Departament de Projectes d'Enginyeria
UNIVERSITAT POLITÈCNICA DE CATALUNYA

Secció de Terrassa

Degree: Aerospace Engineering
Course: Engineering Projects

Title and acronym of the project:
Cubesat Constellation
Astrea

Contents: Report Attachments

Group: G4 EA-T2016

Delivery date: 22-12-2016

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Part I

ANNEX 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 As can be seen, the XY plane coincides with the plane Equatorial with the X axis pointing in the direction of the vernal equinox [1]. The Z axis correspond the axis of rotation of the earth and points to the north (following the right-hand rule).

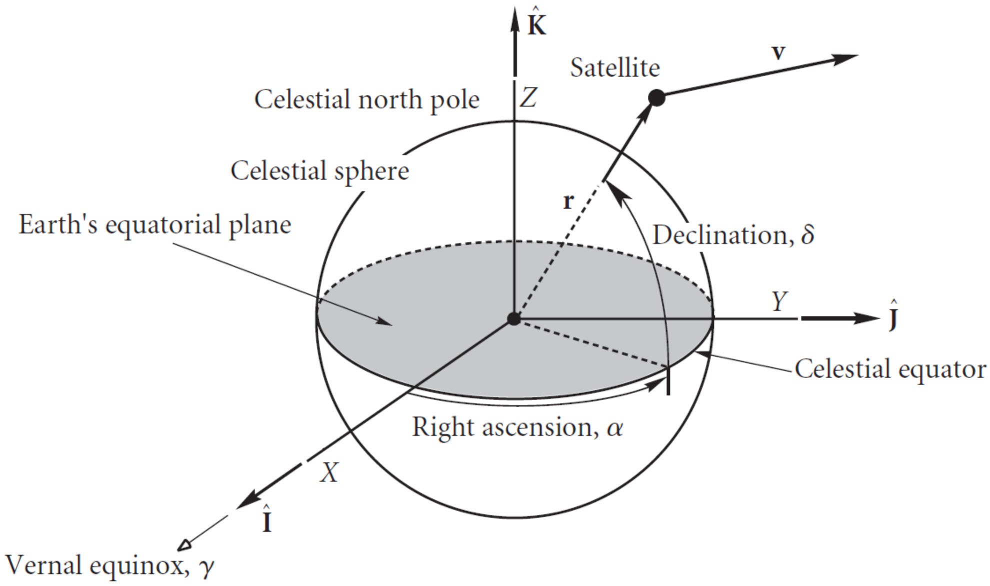


Figure 1.0.1: Geocentric-equatorial frame. Extracted from [1].

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

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

1.1 Keplerian Geometry

The *Classical Orbital elements*, also known as the *Keplerian elements* as an attribution to Johannes Kepler, are six independent quantities which re sufficient to describe the size, shape and orientation of an orbit. This set of elements are shown in the figure 4.3.6 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.

Circular	$e = 1$
Elliptical	$0 < e < 1$
Parabolic	$e = 1$
Hyperbolic	$e > 1$

Table 1.1.1: Eccentricity values depending on the shape of the orbit

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

angle called the eccentric anomaly.

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

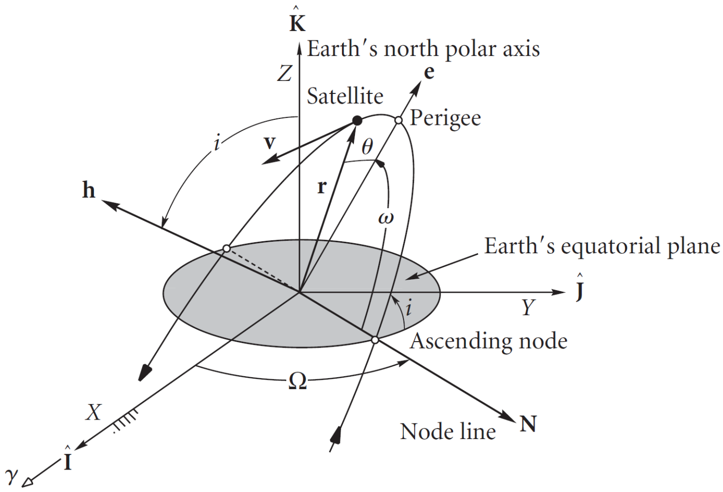


Figure 1.1.1: Geocentric-equatorial frame and the Classical Orbital Elements. Extracted from [1].

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 **??** 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

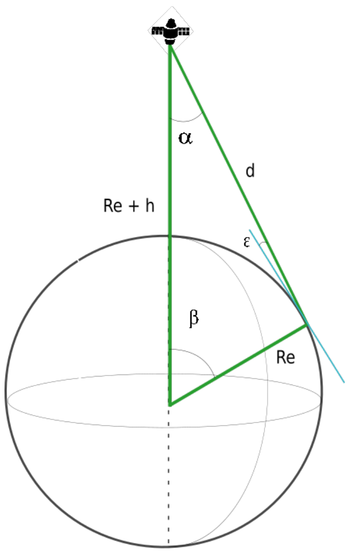


Figure 2.1.1: Single satellite coverage geometry

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.

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

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.

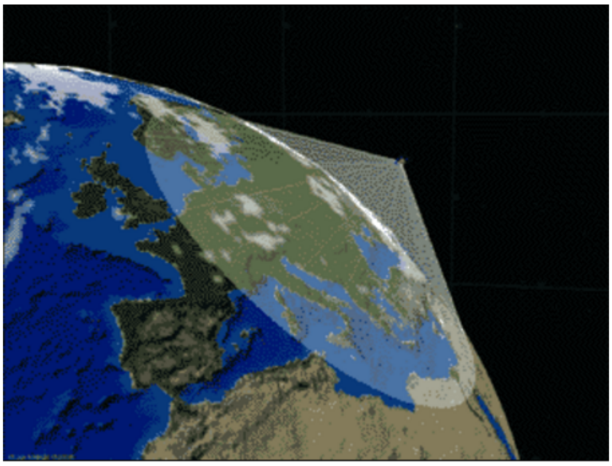


Figure 2.2.1: Elevation angle cone. Source: NOAA

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. First, a description of the different atmospheric conditions will be defined. Then, we will relate these to our bandwidth in order to analyse if they must be taken into account when communicating with ground stations. [elec2013cantero]

– **Atmospheric gases:** water vapour and oxygen absorptions; important when frequencies are above 3 GHz. More information [64] and [07328546]

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

– **Cross Polarization Discrimination:** direct consequence of both terrestrial links and rain. Related to non-spherical rain drops which have a polarization rotated towards the component of the major axis, and hence may attenuate a signal wave.

– **Scintillation** is a rapid fluctuation in signal amplitude at low elevations.

– **Radio Refracting Index:** for elevation angles below 3 degrees (especially those below 1 degree) and depending on the latitude of our satellite we may find big signal losses due to the resulting differential ray bending.

– **Ionosphere layers:**

D layers: 60-90km. Considerable signal absorptions for 10 MHz and below, with progressively less absorption at hihger frequencies and oblique incidencies.

E layers: 90-150 km. Absorptions relevant for frequencies lower than 10 MHz, although for sporadic E propagations this value may be increased to 50MHz.

Sporadic E layers: Reflections of radio waves in this thin-cloud small layer may reach to frequencies up to 225MHz. These layers are usually formed following the E layers altitudes.

F layers: 150-500 km and higher. No absorptions or reflections for these layers. The F2 region allows the longest communication paths, above 210km of altitude.

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.

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.

The minimum elevation angle is applied in low latitude regions for constellations based on polar orbits whereas this value is also applied out of the low-latitude region for inclined orbit constellations [a general evaluation criterion]. The minimum elevation angle is a specific value which is equivalent to the maximum elevation angle needed to fulfill coverage at a given latitude, considering that the distance between planes is maximum at the equator and that it is reduced for higher latitude positions.

This elevation angle is maximum in a Walker Delta constellation when the latitude is equal to the orbital inclination angle[a general evaluation criterion]. This means that the limiting restrictive elevation angle that we need in order to fulfill global coverage is defined at latitude equal to 72 degrees, which is the inclination of our constellation. Otherwise, we can define a constant elevation angle that will apply to the equator, which will then be, for this model, the restrictive condition.

Accordingly, the approach considered is that of a constant elevation angle to fulfill global coverage at 20 degrees. This implies that our constellation is configured and distributed in order to optimize coverage both at the equator and at the maximum elevation angle latitude. This value has been contrasted and discussed considering the atmospheric conditions and analysing experimental data, which contemplates also the rotation of the Earth among others.

This constant elevation angle model will be very useful in order to analyse and calcute the distribution of the constellation. Nevertheless, we need to describe in an accurate way the minimum elevation angle respect to the latitude. This is why a different model must be approached.

Thus, we need to describe the elevation angle respect to the latitude of our constellation taking into account all considerations above. First, for a latitude of 0 degrees the value of the minimum elevation angle will be of 10 degrees. In our model we have considered that this value was of constant 20 degrees, so in fact we have redundant global coverage. At latitudes between the equator and 45 degrees our second model increases linearly to 15 degrees. From 45 to 60 degrees the elevation angle also increases linearly to 22 degrees. Then, from 60 to 70 tha value increases highly reaching a peak at 70 degrees, where the elevation angle will be of 30 degrees. Finally, from 70 to 80 degrees this model is reduced linearly to 15 degrees, and from 80 to the north and south poles it falls to 0 degrees. This is a simple model that will guarantee global coverage, especially at the latitudes of our ground stations

For the distribution of ground stations we need to guarantee that these will be covered either by one satellite or two at any given moment. As discussed before, the model used was based on a constant 20 degree constant elevation angle. However, for this last model that we have described - which is more realistic - we obtain more coverage than for the constant model except for those regions next to the peak. The most restrictive latitude is now 60 degrees - where all the ground stations are set - and has a 22 degree restriction of the angle of elevation, which is higher than the constant model described previously. These facts imply 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. As discussed before, our first model considered a constant 20 degree elevation angle instead of the 22 degrees that now must be corrected. For the previous model coverage was well established with margin. For the latter, the margin has decreased but coverage is still complete. Note: each orbit could be reduced by a number of satellites per plane, but this would endanger the correct and stationary working of the constellation. 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 confirmed as function of the frequencies of the signals and related to the inclination of the orbits.

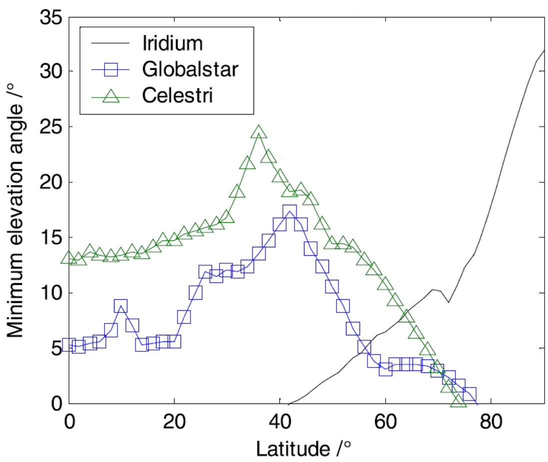


Figure 2.2.2: Minimum elevation angle as function of latitude. Source: [a general evaluation criterion]

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:

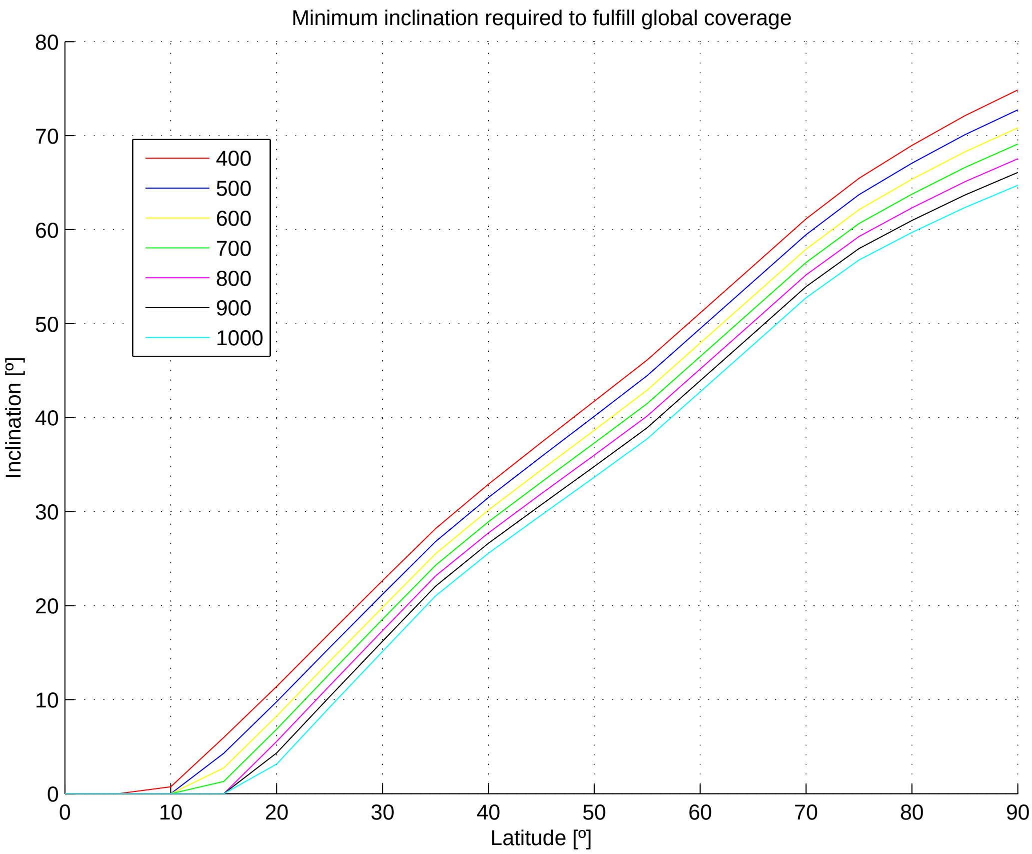


Figure 2.3.1: Minimum Inclination to provide coverage at different latitude for different orbit apogees.

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.

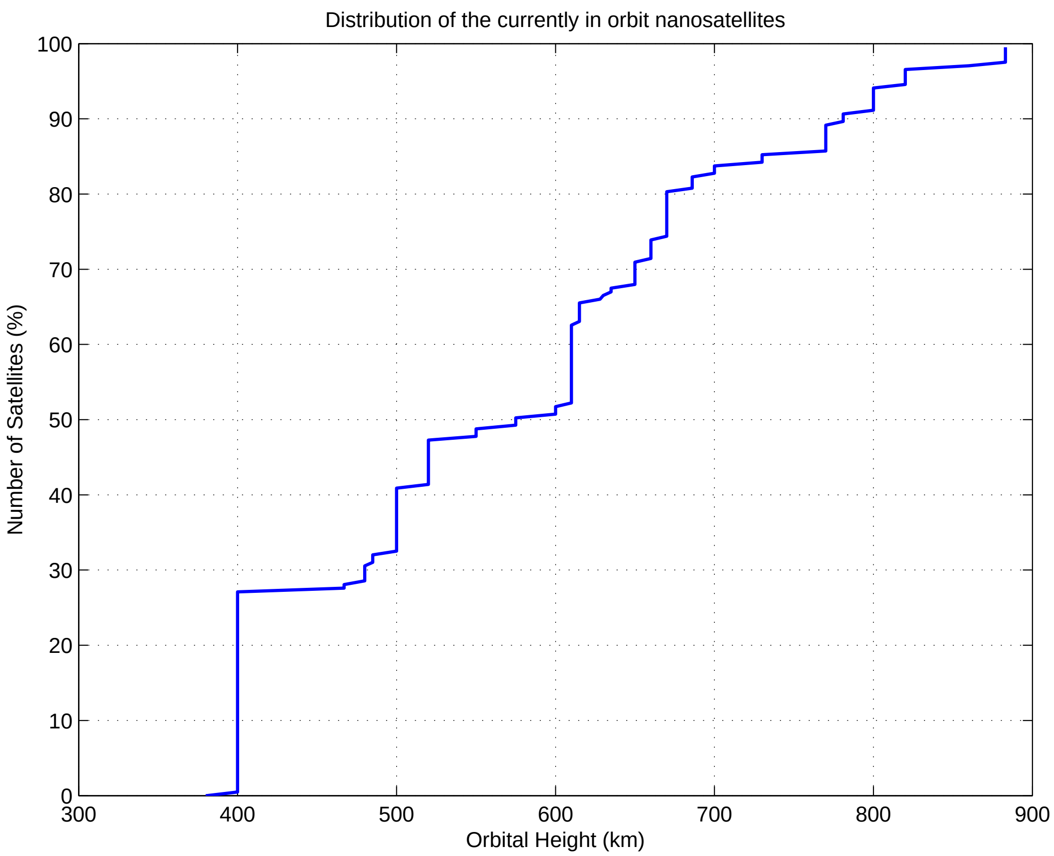


Figure 2.5.1: Distribution of the currently in orbit nanosatellites.

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;

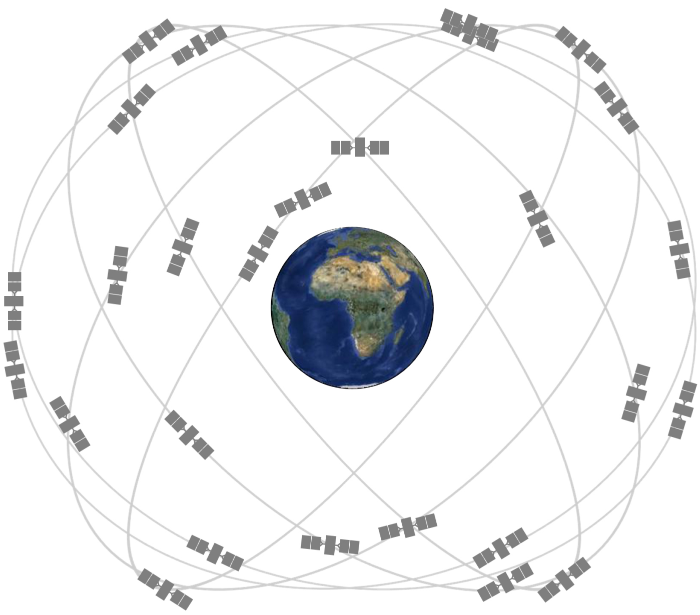


Figure 3.1.1: Distribution of the expanded 24-slot GPS constellation. [?]

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;

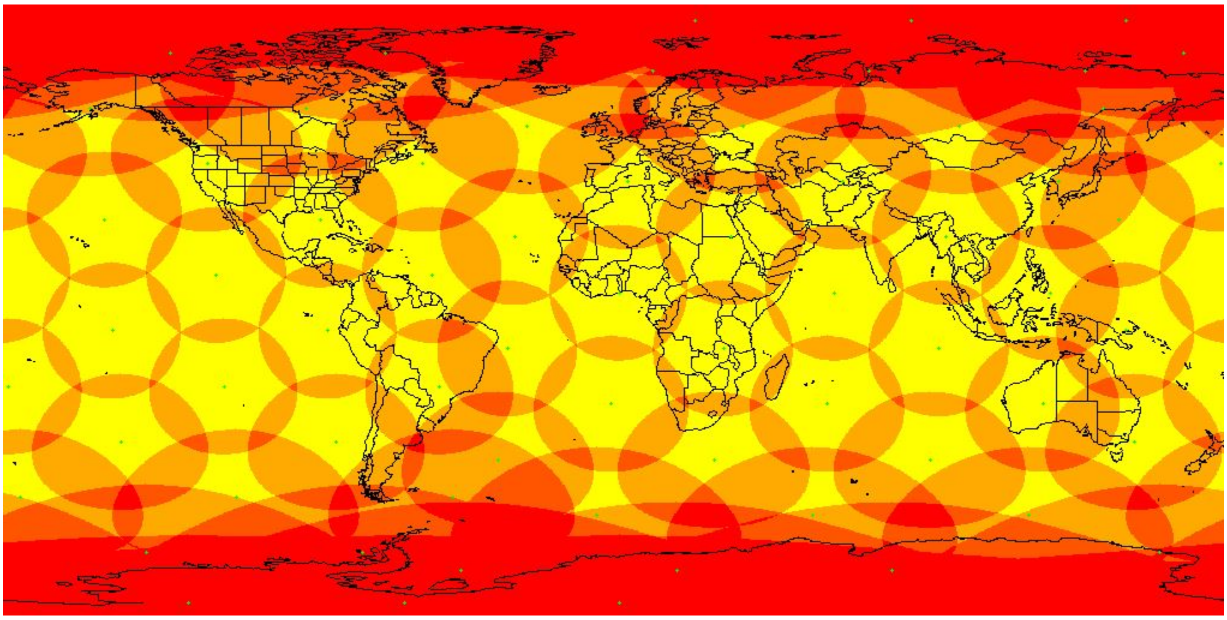


Figure 3.2.1: Distribution of the 66 Iridium constellation satellites. Generated using [?]

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:

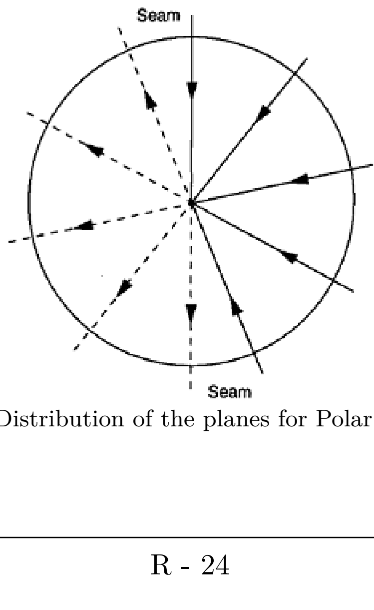


Figure 3.2.2: Distribution of the planes for Polar Orbits design.

3.2.3 The Streets of Coverage Method

This Street of Coverage Method is obtained from [4]. 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 [°]
D_0	Space between antiparallel planes [°]
ε	Elevation angle [°]
λ_{street}	Street of coverage Width [°]
λ_{max}	Maximum footprint Radius [°]

Table 3.2.1: Streets of Coverage Method main variables

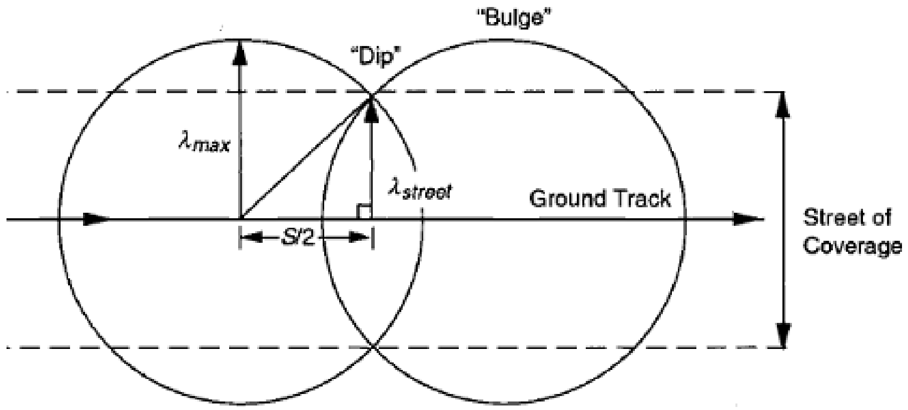


Figure 3.2.3: Single plain street of coverage. The footprints of the satellites superpose leading to a street. [?]

From the figure it can be inferred:

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

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

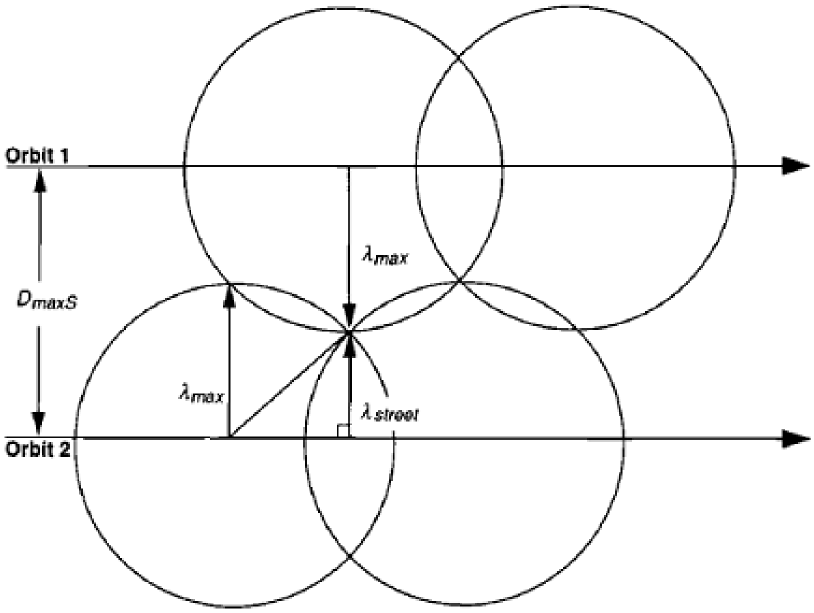


Figure 3.2.4: Two plains streets of coverage. An optimum phasing needs to be obtained. [?]

From this point of view, in general:

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

n For the antiparallel planes:

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

And the overall relationship between planes sums:

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

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

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

3.2.4 Results of Streets of Coverage

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

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

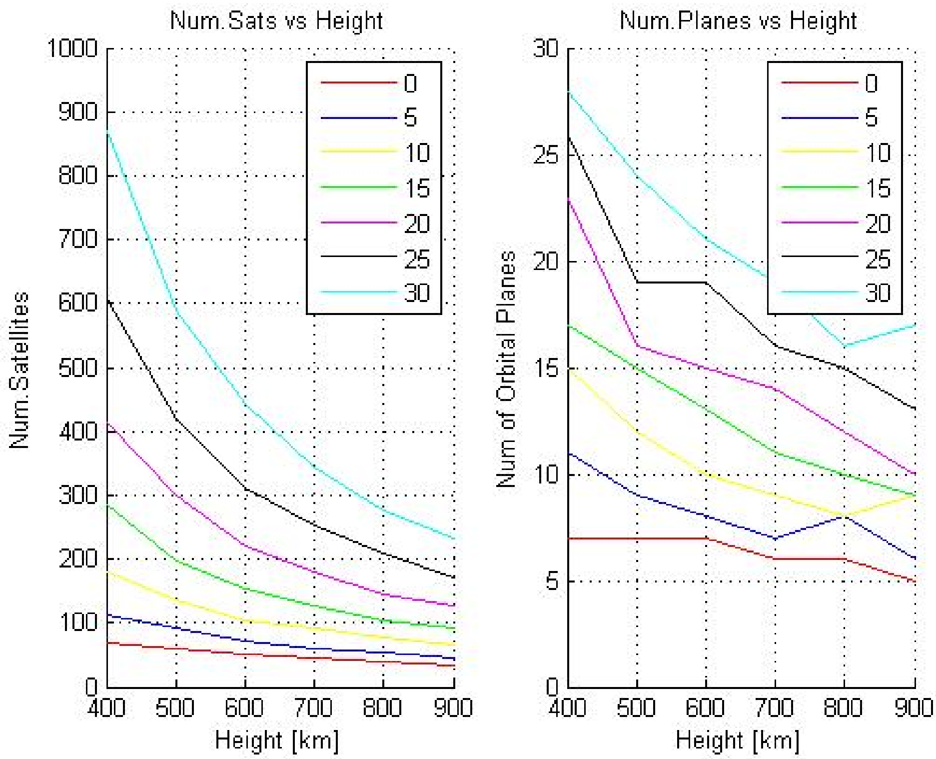


Figure 3.2.5: Variation of number of satellites for different heights and elevation angles

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

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

.

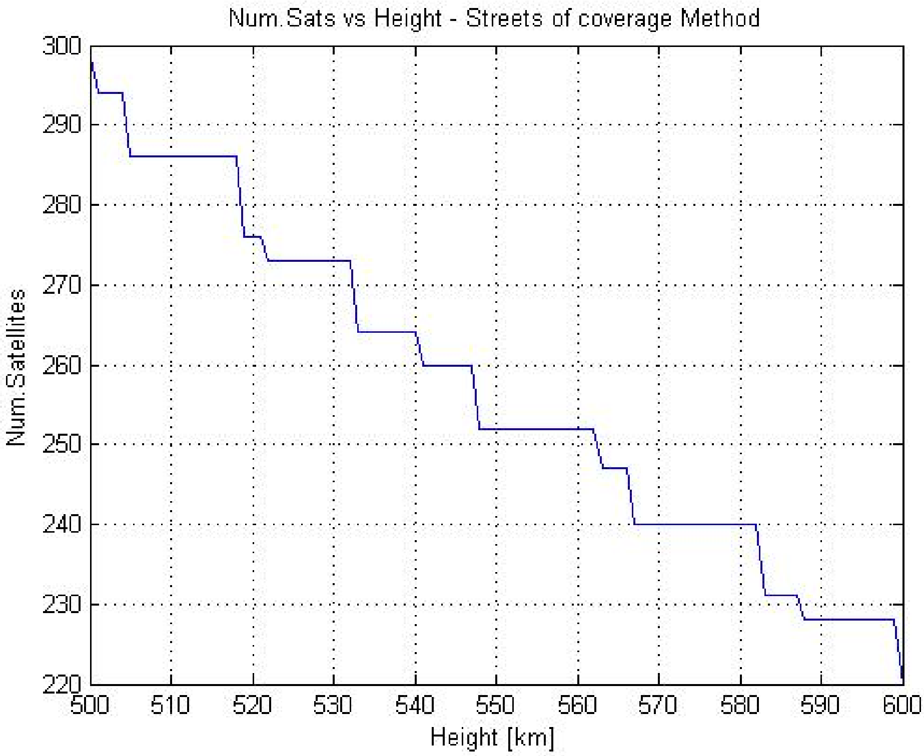


Figure 3.2.6: Variation of number of satellites for different heights between 500 and 600km.

Conclusion

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

3.3 Walker-Delta Constellation

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

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

3.3.1 Full Walker-Delta Constellation

3.3.1.1 Characteristics

A typical delta pattern has the following characteristics:

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

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

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

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

- All orbits have equal inclinations δ to a reference plane. If this plane is the Equator (it usually is), then the inclination δ equals the orbital parameter inclination i [2].

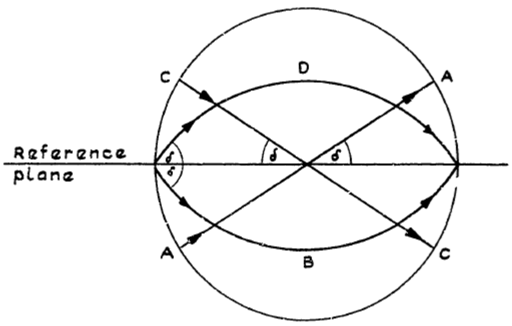


Figure 3.3.1: Definition of the inclination δ. Extracted from [2]

- 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).

