a constellation providing full service to the customers. The problem remains on the fact that in order to use the same strategy, the replacement needs to be achieved in eight weeks, therefore, the new orbital planes cannot be situated into the same position than the old ones. A rapid replacement is also interesting regarding the need of providing full service to the customers without interruption. The solution adopted consists on placing the new planes between the old ones consecutively, following the order of the first placement. In order to clarify the process, a detailed explanation is shown below:
First of all, since different orbital planes are going to be taken into account in this explanation a nomenclature is set: old planes are the ones that have to be replaced, the new ones are the planes that will substitute them. If a plane is named with the number 1, it means that is the first one to be placed (old or new) and so on (2,3,..,21).

- The new plane 1 is placed between the old plane 1 and the old plane 21.
- The new plane 2 is placed between the old plane 1 and the old plane 2 to ensure that at the very moment the first old plane begins to decay, it does not appear a gap.
- At this point, the following new planes are deployed consecutively between the old ones until the constellation is fully renovated. This maneuver is repeated every five years to ensure the continuity of the Astrea Constellation. The following images show the process explained above.

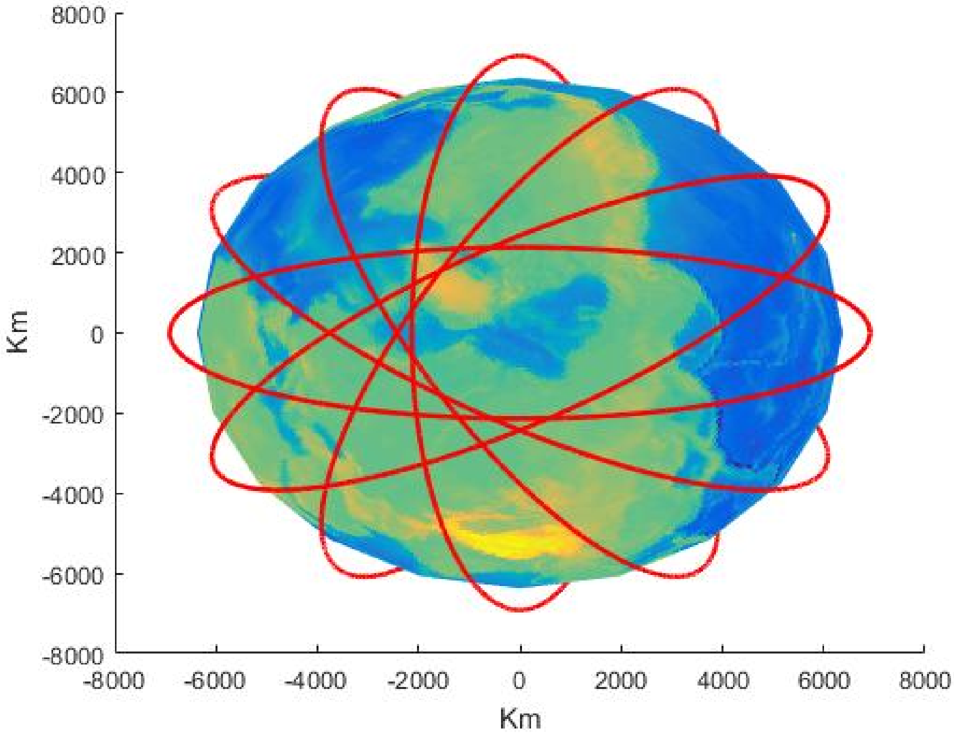


Figure 6.5.1: Old Constellation

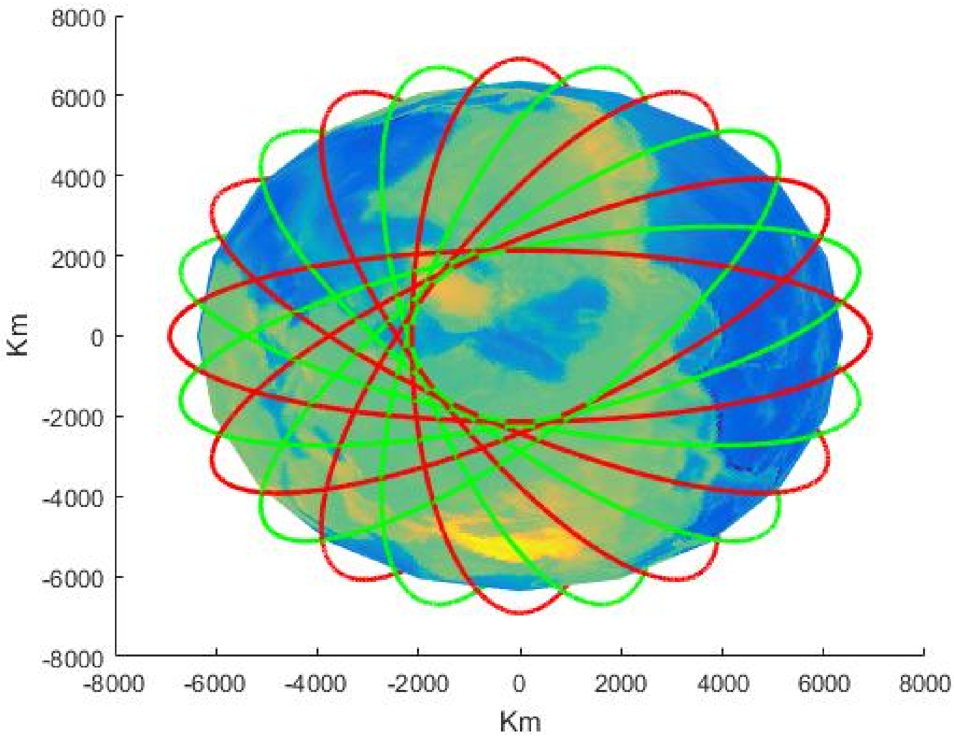


Figure 6.5.2: Old and New Constellations

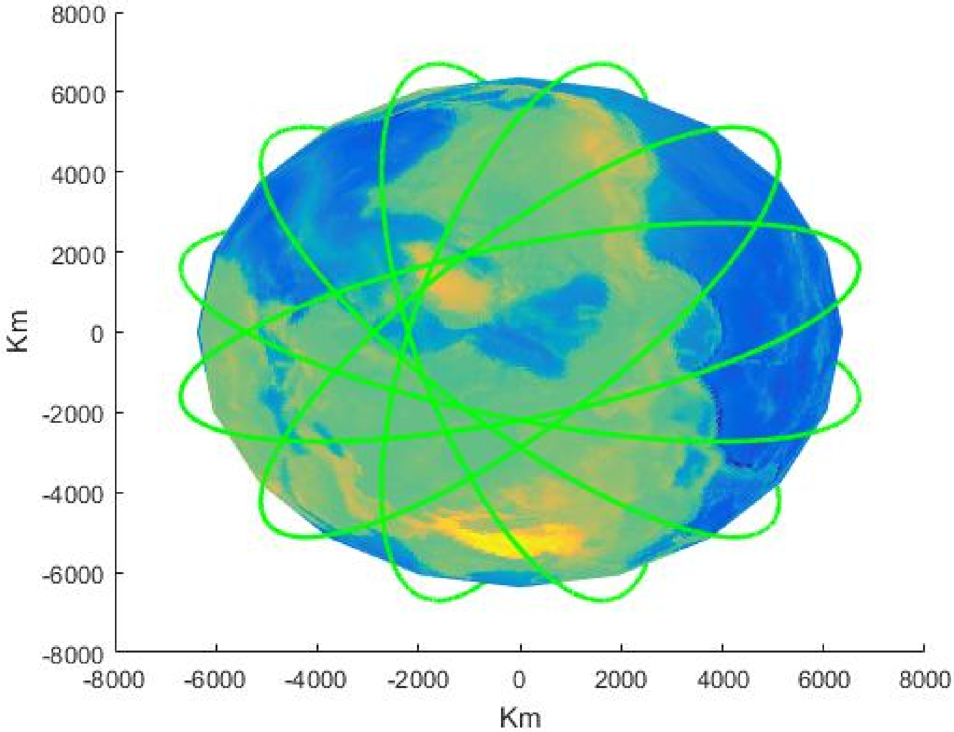


Figure 6.5.3: New Constellation

6.6 Spare Strategy

6.6.1 Introduction

When building a satellite constellation with the target to provide global coverage communication relay between LEO satellites and between LEO satellites and the ground, it is crucial to avoid any deterioration of the service. In order to ensure that any possible fail from the satellites would not spoil the constellation operation for more than 6 hours; a spare strategy has to be done. Nowadays, four different types of spare strategies are known:

- Spare satellites in constellation
- In-orbit spare
- Spare satellites in parking orbits
- Spare satellites on the ground

Each existing spare strategy is valid. Despite, depending on the enterprise prioprities the most suitable has to be choosen. In addition, the decision taken is related to the constellation flexibility to degrade the service to a lower performance level during a certain period and to its cost.

6.6.2 Spare Strategy Alternatives

Spare satellites in constellation:
This configuration consists on designing the constellation to be *"overpopulated"*. As it sounds, this means that the system is established with *extra* operative satellites already orbiting within the constellation. For instance, only two overpopulating configurations had been pictured: ovepopulated by one satellite or overpopulated by two satellites per orbital plane.

- One extra Satellite:
  By adding an extra satellite to the primary design of the orbital plane configuration, one satellite failure is covered with little time delay to recover the plan. In this way, the constellation continues to work at maximum capacity after a short interruption and at a suitable cost.

- Two extra Satellites:
  Usually, by adding two extra satellites per orbital plane the reliability of the service achieves values around the 99.99%. This configuration increases considerably the cost of the project and it is mainly neccessary in cases where the availability of the satellite is essential for the proper operation of the constellation.

Therefore, when designing an overpopulated constallation, the first decition to be made is the number of extra satellite per orbital plane. To guarantee the most optimal configutation a feasibility study is needed.

In-orbit spare:
The main diference between this strategy and the previous one is that in this case spare satellites are not operative. So the idea is to put some spare satellites in a orbit close to the principal one of the constellation in order to avoid possible collisions between operative satellites and spares.

A few things have to be taken into account when using this method. Firstly,even though the spare satellites are not operative, by being in orbit they deteriorate and by the time they are needed their operative lifetime and performability will not be such as the ones of brand new satellites.Secondly, as their are non-controlled satellites their orbital decay has to be predicted to be aware of possible collitions and avoid them. Thirdly, once any spare satellites is needed, it has to be able to do a two Hohmann transfer to achieve the performance orbit; the first one to reach a phasing orbit and the second one to end in the operational altitude.

Spare satellites in parking orbits:
By mading this choice it has to be assumed that the spare satellites can be keeped in parking orbit until they are needed. Two different option are valid: keeping the rocket in a *"parking"* orbit and then try to send it to the corresponding orbit; or keeping it in in-orbit satellites parkings such as the ISS. The main drawback is that the performance takes a long time until the constellation is recovered and depending on the orbit parameters and the launcher it is not possible to use this strategy.

Spare satellites in parking orbits:
The simpliest and easiest one; the only thing that has to be done is to build extra satellites. The spares will remain on ground when the constellation is launched. Only in case the structure collapses due to a satellites failure, an emergency launch will put the spares in orbit. Moreover, this method is expensive because every extra launch has a high cost and it can take weeks to recover the constellation performance.

6.6.3 Spare Strategy Selection

From all those alternatives, two of them are quickly discarded: in-orbit spares and in parking orbit spares. The first one is having a non-working satellite in orbit because not only the satellite has to be purchased, but also it has to be launched to a different orbit than the principal one. That fact will increase the cost of the launch or even worst it could create the necessity of an extra launch. Although, the satellites needs to reach the operative orbit and it is known that cubesats propulsion is not really powerful. Furthermore, this satellites might never be needed. So it is highly probable this investment to be a waist of money and sources and this are the main reasons why it has benn discarded.

The second is not available in the *Astrea Constellation* case. On the one hand, the main parking in orbit will be the ISS which is at an altitude of 400km above the earth and the constellation is situated at among 550km above the earth. Knowing that, this option is immediately discarded. On the other hand, the Electron the rocket that will accomplish the mission to put the satellites in orbit cannot stay in parking orbit before arriving to its final destination. Definitely, the service cannot rely on this option.

Two possible spare strategies remain: pare satellites in the constellation or on ground. In spite deciding if both ones are useful or only one of them is, a feasibility study is done. The objective is analise the diferent kind of failure that have to be covered and determine how the constellation will collapse. Only after that the most suitable strategy method can be designed having as reference the alternatives presented above.

6.6.4 Major failure deffinition

In Project Charter, it has been stated that a major failure can be defined as the loss of a client's satellite coverage because of a failure in the network. However, this deffinition is not enough precise. For example, during a communication, it can happen that a data packet is lost, or has an error and it is discarded. This means that, for that packet, the communication was lost, but it dones not mean that the communication with the client was lost. Another aspect to take into account is that a satellite may fail, but an alternative path can still exist and, therefore, the communication can continue. Morover, if the client satellite loses all communication with all satellites in range, due to the different orbital velocities of the client satellite and the network satellites, the client satellite will eventually be in range of a functional network satellite.

For all this reasons, a more specific criteria is needed. In Project Charter it has also been stated that the network will provide communication between a client satellite and a ground station with a latency lower than 5 minutes (300 seconds, or 300,0000 miliseconds). A major failure will consiste in a failure in the network that causes a message to arrive from a client satellite to a client ground station with more that 5 minutes of delay, or not arrive at all. Derived from this deffinition, a minor failure can also be deffined. It can be defined as a delay of more that 5 minutes in a communication between a client satellite and a ground station without any failure in the network.

6.6.5 Major failure

Because of the different height of the client satellite and the network satellites, if all the network satellites is range of the client satellite fail, the client satellite may come in range of a working network satellite if enough time passes. In some cases, this can happen in less than 5 minutes and, therefore, it will not be considered a major failure. For this reason, a more critical situation will be considered. It will be considered that the client satellite moves at the same speed as the network satellites, in the same orbital plane. In this situation, a major failure can happen because for three reasons: all network satellites in range of the client satellite fail, all ground stations fail, or some satellites fail but the alternative path takes more that 5 minutes to transmit the information.

6.6.5.1 Satellite in range failure

The first reason will be evaluation in the following lines. Depending on the location of the satellite and the distribution of the satellites in the constellation, the number of adjacent satellites may vary. A satellite over the ecuator can have up to six adjacent satellites. If a client satellite only communicates with this network satellite, a major failure will be the failure of this satellite, as it can be seen in Figure 6.6.1. It can also be tha failure of a group of satellites surrounding the transmitting satellite, but this number is larger and, therefore, it would not be considered since the failure of the transmitting satellite is more restrictive.

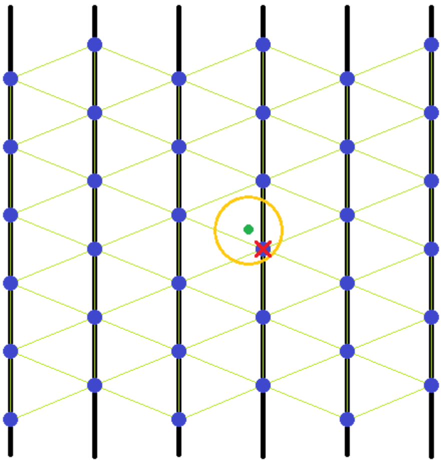


Figure 6.6.1: Failure due to the loss of the only satellite in range of the client satellite.

For antennas with almost half-spherical patterns (an angle of 10º over their horizontal plane has been considered as the minimum angle capable of recieving and transmitting), the minimum height over the satellite network orbit in order to always see more than one satellite is, aproximately, 400 km, considering that our constellation is at 550 km height over the Earth's surface. This means that a significant portion of clients would be in that zone.

For clients that have more than one network satellite in range, the critical failure would be similar as the ones in Figure 6.6.2 and Figure 6.6.3.

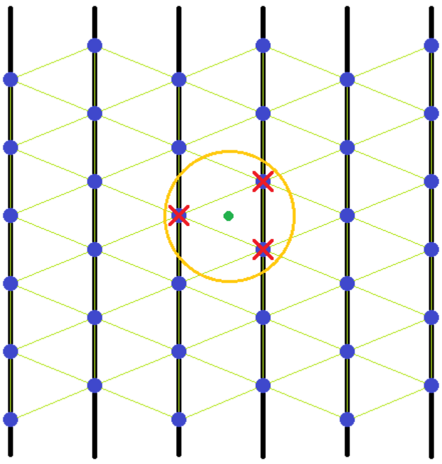


Figure 6.6.2: Failure due to the loss of all possible communication satellites if the client can communicate to three network satellites.

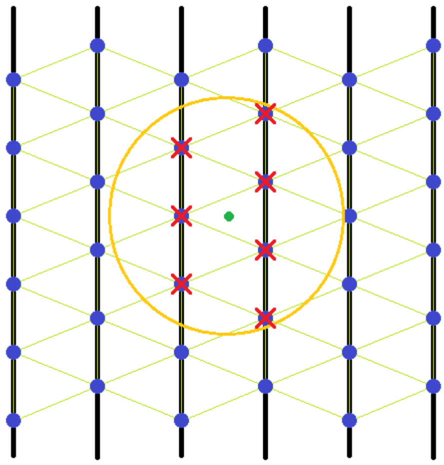


Figure 6.6.3: Failure due to the loss of all possible communication satellites if the client can communicate to seven network satellites.

As it can be seen, the critical failure depends on the communication range of the client satellite. Taking the more restrictive one would mean considering the failure of only one satellite. As this affect a significant amount of potential clients, it can not be neglected.

6.6.5.2 Ground station failure

Since any satellite in the network is able to communicate with any ground station, in order to have a critical failure due to a ground station failure, all ground stations must fail. It will not be considered a failure the loss of connection to a ground station caused by bad weather conditions or radio-fequency interference, since it is not a failure in the network but an anomaly in the medium.

