astrea

ESEIAAT

Escola Superior d'Enginyeries Industrials, Aeroespacial i Audiovisual de Terrassa

UNIVERSITAT POLITÈCNICA DE CATALUNYA

Cubesat Constellation
Astrea

Report

Degree: Aerospace Engineering
Course: Engineering Projects
Group: G4 EA-T2016

Delivery date: 22-12-2016

Students:

Cebrián Galán, Joan
Foreman Campins, Lluís
Fuentes Muñoz, Óscar
Herrán Albelda, Fernando
Martínez Viol, Víctor
Pla Olea, Laura
Puig Ruiz, Josep
Tarroc Gil, Sergi
Urbano González, Eva María
Fontanes Molina, Pol
Fraixedas Lucea, Roger
González García, Sílvia
Kaloyanov Naydenov, Boyan
Morata Carranza, David
Pons Daza, Marina
Serra Moncunill, Josep Maria
Tió Malo, Xavier

Customer: Pérez Llera, Luís Manuel

Contents

List of Tables

List of Figures

Part I

Orbit Design

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).

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

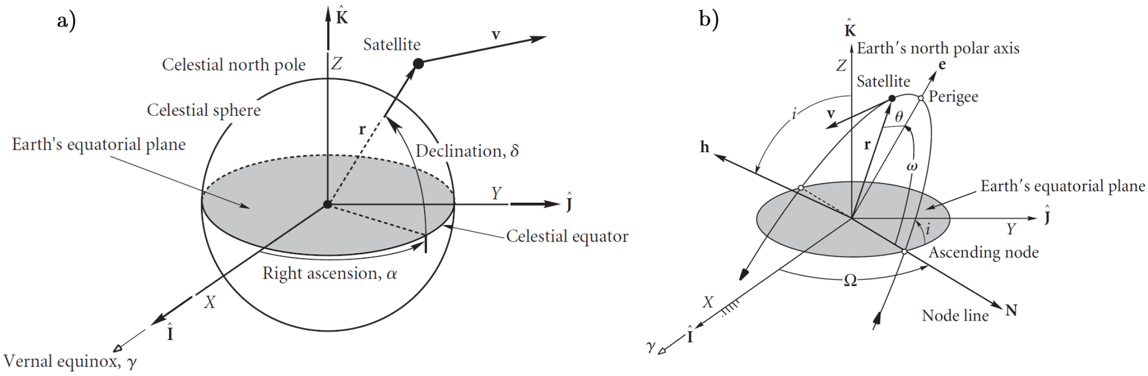


Figure 1.0.1: a) Geocentric-equatorial frame and b) Classical Orbital Elements. Extracted from [2].

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.

1.1 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 or ν): 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.

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

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

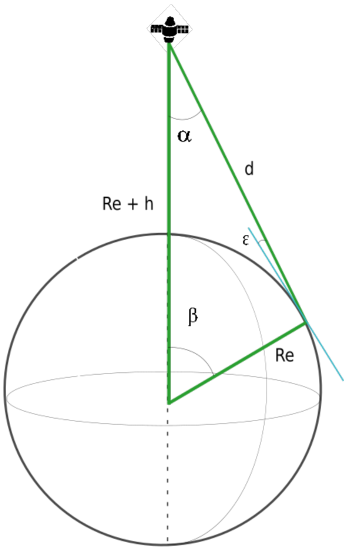


Figure 2.1.1: Single satellite coverage geometry

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.

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

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.

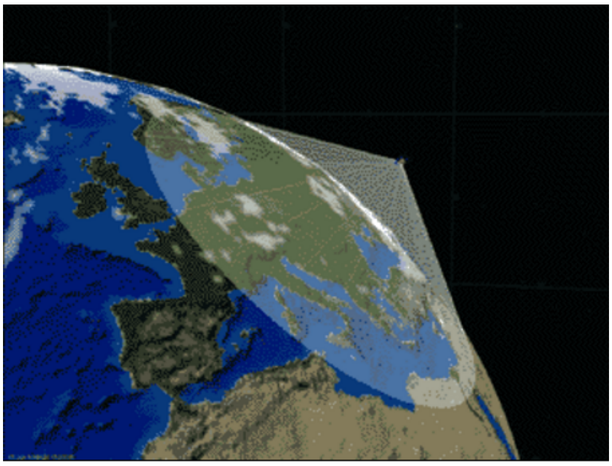


Figure 2.2.1: Elevation angle cone. Source: NOAA

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 [8]. The most important parameters are tge following:

- **Atmospheric gases:** water vapour and oxygen absorptions; important when frequencies are above 3 GHz. More information [9] and [10].

- **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.

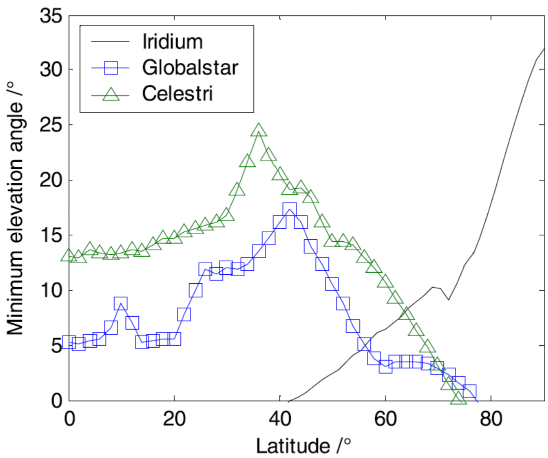


Figure 2.2.2: Minimum elevation angle as function of latitude. Source: [3]

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:

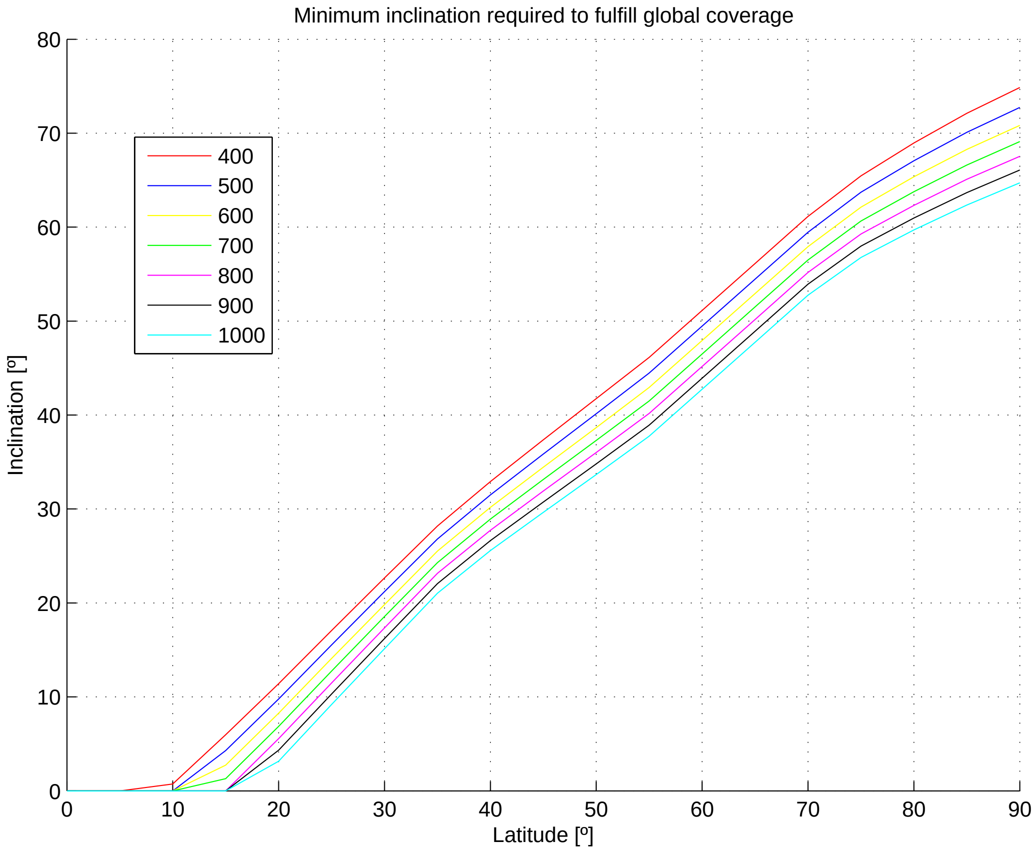


Figure 2.3.1: Minimum Inclination to provide coverage at different latitude for different orbit apogees.

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.

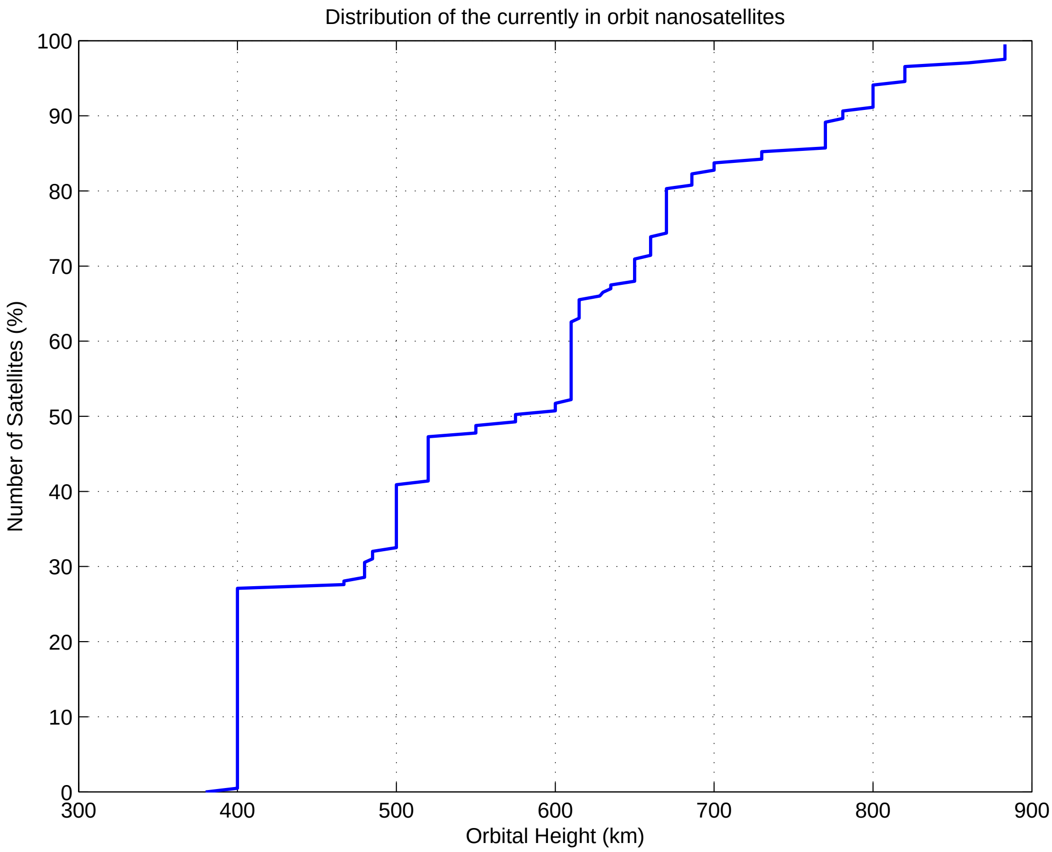


Figure 2.5.1: Distribution of the currently in orbit nanosatellites.

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;

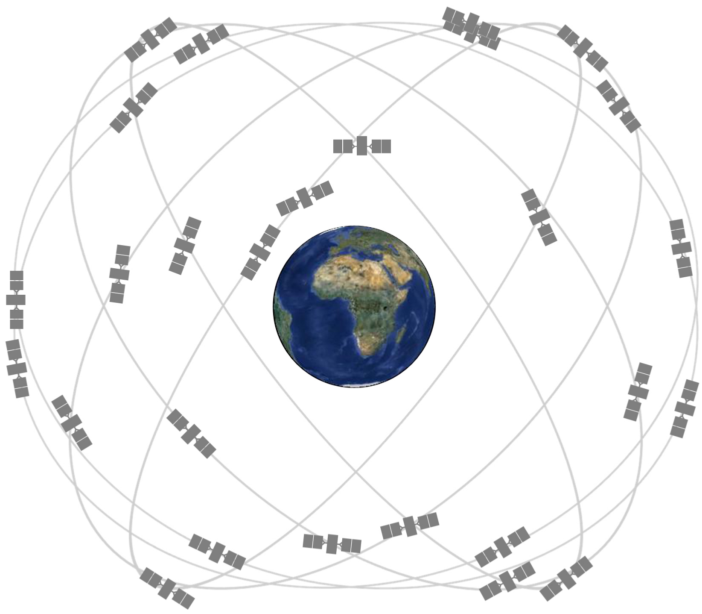


Figure 3.1.1: Distribution of the expanded 24-slot GPS constellation. [?]

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;

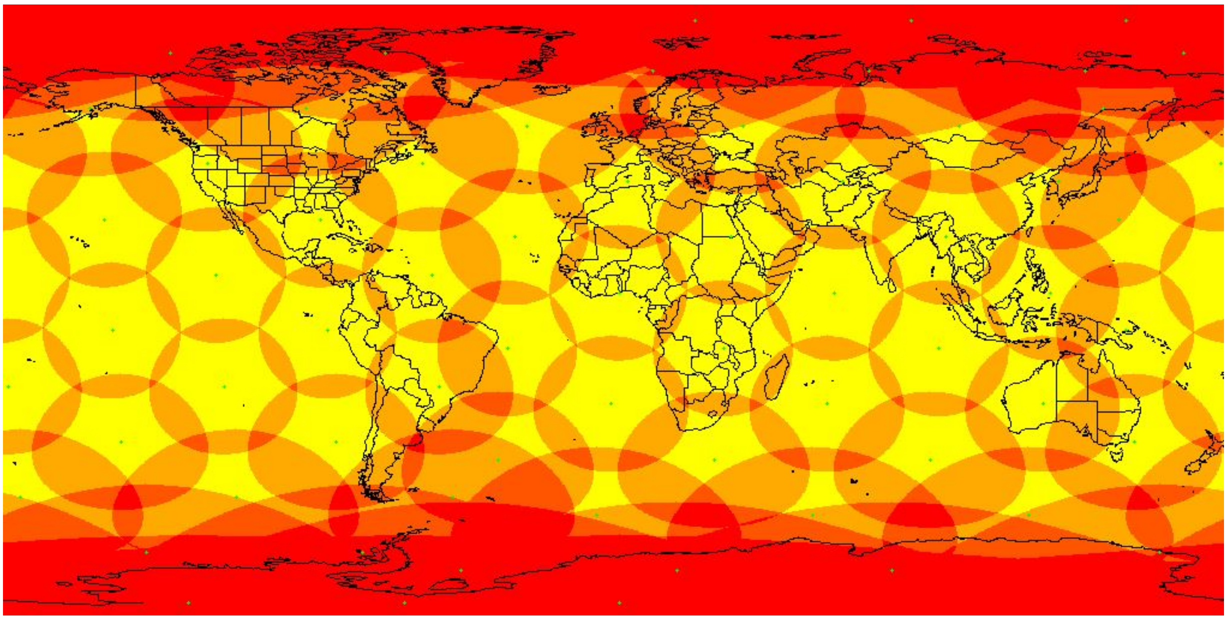


Figure 3.2.1: Distribution of the 66 Iridium constellation satellites. Generated using [?]