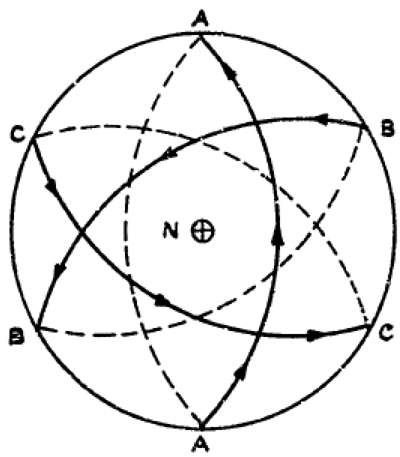


Figure 3.3.2: Delta pattern as seen from the North Pole. Extracted from [3]

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

3.3.1.2 Notation

J.G. Walker developed a notation to define this constellations with only 4 parameters [3]:

$$i : T/P/F$$

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

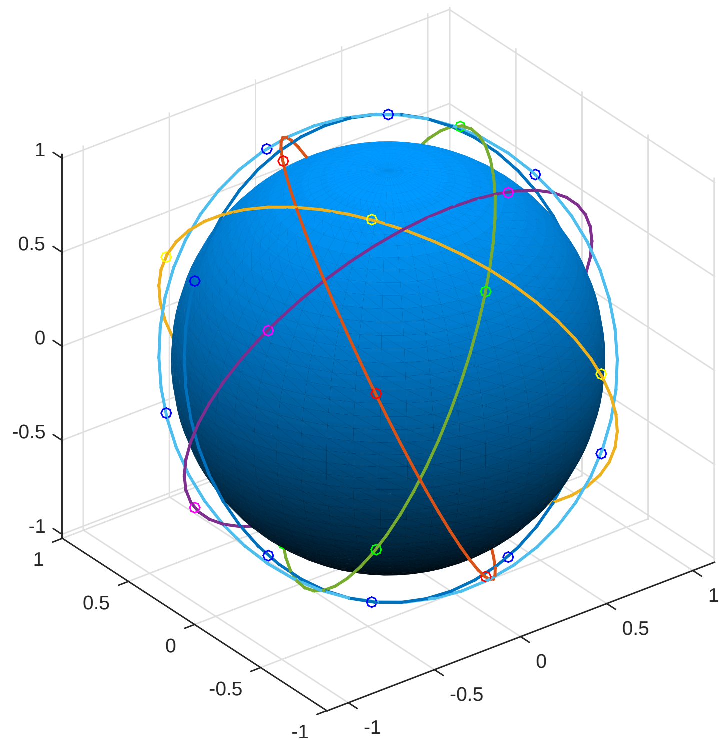


Figure 3.3.3: Delta pattern 65°: 30/6/1

3.3.1.3 Coverage

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

3.3.2 Semi Walker Delta Configuration

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

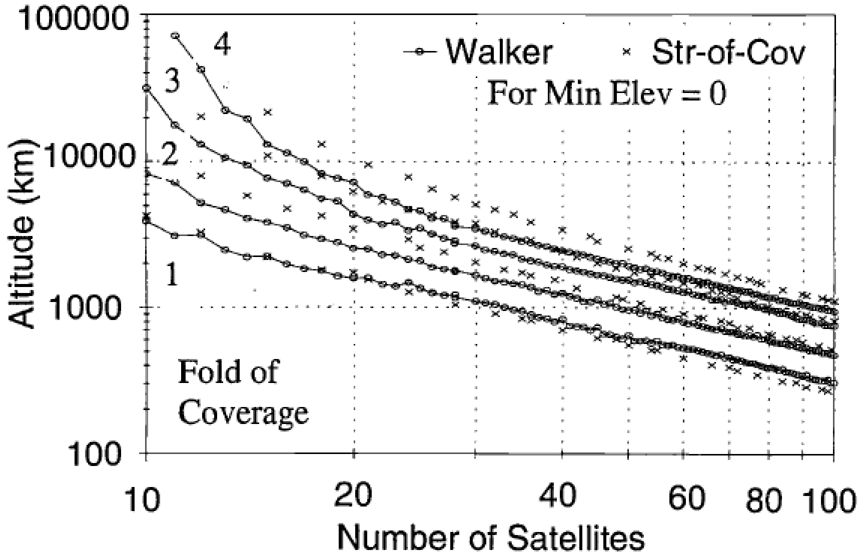


Figure 3.3.4: Minimum altitude for continuous global coverage. Comparison between polar patterns and Walker delta patterns. Extracted from [4]

having some inconvenients.

3.3.2.1 Advantages

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

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

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

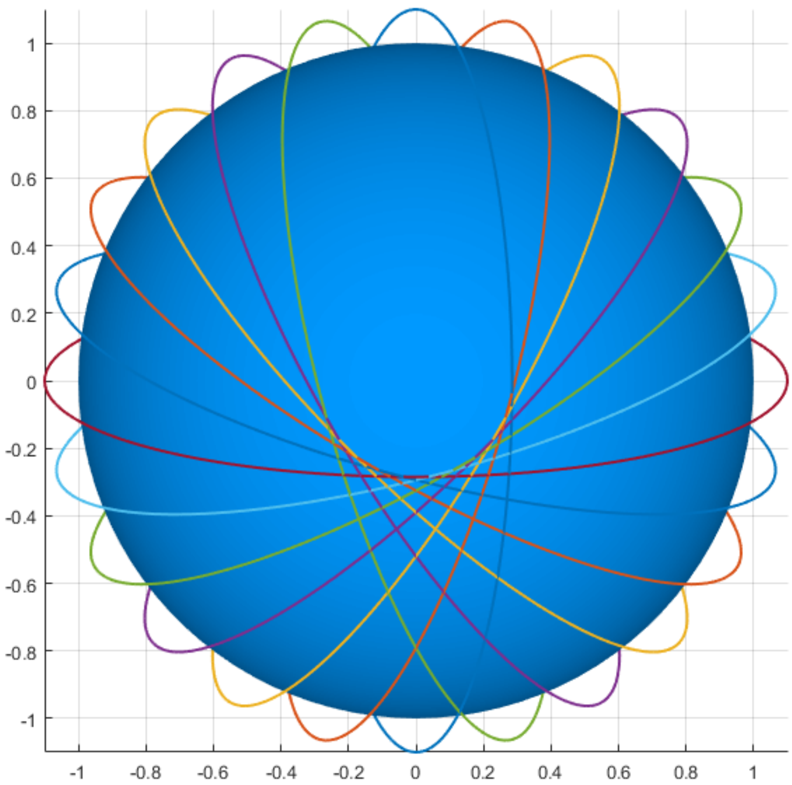


Figure 3.3.5: 12 plane SWDC. Note the gap and the equidistant planes

3.3.2.2 Disadvantages

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

3.3.3 Other Walker Delta Configurations

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

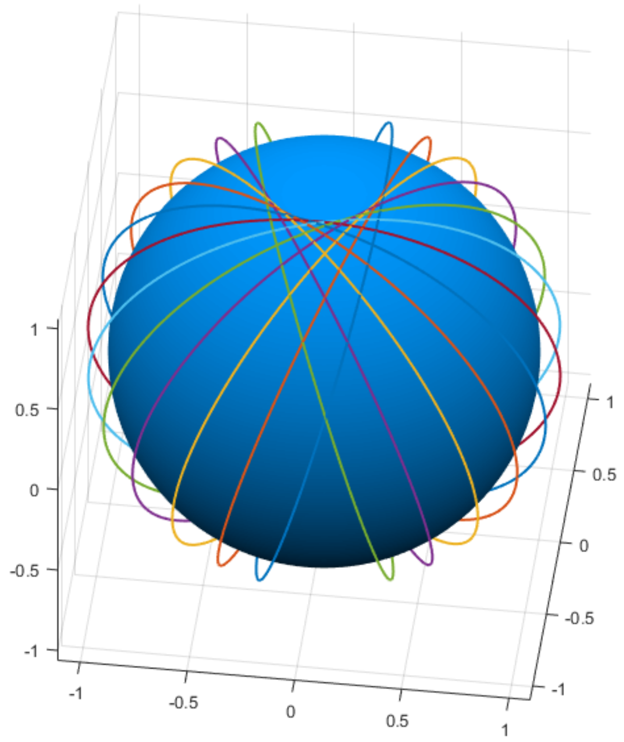


Figure 3.3.6: This geometry distribution induces a large anti-symmetric gap

3.3.3.1 SWDC including an additional polar orbit.

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

3.3.3.2 Mixed Walker Delta.

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

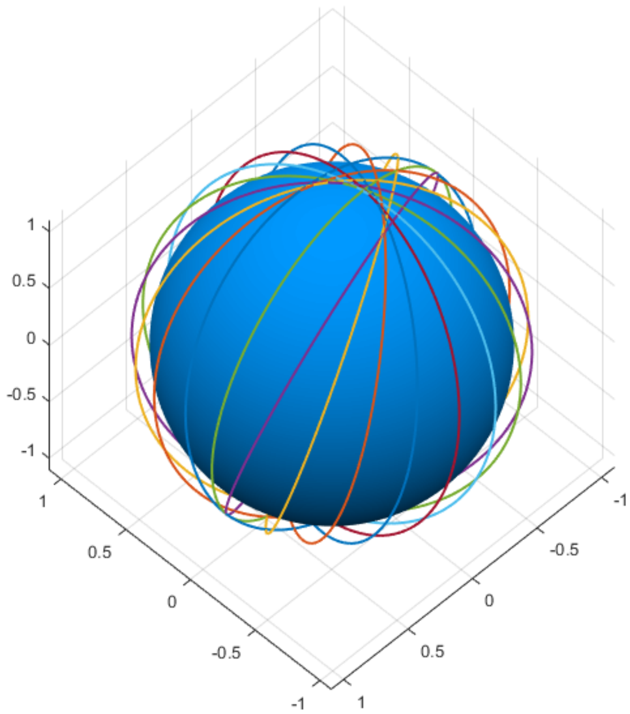


Figure 3.3.7: Added polar orbit to the 11 plane based SWDC

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

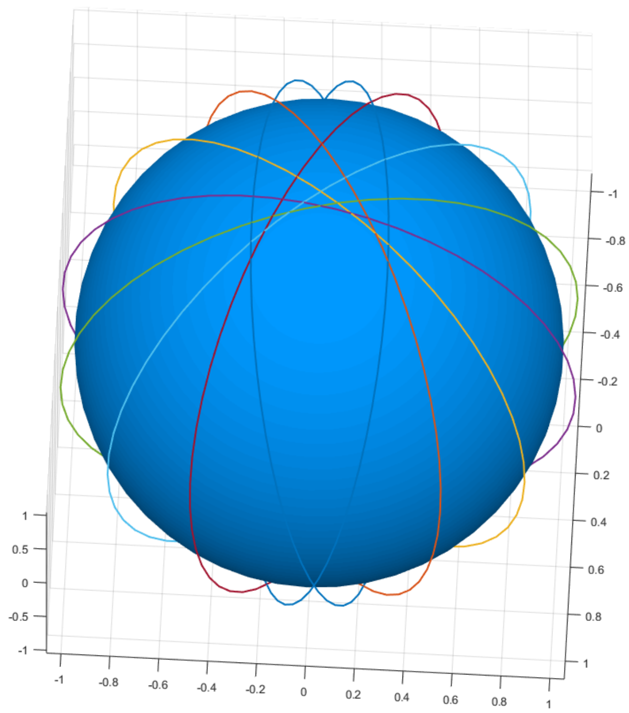


Figure 3.3.8: 8 plane based MWDC generated for 210 degrees

3.4 Testing Method

3.4.1 Introduction

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

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

3.4.2 Method Bases

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

Coverage Testing Method Variables	
typeC	Type of constellation
ε	Elevation angle [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

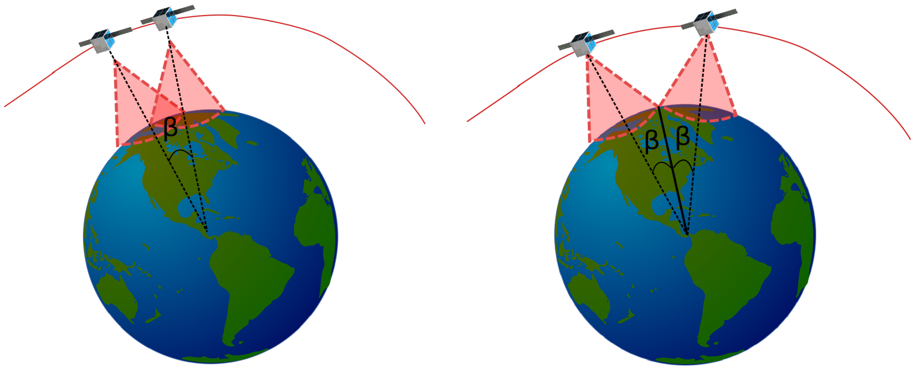


Figure 3.4.1: Geometrical conditions needed to fulfill global coverage.
On the left: Condition between satellites of different planes.
On the right: Condition between satellites of the same plane

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

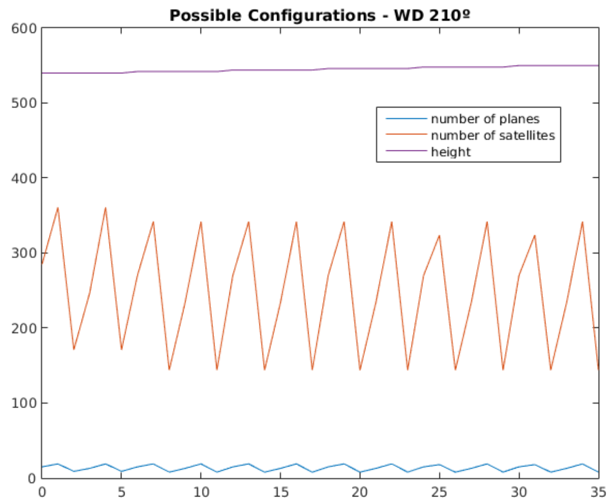


Figure 3.4.2: Possible satellite configurations for a 210° Walker Delta configuration

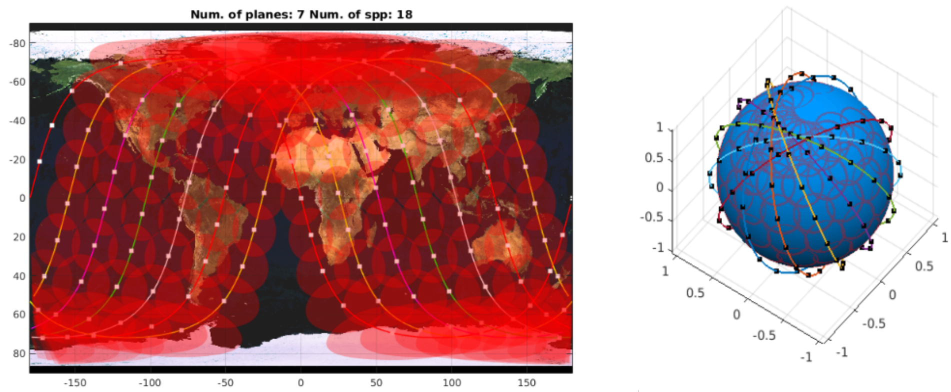


Figure 3.4.3: Ground track and spherical representation for a 180° Walker Delta configuration

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

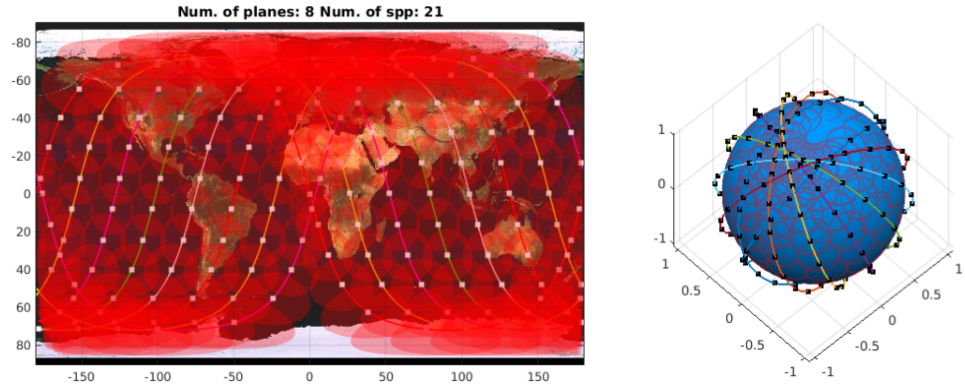


Figure 3.4.4: Ground track and spherical representation for a 210° Walker Delta configuration

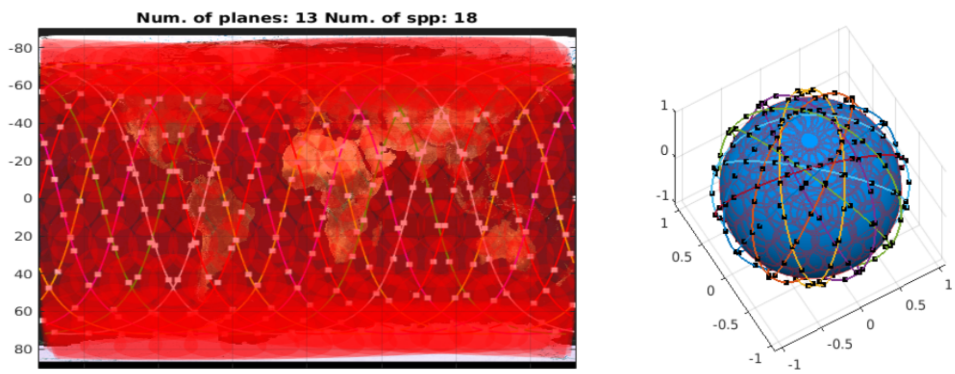


Figure 3.4.5: Ground track and spherical representation for a 360° Walker Delta configuration

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 [?]

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

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

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

Sources of perturbation:

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

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

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

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

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

4.1.2 Gravity Potential of Earth

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

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

General Legendre associated polynomials developed Gravitational Potential

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

General Legendre polynomials developed Gravitational Potential

For Earth, the J_n coefficients are the following:

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

Given this distribution, the only sognificant term J_2.

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

Aproximated Gravitational Potential

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

- $\Delta a = 0$

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

4.1.3 Atmospheric Drag

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

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

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

Modelling the Atmosphere

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

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

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

Where:

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

Table 4.1.1: Exponential Atmosphere Model main Variables

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

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

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

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

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

4.1.4 3rd Body Perturbations

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

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

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

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

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

Table 4.1.2: Third Body Perturbations Coefficients

4.1.5 Other Perturbations

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

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

4.2 Significant Perturbations

Propagation Algorithm

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

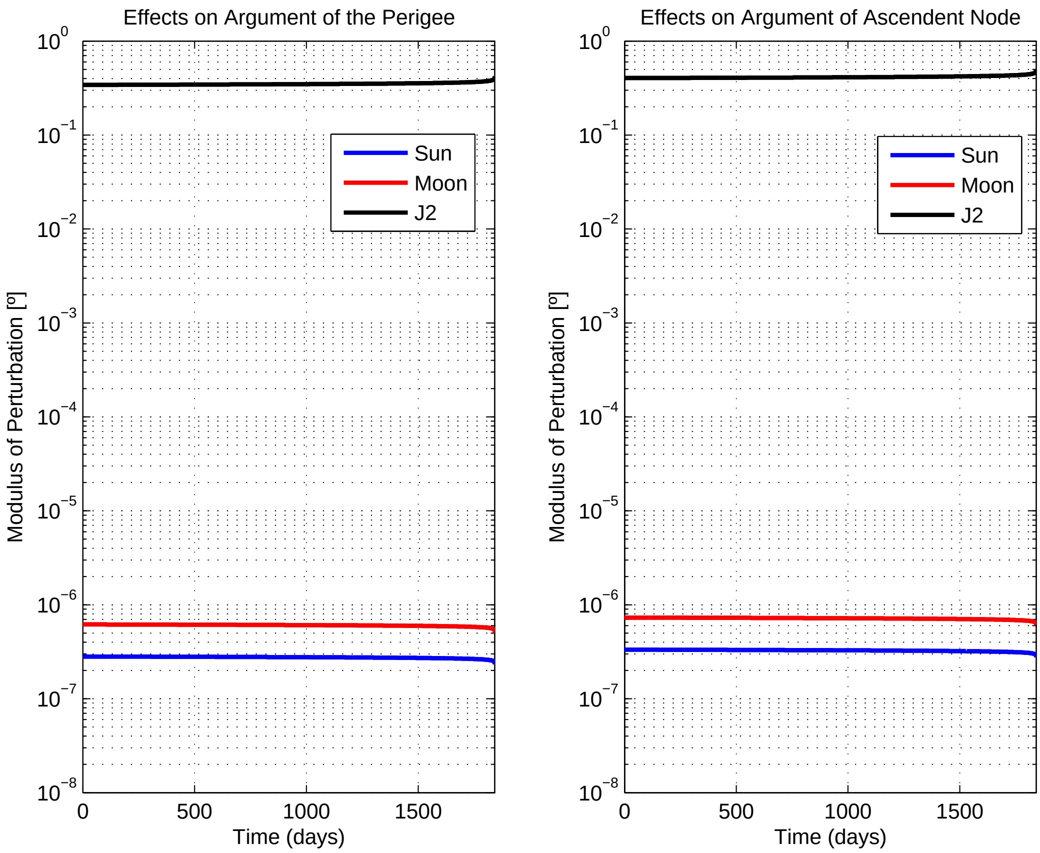


Figure 4.2.1: Logaritmic plot of the modulus of the increases in Angular Arguments of the orbit

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

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

In conclusion

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

4.3 Orbit Decay

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

4.3.1 Effects on the Ascention Node

4.3.1.1 Introduction

Due to the non sphericity of the Earth, two deviations exist in terms of perigee and ascendent node. This pertürtions 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.

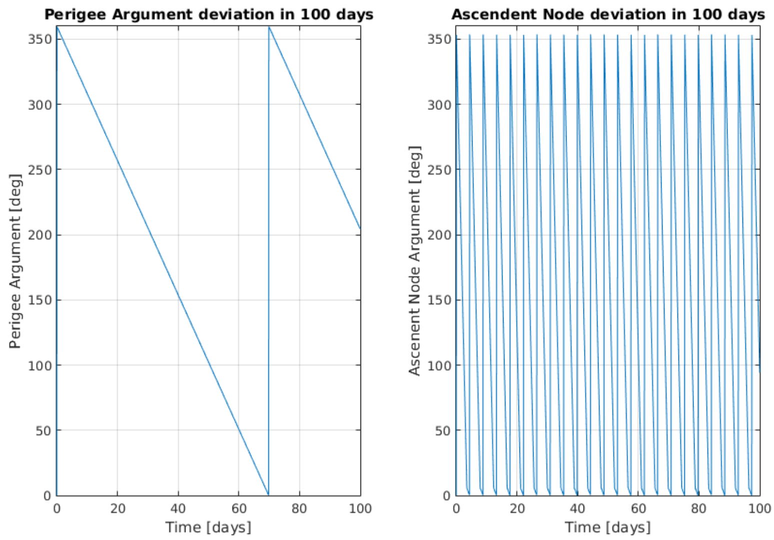


Figure 4.3.1: Ascention node perturbation
On the left: Perigee deviation in terms of time.
On the right: Ascending node deviation in terms of time

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.

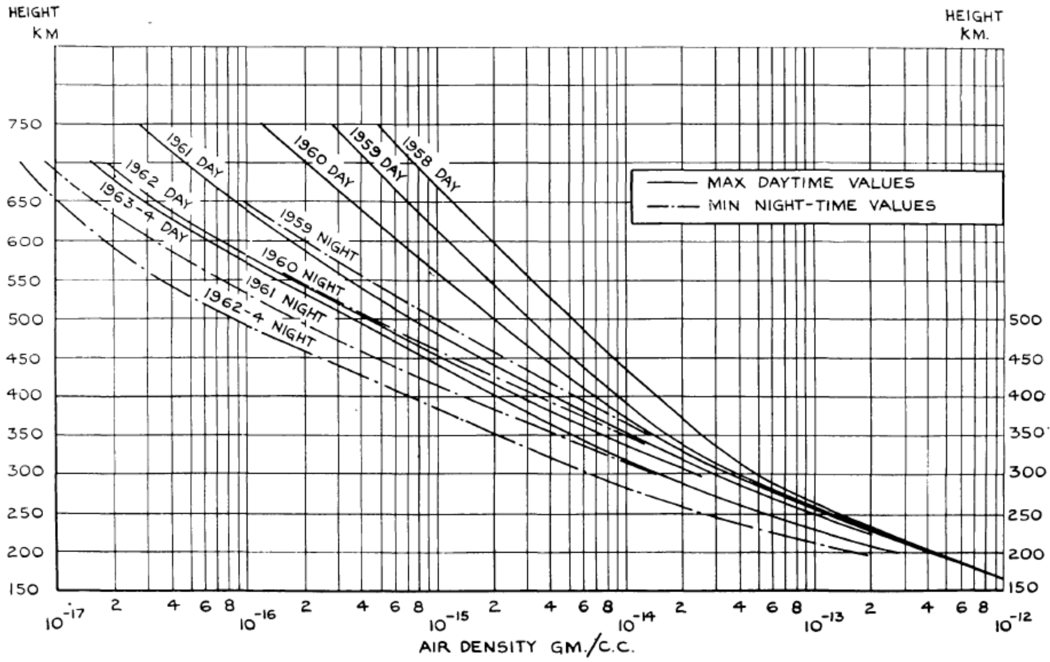


Figure 4.3.2: Deviation of densities in the upper atmosphere due to the 19th solar cycle

At 550 km:

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

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

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

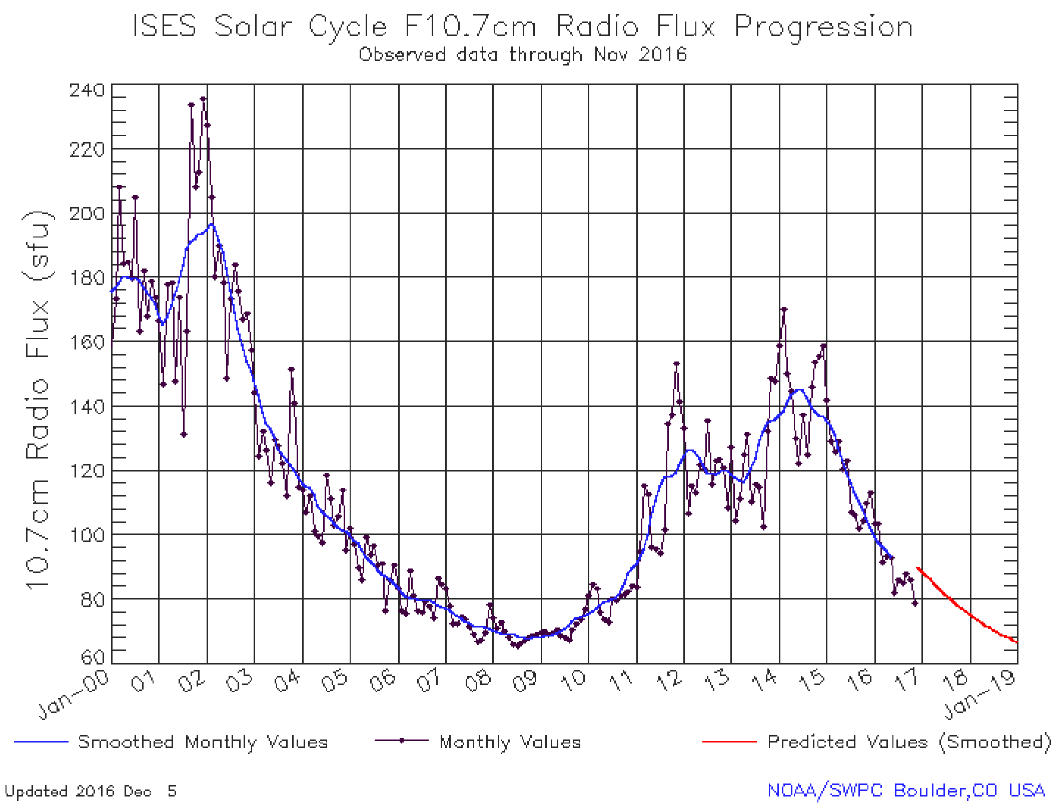


Figure 4.3.3: Measured intensities of the 23rd and 24th solar cycles. Source: NOAA

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

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

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

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

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

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

4.3.3 Orbital Decay Propagation Results

4.3.3.1 Introduction

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

4.3.3.2 Drag Computation Algorithm

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

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

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

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

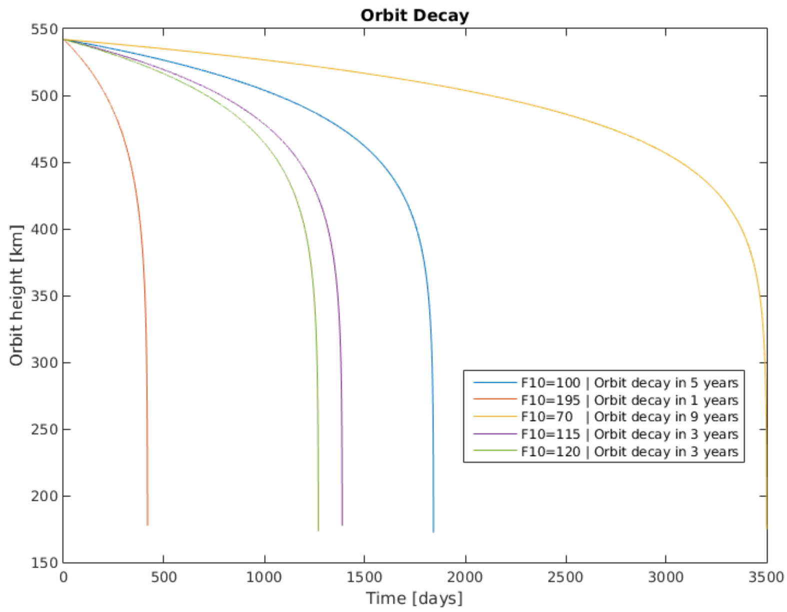


Figure 4.3.4: Orbit Decay computed for several values of

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

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

4.3.4 Dynamic Orbit Decay Computation

4.3.4.1 Introduction

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

Cowell's method: This is the simplest method since it does not require any assumption or approximation. It is based on quantifying the accelerations produced by the perturbations and adding them to the dynamic equation of a Keplerian orbit (see Orbit Design: Chapter 1 equation 1.2.3) leading to:

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

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

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

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

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

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

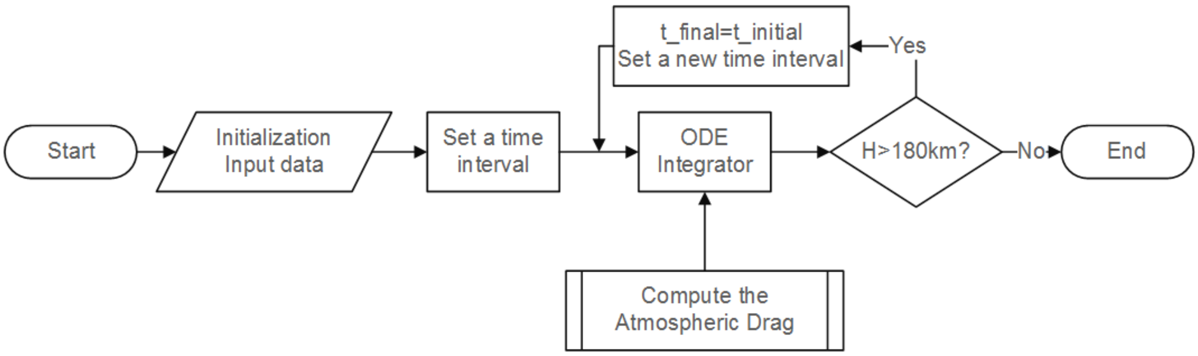


Figure 4.3.5: Algorithm of resolution used to solve the orbital propagation.