Therefore, for a critical failure caused by ground station failures to happen, all ground stations must fail. Since at least three ground stations will be used, the three of them must fail. As the previous case, the time of failure dones not matter, but the fact that they remain unoperative at a given time.

6.6.5.3 Transmitting time failure

In the following lines, a major failure due to a delay supperior to 5 minutes originated by a failure will be evaluated. First of all, it is needed to evaluate the transmission time. The minimmum data rate that will handle the satellites is 25 Mbit/s. Therefore, it will be considered 25 Mbit/s as the data rate of the satellites, since it is the most restrictive.

The protocols chosen, by deffault, cannot handle data units of more than 62,500 bytes, aproximately. This is 500.000 bits. With the data rate chosen, the time to transmit this information is 0.02 seconds. For a path of 20 nodes, and considering that a satellites recives the entire packet before sending it again, the transmission time will be 0.4 seconds. The transmission is done using electromagnetic waves, which move at the speed of light. For this short distances, it can be considered to be instantaneously. The time used to process each data packet has to be taken into account. If each node needs 1 second to process the packet, the total processing time will be 20 seconds.

Finally, the time to recognize a fallen satellite and the time to compute an alternative route is required. By deffault, OSPF protocol requires 40 seconds of no response to label an adjacent node as dead. When this time expires, the fallen link state will be transmitted. When a node recieves this update, it will wait 5 seconds and then it will calculate new routes. If the process requires 100 seconds, the total time until a failure happens and a new route is calculated is 145 seconds. With the processing time of 20.4 seconds, if one node fails, the time to deliver the message is 165.4 seconds. But if another node fails while the message is still being delivered, the total time to deliver the message would be 310.4 seconds, which is superior to 5 minutes.

Therefore, for a critical failure to happen because of a delay of more than 5 minutes in the communication due to a failure in the network, two satellites must fail in less than 160 seconds, and both of them must be in a communication path between a client and a ground station.

6.6.5.4 Conclusion

It can be concluded that a major failure can happen due to various factors:

- The failure of at one satellites.
- The failure of all ground stations. It would be at least 3 ground stations.
- The failure of at least two satellites in a communication route in less than 3 minutes.

6.6.6 Decision

Having studied all the possibilities of failure and taking into account that the porformance of the satellite is guaranteed for four years the conclusion is that there are no spare satellites needed in-orbit because of the fact that the constellation is dimensioned in order to have the capacity to assume some minor expected failures that will not affect the performance of the entire constellation.

However, there has to be always spares on ground for at least two planes so that in case of a major failure there can be a fast reaction to replace the planes afected. Besides, these satellites will not suppose a great increase in the cost of the constellation because if they are not used as spares they can be used for to following replacement.

6.7 End-of-Life Strategy

6.7.1 Introduction

The main objective is to determine the best strategy to implement at the end of the operational lifetime of the satellites forming the constellation. In this way, it is possible to avoid an increase in space debris and in the collision risk between satellites positioned in the same altitude band or nearby.

6.7.2 Space Debris

The Space had been a virgin environment until the middle of the twentieth century. However, it has already been exploited by humanity. During the last sixty years many space research centers –such as NASA, ESA or ROSCOSMOS- have been sending rockets and satellites to explore and understand its foreign environment without thinking on the consequences it could have. Fortunately, at the twenty-first century the concern about space debris has appears. Due to this fact, all those space research centers have begun to develop end-of-life strategies for all the missions that generate debris to restrict its lifetime.

The term Space Debris implicates all man-made objects that are orbiting with no human control. The problem arises from the fact that depending on the orbital parameters this space stuff is subject to more or less perturbations from either the Earth, the Moon, the Sun or the atmospherically drag and, after their operability's death, they might never disappear or completely disintegrate. As the quantity of space debris is huge and varied, they have been classified in four categories: fragmentation debris, non-functional spacecraft, rocket bodies and mission related debris.

The category that concerns the project is the non-functional spacecraft because it refers to all intact structures which have completed their mission. It is noticed that once satellite's operative lifetime arrives to its end, the satellites stop maneuvering and counteracting perturbations to maintain the current orbit. Consequently, they tend to deviate from their nominal orbital parameters, starting an unknown trajectory and important repercussions.

Therefore, by increasing the number of uncontrolled *"dead"* satellites the probability of collision between working satellites and space debris increases at LEO as it is overcrowded. Space debris is small usually and its location can be followed from earth but is impossible

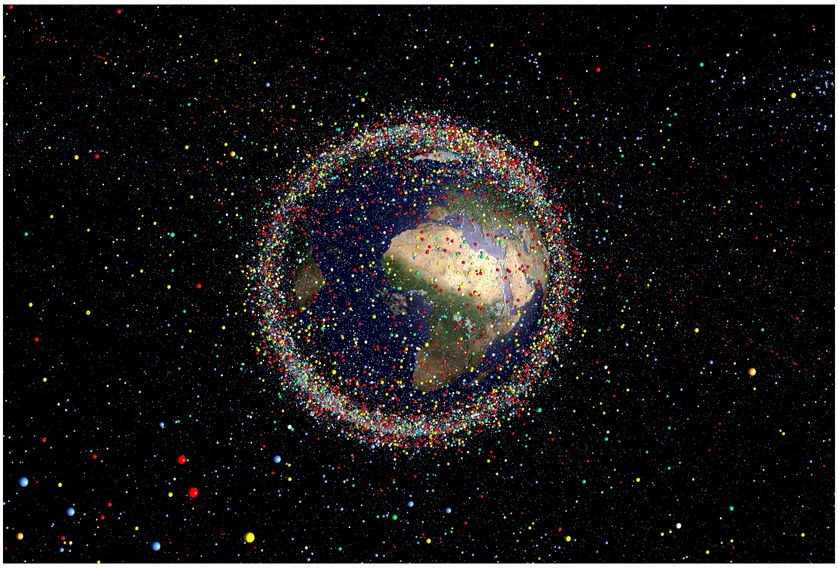


Figure 6.7.1: View of the Space Debris around the Earth

to control it. Meanwhile, it is essential for space assets to be free of any impact because avoidance maneuvers are too complicated to have real success. Thereby, the increasing risk of collision becomes the big threat everyone is fighting against.

6.7.3 End-of-Life Types

End-of-life strategies where implemented taking into account three factors: the time the satellite can orbit, the technical feasibility of active de-orbiting in terms of propellant and sub-systems enhancements and the altitude of its nominal orbital plane.

The first one is related to the fact that the current recommendations say that any space asset that can become a non-functional spacecraft must de-orbit and disintegrate at its twenty-fifth birth on orbit. The second refers to the magnitude of the maneuver that can be developed with the power the thruster system can achieve. The third one is relevant because perturbations in space change according to the distance to the Earth's surface. The closer it is the more perturbations from Earth and drag forces from the atmosphere the satellite suffers and perturbations help to de-orbit and disintegrate space assets.

Based on these premises, two different end-of-life groups had been determined:

- Controlled de-orbit:

  It consists on carrying out a maneuver that leads to steep, controlled re-entry and burn-up in the atmosphere or ground impact. It must be done in a relatively short period of time, usually 1 revolution and it involves significantly high ΔV. This sophisticated maneuver is initiated by a large increment of potential energy to make change the orbital altitude to a lower one well into the atmosphere where the satellite burns. A few calculations are useful to have a numerical result of that ΔV: The velocity in the initial orbit is:

$$V1 = \sqrt{\frac{GM_t}{R_t + h}} = 7593.4m/s$$

Then the semi major axis of the elliptical orbit is obtained:

$$a = \frac{r1+r2}{2} = 6672km$$

The speed at apogee of the elliptical orbit is:

$$V2 = \sqrt{GM_t(\frac{2}{r} - \frac{1}{a})} = 7455m/s$$

Finally, the ΔV is computed:

$$\Delta V = V1 - V2 = 138.4m/s$$

- Uncontrolled de-orbit:

  A simpler and cheaper way to de-orbit satellites is to induce a reduction of the orbit altitude in order to cause a decay and ,finally, a re-entry to the atmosphere. The process is initiated by one or several arc maneuvers at apogee passes and it is carried out without controlling the trajectory. This procedure is appropriate for low-thrust systems and small satellites.
  In addition, when considering satellites placed at LEOs, this strategy takes advantages of the perturbations present in this altitudes (atmospheric drag). This force contributes to the decay increasing the rate of approach to the atmosphere.

In order to make a decision, it has to be considered that the constellation is compounded of very small satellites (3U CubeSats). Those kinds of satellites cannot contain high thrust systems, consequently, the controlled de-orbit is out of its range. Also, the fact that the constellation is placed at LEOs makes easier the application of the uncontrolled de-orbit because of the given reason above. A reason that could force to adopt the controlled de-orbit could be the replacement strategy. If it had been designed so that the de-orbit was rapid, the uncontrolled one would provably not be adequate, nevertheless, the replacement strategy has been designed so as to avoid the need of a quick de-orbit (see Replacement Strategy section). Given all the stated reasons, it is decided to use the uncontrolled de-orbit.

6.8 Conclusions

This final section is intended to put and end to the Constellation Deployment Department activities. First of all, a brief summary of the work done is carried out, secondly, the compliance of the tasks assigned to this department in the Project Charter document is verified. Accomplished tasks:

- Launching System: a launching platform has been chosen regarding all the important parameters. Electron, from the enterprise Rocket Lab is the rocket that will bring Astrea Constellation to life.
- Deployer: a suitable deployer has been selected according to the standards of CubeSat deployment. GPOD deployer, developed by the enterprise GAUSS is in charge of the separation of the CubeSats from the rocket.
- First Placement: the assembly of the satellites will begin approximately 420 days before the first launching. The first placement will consist on eight launchings (one per orbital plane) and will last eight weeks.
- Replacement Strategy: similar to the first placement strategy, new orbital planes are placed between the old ones avoiding the formation of gaps during the decay of the satellites that are being renewed.
- Spare Strategy:
- End of Life Strategy: an uncontrolled de-orbit procedure has been chosen.

The summary shown above demonstrates that the Constellation Deployment Department has fulfilled the requested duties. According to the legislation, both the chosen launcher (Electron) and deployer (GPOD) are certified. Their enterprises designed them strictly following the international requirements.

Part III

Communications

Part IV

Ground Segment Design

Chapter 7

Design of the Ground Segment

7.1 Introduction

The Ground Segment is an indispensable part of almost any space mission. Such is its importance that it can even be seen as a subsystem of the mission.

This subsystem is composed of Ground Stations (GS) and the Mission Control Centre (MCC) and will be responsible of the extraplanetary communications with the spacecrafts. Furthemore, it will operate as a telecommunication port, which means that it will work as a hub, connecting the satellites to the Internet.

In order to establish communication in such high distances ($\approx$ 600km for LEO) high bands radio waves are going to be used. This is a requirement that is going to conditionate the overall Ground Station architecture.

- Since radio waves are going to be used, communication is established only when the Satellite has the Ground Staion in its line-of-sight. That will affect the location. Moreover, the orbits of the satellites will affect the GS location as well. The GS should be placed in a way that it gets maximum coverage time. This point will be further explained.
- Depending on the target band to cover, which is the one used by the satellites for ground segment communication, the GS parts will vary in shape, size and prize significantly.

To use a GS there are two possibilites: building or renting one. In order to know which of the possibilities is the best, in the following lines they will be explained giving some numbers about the cost. First of all, a study about building the Ground Segment will we done, analyzing the location of the GS and the MCC, the legal aspect, the costs and maintenance, and the initial investment necessary to build them. After that, an analysis about renting GS will take place. Finally, a decision will be made.

7.2 Localization of the Ground Stations

The place where the Ground Stations would be placed has to be studied in order to obtain maximum performance of them. This decision will depend mainly of the constellation characteristics, the earth topography and the country legislation and resources. In this chapter the analysis and procedures for arriving to the final decision of where the Ground Stations would be placed are exposed.

Given the constellation topology, the coverage of a Ground Station depending on its longitude and latitude will be studied. The aim of this analysis is to show where a Ground Station would have more coverage and give a first approximation and proposal of the 3 Ground Station placement.

7.2.1 Method

For the purpose above explained, a Matlab algorithm is developed. This algorithm calculates, on a given moment, how many satellites can be seen from a Ground Station. This calculation will be done several times in order to obtain results along time. In order to elaborate the algorithm the steps showed below are followed:

1. Calculate where the satellites are refereed to an inertial Cartesian coordinates system, with the origin at the center of the Earth. This state analysis is done for several time periods with an adequate time-step.
2. Calculate the Ground Station position refereed to the mentioned system. Since the system is inertial, the Ground Station will describe a circle in the rotational plane of the Earth relative to this system. This trajectory depend on the latitude and longitude of the place. This position is calculated for the same time period used before.
3. Calculate, for each time step, how many links can the GS establish. It will depend on the angle between the station and every satellite, and also on the minimum elevation angle.

Once the algorithm is tested and verified, the links during the day for several longitudes and latitudes and how this parameters affect to the coverage of the station are studied. The code used can be found in Attachment XXXXXX, while the study of localization can be found in Attachment XXXXXXX.

7.2.2 Conclusion

To summarize the results of the analysis, for an optimum performance of every Ground Station, they should be located at latitudes between -62.5º and -57.5º or between +57.5º and +62.5º. For a better performance of the system every Ground Station should be 120º of longitude away of the other GSs if they are at the same latitude or 60º of longitude away if they are at the opposite latitude. Taking in account the topography of the Earth, the following options are proposed (every color represent the options for one Ground Station):

Figure 7.2.1: Options for placing the 3 Ground Stations.

Given this possibilities a study of the legislation of the involved countries has to be done in order to know the viability of placing there the Ground Stations. The candidate countries, as is shown in the map, are: Canada, Argentina, Chile, Falkland Islands (Islas Malvinas), United Kingdom, Denmark, Norway, Sweden and Russia.

For the Mission Control Centre, as it does not communicate with any satellite, it has no restrictions on where to build it. It is decided that it will build in Terrassa, since ETSEIAAT is located there and it holds the headquarters of the UPC Space Program.

7.3 Legislation

The legislation will determine the location of the three GS between the locations pre-selected in the previous section. This is done because all the places pre-selected are more or less equivalent, and to choose between them governamental easy will be used. After doing a research on the legislation of all the places where the GS could be placed, only two countries have available legislation: Canada and United Kingdom. For this reason, the location for the 3 Ground Stations are United Kingdom, Falkland Islands and Canada. Falkland Islands are administered by United Kingdom, so the same license must be requested.

7.3.1 United Kingdom Ground Station

Non-Geostationary Earth Stations (Non-Geo). A Non-Geostationary Earth Station is a satellite earth station operating from a permanent, specified location for the purpose of providing wireless telegraphy links with one or more satellites in non-geostationary orbit. Therefore, this is the license required for United Kingdom and Malves Islands.

The form required to ask for the license can be found at [14]. The fees can be obtained from [15] and [16]. The frequency allocation can be found in [17].

7.3.2 Canada Ground Station

The Minister of Industry, through the Department of Industry Act, the Radiocommunication Act and the Radiocommunication Regulations, with due regard to the objectives of the Telecommunications Act, is responsible for spectrum management in Canada. As such, the Minister oversees the development of national policies and goals for spectrum resource use and ensures effective management of the radio frequency spectrum.

In Canada, the fees vary depending on the zone. There are three zones:

- High Congestion Zones: There are six metropolitan areas of Canada designated as zones of intense frequency use. They are in and/or around the following cities: Calgary, Edmonton, Montréal, Toronto, Vancouver and Victoria.
- Medium Congestion Zones: There are 21 areas of Canada designated as zones of moderate frequency usage. These zones can be either stand-alone areas or areas that are adjacent to the six intense frequency use zones listed above. These moderate zones are as follows: Calgary, Chicoutimi, Chilliwack, Edmonton, Halifax, London,

Montréal, Ottawa, the City of Québec, Regina, Saint John, Saskatoon, St. John's, Sudbury, Thunder Bay, Toronto, Trois-Rivières, Vancouver, Victoria, Windsor and Winnipeg.

- Low Congestion Zones: These zones comprise all other areas of Canada.

It would be wise to choose a low congestion zone, which would have additionally less interferences.

The process to fulfill can be found at [18]. The fees might be estimated using [19].

7.4 Annual costs

7.4.1 Annual costs of the Ground Stations and the Mission Control Centre

In order to know the cost that involves having three Ground Station and a Mission Control Centre always operative, an economical study will be done. In this study, parameters such as salaries, electricity and internet services will be taken into account, among othes. This study canbe found in Attachment XXXXXX.

From this study, the following results are obtained:

- Annual cost of the three Ground Stations: 770,000€
- Annual cost of the Mission Control Centre: 460,000€
- Annual cost of the whole Ground Segment: 1,200,000€

7.5 Initial investment

7.5.1 Description of the systems

An S-band system will be used for telemetry and telecommand purposes and for receiving housekeeping data. It is intended to have uplink and downlink capabilities in half-duplex. The model can be found at [20] and [21].

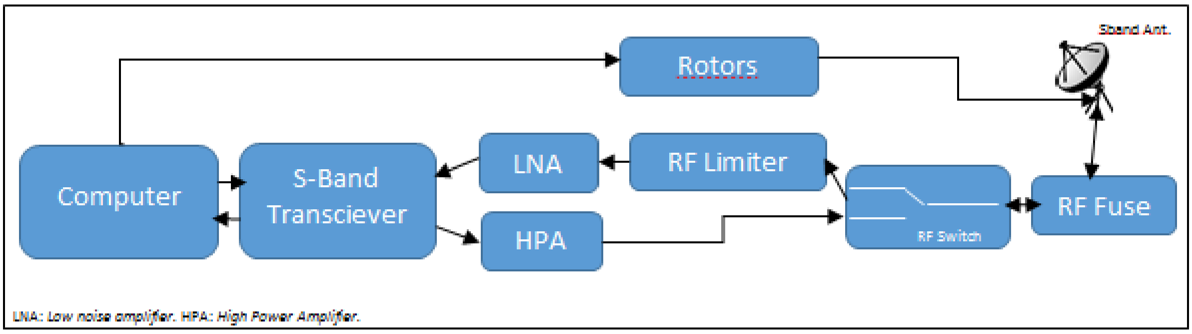


Figure 7.5.1: Equipment needed for S-band communications.