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:

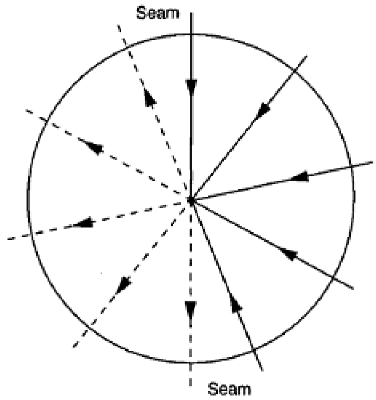


Figure 3.2.2: Distribution of the planes for Polar Orbits design.

3.2.3 The Streets of Coverage Method

This Street of Coverage Method is obtained from [6]. 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

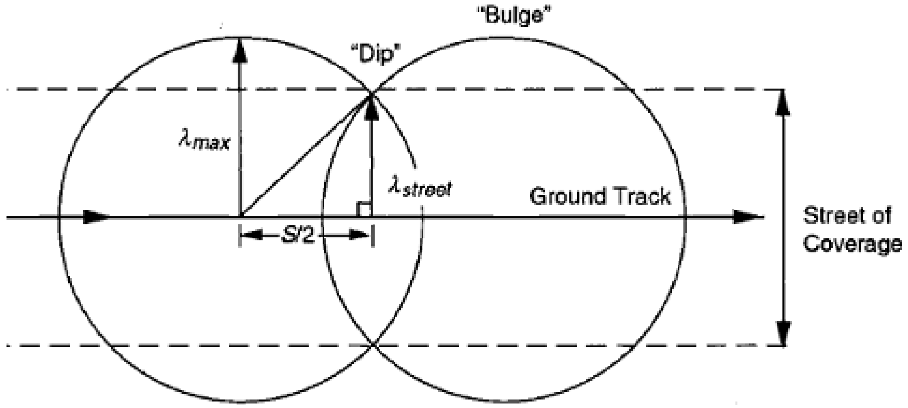


Figure 3.2.3: Single plain street of coverage. The footprints of the satellites superpose leading to a street. [?]

From the figure it can be inferred:

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

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

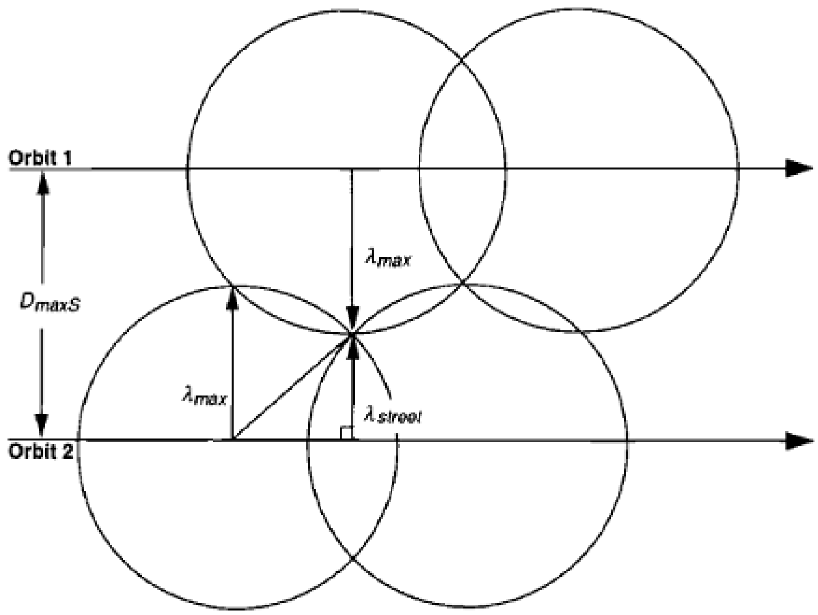


Figure 3.2.4: Two plains streets of coverage. An optimum phasing needs to be obtained. [?]

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.

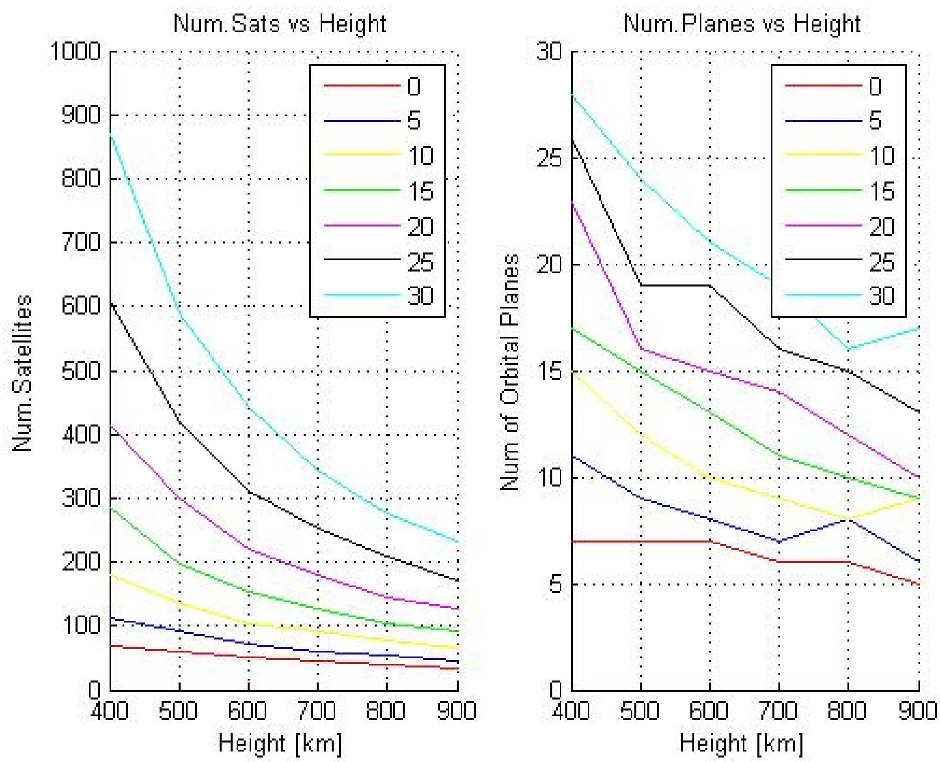


Figure 3.2.5: Variation of number of satellites for different heights and elevation angles

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:

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

.

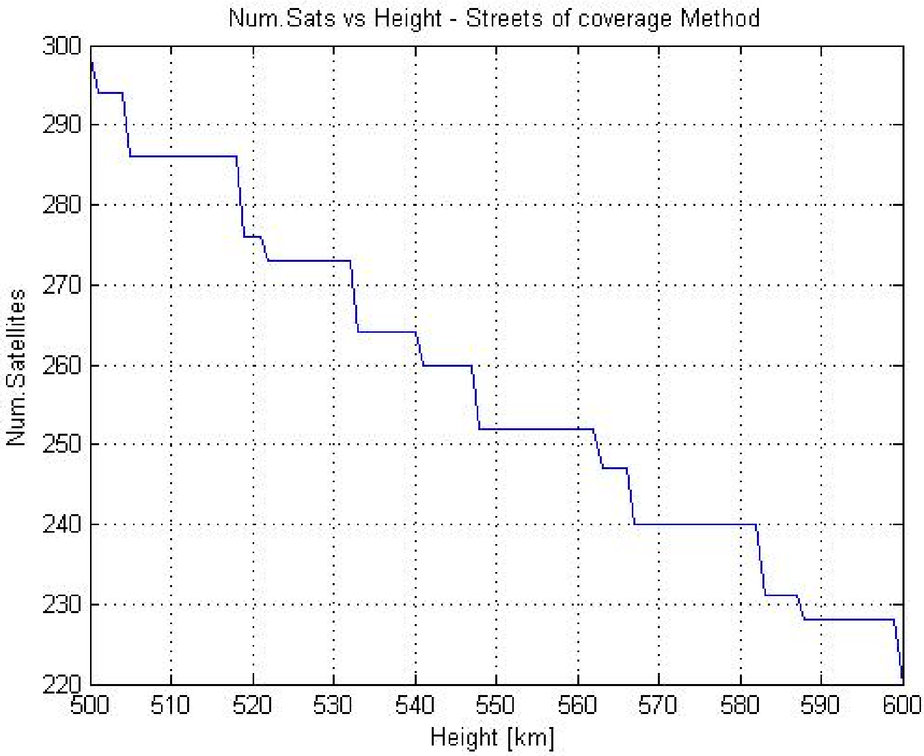


Figure 3.2.6: Variation of number of satellites for different heights between 500 and 600km.

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 [4].

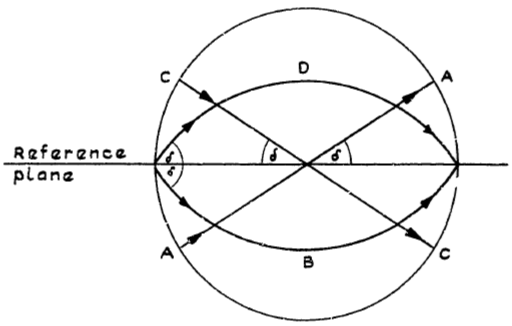


Figure 3.3.1: Definition of the inclination δ. Extracted from [4]

- 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).

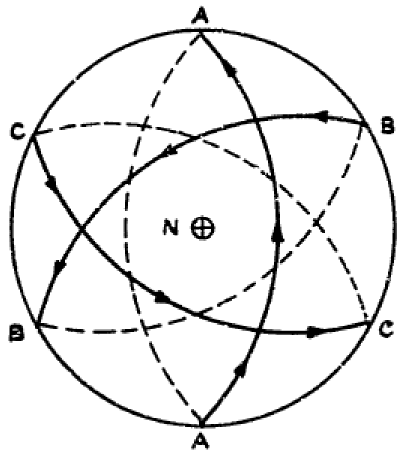


Figure 3.3.2: Delta pattern as seen from the North Pole. Extracted from [5]

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 [**?**].

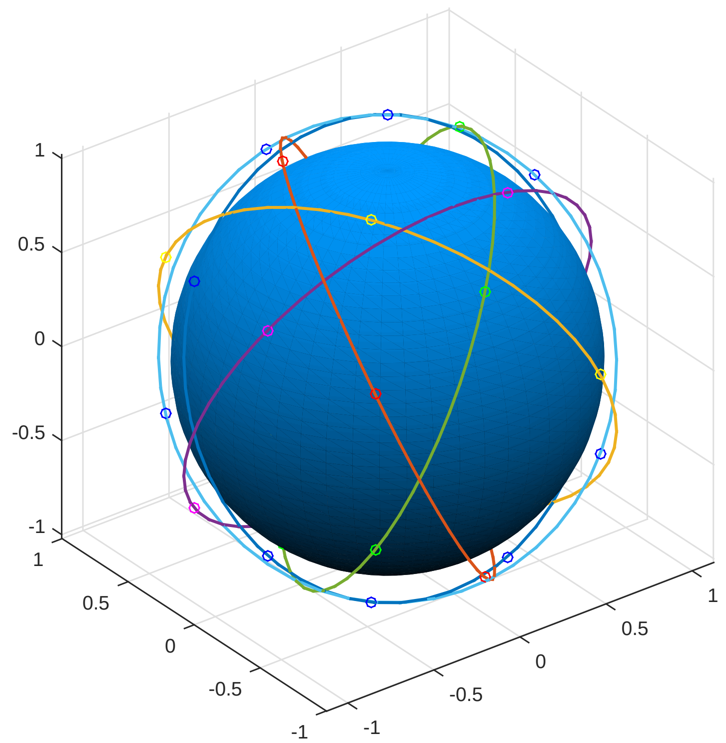


Figure 3.3.3: Delta pattern 65°: 30/6/1

3.3.1.2 Notation

J.G. Walker developed a notation to define this constellations with only 4 parameters [5]:

$$i : T/P/F$$

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.

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.

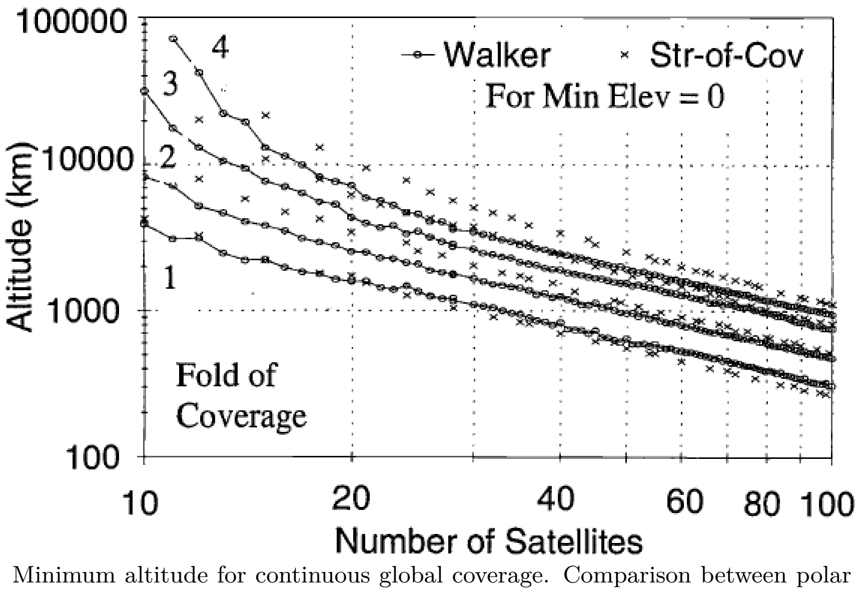


Figure 3.3.4: Minimum altitude for continuous global coverage. Comparison between polar patterns and Walker delta patterns. Extracted from [6]

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 having some inconvenients.

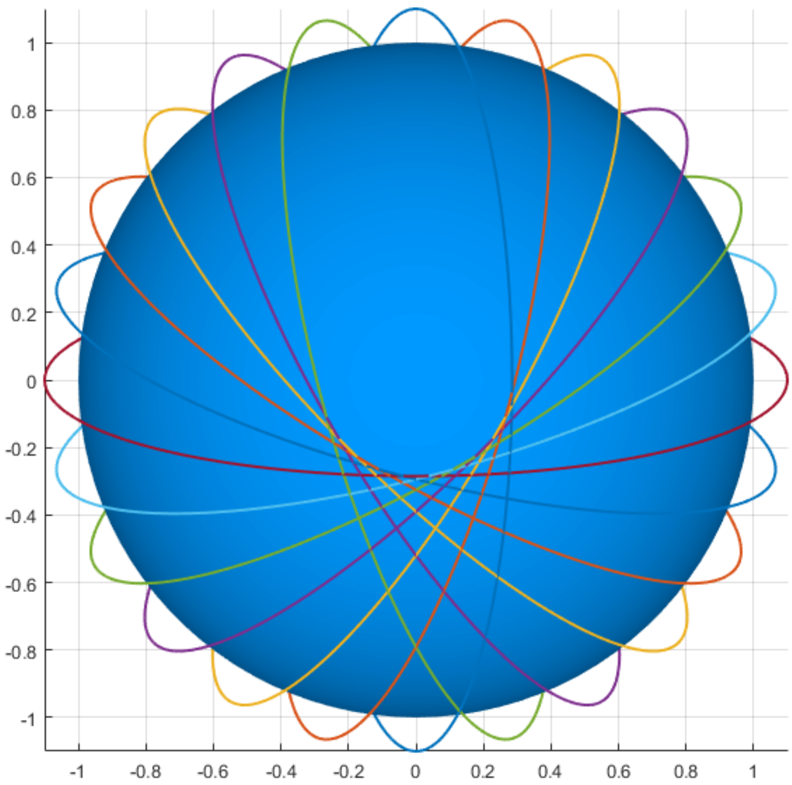


Figure 3.3.5: 12 plane SWDC. Note the gap and the equidistant planes

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.