The disturbances that the routine includes are:

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

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

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

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

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

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

4.3.4.2 Results

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

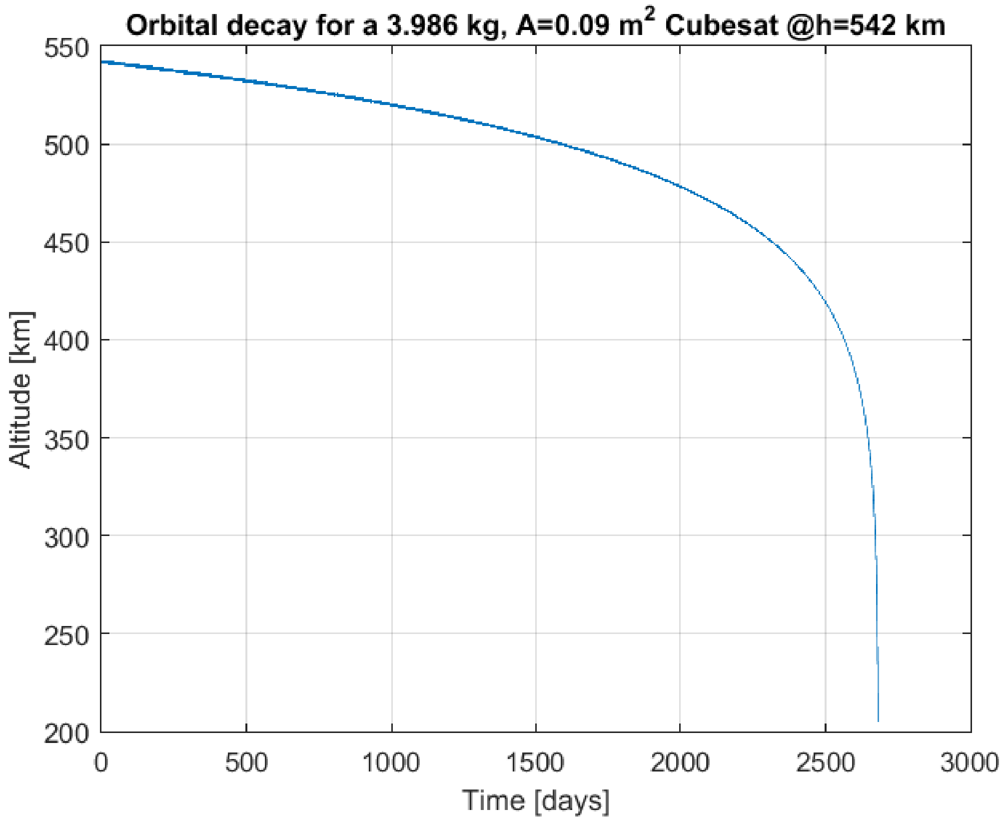


Figure 4.3.6: Orbital decay of the satellite.

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

4.4 Orbital Station-Keeping

We will study:

- Increased height
- Thrusters

4.4.1 Raising the orbit height to increase Lifetime

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

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

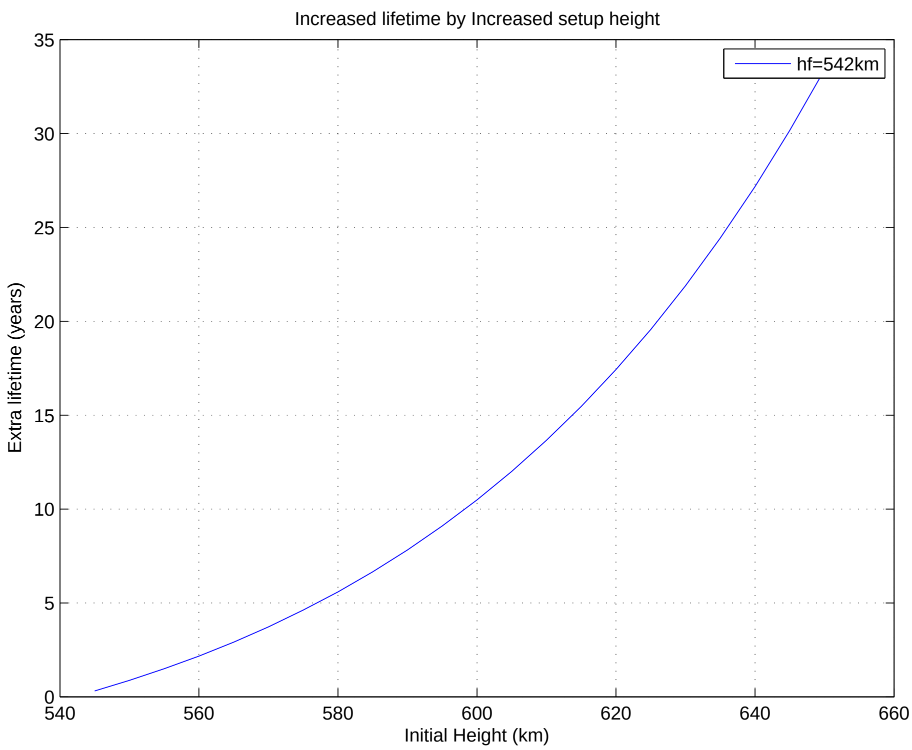


Figure 4.4.1: Increase in the Lifetime obtained by setting the constellation in a higher orbit

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

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

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

4.4.2 Using Thrusters to increase Lifetime

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

4.4.2.1 Energy equation

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

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

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

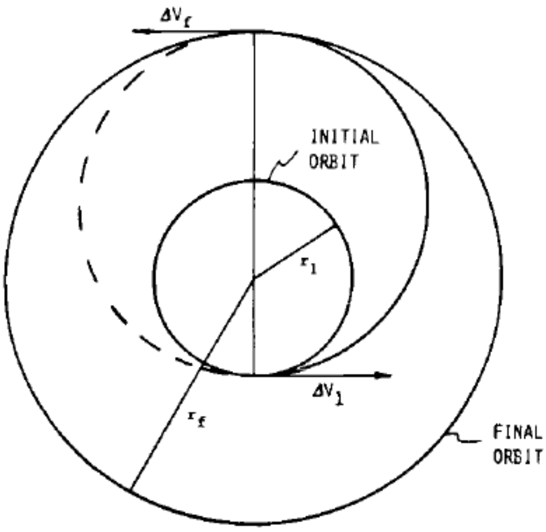


Figure 4.4.2: Hohmann transfer. Extracted from [4]

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

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

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

4.4.2.2 Delta-V

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

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

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

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

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

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

The velocities in the perigee and apogee are:

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

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

Therefore the velocity increments are:

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

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

4.4.2.3 Time

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

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

4.4.2.4 Propellant

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

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

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

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

Thrust	100 μN
Specific Impulse	2150 s

Table 4.4.1: Simulation Thruster Parameters

4.4.2.5 Orbit maintenance

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

4.4.2.6 Results

The results are computed for a 3U CubeSat with an ion thruster. The characteristics of the thruster are the following ones (for more characteristics of the thruster refer to the section **??**.):

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.

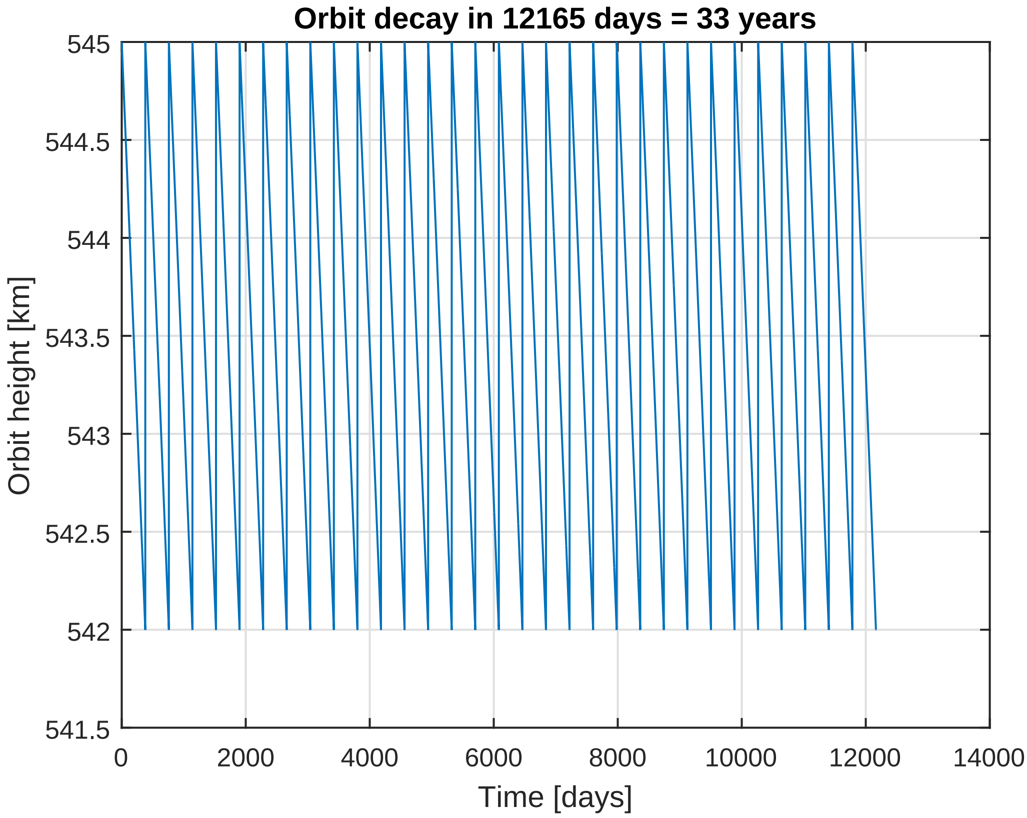


Figure 4.4.3: Height variation of the satellite

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:

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

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

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.

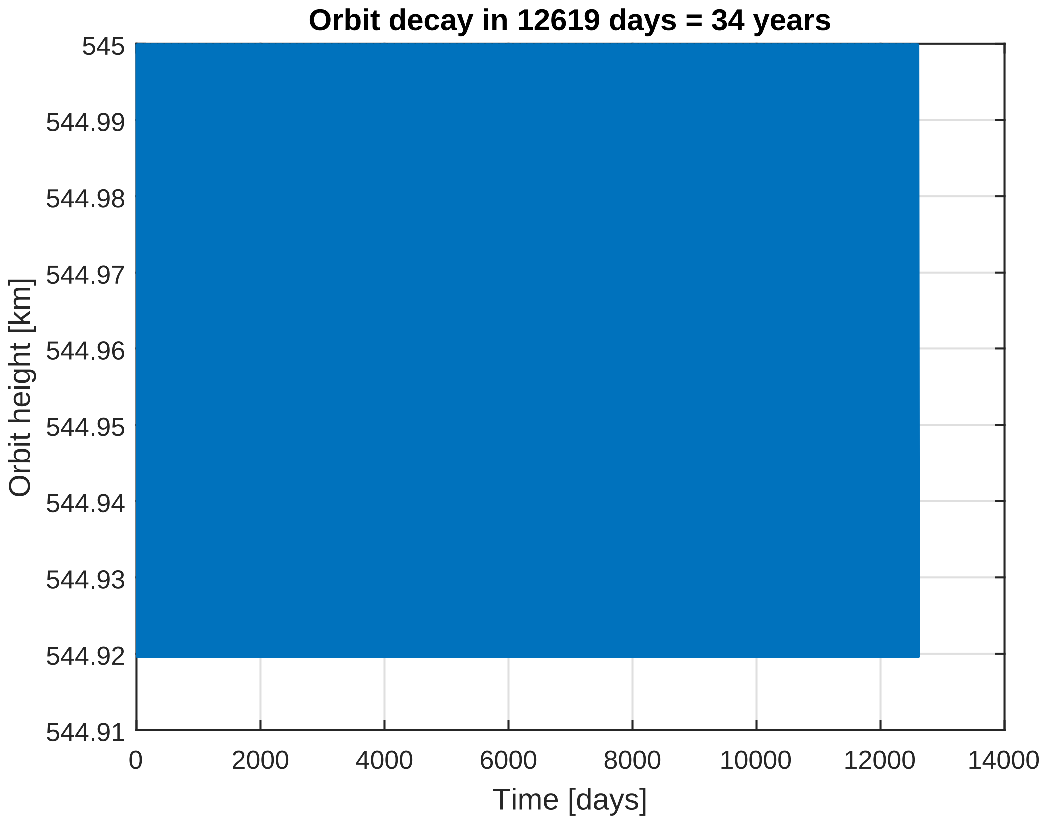


Figure 4.4.4: Height variation of the satellite with a more restrictive minimum height

Chapter 5

Constellation Design Decision

"Aerospace Engineering is the way to the universe."

Marc Cortés Fargas, 2012

5.1 Considered Designs

5.1.1 Introduction

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

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

5.1.2 Candidate 1: Polar - Global Coverage

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

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

5.1.3 Candidate 2: Polar - GS Coverage

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

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

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

5.1.4 Candidate 3 and 4: Walker-Delta GS Coverage

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

Candidate 3

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

Candidate 4

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

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

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

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

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

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

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

5.1.7 Candidate 7: Walker-Delta GS Coverage 3

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

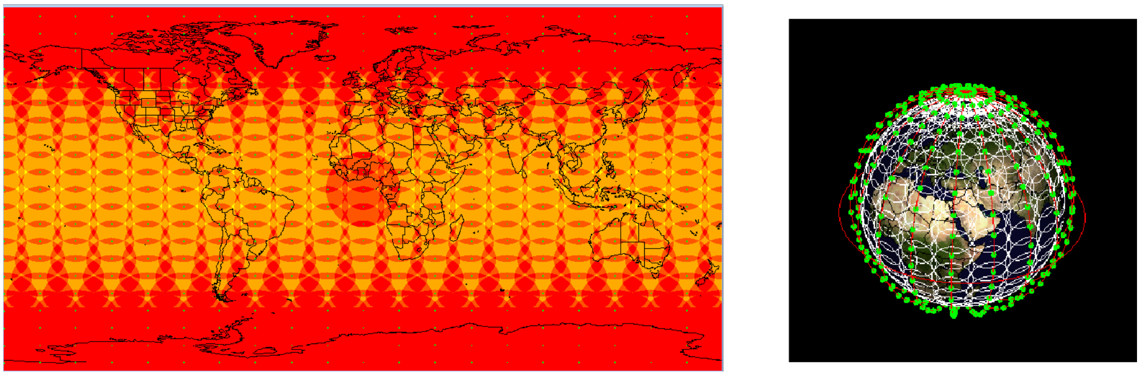


Figure 5.1.1: Candidate 1. Full Polar constellation with global coverage. h= 560km; Np=20; Npp=21; Tsat=420

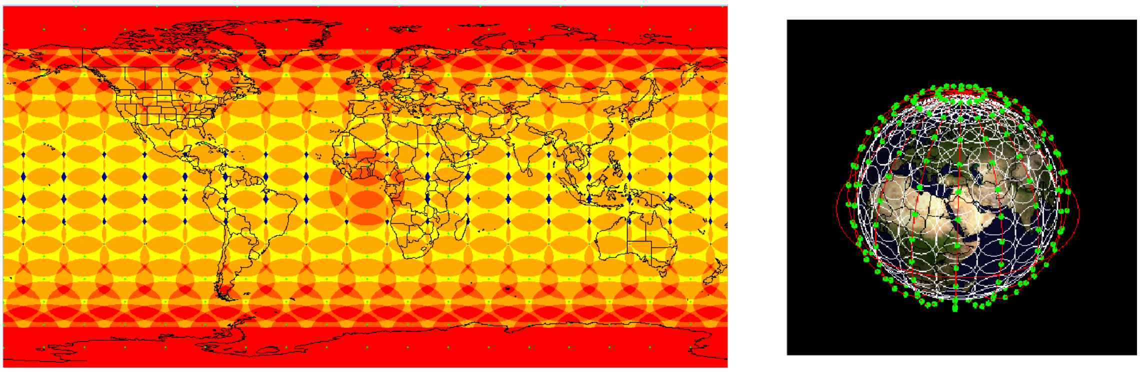


Figure 5.1.2: Candidate 2. Full Polar constellation with total ground station coverage. h= 550km; Np=18; Npp=20; Tsat=288

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

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

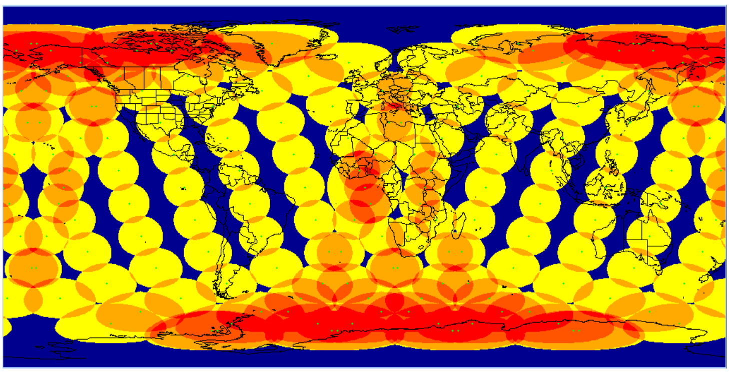
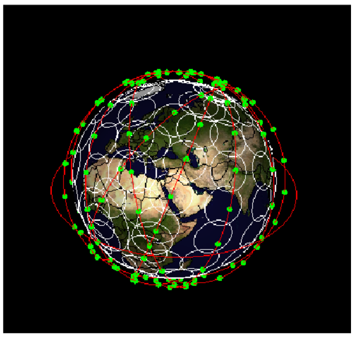



Figure 5.1.3: Candidate 3. 210° Walker-Delta constellation configuration. h= 542km; in=72; Np=8; Npp=21; Tsat=168

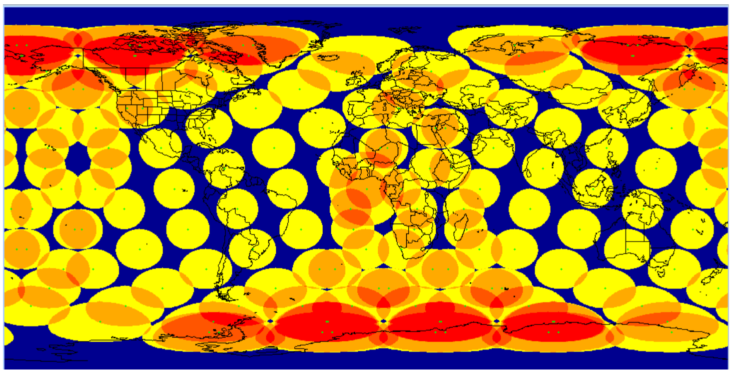
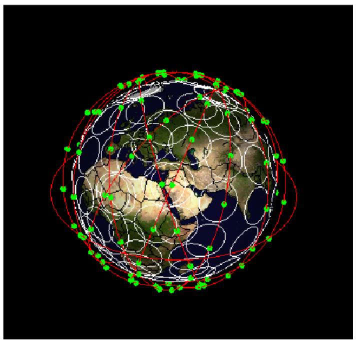



Figure 5.1.4: Candidate 4. 225° Walker-Delta constellation configuration. h= 542km; in=72; Np=9; Npp=17; Tsat= 153

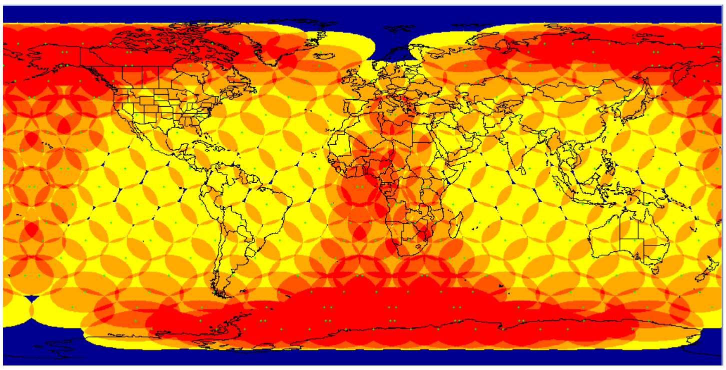
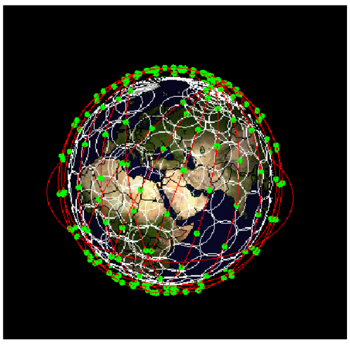



Figure 5.1.5: Candidate 5. 210° Walker-Delta constellation configuration with total coverage of the lattitudes from 0 to 52 degrees. h= 560km; in=72; Np=9; Npp=17; Tsat= 153

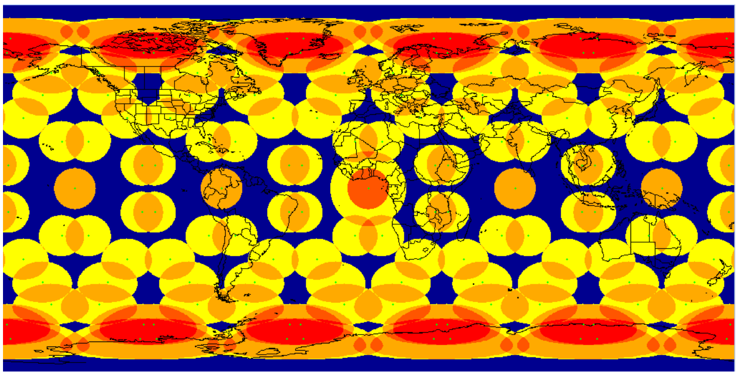
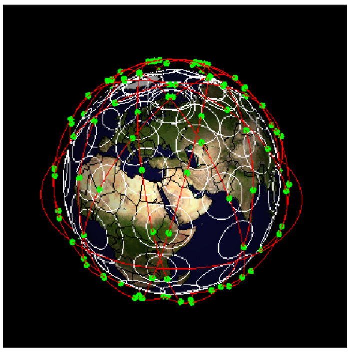



Figure 5.1.6: Candidate 6. 225° Walker-Delta constellation configuration.
h= 542km; in=72; Np=9; Npp=21; Tsat= 189

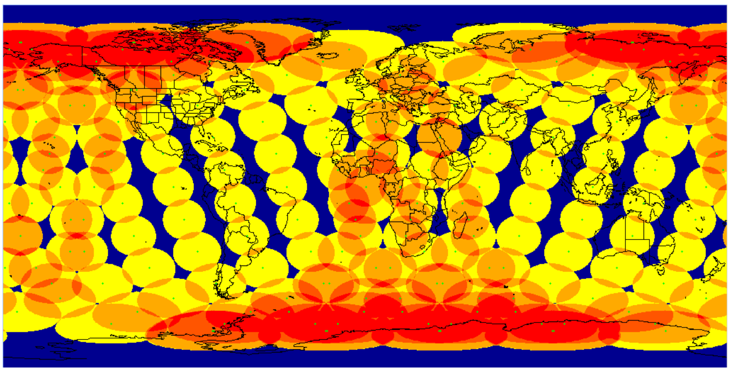
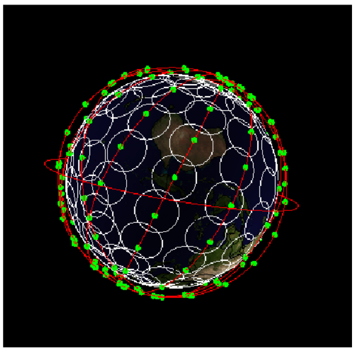



Figure 5.1.7: Candidate 7. Full Walker-Delta constellation configuration.

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	$i = 75^{\circ}$
GS Latitude	$\lambda = 80^{\circ}$
GS Longitude	$\phi = 0^{\circ}$

Table 5.2.1: Constellation parameters for the Example Constellation

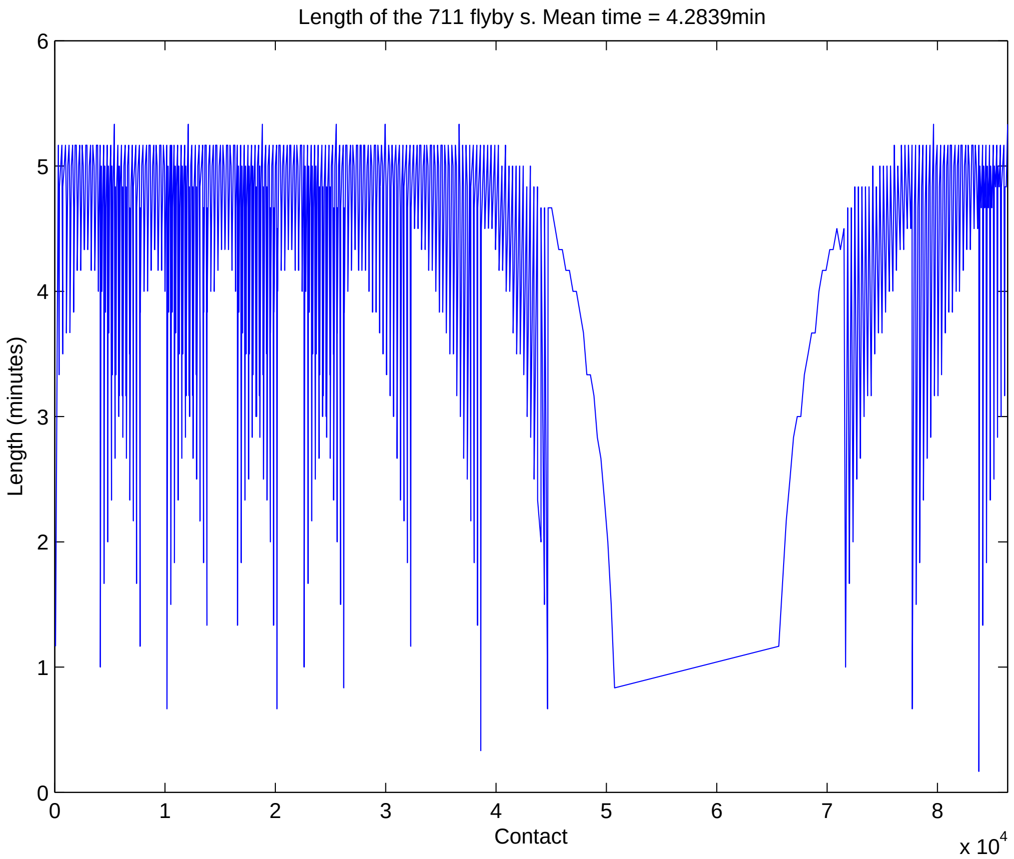


Figure 5.2.1: Length of the passes on the example GS.

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

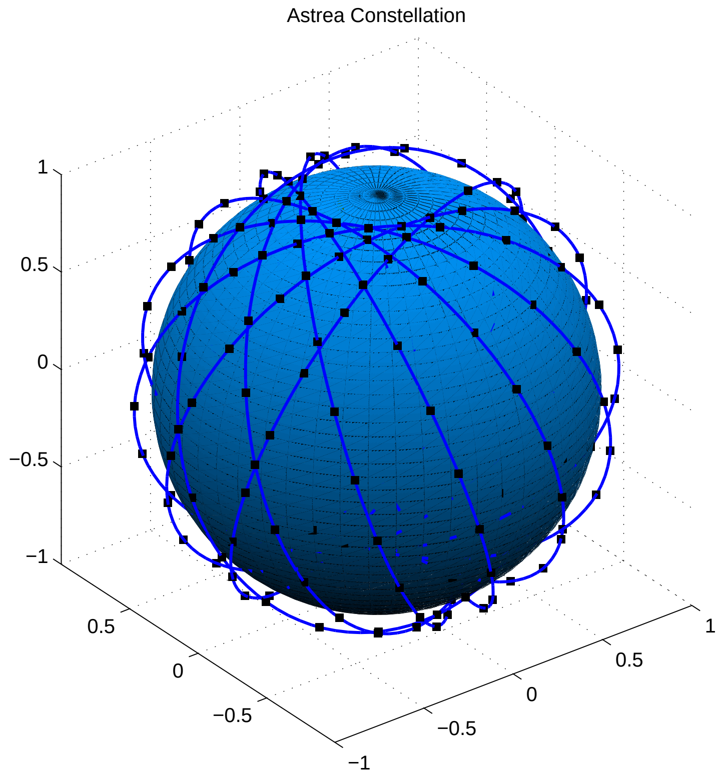


Figure 5.3.1: Astrea Constellation Final Configuration.

Part II

ANNEX II: Constellation Deployment

Chapter 6

Constellation Deployment

6.1 Constellation Deployment Department

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

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

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

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

6.2 Launching System

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

6.2.1 Launch site and vehicle analysis

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

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

The following table displays the first seven candidates.

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

Table 6.2.1: List of Launchers

6.2.2 Last candidates and selection

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

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

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

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

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

Table 6.2.2: Criteria

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

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

Figure 6.2.1: Electron Rocket

6.2.3 Launcher overview

Following, a brief description of Electron is provided.