A X-band system will be used for receiving the data requested by the client from the satellites. It will only have downlink capabilities. The model can be found at [22]. beginfigure[H]

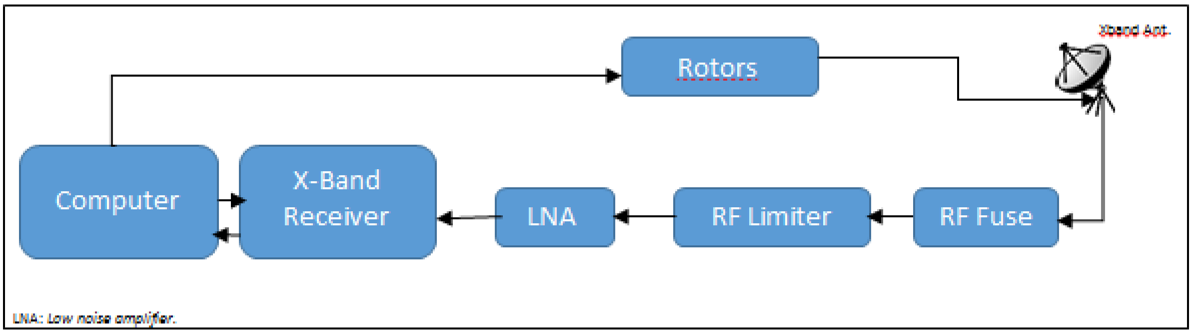


Figure 7.5.2: Equipment needed for X-band communications.

7.5.2 Investment

In order to calculate the initial investment in order to build the entire Ground Segment, another economical study must be done. Same as the previous studies, this one can also be found in Attachment XXXXX. The results of said study will be exposed in the following lines:

- Initial investment of one Ground Station: 356,000€
- Initial investment of the three Ground Stations: 1,070,000€
- Initial investment of the Mission Control Centre: 150,000€
- Initial investment of the whole Ground Segment: 1,220,000€

7.6 Renting of a Ground Station

There are a lot of ground stations spared all over the world. In Atachment XXXXX, a list of the most important Ground Stations and their spacifications can be found.

7.6.1 Contact with GS companies

Some companies that own a Ground Station have been contacted in order to get some information about costs and conditions of renting their stations. However, is important to notice that no answer is given for this type of project (students project). Moreover, information is not available on the Internet. If the project goes ahead, more information could be given to these companies and a cost can be obtained, so the option of renting one of the above cited GS is not discharged. Nevertheless, a cost is needed to know if is better to rent the GS or to build one. To do so, a company named LeafSpace will be used.

7.6.2 LeafSpace

LeafSpace is an italian company which provides a GS network, specifically designed to exchange data with micro and nanosatellites in a fast and simple way. Their global distribution ensures a high visibility time for a wide range of orbits, allowing their customers to download massive amounts of data.
This means that LeafSpace lets customers use their GS to download data, but does not permit to rent them in exclusive, which is the main idea of this project. Due to the small amount of information existing, LeafSpace will be considered in order to get a first approximation and to develop an OWA to decide.

7.6.2.1 Features

Antenna LeafSpace allows to receive data from VHF (137-144 MHz), UHF (400-402 MHz), S-Band (2.2-2.4 GHz) and X-Band (8.025-8.5 GHz), but only can transmit UHF (401-403 MHz) and S-Band (2.025-2.11 GHz). The polarization is RHCP/LHCP (Right and Left Hand Circular Polarization, respectively). The moduation and the protocol are totally configurable. The datarates depend on the bandwidth: for UHF, up to 100 Kbps; for S-Band, up to 30Mbps; and for X-Band, up to 100Mbps.

Pricing The prices, expressed in euros/Mbyte, depend on the bandwidth too: for receiving, VHF 5, UHF 5, S-Band 0.4 and X-Band 0.1, while for transmitting it is UHF

20 and S-Band 2 (recall that they can only transmit in those two bandwidths).

Nevertheless, it is also stated that customized subscriptions are available for missions with large data transfers and constellations. Then, it is highly probable that a better pricing can be achieved.

Boost Performance Within 2017, 20 Ground Stations are scheduled to be implemented all around the World, ensuring a telecommunication service with a considerable increase of visibility time, together with a drastic reduction of communication latency for a wide range of Low Earth Orbits.

Way of use Data management is achieved with a user-friendly web-based interface, along with cloud storage granting direct access to download data at any time.

Since this is all granted by LeafSpace, there would be no need to develop the Ground Segment discussed before.

Services It is claimed to be 24/7 full availability of downloaded data, API access for constellations management, full redundant cloud storage for up to 10 days, advanced levels of data encrypting on demand, automatic scheduling, upling and downling, ranging and tracking, and 24/7 alert service.

Map In the following image there is the planification of Ground Stations to be built in the following years by LeafSpace.

Figure 7.6.1: List of planned LeafSpace Ground Stations.

Operation No information relative to operation is given. It is certainly stated that its working way is automatic. Despite so, some maintenance is surely required, though its cost is probably low.

7.7 Decision taking

In this subsection the decision between building GS or renting existent ones will be taken. There are a few things to be taken into account before starting to talk about the benefits and drawbacks of each of the options.

First of all, the number of ground stations required is needed. If there is no communication with the satellites, the mission would not be accomplished. For this reason, the nodes of the ground stations are very important. The number of ground stations required is the minimum number that, with two failures, can still transfer the data from the satellite of the client to the client itself in less than 5 minutes. Supposing that three ground stations are built or rent, if two of them fail the communication between the client and its satelline can still be done using the left ground station.

Regarding the latency, as it has been already exposed, the communication will take place with a latency of less than 5 minutes, as only one ground station that may fall will be in the communication path and is very improbable that if the ground station fails and the information is redirected to another, the latter falls too in less than three minutes. Regarding the position of the ground station, as the code developed shows, the ideal will be to have them close to the equator, because they would be capabile to stablish more links with different satellites and then the communication to the client's satellite is assured.

The decision will be performed using an Ordered Weighted Average method (OWA) that can be found in Attachment XXXXX. The conclusion of the decision process are explained in the following lines:

7.7.1 Decision

The results of the OWA have been the following ones:

- Building a GS: 0.83
- Renting a GS: 0.67

Looking at the results, building a ground station is the best option for Atrea Constellation in order to accomplish its requirements and to give a high-quality service.

Part V

Satellite design

Chapter 8

Satellite design

8.1 Structure and mechanics

The design and operation of a CubeSat is a complex process that must be completed keeping in mind the different subsystems as well as the role they will play during the lifetime of the mission. And since these systems will operate in space, they have to be prepared and certified to withstand extreme temperature and radiation conditions.

The satellite used by Astrea must have high compatibility between all the systems to avoid potential problems and has to be tested (either all the systems together or one by one) and their correct functioning has to be ensured. Given that the lifetime of the mission should be greater than four years, the critical systems such as the solar arrays, batteries and antennas should be fully operational until the end of the mission.

8.1.1 Structure

The mission of the structure is to sustain and protect all the electronic devices carried by the satellite in order to fulfill the mission requirements. In order to ensure that all the electronic and mechanic systems can be mounted upon the structure, a high compatibility between these systems is required. Given that the configuration of the current CubeSat is not as common as other configurations of actual commercial or operational CubeSats, it is a really important point that the structure is highly flexible regarding the arrangement of the subsystems.

The structure chosen is manufactured by **Innovative Solutions In Space (ISIS)**. Among its features it is worth mentioning that it can withstand the high range of temperature it will face in the space (from -40ºC to 80ºC) and it is highly compatible; almost every physical system used can be placed within the structure or on its faces (such as the antennas or the deployable solar arrays). Finally, the mass of the structure is relatively low, and given that the mass of the other subsystems is sometimes a drawback, it is plus point.

8.1.2 Thermal protection

The thermal protection system consists of various insulating materials that aim to protect the CubeSat from potential thermal shocks. The satellite must remain within an optimal range of temperature, despite of the variation of the external temperature, in order to work properly. Operating in space, the CubeSat is vulnerable to suffer extreme

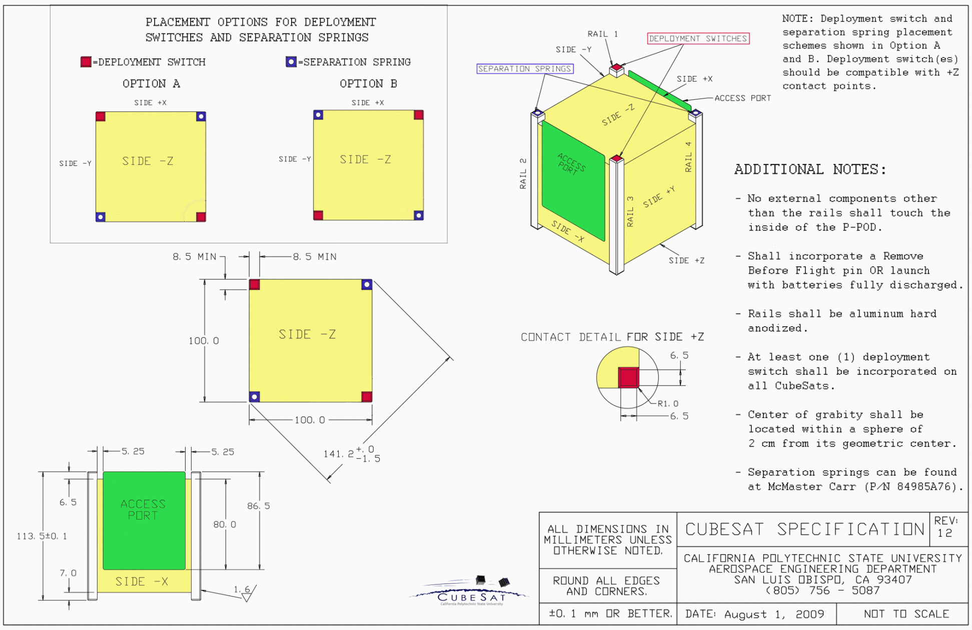


Figure 8.1.1: Dimensions of a 1U CubeSat
[23]

temperatures, both below zero and above zero, and thermal protection must guarantee that all subsystems are protected. Furthermore, the thermal protection system should also dissipate the heat produced by the other systems.

Currently, the most used element as thermal protection in the aerospace industry is the multilayer insulation (MLI), a set of multiple thin insulation layers. The MLI fulfills all the requirements that were previously stated and its main objective is to reduce the heat generated by radiation since the heat generated by convection or conduction does not have such a high impact on the on-board systems.

After a market study, *Dunmore Aerospace* company has been chosen to provide us its MLI product. Specially, the product is the **Dunmore Aerospace Satkit** and it is made for small satellites for LEO and it will provide the CubeSat with the protection required during operation

8.1.3 Study of the commercial available options and options chosen

A broad marked study is needed since all the options have to be considered. For this reason, and with the aim to show all the information and features of each system that has been considered in this section, the table 8.1.1 is presented below.

Brand and model	Features	Total price (€)
Structure		
ISIS 3U structure	Low mass (304.3g) Highly compatible High temperature range	3900
Gomspace GOMX-Platform	High mass (1500g) Comes fully equipped (basic systems) High temperature range	11000
Thermal protection		
Dunmore Aerospace Satkit	Lightweight Durability Made for small satellites	1000
Dupont Kapton Aircraft Thermal	Lightweight Durability Non-flammable	1400

Table 8.1.1: Options studied for the structure and thermal protection

Finally, the options chosen are presented in the table 8.1.2.

System	Brand and model	Price per unit (€)	N. of units
3U Structure	ISIS	3900	1
Thermal Protection	Dunmore Satkit	1000	1

Table 8.1.2: Options chosen for the structure and thermal protection

8.2 Electrical Power System

The electric power system of the satellite must provide and manage the energy generated efficiently in order to have all the systems operating under normal conditions during the lifetime of the mission. The EPS of the Cubesat is, probably, the most fundamental requirement of the satellite, since its failure would result in a mission failure.

The energy collection system and the power management and collection systems compose the EPS and their role is to control and distribute power to the Cubesat, to suppy a continuous source of electrical power during the length of the mission, to protect the satellite against electrical bus failiures and to monitor and communicate the status of the EPS to the on-board computer.

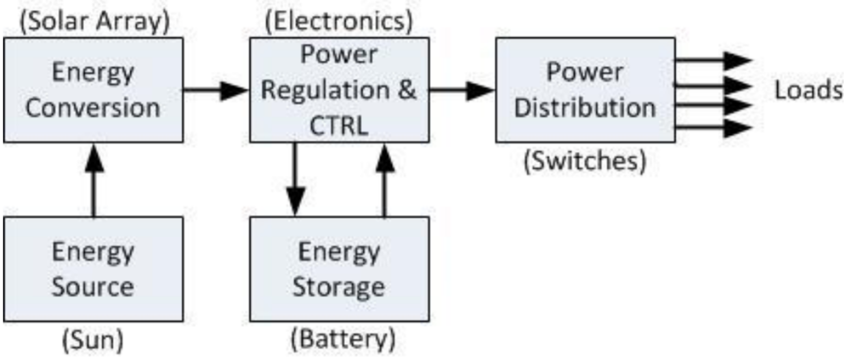


Figure 8.2.1: Basic schematics of the EPS
[24]

8.2.1 Estimation of the power required

To select the adequate electrical power systems it is essential that the power consumed by the CubeSat is known *a priori*. Thus, to select the solar arrays and the batteries, as well as the power management system, an estimation of the power consumed has to be made.

The vast majority of the time the satellite will work under typical operation conditions. However, the estimation of the power consumption provided in the table 8.2.1 has been made for typical-high conditions in order to have a power margin and a more reliable estimation.

System (number of units)	Typical power consumption per unit (W)
Payload	
Patch antenna (8)	4
Payload power consumption	32
Electrical Power System	
NanoPower P60 Power Module (1)	2
Battery (2)	-
Solar arrays (4)	-
EPS power consumption	2
Data Handling Systems	
Transceiver inner-satellite (3)	4
Transceiver space to ground (1)	4
Data handling system (1)	4
DHS power consumption	15
Propulsion and ACDS	
Thruster (1)	20
ADACS (1)	3
OACDS power consumption	3
Estimated total power consumption	52

Table 8.2.1: Estimation of the power consumption under typical working conditions

Additionally, it is worth mentioning that the thrusters are not included in the final estimated power. The thruster will only be active for shorts periods of time to maintain the orbit, and when it ignites, the other subsystems will not perform in typical conditions. The CubeSat will manage to send only the essential information to the other satellites and, since it is unlikely that their thruster is ignited, the communication is ensured during the maneuver.

8.2.2 Solar arrays

Given that the space of a 3U CubeSat is very limited, the primary source of electrical power has to be photovoltaic cells. The photovoltaic cells will collect and convert the energy of the sun into electrical energy and they have to be correctly selected to prevent failure given their importance.

The solar arrays used must have a decent efficiency and capacity to collect the energy from the sun, have to keep their mass relatively low, must have a protective radiation shield to ensure their full efficiency for at least 4 years, a proper deployment system, the ability to withstand space conditions and also must be highly compatible with all the other systems used, especially the power management system (the *NanoPower P60*).

The option selected for the mission is a set of deployable solar panels provided by **EXA (Agencia Espacial Civil Ecuatoriana)**. These solar arrays fulfill all the requirements mentioned above: they are low mass (135g per unit), they have a protective radiation shield (NEMEA Anti Radiation Shield protects the solar panels of EM, High Gamma, X-Ray, Alfa, Beta and low neutron radiation) they can withstand a very high temperature range (from -80ºC to 130ºC) ensuring that they can operate in space, they have a gentle release and deployment system with artificial muscles (developed by EXA) and they provide a power of 16.8W each (19.2V@0.5A).

Every cubesat will come with at least 4 deployable solar panels providing it with 67.2W of power, approximately, to supply peak demands during the lifetime of the mission. Additionally, it is worth mentioning that these solar arrays are compatible with the hardward used (the structure and the power management system).
Note that these 4 deployable solar panels are a basic requirement. If more space is available on the faces of the satellite, additional 1U non-deployable solar arrays (giving an extra power of 2.3W per array, approximately) or 1U deployable arrays (giving an extra power of 16.8W or 10W) will be placed. They are also low mass equipment (about 80g per array) as the deployable solar arrays and highly compatible with the CubeSat. Their current and voltage are different but given that the CubeSat will be equipped with the NanoPower P60, that should not be a problem. The only drawback of these arrays is that they may be only fully operational for 2 years in LEO. However, that does not mean they will not work anymore after these 2 years; it means that they will start losing efficiency.

8.2.3 Power management system

The role of the power management system is to distribute the power and supply the energy to the different systems used in the CubeSat. Since the systems of the CubeSat have different power and energy needs, the power management system has to be highly compatible and have a number of buses high enough to supply the different voltage and intensity required to the systems.

The selected option for the mission is the **NanoPower P60** by **Gomspace**, a high-power EPS for small satellites that comes with 1 motherboard, 1 ACU module (Array Conditioning Unit) and 1 PDU (Power Distribution Unit), allowing multiple configurations in just one motherboard; saving a lot of space.

The motherboard supports up to 4 ACU and PDU modules and has different regulated outputs (3.3V and 5V). It means that with one single motherboard, several conditioning and distributing units can be connected. That ensures that additional equipment (ACU and PDU) could be linked to the motherboard if something failed in the assembly process.

The ACU module 6 different inputs per unit with a high voltage solar input (up to 16V or 32V). Additionally, each input can withstand a maximum current of 2A and current and voltage inputs are measured on each input channel and the measurements can be communicated to the onboard computer.