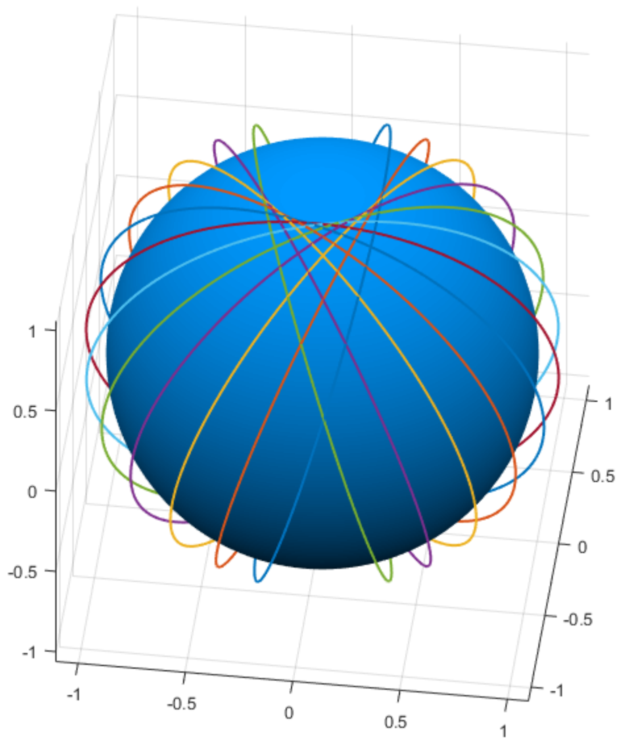


Figure 3.3.6: This geometry distribution induces a large anti-symmetric gap

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.

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.

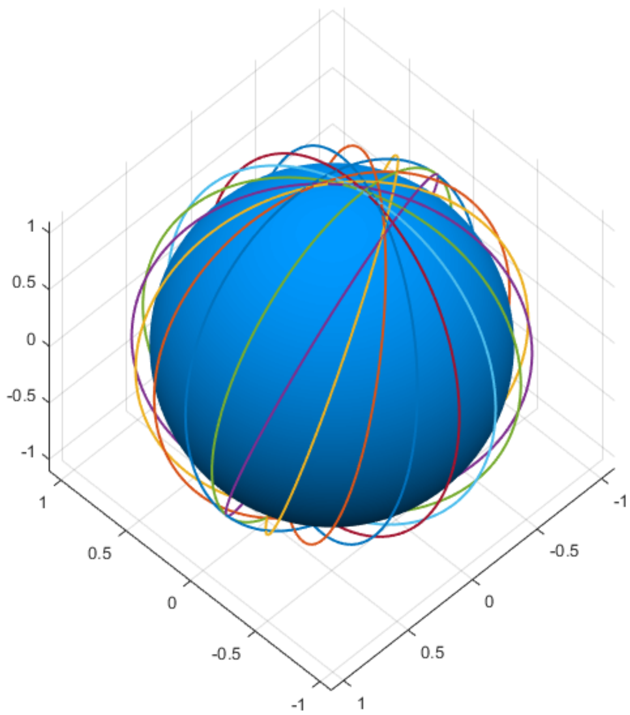


Figure 3.3.7: Added polar orbit to the 11 plane based SWDC

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.

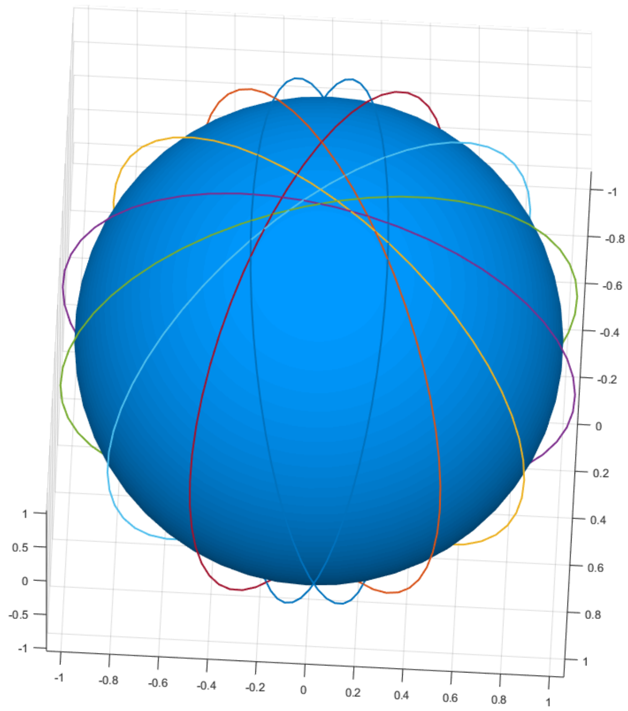


Figure 3.3.8: 8 plane based MWDC generated for 210 degrees

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.

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 [º]
h	Height [km]
in	Inclination angle [º]
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

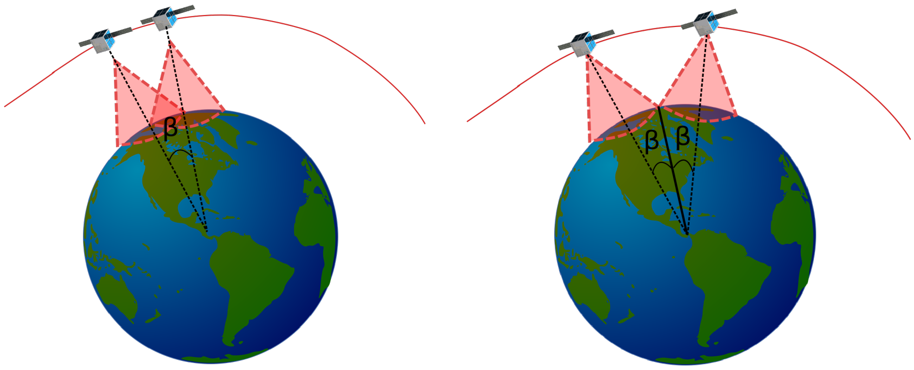


Figure 3.4.1: Geometrical conditions needed to fulfill global coverage.
On the left: Condition between satellites of different planes.
On the right: Condition between satellites of the same plane

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

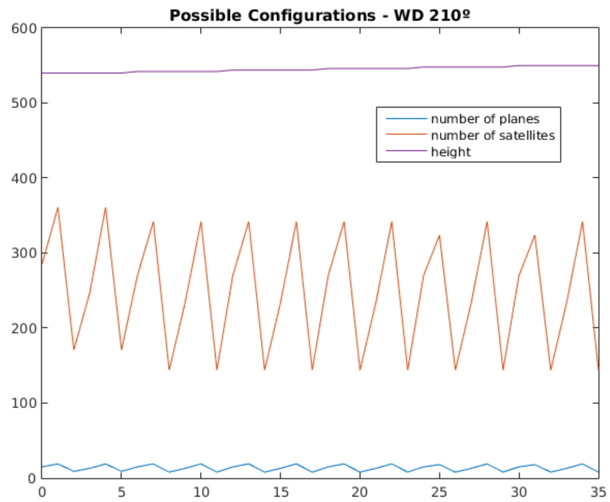


Figure 3.4.2: Possible satellite configurations for a 210° Walker Delta configuration

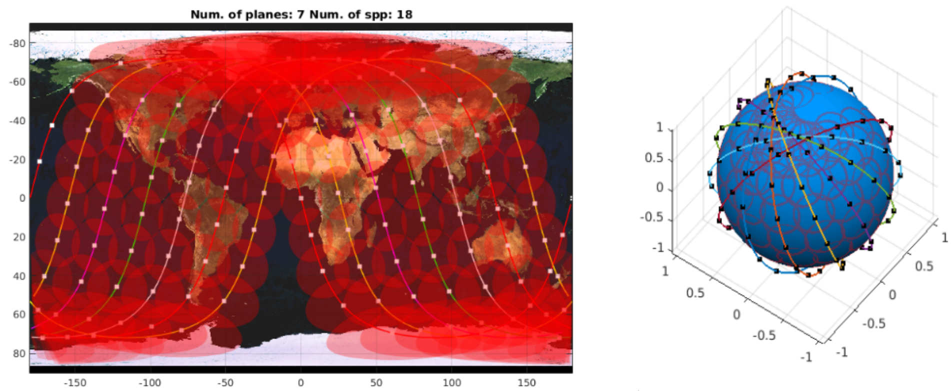


Figure 3.4.3: Ground track and spherical representation for a 180° Walker Delta configuration

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

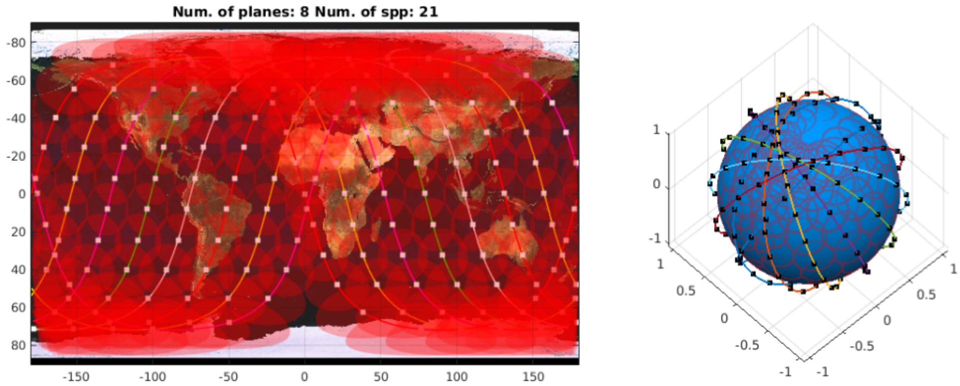


Figure 3.4.4: Ground track and spherical representation for a 210° Walker Delta configuration

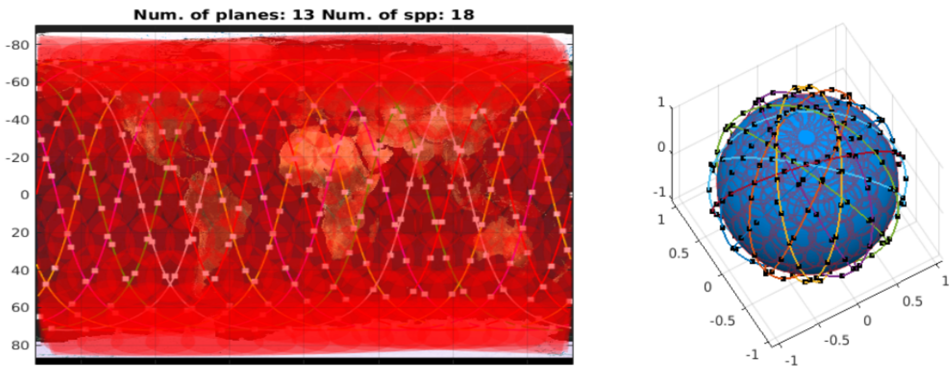


Figure 3.4.5: Ground track and spherical representation for a 360° Walker Delta configuration

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 [1]

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 Perturbation: 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 approximations 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:

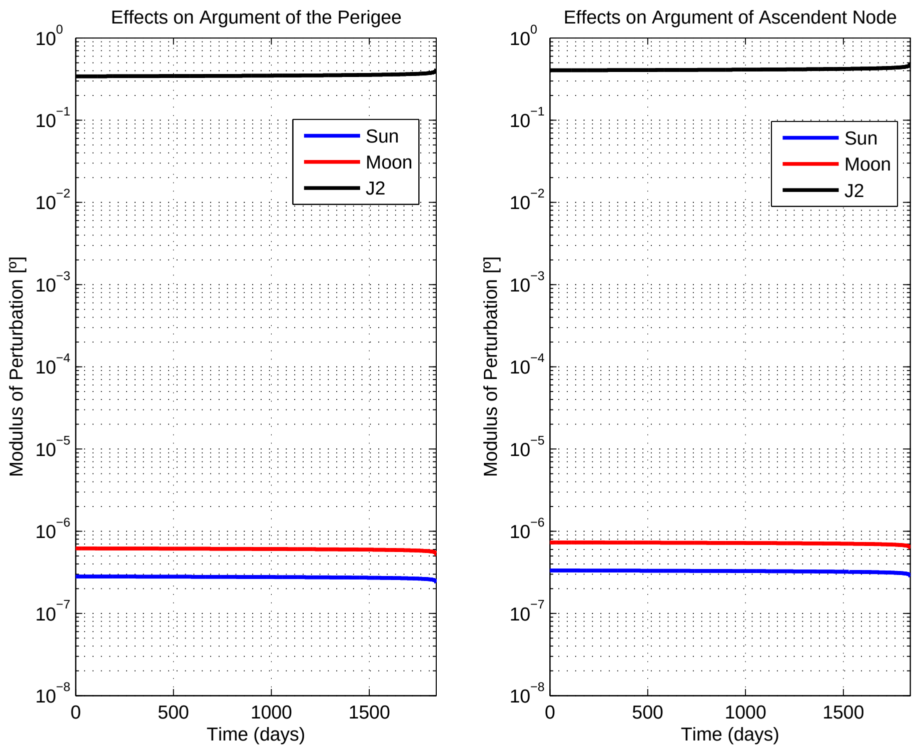


Figure 4.2.1: Logaritmic plot of the modulus of the increases in Angular Arguments of the orbit

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 observation 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.

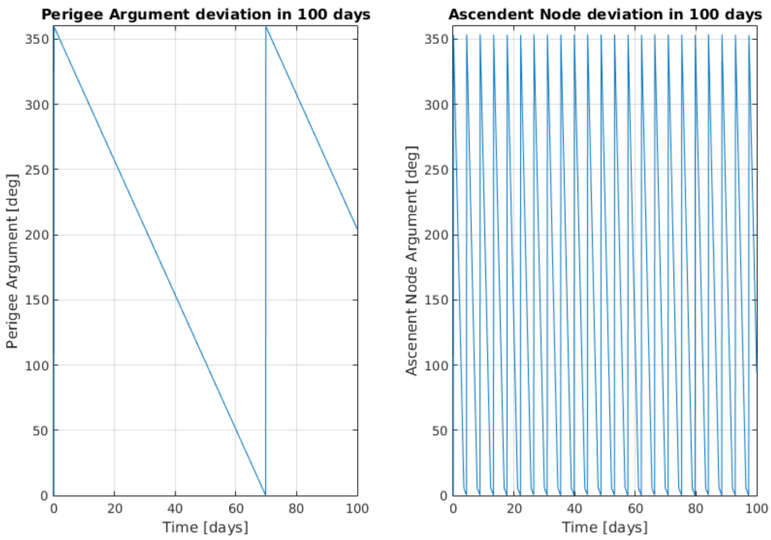


Figure 4.3.1: Ascention node perturbation
On the left: Perigee deviation in terms of time.
On the right: Ascending node deviation in terms of time

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 [?].