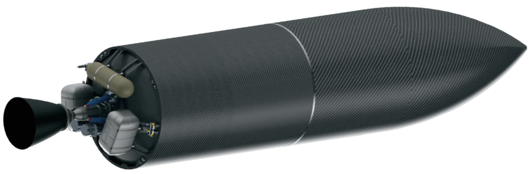


Figure 6.2.2: Second Stage

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

Figure 6.2.3: Electron Rocket Fairing

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

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

Table 6.2.3: Flight Profile

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

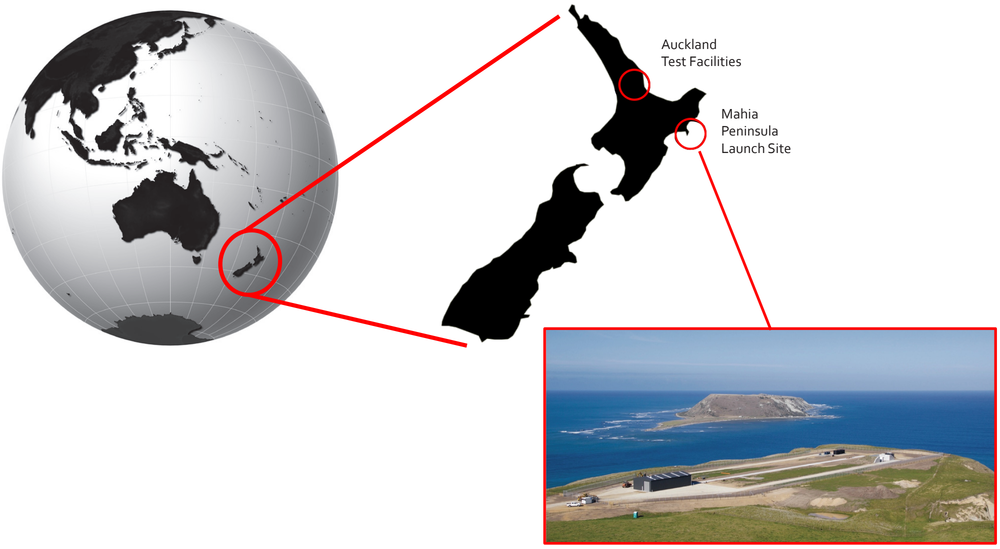


Figure 6.2.4: Rocket Lab Facilities

6.3 Deployer

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

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

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

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

Figure 6.3.1: ISIPOD

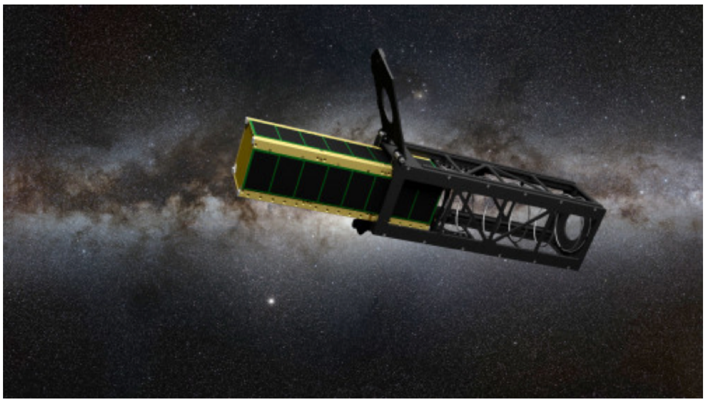


Figure 6.3.2: GPOD

6.4 First Placement

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

6.4.1 First Placement logistics

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

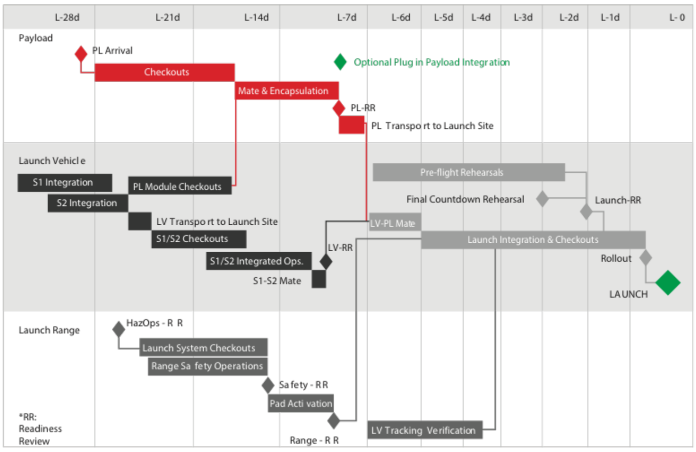


Figure 6.4.1: Launch Range Operations Flow/Schedule

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

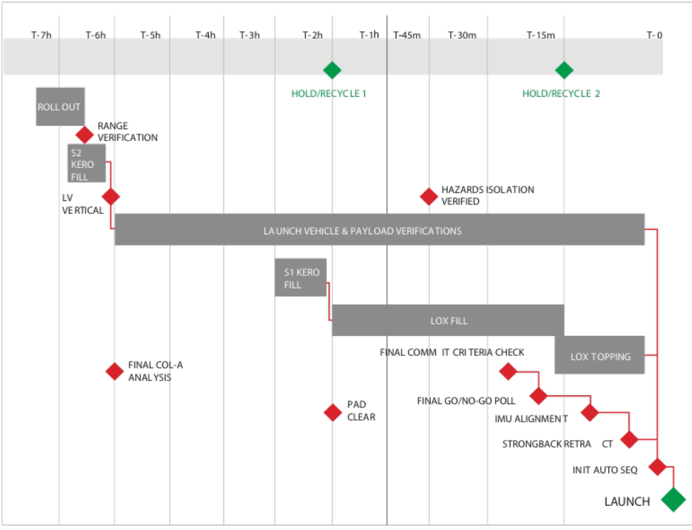


Figure 6.4.2: Countdown Operations Flow

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

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

6.4.2 1st Placement Maneuver

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

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

6.4.3 In-Orbit Injection

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

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

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

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

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

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

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

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

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

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

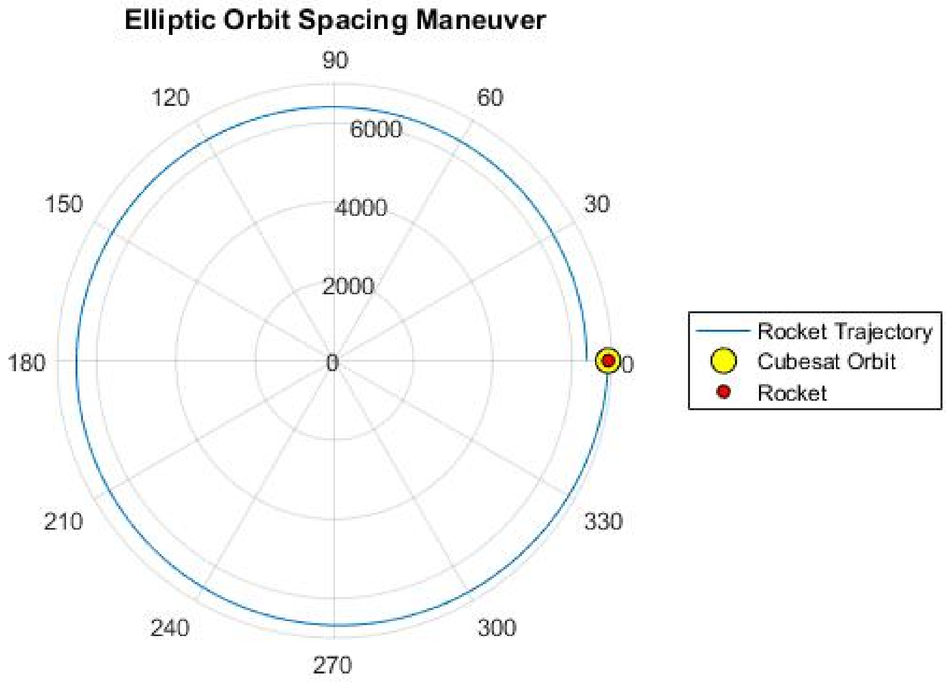


Figure 6.4.3: Rocket's trajectory from lift-off to final orbit.

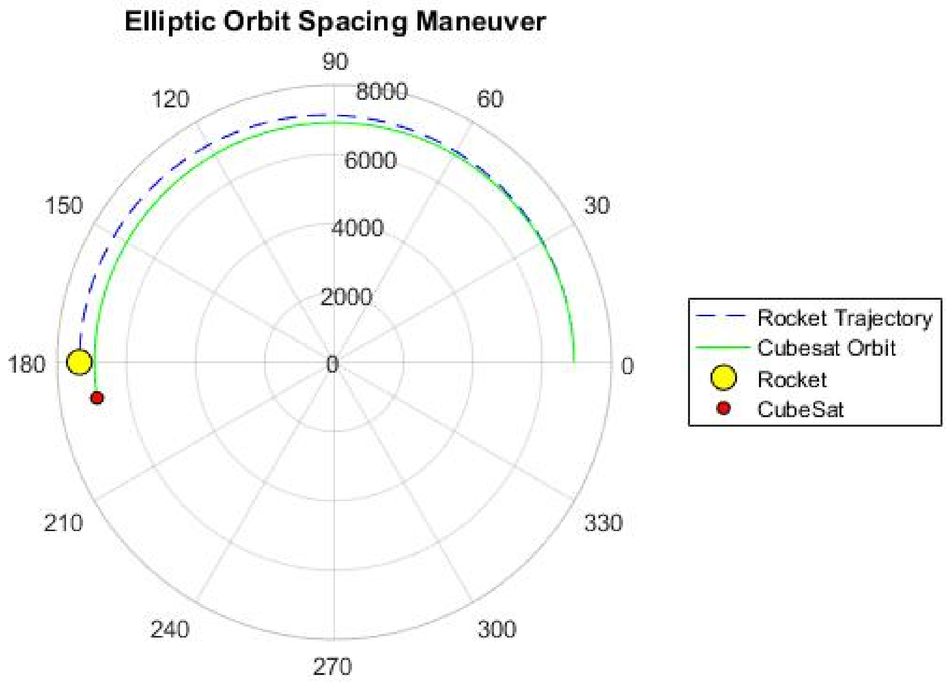


Figure 6.4.4: Half of a revolution of the rocket in the elliptical spacing orbit.

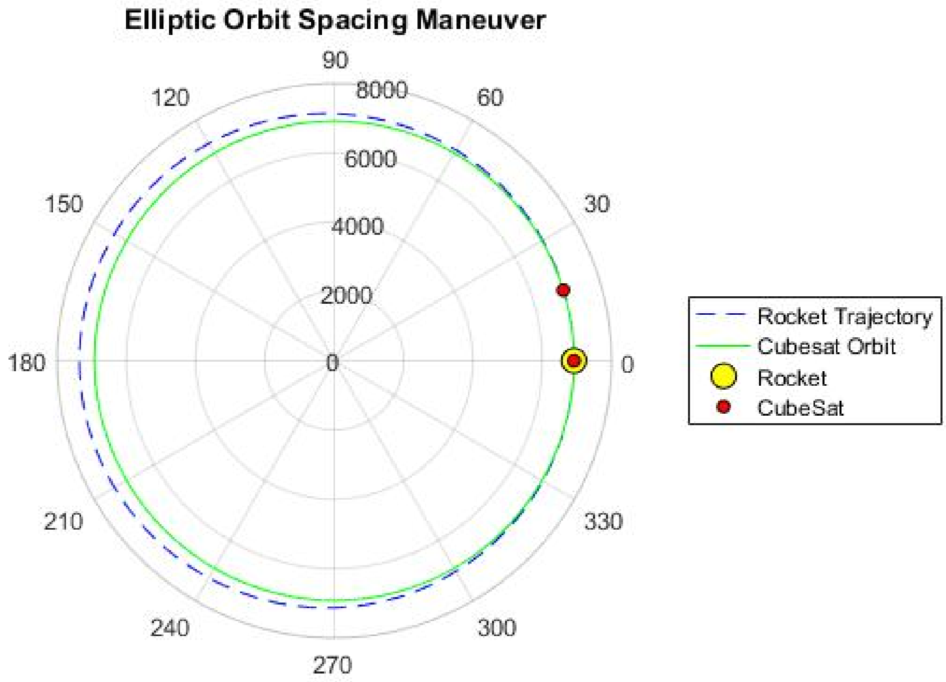


Figure 6.4.5: Deployment of the second satellite.

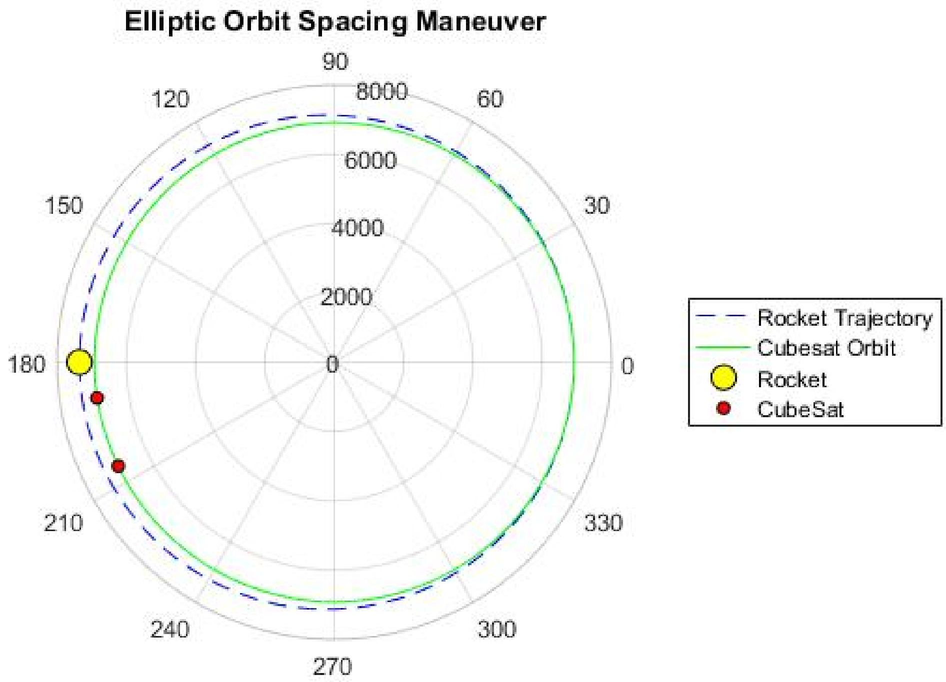


Figure 6.4.6: Half of a revolution of the rocket after the deployment of the second satellite.

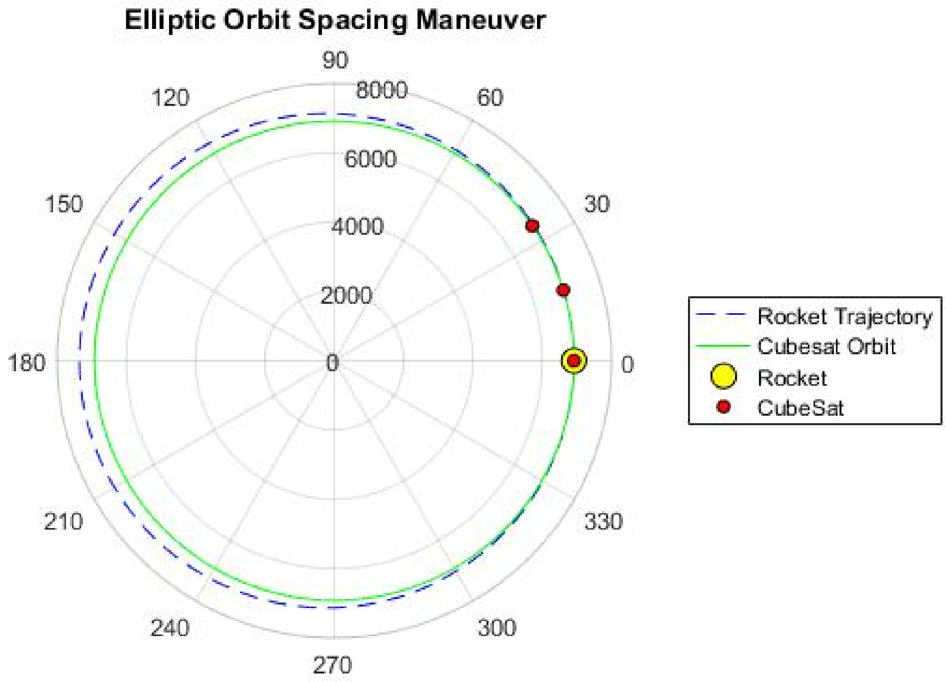


Figure 6.4.7: Deployment of the third satellite.

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

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

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

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

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

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

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

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

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

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

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

6.4.3.1 Plane Order

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

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

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

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

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

6.5 Replacement Strategy

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

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

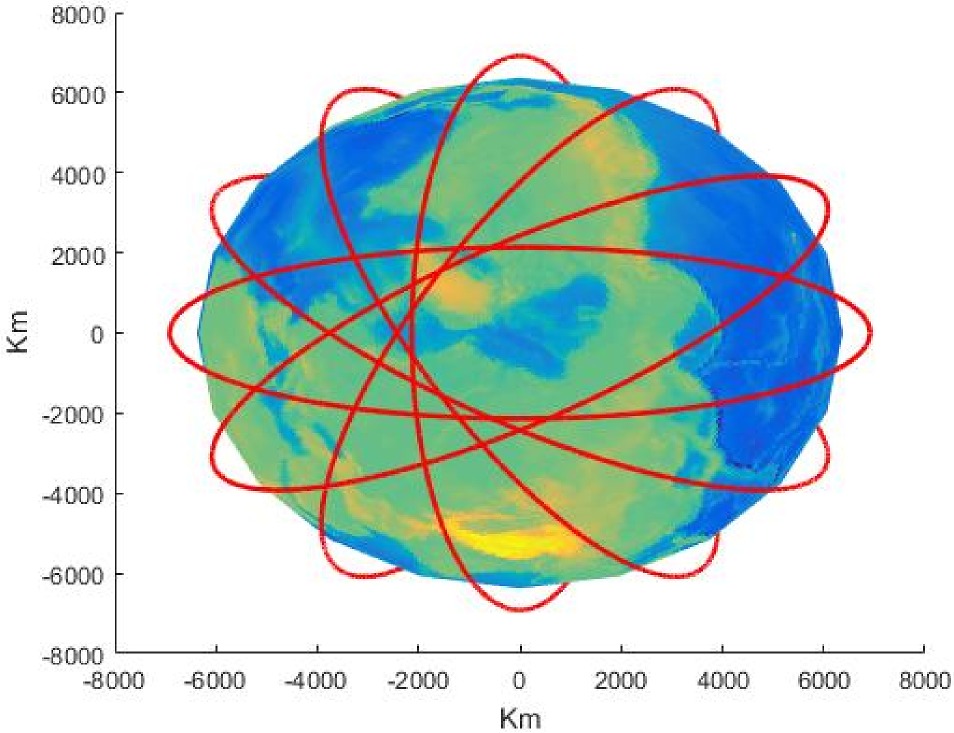


Figure 6.5.1: Old Constellation

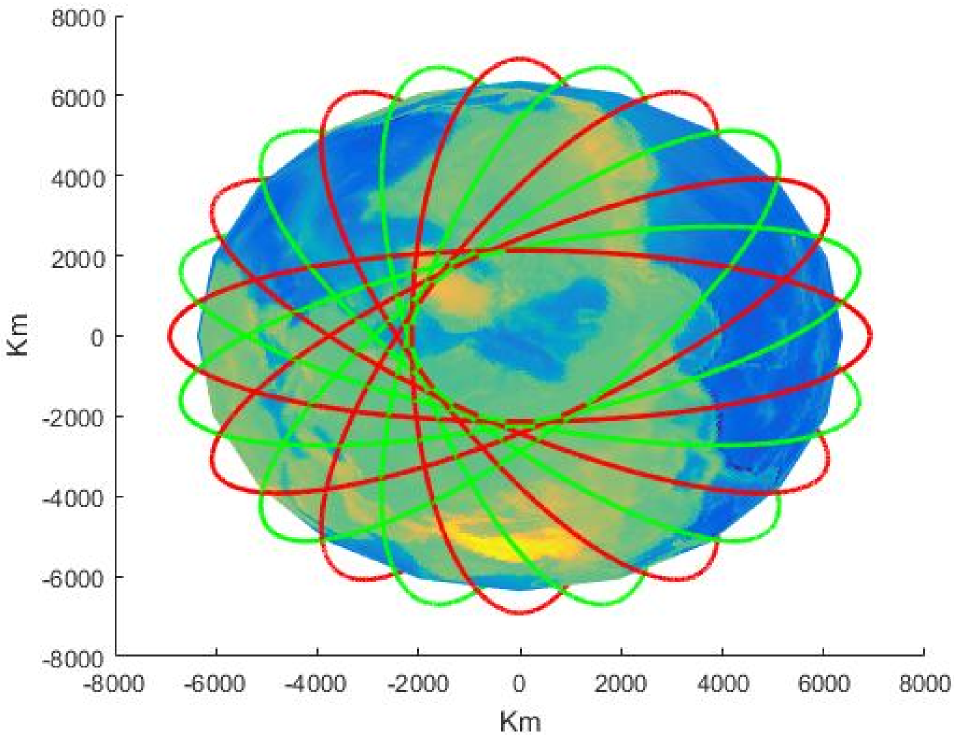


Figure 6.5.2: Old and New Constellations

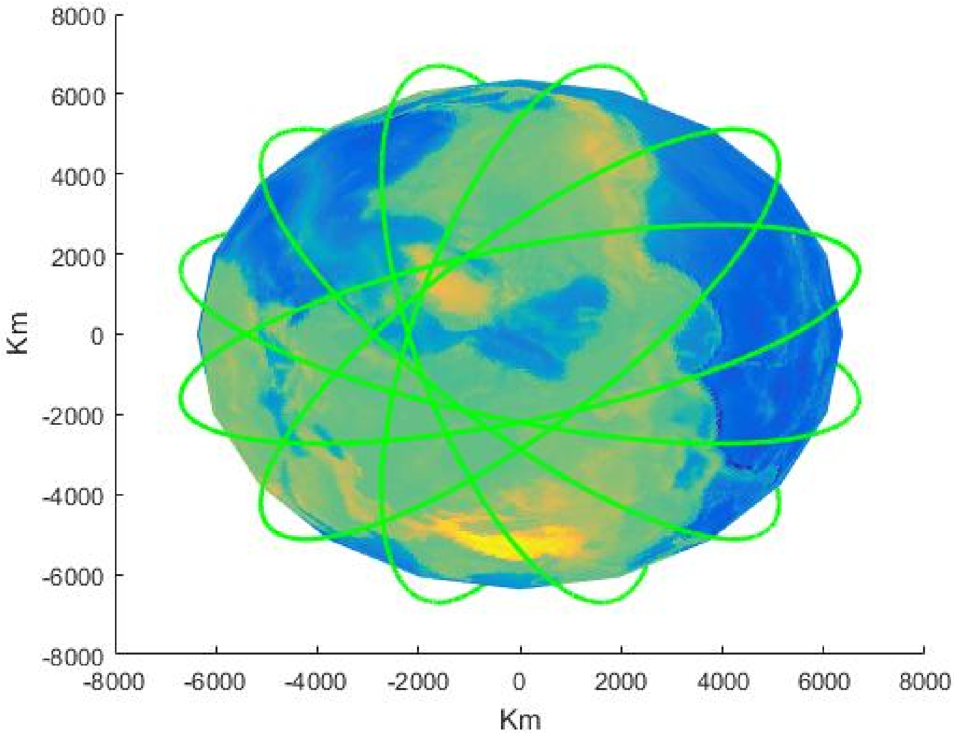


Figure 6.5.3: New Constellation

6.6 Spare Strategy

6.6.1 Introduction

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

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

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

6.6.2 Spare Strategy Alternatives

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

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

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

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

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

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

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

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

6.6.3 Spare Strategy Selection

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

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

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

6.6.4 Major failure deffinition

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

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

6.6.5 Major failure

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

6.6.5.1 Satellite in range failure

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

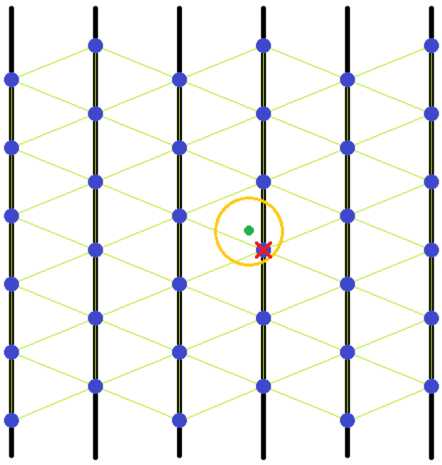


Figure 6.6.1: Failure due to the loss of the only satellite in range of the client satellite.

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

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

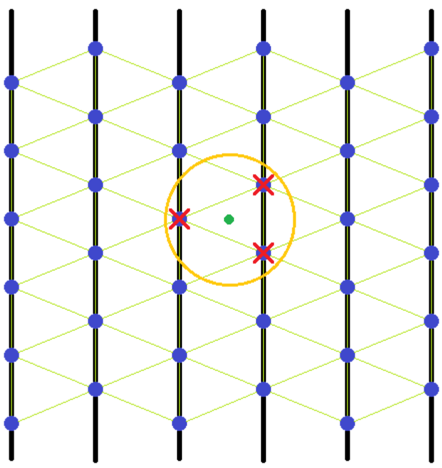


Figure 6.6.2: Failure due to the loss of all possible communication satellites if the client can communicate to three network satellites.

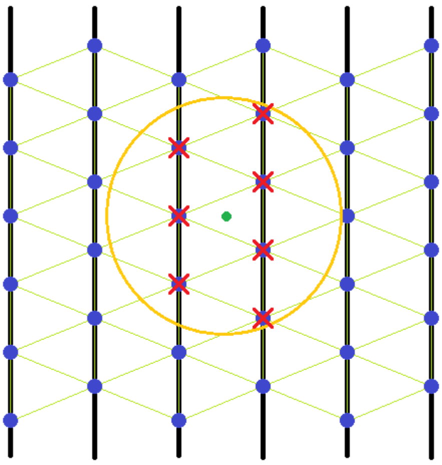


Figure 6.6.3: Failure due to the loss of all possible communication satellites if the client can communicate to seven network satellites.

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

6.6.5.2 Ground station failure

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

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

6.6.5.3 Transmitting time failure

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

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

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

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

6.6.5.4 Conclusion

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

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

6.6.6 Decision

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

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

6.7 End-of-Life Strategy

6.7.1 Introduction

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

6.7.2 Space Debris

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

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

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

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

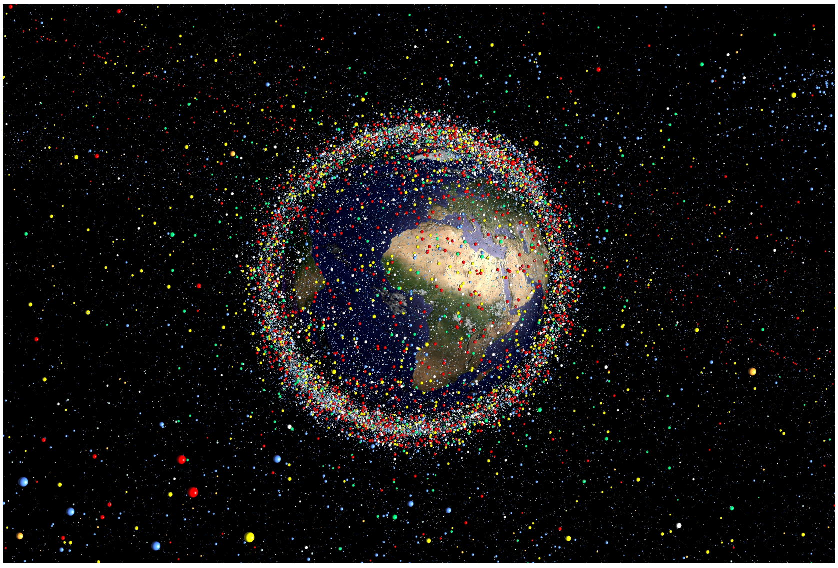


Figure 6.7.1: View of the Space Debris around the Earth

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

6.7.3 End-of-Life Types

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

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

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

- Controlled de-orbit:

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

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

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

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

The speed at apogee of the elliptical orbit is:

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

Finally, the ΔV is computed:

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

- Uncontrolled de-orbit:

  A simpler and cheaper way to de-orbit satellites is to induce a reduction of the orbit altitude in order to cause a decay and ,finally, a re-entry to the atmosphere. The process is initiated by one or several arc maneuvers at apogee passes and it is carried out without controlling the trajectory. This procedure is appropriate for low-thrust systems and small satellites.

  In addition, when considering satellites placed at LEOs, this strategy takes advantages of the perturbations present in this altitudes (atmospheric drag). This force contributes to the decay increasing the rate of approach to the atmosphere.

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

6.8 Conclusions

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

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

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

Part III

ANNEX 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 Layer 2: Data Link

7.1.1 Functions of the DLL

The explained functions are:

- **Framing**: Data-link layer takes packets from Network Layer and encapsulates them into Frames.Then, it sends each frame bit-by-bit on the hardware. At receiver end, data link layer picks up signals from hardware and assembles them into frames.
- **Adressing**: Each device on a network has a unique number, usually called a hardware address or MAC address, that is used by the data link layer protocol to ensure that data intended for a specific machine gets to it properly.
- **Synchronization**: When data frames are sent on the link, both machines must be synchronized in order to transfer to take place.
- **Error control**: Sometimes signals may have encountered problem in transition and the bits are flipped. These errors are detected and attempted to recover actual data bits.
- **Flow control**: Stations on same link may have different speed or capacity. Data-link layer ensures flow control that enables both machine to exchange data on same speed.

7.1.2 Working procedure

Working procedure is explained deeply now. All the images have been extracted from [7].

7.1.2.1 Simplest Protocol

This protocol has no error or flow control. It is supposed that the frames are traveling only in one direction, from the sender to the receiver. It is also supposed that the receiver can immediately handle the frames received, so there is no overwhelming. The DLL of the sender site gets data from its network layer, makes a frame out of the data and sends it. The DLL at the receiver site receives a frame from its physical layer, extracts data from the frame and delivers the data to its network layer. The problem here is that the sender site cannot send a frame until its network layer has a data packet to send and the receiver site cannot deliver a data packet to its network layer until a frame arrives. There is the need to introduce the idea of events in the protocol. The procedure at the sender site is constantly running; there is no action until there is a request from the network layer.

The procedure at the receiver site is also constantly running, but there is no action until notification from the physical layer arrives.

```
while (true)                        // Repeat forever
{
 WaitForEvent()i                    // Sleep until an event occurs
  if(Event(RequestToSend»           //There is a packet to send
  {
     GetData()i
     MakeFrame()i
     SendFrame()i                   //Send the frame
  }
}
```

Figure 7.1.1: Sender algorithm for the simplest protocol.

```
while(true)                         // Repeat forever
{
 WaitForEvent()i                    II Sleep until an event occurs
 if(Event(ArrivalNotification»      IIData frame arrived
 {
     ReceiveFrame()i
     ExtractData()i
     DeliverData ()i                //Deliver data to network layez
 }
}
```

Figure 7.1.2: Receiver algorithm for the simplest protocol.

7.1.2.2 Stop-and-Wait Protocol

If data frames arrive at the receiver site faster than they can be processed, the frames must be stored until their use. Normally, the receiver does not have enough storage space, especially if it is receiving data from many sources. This may result in either the discarding of frames or denial of service. To prevent the receiver from becoming overwhelmed with frames,we somehow need to tell the sender to slow down. There must be feedback from the receiver to the sender.

In the Stop-and-Wait Protocol the sends one frame, stops until it receives confirmation from the receiver and then sends the next frame. We still have unidirectional communication for data frames, but auxiliary ACK frames (simple tokens of acknowledgment) travel from the other direction. We add flow control to our previous protocol. In this case the algorithms of the sender and the receiver are the following ones.

```
while(true)                        //Repeat forever
canSend = true                     //Allow the first frame to go
{
  WaitForEvent();                  // Sleep until an event occurs
  if(Event(RequestToSend) AND canSend)
  {
     GetData();
     MakeFrame();
     SendFrame();                  //Send the data frame
     canSend = false;              //cannot send until ACK arrives
  }
  WaitForEvent();                  // Sleep until an event occurs
  if(Event(ArrivalNotification)    // An ACK has arrived
   {
     ReceiveFrame();               //Receive the ACK frame
     canSend = true;
   }
}
```

Figure 7.1.3: Sender algorithm for the Stop-and-Wait Protocol.

```
while(true)                        //Repeat forever
{
  WaitForEvent();                  // Sleep until an event occurs
  if(Event(ArrivalNotification))   //Data frame arrives
  {
     ReceiveFrame();
     ExtractData();
     Deliver(data);                //Deliver data to network layer
     SendFrame();                  //Send an ACK frame
  }
}
```

Figure 7.1.4: Receiver algorithm for the Stop-and-Wait Protocol.

The two protocols explained are protocols that can be suitable for noiseless channels. However, noiseless channels are nonexistent. There is a need to add error control to the protocol. Three protocols are discussed with the aim of doing so.

7.1.2.3 Stop-and-Wait Automatic Repeat Request

The Stop-and Wait ARQ adds a simple error control mechanism to the Stop-and-Wait Protocol. To detect and correct corrupted frames, we need to add redundancy bits to our data frame. When the frame arrives at the receiver site, it is checked and if it is corrupted, it is silently discarded. The detection of errors in this protocol is manifested by the silence of the receiver. Frames are also numbered so if the receiver receives a data frame that is out of order, this means that frames were either lost or duplicated. What is done to solve the error is that when the sender sends a frame, it keeps a copy of the sent frame. At the same time, it starts a timer. If the timer expires and there is no ACK for the sent frame, the frame is resent, the copy is held, and the timer is restarted. Since the protocol uses the stop-and-wait mechanism, there is only one specific frame that needs an ACK even though several copies of the same frame can be in the network. Since an ACK frame can also be corrupted and lost, it too needs redundancy bits and a sequence number. In the following figure is possible to see more clearly what is going on with this protocol.