The PDU module has 9 different outputs per unit that are highly configurable. Each module has 3 configurable output voltages (3.3V, 5V, 8V, 12V, 18V, 24V) and each of the outputs can withstand a maximum current of 1A or 2A (programmable). Additionally, like the ACU module, current and voltage outputs are measured on each output channel and can be effectively communicated to the onboard computer.

All these features make the **NanoPower P60** a very efficient and configurable power management unit that fulfills the mission requirements. Furthermore, given this capacity to configure each input and output channel and the high number of channels that it has, the compatibility between all the systems used in the satellite is ensured. Additionally, the communication between this system and the onboard computer in order to detect potential failures is a really adequate feature.
With the NanoPower P60 we aim to distribute the energy to all of the subsystems of the CubeSat.

8.2.4 Batteries

Batteries are essential for a proper mission operation. They will provide the spacecraft subsystems with the power needed when the solar arrays are working less efficiently or not properly. Astrea is looking for decent capacity batteries that provide a slightly high typical energy and power supply, since all the systems will not usually operate under peak conditions. Additionally, through the lifetime of the mission, the solar arrays will face an important unfavorable condition; in the worst case scenario, the satellite will be in the dark during half of the period of the orbit. So, it is clear that the batteries are a critical system of the CubeSat

Among all the commercial options, Astrea has chosen the **BA01/D** batteries manufactured by **EXA-Agencia Espacial Civil Ecuatoriana**. The CubeSat will have two of these batteries, with a total capacity of 28800mAh or 106,4Wh. Each battery has a total of 16 cells, highly stackable and with a very low mass (155g per unit). They also come with unique thermal transfer bus, that will transfer the heat of the other subsystems to the batteries to keep their temperature under efficient working conditions.

The output voltage can be configured (3.7V and 7.4V) and they are perfectly compatible with the solar arrays. Furthermore, they come with a protective radiation shield (NEMEA) that ensures at least 4 years working under full efficiency conditions in a LEO. It is also worth mentioning that if the company that will assemble the CubeSat faces problems during this part of the process, the batteries can be customized by contacting EXA.

As mentioned above, if the satellite was in the dark during half of the period of the orbit, the estimated energy that it would need would be 50W. Thereby, the capacity of the batteries is more than enough to supply the required energy in the worst case scenario. In fact, they will supply energy when the energy demand of the CubeSat is higher than the energy collected by the solar cells. And logically, they will store the energy collected by the solar arrays when the energy demand of the systems is lower than the energy collected.

8.2.5 Study of the commercial available options and options chosen

A broad marked study is needed since all the options have to be considered. For this reason, and with the aim to show all the information and features of each system that has been considered in this section, the table 8.2.2 is presented below.

Brand and model	**Features**	**Total price (€) per unit**
Solar arrays		
EXA-Agencia Espacial Ecuatoriana	Total power of 67.2W (4units) Mass of 270g (p.unit) Included thermal protection At least 4 years lifetime	17000
ISIS	Total ower of 30W (4units) Mass of 150g (p.unit) No thermal protection At least 2 years lifetime	9000
Power management		

Crystalspace P1 Vasik	Mass of 80g Full redundancy Low volume 6x outputs Up to 10W input High temperature range	5400
Gomspace NanoPower P60	Mass of 176g 9x configurable outputs 6x inputs per module EMI shielding High temperature range	16000
Batteries		
Gomspace NanoPower BP4	Total capacity of 77Wh (2u) Automatic heat regulation Highly stackable Mass of 270g (p.unit)	3250
EXA-Agencia Espacial Ecuatoriana	Total capacity of 106.4Wh (2u) Automatic heat regulation Highly stackable Total mass of 155g	6300

Table 8.2.2: Options studied for the Electric Power System

Finally, the options chosen are presented in the table 8.2.3.

System	Brand and model	Price per unit (€)	N. of units
Solar arrays	EXA	17000	4
Additional solar arrays	-	4000-12000	depends
Batteries	EXA	6300	2
Power Management	Gomspace NanoPower P60	16000	1

Table 8.2.3: Options studied for the Electric Power System

8.3 Propulsion Systems

8.3.1 Requirements

There is a big risk of a collision with space debris while a spacecraft is operating in Low Earth Orbits. The Inter-Agency Space Debris Coordination Committee recommended to the United Nations (section 5.3.2 'Objects Passing Through the LEO Region'): "Whenever possible space systems that are terminating their operational phases in orbits that pass through the LEO region, or have the potential to interfere with the LEO region, should be de-orbited (direct re-entry is preferred) or where appropriate manoeuvred into an orbit with a reduced lifetime. Retrieval is also a disposal option." and "A space system should be left in an orbit in which, using an accepted nominal projection for solar activity, atmospheric drag will limit the orbital lifetime after completion of operations. A study on the effect of post- mission orbital lifetime limitation on collision rate and debris population growth has been performed by the IADC. This IADC and some other studies and a number of existing national guidelines have found 25 years to be a reasonable and appropriate lifetime limit." [25]

Thus, a proper propulsion system is needed both for maintaining the satellite's orbit and for de-orbiting after the mission's lifetime.

Given the size of the CubeSat, not many effective options are available and a committed solution has to be found in order to follow the recommendations by the IADC.

8.3.2 Thrusters

Thruster is a main part of the structure because it is needed to allow the satellite to realise different maneuvers how incorporate it adequatly to the orbit after the deployment of the rocket, can obtain the optimal orientation or to mantain the satellite in the orbital and avoid its fallen.

The main parameters that must consider are thrust, total specific impulse,power required, weight of the propulsion subsystem and its volume.

At the moment, the most used and more modern thrusters for satellites are: ionic, pulsed plasma, electrothermal and green monopropellant thrusters. An important aspect to consider is that the goal is to reduce the mass required although this will cause minor accelerations than conventional engines but it will be suitable for small satellites.

After a market study,the best two options to consider are the green monopropellant thruster BGT-X5 and the ion thruster BIT-1, both from Busek company. These two thruster are among the most used in the aerospace industry for small satellites. The main difference between both are the thrust and the specific impulse. On the one hand, the BIT-1 thruster provides a lower thrust but with a high specific impulse. On the other hand, BGT-X5 thruster provides a high thrust, around 0.5 N but with a lower specific impulse.

Finally, BGT-X5 has been chosen how the CubeSat thruster. With the high thrust and delta V that BGT-X5 provides, the CubeSat will be able to carry out the necessary actions to keep the satellite in orbit, to relocate the satellite or to change its orbit.

The following table 8.3.1 shows the main parameters of this thruster.

BGT-X5	
PARAMETERS	**VALUE**
Total thruster power	20 W
Thrust	0.5 N
Specific impulse	225 s
Thruster Mass	1500 g
Input voltage	12 V
Delta V	146 m/s

Table 8.3.1: Main features of BGT-X5

8.3.3 Study of the commercial available options

A broad marked study is needed since all the options have to be considered. For this reason, and with the aim to show all the information and features of each system that has been considered in this section, the table 8.3.2 is presented below.

Brand and model	Features	Total price (€)
Propulsion		
Busek ion thruster BIT-1	Volume 1/2 U High Isp (2150 s) Low thrust (100 uN)	58000

Busek BGT-X5	Volume 1 U High thrust (0.5 N) High delta V (146 m/s)	50000

Table 8.3.2: Options studied for the propulsion system

Finally, the option chosen is presented in the table 8.3.3.

System	**Brand and model**	**Price per unit (€)**
Propulsion	Busek BGT-X5	50000

Table 8.3.3: Option chosen for the propulsion system

8.4 Attitude and Orbital Control Systems

Attitude and orbital control subsystem is needed to enable the satellite to keep a specific position within its orbit and to control the antennas in order to remain oriented to assigned area, because the satellite tends to change its orientation due to torque. The AOCS receives telecommands from the central computer and acquires measurements (satellite attitude and orbital position) from sensors. We will also refer to the attitude control as ADACS (Attitude Determination and Attitude control system).

Attitude control for CubeSats relies on miniaturizing technology without significant performance degradation. Tumbling typically occurs as soon as a CubeSat is deployed, due to asymmetric deployment forces and bumping with other CubeSats. Some CubeSats operate normally while tumbling, but those that require pointing in a certain direction or cannot operate safely while spinning, must be detumbled. Systems that perform attitude determination and control include **reaction wheels**, **magnetorquers**, **thrusters**, **star trackers**, **Sun sensors**, **Earth sensors**, **angular rate sensors**, and **GPS receivers and antennas**. Combinations of these systems are typically seen in order to take each method's advantages and mitigate their shortcomings. **Reaction** wheels are commonly utilized for their ability to impart relatively large moments for any given energy input, but reaction wheel's utility is limited due to saturation, the point at which a wheel cannot spin faster. Reaction wheels can be desaturated with the use of thrusters or magnetorquers. **Thrusters** can provide large moments by imparting a couple on the spacecraft but inefficiencies in small propulsion systems cause thrusters to run out of fuel rapidly. Commonly found on nearly all CubeSats are **magnetorquers** which run electricity through a solenoid to take advantage of Earth's magnetic field to produce a turning moment. Attitude-control modules and solar panels typically feature built-in magnetorquers. For CubeSats that only need to detumble, no attitude determination method beyond an angular rate sensor or electronic gyroscope is necessary (*wikipedia extract*, [**?**]).

Pointing in a specific direction is necessary for Earth observation, orbital maneuvers, maximizing solar power, and some scientific instruments. <u>Directional pointing accuracy</u> can be achieved by sensing Earth and its horizon, the Sun, or specific stars. <u>Determination of a CubeSat's location</u> can be done through the use of on-board GPS, which is relatively expensive for a CubeSat, or by relaying radar tracking data to the craft from Earth-based tracking systems (*wikipedia extract*, [**?**]).

8.4.1 Orbital Control

Orbital control will be achieved as a combination of two systems. ADCS will orient the thrust, this thrust will be given by the propulsion system and all the operation will be controlled on the On-Board Computer. Principally, the orbit control will be necessary to mitigate orbital debris effect on every satellite.

8.4.2 Study of the commercial available options

Because AOCS involve so many systems working together, full assembled module had been considered in order to avoid compatibility issues.

ADACS options		
Features	**CUBE ADCS**	**MAI-400 ADACS**
Power	3.3/5 VDC Peak: 7.045W	5 VDC Peak: 7.23W
Mass	506 g	694 g
Size	90 x 90 x 58 mm	10 x 10 x 5.59 cm
Sensors	3-Axis Gyro Fine Sun & Earth sensor Magnetometer 10x Coarse Sun Sensors Star tracker(optional)	3-axis magnetometer Coarse sun sensor EHS Camera
Actuators	3 reactions wheels 2 torque rods	3 reactions wheels 3 torque rods
Computer	4-48 MHz full ADCS + OBC	4Hz Provides telemetry
Control Board	Works as OBC included	MAI-400 not included

Table 8.4.1: Main ADACS features

Decision After the study of commercial options available, the previous two where the unique that fitted in AstreaSAT requirements, so a decision between these two must be done. Since all the features tabulated on 8.4.1 are critical, the same weights are given. Therefore, we will compare directly the two alternatives for choosing the best alternatives. Taking into account that we need: low power consumption, low, weight and size, high pointing accuracy and really versatile systems that can integrate multiple subsystems; **CUBE ADCS** is chosen. It has the lowest mass and power consumption, it also offers a higher attitude determination systems, redundancy is a key fact because we can not loose precision during the life time of each satellite. Finally, the fact that CUBE ADCS integrates also and On-Board Computer (OBC) is the turning point, because we have size and weight limitations, having and integrated, high performance OBC in this system will make able TT&C with the ground stations and the control af every system on board. http://www.cubespace.co.za/cubecomputer

8.5 Payload

Aim AstreaSAT payload, needs to provide a radio link to the client satellites, for real time data relay with no less than 25MB/s of data rate. For achieving its porpoise, the payload will consist on a pack of arrays of antennas and data handling computers.

AstreaSAT payload will have to have three types of radio links for transmitting in every condition the data received from the clients:

- **Space to Ground link**: Connection between satellite and Ground Station when it is possible.
- **Inter-satellite Space to Space link**: Communication between Astrea satellites for data relay, looking for the nearest satellite with Ground Station link available, to transmit the data.
- **Client Space to Space link**: Communication between client and Astrea satellites.

The radio frequencies that we can use to establish the previous described links are regulated in [26] by frequency, bandwidth and type of communication . So, for the **Space to Ground link** we can use frequencies from **70MHz** to **240GHz**; for **Inter-satellite Space to Space link** plus data relay type of communication, frequencies are **2-2.4GHz**, **4-4,4GHz** and **22-240GHz**. Finally, **Client Space to Space link**, they exist to cases; on the one hand, the client points towards the Earth like a standard satellite, we capture its signal and make the data relay, since it is like a Space to Ground communication and also like a inter-satellite communication, we can combine the two previous restrictions. On the other hand, if the client satellite is below our constellation, we only had inter-satellite communication, therefore **Inter-satellite Space to Space link** rules are applied.

Finally, the Payload will consist on a combination of antennas, transceivers and data handling systems which will combine to create a data relay module.

8.5.1 Antennas

The antennas are essential in this mission, since their role is to transmit and receive the data from other satellites as well as the ground stations. In order to provide fast and reliable communication, several options have been studied and information about their main parameters is presented below.

It has to be kept in mind that the mass of the antennas should be as low as possible given that there are already a lot of subsystems in the CubeSat and the mass limitation is about 4kg. Additionally, the power consumption has to be kept as low as possible given the limitations regarding to the power supply of the CubeSat. The antennas must be certified to work under space conditions (high temperature range and radiation protection shield). Preliminary, after a first satellite preliminary design, seems that patch and turnstile antennas will cover the needs of AstreaSAT.

8.5.1.0.1 Basic parameters

The **frequency range** is one of the most important parameters, since it is related to an effective satellite-satellite and satellite-ground station communication. The frequency range should be between 1GHz and 10GHz, which is a very demanding condition given that the CubeSat has a limited space and power supply. Those frequencies, assure the desired data rates an negligible atmosphere attenuations.
For an effective communication, the signal has to be able to trespass the atmosphere without a high number of losses and interference. The high frequency range allows the signal to go through this barrier and reach the ground stations.

The **bandwidth** is the frequency range in which the highest power of the signal is found. It is really important to have a high bandwidth to have a great performance and avoid extremely high signal losses.

The **gain** of an antenna is the ratio between the power density radiated in one direction and the power density that would radiate an isotropic antenna. The best option is to have a high gain.

The **polarization** of an antenna is the orientation of the electromagnetic waves when they are leaving it. There are three types of polarization: linear, circular and elliptical. For a high performance, the receiver antenna and the transmitter antenna should have the same polarization. It has been derived that the best option for the project is an antenna with circular polarization; these types of antennas are able to keep the signal constant regardless of the appearance of different adverse situations such as the relative movement of the satellites with respect to the ground station.

8.5.1.0.2 Patch antenna

A **patch antenna** is a type of radio antenna with a low profile, which can be mounted on a flat surface, It consists of a flat rectangular sheet or "patch" of metal, mounted over a larger sheet of metal called a ground plane. They are the original type of microstrip antenna described by Howell in 1972. [**?**, wikipedia]

Patch antenna AntDevCo	
Features	**Value**
Bands	L,S,C,X
Frequency range	1-12 GHz
Bandwidth	20 MHz
Gain	6 dBi
Polarization	Circular
Maximum power consumption	10 W
Impedance	50 Ohms
Operational temperature range	-65°C to +100°C
Mass	<250 grams

Table 8.5.1: Main features of the patch antenna

8.5.1.0.3 Turnstile antenna

A **turnstile antenna**, or crossed-dipole antenna, is a radio antenna consisting of a set of two identical dipole antennas mounted at right angles to each other and fed in phase quadrature; the two currents applied to the dipoles are 90^o out of phase.

Turnstile antenna ANT430	
Features	**Value**
Frequency range	400-480 MHz
Bandwidth	5 MHz
Gain	1.5 dBi
Polarization	Circular
Maximum power consumption	10 W
Impedance	50 Ohms
Operational temperature range	-40°C to +85°C
Mass	30 grams

Table 8.5.2: Main features of the turnstile antenna

8.5.2 Antenna selection

After a market study, the best two antennas to add in the CubeSat are the patch antenna AntDevCov and the turnstile antenna ANT430 Gomspace. The number of units of each antenna are 4 and 2 respectively. The 4 patch antennas will be placed on each side face of the CubeSat and they will occupy a 1U face. The 2 turnstile antennas will be placed on the upper and lower face of the CubeSat and, as they do not occupy space, other systems such as a solar panel or the thruster can be placed on those faces.

Other antenna types, like helicoidal deployable antennas, parabolic antennas or monopole antennas, had been discarded because of their big volume and mass or because the don't accomplish the preliminary requirements stated on the project charter.

Nevertheless, this is only a preselection. After the link budget study and negotiation with communications department changes can be made if it is necessary.

8.5.3 Payload Data Handling Systems

Every AstreaSAT will act as a router to transmit client data to the ground. This initial raw data, should be temporally stored into the satellite in order to process it, if necessary. Since, to down-link the data, first the satellites need to establish connection, data can not be directly retransmitted to other sources (Ground Station or satellite) as it enters to the satellite. Furthermore, non loss compression algorithms can be applied to reduce the data size load and achieve higher data transmission velocities.

To sum up, Payload Data Handling System of every AstreaSAT (PDHS) will be able to receive, process and send the client data, using the integrated transceivers (transmitter + receiver) for sending the data and the PDHS computer to process it. PDHS have a hard disk associated which will temporally store the client data.

Finally, is necessary to find the transceivers and PDHS computers compatible combination in order to achieve the specifications stated on the Project Charter.