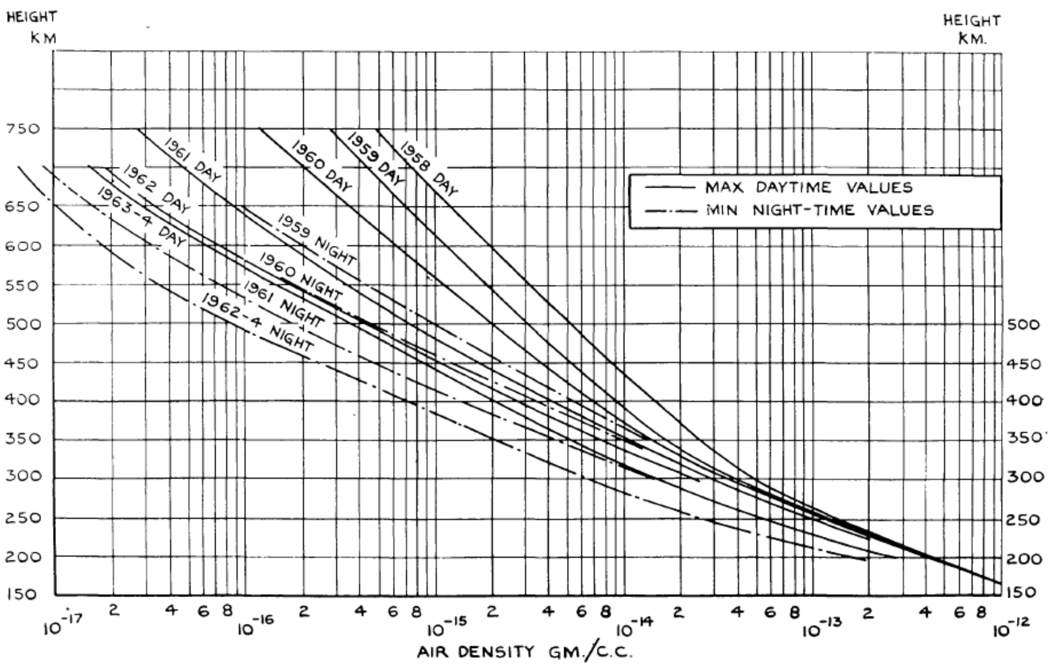


Figure 4.3.2: Deviation of densities in the upper atmosphere due to the 19th solar cycle. Source: [?]

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.

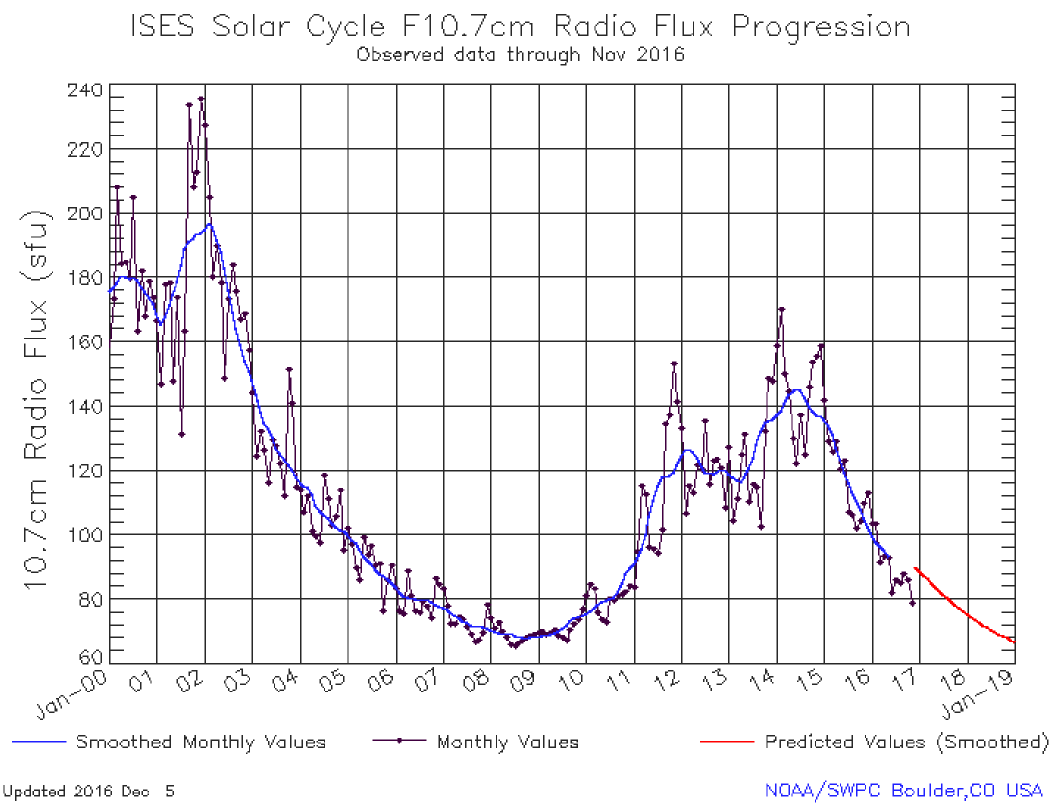


Figure 4.3.3: Measured intensities of the 23rd and 24th solar cycles. Source: NOAA

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 4.3.3

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 approach of the drag computation have been done in order to determine the orbit decay and consequently compute how much time a satellite last until it reenters the Earth atmosphere.

4.3.3.2 Drag Computation Algorithm

Given the definitions to calculate orbital perturbations described in 4.1.1 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 depends 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 order 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 4.3.3).

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

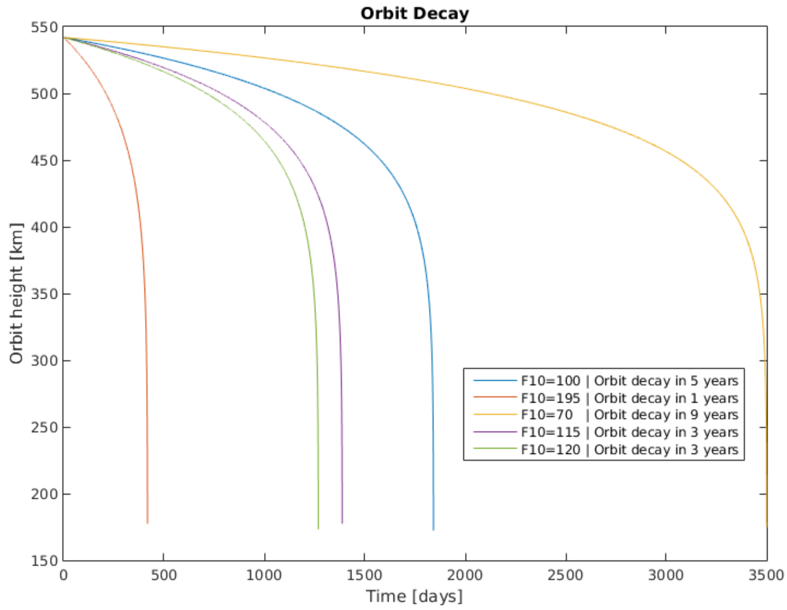


Figure 4.3.4: Orbit Decay computed for several values of

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 approximated orbit decay of 5 years.

In order to verify if the results obtained by the approximation used are valid, a more advanced analysis has been carried out using what was previously defined (in 4.1.1) as General Perturbations method. This method is based on propagating the perturbations making use of the numerical integration on the dynamics equations. Both the algorithm and the obtained results can be consulted in the Attachment I of this Report.

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:

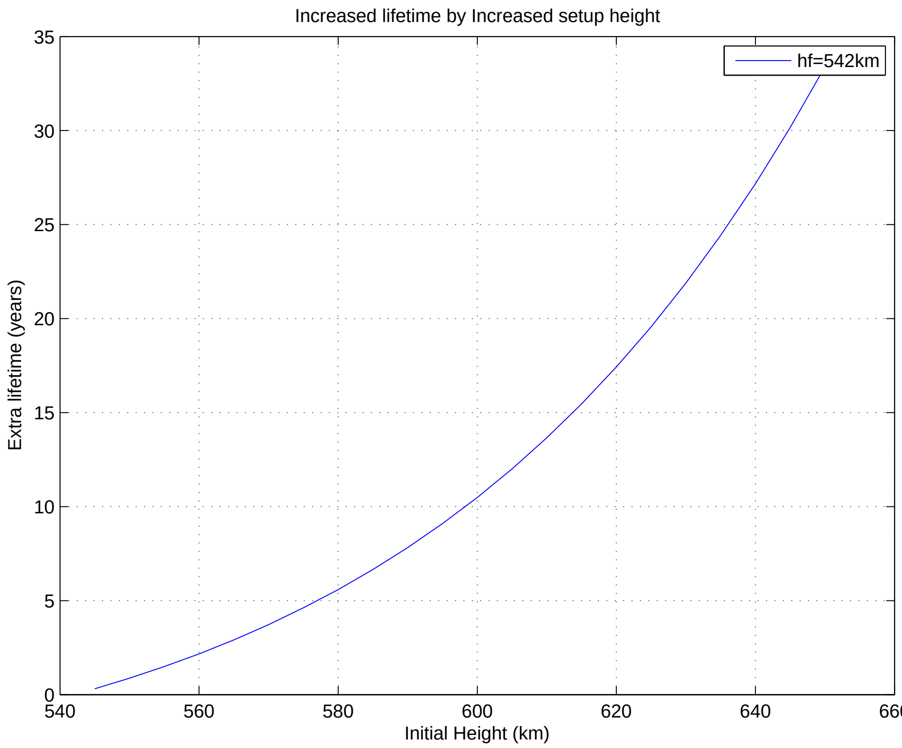


Figure 4.4.1: Increase in the Lifetime obtained by setting the constellation in a higher orbit

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).

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:

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

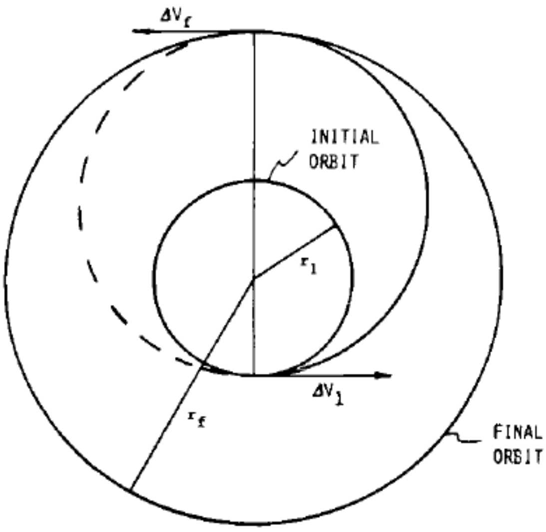


Figure 4.4.2: Hohmann transfer. Extracted from [6]

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.

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

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

Thrust	100 μN
Specific Impulse	2150 s

Table 4.4.1: Simulation Thruster Parameters

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 4.4.4 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:

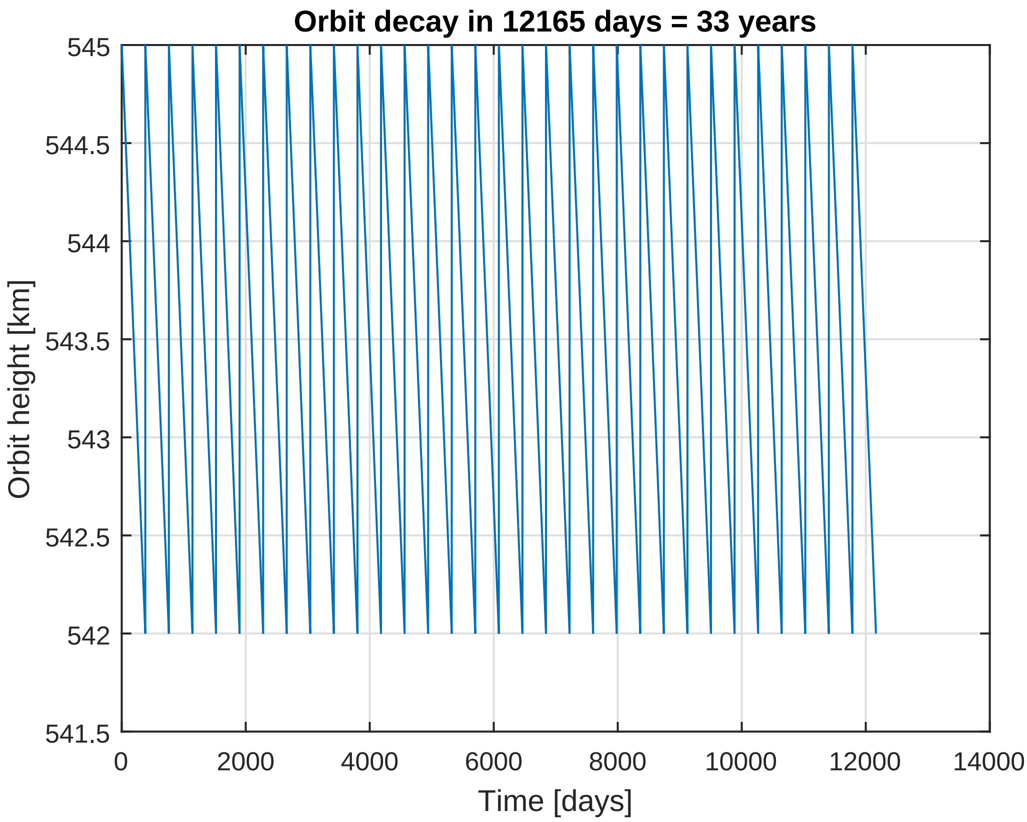


Figure 4.4.3: Height variation of the satellite

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

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

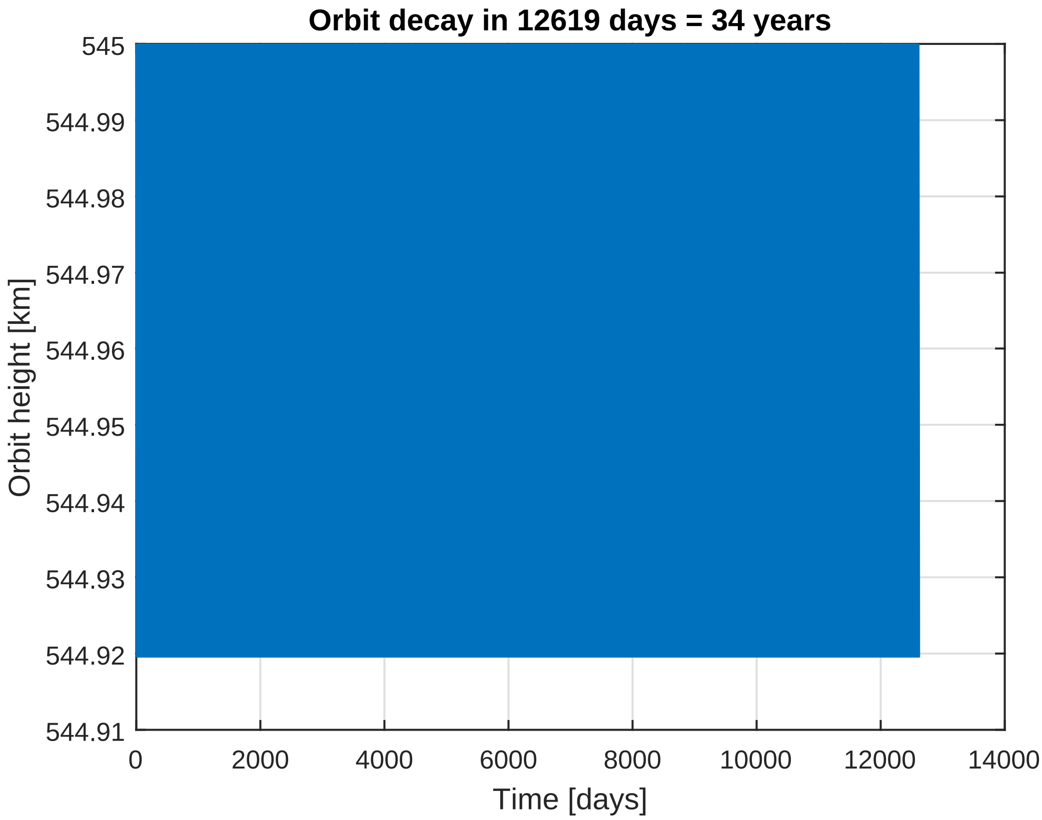


Figure 4.4.4: Height variation of the satellite with a more restrictive minimum height