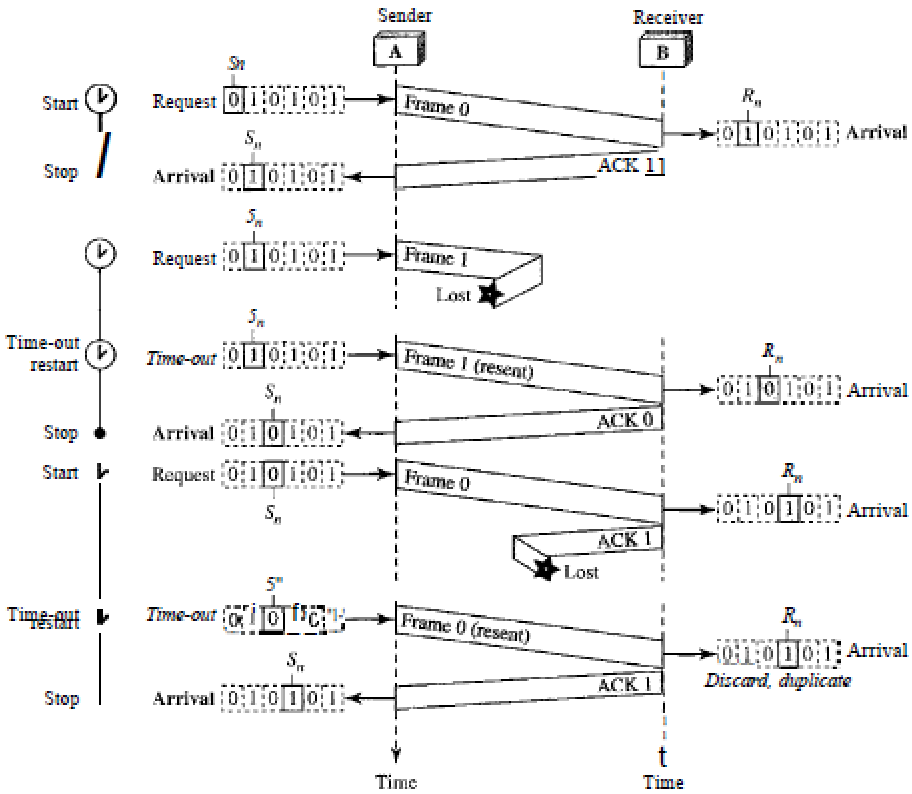


Figure 7.1.5: Flow diagram of the Stop-and Wait ARQ.

The main problem of this protocol is its efficiency. The Stop-and-Wait ARQ is very inneficinet if our channel is thick and long. The product of thickness and longitud is called the bandwidth-delay product. We can think of the channel as a pipe. The bandwidth-delay product then is the volume of the pipe in bits. The pipe is always there. If we do not use it, we are inefficient.

7.1.2.4 Go-Back-N Automatic Repeat Request

To improve the efficiency of transmission (filling the pipe), multiple frames must be in transition while waiting for acknowledgment. In other words, we need to let more than one frame be outstanding to keep the channel busy while the sender is waiting for acknowledgment. In the Go-Back-N Automatic Repeat Request the sender sends several frames before receiving acknowledgments. It also keeps a copy of these frames until the acknowledgments arrive. Although there can be a timer for each frame that is sent, in this protocol only one is used. The reason is that the timer for the first outstanding frame always expires first and then all outstanding frames when this timer expires are sent again. The receiver sends a positive acknowledgment if a frame has arrived safe and sound and in order. If a frame is damaged or is received out of order, the receiver is silent and will discard all subsequent frames until it receives the one it is expecting. The silence of the receiver causes the timer of the unacknowledged frame at the sender site to expire. This, in turn, causes the sender to go back and resend all frames, beginning with the one with the expired timer. The receiver does not have to acknowledge each frame received. It can send one cumulative acknowledgment for several frames. That is the reason why the protocol is called Go-Back-N. The flow diagram and the and algorithms of the sender and the receiver are shown next.

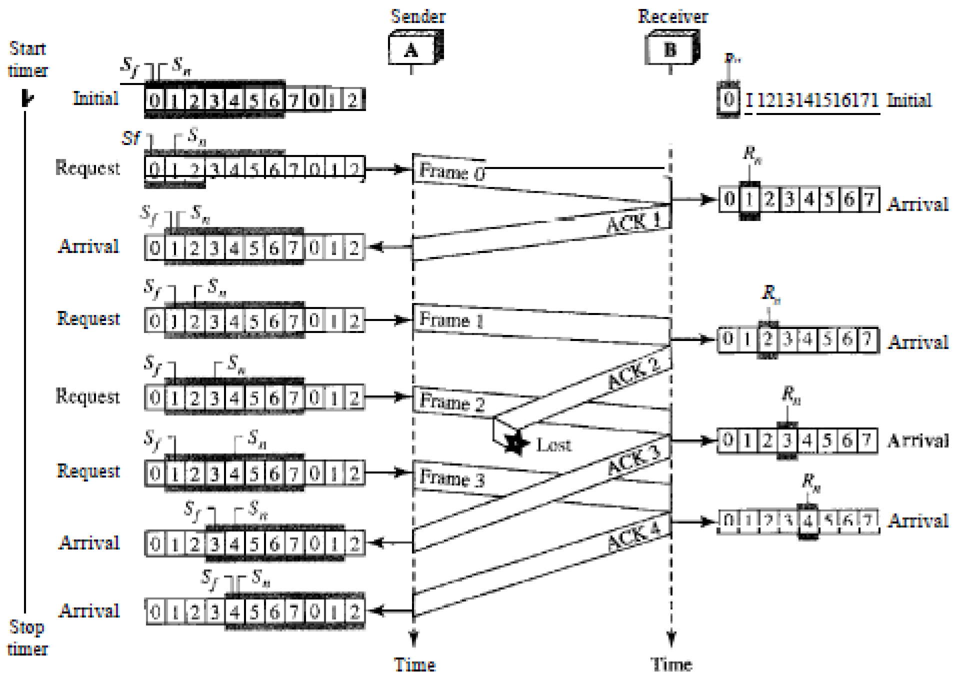


Figure 7.1.6: Flow diagram of the Go-Back-N ARQ.

```
Rn = 0;

while (true)                             IIRepeat forever
{
  WaitForEvent();

  if(Event{ArrivalNotification»          /Data frame arrives
  (
     Receive(Frame);
     if(corrupted(Frame»
          Sleep();
     if(seqNo == Rn)                     IIIf expected frame
     {
       DeliverData()i                    IIDeliver data
       Rn =Rn + 1;                       IISlide window
       SendACK(Rn);
     }
  }
}
```

Figure 7.1.7: Receiver algorithm for the Go-Back-N ARQ.

```
Sw = 2m - 1;
Sf = 0;
Sn = 0J

while (true)                          //Repeat forever
{
 WaitForEvent();
  if(Event(RequestToSend»             //A packet to send
  {
     if(Sn-Sf >= Sw)                  ///If window is full
           Sleep () ;
     GetData() ;
     MakeFrame (Sn) ;
     StoreFrame (Sn) ;
     SendFrame(Sn) ;
     Sn = Sn + 1;
     if(timer not running)
          StartTimer ();
  }

  if{Event{ArrivalNotification»       IIACK arrives
  {
     Receive (ACK) ;
     if{corrupted{ACK»
          Sleep ();
     if{{ackNo>sf)&&{ackNO<=Sn»       ///If a valid ACK
     While(Sf <= ackNo)
      {
       PurgeFrame{Sf);
       Sf = Sf + 1;
      }
      StopTimer();
  }

  if{Event{TimeOut»                   liThe timer expires
  {
   StartTimer() ;
   Temp = Sf;
   while{Temp < Sn);
    {
     SendFrame(Sf);
     Sf = Sf + 1;
    }
  }
}
```

Figure 7.1.8: Sender algorithm for the Go-Back-N ARQ.

7.1.2.5 Selective Repeat Automatic Repeat Request

Go-Back-N ARQ simplifies the process at the receiver site. The receiver keeps track of only one variable, and there is no need to buffer out-of-order frames; they are simply discarded. However, this protocol is very inefficient for a noisy link. In a noisy link a frame has a higher probability of damage, which means the resending of multiple frames. In the case of these protocol, the Selective Repeat ARQ, the processing at the receiver is more complex but is more efficient for noisy links. The Selective Repeat Protocol allows a number of frames to arrive out of order and be kept until there is a set of in-order frames to be delivered to the network layer. The handling of the request event is similar to that of the previous protocol except that one timer is started for each frame sent. The arrival event is more complicated here. An ACK or a NAK frame may arrive. If a valid NAK frame arrives, the corresponding frame is resent. If a valid ACK arrives the corresponding timer stops. When the time for a frame has expire, only this frame is resent.

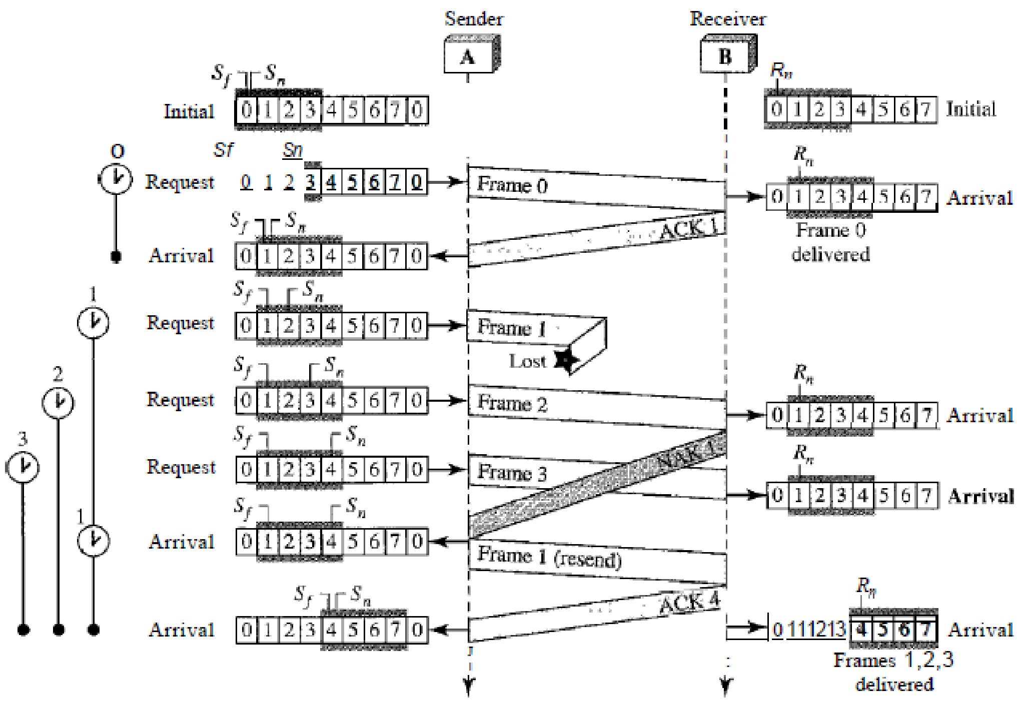


Figure 7.1.9: Flow diagram of the Selective Repeat ARQ.

```
    = 2m-1 ;
    = 0;
    = 0;

 hile (true)                          //Repeat forever
 {
   WaitForEvent();
   if(Event(RequestToSend)}           //There is a packet to sen
   {
```

```
    if{Sn-S;E >= Sw)                I/If window is full
          Sleep ();
    GetData () ;
    MakeFrame (Sn) ;
    StoreFrame{Sn);
    SendFrame (Sn) ;
    Sn = Sn + 1;
    StartTimer{Sn);
 }

 if(Event{ArrivalNotification»   IIACK arrives
 {
    Receive{frame);                 I/Receive ACK or NAK
    if{corrupted{frame»
         Sleep ();
    if (FrameType == NAK)
       if (nakNo between Sf and So)
       {
        resend{nakNo);
        StartTimer{nakNo);
       }
    if (FrameType == ACK)
       if (ackNo between Sf and So)
       {
         while{sf < ackNo)
         {
          Purge (sf);
          stopTimer (Sf) ;
          Sf = Sf + 1;
         }
       }
 }

 if(Event{TimeOut{t»)            liThe timer expires
 {
  StartTimer{t);
  SendFrame{t);
 }
}
```

Figure 7.1.10: Sender algorithm for the Selective Repeat ARQ.

```
Rn = 0;
NakSent = false;
AckNeeded = false;
Repeat(for all slots)
    Marked(slot) = false;

!while (true)                              //Repeat forever
{
  WaitForEvent()i

  if{Event{ArrivalNotification»            jData frame arrives
  {
     Receive(Frame);
     if(corrupted(Frame»&& (NOT NakSent)
     {
      SendNAK(Rn);
      NakSent = true;
      Sleep();
     }
     if(seqNo <> Rn)&& (NOT NakSent)
     {
      SendNAK(Rn);
      NakSent = true;
      if {(seqNo in window)&&(IMarked(seqNo»
      {
       StoreFrame{seqNo)
       Marked(seqNo)= true;
       while(Marked(Rn))
       {
        DeliverData(Rn);
        Purge(Rn);
        Rn = Rn + 1;
        AckNeeded = true;
       }
        if(AckNeeded);
        {
        SendAck(Rn);
        AckNeeded =false;
        NakSent = false;
        }
      }
     }
  }
}
```

Figure 7.1.11: Receiver algorithm for the Selective Repeat ARQ.

7.1.2.6 Bidirecional links: Piggybacking

Piggybacking is not a protocol, is a technique. All que protocols explained until now are all unidirectional: data frames flow in only one direction although control information such as ACK and NAK frames can travel in the other direction. In real life, data frames are normally flowing in both directions: from node A to node B and from node B to node A. This means that the control information also needs to flow in both directions. Piggybacking is usedto improve the efficiency of the bidirectional protocols. When a frame is carrying data from A to B, it can also carry control information about arrived (or lost) frames from B; when a frame is carrying data from B to A, it can also carry control information about the arrived (or lost) frames from A.

7.1.3 TC Space Data Link Protocol

Now some specifications of the chosen protocol will be exposed in order to know how it is structured and how many bits it adds to the original data. Further information of the protocol can be found in [8]. The protocol specifications will be explained when it is used with the support of the SDLS protocol. In this section is important to know that 1 octet is an eight-bit word. The structure of the transfer frame in this protocol is the following one:

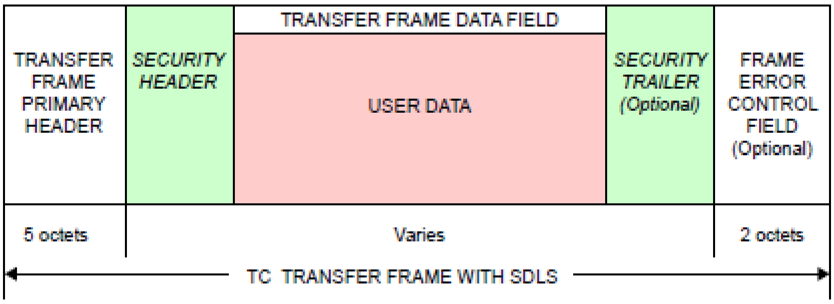


Figure 7.1.12: Transfer frame structure of the TC Space DL Protocol with SDLS.

In the transfer frame primary header, the following information is contained:

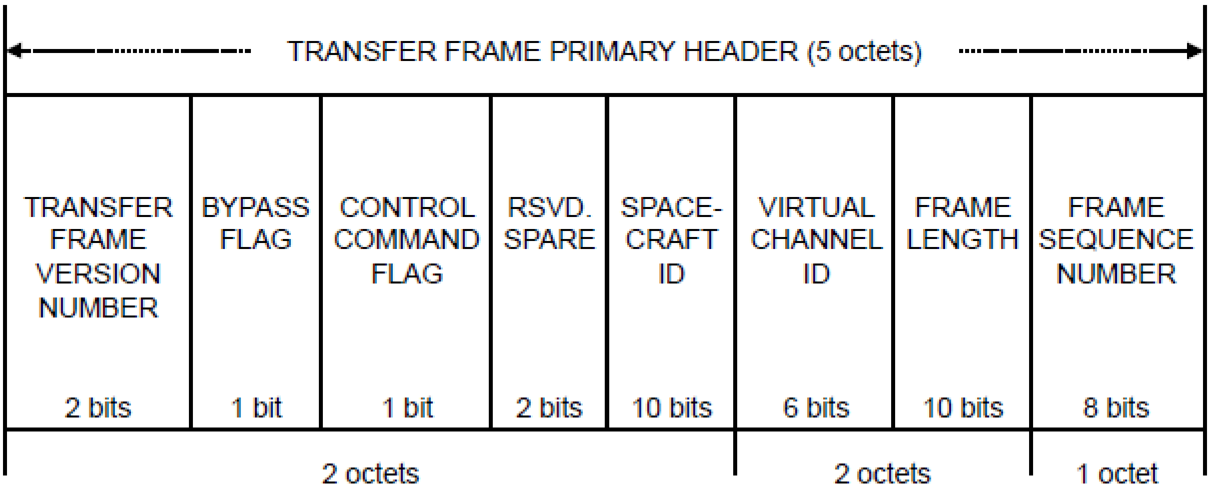


Figure 7.1.13: Transfer frame primary header.

With this data, is possible to say that the TC Space Data Link Protocol will add to data coming from the Network layer at least 5 octets (40 bits).

7.1.4 TC Sync and Channel Coding

This protocol is the corresponding to the Synchronization and Channel Coding Sublayer that has be used with the TC Space and Data Link Protocol. It has functions as for example, encapsuate the data units so that the start and end can be detected by the receiving end, ensure there are sufficient bit transitions in the transmitted bit stream so that the receiver can mantain bit synchronization during the reception of the data unit, etc. In a nutshell, one instance of the Synchronization and Channel Coding Sublayer processes the data stream for a single Physical Channel, making it a stream of bits that can be transferred over a space link in a single direction. The procedures can be differenciated between the ones that occur in the sending end and the one that occur in the receiving end. The procedures are the following ones:

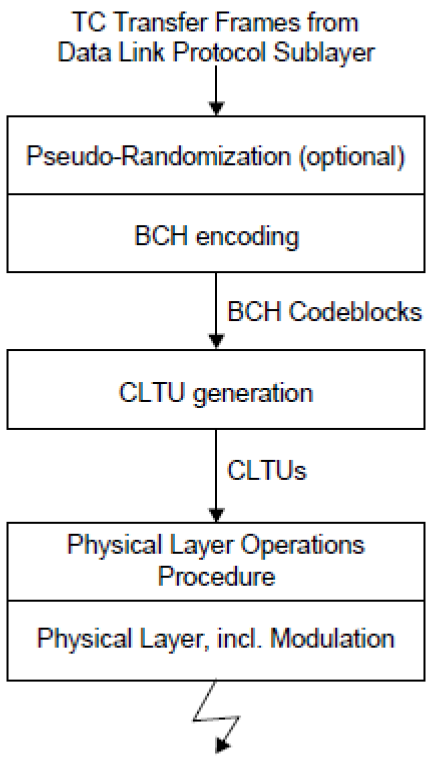


Figure 7.1.14: Procedure at the sending end.

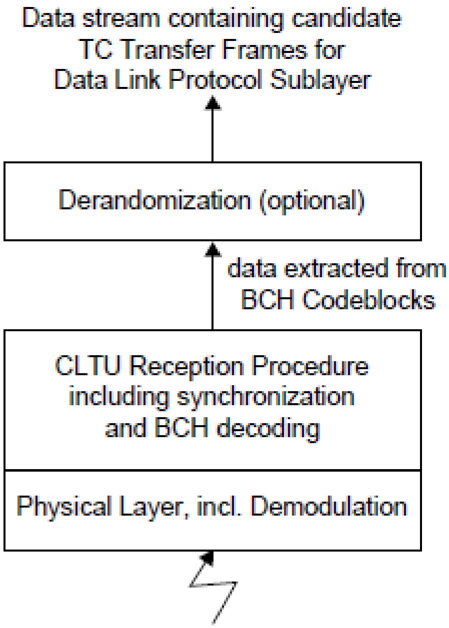


Figure 7.1.15: Procedure at the receiving end.

Is possible to see that two packets of data are created, BCH Codeblocks and CLTUs. From the point of view of the Synchronization and Channel Coding Sublayer, the content of the Frames parameter is a single block of data. For a single Channel Access request, the Synchronization and Channel Coding Sublayer generates a set of BCH Codeblocks, and that set of BCH Codeblocks is placed in a single CLTU. One of the managed parameters for the Physical Channel is the maximum length of a CLTU. The lenght of the CLTU can be calculated as follows (in octets):

$$LenghtoftheCLTU = 10 + 8 \cdot (\frac{Totallenghtoftheframes + 6}{7}) \tag{7.1.1}$$

Since with the TC Space Data Link protocol the frames can have different sizes, the CLTU can also have different sizes. More information about this sublayer of the DLL can be found in reference [9]

7.2 Layer 3: The Network

7.2.1 Functions of the Network Layer

The Network layer provides the following functions:

- **Routing**: Selects the best path between two nodes in a network, often using itermediate nodes called routers.
- **Network fow control**: Routers may indicate a transmitting node to reduce its transmission when the router's buffer becomes full.
- **Package fragmentation**: If the message to be transmitted is too large to be transmitted in the Data link layer, the network may split it into several packages in one node, send them independently an reassemble them in another node. Optionally, it can provide error control.
- **Logical-physical adress allocation**: Translates the logical adress (or names) of the network nodes into a unique physical adress.
- **Message forwarding**: A network may be divided into subnetworks, connected through specialized hosts, called gateways or routers, that forward packets between those subnetworks.

7.2.2 Protocols

7.2.2.1 Main protocols

Space Packet Protocol (SPP) [10]

The Space Packet Protocol (SPP) is a protocol designed to efficiently transfer application data over a network of space links. SPP provides a unidirectional data transfer service from a single source user application to one or more destination user applications through one or more subnetworks. The path from the source user application to the destination user application is called a Logical Data Path (LDP). Every LDP is uniquely identified by a Path Identifier (Path ID). The protocol data unit used by this protocol is the Space Packet. Each Space Packet is defined by a header section and a data section.

Each LPD is uniquely identified by a Path ID. A Path ID consists of an Application Process Identifier (APID) and an optional APID Qualifier. APID Qualifiers identify the naming domain for an APID. APIDs are unique in a single naming domain. The APID is part of the header of the Space Packet, but the APID Qualifier must be carried by a protocol of an underlying layer.

The following features are common to the services of the SPP:

- Pre-configured Services. The user can send or receive data only through a preconfigured LDP established by management.
- Unidirectional Services. One end of an LDP can send, but not receive, data through the LDP, while the other end can receive, but not send. This means A can send to B through a LPD, but for B to send to A has to use a different LDP
- Asynchronous Services. There are no predefined timing rules for the transfer of service data units supplied by the service user. The user may request data transfer at any time it desires, but there may be restrictions imposed by the provider on the data generation rate.
- Unconfirmed Services. The sending user does not receive confirmation from the receiving end that data has been received.
- Incomplete Services. The services do not guarantee completeness, nor do they provide a retransmission mechanism.
- Non–sequence Preserving Services. The sequence of service data units supplied by the sending user may not be preserved through the LDP.

The following services are assumed from the underlying layers:

- Adressing and routing capabilities for establishing LDPs
- Capability for associating an APID Qualifier for each Space Packet.

The structure of a Space Packet consists of a Packet Primary Header, and a Packet Data Field, which can contain an optional Secondary Header. Figure 7.2.1 shows the structure of the SPP primary header:

<table>
<tr><th>Offsets</th><th>Octet</th><th colspan="8">0</th><th colspan="8">1</th></tr>
<tr><th>Octet</th><th>Bit</th><th>0</th><th>1</th><th>2</th><th>3</th><th>4</th><th>5</th><th>6</th><th>7</th><th>8</th><th>9</th><th>10</th><th>11</th><th>12</th><th>13</th><th>14</th><th>15</th></tr>
<tr><td>0</td><td>0</td><td colspan="3">Packet Version Number</td><td>Packet Type</td><td>Secondary Header Flag</td><td colspan="11">Application Process Identifier (APID)</td></tr>
<tr><td>4</td><td>32</td><td colspan="2">Sequence Flags</td><td colspan="14">Packet Sequence Count or Packet Name</td></tr>
<tr><td>8</td><td>64</td><td colspan="16">Packet Data Length</td></tr>
</table>

Figure 7.2.1: Example of a header for an SPP Space Packet.

Internet Protocol version 4 (IPv4) [11]

The Internet Protocol version 4 (IPv4) is the fourth version of the Internet Protocol (IP). It is one of the core protocols of standards-based internetworking methods in the Internet. Despite the ongoing deployment of a successor protocol (IPv6), the IPv4 still routes most of the Internet traffic. IPv4 is a conectionless protocol and does not guarantee delivery, nor does it assure proper sequencing or avoidance of duplicate delivery. These aspects are adressed by a transport layer protocol.

One of the features of IPv4 are adresses. Network adresses are the identification number of any device that is part of a network. IPv4 uses 32-bit (4 byte) adresses. Therefore, the adress space is limited to 4294967296 (2^{32}) adresses. A IPv4 adress is usually represented in two ways: in binary notation, where each group of 8 bits is separated by a dot, or in decimal notation, where each 8-bit binary number is translated to decimal, as it can be seen in Table 7.2.1.

IP adress	10101100000100001111111000000001
Dot-binary notation	10101100.00010000.11111110.00000001
Dot-decimal notation	172.16.254.1

Table 7.2.1: IP adress notation in dot-decimal and dot-binary.

Packets in the IPv4 consist of a header section and a data section. There is no footer at the end of the data section since the protocols in the data link layer and the transport layer provide error correction controls. Headers in a IPv4 packet contain 14 fields, one of them being optional. The fields are packed with the most significant byte first, and the most significant bit is also the first. Headers have a lenght between 20 and 60 bytes. The data section comes after the header, and its format depends on the protocol used (for example, ICMP, IGMP, TCP, etc.). FIgure 7.2.2 shows the structure of a IPv4 header.

Offsets	Octet	0								1								2								3							
Octet	**Bit**	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31
0	**0**	Version				IHL				DSCP						ECN		Total Lenght															
4	**32**	Identification																Flags			Fragment Offset												
8	**64**	Time to Live								Protocol								Header Checksum															
12	**96**	Source IP Adress																															
16	**128**	Destination IP Adress																															
20	**160**	Options (if IHL>5)																															
24	**192**																																
28	**224**																																
32	**256**																																

Figure 7.2.2: Example of a header for an IPv4 packet. In this case, it has a lenght of 36 bytes.

Internet Protocol version 6 (IPv6) [12]

- Larger adress space: The length of IPv6 adresses is 128 bits, which is four times the length of IPv4 adresses. It offers a capacity of 2^{128} adresses.
- Multicasting: IPv6 acomplishes multicasting without using other protocols (such as IGMP for IPv4)
- Stateless address autoconfiguration (SLAAC): IPv6 hosts can configure themselves automatically when they are connected to a IPv6 network using the Neighbor Discovery Protocol via Internet Control Message Protocol version 6 (ICMPv6) router discovery messages. When a host is connectes for the first time, it sends a link-local router solicitation multicast request for its configuration parameters. Then, routers respond to the request with a router advertisement packet that contains Internet Layer configuration parameters.
- Network-layer security: Internet Protocol Security was developed for IPv6 before it was adapted for IPv4.
- Simplified processing by routers: Packet headers and the process of packet forwarding have been simplified, so packet processing by routers is more efficient. Headers now have a fixed length of 40 bytes, and may have an optional section aimed for options between the header section and the data section. Figure 7.2.3 shows the structure of a IPv6 header. IPv6 routers do not perform fragmentation.
- Mobility: Mobile IPv6 avoids triangular routing (unlike IPv4) and is as efficient as native IPv6.
- Options extensibility: IPv6 headers have astructure capable of extending the protocol in the future without affecting the core packet structure.
- Jumbograms: IPv4 limits packets to (2 power 16) - 1 octets per payload. A IPv6 node can handle packets of (2 power 32) - 1 octets (called jumbograms).

Offsets	Octet	0								1								2								3							
Octet	Bit	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31
0	0	Version				Traffic Class								Flow Label																			
4	32	Payload Length																Next Header								Hop Limit							
8	64	Source Adress																															
12	96																																
16	128																																
20	160																																
24	192	Destination Adress																															
28	224																																
32	256																																
36	288																																

Figure 7.2.3: Example of a header for an IPv6 packet.

7.2.2.2 Auxiliary protocols

Encapsulation service [13]

The Encapsulation Service is a service used to transfer data units that can not be directly transferred by the CCSDS Space Data Link Protocols. In order to be directly transferred by a Space Data Link Protocol, a data unit must have a Packet Version Number authorized by the CCSDS (a list of PVN authorized by CCSDS is contained in [14]). With the Encapsulation Service, data units that do not have an authorized VPN can be transmitted with Space Data Link Protocols. The data unit to be transmitted must be of an integral number of octets.

A user of the Encapsulation Service is identified by the combination of the following:

- A Packet Version Number (PVN) that indicates whether Space Packets (PVN=1) or Encapsulation Packets (PVN=8) are used for encapsulation,
- An Encapsulated Protocol Identifier (EPI), whichis either:
  - An Application Process Identifier (APID) defined in reference (if Space Packets are used).
  - a Protocol ID defined in section 4 of this document (if Encapsulation Packets are used).

The APIDs used by the Encapsulation Service must be registered as ‘reserved APIDs’ in [15]. The Protocol IDs used by the Encapsulation Service must be registered as ‘defined Protocol IDs’ in [16].

If the Data Unit is encapsulated in a Space Packet, the header format of the Space Packet is the same as the one used by Space Packet Protocol, only that the values of the parameters are restricted to some values. On the other hand, if the Data Unit is encapsulated in a Encapsulation Packet, a different header format will be used. This header have a length of 1-8 octets, and for the case of 8 octet it can be shown in Figure 7.2.4

	Bit							
Octet	0	1	2	3	4	5	6	7
0	Packet VersionNumber			Protocol ID			Length of length	
1	User defined fields				Protocol ID extension			
2	CCSDS defined field							
3								
4	Packet length							
5								
6								
7								

Figure 7.2.4: Example of a header for an Encapsulation Packet of maximum length. Some parameters may vary its length in other cases.

IP over CCSDS (IPoC) [17]

The IP over CCSDS is used to transfer IP Data Units over CCSDS Space Data Link Protocols. IP Data Units are encapsulated in Encapsulation Packets and sent through Space Data Link Protocols. IPoC uses the CCSDS Internet Protocol Extension (IPE) convention in conjunction with the CCSDS Encapsulation Service. The IPE convention is used to add IPE octecs at the beginning of a IP Data Unit, encapsulate the result in an Encapsulation Packet, and transmit it with a CCSDS Space Data Link Protocol. It is used because not all protocols that use an IP datagram have a Protocl ID used by the Encapsulation Packet.

IPoC adds a header at the beginning of the IP Data Unit, called IPE header. The sum of the IP Data Unit and the IPE header is the Data Unit used by the Encapsulation Service. In other words, for the Encapsulation Service, the IPE header and the IP Data Unit are a whole.