8.5.3.0.1 Transceivers

A transceiver is a device comprising both a transmitter and a receiver that are combined and share common circuitry or a single housing. For the preliminary design, because we know that they should satisfy all the connectivity options, we are restricted to the S, K or higher bands for **Inter-satellite communication** and not restriction virtually at all for **Space to Ground** communication. Nevertheless, together with the communications department, X band is chosen as the frequency to talk to the floor because several factors: the use in

Transceivers options - Inter-satellite comm.(S band)		
Features	**NanoCom TR-600**	**SWIFT-SLX**
Band	70 - 6000 MHz	1.5 - 3.0 GHz
Bandwidth	0.2 - 56 MHz	10+ MHz
Vcc	3.3V	6 - 36V
Max. Power consumption	14W	10.8W
Dimensions	65 x 40 x 6.5 mm	86 x 86 x 25-35mm
Operational temperature range	-40°C to +85°C	-35°C to +70°C
Mass	16,4 grams	250 grams

Table 8.5.3: Main inter-satellite communication transceivers features

NanoCom TR-600 has an additional advantage, GOMspace, the supplier, offers it in combination with the NanoMind Z7000 seen in PDHS computers section. Both integrated on a board able to hold three TR-600 transceivers and one computer. The low dimensions, high bandwidth (associated to high data rates) and low mass of TR-600 versus SWIFT-SLX, makes the first, a great choice for Inter-Satellite communication.

Transceivers options - Space to Ground comm.(X band)		
Features	**SWIFT-XTS**	**ENDUROSAT**
Band	7 - 9 GHz	8.025 - 8.4 GHz
Bandwidth	10 - >100 MHz	10+ MHz
Vcc	3.3V	12V
Max. Power consumption	12W	11.5W
Dimensions	86 x 86 x 45mm	90 x 90 x 25mm
Operational temperature range	-40°C to +85°C	-35°C to +70°C
Mass	350 grams	250 grams

Table 8.5.4: Main space to ground communication transceivers features

SWIFT-XTS is pretty similar to ENDUROSAT, but presents some advantages. The higher Bandwidth, will make possible higher communication data rates. The higher mass respect to ENDUROSAT could be a problem, from the link budget analysis a decision will could be made, because the most important factor is the possibility to transmit with low losses to the ground.

8.5.3.0.2 PDHS computers

PDHS computers will process and store the clients data before the data relay is done.

PDHS computers options		
Features	**NanoMind Z7000**	**ISIS iOBC**
Operating System	Linux	FreeRTOS
Storage	4GB to 32 GB	16GB
Processor	MPCoreA9 667 MHz	ARM9 400 MHz
Vcc	3.3V	3.3V
Max. Power consumption	30W	0.55W
Dimensions	65 x 40 x 6.5mm	96 x 90 x 12.4mm
Operational temperature range	-40°C to +85°C	-25°C to +65°C
Mass	28.3 grams	94 grams

Table 8.5.5: Main PDHS computers features

The main advantage of NanoMind Z7000 over ISIS iOBC is the computing availability, because of its two 667MHz processor Z7000 can handle higher data payloads and processit at higher velocities, reducing in last term delay between communications. Also, Z7000 presents a lower mass, critical think in our mass limitation of 4kg. But the turning point is, as stated before, Z7000 comes integrated on a single board with a maximum of three NanoMind TR-600 transceivers, fact that makes it a perfect option to build a data relay module payload.

8.5.4 Study of the commercial available options and options chosen

A broad marked study is needed since all the options have to be considered. For this reason, and with the aim to show all the information and features of each system that has been considered in this section, the table 8.5.6 is presented below.

Brand and model	Features	Total price (€)
Antennas		

Patch antenna AntDevCo	High frequency range (L,S,C,X bands) High bandwidth High mass (120 g)	18000 (7000)
ISIS monopole deployable antenna	Low frequency range (10MHz) Higher mass than ANT430 (100 g) Deployable Not occupy space	17000
Turnstile antenna ANT340 Gomspace	Low frequency range (400-480 MHz) Low mass (30 g) Deployable Not occupy space	9500
Transceiver inter-satellite		
NanoCom TR-600	SDR including S band High Bandwith Low mass and dimensions Integrated with other PDHS	8545
SWIFT-SLX	Low power consumption High mass and dimensions Narrow bandwidth	7800
Transceiver space to ground		
SWIFT-XTS	High bandwith High mass Standard dimensions	5500
ENDUROSAT	Narrow bandwidth Lower mass Standard size	22500
PDHS Computers		
NanoMind Z7000	LinuxOS High processing velocity High power consumption Low mass and dimensions	5000
ISIS iOBC	FreeRTOS OS Less computing velocity High dimensions and mass	9400

Table 8.5.6: Options studied for the payload

Finally, with the aim to clarify all the information of this section, the chosen systems and components are presented in the table 8.5.7.

System	Brand and model	Price per unit (€)	N. of units
Antenna	Patch antenna AntDevCo	TO REQUEST!	8
Transceiver	NanoCom TR-600	TO REQUEST!	3
Transceiver	SWIFT-XTS	TO REQUEST!	1
PDHS	NanoMind Z7000	TO REQUEST!	1

Table 8.5.7: Options chosen for the payload

8.6 Communication module

The telemetry subsystem analyses the information of the ground station and other sensors of the satellite in order to monitor the on-board conditions. With this system, the CubeSat is able to transmit the status of the on-board systems to the ground station.

The command and control subsystem (TT&C) allows the ground station to control the satellite.

Every Astrea satellite (AstreaSAT) of the constellation, will need to report its operating status to the ground and receive commands from the ground. TT&C operations will usually be performed when the satellite flights over the coverage of the constellation ground station, but since the satellites are interconnected, there is the possibility to perform this operations via data relay links between satellites. As a collaboration with the communications department, S band frequency is chosen for TT&C operations, since there is no need for high data rates, the lower band will significantly reduce the power consumption.

Communication to the ground will be perform with a NanoCom TR-600 transceiver module attached to AntDevCo Patch antenna, both configured for S band frequency communication.

8.7 Link Budget

Astrea constellation main satellite must be able to stablish three different telecommunications link:

- Space to Ground link for payload and TT&C data.
- Space to Space link between Astrea satellites.
- Space to Space link between client and Astrea satellites.

8.7.1 Communications Basics

When evaluating a wireless link, the three most important questions to be answered are: [10]

1. How much radio frequency (RF) power is available? Up to 2W for S band or up to 12W for Xband.
2. How much bandwidth is available?
   Available 400MHz with 28 channels of 14MHz or 228 channels of 1.75MHz for inter-satellite communication at S band. For X band, there's more than 4GHz available [26]. In fact is limited by the TR-600 transceiver at 56MHz for S band and to 100MHz by SWIFT - XTS at X band.
3. What is the required reliability (as defined by Bit Error Rate, or BER)?
   Required reliability for space systems$E_b/N_o \geq 10$, so $BER = 5.5x10^{-6}$ for a MSK, PSK (worst case) modulation as shown in Fig.8.7.5.

The upper limit in terms of data rate is given by Shannon's Channel Capacity Theorem:

$$C = Blog_2(1 + S/N) \tag{8.7.1}$$

where:

C = channel capacity (bits/s)
B = channel bandwidth (Hz)
S = signal strength (watts)
N = noise power (watts)

With all data known, the minimum required sensitivity of a receiver using the Eq. 8.7.1 will be stated in the Link Budget calculation.

Transmission Losses In any satellite transmission, there are always losses from various sources. Some of those losses may be constant, others are dependent of statistical data and others vary with the weather conditions, especially with rain.

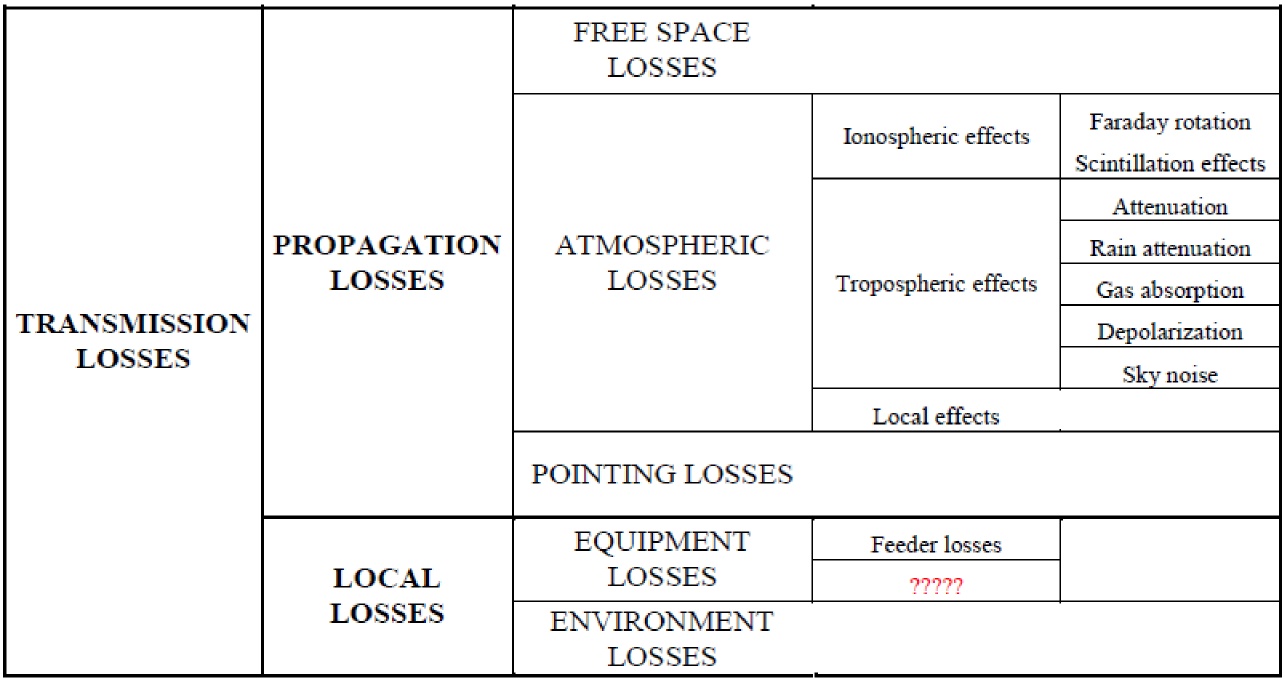


Figure 8.7.1: Principal losses in the received signal [9]

8.7.2 Propagation losses

8.7.2.0.1 Free Space Losses

Range and Path Loss Another key consideration is the issue of range. As radio waves propagate in free space, power falls off as the square of range. For a doubling of range, power reaching a receiver antenna is reduced by a factor of four. This effect is due to the spreading of the radio waves as they propagate, and can be calculated by [10]:

$$L = 20log_{10}(4\pi D/\lambda) \tag{8.7.2}$$

where:

$$\begin{aligned} D &= \text{the distance between receiver and transmitter} \\ \lambda &= \text{free space wavelength} = \text{c/f} \\ c &= \text{speed of light}(3\text{x}10^8 m/s) \\ f &= \text{frequency (Hz)} \end{aligned}$$

8.7.2.0.2 Atmospheric Losses

This kind of losses derives from the absorption of energy by atmospheric gases. They can assume two different types:

- Atmospheric attenuation.
- Atmospheric absorption.

The major distinguishing factor between them is their origin. Attenuation is weatherrelated, while absorption comes in clear-sky conditions. Likewise, these losses can be due to ionospheric, tropospheric and other local effects. [9]

Ionospheric Effects All radio waves transmitted by satellites to the Earth or vice versa must pass through the ionosphere, the highest layer of the atmosphere, which contains ionized particles, especially due to the action of sun's radiation. Free electrons are distributed in layers and clouds of electrons may be formed, originating what is known as travelling ionospheric disturbances, what provoke signal fluctuations that are only treated as statistical data. The effects are:

- **Polarization rotation**: When a radio wave passes through the ionosphere, it contacts the layers of ionized electrons that move according to the Earth's magnetic field. The direction these electrons move will no longer be parallel to the electric field of the wave and therefore the polarization is shifted, in what is called Faraday rotation (θ_F). ;
- **Scintillation effects**: Differences in the atmospheric refractive index may cause scattering and multipath effect, due to the different directions rays may take through the atmosphere. They are detected as variations in amplitude, phase, polarization and angle of arrival of the radio waves. It is often recommended the introduction of a fade margin so atmospheric scintillation can be a tolerated phenomenon.;
- Absorption
- Variation in the direction of arrival
- Propagation delay
- Dispersion
- Frequency change

These effects decrease usually with the increase of the square of the frequency and most serious ones in satellite communications are the polarization rotation and the scintillation effects, and those are the ones that will be treated in this dissertation. [9]

Tropospheric Effects [9] Troposphere is composed by a miscellany of molecules of different compounds, such as hail, raindrops or other atmospheric gases. Radio waves that pass by troposphere will suffer their effects and will be scattered, depolarized, absorbed and therefore attenuated.
Attenuation: As radio waves cross troposphere, radio frequency energy will be converted into thermal energy and that attenuates signal.
Rain attenuation: Ground stations had been chosen in order that the attenuation caused by rainfall will be very punctual. Also, the fact that there are three ground stations makes really difficult that a satellite can not communicate to the ground in all the orbit period.
Gas absorption: Under normal conditions, only oxygen and water vapour have a significant contribution in absorption. Other atmospheric gases only become a problem in very dry air conditions above 70 GHz. Thereby, losses caused by atmospheric absorption vary with frequency and the collection of data already received allows the elaboration of the graphic that follows:

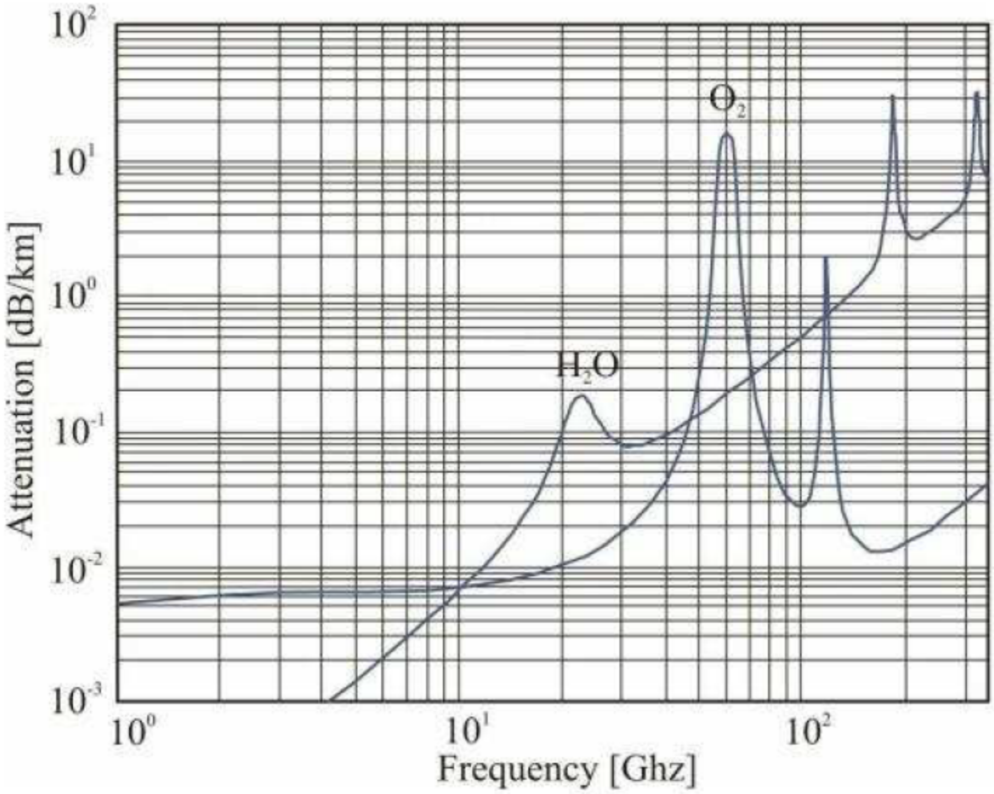


Figure 8.7.2: Specific attenuation for different frequencies [9]

Once these values depend on atmosphere thickness, it becomes necessary to perform all calculations taking into account troposphere's thickest layer (T_{trop}), which has 20 km.
It is also mandatory to refer that this graph represents the absorption for a satellite in the zenith, in other words, for an elevation angle of 90° ($\theta = 90^o$). For lower angles, the atmospheric absorption (L_{abs}) is given by [9]:

$$L_{abs}(dB) = L_{abs|90^o}(dB/km)\ cosec(\theta)\ T_{trop}(km) \tag{8.7.3}$$

For AstreaSAT, 5x10$^{-3}dB/km$ attenuation factor is considered for S band due to the O_2 specific attenuation. On the other hand, 4x10$^{-3}dB/km$ attenuation factor is considered for X band due to the H_2O and to the O_2 specific attenuations. An study of the critical elevation angle will lately be performed.

For AstreaSAT ground station, communication starts at an elevation angle of $\theta = 10^o$ (worst case scenario). Consequently, $cosesc(\theta)$ will go from 5.76 to 1 (best reception case). In that case, we assume:

$$L_{abs} = 2 \cdot 4\text{x}10^{-3} \cdot 5.76 \cdot 20 = \mathbf{0.92dB} \quad \text{X band}$$
$$L_{abs} = 5\text{x}10^{-3} \cdot 5.76 \cdot 20 = \mathbf{0.58dB} \quad \text{S band}$$

Polarization: Satellite communications use linear and circular polarization, but undesirable effects may transform it into an elliptical polarization. Depolarization may occur when an orthogonal component is created due to the passing of the signal through the ionosphere. There are two ways to measure its effect, cross polarization discrimination (XPD) and polarization isolation (I) [9]. To overcome this attenuation problems a circular polarization is the best option. AstreaSAT patch antennas will mitigate this problem, therefore this loses are considered negligible.

Sky noise: Sky noise is a combination of galactic and atmospheric effects, according as both these factors influence the quality of the signal in the reception. Galactic effects decrease with the increase of frequency. They are due to the addition of the cosmic background radiation and the noise temperature of radio stars, galaxies and nebulae. This value is quite low and a good approximation is **3 K**.