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)

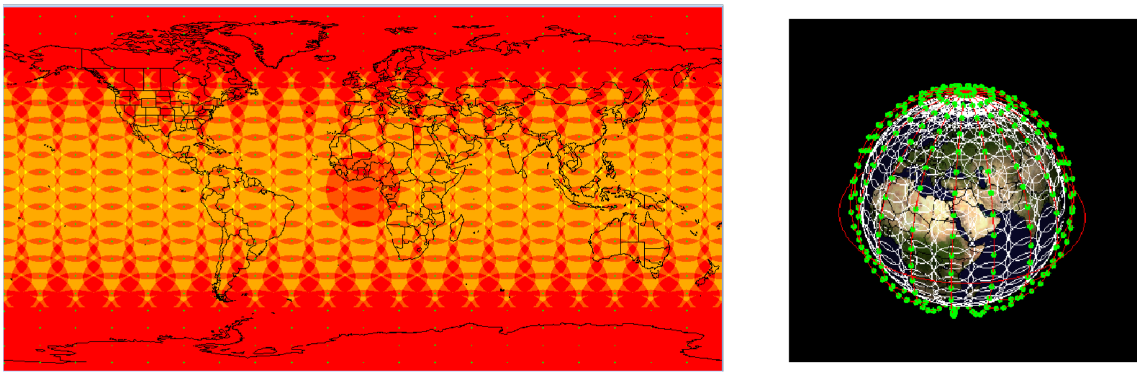


Figure 5.1.1: Candidate 1. Full Polar constellation with global coverage. h= 560km; Np=20; Npp=21; Tsat=420

- 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 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

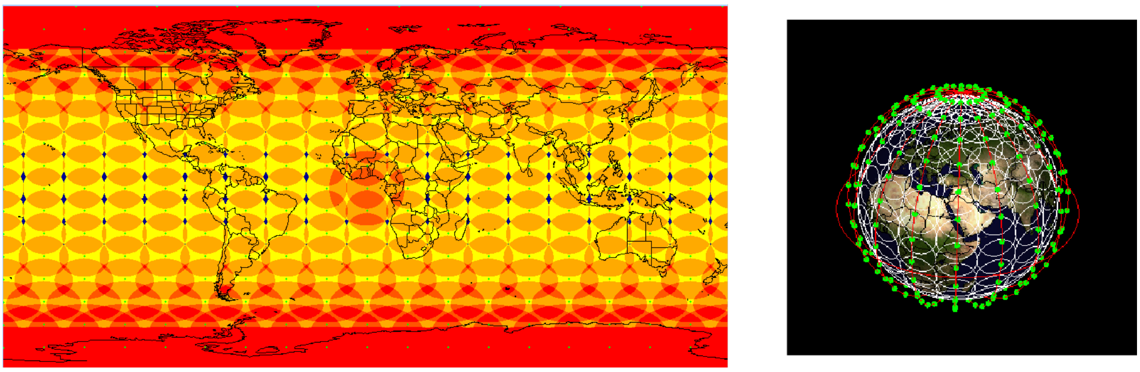


Figure 5.1.2: Candidate 2. Full Polar constellation with total ground station coverage. h= 550km; Np=18; Npp=20; Tsat=288

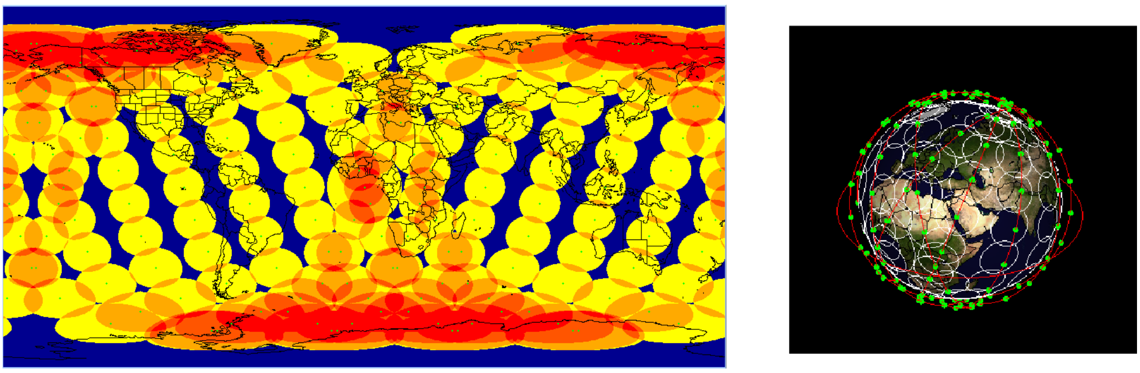


Figure 5.1.3: Candidate 3. 210° Walker-Delta constellation configuration. h= 542km; in=72; Np=8; Npp=21; Tsat=168

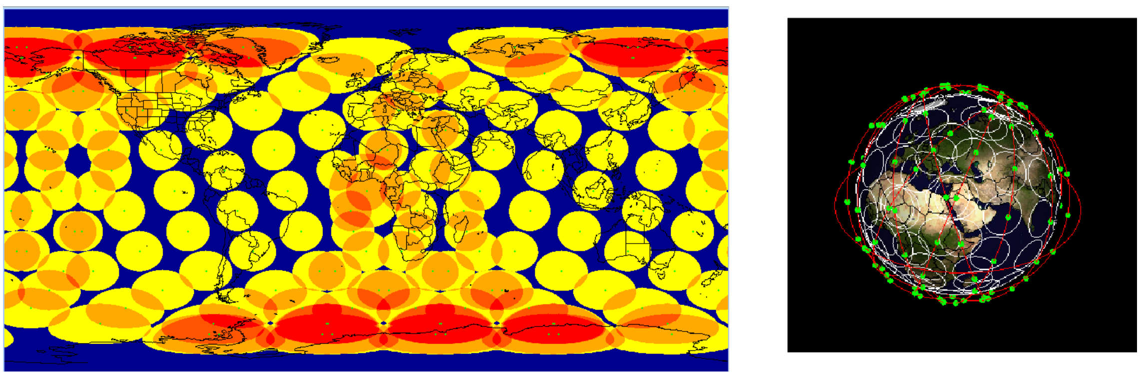


Figure 5.1.4: Candidate 4. 225° Walker-Delta constellation configuration. h= 542km; in=72; Np=9; Npp=17; Tsat= 153

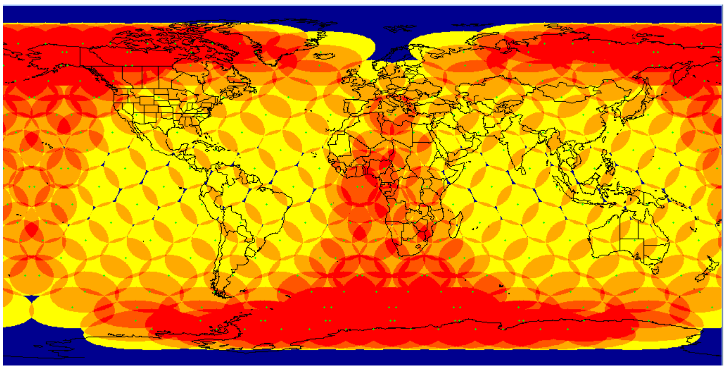
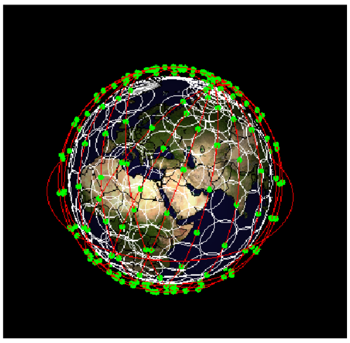



Figure 5.1.5: Candidate 5. 210° Walker-Delta constellation configuration with total coverage of the lattitudes from 0 to 52 degrees. h= 560km; in=72; Np=9; Npp=17; Tsat= 153

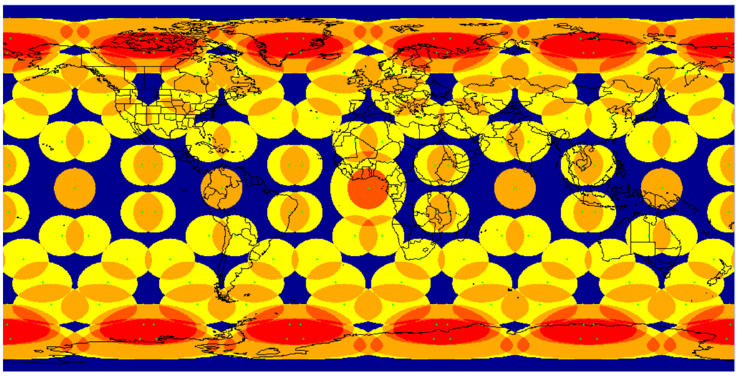
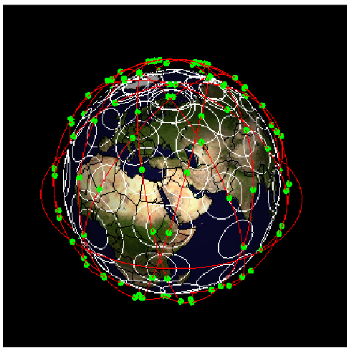



Figure 5.1.6: Candidate 6. 225° Walker-Delta constellation configuration.
h= 542km; in=72; Np=9; Npp=21; Tsat= 189

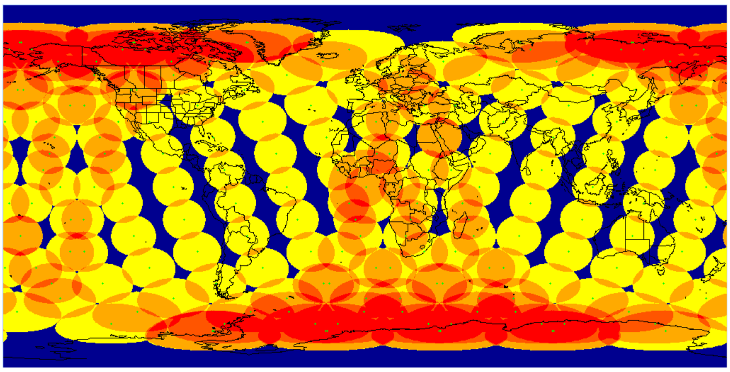
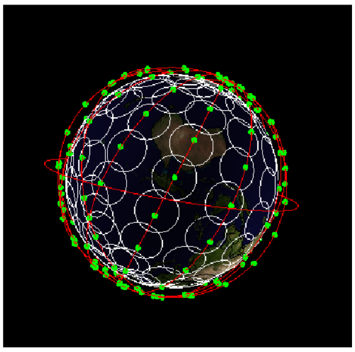



Figure 5.1.7: Candidate 7. Full Walker-Delta constellation configuration.

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

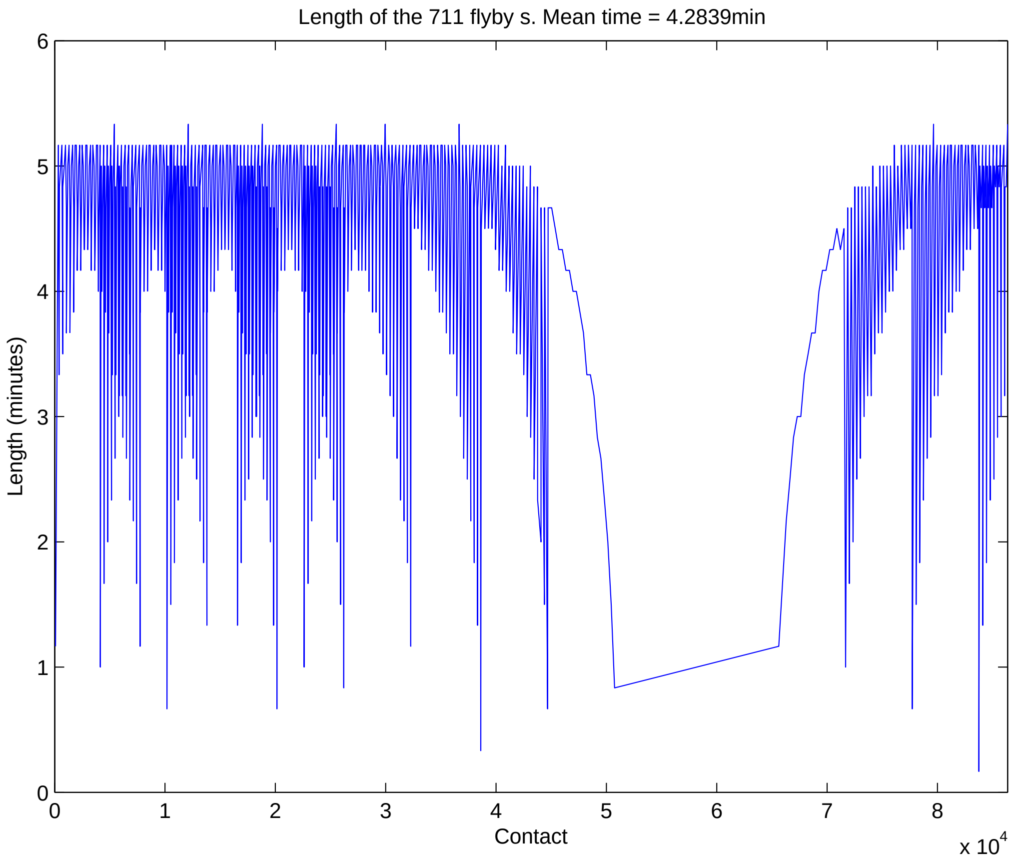


Figure 5.2.1: Length of the passes on the example GS.

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

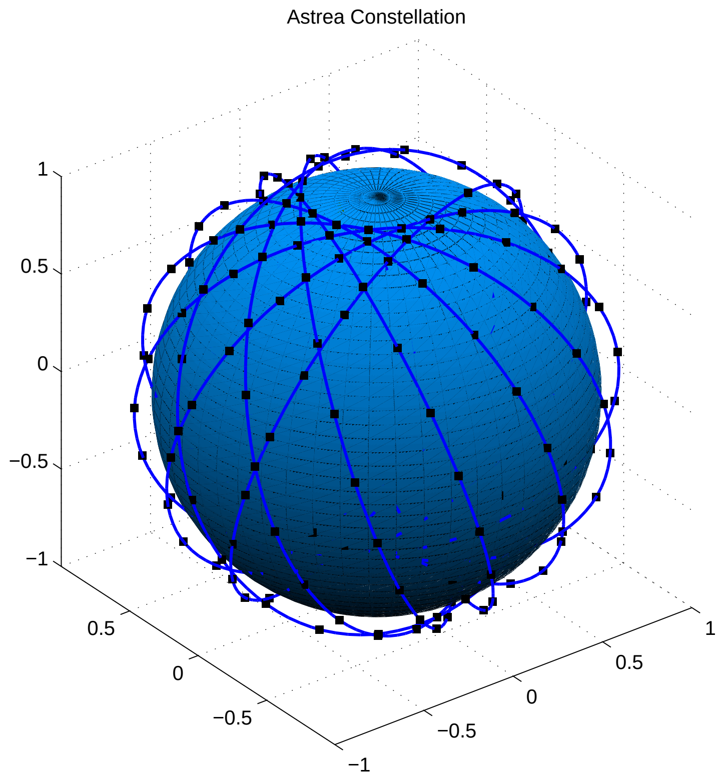


Figure 5.3.1: Astrea Constellation Final Configuration.

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 $\pm$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.