The structure of the IPE header will be the following. It must be of a length of an integral number of octets, with a minumum length of 1 octet. Each octet will be divided into two parts: the first seven bits (bits 0-6), and the least significant bit (LSB, bit 7). If more octets are added, the LSB of all octets except the last octet are set to '0'. The value of the IPE header is the decimal value of all the octets. The value of the IPE header must be one of the possible values in [18].

Internet Control Message Protocol (ICMP) [19]

The Internet Control Message Protocol (ICMP) is one of the main protocols of the TCP/IP protocol suite. It is used to send error messages to the source IP of the data packet. It is assigned IP protocol number 1. ICMP messages are typically used for diagnostic, control purposes or generated in response to errors in IP operations. They are processed differently than normal IP proceessing.

There are many types of control messages that the ICMP can send:

- Souce quench: Used to request the sender to decrease the rate of messages sent to a router.
- Redirect: Used to request the sender to send the data to another router.
- Time exceeded: Used by a gateway to inform the sender of a discarded datagram due to the time to life field reaching zero. It is also used to inform the sender that a fragment of a message has not been reassembled within the time limit
- Timestamp: Used for time syncronization. The sender sends the timestamp it last touched the packet (in miliseconds since midnight)
- Timestamp reply: Used to reply a timestamp. The reciever of the timestamp message replies the sender with the original timestamp, the timestamp when the message was recieved, and the timestamp when the reply was sent.
- Adress mask request: Used by a host to obtain the subnet mask of a router
- Adress mask reply: Used to reply the adress mass request returning the subnet mask.
- Destination unreachable: Used by the host or its inbound gateway to inform the client that the destination is unreachable.

Internet Protocol Security (IPsec) [20]

IPsec uses the following protocols to perform various functions;

- Authentication Headers (AH): Provides connectionless data integrity and data origin authentication for IP datagrams, and provides protection against replay attacks.
- Encapsulating Security Payloads (ESP): Provide confidentiality, data-origin authentication, connectionless integrity, an anti-replay service, and limited traffic-flow confidentiality.
- Security Associations (SA): Provides the bundle of algorithms and data that provide the parameters necessary for AH and ESP operations.

Protocol Independent Multicast (PIM) [21] [22]

There are four variants of PIM:

- PIM Sparse Mode (PIM-SM): It builds unidirectional shared trees rooted at a rendezvous point (RP) per group, and optionally creates shortest-path trees per source. It is called sparse-mode because it is suitable for groups where low percentage of the nodes will subscribe to the multicast session.
- PIM Dense Mode (PIM-DM): It uses dense multicast routing. It builds shortest-path trees by flooding multicast traffic domain wide, and then pruning back branches of the tree where no receivers are present. Dense mode is ideal for groups where many of the nodes will subscribe to receive the multicast packets.
- Bidirectional PIM: It builds shared bi-directional trees. It never builds a shortest path tree, so may have longer end-to-end delays than PIM-SM.
- PIM Source-Specific Multicast (PIM-SSM): It builds trees that are rooted in just one source, offering a more secure model for a limited amount of applications (mostly broadcasting of content). In SSM, an IP datagram is transmitted by a source S to an SSM destination address G, and receivers can receive this datagram by subscribing to channel (S,G).

7.2.2.3 Routing protocols

Enhanced Interior Gateway Routing Protocol (EIGRP) [23]

The additional two tables are:

- Neighbour table. It stores the IP adress of the routers that have a direct connection with this router. If a router is connectedto another with an intermediate router, it will not be recorded in this table.

- Topology table. It keeps record of routes that has learned from neightbouring router tables, and also records the distance (number of intermediate routers) of each route, the feasible successor and the successors (other routes that have the same destination and are loop free). Routes in this table are either labelled as "passive" or "active". Passive means that EIGRP has determined the path for the specific route and has finished processing. Active means that EIGRP is still trying to calculate the best path for the specific route. The router dones not use the routes in this table. A route in this table will be inserted in the routing talbe when is marked as passive, is not a feasible successor and does not have a higher distance than an equivalent path

If there is a change in the network (a link fails, or a router is disconnected), the path becomes unavaiable, and is removed from the routing table. The routing table of a router will be updated, and only the changes since the previous update will be transmitted to the neightbouring routers. The information about the changes in the routing table is not transmitted periodically, but only when a change actually occurs.

EIGRP supports the following features:

- Support for Classless Inter-Domain Routing (CIDR) and variable length subnet masking. Routes are not summarized at the classful network boundary unless auto summary is enabled.
- Support for load balancing on parallel links between sites.
- The ability to use different authentication passwords at different times.
- MD5 authentication between two routers.
- Sends topology changes, rather than sending the entire routing table when a route is changed.
- Periodically checks if a route is available and propagates routing changes to neighboring routers if any changes have occurred.
- Runs separate routing processes for Internet Protocol (IP), IPv6, IPX and AppleTalk through the use of protocol-dependent modules (PDMs).

EIGRP does not operate using the Transmission Control Protocol (TCP) or the User Datagram Protocol (UDP). This means that EIGRP does not use a port number to identify traffic. Rather, EIGRP is designed to work on top of layer 3. Since EIGRP does not use TCP for communication, it implements Cisco's Reliable Transport Protocol (RTP) to ensure that EIGRP router updates are delivered to all neighbors completely.

Open Shortest Path First (OSPF) [24] [25]

OSPF supports complex networks with multiple routers, including backup routers, to balance traffic load on multiple links to other subnets. Routers form adjacencies when they have detected each other. This detection is initiated when a router identifies itself in a Hello protocol packet. Upon acknowledgment, this establishes a two-way state and the most basic relationship. The routers in an Ethernet or Frame Relay network select a Designated Router (DR) and a Backup Designated Router (BDR) which act as a hub to reduce traffic between routers. OSPF establishes and maintains neighbor relationships for exchanging routing updates with other routers. The neighbor relationship table is called an adjacency database. Two OSPF routers are neighbors if they are members of the same subnet and share the same area ID, subnet mask, timers and authentication. OSPF adjacencies are formed between selected neighbors and allow them to exchange routing information. Two routers become adjacent if at least one of them is Designated Router or Backup Designated Router (on multiaccess type networks), or they are interconnected by a point-to-point or point-to-multipoint network type.

OSPF does not carry data via a transport protocol. Instead, OSPF forms IP datagrams directly, packaging them using protocol number 89 for the IP Protocol field. OSPF defines five different message types, for various types of communication:

- Hello: It is used to allow a router to discover other adjacent routers on its local links and networks. The messages establish adjacencies between neighboring devices. uring normal operation, routers send hello messages to their neighbors at regular intervals. If a router stops receiving hello messages from a neighbor, after a set period the router will assume the neighbor has gone down.
- Database Description: It contain descriptions of the topology of the autonomous system or area. They convey the contents of the link-state database (LSDB) for the area from one router to another. Communicating a large LSDB may require several messages to be sent.
- Link State Request: These messages are used by one router to request updated information about a portion of the LSDB from another router. The message specifies exactly which link about which the requesting device wants more current information.
- Link State Update: These messages contain updated information about the state of certain links on the LSDB. They are sent in response to a Link State Request message, and also broadcast or multicast by routers on a regular basis. Their contents are used to update the information in the LSDBs of routers that receive them.

- Link State Acknowledgment: These messages provide reliability to the link-state exchange process, by explicitly acknowledging receipt of a Link State Update message.

Routing Information Protocol (RIP) [26] [27]

Rules:

- If there are no route entries matching the one received then the route entry is added to the routing table automatically, along with the information about the router from which it received the routing table.
- If there are matching entries but the hop count metric is lower than the one already in its routing table, then the routing table is updated with the new route.
- If there are matching entries but the hop count metric is higher than the one already in its routing table, then the routing entry is updated with hop count of 16 (infinite hop). The packets are still forwarded to the old route. A Holddown timer is started and all the updates for that route from other routers are ignored. If after the Hold-down timer (per deffect 180 seconds) expires and still the router is advertising with the same higher hop count then the value is updated into its routing table. Only after the timer expires, the updates from other routers are accepted for that route.

If the Invalid timer (per deffect 180 seconds) expires and a routing entry has not been updated, the hop counter of that route will be set to 16, marking the route as invalid. Then, if the Flush timer (per deffect 240 seconds) expires, the invalid route entry will be removed

7.2.3 Final structure

As the protocols have already been chosen, it is time to establish how will be the headers of the different protocols.

The IPv6 header will depend greatly on the protocol of the upper layers, or the auxiliar protocol (OSPF, ICMPv6). The main parameters of the IPv6 header, that can be seen in Figure 7.2.3, are the following:

- **Version** Current version of IP, which for IPv6 is 6 (bit sequence 0110).

- **Traffic Class**. The bits of this field hold two values. The 6 most-significant bits are used for differentiated services, which is used to classify packets. The remaining two bits are used for ECN; priority values subdivide into ranges: traffic where the source provides congestion control and non-congestion control traffic.
- **Flow Label**. The flow label when set to a non-zero value now serves as a hint to routers and switches with multiple outbound paths that these packets should stay on the same path so that they will not be reordered.
- **Payload Length**. The size of the payload in octets, including any extension headers. The length is set to zero when a Hop-by-Hop extension header carries a Jumbo Payload option.
- **Next Header**. Specifies the type of the next header. This field usually specifies the transport layer protocol used by a packet's payload. When extension headers are present in the packet this field indicates which extension header follows. The values are shared with those used for the IPv4 protocol field, as both fields have the same function (see List of IP protocol numbers in [28]).
- **Hop Limit**. This value is decremented by one at each intermediate node visited by the packet. When the counter reaches 0 the packet is discarded.
- **Source Address**. The IPv6 address of the sending node.
- **Destination Address**. The IPv6 address of the destination node.

It has been stated that, since Astrea network is a private network that will not be connected to the Internet, IP adresses will be arbitrary assigned to the nodes of the network.

For the IPoC heaver, the value for IPv6 datagrams is 87, so the header of OPoC will be 01010111

For the Encapsulation Service, depending of the length of the data unit transmitted, the header will very. For data units up to 65531 octets, the Encapsulation Service header will be the following: 11101010-00000000-XXXXXXXX-XXXXXXXX, where XXXXXXXX-XXXXXXXX is the binary number of the total length of the Encapsulation Packet, including the Encapsilation Packet header.

7.3 Layer 4: Transport and Session

7.4 Session and Transport Layer

7.4.1 User Datagram Protocol (UDP)

The User Datagram Protocol (UDP) is a connectionless, unreliable transport protocol. The only new feature regarding IP is that it provides process-to-process communication instead of host-to-host communication, and performs a very limited error checking. It might seem a powerless protocol, but its main point is that is a very simple protocol using a minimum of overhead. Therefore, if a process wants to send a small message and no extremely reliability is required, UDP is a good choice.

Nevertheless, regarding the aim of this project, it is unacceptable to use UDP, since reliability is a key factor and must be taken into account.

7.4.2 Stream Control Transmission Protocol (SCTP)

The Stream Control Transmission Protocol is a new reliable, message-oriented transport layer protocol. Nevertheless, it has been designed and implemented mostly for Internet applications, such as IUA or SIP. But precisely it does not fit the goal of this project.

Therefore, as there is a better choice (which will be deeply and widely explained in the following section), this protocol will not be considered.

7.4.3 Transmission Control Protocol (TCP)

The Transmission Control Protocol is again a process-to-process protocol. Consequently it uses port numbers. The main difference with the UDP is that TCP is a connection-oriented protocol, which means that creates a virtual connection between two TCP's in order to send data. Moreover, TCP uses flow and error control mechanisms. It is then a more reliable protocol that UDP. It adds connection-oriented and reliability features to the services of IP.

This will be the protocol chosen for this project, so it will be explained in detail in this section.

7.4.3.1 TCP Services

Process-to-process communication: Like UDP, TCP provides this type of communication, using port numbers. In the following image there are the main well-known port numbers used by TCP.

Port	*Protocol*	*Description*
7	Echo	Echoes a received datagram back to the sender
9	Discard	Discards any datagram that is received
11	Users	Active users
13	Daytime	Returns the date and the time
17	Quote	Returns a quote of the day
19	Chargen	Returns a string of charaters
20	FIP, Data	File Transfer Protocol (data connection)
21	FIP, Control	File Transfer Protocol (control connection)
23	TELNET	Tenninal Network
25	SMTP	Simple Mail Transfer Protocol
53	DNS	Domain Name Server
67	BOOTP	Bootstrap Protocol
79	Finger	Finger
80	HTTP	Hypertext Transfer Protocol
111	RPC	Remote Procedure Call

Stream Delivery Service: as has been mentioned before, TCP, unlike UDP, is a stream-oriented protocol. UDP does not recognize any relationship between the datagrams. TCP, in contrast, allows the sending process to deliver data as a stream of bytes and allows the receiving process to obtain data as a stream of bytes. A way of explaining this would be an environment in which the two processes seems to be linked by and imaginary "tube" that carries the data across the Internet. The sending process produces the stream of bytes and the receiving process consumes them. This is, the first writes and the last reads.

Sending and Receiving Buffers: Since the sending and receiving processes might not write or read data at the same speed, there is a need for storage in TCP. Therefore, TCP includes two buffers, the sending buffer and the receiving buffer. A deeper look into those buffers can be performed by looking at the bibliography.

Full-Duplex Communication: TCP allows full-duplex service, so that data can flow in both directions at the same time. Each TCP has a sending and receiving buffer, and segments move in both directions. This feature is very important for the goal of this project.

Segments: Although buffering solves the problem of different speeds of producing and consuming, there is still one impotant feature to be discussed. The data needs to be sent in packets, not as a endless stream of bytes. Therefore, TCP groups a number of bytes together into a packet called a segment. A header is added to each segment for control purposes.

7.4.3.2 TCP features

In order to provide the services that have been explained, TCP has some features that will be briefly discussed.

Numbering Systems

TCP keeps track of the segments being transmitted or received, using the header previoulsy discussed. There are in addition two fields, the sequence number and the acknowledgement number, which refer to the byte number, not the segment number.

TCP numbers all data bytes that are transmitted in a connection. Numbering is independent in each direction. When TCP receives bytes of data from a process, it stores them in the sending buffer and numbers them. Typically, it generates randomly a number between 0 and $2^{32} - 1$ for the number of the first byte. For example, if the random number happens to be 1427 and the total data to be sent are 5000 bytes, the bytes are numbered from 1427 to 6426. This system is used for flow and error control.

After the bytes have been numbered, TCP assigns a sequence number to each segment that is being sent. The sequence number for each segment is the number of the first byte carried in that segment. This is, the value in the sequence number field of a segment defines the number of the first data byte contained in that segment.

The value of the acknowledgement field in a segment defines the number of the next byte a party expects to receive. It is a cumulative number.

Flow Control

TCP provides flow control, which means that the receiver can control the amount of data that if to be sent by the sender. The purpose of this is to avoid over-whelmed receivers.

Error Control

In order to provide a reliable service, TCP implements an error control mechanism. It considers a segment as the unit of data for error detecting, even though there is also a byte-oriented control mechanism.

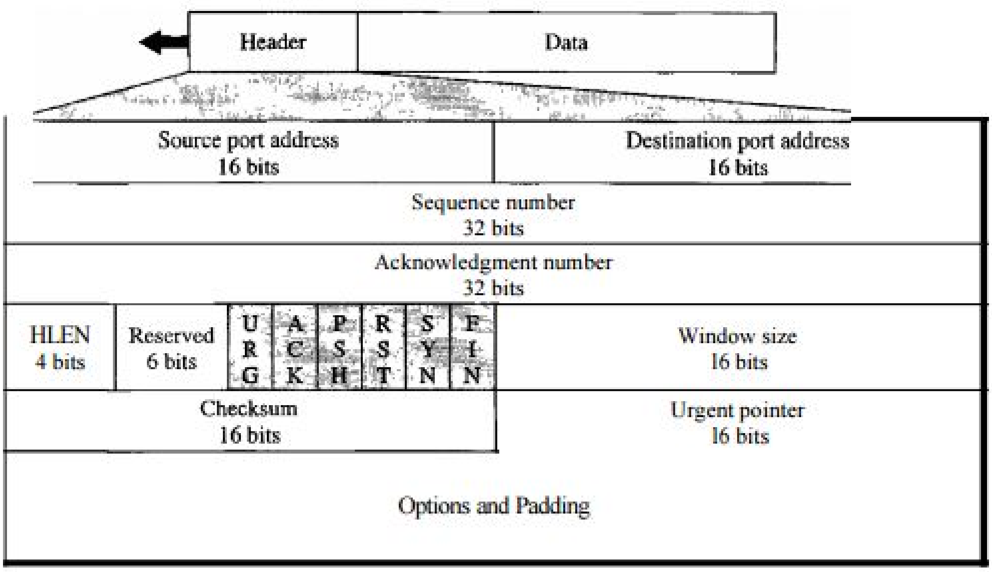
Congestion Control

TCP also takes into account congestion in the network, by the detenning of the flow depending on the level of congestion in the network.

Segment

As has been explained before, a packet in TCP is called a segment. The aim of this point is to explain in detail what a segment is and how its structure is.

The typical format of the segment is shown in the next figure.



The segment consists of a 20 to 60-byte header, followed by data frim the application program. The byte is 20-byte long if there are no options, and up to 60-bytes if there are options.

The main parts of the format are to be discussed in the following lines.

Source Port Adress

This is a 16-bit field that states the port number of the application program in the host that is sending the segment.

Destination Port Adress

It is also a 16-bit that defines the port number of the application program in the host that is receiving the segment.

Sequence Number

This 32-bit field defines the number assigned to the first byte of the data contained in the segment considered. This numeration has been previously explained.

Acknowledgement Number

This is a 32-bit field that defines the byte number that the receiver of the segment is expecting to receive from the other party. If the receiver of the segment has successfully received byte number x, it defines x + I as the acknowledgement number.

Header Length

A 4-bit field that indicates the number of 4-byte words in the TCP header. As seen, the length of the header can be between 20 and 60 bytes. Then, the value of this field can be between 5 and 15 (since $5 \times 4 = 20$, and $15 \times 4 = 60$).

Reserved

This is a 6-bit field reserved for future usage.

Control

This field defines 6 different control bits or flags. One or more of those bits can be set at a time.

Window Size

This fiels defines the size of the window, in bytes, that the other party must maintain. Since the length of this field is 16 bits, the maximum size of the windows is $2^{16} = 65535$ bytes.

Urgent Pointer

Another 16-bit field, which is only valid if the urgent flag is set, which means that the segment contains urgent data. In actually defines the number that must be added to the sequence number to obtain the number of the last urgent byte in the data section of the segment.

Options

As has been explained, there can be up to 40 bytes of optional information in the TCP Header. This is the purpose of this last field.

Adaptation to space needs

TCP was estblished for wired connections initially. Therefore, in order to be eligible for the purpose of this project, it is highly recommended that some slight modifications are done. The Space Communications Protocols Specification (SCPS) defines a set of revisions to the protocols to enable them to operate properly. This is, SCPC-TCP becomes an "upgraded" TCP, specially designed for space application.

With SCPS, TCP the bandwidth of an existing link will be utilized to a significantly higher percentage and more efficiently. It also supports end-to-end communications between applications and is designed to meet the needs of a broad range of space missions.

This is all achieved because of an extension that is added to the header shown before. This extension header is shown next. Each line is a octet of bits; i.e, 8 bits:

SCPS Option Type (20)							
SCPS Option Length							
BETS	SN1	SN2	Com	NL TS			ext

7.5 Global Overview

For the sake of clarification, all the elected options are going to be put together obtaining the desired fully desgined **protocol stack**.

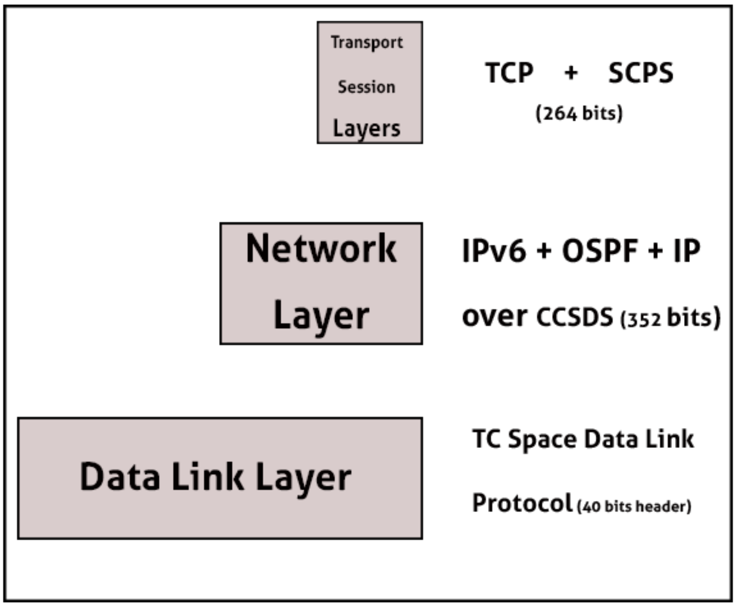


Figure 7.5.1: Overall space communication protocol stack

In total, the overhead is **656 bits**, whith conservative calculations. Hence, the quantity is negligible in comparison to the data rate.

Chapter 8

Ground Segment Protocols

8.1 Ground Segment protocols

8.1.1 File Transfer Protocol (FTP)

The File Transfer Protocol (FTP) is a standard network protocol used to transfer computer files between a client and server on a computer network.

FTP is built on a client-server model architecture and uses separate control and data connections between the client and the server. FTP users may authenticate themselves with a clear-text sign-in protocol, normally in the form of a username and password, but can connect anonymously if the server is configured to allow it. For secure transmission that protects the username and password, and encrypts the content, FTP is often secured with SSL/TLS (FTPS). SSH File Transfer Protocol (SFTP) is sometimes also used instead, but is technologically different.

Setting up an FTP control connection is quite slow due to the round-trip delays of sending all of the required commands and awaiting responses, so it is customary to bring up a control connection and hold it open for multiple file transfers rather than drop and re-establish the session afresh each time.
See more about FTP in [?]

8.1.2 Secure Shell (SSH)

Secure Shell (SSH) is a cryptographic network protocol for operating network services securely over an unsecured network. The best known example application is for remote login to computer systems by users.

SSH provides a secure channel over an unsecured network in a client-server architecture, connecting an SSH client application with an SSH server. Common applications include remote command-line login and remote command execution, but any network service can be secured with SSH.

See more about SSH in [?]

8.1.3 Simple Mail Transfer Protocol (SMTP)

Simple Mail Transfer Protocol (SMTP) is an Internet standard for electronic mail (email) transmission. Email is submitted by a mail client (mail user agent, MUA) to a mail server (mail submission agent, MSA). The MSA delivers the mail to its mail transfer agent (mail transfer agent, MTA). Often, these two agents are instances of the same software launched with different options on the same machine. Local processing can be done either on a single machine, or split among multiple machines; mail agent processes on one machine can share files, but if processing is on multiple machines, they transfer messages between each other using SMTP, where each machine is configured to use the next machine as a smart host. Each process is an MTA (an SMTP server) in its own right.

SMTP is a connection-oriented, text-based protocol in which a mail sender communicates with a mail receiver by issuing command strings and supplying necessary data over a reliable ordered data stream channel. An SMTP session consists of commands originated by an SMTP client (the initiating agent, sender, or transmitter) and corresponding responses from the SMTP server (the listening agent, or receiver) so that the session is opened, and session parameters are exchanged.

See more about SCTP in [**?**]

8.1.4 Hypertext Transfer Protocol (HTTP)

The Hypertext Transfer Protocol (HTTP) is an application protocol for distributed, collaborative, hypermedia information systems. HTTP is the foundation of data communication for the World Wide Web.

HTTP functions as a request–response protocol in the client–server computing model. A web browser, for example, may be the client and an application running on a computer hosting a website may be the server. The client submits an HTTP request message to the server. The server, which provides resources such as HTML files and other content, or performs other functions on behalf of the client, returns a response message to the client. The response contains completion status information about the request and may also contain requested content in its message body.

8.1.5 Transport Layer Security (TLS)

Transport Layer Security (TLS)is a cryptographic protocol that provides communications security over a computer network. Several versions of the protocol find widespread use in applications such as web browsing, email, Internet faxing, instant messaging, and voice-over-IP (VoIP). Major websites use TLS to secure all communications between their servers and web browsers.

The Transport Layer Security protocol aims primarily to provide privacy and data integrity between two communicating computer applications. When secured by TLS, connections between a client and a server have one or more of the following properties:

- The connection is private (or secure) because symmetric cryptography is used to encrypt the data transmitted. The keys for this symmetric encryption are generated uniquely for each connection and are based on a shared secret negotiated at the start of the session. The server and client negotiate the details of which encryption algorithm and cryptographic keys to use before the first byte of data is transmitted . The negotiation of a shared secret is both secure (the negotiated secret is unavailable to eavesdroppers and cannot be obtained, even by an attacker who places themselves in the middle of the connection) and reliable (no attacker can modify the communications during the negotiation without being detected).
- The identity of the communicating parties can be authenticated using public-key cryptography. This authentication can be made optional, but is generally required for at least one of the parties (typically the server).
- The connection ensures integrity because each message transmitted includes a message integrity check using a message authentication code to prevent undetected loss or alteration of the data during transmission.

8.1.6 Hypertext Transfer Protocol Secure (HTTPS)

HTTPS is a protocol for secure communication over a computer network which is widely used on the Internet. HTTPS consists of communication over Hypertext Transfer Protocol (HTTP) within a connection encrypted by Transport Layer Security. The main motivation for HTTPS is authentication of the visited website and protection of the privacy and integrity of the exchanged data.

See more about HTTP and HTTPS in [**?**]

8.2 Delivery of the data method

The advantages and drawbacks of each of the systems are:

- **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. The advantages are:
  - It would have a really friendly use for the costumer.
  - It could include friendly information for the user us who we are, how to contact, FAQs, etc.
  - It could be very automatized.
  - The information could be protected with the adequate security protocols.
  - The client would not need any special software.

  The disadvantages are:
  - The web would be vulnerable to some type of attacks or problems that would compromise the data. This could aviod the communication between the user and the network.
  - It would need several maintenance.
  - There would be some type of data, like videos and photos, which the client would want to download as a file. So the web would have to be complemented with a file transfer protocol.
  - The web would have to be designed.
- **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. The advantages are:
  - It would be very secure and stable.
  - The mail could not fall as a web does.
  - The client would not need any special software.
  - The information could be sent and received as a text or as a file.

  The disadvantages are:
  - It could not be automatized, and this make it inefficient.

  - It is not very friendly to use for a client.
  - If there is some information missing in the request the client would have to wait for an answer and then complete 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 advantages are:

  - It would be really friendly use for the costumer.
  - It would be really secure and stable.
  - It could include friendly information for the user as: who we are, how to contact, FAQs, etc.
  - The information could be sent and received as a text or a files.

  The disadvantages are:

  - It would need to be downloaded and installed.
  - It would need some maintenance.
  - It would need to be designed.

Part IV

ANNEX IV: Ground Segment Design

Chapter 9

Design of the Ground Segment

9.1 Study of localization of Ground Stations

In the following lines, a study of the location of the Ground Sations will be carried in order to know the optinal location for them. As it has been stated in the report, the study has been done using a Matlab code that can be found in Atachment XXXXXXX.

9.1.1 Latitude analysis

Is easy to see that the effect of changing the latitude is practically independent for the longitude. For this reason, the links during the day for a given longitude are studied independently of the latitude and viceversa. Doing the analysis for latitudes between 0° and 90° during 2 days, with 5 minutes time-step, this are the results:

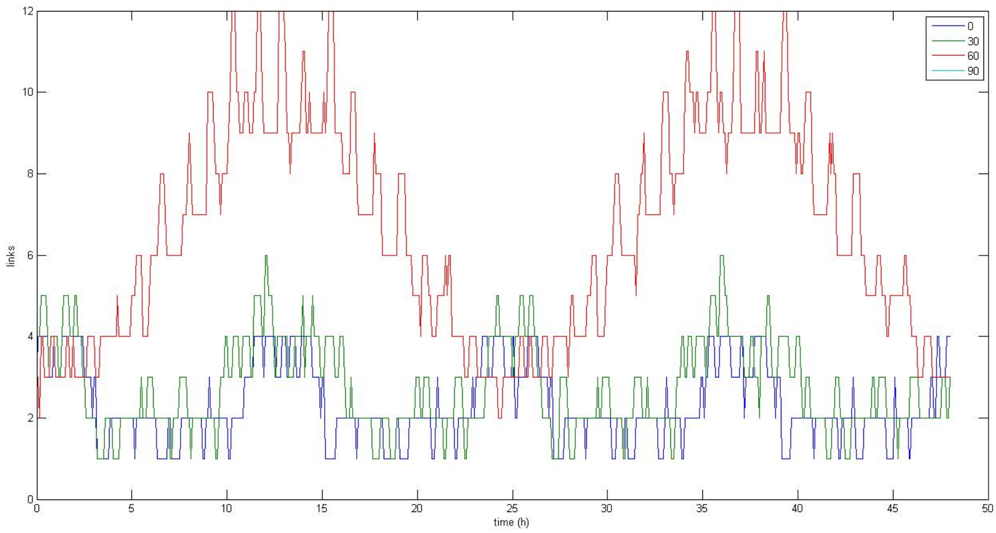


Figure 9.1.1: Links vs time for latitudes from 0° to 90°

As is shown in Figure 1.1, the behaviour is not constant during the day. For every day there is a peak and a valley. This is produced for the cylindrical asymmetry of the constellation. It can also be seen that the pole is not covered. This fact was considered and assumed at the design of the constellation since it doesn't involve any problem at the performance of the system. It can also be seen that for an equatorial latitude there is always 1 link, at least. The equator is the most critical place because is where satellites from different planes are more separated. Global coverage can be ensured, but is important to appreciate that for higher latitudes the coverage is better.

Doing the same analysis but for negative latitudes, the following results are obtained:

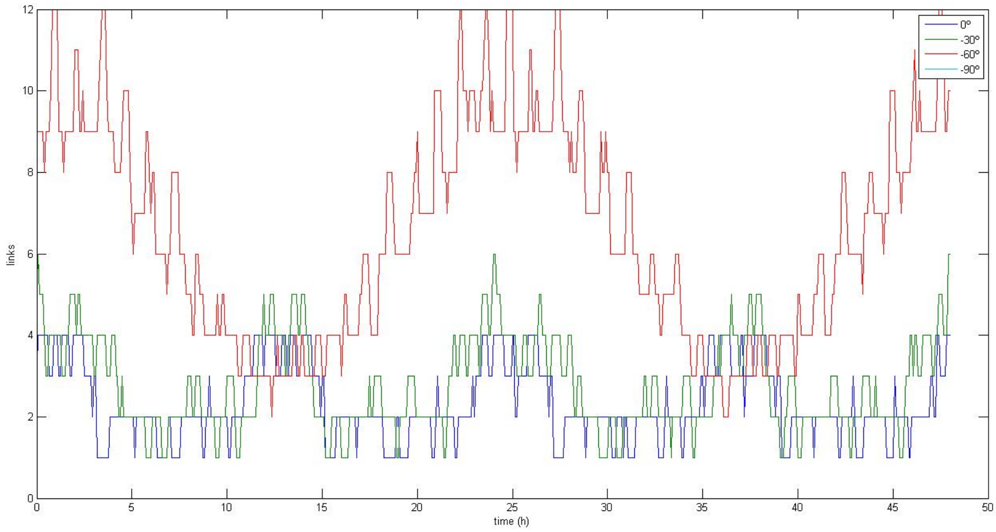


Figure 9.1.2: Links vs time for latitudes from 0º to -90º

Comparing the results of Figure 1.2 with the ones of Figure 1.1 it is seen that they are practically the same but with an offset of 12 hours. They are also seen small local deviations, but these are not much significant because of the time-step. This time-step is of 5 minutes for a first sight of the tendencies, and it do not allow extremely precise results.