AstreaSAT noise temperature A good approximation based on Fig.8.7.3 is that Galaxy noise is 3K for S band and almost 1K for X band. Furthermore, for the previous

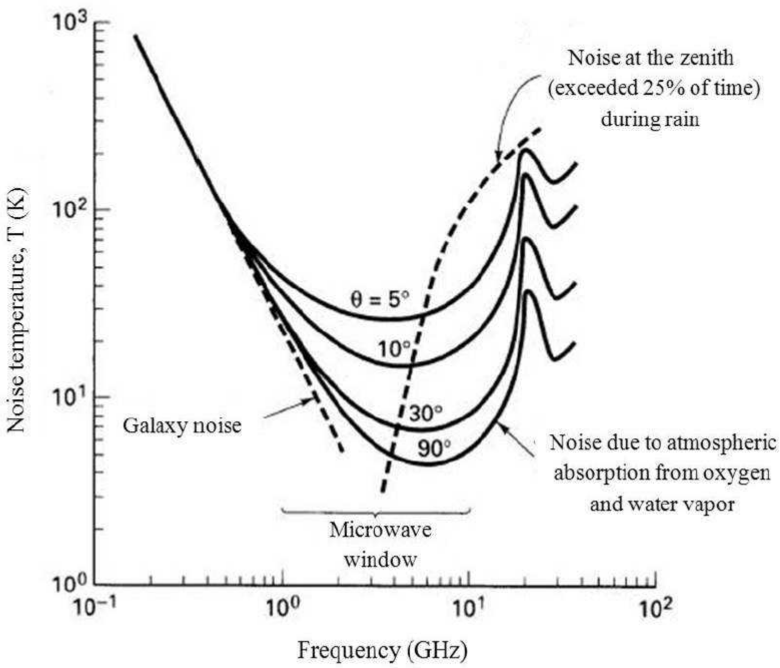


Figure 8.7.3: Galaxy noise influence in noise temperature [9]

worst case scenario stated before $\theta = 10$, noise temperature due to atmospheric absorption is 19K for both bands (S and X).

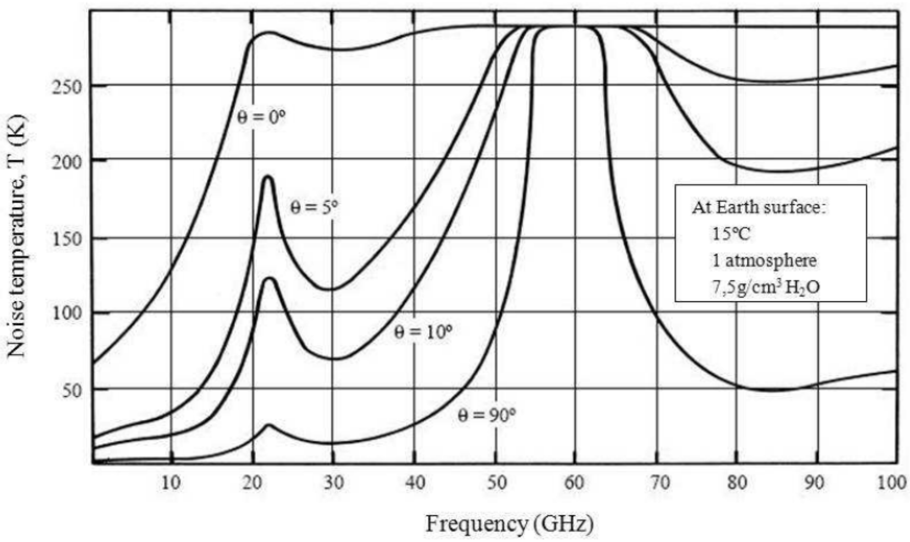


Figure 8.7.4: Noise temperature variation with frequency [9]

Local Effects These effects refer to the proximity of the local ground stations, possible sources that may interfere with the received signal and buildings that may block the signal. If the ground station is on a free external interferences zone, for satellite communications this factor may be negligible.

8.7.2.0.3 Pointing Losses

Ideal reception implies that the value for misalignment losses would be 0 dB which means maximum gain at the ground station is achieved when both the transmitter and the receiver antennas are 100% aligned. Realistically it is virtually impossible to achieve a perfect alignment between the antennas of the ground station and the satellite, especially in the case of CubeSats, due to their fast movement of nearly 8000 ms^{-1}.

Antenna misalignment losses (L_{aml}) are calculated using statistical data, so these values are an approximation based on real data observed in several GS. Ergo, these values are not calculated, but estimated. [9]

Based on a estimation from [**?**] a $L_{aml} = 1dB$ is a good approximation.

8.7.2.0.4 Multipath and Fade Margin

Multipath occurs when waves emitted by the transmitter travel along a different path and interfere destructively with waves travelling on a direct line-of-sight path. This is sometimes referred to as signal fading. This phenomenon occurs because waves travelling along different paths may be completely out of phase when they reach the antenna, thereby cancelling each other.

The amount of extra RF power radiated to overcome this phenomenon is referred to as fade margin. The exact amount of fade margin required depends on the desired reliability of the link, but a good rule-of-thumb is 20dB to 30dB.

8.7.3 Local Losses

8.7.3.0.1 Equipament Losses

The receiving and emitting equipments also introduces some losses to the signal.
Feeder Losses: Feeder losses occur in the several components between the receiving antenna and the receiver device, such as filters, couplers and waveguides. These losses are similar to the ones which occur also in the emission, between the emitting antenna and the output of the high power amplifier (HPA). [9]

8.7.3.0.2 Environment Losses

This item is related to the specific region of the globe where the ground station is placed (equatorial, tropical, polar…). Depending on its latitude, each region has its own characteristics (e.g. temperature, moisture, thickness of atmospheric ice layer…), which may provoke variation in signal reception. [9]

Communications department, had chosen the best locations over the globe, with stable good weather conditions to neglect this fact.

8.7.4 Modulation Technique

Modulation technique is a key consideration. This is the method by which the analogue or digital information is converted to signals at RF frequencies suitable for transmission. Selection of modulation method determines system bandwidth, power efficiency, sensitivity, and complexity. Most of us are familiar with Amplitude Modulation (AM) and Frequency Modulation (FM) because of their widespread use in commercial radio. Phase Modulation is another important technique. It is used in applications such as Global Position System (GPS) receivers and some cellular telephone networks. [10]

For the purposes of link budget analysis, the most important aspect of a given modulation technique is the Signal-to- Noise Ratio (SNR) necessary for a receiver to achieve a specified level of reliability in terms of BER.

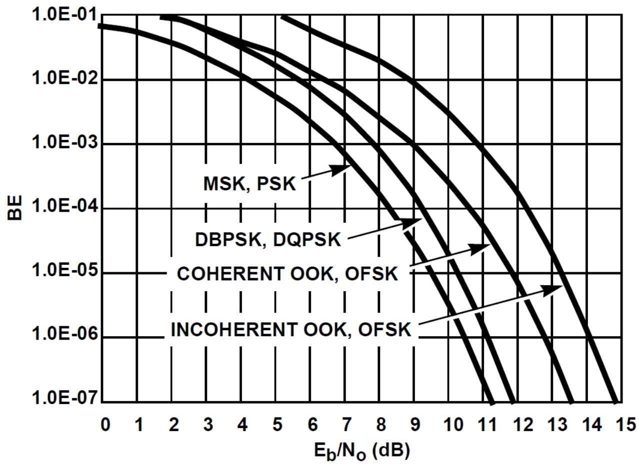


Figure 8.7.5: Probability of bit error for common modulation methods [10]

A graph of E_b/N_o vs BER is shown in Figure 8.7.5. E_b/N_o is a measure of the required energy per bit relative to the noise power. Note that E_b/N_o is independent of the system data rate. In order to convert from E_b/N_o to SNR, the data rate and system bandwidth must be taken into account as shown below:

$$SNR = (E_b/N_o)(R/B_T) \tag{8.7.4}$$

where:

$$\begin{aligned} E_b &= \text{Energy required per bit of information} \\ N_o &= \text{thermal noise in 1Hz of bandwidth} \\ R &= \text{system data rate} \\ B_T &= \text{system bandwidth} \end{aligned}$$

AstreaSAT is equipped with Software Defined Radios, it has the ability to change the modulation methods when its flying, for calculus MSK and PSK modulations will be considered, because of their more restrictive conditions.

8.7.5 System Noise

The system noise temperature (T_S) is the sum of the antenna noise temperature (T_A) and the composite temperature of other components (T_{comp}), according to: [9]

$$T_S = T_A + T_{comp} \tag{8.7.5}$$

T_A may be known if the total attenuation due to rain and gas absorption (A), the temperature of the rain medium (T_m) and the temperature of the cold sky (T_C) are also known. Then, the following expression may be applied:

$$T_A = T_m\ (1 - 10^{-A/10}) + T_C\ 10^{-A/10} \tag{8.7.6}$$

Usually, for clouds it is considered $T_m = 280K$ and for the rain $T_m = 260K$. The sky noise tends to be $T_C = 10K$. Taking into account the values from Fig.8.7.3 and Fig.8.7.2 the following estimation can be made:

$$T_A = 280 \cdot (1 - 10^{-(5x10^{-3})/10}) + 22 \cdot 10^{-(5x10^{-3})/10} = \mathbf{22.29K} \quad \text{S band}$$
$$T_A = 280 \cdot (1 - 10^{-2\cdot(4x10^{-3})/10}) + 20 \cdot 10^{-2\cdot(4x10^{-3})/10} = \mathbf{20.48K} \quad \text{X band}$$

According to [9] a good components temperature approximation for a typical ground station is $T_{comp} = 65.5K$.

AstreaSAT system temperature will be considered as $T_S = 22.29 + 65.5 = \mathbf{87.79K}$ for S band and $T_S = 20.48 + 65.5 = \mathbf{85.98K}$ for X band. Since both frequencies are part of the microwave spectrum, we see that system temperatures are pretty much the same.

Channel Noise All objects which have heat emit RF energy in the form of random (Gaussian) noise. The amount of radiation emitted can be calculated by [10]:

$$N = kTB \tag{8.7.7}$$

where:

$$N = \text{noise power (watts)}$$
$$k = \text{Boltzman's constant}(1.38x10^{-23}J/K)$$
$$T = \text{system temperature, usually assumed to be 290K}$$
$$B = \text{channel bandwidth (Hz)}$$

This is the lowest possible noise level for a system with a given physical temperature. For most applications, temperature is typically assumed to be room temperature (290K). Equations 8.7.1 and 8.7.7 demonstrate that RF power and bandwidth can be traded off to achieve a given performance level (as defined by BER). [10]

8.7.6 Link Budget Calculation

Methodology From the expected requirements fixed on the Project Charter, general radio systems parameters will computed, in order to have a reference to look for the best communications system on board the Astrea satellites. As background, general losses parameters had been calculated on previous sections.

The most important concern on AstreaSAT link Budget is how far every satellite can emit on the desired frequencies. This is a key factor to know the utility of the modules selected. At least, Project Charter communication requirements must be accomplish.

$EIRP = P_T - L_T - G_T$
FRIIS EQUATION + GRAPH RANGE
SENSITIVITY CALCULUS A PARTIR DE LA DE CAPACITAT + NOISE ADJUDICANT BANDWITH

8.8 Budget

System	Cost/unit (€)	Total cost (€)	N. of units
STRUCTURE AND MECHANICS			
Structure	3900	3900	1
Thermal protection	1000	1000	1
Total		4900	
ELECTRIC POWER SYSTEM			
Solar arrays	17000	68000	4
Batteries	6300	12600	2
Power management	16000	16000	1
Total		96600	
PAYLOAD			
Patch antenna	18000 1st unit 7000 others	67000	8
Transceiver inter-satellite	8545	25635	3
Transceiver space to ground	5500	5500	1
Data handling system	5000	5000	1
Antenna Deployable	3000	3000	1
Variable expenses	4000	4000	1
Total		110135	
AOCDS			
Thruster	1350	50000	1
ADACS	280	15000	1
Total		65000	
TOTAL		276635	
TOTAL ESTIMATION		297000	
+Fixed cost	(includes all CubeSats)	150000	

The difference between the total cost and the total estimation is due to the fact that every satellite has to go through a process to be ready for operation. This is, the CubeSat has to be assembled and has to be tested as well to ensure that all the systems are working properly. Thus, an estimation of the costs related with this operation has to be made.

The fixed cost for assembling the satellites will be 150000€(cost of renting the building, the electricity, ...) and an additional cost 20000€$/unit$, which will include the wages of the people assembling and testing the satellite and also other variable costs that may appear in the process, is added to every satellite. Furthermore, this extra 20000€includes the costs of transport to launch site.

Several options have been studied for assembling and testing the satellite, and the option chosen is *OpenCosmos*. Astrea is committed to encourage the growth of the local economy and we are sure that *OpenCosmos* would be a perfect partner for the mission. They provide companies and individuals with simple and affordable access to space offering integration and testing services.

8.9 Astrea satellite Final Configuration

System	Weight/unit (g)	Sizes (mm)	N. of units
STRUCTURE AND MECHANICS			
Structure	304.3	100 x 100 x 300	1
Thermal protection	38	Covers all	1
Total	342.3		
ELECTRIC POWER SYSTEM			
Solar arrays	175	98 x 83 x 8.50	4
Batteries	155	90 x 63 x 12.02	2
Power management	126	92.0 x 88.9 x 20.5	1
Total	1136		
PAYLOAD			
Patch antenna	30	90 x 90 x4.35	8
Transceiver inter-satellite	16.4	65 x 40 x 6.5	3
Transceiver space to ground	101.5	86 x 86 x 45	1
Data handling system	28.3	65 x 40 x 6.5	1
Antenna Deployable	83	100 x 83 x 6.5	1
Variable	150	-	1
Total	652		
AOCDS			
Thruster	1350	90 x 90 x 95	1
ADACS	506	90 x 90 x 58	1
Total	1856		
TOTAL ESTIMATION	**3986.3**		

Part VI

Financial and Other Considerations

"The first rule is to never lose money. The second rule is to never forget the first one."

Warren Buffett

Over this chapter, the **financial study** is going to be performed. The costs and the profits will be analyzed, and some important figures will be acquired.

Moreover, some other important considerations, such as social and security issues or environmental impact will be studied too.

Chapter 9

Financial Study

The different departments have estimated the main costs of the project. It is high time to start performing a deep analysis on the economical solvency of the project. The analysis carried on will be of 10 years.

Up to this point, it is important to determine how this product will be sold, so as to quantify the benefits of the project and be able to determine some figures such as the Pay Back Time or the Net Present value, and be able to make some conclusions.

9.1 Selling the product

The aim of the project is to be able to sell to the customers the chance of both sending and receiving data from satellites. Therefore, it seems logic that the price of the product has to be somehow related to the amount of data passed on. Then, there will be a price for every Mbit, either sent or received.

From the Communications Department, there is a limitation of 3 Ground Stations operating, and each one can carry up to 25 Mbits/second. Accepting that those Ground Stations will fully operating the whole year, and calculating the amount of seconds that there are in a normal year:

$$365 \cdot 24 \cdot 60 \cdot 60 = 31536000s \tag{9.1.1}$$

It can be easily calculated the amount of Mbits that Astrea Constellation is able to either send or receive:

$$31536000 \cdot 75 = 2365200000Mbits \tag{9.1.2}$$

This means that no more than 2365200000 Mbits can be sold. This is the maximum supply.

But how can the demand be estimated? There is a need to make assumptions.

9.1.1 Estimation of demand

9.1.1.1 Universities

Firstly, it has been thought that the service offered has great academic interests. In fact, any student could build a satellite with a certain payload, send it to space and then receive data from the satellite at any time thanks to Astrea constellation.

In order to study the possible demand of Mbits, an estimation of the possible universities that would want to use the services has been done. Fortunately, the list of universities that offer studies in the aerospace field goes back a total of 400 schools approximately. Nevertheless, it is highly improbable that all those colleges become clients because not all universities have the same sources or interests. Therefore, the following list presents the number of existing colleges having an aerospace degree in each continent.

By analyzing this information, it can be determined that the continents with countries with higher PIB have more colleges interested in the space field. It is noticed that Asia is the continent with more colleges because, even if it is mostly poor, it is so big that it has rich countries such as Japan, Korea or China and the United Arab Emirates. Moreover,

Continent	Number of Universities
Europe	124
Asia	138
North America	97
South America	18
Australia	8
Africa	12

Table 9.1.1: Table. List of Universities with Aerospace Degrees

Europe and North America are not so extensive but have a higher aerospace culture and interest.

On the basis the service is affordable for many prestigious colleges and it permits to provide their students with the chance to improve their knowledge by doing their own experiments, it has been estimated that about 150 universities will end up contracting Astrea's service in the next years. If we assume that each university would be interested in sending or receiving a total of 630720 Mbits anually, therefore the number of Mbits for universities, anually, will be of 94608000 Mbits.

9.1.1.2 Particular customers

Another extremely important sector of clients are the private ones. It is harder to make an assumption on the number of Mbits consumed by this sector. Nevertheless, some figures are needed in order to perform a good feasibility study.

According to the Union of Concerned Scientists of the United States of America, right now there are about 1500 satellites orbiting around the Earth. But every day space technology is more affordable and feasible, which leads to think that in the next years a good figure of satellites would be of roughly 2000. Nonetheless, around 40% of those missions would benefit of a faster communication with their satellites. As Astrea provides a very competitive price, it seems reasonable to think that a good percentage of those satellites would be interested. In order to be conservative, a 50% of those would be potencial clients. This means that 400 full operating satellites would use Astrea, and assuming also than the average amount of data that those satellites would either send or receive anually is of 946080 Mbits, the number of Mbits for particular clients, anually, will be of 378432000 Mbits.

It can be checked that the sum of the amounts of Mbits for universities and for particular clients is lower than the maximum amount of Mbits due to the 3 Ground Stations (as has been stated before), this is, 2365200000 Mbits. In particular, it turns out to be a fifth of this quantity.