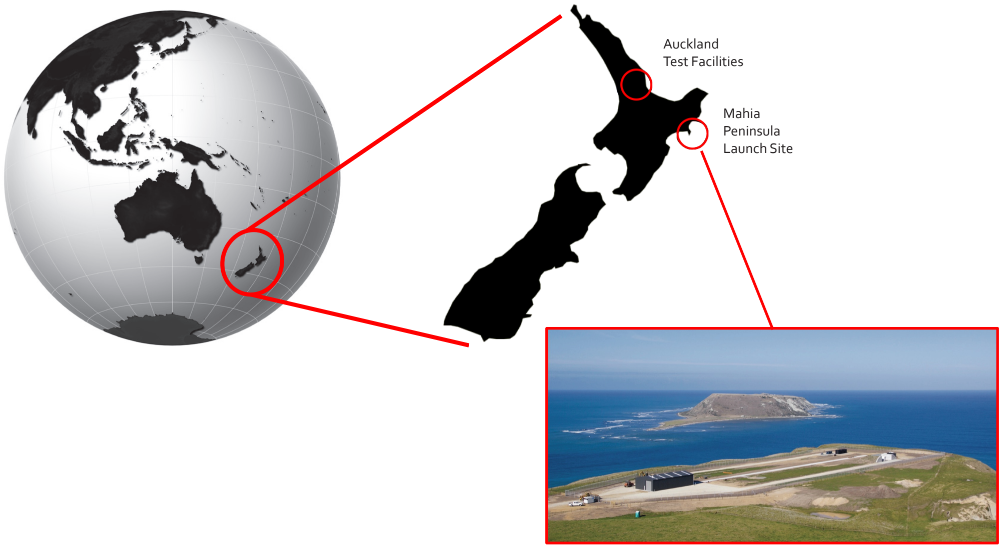


Figure 6.2.4: Rocket Lab Facilities

6.3 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.

6.3.1 Launch site and vehicle analysis

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. A table of this seven candidates is shown in the ANNEX.
Once this first selection is done, more accurate information is needed so as to reach a reliable conclusion. However, none of the 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.
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 8 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 6.3.1: 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.
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, 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. 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. 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. Some images of Electron are showed in the ANNEX, a picture of the launching site is also showed.

6.4 Deployer

The objective of this section is to give a brief explanation of what is a deployer and how it works. 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. The main characteristics of the two deployers are listed in the ANNEX, also two pictures of them are shown there. 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. As seen in the ANNEX, GPOD has accessible panels whereas ISIPOD is fully closed, hence, ISIPOD is not suitable for the needs of the Astrea constellation. 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. The price for the GPOD deployer is 16000 US dollars per unit.

6.5 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 manoeuvre required so as to deploy the satellites into orbit.

6.5.1 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 which are included at the corresponding attachment on which their launching procedure is explained.

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 launch per orbital plane, consequently, the first placement consists on 9 launches and all the logistics around them. Rocketlab is capable of launching once per 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 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).

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 Launch minus 423 days.
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.

6.5.2 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.

6.5.3 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.

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 can be computed.

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.

6.5.3.1 Plane Order

Planes are going to be placed consecutively from the first one to the last one. This way allows a growing wider range of communication during the time the constellation is being set up. The whole discussion on which would be the best way to set up the orbital planes on the first placement can be found in the attachments section.

6.6 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.

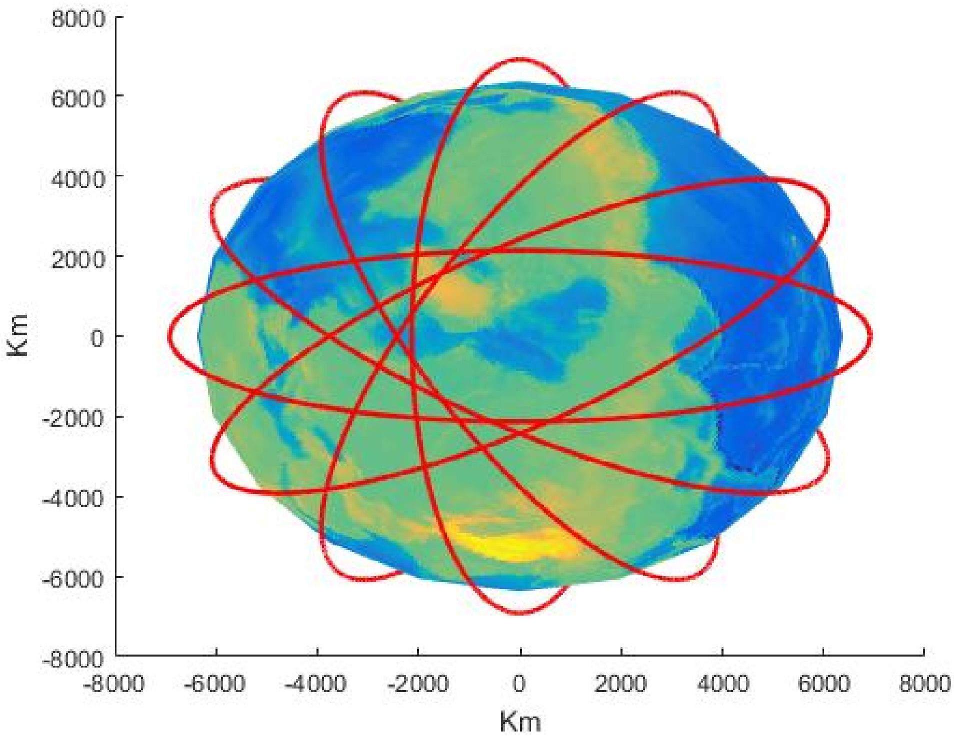


Figure 6.6.1: Old Constellation

6.7 Spare Strategy

6.7.1 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

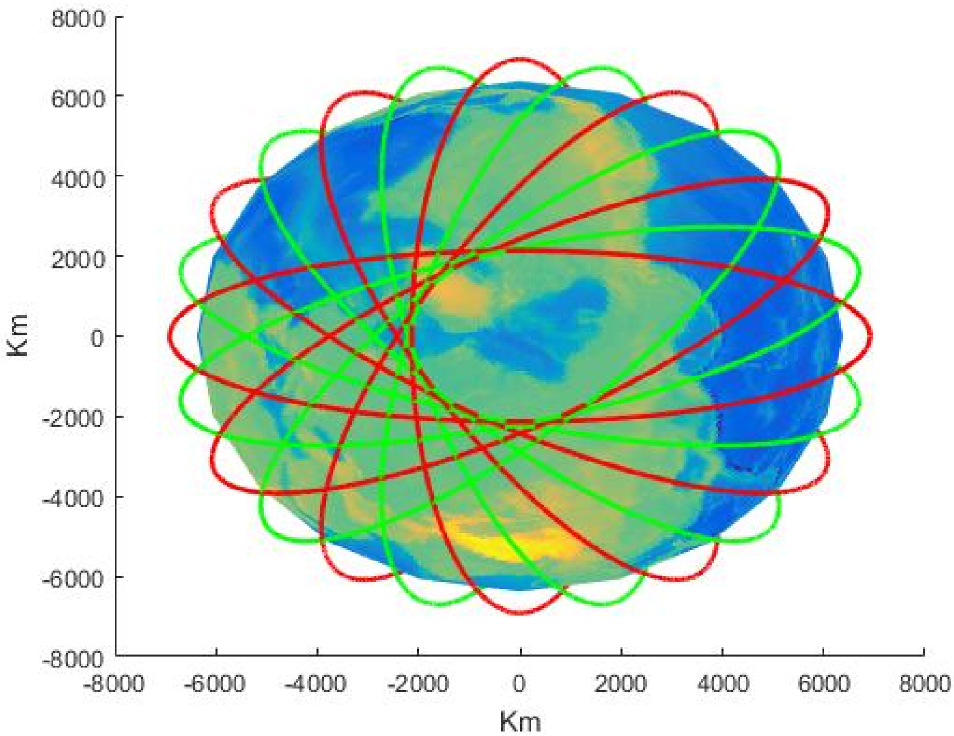


Figure 6.6.2: Old and New Constellations

constellation flexibility to degrade the service to a lower performance level during a certain period and to its cost.

6.7.2 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

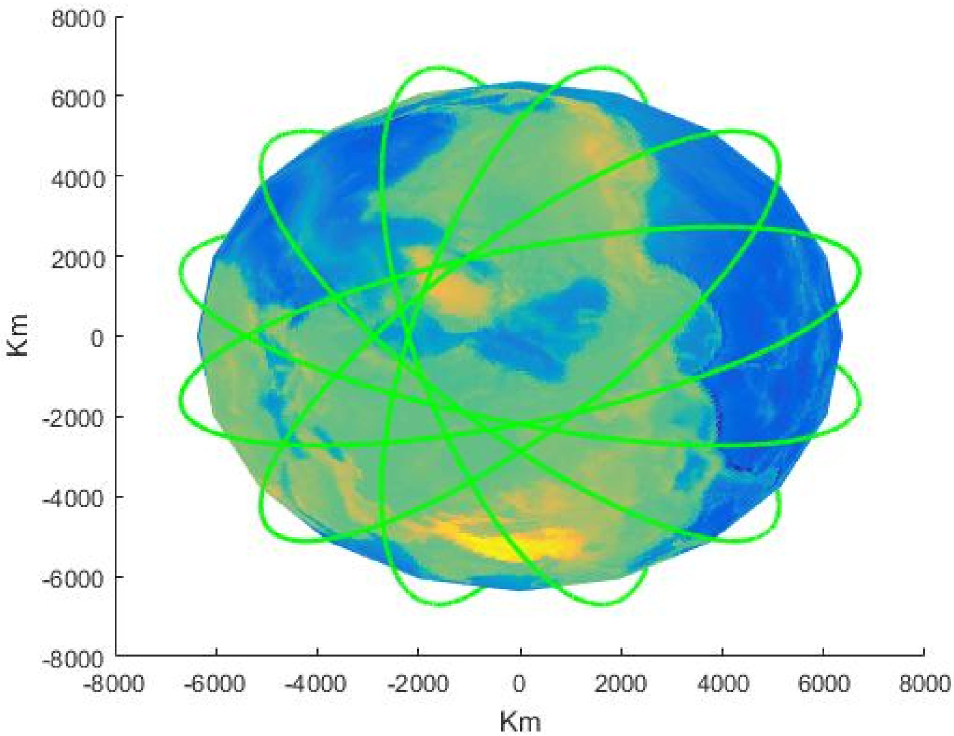


Figure 6.6.3: New Constellation

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.

6.7.3 Major failure deffinition

It can be stated 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.

6.7.4 Decision

Having studied all the possibilities of failure and taking into account that the performance 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.

6.8 End-of-Life Strategy

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.
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. Moreover, due to the fact the replacement strategy has been designed so as to avoid the need of a quick de-orbit, to adopt a controlled de-orbit decay it is not necessary and the uncontrolled one is still being adequate. Finally, the fact that the constellation is placed at LEOs makes easier the application of the uncontrolled de-orbit because the perturbations present in this altitudes increase the satellites decay rate of approach. As a result, given all the stated reasons, it is decided to use the uncontrolled de-orbit.

6.9 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

Chapter 7

Space Segment Protocol Stack

"The wonder is, not that the field of stars is so vast, but that man has measured it."

Anatole France, 1894

7.1 Introduction

Over this chapter, the **space communication protocols** are going to be defined. That is, a set of rules are going to be established in order to achieve the actual node-to-node communication. Altough the scope of the chapter is limited to the space segment, this inital introduction on the protocol definition is usfeul for the ground segment. Having said that, several factors constrain the design of this relation of rules:

- **Speed:** As it has already been mentioned, each node should be capable of handling at least **25** Mbit/s. Even though this doesn't mean that the design should be able to fit 25 Mbit/s of pure costumer data, it is still a strong requirement with many effects over the system. For example, some protocols are just too slow establishing the connection; those will be directly discarded.
- **Reliability:** The protocols have to assure that the messages are going to arrive to their destination. In order to achieve this, a routing protocol has to be used as well.
- **Security:** Messages are not just required to arrive to their destination but they also must be ordered and coherent when they reach the client. That is the reason why error control is taken into consideration very seriously along the design process.

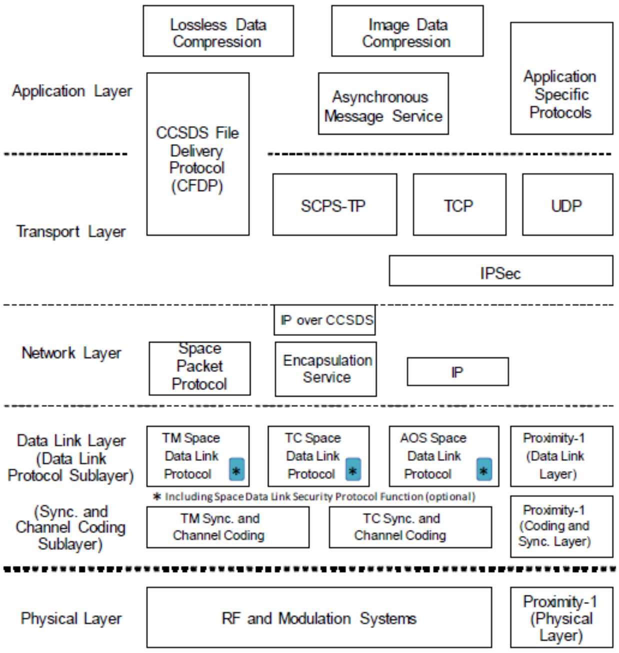


Figure 7.1.1: Protocols recommended by the CCSDS, classified in their respective OSI layer. Extracted from [7]

In order to define the protocols, the standards of the Consultive Committee for Space Data Systems will be followed. The CCSDS define a set of standards that become part of ISO and enhance operatibility between different satellites. The protocols recommended by the CCSDS and the possible combinations between them are shown in the Figures at the end of this section.

As far as *Astrea constellation* is concerned, the physical layer is already defined in detail in the **Satellite Desgin** part. Since the **Data Link layer**, the **Network layer** and the **Transport/Session layer** are of vital importance for the communication to work, the aim of the next sections will be to define the protocol that the constellation will be using for each one of those layers.

The presentation and the application layer are more client oreinted. In other words, if one client's satellite sends some data formatted with an unknown application protocol, *Astrea* will not be affected in any way. What astrea will do is add to this stream of bits, some headers, in order for the message to arrive in time to its destination. This methodology is undoubtedly positive for *Astrea* since the respobility of the application data will be solely for the customer.

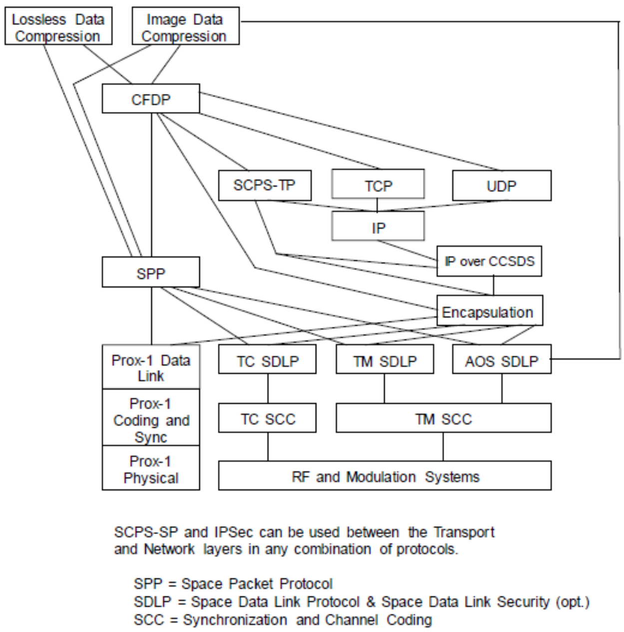


Figure 7.1.2: Possible combinations of the CCSDS recommended protocols. Extracted from [7]

7.2 Layer 2: Data Link

7.2.1 Functions of the DLL

The functions of the Data Link Layer are the following ones:[1].