Taking into account that the results of positive latitudes can be extrapolated to negative ones, the rest of the analysis will be done only for positive latitudes.Is important to know at which latitude, close to the poles, the coverage is lost due to the geometry of the constellation.

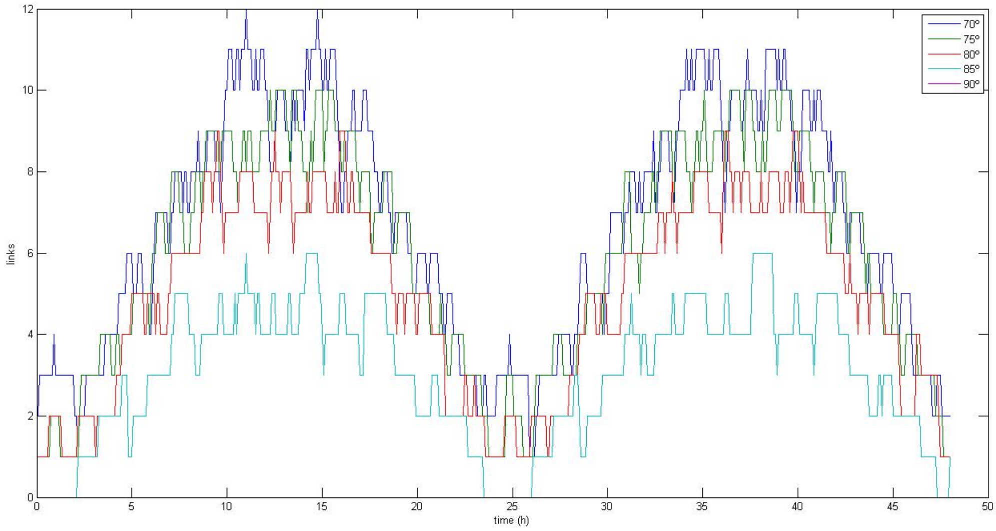


Figure 9.1.3: Links vs time for latitudes from 70° to 90°

It is seen that over 80° of latitude the system starts to lose coverage. It does not cause any problem because there are not inhabited zones over +80° or under -80°. For situating the Ground Stations it has to be considered this restriction.

Now, the latitudes that can provide more links are, around 60°:

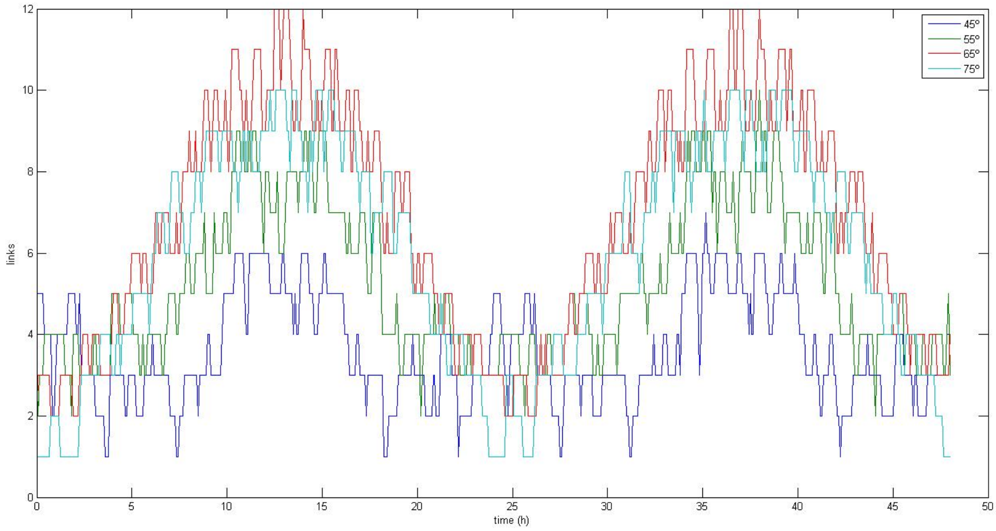


Figure 9.1.4: Links vs time for latitudes from 45° to 75°

As it can be seen in Figure 1.4, the optimal latitude must be between 55º and 75º. Expanding the analysis:

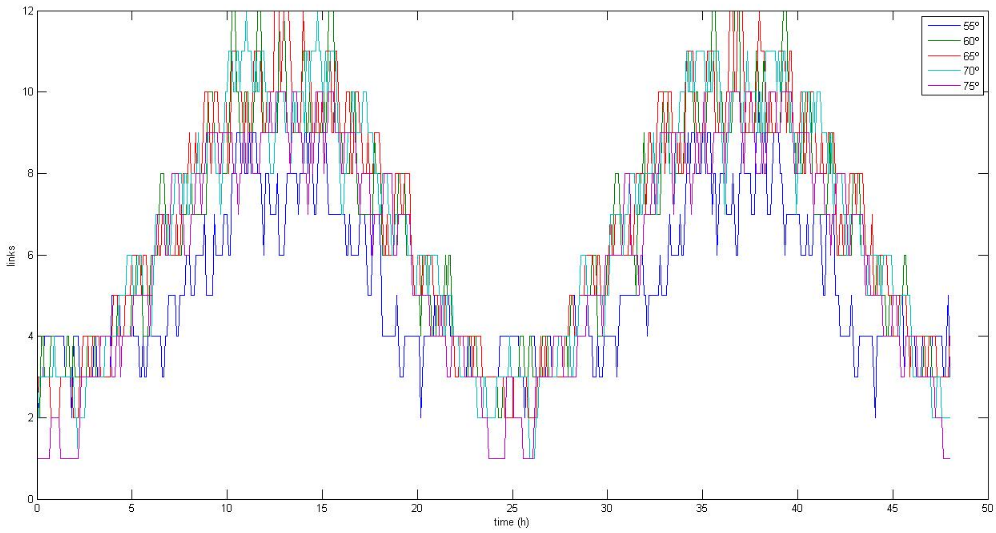


Figure 9.1.5: Links vs time for latitudes from 55º to 75º

The better performance is registered around 60º and 65º. Figure 1.5 suggest that between 50º and 60º there is always at least 1 link. But looking it carefully, at the hour 37, there is a local deviation to 0 links. This requires a more accurate analysis decreasing the time-step. For 30 seconds time-step:

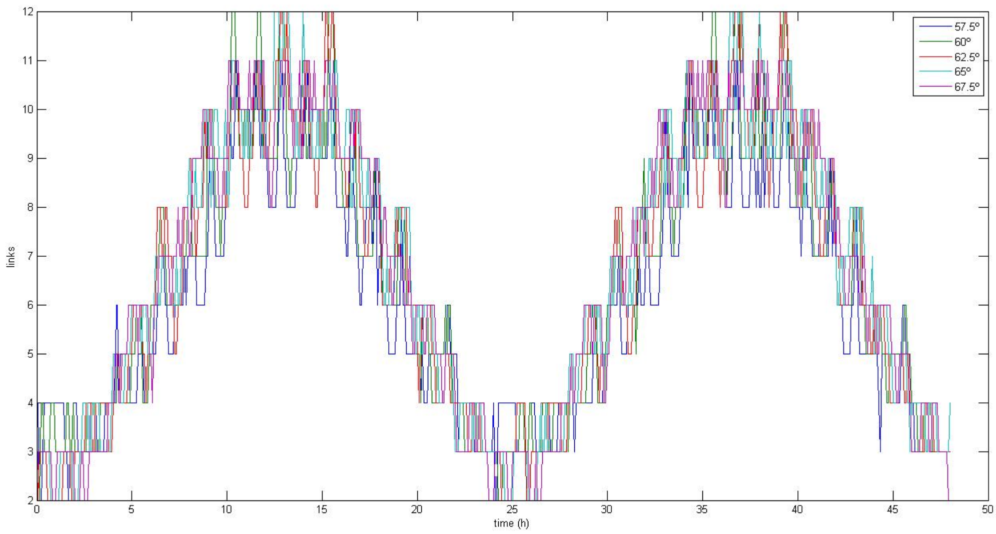


Figure 9.1.6: Links vs time for latitudes from 57.5º to 67.5º

In Figure 1.6 there is no problem with the coverage. For ensuring the results and to avoid possible loses of links locally in time, the same ragne of latitudes is analyze with a smaller time-step.

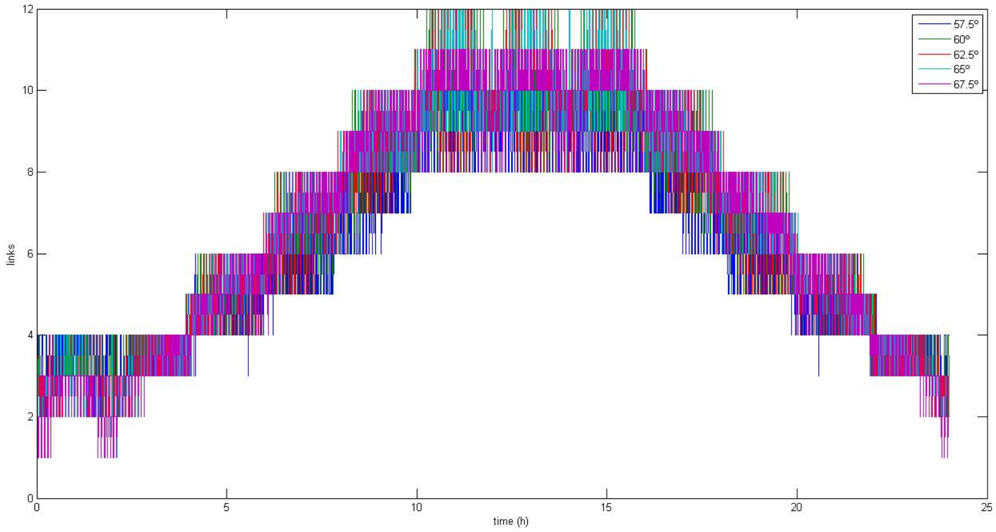


Figure 9.1.7: Links vs time for latitudes from 57.5° to 67.5° with 30 seconds time-step

It can be seen that between 65° and 67.5° the system loses the 2nd link and for a wile the station would be connected only to 1 satellite. It is optimum to place the stations between +57.5° and +62.5° of latitude. In order to verify the results for the opposite latitudes:

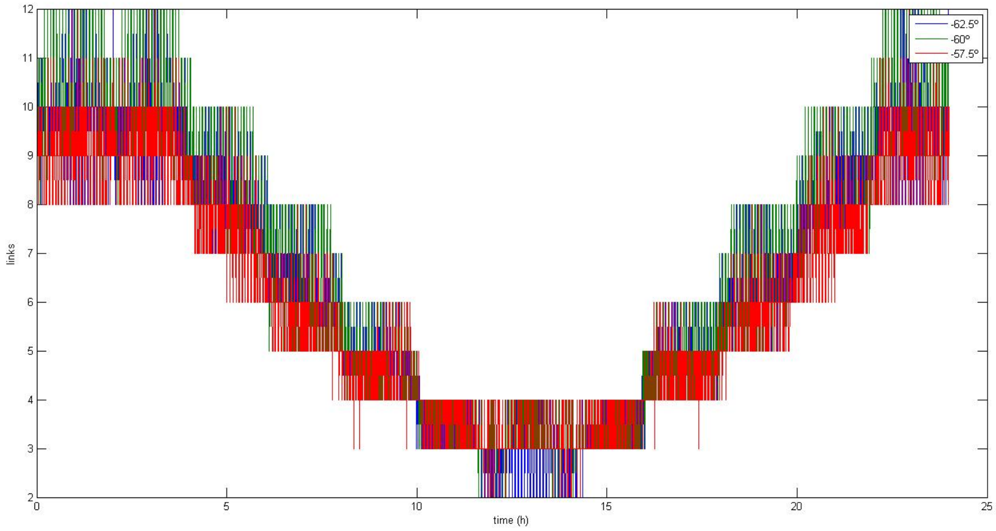


Figure 9.1.8: Links vs time for latitudes from -62.5° to -57.5° with 30 seconds time-step

In conclusion, the optimum latitudes for the Ground Station are:

- Between -62.5° and -57.5°
- Between +57.5° and +62.5°

9.1.2 Longitude analysis

It is intuitive to think that the effect of changing the longitude is delaying the evolution of the coverage. This effect is verified by the algorithm:

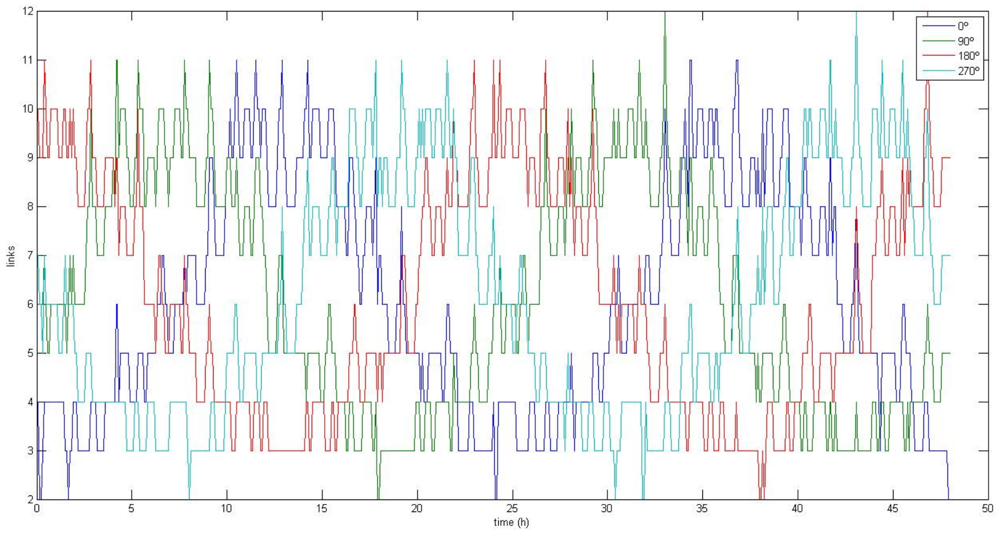


Figure 9.1.9: Links vs time for longitudes from 0° to 270°

As it is seen in Figure 1.8 the delay has a reason of 3 hours for every 45° of longitude. This effect can be used in order to optimize the performance of the Ground Stations. During the day every station will have a peak and a valley in the coverage. Placing the stations with a relative longitude of 120° would ensure that when one is at the valley another one is at the peak:

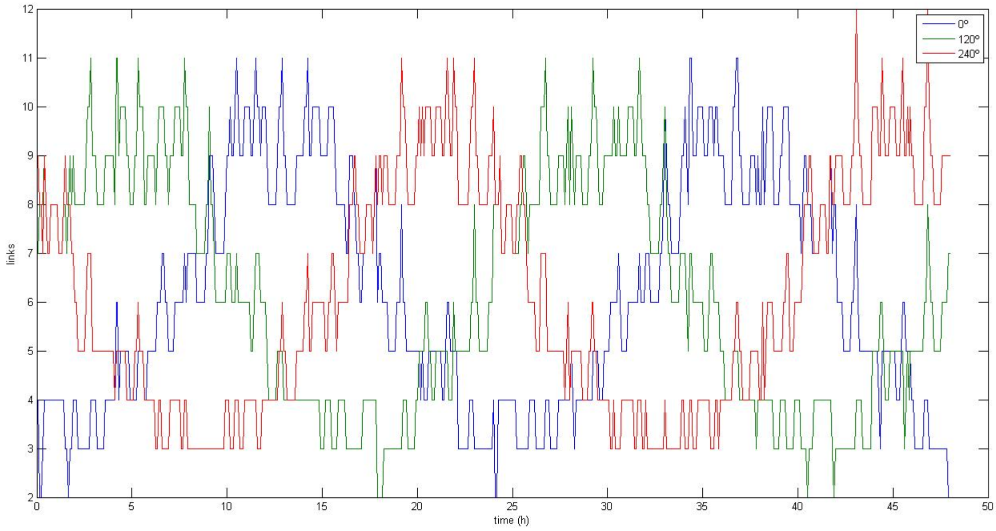


Figure 9.1.10: Links vs time for longitudes of 0°, 120° and 240°

In conclusion, the Ground Stations should be separated 120° longitude between them. It has to be taken into account that this analysis is done for stations at the same latitude. A Ground Station in a given latitude has the same coverage behaviour as an other one at the opposite latitude and 180° of longitude away. To exemplify, in the following table coordinates of equivalent places from the Ground Station point of view are showed.

	GS1		GS2		GS3	
	Latitude	Longitude	Latitude	Longitude	Latitude	Longitude
Option 1	55	0	55	120	55	240
Option 2	-55	180	55	120	55	240
Option 3	55	0	-55	300	55	240
Option 4	55	0	55	120	-55	60
Option 5	-55	180	-55	300	55	240
Option 6	-55	180	55	120	-55	60
Option 7	55	0	-55	300	-55	60
Option 8	-55	180	-55	300	-55	60

Table 9.1.1: Equivalent coordenates

9.2 Study of annual costs

9.2.1 Energy and Maintenance

In this section the maintenance of the ground stations and the control center, which is located in Terrassa, will be explained and its costs will be approximated. Is important to notice that the prices are not exact numbers, but just an approximation of the real value of the costs. For these reasons, some of the calculations as for example the cost of Internet in Scotland or Canada, is done using the value of the Internet cost in Spain, as they will be of the same order but can have a slow variation.

9.2.1.1 Mission Control Center

The control center will be located in Terrassa and it will act as a center from which the activity of the Astrea group will be monitored. The most important cost in this building will be the energy consumption. To approximate the energy consumption the energy use intensity (EUI) can be used. The EUI is a recommended benchmark metric for all type of buildings and tells the amount of energy used in buildings per meter square during one year. The EUI is calculated depending on the type of building (hospital, school, etc). The type of building of the control center can be considered as a set of offices, because the most important features of it will be the computers and the internet communications. Taking as a reference an usual office floor from a building, the average surface it occupies is 500 m^2. The EUI has been obtained from [29] and is 212 kWh/m^2. The cost of a kWh according to [30] is of 0,141033 €/kWh, taking into account that the main type of consumption is of electricity. Then, doing the calculation:

$$212 \cdot 500 \cdot 0,141033 = 14960 \tag{9.2.1}$$

This is the cost of the energy consumed. However, the fixed term has also to be taken into account. This term is of 3,170286 €/month/kW. It does not depend on the kW consumed, but the ones that have been contracted. Considering a tariff of 11,5 kW, the cost per year will be of 440 €. Then, the total cost of electricity per year is 15400 €. This is the cost without taxes. Taxes applied to the consume of electricity in Spain are the excise duty on electricity (4,864%) and the value added tax (21%). With these data, the resulting cost is of 20540 €.

Another important cost is the one of the maintenance. The maintenance include cleaning service, industrial maintenance and possible failures of the systems that would need to be repaired. There are companies that offer these services, so to know the cost of the maintenance a research on the market will be done. In most of these companies, no available information about the cost can be found if no information about the exact needs is provided. However, there are some of them that have few standards tariff that can be used. The maintenance will be divided into two: informatic maintenance and cleaning service. The cost of informatic maintenance for a business extracted from [31] is of 206 €per month. So in one year the cost will be of 2500 €. For the cleaning service, the average market cost is of 10 €per hour according to [32], for contracted maintenance. If there are 250 laborable days and every day there is 2 hour of cleaning service, the total cost of it is of 5000 €.

The other cost that has to be taken into account is the Internet connexion. To give an approximation of this cost, some Internet providers are consulted and the resulting price is of 55 €/month, that are 660 €per year.

In the following table the results are exposed:

Concept	**Cost€**
Energy:	20540
Maintenance:Informatics	2500
Maintenance:Cleaning	5000
Internet connexion	660
Total cost	**28700**

Table 9.2.1: Costs per year for the control centre

9.2.1.2 Ground Stations

The same procedure as the previous one will be done. The costs of maintenance (informatics and cleaning) and of the Internet connexion will be the same, but the difference will be on the energy consumed. The EUI of the site itself, without taking into account the antennas, will also be the same: 212 kWh/m^2. The surface of the building of the ground station will be of approximately 100 m^2, enough for the comfortability of 4 people working there. Then, the energy consumption per year will be of 21200kWh. The consumption of the antennas has also to be taken into account. Each antenna consumes 770 W approximately and each GS has four antennas, considering that they will be working 24 h/day during the whole year, the consumtion during one year can be calculated.

$$\frac{4 \cdot 770 \cdot 24 \cdot 365}{1000} = 26981 kWh/year \tag{9.2.2}$$

Then the total consumption in kWh of one ground station is:

$$26981 + 21200 = 48181 kWh/year \tag{9.2.3}$$

Now the cost of the kWh is needed, and it depends on the countries, so in the following lines the cost will be calculated for each of the ground stations. The cost of kWh supplied has been extracted from [33] and is an average because it depends on many factors as for example the company selected, the type of tariff, the fixed term, taxes, etc.

Canada In Canada, the average cost of 1kWh is of 10 US cents, that are 0,0945 €. Doing the calculation:

$$48181 \cdot 0,0945 = 4550 \tag{9.2.4}$$

The total cost of energy will be of 4550 €.

United Kingdom and Falkland Islands As the other two ground stations are located under the administration of the United Kingdom, its costs will be used. In the UK the average cost per kWh is of 20 US cents, that are 0,189 €. Doing the calculation:

$$48181 \cdot 0,189 = 9100 \tag{9.2.5}$$

The total cost of energy will be of 9100€.

Total annual cost In the following table all the data that has been calculated is exposed in order to know the annual cost of the control centre (MCC) and the ground stations (GS).

Concept	MCC	GS Canada	GS Scotland	GS Malvinas
Energy	20540€	4550€	9100 €	9100 €
Maintenance	7500€	7500€	7500€	7500€
Internet	660€	660€	660€	660€
Total	28700€	12710€	17260€	17260€

Table 9.2.2: Annual costs

Total annual cost	**75930 €**

Table 9.2.3: Total annual cost of the ground segment consumption and maitenance

9.2.2 Salaries

In order to work properly, each ground station will require an electrical engineer, a computer technician, a manager and a secretary. Due to the nature of the constellation, the GS will need to be always functioning and, therefore, it can potentially fail at any moment. For this reason, the presence of an electrical engineer and a computer technician is required all the time. Four engineers and four computer technician will be hired so that for each job three of them will work all the day in 8 hours shifts while the other has the day off.

The salaries for each employee will be the average salary for each job in their respective countries. Those can be seen in Figure 9.2.4.

	Canada	United Kingdom	Argentina
Electrical engineer	47,700€	36,900€	12,300€
Computer technician	30,100€	21,800€	7,100€
Manager	34,500€	28,800€	14,100€
Secretary	28,000€	22,300€	9,500€

Table 9.2.4: Salaries for the different jobs according to the country.

Taking into account that each GS will have a manager, a secretary, four electrical engineers and four computer technicians, and that everyday will be an engineer and a technnician working during night, the total cost per ground station would be the following:

- Canada: 381,500€
- United Kingdom: 226,400€
- Argentina: 81,800€

The Mission Control Centre will consist of a building with a manager, a secretary and three aerospace engineers working. Because of the same reason as the ground station, it will be needed to hire twelve engineers, so as to have always three of them working all the time. Taking into account the average salary of each job in Spain, the cost of the salaries can be seen in Figure 9.2.5:

	Spain
Aerospace engineer	30,600€
Manager	30,500€
Secretary	23,000€

Table 9.2.5: Salaries for the different jobs in Spain.

The annual cost of all the salaries can be seen below:

- Annual cost of all Ground Stations salaries: 689,700€
- Annual cost of the Mission Control Centre salaries: 429,900€

9.3 Study of initial investment

The following items are needed:

- S-band system: 46,500€
- X-band system: 100,000€
- Computers and office material: 13,000€
- Building: 50,000€

Because of the time interval in which an antenna will be reorientating itself to point to the next satellite when the current satellite gets out of range, that antenna will not function until it finishes the reorientation. For this reason, two S-band and X-band systes are required for each ground station to be always operative. Therefore, each ground station needs two X-band systems, two S-band systems, computers and office material and a building.

The initial investment of one ground station will be 356,000€. The initial investment of the three ground stations will be 1,070,000€.

For the Mission Control Centre, the following costs are assumed:

- Computers and office material: 50,000€
- Building: 100,000€

The initial investment of the mission control centre will be 150,000€. The initial investment of all the ground segment will be 1,220,000€.

9.4 List of existing Ground Stations

9.4.1 ESA Ground Stations

- **Kiruna Station**
  - Coordinates: 67^o 51' 25.66" N, 20^o 57' 51.57" E.
  - Number of antennas: 2.
  - Size of the antennas: 15 meters. 13 meters.
  - Frequencies: S band transmission and S andX band reception. S band transmission and S andX band reception.
- **Kourou Station**
  - Coordinates: 5^o 15' 05.18" N, 52^o 48' 16.79" W.
  - Number of antennas: 1.
  - Size of the antennas: 15 meters.
  - Frequencies: S and X band transmission and reception.
- **Maspalomas Station**
  - Coordinates: 27^o 45' 46.40" N, 15^o 38' 01.68" W.
  - Number of antennas: 1.
  - Size of the antennas: 15 meters.
  - Frequencies: S band transmission and S and X band reception.
- **Redu Station**
  - Coordinates: 50^o 00' 01.64" N, 5^o 08' 43.24 E.
  - Number of antennas: 3.
  - Size of the antennas: 15 meters. 13.5 meters. 2.4 meters.
  - Frequencies: S band reception and transmission. Ka band reception and transmission. S band reception and transmission.
- **Santa Maria Station**
  - Coordinates: 36^o 59' 50.10" N, 25^o 08' 08.60" W.
  - Number of antennas: 1.
  - Size of the antennas: 5.5 meters.
  - Frequencies: S band reception.

- **Villafranca Station**
  - Coordinates: 40^o 26' 33.23" N, 03^o 57' 05.70" W.
  - Number of antennas: 2.
  - Size of the antennas: 15 meters. 15 meters.
  - Frequencies: S band transmission and reception. S band transmission and reception.

9.4.2 KSAT Ground Stations

- **Svabard Satellite Station**
  - Coordinates: 78^o N,15^o E.
  - Number of antennas: 31+
  - Size of the antennas: -
  - Frequencies: C, L, S, X and Ka band.
- **Tromsø Satellite Station**
  - Coordinates: 69^o N, 18^o E.
  - Number of antennas: 30+
  - Size of the antennas: -
  - Frequencies: L, S and X band.
- **Troll Satellite Station**
  - Coordinates: 72^o S, 2^o E.
  - Number of antennas: 3?
  - Size of the antennas: 7.3 meters.
  - Frequencies: S and X band.
- **Grimstad**
  - Coordinates: 58^o N, 8^o E.
  - Number of antennas: 1.
  - Size of the antennas: 3.2 meters.
  - Frequencies: X band.
- **Hartebeesthoek**
  - Coordinates: 25^o S, 27^o E.

- Number of antennas: 1?
- Size of the antennas: -
- Frequencies: S and X band.

- **Dubai**
  - Coordinates: 25^o N, 55^o E.
  - Number of antennas: 1?
  - Size of the antennas: -
  - Frequencies: S and X band.
- **Mauritius**
  - Coordinates: 20^o S, 57^o E
  - Number of antennas: 1?
  - Size of the antennas: -
  - Frequencies: S and X band.
- **Singapore**
  - Coordinates: 1^o N, 103^o E.
  - Number of antennas: 1?
  - Size of the antennas: -
  - Frequencies: S and X band.

9.4.3 NASA Ground Stations

- **Alaska Satellite Facility**
  - Coordinates: 64^o N, 147^o W.
  - Number of antennas: 3.
  - Size of the antennas: 11 meters. 11 meters. 10 meters.
  - Frequencies: S and X band. S and X band. S and X band.
- **McMurdo Ground Station**
  - Coordinates: 77^o 50' 20.87" S, 193^o 19' 58.50" W.
  - Number of antennas: 1.
  - Size of the antennas: 10 metes.
  - Frequencies: S band transmission and S and X band reception.

- **Wallops Ground Station**
  - Coordinates: 35^o N, 75^o W.
  - Number of antennas: 1.
  - Size of the antennas: 18.3 meters.
  - Frequencies: UHF.
- **White Sands Ground Station**
  - Coordinates: 33^o N, 107^o W.
  - Number of antennas: 2.
  - Size of the antennas: 18.3 meters. 18.3 meters.
  - Frequencies: VHF, S and Ka band. VHF, S and Ka band.

9.4.4 SSC Ground Stations

- **Clewiston Satellite Station**
  - Coordinates: 26.7^o N, 81.0^o W.
  - Number of antennas: 1.
  - Size of the antennas: -
  - Frequencies: S and X band.
- **Esrange Satellite Station**
  - Coordinates: 67^o 53" N, 21^o 04" E.
  - Number of antennas: 12.
  - Size of the antennas: -
  - Frequencies: 6x S band. 6x S and X band.
- **Inuvik Satellite Station**
  - Coordinates: 68^o 24" N, 133^o 30" W.
  - Number of antennas: 1.
  - Size of the antennas: 13 meters.
  - Frequencies: S and X band.
- **North Pole Satellite Station**
  - Coordinates: 64^o 48’ N, 147^o 30’ W.
  - Number of antennas: 2.

  - Size of the antennas: -
  - Frequencies: S band transmission and S and X band reception. S band transmission and S and X band reception.

- **Punta Arenas Satellite Station**
  - Coordinates: 53^o S, 71^o W.
  - Number of antennas: 1.
  - Size of the antennas: 7.3 meters.
  - Frequencies: S and X band.

- **Santiago Satellite Station**
  - Coordinates: 33^o 08" S, 70^o 40" W.
  - Number of antennas: 3.
  - Size of the antennas: -
  - Frequencies: S band transmitting and recepting.

- **South Point Satellite Station**
  - Coordinates: 19^o N, 156^o W.
  - Number of antennas: 2.
  - Size of the antennas: -
  - Frequencies: S, X and Ku band transmitting and receiving. S, X and Ku band transmitting and receiving.

- **Dongara Satellite Station**
  - Coordinates: 29^o 03' S, 115^o E.
  - Number of antennas: 3.
  - Size of the antennas: -
  - Frequencies: S, X, Ku and Ka band transmitting and receiving. S, X, Ku and Ka band transmitting and receiving. S, X, Ku and Ka band transmitting and receiving.

- **Yatharagga Satellite Station**
  - Coordinates: 29^o S, 115^o E.
  - Number of antennas: 1.
  - Size of the antennas: 13.56 meters
  - Frequencies: S band transmitting and S, X and Ka band reception.

9.4.5 Other Ground Stations

- **Goonhill Earth Station**
  - Coordinates: 50^o N, 5^o W.
  - Number of antennas: 28.
  - Size of the antennas: 3.7 meters   32 meters.
  - Frequencies: L, S, X, C, Ku and Ka band.

9.5 Decision taking

In the following lines the factors to take into account to decide the ground stations will be explained. After doing so, an OWA will be done if needed.

9.5.1 Availability

9.5.1.1 Building a ground station

If the decision to build a ground station is taken, it will be available as soon as it is constructed. The time taken to construct the ground stations depend on the efforts employed, but the three ground stations will be surely completed at the time the satellite network is completely deployed. From the moment the ground stations are built, they are totally available to accomplish the missions of Astrea constellation.