9.1.1.3 Demand

Taking into account both the universities and the particular clients, and making a conservative assumption, the estimation of the demand, in Mbits, is of a fifth of the maximum capacity of Astrea, this is, 473040000 Mbits anually. Also, in order to simulate the unceirtainty of the company during the first years (as years pass, the company gets reputation and therefore its amount of clients also enlarges, a percentage is applied during the first years. This means that first year only a 75% of the potential customers exposed before will be achieved, the second year a 80%, and so on, until the sixth year, in which a 100% is achieved.

9.1.2 Pricing the service

The determination of the price is made upon the feasibility study, in order to get a reasonable Pay Back Time and benefit. Nevertheless, it is a fact that the fare of Astrea service must fulfill a condition: it has to be competitive.

Comparing with some others constellations that offer a similar service, in order to provide a competitive fare, it seems reasonable a price per Mbit of no more than 0.5 €per Mbit, as an upper tape.

9.2 Economic Feasibility Report

In order to perform the analysis on the economical solvency of the project, following there is a table which contains the main costs of the project, as well as the numerical operations that allow to calculate some important financial parameters, such as the Net Present Value (NPV), the Internal Rate of Retorn (IRR), the Simple Pay Back Time (PBT), the Updated Pay Back Time (UPBT) and the Break Even Point (BEP). From this data, some conclusions will be drawn.

Firstly, though, there is need to take into account some costs that are not included in the other departments, and which are key to analyzing the costs and benefits.

9.2.1 Previous costs

9.2.1.1 Engineering hours

The engineering hours, which were specified in the Project Charter, are again synthesized in the following table:

Engineering hours budget	Hours	Labor cost (€)
MANAGEMENT		
Meetings documentation		
Meetings	340	6800
Meetings preparation		
Agendas	10	200
Minutes	10	200
Task Tracking and scheduling		
Project Charter	170	3400
Team tasks monitoring	20	400
WBS and Gantt update	10	200
SATELLITE DEVELOPMENT		
Spacecraft subsystems	180	3600
Payload		
Antenna	40	800
PHDS	50	1000
ORBITAL DESIGN		
Constellation geometry	220	4400
Orbit parameters		
General parameters	120	2400

Engineering hours budget	Hours	Labor cost (€)
Drift	100	2000
Legislation	50	1000
LAUNCH SYSTEMS		
Vehicle	60	1200
Satellite deployer	10	200
Replacement strategy	100	2000
OPERATION		
Communication protocol	100	2000
Ground station	80	1600
End of life strategy	80	1600
FINANCIAL PLAN		
Costs		
Fix		
Maintenance and cost analysis	10	200
Insurance cost analysis	15	300
Administration cost analysis	15	300
Taxes cost analysis	25	500
Variable		
Manufacturing cost report	10	200
Launching cost report	10	200
Income		
Price analysis	25	500
Revenue forecast	25	500
Economic feasibility report	40	800
Marketing Plan	20	400
PROJECT EXHIBITION		
Constellation simulation	30	600
TOTAL	1975	39500

9.2.1.2 Administrarion costs

It has to be taken in account that administrating the company will require resources and manpower. To budged this costs there have been considered the following factors:

- **Manpower**. It is estimate that it will be needed 6 people working at full time. 3 for the administration of the stations, 2 more for the clients, and an other one for the purchases of new satellites and contracting launchings. The annual salary of each worker would be of 24000 €, which make a total of 144,000€

- **Financial costs**. The treasury of the company will require a bank, with its associated costs. This is estimated in 100000€per year.
- **Local** The place where the administrators will work would cost annually around 10000€
- **Supplies** The water, electricity, internet and telephone would cost 5000€

This result in 259.000€/year

9.2.1.3 Taxes

The headquarters of effective management is located in Spanish territory, so it is crucial to take into consideration the corresponding taxes. It is known that any entity that directs and controls all of its activities of effective management in Spanish territory is considered as resident. Consequently by having the residence there they are subjected to the Spanish Corporation Tax. It has to be known that this tax is an annual and proportional tribute belonging to the Spanish tax system that taxes the income of the companies.

Moreover, by following the Article 29 of the Law 27/2014 on the CT it is possible to determine the tax rate that is going to be paid. As a result, for any company located in Catalonia the annual fee to be paid is 25% of annual profits. However, for being a company of new creation, the first two years the tax will be 15% of profits only. It is important to notice that this kind of tax will be paid when the taxpayer begins to obtain benefit, in other words since the enterprise starts to be profitable.

9.2.1.4 Insurance

The responsability for possible damages or errors is an important aspect to consider. In a satellite, there are different stages that need an insurance because they have possabilities to fail and cause high damages.

From an international point of view, from 1972 there is a treaty, *The Space Liability Convention*, which says that the states must assume their responsability of their space objects launched in their territories. This liability was created to provide compensation to parties injured by space activities. This treaty was ratified in Jaunary 2013 by 89 states and signed but not ratified by 22 states. [?]

As a private company, Astrea should provide a compensation to third people if they are injured by one of the CubeSats. Furthermore, how has been explained in *Social and security considerations*, there are some little risks in differents stages of a Cubesat (launch and in-orbit) and it might be advantageous to have economic security contracting a insurance.

Currently, there are a lot of insurance companies that provide their services to space companies and specifically to satellites companies. After a market study, there are two companies to consider, *SpaceCo*, a subsidiary of *Allianz* company and *Marsh*. Both provide the main services that we need: satellite launch and in-orbit insurance and satellite third party iability insurance.

Finally, *SpaceCo* has been chosen as Astrea insurer company, due to it is considered one of the best insurer for space companies and it has more experiences than others.

This insurer provide a great coverage, in which highlights:

- Launch and commissioning – cover for the launch systems and commissioning equipment.
- In-orbit – operational life insurance for the space satellite.
- In-orbit incentives – cover for the manufacturer's obligation to the client in the event of malfunction or non-performance.
- Liability – cover for third party liability during a launch or in-orbit activities.
- Captive services – assisting cover for companies that self-insure space risks. [?]

The cost of the insurance is around a 20% of cubesat value, which is 297000 €, to pay in the 5 life-years of each. Then, the total cost of the constellation insurance would be:

N. of CubeSats	189
Cost per Cubesat	59400 €
Total cost in 5 years	11226600 €
Cost per year	2245320 €

9.2.2 Economic feasibility study

Finally, the mentioned financial table can be made. The costs are the ones taken from every department, as well as the costs just explained.

TIME	**Year 0**	**Year 1**	**Year 2**	**Year 3**	**Year 4**	**Year 5**	**Year 6**	**Year 7**	**Year 8**	**Year 9**	**Year 10**	**Year 11**	**Year 12**
INVESTMENT	-4,07												
INCOME													
Percentage (learning curve)		0,75	0,80	0,85	0,90	0,95	1,00	1,00	1,00	1,00	1,00	1,00	1,00
Number of Mbits hired		221.737.500,00	236.520.000,00	251.302.500,00	266.085.000,00	280.867.500,00	295.650.000,00	295.650.000,00	295.650.000,00	295.650.000,00	295.650.000,00	295.650.000,00	295.650.000,00
Gain (M euros)		22,17	23,65	25,13	26,61	28,09	29,57	29,57	29,57	29,57	29,57	29,57	29,57
Total	**0,00**	**22,17**	**23,65**	**25,13**	**26,61**	**28,09**	**29,57**	**29,57**	**29,57**	**29,57**	**29,57**	**29,57**	**29,57**
COSTS													
n planes/year	9					9					9	0	0
Satellites/year	189	0	0	0	0	189	0	0	0	0	189	0	0
Engineering hours	-0,0395												
Administration		-0,259	-0,259	-0,259	-0,259	-0,259	-0,259	-0,259	-0,259	-0,259	-0,259	-0,259	-0,259
Insurance		-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532
Web hosting, maint. and promotion	-0,005	-0,005	-0,005	-0,005	-0,005	-0,005	-0,005	-0,005	-0,005	-0,005	-0,005	-0,005	-0,005
Launching													
Planes	-48,256	0,000	0,000	0,000	0,000	-48,256	0,000	0,000	0,000	0,000	-48,256	0,000	0,000
Satellites	-3,024	0,000	0,000	0,000	0,000	-3,024	0,000	0,000	0,000	0,000	-3,024	0,000	0,000
System													
Assembly (individual)	-3,78	0,00	0,00	0,00	-3,78	0,00	0,00	0,00	0,00	-3,78	0,00	0,00	0,00
Assembly (constellation)	-0,15	0,00	0,00	0,00	-0,15	0,00	0,00	0,00	0,00	-0,15	0,00	-0,15	-0,15
Structure	-0,737	0,000	0,000	0,000	-0,74	0,000	0,000	0,000	0,000	-0,74	0,000	0,000	0,000
Thermal protection	-0,189	0,000	0,000	0,000	-0,19	0,000	0,000	0,000	0,000	-0,19	0,000	0,000	0,000
Electric power system													
Solar arrays	-12,852	0,000	0,000	0,000	-12,85	0,000	0,000	0,000	0,000	-12,85	0,000	0,000	0,000
Batteries	-2,381	0,000	0,000	0,000	-2,38	0,000	0,000	0,000	0,000	-2,38	0,000	0,000	0,000
Power management	-3,024	0,000	0,000	0,000	-3,02	0,000	0,000	0,000	0,000	-3,02	0,000	0,000	0,000
Payload													
Patch antenna	-10,595	-0,011	-0,011	-0,011	-10,60	0,000	-0,011	-0,011	-0,011	-10,60	0,000	-0,011	-0,011
Antenna deployment	-0,567	0,000	0,000	0,000	-0,57	0,000	0,000	0,000	0,000	-0,57	0,000	0,000	0,000
Transciever inter-satellite	-4,845	0,000	0,000	0,000	-4,85	0,000	0,000	0,000	0,000	-4,85	0,000	0,000	0,000
Transciever space to ground	-1,040	0,000	0,000	0,000	-1,04	0,000	0,000	0,000	0,000	-1,04	0,000	0,000	0,000
Data handling system	-0,945	0,000	0,000	0,000	-0,95	0,000	0,000	0,000	0,000	-0,95	0,000	0,000	0,000
Variable expenses	-0,756	0,000	0,000	0,000	-0,76	0,000	0,000	0,000	0,000	-0,76	0,000	0,000	0,000
AOCDS													
Thruster	-9,450	0,000	0,000	0,000	-9,45	-9,450	0,000	0,000	0,000	-9,45	0,000	0,000	0,000
CubeSpace ACDS	-2,835	0,000	0,000	0,000	-2,84	-2,835	0,000	0,000	0,000	-2,84	0,000	0,000	0,000
Communications													
Maintenance GS Canada		-0,011	-0,011	-0,011	-0,011	-0,011	-0,011	-0,011	-0,011	-0,011	-0,011	-0,011	-0,011
Maintenance GS Scotland (UK)		-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015
Maintenance GS Malvines		-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015	-0,015
Salaries GS Canada		-0,382	-0,382	-0,382	-0,382	-0,382	-0,382	-0,382	-0,382	-0,382	-0,382	-0,382	-0,382
Salaries GS Scotland (UK)		-0,226	-0,226	-0,226	-0,226	-0,226	-0,226	-0,226	-0,226	-0,226	-0,226	-0,226	-0,226
Salaries GS Malvines		-0,082	-0,082	-0,082	-0,082	-0,082	-0,082	-0,082	-0,082	-0,082	-0,082	-0,082	-0,082
Salaries MCC		-0,430	-0,430	-0,430	-0,430	-0,430	-0,430	-0,430	-0,430	-0,430	-0,430	-0,430	-0,430
Licenses		-0,010	-0,010	-0,010	-0,010	-0,010	-0,010	-0,010	-0,010	-0,010	-0,010	-0,010	-0,010
TOTAL	**-109,54**	**18,48**	**19,96**	**21,44**	**-31,22**	**-39,16**	**25,87**	**25,87**	**25,87**	**-28,26**	**-25,39**	**25,72**	**25,72**
CASH FLOW	**-113,61**	**40,66**	**43,61**	**46,57**	**-4,61**	**-11,07**	**55,44**	**55,44**	**55,44**	**1,30**	**4,17**	**55,29**	**55,29**
DISC CF	**-113,61**	**38,36**	**38,82**	**39,10**	**-3,65**	**-8,27**	**39,08**	**36,87**	**34,78**	**0,77**	**2,33**	**29,13**	**27,48**
CUM CF	**-113,61**	**-72,95**	**-29,34**	**17,23**	**12,62**	**1,55**	**56,99**	**112,43**	**167,87**	**169,18**	**173,35**	**228,63**	**283,92**
DIS CUM CF	**-113,61**	**-75,25**	**-36,44**	**2,67**	**-0,99**	**-9,26**	**29,82**	**66,69**	**101,48**	**102,25**	**104,58**	**133,70**	**161,18**

Table 9.2.3: Feasibility Study

As it has been said, upon this financial table, in order to get a good feasibility situation, the pricing of the service is decided to be of 0.1 €per Mbit.

9.3 Conclusions of the financial study

As a result of a few iterations of this table, changing some parameters, it has been found that:

9.3.1 Pay Back Time (PBT)

From the shown table, it can be seen that between years 3 and 4, the Cumulative Cash Flow goes from a negative value to a positive one. Therefore, the Pay Back Time is between those two years. This gives a rough approximation of when will the investment be recouped. To be more precise about it, it can be linearly interpolated:

$$\frac{25.42-(-7.08)}{4-3}=\frac{25.42-0}{4-x} \tag{9.3.1}$$

Solving for x, the result is of a PBT of 3.22 years.

Nevertheless, it can also be seen that in year 5, the Cumulative Cash Flow again becomes negative, due to the increase of costs because of the re-launching of the satellites. Thus, a second Pay Back Time could be found, between years 6 and 7. Interpolating again:

$$\frac{18.08-(-13.70)}{7-6}=\frac{18.08-0}{7-x} \tag{9.3.2}$$

Solving for x again, the result is of a PBT2 of 6.43 years. However, the important one is the first PBT, since it is the point from which there starts to be benefit.

In year 10, though, the profits are high enough to cover the increase of cost due to third launching, which would make that Cumulative Cash Flow does not become negative.

The value of the first PBT found seems reasonably acceptable, taking into account that this project requires a great budget, as all space projects do, due to its own nature.

9.3.2 Updated Pay Back Time (UPBT)

Taking into account now the discount rate (6% anual), there is the Discounted Cumulative Cash Flow. It can be seen that between years 3 and 4, this value goes from a negative value to a positive one. Thus, the Updated Pay Back Time is between those two years. It can be linearly interpolated to gain some precision:

$$\frac{7.25 - (-18.49)}{4 - 3} = \frac{7.25 - 0}{4 - x} \tag{9.3.3}$$

Solving for x, the result is of a UPBT of 3.72 years.

Again, it can be seen that in year 5, the Discounted Cumulative Cash Flow again becomes negative, due to the increase of costs of the re-launching of the satellites, which allows to calculate a second Updated Pay Back Time, between years 7 and 8. Interpolating again:

$$\frac{17.75 - (-2.19)}{8 - 7} = \frac{17.75 - 0}{8 - x} \tag{9.3.4}$$

Solving for x, the result is a UPBT2 of 7.11 years.

Now, in contrast to the Pay Back Time, there will be a third Updated Pay Back Time. When taking into account the discount rate, the benefits in year 10 do not cover the increase in cost of the third re-launching, forcing Discounted Cumulative Cash Flow to be negative again, and a third Updated Pay Back Time might be found. However, this third date can not be determined with this study, since the reach of this feasibility exercise is performed for just the first 10 years.

When analyzing the NPV of the feasibility study, a graphic with those phenomenon will be shown.

Again, that first value of UPBT seems reasonably acceptable, because of the nature of the project, the space sector, a very demanding and expensive one.

9.3.3 Break Even Point (BEP)

The Break Even Point is the point at which total cost and total revenue are equal, there is no net loss or gain. This figure represents the sales amount (quantity) required to cover total costs, consisting of both fixed and variable costs to the company. At this point, the total profit is zero.

In Astrea's case, the Break Even Point is the number of Mbits sold the first year so that the Cash Flow of that year is just 0 (or approximately).

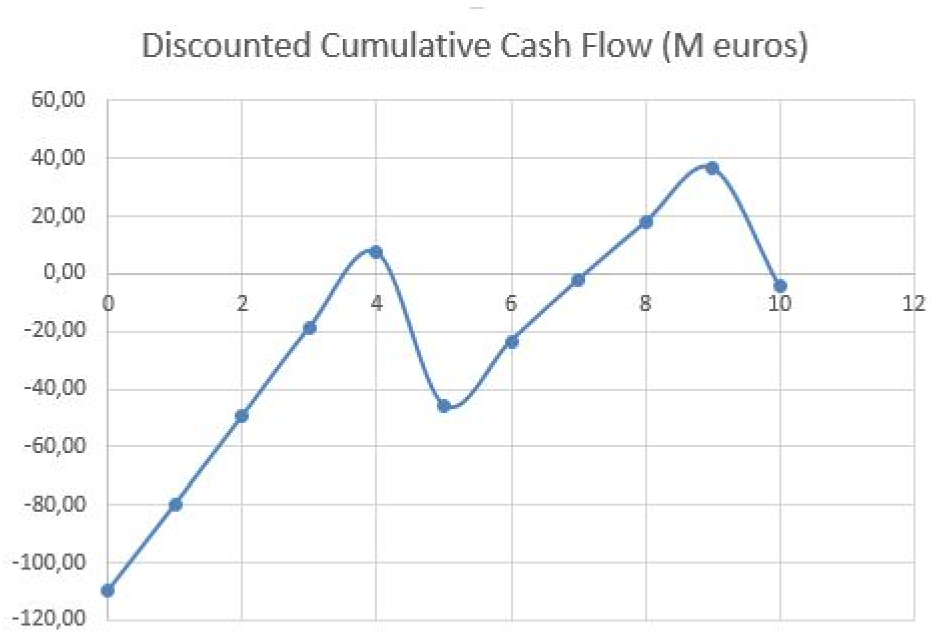
By changing manually the parameter "Number of Mbits hired" of first year, it is found that the Break Even Point is of 36907600 Mbits (with this value, the Cash Flow is approximately 0). This means that under no account there can be less Mbits hired, otherwise, the Cash Flow would be negative and the Cumulative Cash Flow, negative since first year is fully invest, would never reach a positive value, generating losses.