- **Framing**
- **Adressing**
- **Synchronization**
- **Flow control**

7.2.2 Working procedure

There are five possible working procedures:

- Simplest Protocol
- Stop-and-Wait Protocol

[1]They are explaied in Annex II 7.1.1

- Stop-and-Wait Automatic Repeat Request
- Go-Back-N Automatic Repeat Request

They are explained in an extensive manner in Annex II 7.1.2. These working procedures had to be ordered according to its suitability for their application in Astrea constellation. To do so, and Ordered Weight Average (OWA) have been done. The decisive factors and its weights are:

- Efficiency: 40
- Time: 30
- Error correction:60

Then, the results of the OWA are the following ones:

Protocol	Efficiency	Time	Error correction	OWA
Stop-and-Wait Protocol	0	0	0	0
Stop-and-Wait ARQ	0	0	1	0,46
Go-Back-N ARQ	1	0	1	0.69
Selective Repeat ARQ	1	1	1	1

Table 7.2.1: OWA of the DLL protocols.

7.2.3 Protocols

The standards of the CCSDS will be followed in order to allow interoperability with other satellites such as the one of the client. The CCSDS has developed four protocols for the Data Link Protocol Sublayer of the Data Link Layer [11]:

- TM Space Data Link Protocol
- TC Space Data Link Protocol
- AOS Space Data Link Protocol
- Proximity-1 Space Link Protocol-Data Link Layer

These protocols provide the capability to send data over a single space link. TM, TC, and AOS can have secured user data into a frame using the Space Data Link Security (SDLS) Protocol.

CCSDS has also developed three standards for the Synchronization and Channel Coding Sublayer of the DLL:

- TM Synchronization and Channel Coding
- TC Synchronization and Channel Coding
- Proximity-1 Space Link Protocol—Coding and Synchronization Layer

TM Synchronization and Channel Coding is used with the TM or AOS Space Data Link Protocol, TC Synchronization and Channel Coding is used with the TC Space Data Link Protocol and the Proximity-1 Space Link Protocol—Coding and Synchronization Layer is used with the Proximity-1 Space Link Protocol—Data Link Layer.

After comparing the TC and Proximity-1 Protocols (more information at Annex II 7.1.3), the decision taken is to use the TC Space Data Link Protocol with the TC sync and channel coding together with the Space Data Link Security Protocol. The reasons for doing so are mainly:

- Security: Incorporing the SLDS authentication and confidentality is provided.
- More virtual channels: This feature allow more clients communicating with their satellites at the same time.

More information about the chosen protocols such as the amount of bits occupied by the header, its configuration and total lenght can be found at Annex II 7.1.4 and Annex II 7.1.5.

7.3 Layer 3: The Network

7.3.1 Functions of the Network Layer

The Network layer provides the following functions[2]

- **Routing**
- **Network fow control**
- **Package fragmentation**

[2]Explained in Annex II 7.2.1

- **Logical-physical adress allocation**
- **Message forwarding**

7.3.2 Protocols

According to the CCSDS the protocols to use can be divided into three different protocols[3]

- Main protocol
- Routing protocol
- Auxiliary protocols

The possible protocols are:

- Main protocol: Space Packet Protocol (SPP), Internet Protocol version 4(IPv4) and Internet Protocol version 6(IPv6).
- Routing protocol: Enhanced Interior Gateway Routing Protocol (EIGRP), Open Shortest Path First (OSPF) and Routing Information Protocol (RIP).
- Auxiliary protocols: Encapsulation service, IP over CCSDS (IPvC), Internet Control Message Protocol (ICMP), Internet Control Message Protocol version 6(ICMPv6), Internet Group Management Protocol (IGMP), Internet Protocol Security (IPsec) and Protocol Independent Multicast (PIM).

7.3.3 Protocol Selection

7.3.3.1 Choice of the main protocol

The choice of the main protocol will be between SPP, IPv4 and IPv6. To make the choice, it is important to take into account that the Astrea constellation is a network that can be of more than two hundred satelites, which will communicate point-to-point. Each node can be the source ,the destination or an intermediate node of a communication route. For this reason, the SPP has to be discharged, because it requires a Path ID. In Astrea constellation there are 29800 possible routes while the Path ID parameter only has 11 bits, that means that could work in a network with a maximum of 2048 routes. Between IPv4

[3] They are explained ad Annex II 7.2.2

and IPv6, is logical to choose IPv6 for its amplified benefits. Then, IPv6 will be the main protocol of the network layer. More details about this decision can be found in Annex 7.2.3.1.

7.3.3.2 Choice of routing protocol

First of all, RIP will be discharged because it has poor scalability and needs more time to converge. Then, the decision remains between EIGRP and OSPF. The difference between these two protocols is the way they update the routing table. With the EIGRP, 2000 entries are updated frequently while with OSPF, only 205 entries are needed to be updated. For this reason, OSPF is chosen. From more details about this selection, see Annex 7.2.3.2.

7.3.3.3 Choice of complementary protocols

The choice of which protocols include will depend on the main protocol of the network layer and the degree of services featured by the communication process.

Since IPv6 has been chosen, IP over CCSDS and Encapsulation Service are necessary. Additionally, ICMPv6 greatly expand the features of IPv6 such as flow control. Security features are already provided in the Data Link layer and, therefore, IPsec is not necessary. Also, no multicast features are required, so no multicast protocols will not be used.

7.3.3.4 Conclusion

It has been decided that IPv6 will be the network layer protocol, complemented with IPoC, Encapsulation Service and ICMPv6, and with OSPF as the routing protocol. In Annex II 7.2.4 the headers of the different protocols are shown.

7.4 Layer 4: Transport and Session

The objective of this layer is to provide and guarantee a reliable and cheap flow of the data.

The transport layer is responsible for process-to-process delivery, i.e, the delivery of a packet, part of a message, from one process to another. Two processes communicate in a client/server relationship.

More information about the functions of the Transport Layer and of the protocols recommended by the CCSDS can be found at Annex II 7.3.

7.4.1 Protocols

The protocols recommended to be used over this layer are:

- User Datagram Protocol (UDP)
- Stream Control Transmission Protocol (SCTP)
- Transmission Control Protocol (TCP)

7.4.2 Choice of protocol for the transport layer

The UDP has some disadvantages which make it not suitable for the purpose of the project, such as the fact that no reliabililty is guaranteed, for example, amongst others. The SCTP is designed mostly for Internet applications, which does not fit the goals of this project. Therefore, the only candidate suitable for the project is the TCP, Transmission Control Protoco. More information had to be compiled in order to know the full suitability of it to the project. The exposition of all its parameters is not shown in this report for its large length. See Annex II 7.3.3 for more information. As it has the required features that the project demands, it is the chosen protocol for this layer. Also, as it has been established during the deep research done, it is very recommended to use the extension SCPS, due to adaptation to space needs.

Chapter 8

Ground Segment Protocols

8.1 Ground Segment Protocols

8.1.1 Introduction

In the previous chapter the space protocols have been selected, so in this one the focus will be on the ground segment protocols. The information will be transmitted to the client using the Internet, so a part of the protocol is already established by the system. However, a secure protocol has to be defined above the Internet protocol to assure confidenciality to the client. The protocol used in the Internet is the TCP/IP protocol suite, that provides an end-to-end data communication specifying how data should be packeted, addressed, transmitted, routed and received. In the following lines the characteristics of the different available protocols that can be adapted to the needs of the project are presented. More information about the suitability of them can be consulted in Annex II 8.2.

8.1.2 Protocols

The protocols are the following ones:

- File Transfer Protocol (FTP)
- Secure Shell (SSH)
- Simple Mail Transfer Protocol (SMTP)
- Hypertext Transfer Protocol (HTTP)
- Transport Layer Security (TLS)

- Hypertext Transfer Protocol Secure (HTTPS)

8.2 Delivery of the data method

At first, it has to be take in account that this layer provides the plataform in which the client will make contact with the service. At this point, not only the technical criteria should be considered, but also how do the service is presented. It has to be found a friendly use method for the client keeping the technical efficiency.

Analazyng the previous protocols, avoiding the techincal details of each one, there are considered this 3 ways of working, with its advantages and drawbacks. In the following page the systems proposed are shown toguether with the decision, but in Annex II 8.3 the advantages and drawbacks of each method is extensively explained.

- **Web.** This system would be based in HTTP an implemented with the corrseponding security protocols in order to ensure the privacy of the data. In this case the client wolud entry with its computer a https adress where he/she wolud sign in with an account. When the user is verified, the client could request to download informaton of his satellite.
- **Mail**. This method would be implemented over a SMTP with the corresponding security protocols. If the client wants to download data of his satellite, he/she would have to send a mail specifing the request. Then the client will receive an email with the information.
- **Application**. The idea is that the cient would operate in his computer with this software, and when he/she want to upload or download something, the program would use a secure internet channel to transfer the information. This system wolud be implemented over a FTP or a SSH. For using this method it has to be implemented a plataform for the client use.

The chosen method to deliver the data is using an application that would be instaled in the client's computer and where data could be extracted. This is a secure, efficient and user-friendly solution. The application will ensure a high security of the data and a robust access to it. Moreover, for the point of view of the Astrea team, the efforts in mainting a computer application are less than in other cases such as the Mail, where automatization is more difficult to implement.

This system could work with a FTP or with a SSH. Both would work properly in the system and have very similar characteristics, but SSH is more secure than FTP, so the system would be ruled by a SSH protocol.

Chapter 9

Design of the Ground Segment

9.1 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.

9.2 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.

9.2.1 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 [REF TO ANNEX of matlab codes. Chapter to be determined. Section to be determined], while the study of localization can be found in [REF TO ANNEX III. Chapter 9. Section 1].

9.2.2 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 9.2.1: 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.

9.3 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.

9.3.1 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 [12]. The fees can be obtained from [13] and [14]. The frequency allocation can be found in [15].

9.3.2 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 [16]. The fees might be estimated using [17].

9.4 Annual costs

9.4.1 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 [REF TO ANNEX III. Chapter 9. Section 2].

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€

9.5 Initial investment

9.5.1 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 [18] and [19].

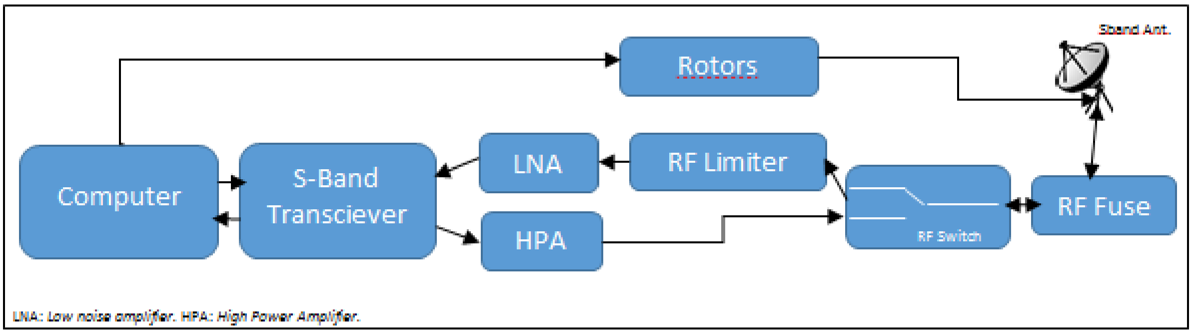


Figure 9.5.1: Equipment needed for S-band communications.

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 [20]. beginfigure[H]

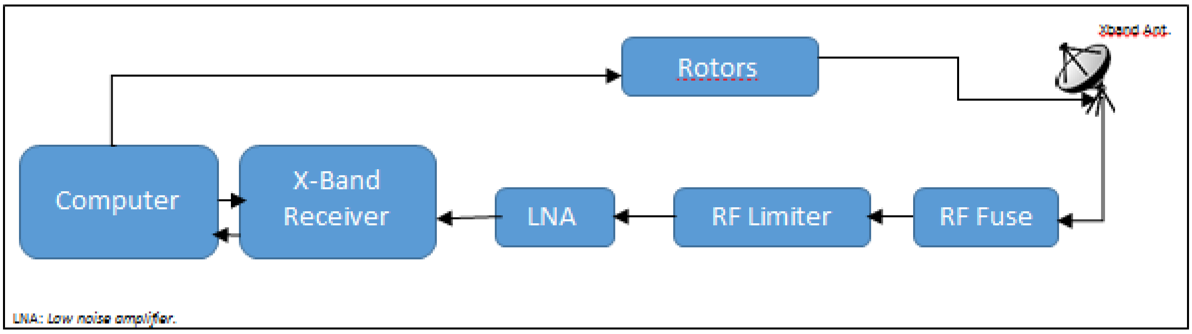


Figure 9.5.2: Equipment needed for X-band communications.

9.5.2 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 [REF TO ANNEX III. Chapter 9. Section 3]. 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€

9.6 Renting of a Ground Station

There are a lot of ground stations spared all over the world. In [REF TO ANNEX III. Chapter 9. Section 4], a list of the most important Ground Stations and their spacifications can be found.

9.6.1 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.

9.6.2 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.

9.6.2.1 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 9.6.1: 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.

9.7 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 [REF TO ANNEX III. Chapter 9. Section 5]. The conclusion of the decision process are explained in the following lines:

9.7.1 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 IV

Satellite design

Chapter 10

Satellite design

10.1 Structure and mechanics

The design and operation of a CubeSat is a complex process that must be completed keeping in mind the huge differences between all 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). Their correct functioning has to be ensured, especially the critical systems such as the solar arrays, batteries and antennas should be fully operational for at least four years.

10.1.1 Structure

The mission of the structure is to sustain and protect all the electronic devices carried by the satellite. In order to ensure that all the electronic and mechanical systems can be mounted upon the structure, a high compatibility between these systems is required; therefore, the structure must be very 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.

10.1.2 Thermal protection

The CubeSat is vulnerable to suffer extreme temperatures while operating in space, both below zero and above zero. The thermal protection system consists of a set of layers (MLI) made of insulating materials and it aims 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. Furthermore, the thermal protection system should also dissipate the heat produced by the other systems.

Dunmore Aerospace has been chosen to provide us its MLI product. The product, **Dunmore Aerospace Satkit**, has been designed for small satellites operating in LEO and it will provide the CubeSat with the protection required during operation.

10.1.3 Options chosen for the structure and thermal protection

The options chosen are presented in the table 10.1.1.

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

Table 10.1.1: Options chosen for the structure and thermal protection

10.2 Electrical Power System

The Electrical Power System (EPS) of the satellite must provide and manage the energy generated efficiently in order to have all the systems operating under typical conditions during the lifetime of the mission. The role of this system is to control and distribute a continuous power to the Cubesat, to protect the satellite against electrical bus failiures and to monitor and communicate the status of the EPS to the on-board computer. The EPS of the Cubesat is, probably, the most fundamental requirement of the satellite, since its failure would result in a mission failure.

10.2.1 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 total power required is 52W and it has been estimated considering that all the subsystems are working under typical working conditions.

10.2.2 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 fully efficient for at least four years (this is: it has to be ensured that the missions does not run out of power for at least four years).

Every cubesat will come with at least 4 deployable solar panels (manufactured by **EXA-Agencia Espacial Civil Ecuatoriana** providing it with 67.2W of power, approximately, to supply peak demands. Note that these 4 deployable solar panels are a basic requirement. If more space is available on the faces of the satellite, additional panels can be placed providing extra power.

10.2.3 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 these systems have different power and energy needs, the power management system has to be highly compatible and must have a high enough number of buses 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 and saving a lot of space.

10.2.4 Batteries

The role of the batteries is to provide the subsystems of the satellite 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 higher* than typical energy and power supply, since all the systems will not usually operate under peak conditions.

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.

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. If the satellite was in the dark during half of the period of the orbit, the estimated energy that it would need would be 50Wh. Thereby, the capacity of the batteries is more than enough to supply the energy required in the worst case scenario. Furthermore, 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.

10.2.5 Options chosen for the EPS

Finally, the options chosen are presented in the table 10.4.1.

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 10.2.1: Options studied for the Electric Power System

10.3 Propulsion Systems

The propulsion system is an important part of the satellite given the needs of the satellite to perform different maneuvers, such as reach the desired orbit after it has been deployed from the rocket, and to maintain the orbit and avoid falling to the Earth. The main parameters that have to be consider are thrust, total specific impulse, power required, weight of the propulsion system and its volume, since the size and weight of the CubeSat are very restrictive.

At the moment, the most used and modern thrusters for small 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 imply smaller accelerations than conventional propulsion systems.

The **BGT-X5**, a green monopropellant propulsion system, has been chosen as the thruster for the CubeSat. The high thrust and delta V that BGT-X5 provides has predominated at expenses of other variables as weight or specific impulse where others options were better. With this thruster 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 option chosen is presented in the table below 10.3.1.

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

Table 10.3.1: Option chosen for the propulsion system

10.4 Attitude and Orbital Control Systems

The attitude and orbital control system (AOCS) 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 the 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 different sensors. The AOCS can also be referred as the 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 to 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. (*wikipedia extract*, [**?**]).

Pointing to a specific direction is necessary for Earth observation, orbital maneuvers, maximizing solar power, and some scientific instruments. Directional pointing accuracy can be achieved by sensing Earth and its horizon, the Sun, or specific stars. Determination of a CubeSat's location 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*, [**?**]).

10.4.1 Orbital Control

Orbital control will be achieved as a combination of two systems. ADCS will orient the thrust (given by the propulsion system) and the operation will be controlled by the on-board computer. Mainly, the orbit control will be necessary to mitigate orbital debris effect on every satellite.

Taking into account several very restrictive variables: 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 and integrates also and On-Board Computer (OBC).

Finally, the options chosen are presented in the table 10.4.1.

System	Brand and model	Price per unit (€)	N. of units
ACDS	CUBE ADCS	15000	1

Table 10.4.1: Options studied for the AOCDS

10.5 Payload

The payload needs to provide a radio link with the client's satellite(s) for real time data relay with no less than 25MB/s of data rate. For achieving its purpose, the payload will consist on a pack of arrays of antennas and data handling computers.

The CubeSat will 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 used to establish the previous described links are regulated in [21] by frequency, bandwidth and type of communication . So, for the **Space to Ground link** frequencies from **70MHz** to **240GHz** will be used; 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.

10.5.1 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.

10.5.1.1 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]

The antenna that will be equipped in the CubeSat is the patch antenna manufactured by **AntDevCov**. The satellite will come with 8 of this type of antenna; 4 of them will be placed on each side face of the CubeSat and they will occupy a 1U face and the other 4 of them will be placed on the top and the bottom.

10.5.2 Payload Data Handling Systems

Every satellite 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 nor 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 (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.

The PDHS selected for the mission is the **Nanomind Z7000** because of its two 667MHz processor can handle a high data payloads and processit at a high velocity velocities, reducing in last term delay between communications.

10.5.2.1 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

The **NanoCom TR-600** by **GOMspace** has has been selected as the inner-satellite tranceiver since its manufacturer offers it in combination with the **NanoMind Z7000**. 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 makes it a great choice for Inter-Satellite communication.

The **SWIFT-XTS** has been selected as the space-to-ground transceiver. Its high bandwidth will make possible higher communication data rates.

10.5.3 Options chosen for the payload module

Finally, with the aim to clarify all the information of this section, the chosen systems and components are presented in the table 10.5.1.

System	Brand and model	Price per unit (€)	N. of units
Antenna	Patch antenna AntDevCo	18000 1st (7000 others)	8
Transceiver	NanoCom TR-600	8545	3
Transceiver	SWIFT-XTS	5500	1
PDHS	NanoMind Z7000	5000	1

Table 10.5.1: Options chosen for the payload

10.6 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 CubeSat 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.

10.7 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**		

Additional information regarding the systems used in the CubeSat and the final configuration, calculations and decision taking can be found on the appendix.

10.8 Budget

System	Cost/unit (€)	Total cost (€)	N. of units
Structure and mechanics			
Structure	3900	3900	1
Thermal protection	1000	1000	1
Total		4900	
Electrical 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	50000	50000	1
ADACS	15000	15000	1
Total		65000	
Total CubeSat		276635	
Total estimation CubeSat		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 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 (such as the purchase of additional solar arrays), is added to every satellite. Furthermore, this extra 20000€includes the costs of transport to launch site. The payload's variable expenses stand for cable, adaptors and several minor components that have to be purchased.

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 *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.

Part V

Financial and Other Considerations

Chapter 11

Financial Study

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

Warren Buffett

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 12 years. The decision of this amount of years has been done taking into account that every 5 years there is a re-launching of the whole constellation, which means a considerable increase in the costs. In order to have a good view of the whole project, it has been decided that at least two full cycles had to appear complete (years 0 to 4 and 5 to 9) and also to see the tendency it was recommended to add some more time. This is the reason why the studied is carried on for 12 years.

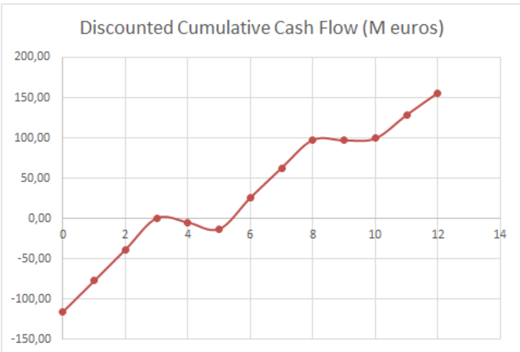
From an analysis carried on by Astrea's team, it has been found that the anual estimation of the demand can be approximated by 295650000 Mbits hired per year. An economical study is going to be performed using that estimation as the sales. The determination of the price of Astrea'a service is made upon this feasibility study.

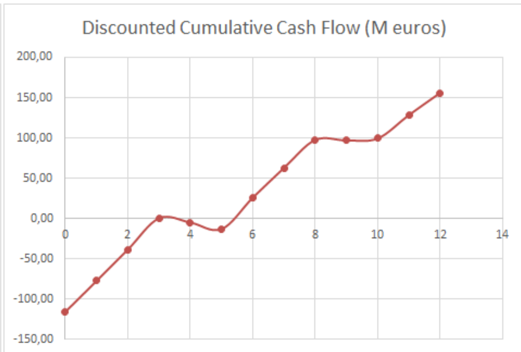
When performing the feasibility study, all the costs must be taken into account. This means that all the costs derived from the different departments (communications, satellites, launching, ...) are considered, as well as other costs such as the engineering hours, which were stated in the Project Charter and the Gantt Chart. All those costs have already been explained. Nevertheless, some last costs which must be taken into account are the administration costs and the insurance service. The estimations of those costs are of 259000 and 2245320 €per year, respectively. For further details on the studies mentioned, see the Financial Annex. All the costs can be found in the Budget as well.

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 11.0.1: Feasibility Study

- Pay Back Time: The Cumulative Cash Flow is plotted next. It can be seen that the Pay Back Time is between years 2 and 3. Assuming linearly distributed, it will be of 2.68 years. The value of the first PBT (2.63 years) found seems reasonably acceptable, taking into account that this project requires a great budget, as all space projects do, due to its own nature. In year 4, the satellites that are going to be launched in year 5 are made. This means an increase in the cost, due to the fabrication and assembly. Moreover, in year 5, there is also a increase, even more significant, because of the re-launching itself of those satellites. This is what causes that interval of time in which the Cumulative Cash Flow does not increase, but even decreases a bit: that year, the income is slighly smaller than the re-launching costs. A second Pay Back Time might be found of 5.05 years. Since the investment was done in year 4, it means an actual Pay Back Time of 1.05 years, for the second investment. In years 9 and 10, the same process takes place again: the fabrication and re-launching of the new satellites. Nevertheless, profit is high enough at this point so that the Cumulative Cash Flow does not get to negative values again.





- Updated Pay Back Time. The Discounted Cumulative Cash Flow has already been plotted too. It can be seen that the Updated Pay Back Time is between 2 and 3 years. If again linearly distribution is assumed, the Updated Pay Back time will be of 2.99 years. This value seems reasonably acceptable, because of the nature of the project, within the space sector, a very demanding and expensive one. This time, because of the significat costs during year 4 (building and assembling of satellites) and especially year 5 (re-launching of satellites), added to the fact that the discount rate is now being taken into account to, it makes the Discounted Cumulative Cash Flow negative again, as can be seen in the graphic. Therefore, there will be another Updated Pay Back Time for the inversion done during years 4 and 5, exactly at year 5.33. Taking into account that this inversion was made during the year 4, it turns out to be an actual UPBT of 1.33 years. There won't be a third UPBT. In years 9 and 10 there is a big investment again, but by then, the profit is high enough so as to make it possible for the Discounted Cumulative Cash Flow to not become negative any more.

- Break Even Point. By changing manually the parameter "Number of Mbits hired" of first year, it is found that the Break Even Point is of 18375000 Mbits (with this value of Mbits hired the first year, the Cash Flow is approximately 0). This means that under no account there can be less Mbits hired that that, because otherwise the Cash Flow would be negative and the Cumulative Cash Flow, negative at beginning since first year is fully just 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. Thus, the cash flow will be positive and consequently there will be a time in which there is benefit.

- Net Present Value. From the table, it can be immediately seen that the Net Present Value (for the period of time studied, of 12 years) is of +156.01M€. 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. It is clearly positive, which indicates that the project earnings generated by the investment exceeds the costs. In other words, it will be profitable. The Net Present Value coincides with the value of the Discounted Cumulative Cash Flow of the twelfth year.

- Internal Rate of Retorn. 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. The studied carried on uses a 6% as the discount rate, and this value gives a positive Net Present Value, as has been already explained. By manually changing the discount rate, it can be achieved that the Net Present Value gets to 0, or very closely. This value of the discount rate is the Internal Rate of Return, and is used to evaluate the attractiveness of a project. In this case, it has been found that the IRR is of 26.80%. The higher the IRR is, the more desirable is to undertake the proeject. In any case, it must be greater than the actual discount ratio. In this case, not only the IRR is greater, but it is actually a very high rate. Therefore, the project is very attractive, feasibly speaking.

The Pay Back Time and the Updated Pay Back Time are of approximately 3 years, quite acceptable in the space sector. The Net Present Value is really great, as well as the Internal Rate of Return. Moreover, the estimation of demand is far higher than the Break Even Point. In conclusion, all the parameters analysed happen to indicate the same thing: the investment is strongly recommended and the project is feasible. Nevertheless, it must not be forgotten than this study is done upon some important assumptions, which provide a certain degree of uncertainty. Nonetheless, the results of the study have been so positive that a slight variation of the hypothesis done could not lead to a unfeasible situation.

Chapter 12

Environmental Impact Study

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. The entire study can be seen at the FINNANCIAL ANNEX, since in this document are only exposed the conclusions of it. The study is focused in these 3 aspects:

- **Ground stations**. It is concluded that they would not represent any environmental impact if they are in a place where they not interfere with the normal behaviour of any ecosystem.
- **Satellites**. The study analyses the impact of the fabrication and the performance. The satellites are certificated by all the corresponding regulations and there is no significant environmental impact during the fabrication. On the other hand the satellites are designed to be burned out in the atmosphere in a period of 5 years, and they would not be orbital waste.
- **Launching**. The launching is the most critical part in environmental terms and it is depth analysed. Basing in a Environmental Assessment done by the Ministry for the Environment of New Zealand [**?**] it is concluded that there is no significant damage in the environment at short term.

In conclusion the project is environmentally viable.

Chapter 13

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.

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 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 different factors, where CubeSats could be in danger. The launch stage is one of the most important, because it is where the mission has more probability to fail. In the Financial Annex it 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.

Once the constellation is in orbit, CubeSats can find dangers how colliding with other satellites, which is not probably due to the distance between satellites is around hundred of miles, or with space debris, whose movement 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*.

Finally, the ground stations will have its operator, to control the operation of the installation, and a security system, to avoid intrusions.

Chapter 14

Bibliography

[1] D.A. Vallado. *Fundamentals of Astrodynamics and Applications.* Springer-Verlag New York, 3 edition, 2007.

[2] Howard D. Curtis. *Orbital Mechanics for Engineering Students*, volume 3rd editio. Elsevier, 2014.

[3] Yongjun Li, Shanghong Zhao, and Jili Wu. A general evaluation criterion for the coverage performance of LEO constellations. *Aerospace Science and Technology*, 48:94–101, 2016.

[4] J.G. Walker. Some Circular Orbit Pattern Providing Continuous Whole Earth Coverage. *Journal of the British Interplanetary Society*, 24:369–384, 1971.

[5] J.G. Walker. Continuous whole-Earth coverage by circular-orbit satellite patterns. *Royal Aircraft Establishment. Technical report 77044*, 1977.

[6] Vladimir A Chobotov. *Orbital Mechanics.* 2002.

[7] CCSDS. *Report Concerning Space Data System Standards - Overview of Space Communications Protocols.* Number CCSDS 130.0-G-3. 2014.

[8] Jorge Cantero Gómez. Communication link design at 437 . 5 MHz for a nanosatellite. (June), 2013.

[9] Muhammad Zubair, Zaffar Haider, Shahid a Khan, and Jamal Nasir. Atmospheric influences on satellite communications. *PRZEGLĄD ELEKTROTECHNICZNY (Electrical Review)*, 87(5):261–264, 2011.

[10] Lorenzo Luini, Carlo Capsoni, Carlo Riva, and Luis David Emiliani. Predicting total tropospheric attenuation on monthly basis. pages 1–6, 2015.

[11] CCSDS Secretariat. Overview of Space Communications Protocols. (CCSDS 130.0-G-3):43, 2014.

[12] UK Government. https://www.ofcom.org.uk/___data/assets/pdf_file/0023/47138/ofw564.pdf.

[13] UK Government. https://www.ofcom.org.uk/___data/assets/pdf_file/0020/27461/fees.pdf.

[14] Government UK. https://www.ofcom.org.uk/___data/assets/pdf_file/0038/66899/fees_for_grant_of_

[15] UK Government. https://www.ofcom.org.uk/___data/assets/pdf_file/0028/44875/ofw_241_mar_201

[16] Canada Government. http://www.ic.gc.ca/eic/site/smt-gst.nsf/eng/sf00023.html.

[17] Canada Government. http://www.ic.gc.ca/eic/site/smt-gst.nsf/eng/sf01027.html#a4.

[18] ISISpace. https://www.isispace.nl/product/full-ground-station-kit-s-band/.

[19] IQ wireless. http://www.iq-wireless.com/images/pdf/SLINK-PHY-Datasheet.pdf.

[20] Dartcom. http://www.dartcom.co.uk/files/DartcomXBandEOSSystemBrochure.pdf.

[21] Secretaría de Estado de telecomunicaciones y para la sociedad de la información. Cuadro Nacional de Atribución de Frecuencias (CNAF) revisado 2015. pages 3–110, 2015.

[22] Craig Clark. Constellations of CubeSats are revolutionising how we use satellites.