From the assumptions of demand already explained, it can be seen that having a greater demand than the BEP is very likely to happen.

9.3.4 Net Present Value (NPV)

The Net Present Value is the difference between the present value of cash inflows and the present value of cash outflows over a period of time (in this case, of 10 years). It is useful to analyze the profitability of a project. A positive NPV indicates that the project earnings generated by a investment exceeds the costs. The Internal Rate of Return must also be taken into account when calculating the NPV. In this project, a IRR of 6% has been considered.

From the table, it can be immediately seen that the Net Present Value (for a period of time of 10 years) is of -4.55M€. It is clearly not positive, which theorically would say that the project is not feasible within the 10 years considered. Nevertheless, as it has been explained in the pay back times, this is due to the fact that in years 0, 5, 10, 15... a re-launching of the whole constellation is performed. Therefore, just in year 9 the Discounted Cumulative Cash Flow is of 36.57M€, which means that if the period of time of the study would have been of 9 years, the NPV would be clearly positive. What is trying to be explained is that the NPV of the study is negative just because the last year coincides with a year of re-launching. Moreover, compared to the Discounted Cumulative Cash Flow of year 5, it is clearly much bigger. For sure, in year 11 it will be positive, and in year 15, of re-launching again, there won't be a Discounted Cumulative Cash Flow negative. This phenomenom is shown in the following graphic, that shows the Discounted Cumulative Cash Flow of the first 10 years so as to see the tendency of it:



In that graphic, it can be seen what has just been explained. In year 15 there will be a new decrease, but this time, its lowest point (locally) will be positive, and from that point, there will always be a positive balance.

9.3.5 Internal Rate of Return (IRR)

The internal rate of retorn is the interest rate at which the Net Present Value of all the cashflows is equal to zero. This is used to evaluate the attractiveness of a project. If the Internal Rate of Return of a project exceeds a company's required rate of return, the project is desirable, and if on the other hand the IRR falls below the required rate of return, the project should be rejected.

For the study carried on, the discount rate has been a 6% anual. Because of what has been said in the NPV, since the NPV is negative, the IRR will be a smaller quantity. According to the theory, the project should be rejected. But once again, because of the re-launching of the tenth year, it is not a good indicative figure. It should have been a better idea to perform a 9 or 11 years analysis, but it was also interesting to do a economical study of the first two completes lifes of the satellites.

Changing manually the parameter d of the table, it is found that for a discount ratio of 3.84%, the NPV is zero, which means that this is the IRR. It is smaller than the actual discount ratio, just as was predicted and explained.

Chapter 10

Marketing Plan

10.1 Executive Summary

Astrea is the result of an enormous amount work and effort from its 17 co-founders and its name needs to be spread all over the world in order to start selling its services. In order to do that it is important to define the target customers to whom the service offered is going to be sold. Being the latter clear, it's essential to point out what does Astrea offer that makes it stand out from the rest of companies in the sector, that is, making an assessment of the strong points of the company. Moreover, it is necessary to establish the price at which the service is going to be sold to the customers and defining the position of the company among its competitors in the sector.

Of course none of the above would make sense without defining a distribution plan in which the way customers buy from us is defined. In addition to that, the marketing materials used also have to be defined along with the online marketing strategy.

A conversion strategy has to be defined too, that is, defining a way to turn prospective customers into paying ones. Finally, possible partnerships or future partnership plans will be assessed.

10.2 Target Customers

One of the most important items when it comes to selling a product or service is to whom it may be of interest. Since the service sold is essentially a communication bridge between satellite-to-satellite, Earth-to-satellite or Earth-to-Earth, it is well obvious that the average customer is not going to be an average consumer.

Instead the service offered is projected towards public or private institutions such as aerospace universities who would like to execute experiments which require a reliable communication between their own spacecraft and their ground stations. Also towards start-up enterprises who would like to enter into the aerospace industry and need Astrea's infrastructure to accomplish their own projects.

In addition to all of the above, the service is also targeted towards space agencies who plan on doing pilot missions with which Astrea could help with. Also aerospace enterprises who nonetheless would like to test their technologies and need real time feedback from them. Finally another targeted sector would be the communications enterprises who would like to acquire real time information from Earth's surface or outer space.

10.3 Unique Selling Proposition

The USP is, as the title appoints, what Astrea has to offer that sets it apart from other companies in its sector. Everyone in Astrea knows what the company is capable of and what it can offer and this is no more and no less than:

- Global signal coverage: Astrea's constellation covers every single spot on Earth's surface. This means that every ground station will have full-time signal coverage.
- Ground station support: Astrea offers ground stations to its customers. For advanced users, custom ground stations are also available.
- High reliability: Astrea's constellation is robust. Therefore, reliability is guaranteed.
- Cheapest price on sector: Astrea brings global communication to customers at the lowest and most affordable price.

10.4 Pricing & Positioning Strategy

The communication service Astrea offers is set to a price of 0.1€/Mb. Since there are no other companies offering the same kind of service it is not possible to make a comparison as of now.

10.5 Distribution Plan

Since what Astrea offers is not a conventional service, people will not be able to purchase it directly. Instead, we use our website to get people to know what Astrea does as well as a way for our customers to get in touch with us. When a customer contacts us we provide them with all the necessary information on how to properly use our systems. Once the contract is made they can start using our communication systems right away. The payment is done monthly much like a regular mobile carrier. Customers will get their invoices with their total data consumption and price.

10.6 Marketing Materials

The marketing materials we count on are:

- A website: http://astrea.upcprogram.space/
- An informative and encouraging video.
- Brochures.
- A poster.

10.7 Online Marketing Strategy

Given the fact that our distribution plan is executed in an essentially online manner, it makes sense to elaborate an Online Marketing Strategy. The key components to our online marketing strategy are:

1. Keyword Strategy: it is important to identify the keywords to optimize our website for. In our case the keywords would be: "Astrea", "constellation", "reliability", "CubeSat" and "communication".
2. Search Engine Optimization: document updates will be made to the website in order to appear more prominently in online search engines.
3. Social Media Strategy: nowadays it is crucial to be in the social media. The world is permanently connected through the social media and it can be one of most powerful ways to show off what we've produced. Therefore, Astrea will have its own Twitter, Instagram and Facebook accounts.

10.8 Conversion Strategy

The technique we use to turn prospective customers into actual paying customers will be showing testimonials from actual customers who were satisfied with our service in our website. In addition to that, we will post in our website every successful project we provide service to. This will show the reliability of the service to the insecure customers and hopefully turn them into actual customers

10.9 Joint Ventures & Partnerships

Right at its beginning Astrea does not count on any Joint Ventures nor Partnerships with other enterprises. Nevertheless, Astrea is open to future partnerships with businesses who would like to work in collaboration with us.

Chapter 11

Environmental Impact Study

11.1 Introduction

This chapter pretends to assess the environmental consequences (positive and negative) of developing the project. The target of this study is to identify, predict, evaluate and mitigate the biophysical and social negative effects that the project could generate during the execution of it.

11.2 Ground Stations

At first sight the Ground Stations do not represent any environmental problem. The main factor that has to be taken into account is the placement of the stations. They have to be located in a place where they do not interfere with the ecosystem. The placement of the stations has to be adequate with the environmental legislation of the countries.

11.3 Satellites

For analysing the impact of the satellites it has to be studied the possible environmental impact during the fabrication and during the orbital life.

Since the fabrication of the satellites is externalized to other companies, the responsibility of the environmental consequences derived of this manufacturing is over these companies. For commercializing these products they must pass all the controls required.

During the orbital performance of the satellites, it has to be taken into account whether or not they would become orbital waste. The satellites are designed to burn out in the atmosphere at the end of their useful life. This burnt should not leave any solid residue that could precipice over the surface. The deorbit would be forced and controlled by the propulsion system of the satellite. In the case that this system fails, given that they will orbit in a LEO, they will be deorbited and burnt out naturally in a period around 5 years.

11.4 Launch system

The most critical part of the entire process, in environmental terms, is the launch of the satellites. For this reason the main relevance in this report is given to the spacecraft that will put the satellites in orbit, the Electron rocket of Rocket-Lab.

The company operate in New Zeeland, and for doing it, the Ministry for the Environment make an accurate study of the environmental impact of the Electron launching. The entire document can be seen at [**?**].

In this document are analysed the critical components of the spacecraft:

- **Structure**. The primary structural material is carbon fibre reinforced polymer. The carbon filaments are chemically inert and do not react to seawater.
- **Propellants**. Liquid oxygen and kerosene (RP-1 analogue) propellants are used on both the first and second stages of the launch vehicle. Liquid oxygen, if released to the atmosphere, rapidly boils and returns to the atmosphere as gaseous oxygen. RP-1 kerosene is a highly refined grade of hydrocarbon with low density, a thin surface film and rapid evaporation.
- **Pneumatics**.All inflight pneumatic systems use stored pressurised cold gases to provide tank pressurisation, cold-gas manoeuvring thrust in space, and for stage separation mechanisms. All gases are non-toxic.
- **Engines**. The launch vehicle uses nine engines for stage 1 and a single engine for stage 2. The engines are constructed of inconel, an inert high performance, corrosion resistant nickel alloy. At stage 1 separation, the thrust section is likely to separate from the stage, return to Earth's surface and land in the Exclusive Economical Zone.
- **Batteries**. The first stage batteries are highly likely to burn-up before returning to Earth's surface. The stage 2 batteries will entirely burn-up downrange, with only the first battery potentially landing in the EEZ. The batteries are lithium-based, and contain no lead, acid, mercury, cadmium, or other toxic heavy metals.

The document also evaluates the following possible risks:

- **Risk of toxic effects**. The toxic effects of the materials comprising stage 1, the fairings and the two stage 2 LithiumIon batteries were assessed as low at all levels of launch activity.
- **Risk of ingestion of materials and provision of floating shelter**. Floating jettisoned materials as shelter for pelagic organisms and the ingestion of jettisoned

materials were both evaluated as having low ecological risk at all levels of launch activity.

- **Environmental effect of the displacement of fishing activities.** For the demersal fish and mobile invertebrate community, marine mammals and seabirds, the effects of fishing displacement would be low because these populations could also be impacted in the areas to which fishing is displaced. In the eastern jettison zone there is less fishing activity so the consequences of fishing displacement on the seabed community, demersal fish and mobile invertebrates, marine mammals and seabirds are negligible, reaching minor impacts after 1000 or more launches.
- **Effect of the provision of hard substrates.** Another potential positive outcome for seafloor biota requiring hard substrates is that the jettisoned materials would provide further attachment sites. However, even after 10,000 launches this would provide only about 50 ha of additional attachment surface, leading to a moderate benefit at most.
- **Disturbance to marine fauna.** Noise and disturbance to marine fauna above and below water is a potential consequence of the jettisoned materials falling into the jettison zone. The chance of repeated disturbance to the same individuals or groups of marine mammals or seabirds increases with the number of launches. This was assessed as a low risk for up to 100 launches over two years, a moderate risk for up to 1000 launches over almost 20 years, and a high risk for up to 10,000 launches over almost 200 years.
- **Risk of direct strikes causing mortality to components of the ecosystem.** Direct strikes causing mortality are a low risk for all components of the ecosystem up to 1000 launches over an almost 20 year period. Direct strikes reach moderate levels of risk for the benthic invertebrate community, sensitive benthic environments, and a rare threatened species, the magenta petrel, after 10,000 launches over a period of almost 200 years.
- **Risk of smothering of sea floor organisms.** Smothering the feeding or respiratory structures of sea floor organisms by jettisoned materials was assessed as a low risk for all levels of launches up to 1000 launches and a moderate risk by 10,000 launches. This is likely to be a factor principally in areas of hard substrate where the jettisoned materials are unlikely to become buried in sediment so will be important principally on the Bounty Platform.

New Zealand legislation does not yet regulate these activities, since Rocket Lab is the first company that pretends to operate rocket launchings in the territory. The study concludes that the environmental effects of the activity may become significant after 10,000 launches, this would take 200 years to reach at one launch per week. The regulatory regime would have been reviewed well before this number of launches. During this review the Ministry allows the activity of the company.

Chapter 12

Social and Security Considerations

12.1 Social and security considerations

The potential of the CubeSats is very high and they might be the future of satellites. Their low cost and the easiness to construct them, compared to large satellites, make them accessible to countries with fewer resources, universities and people in general, making them able to explore the space and to pursue different missions. The assembly of a CubeSat is not very complicated but requires a minimum knowledge about the subject; in other words, now "you've got a user manual, a datasheet and a 3D model that you can download, and you've got an online shop where people can buy their power systems, etc with their credit card" [27].

This project is based on the design of a satellite constellation dedicated to communications relay between LEO satellites and between LEO satellites and the ground. This project is helping to develop the CubeSat industry and its use and it will demonstrate that these small satellites can carry out different missions that were previously done only by large satellites, as for example the communication.

Currently, the constellations of CubeSats dedicated to the communication are in development and the market is not very extensive, this is why this project, and the global coverage that it provides, could have a privileged place in this industry. The main commitment that this project has with the customers is to ensure that they will be able to communicate with any part of the world without problems.

Another important aspect to consider is that the constellation will provide total privacity to the costumers, ensuring that they make a correct use of it and avoiding that third people interfering in the communication.

In relation to security, it must ensured the proper functioning of the constellation. To do this, it must be considered three main factors, where CubeSats could be in danger: the launch of the payload, the permanence of the CubeSats in space and the ground stations.

The launch stage is one of the most important, because it is where the mission has more probability to fail. In the next figure can be observed the succes rate of orbital launches in the last 57 years. In 2014, there were a total of 92 unmaned launches and only 4 of them were failed. This indicates that the fail rate is only a 4,34 %, which is very low.

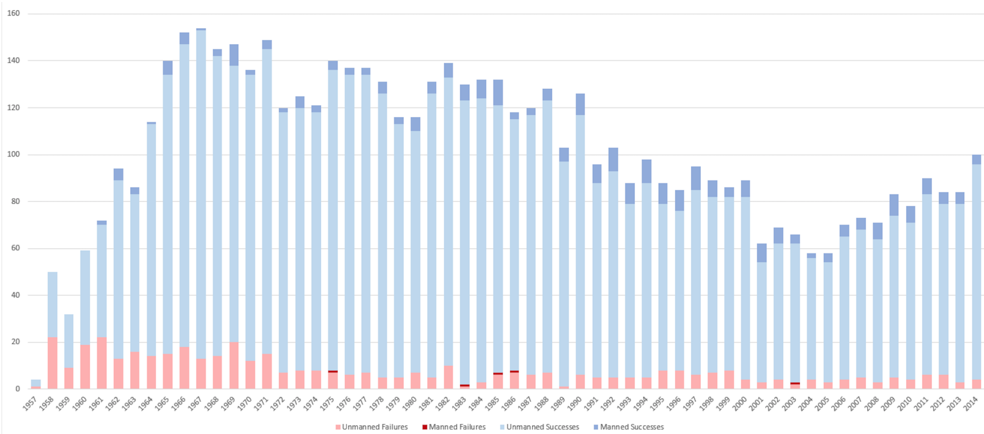


Figure 12.1.1: Orbital Launch Summary by Year

Once the constellation is in orbit, CubeSats can find dangers how colliding with other satellites or with space debris. The distances between most satellites is around hundreds of miles and there is not danger of collision, but the movement of space debris is unpredictable. In order to avoid this space debris, a CubeSat can perform a Debris Avoidance Manoeuvre (DAM). The responsible to control these fragmentation debris is *The United States Space Surveillance Network*. It consists of ground-based radars and optical sensors at 25 sites worldwide and Currently tracks more than 8000 orbiting objects.

Finally, the ground stations are a key element for the correct operation of the constellation and they must prevented from stop working. To do this, each ground station will have its operator, to control the operation of the installation, and a security system, to avoid intrusions.

12.2 Legislation

The legislation concerning activities related to space is the Space Law. Space Law is an international law comprised of international treaties and agreements. Its most important rules are the five international treaties, which have been developed under the supervision of the United Nations. The body that promotes these regulations is the United Nations Office for Outer Space Affairs (UNOOSA).

The international law is only applicable to the states that are parties to the treaties. According to the Outer Space Treaty, states are responsible for their national space activities, public or private. For this reason, each state usually adopts its national space regulations.

In the case of the Astrea constellation, since the company is based in Spain (a party of the Space Law), the current legislation is the *Real Decreto 278/1995* of 24 February 1995. According to this Royal Decree, the objects launched from Spain or whose launch has been promoted by Spain, should be registered in the *Registro Español de Objetos Espaciales Lanzados al Espacio Ultraterrestre* (Spanish Registry of Objects Launched into Outer Space). The necessary data to register the satellite must be provided to the *Dirección General de Tecnología Industrial del Ministerio de Industria y Energía* (Department of Industrial Technology of the Ministry of Industry and Energy). This department will notificate the registry to the Secretary-General of the United Nations.

The registration has to contain the following data:

a) Name of launching State or States;

b) An appropriate designator of the space object or its registration number;

c) Date and territory or location of launch;

d) Basic orbital parameters, including:

   I) Nodal period;
   II) Inclination;
   III) Apogee;
   IV) Perigee;

e) General function of the space object.

and any other useful information. For example, in the case of one of the Astrea satellites, the registration will be:

a) Spain

b) AstreaSAT 1

c) 22 February 2018,

d) Basic orbital parameters: Low Earth Orbit

   I) 95,4815 minutes

   II) 72 degrees

   III) 6.913,0 km

   IV) 6.913,0 km

e) CubeSat 3U, part of the communications constellation Astrea

Chapter 13

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