9.5.1.2 Renting a ground station

The sections regarding the renting of a ground station will be done considering LeafSpace (as it has been already said). LeafSpace is a company that does not work only with Astrea constellation, so total availability of the antenna's and its transmissions can not be assured. For this reason, is not possible to assure that the communication rate established in the project charter will be accomplished. Moreover, LeafSpace's Ground Stations are still non-existent, and they predict that the first ones will be available next year.

9.5.2 Cost

9.5.2.1 Building a ground station

The costs of building a ground station can be divided into an initial investment an a maintenance. The initial investment have been estimated in 190940 €and the maintenance in 30000€/year. The Net Present Cost (NPC) in 10 years will be calculated in order to compare this option with the option of renting a ground station. The discount rate used to do so will be 12%.

$$NPC = +I_o + \sum_{i=1}^{10} \frac{CF_i}{(1+r)^i} \tag{9.5.1}$$

$$NPC = 190940 + \sum_{i=1}^{10} \frac{30000}{(1+0.12)^i} = 360500 \tag{9.5.2}$$

9.5.2.2 Renting a ground station

In this case maintenance is not needed as it is carried out by the owners of the ground station. The cost, however, comes from the amount of data that is transfered from the satellites to the client. The estimation of the Mbyte transfered over a whole year is difficult to calculate. LeafSpace provides a minimum cost per month of 2400 €. This has been calculated for small communications with X-band. To calculate an approximation, this number will be increased a 40% because Astrea constellation will probably has quite higher transfer of data. The cost per year is, then 40320 €. The NPC will be calculated too:

$$NPC = \sum_{i=1}^{10} \frac{40320}{(1+0.12)^i} = 227820 \tag{9.5.3}$$

9.5.3 Position

9.5.3.1 Building a ground station

In the case the ground station is constructed an operated for the Astrea constellation, there is the possibility of build them in latitudes close to the ideal ones (from 45° to 70°), so more links will be available during more time. Moreover, there is also the possibility to build them in different longitudes (approximately with a difference of 120°).

9.5.3.2 Renting a ground station

In the case the ground station is rented, there is no possibility to choose the position of the ground station. In the case of LeafSpace, most of the ground stations that will be built in 2017 are located at 45° north. This can seem quite good from the point of view of visibility and links. However, all of them are more or less in the same longitude, so at the same time the links at the different ground stations are the same. With ground station at different longitudes, the performance of the constellation would be better than having them in the same longitude.

9.5.4 Ease to improve

9.5.4.1 Building a ground station

The fact of building a ground station implies that it can be improved and adapted to the constellation and the needs of the clients along the developement of the mission.

9.5.4.2 Renting a ground station

If the ground station is rented, it can not be improved according to the needs of the constellation, and maybe the constellation will have to be adapted to the ground station in order to accomplish the mission. The improvement in this case is, then, difficult and probably impossible.

9.5.5 Decision

The factors used to decide will be the ones presented previously. They will be rated from 1 to 2, being 2 the best option and 1 the worst option. As there are only two options, no linear interpolation is needed. Taking into account the requirements and needs of the project, the weights are the following ones:

- Availability: 6
- Cost: 9
- Position: 6
- Ease to improve: 5

The rating and the OWA of the decision between building a ground station or renting an existent one is:

	Availability	**Cost**	**Position**	**Ease to improve**	**OWA**
Build	2	1	2	2	0.83
Rent	1	2	1	1	0.67

Table 9.5.1: OWA of the GS

Part V

ANNEX V: 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 different subsystems as well as the role they will play during the lifetime of the mission. And since these systems will operate in space, they have to be prepared and certified to withstand extreme temperature and radiation conditions.

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

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

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

10.1.2 Thermal protection

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

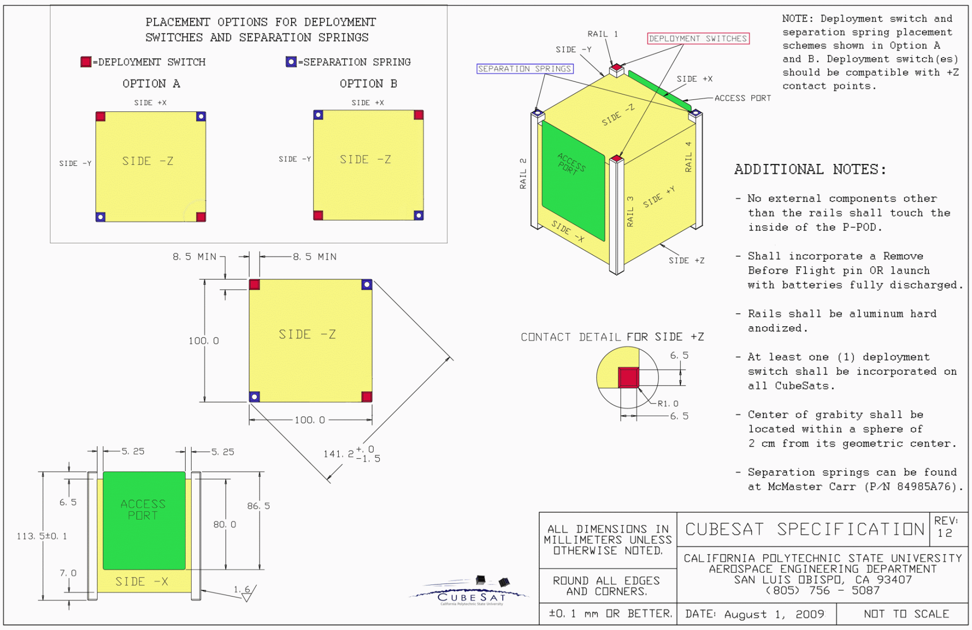


Figure 10.1.1: Dimensions of a 1U CubeSat
[34]

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

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

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

10.1.3 Study of the commercial available options and options chosen

A broad marked study is needed since all the options have to be considered. For this reason, and with the aim to show all the information and features of each system that has been considered in this section, the table 10.1.1 is presented below.

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

Table 10.1.1: Options studied for the structure and thermal protection

Finally, the options chosen are presented in the table 10.1.2.

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

Table 10.1.2: Options chosen for the structure and thermal protection

10.2 Electrical Power System

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

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

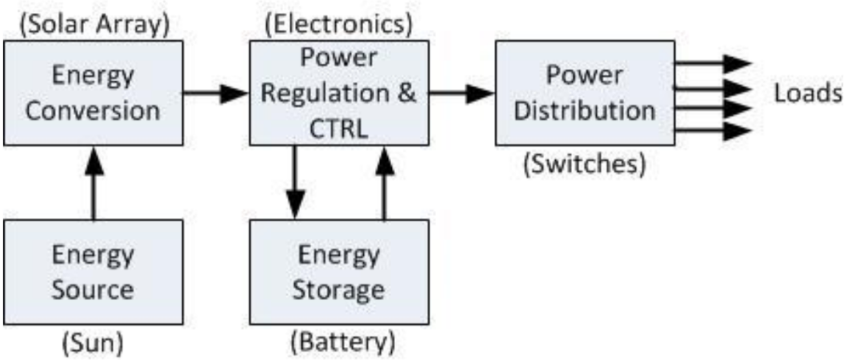


Figure 10.2.1: Basic schematics of the EPS
[35]

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 vast majority of the time the satellite will work under typical operation conditions. However, the estimation of the power consumption provided in the table 10.2.1 has been made for typical-high conditions in order to have a power margin and a more reliable estimation.

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

Table 10.2.1: Estimation of the power consumption under typical working conditions

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

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 correctly selected to prevent failure given their importance.

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

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

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

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

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

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

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

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

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

10.2.4 Batteries

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

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

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

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

10.2.5 Study of the commercial available options and options chosen

A broad marked study is needed since all the options have to be considered. For this reason, and with the aim to show all the information and features of each system that has been considered in this section, the table 10.2.2 is presented below.

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

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

Table 10.2.2: Options studied for the Electric Power System

Finally, the options chosen are presented in the table 10.2.3.

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

Table 10.2.3: Options studied for the Electric Power System

10.3 Propulsion Systems

10.3.1 Requirements

There is a big risk of a collision with space debris while a spacecraft is operating in Low Earth Orbits. The Inter-Agency Space Debris Coordination Committee recommended to the United Nations (section 5.3.2 'Objects Passing Through the LEO Region'): "Whenever possible space systems that are terminating their operational phases in orbits that pass through the LEO region, or have the potential to interfere with the LEO region, should be de-orbited (direct re-entry is preferred) or where appropriate manoeuvred into an orbit with a reduced lifetime. Retrieval is also a disposal option." and "A space system should be left in an orbit in which, using an accepted nominal projection for solar activity, atmospheric drag will limit the orbital lifetime after completion of operations. A study on the effect of post- mission orbital lifetime limitation on collision rate and debris population growth has been performed by the IADC. This IADC and some other studies and a number of existing national guidelines have found 25 years to be a reasonable and appropriate lifetime limit." [36]

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

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

10.3.2 Thrusters

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

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

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

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

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

The following table 10.3.1 shows the main parameters of this thruster.

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

Table 10.3.1: Main features of BGT-X5

10.3.3 Study of the commercial available options

A broad marked study is needed since all the options have to be considered. For this reason, and with the aim to show all the information and features of each system that has been considered in this section, the table 10.3.2 is presented below.

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

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

Table 10.3.2: Options studied for the propulsion system

Finally, the option chosen is presented in the table 10.3.3.

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

Table 10.3.3: Option chosen for the propulsion system

10.4 Attitude and Orbital Control Systems

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

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

Pointing in a specific direction is necessary for Earth observation, orbital maneuvers, maximizing solar power, and some scientific instruments. 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, this thrust will be given by the propulsion system and all the operation will be controlled on the On-Board Computer. Principally, the orbit control will be necessary to mitigate orbital debris effect on every satellite.

10.4.2 Study of the commercial available options

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

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

Table 10.4.1: Main ADACS features

Decision After the study of commercial options available, the previous two where the unique that fitted in AstreaSAT requirements, so a decision between these two must be done. Since all the features tabulated on 10.4.1 are critical, the same weights are given. Therefore, we will compare directly the two alternatives for choosing the best alternatives. Taking into account that we need: low power consumption, low, weight and size, high pointing accuracy and really versatile systems that can integrate multiple subsystems; **CUBE ADCS** is chosen. It has the lowest mass and power consumption, it also offers a higher attitude determination systems, redundancy is a key fact because we can not loose precision during the life time of each satellite. Finally, the fact that CUBE ADCS integrates also and On-Board Computer (OBC) is the turning point, because we have size and weight limitations, having and integrated, high performance OBC in this system will make able TT&C with the ground stations and the control af every system on board. http://www.cubespace.co.za/cubecomputer

10.5 Payload

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

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

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

The radio frequencies that we can use to establish the previous described links are regulated in [37] by frequency, bandwidth and type of communication . So, for the **Space to Ground link** we can use frequencies from **70MHz** to **240GHz**; for **Inter-satellite Space to Space link** plus data relay type of communication, frequencies are **2-2.4GHz**, **4-4,4GHz** and **22-240GHz**. Finally, **Client Space to Space link**, they exist to cases; on the one hand, the client points towards the Earth like a standard satellite, we capture its signal and make the data relay, since it is like a Space to Ground communication and also like a inter-satellite communication, we can combine the two previous restrictions. On the other hand, if the client satellite is below our constellation, we only had inter-satellite communication, therefore **Inter-satellite Space to Space link** rules are applied.

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

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.

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

10.5.1.0.1 Basic parameters

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

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

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

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

10.5.1.0.2 Patch antenna

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

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

Table 10.5.1: Main features of the patch antenna

10.5.1.0.3 Turnstile antenna

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

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

Table 10.5.2: Main features of the turnstile antenna

10.5.2 Antenna selection

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

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

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

10.5.3 Payload Data Handling Systems

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

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

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

10.5.3.0.1 Transceivers

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

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

Table 10.5.3: Main inter-satellite communication transceivers features

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

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

Table 10.5.4: Main space to ground communication transceivers features

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

10.5.3.0.2 PDHS computers

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

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

Table 10.5.5: Main PDHS computers features

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

10.5.4 Study of the commercial available options and options chosen

A broad marked study is needed since all the options have to be considered. For this reason, and with the aim to show all the information and features of each system that has been considered in this section, the table 10.5.6 is presented below.

Brand and model	Features	Total price (€)
Antennas		

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

Table 10.5.6: Options studied for the payload

Finally, with the aim to clarify all the information of this section, the chosen systems and components are presented in the table 10.5.7.

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

Table 10.5.7: 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 Astrea satellite (AstreaSAT) of the constellation, will need to report its operating status to the ground and receive commands from the ground. TT&C operations will usually be performed when the satellite flights over the coverage of the constellation ground station, but since the satellites are interconnected, there is the possibility to perform this operations via data relay links between satellites. As a collaboration with the communications department, S band frequency is chosen for TT&C operations, since there is no need for high data rates, the lower band will significantly reduce the power consumption.

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

10.7 Link Budget

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

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

10.7.1 Communications Basics

When evaluating a wireless link, the three most important questions to be answered are: [6]

1. <u>How much radio frequency (RF) power is available?</u> Up to 2W for S band or up to 12W for Xband.
2. <u>How much bandwidth is available?</u>
   Available 400MHz with 28 channels of 14MHz or 228 channels of 1.75MHz for inter-satellite communication at S band. For X band, there's more than 4GHz available [37]. In fact is limited by the TR-600 transceiver at 56MHz for S band and to 100MHz by SWIFT - XTS at X band.
3. <u>What is the required reliability (as defined by Bit Error Rate, or BER)?</u>
   Required reliability for space systems$E_b/N_o \geq 10$, so $BER = 5.5x10^{-6}$ for a MSK, PSK (worst case) modulation as shown in Fig.10.7.5.

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

$$C = Blog_2(1 + S/N) \tag{10.7.1}$$

where:

$$C = \text{channel capacity (bits/s)}$$
$$B = \text{channel bandwidth (Hz)}$$
$$S = \text{signal strength (watts)}$$
$$N = \text{noise power (watts)}$$

With all data known, the minimum required sensitivity of a receiver using the Eq. 10.7.1 will be stated in the Link Budget calculation.

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

TRANSMISSION LOSSES	PROPAGATION LOSSES	FREE SPACE LOSSES		
		ATMOSPHERIC LOSSES	Ionospheric effects	Faraday rotation
				Scintillation effects
			Tropospheric effects	Attenuation
				Rain attenuation
				Gas absorption
				Depolarization
				Sky noise
			Local effects	
		POINTING LOSSES		
	LOCAL LOSSES	EQUIPMENT LOSSES	Feeder losses	
			?????	
		ENVIRONMENT LOSSES		

Figure 10.7.1: Principal losses in the received signal [5]

10.7.2 Propagation losses

10.7.2.0.1 Free Space Losses

Range and Path Loss Another key consideration is the issue of range. As radio waves propagate in free space, power falls off as the square of range. For a doubling of range, power reaching a receiver antenna is reduced by a factor of four. This effect is due to the spreading of the radio waves as they propagate, and can be calculated by [6]:

$$L = 20log_{10}(4\pi D/\lambda) \tag{10.7.2}$$

where:

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

10.7.2.0.2 Atmospheric Losses

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

- Atmospheric attenuation.
- Atmospheric absorption.

The major distinguishing factor between them is their origin. Attenuation is weatherrelated, while absorption comes in clear-sky conditions. Likewise, these losses can be due to ionospheric, tropospheric and other local effects. [5]

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

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

These effects decrease usually with the increase of the square of the frequency and most serious ones in satellite communications are the polarization rotation and the scintillation effects, and those are the ones that will be treated in this dissertation. [5]

Tropospheric Effects [5] Troposphere is composed by a miscellany of molecules of different compounds, such as hail, raindrops or other atmospheric gases. Radio waves that pass by troposphere will suffer their effects and will be scattered, depolarized, absorbed and therefore attenuated.
Attenuation: As radio waves cross troposphere, radio frequency energy will be converted into thermal energy and that attenuates signal.
Rain attenuation: Ground stations had been chosen in order that the attenuation caused by rainfall will be very punctual. Also, the fact that there are three ground stations makes really difficult that a satellite can not communicate to the ground in all the orbit period.
Gas absorption: Under normal conditions, only oxygen and water vapour have a significant contribution in absorption. Other atmospheric gases only become a problem in very dry air conditions above 70 GHz. Thereby, losses caused by atmospheric absorption vary with frequency and the collection of data already received allows the elaboration of the graphic that follows:

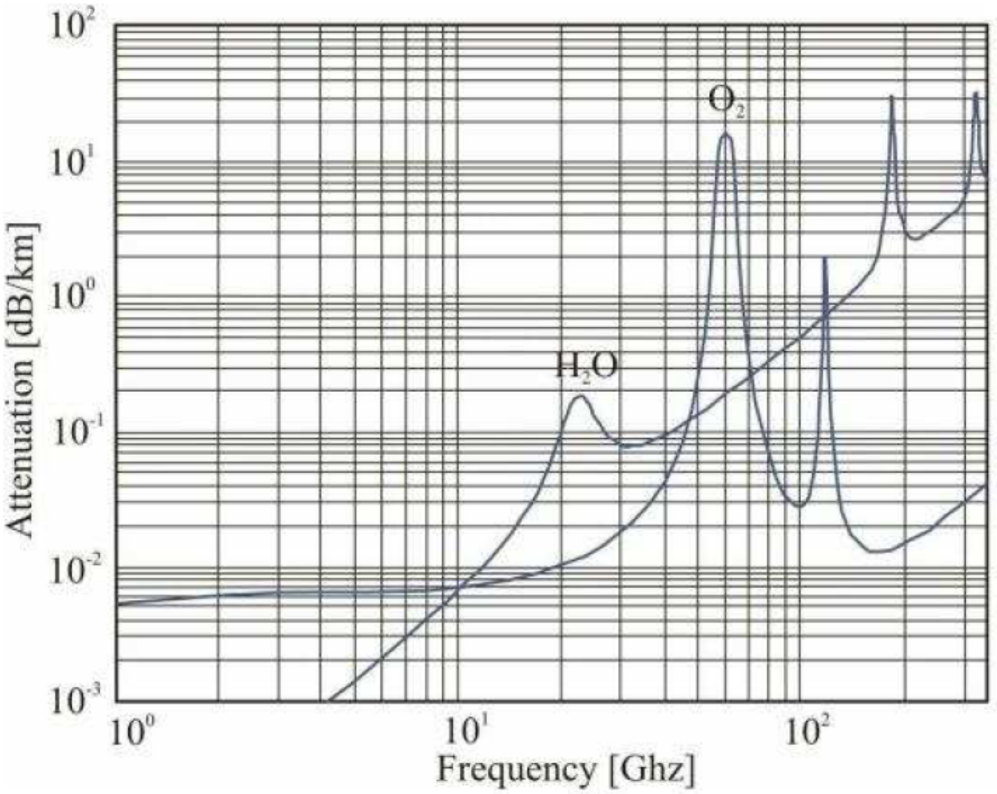


Figure 10.7.2: Specific attenuation for different frequencies [5]

Once these values depend on atmosphere thickness, it becomes necessary to perform all calculations taking into account troposphere's thickest layer (T_{trop}), which has 20 km.
It is also mandatory to refer that this graph represents the absorption for a satellite in the zenith, in other words, for an elevation angle of 90° ($\theta = 90^o$). For lower angles, the atmospheric absorption (L_{abs}) is given by [5]:

$$L_{abs}(dB) = L_{abs|90^o}(dB/km)\ cosec(\theta)\ T_{trop}(km) \tag{10.7.3}$$

For AstreaSAT, 5x$10^{-3}dB/km$ attenuation factor is considered for S band due to the O_2 specific attenuation. On the other hand, 4x$10^{-3}dB/km$ attenuation factor is considered for X band due to the H_2O and to the O_2 specific attenuations. An study of the critical elevation angle will lately be performed.

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

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

Polarization: Satellite communications use linear and circular polarization, but undesirable effects may transform it into an elliptical polarization. Depolarization may occur when an orthogonal component is created due to the passing of the signal through the ionosphere. There are two ways to measure its effect, cross polarization discrimination (XPD) and polarization isolation (I) [5]. To overcome this attenuation problems a circular polarization is the best option. AstreaSAT patch antennas will mitigate this problem, therefore this loses are considered negligible.

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

AstreaSAT noise temperature A good approximation based on Fig.10.7.3 is that Galaxy noise is 3K for S band and almost 1K for X band. Furthermore, for the previous

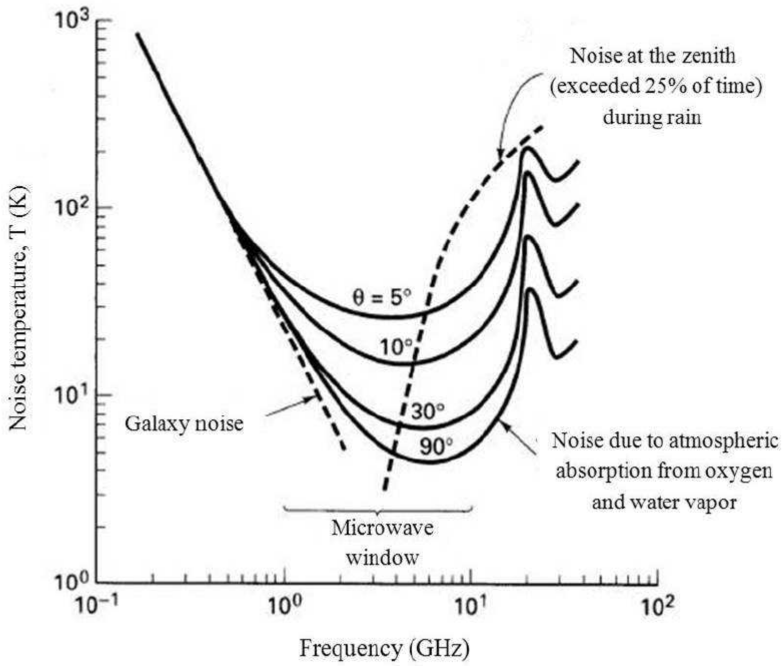


Figure 10.7.3: Galaxy noise influence in noise temperature [5]

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

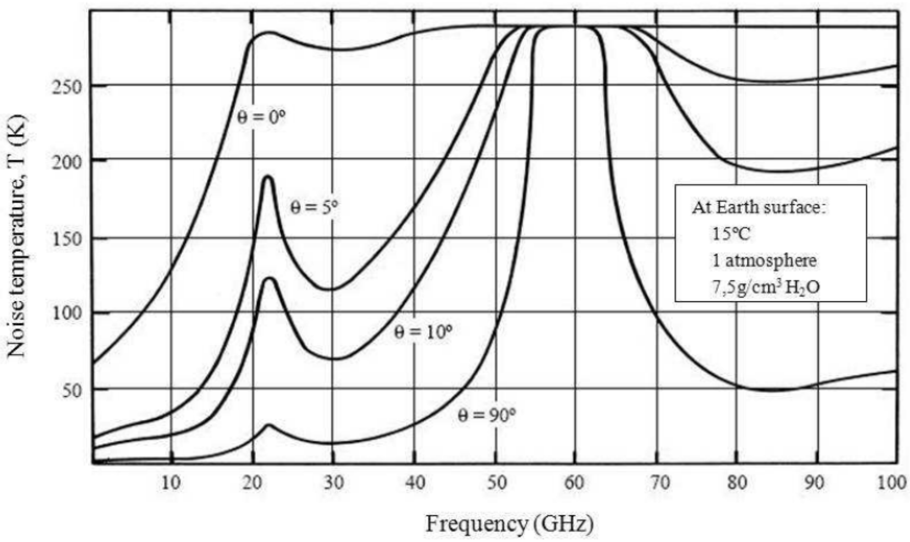


Figure 10.7.4: Noise temperature variation with frequency [5]

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

10.7.2.0.3 Pointing Losses

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

Antenna misalignment losses (L_{aml}) are calculated using statistical data, so these values are an approximation based on real data observed in several GS. Ergo, these values are not calculated, but estimated. [5]

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

10.7.2.0.4 Multipath and Fade Margin

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

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

10.7.3 Local Losses

10.7.3.0.1 Equipament Losses

The receiving and emitting equipments also introduces some losses to the signal.
Feeder Losses: Feeder losses occur in the several components between the receiving antenna and the receiver device, such as filters, couplers and waveguides. These losses are similar to the ones which occur also in the emission, between the emitting antenna and the output of the high power amplifier (HPA). [5]

10.7.3.0.2 Environment Losses

This item is related to the specific region of the globe where the ground station is placed (equatorial, tropical, polar...). Depending on its latitude, each region has its own characteristics (e.g. temperature, moisture, thickness of atmospheric ice layer...), which may provoke variation in signal reception. [5]

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

10.7.4 Modulation Technique

Modulation technique is a key consideration. This is the method by which the analogue or digital information is converted to signals at RF frequencies suitable for transmission. Selection of modulation method determines system bandwidth, power efficiency, sensitivity, and complexity. Most of us are familiar with Amplitude Modulation (AM) and Frequency Modulation (FM) because of their widespread use in commercial radio. Phase Modulation is another important technique. It is used in applications such as Global Position System (GPS) receivers and some cellular telephone networks. [6]

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

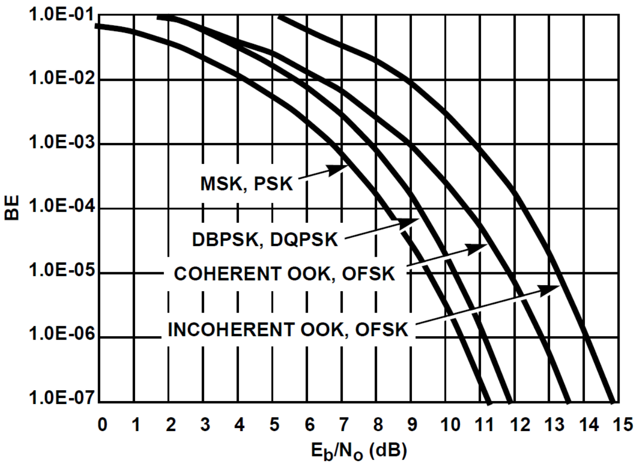


Figure 10.7.5: Probability of bit error for common modulation methods [6]

A graph of E_b/N_o vs BER is shown in Figure 10.7.5. E_b/N_o is a measure of the required energy per bit relative to the noise power. Note that E_b/N_o is independent of the system data rate. In order to convert from E_b/N_o to SNR, the data rate and system bandwidth must be taken into account as shown below:

$$SNR = (E_b/N_o)(R/B_T) \tag{10.7.4}$$

where:

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

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

10.7.5 System Noise

The system noise temperature (T_S) is the sum of the antenna noise temperature (T_A) and the composite temperature of other components (T_{comp}), according to: [5]

$$T_S = T_A + T_{comp} \tag{10.7.5}$$

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

$$T_A = T_m\ (1 - 10^{-A/10}) + T_C\ 10^{-A/10} \tag{10.7.6}$$

Usually, for clouds it is considered $T_m = 280K$ and for the rain $T_m = 260K$. The sky noise tends to be $T_C = 10K$. Taking into account the values from Fig.10.7.3 and Fig.10.7.2 the following estimation can be made:

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

According to [5] a good components temperature approximation for a typical ground station is $T_{comp} = 65.5K$.

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

Channel Noise All objects which have heat emit RF energy in the form of random (Gaussian) noise. The amount of radiation emitted can be calculated by [6]:

$$N = kTB \tag{10.7.7}$$

where:

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

This is the lowest possible noise level for a system with a given physical temperature. For most applications, temperature is typically assumed to be room temperature (290K). Equations 10.7.1 and 10.7.7 demonstrate that RF power and bandwidth can be traded off to achieve a given performance level (as defined by BER). [6]

10.7.6 Link Budget Calculation

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

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

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

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

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

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

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

10.9 Astrea satellite Final Configuration

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

Part VI

ANNEX VI: Financial and Other Considerations

> *"The first rule is to never lose money. The second rule is to never forget the first one."*
>
> Warren Buffett

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

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

Chapter 11

Financial Study

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

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

11.1 Selling the product

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

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

$$365 \cdot 24 \cdot 60 \cdot 60 = 31536000s \tag{11.1.1}$$

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

$$31536000 \cdot 75 = 2365200000Mbits \tag{11.1.2}$$

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

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

11.1.1 Estimation of demand

11.1.1.1 Universities

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

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

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

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

Table 11.1.1: Table. List of Universities with Aerospace Degrees

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

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

11.1.1.2 Particular customers

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

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

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

11.1.1.3 Demand

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

11.1.2 Pricing the service

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

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

11.2 Economic Feasibility Report

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

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

11.2.1 Previous costs

11.2.1.1 Engineering hours

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

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

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

11.2.1.2 Administrarion costs

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

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

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

This result in 259.000€/year

11.2.1.3 Taxes

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

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

11.2.1.4 Insurance

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

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

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

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

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

This insurer provide a great coverage, in which highlights:

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

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

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

11.2.2 Economic feasibility study

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

TIME	Year 0	Year 1	Year 2	Year 3	Year 4	Year 5	Year 6	Year 7	Year 8	Year 9	Year 10
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
Number of Mbits hired		354.780.000,00	378.432.000,00	402.084.000,00	425.736.000,00	449.388.000,00	473.040.000,00	473.040.000,00	473.040.000,00	473.040.000,00	473.040.000,00
Gain (M euros)		35,48	37,84	40,21	42,57	44,94	47,30	47,30	47,30	47,30	47,30
Total	**0,00**	**35,48**	**37,84**	**40,21**	**42,57**	**44,94**	**47,30**	**47,30**	**47,30**	**47,30**	**47,30**
COSTS											
n planes/year	9					9					9
Satellites/year	189	0	0	0	0	189	0	0	0	0	189
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
Insurance		-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532	-2,24532
Taxes											
Percentage of profit					0,15	0,15	0,25	0,25	0,25	0,25	0,25
Cost of taxes		0	0	0	-6,38604	-6,74082	-11,826	-11,826	-11,826	-11,826	-11,826
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
Launching											
Planes	-48,256	0,000	0,000	0,000	0,000	-48,256	0,000	0,000	0,000	0,000	-48,256
Satellites	-3,024	0,000	0,000	0,000	0,000	-3,024	0,000	0,000	0,000	0,000	-3,024
System											
Structure and mechanics											
Assembly (individual)	-3,78	0,00	0,00	0,00	0,00	-3,78	0,00	0,00	0,00	0,00	-3,78
Assembly (constellation)	-0,15	0,00	0,00	0,00	0,00	-0,15	0,00	0,00	0,00	0,00	-0,15
Structure	-0,737	0,000	0,000	0,000	0,000	-0,737	0,000	0,000	0,000	0,000	-0,737
Thermal protection	-0,189	0,000	0,000	0,000	0,000	-0,189	0,000	0,000	0,000	0,000	-0,189
Electric power system											
Solar arrays	-12,852	0,000	0,000	0,000	0,000	-12,852	0,000	0,000	0,000	0,000	-12,852
Batteries	-2,381	0,000	0,000	0,000	0,000	-2,381	0,000	0,000	0,000	0,000	-2,381
Power management	-3,024	0,000	0,000	0,000	0,000	-3,024	0,000	0,000	0,000	0,000	-3,024
Payload											
Patch antenna	-10,595	-0,011	-0,011	-0,011	-0,011	-10,595	-0,011	-0,011	-0,011	-0,011	-10,595
Antenna deployment	-0,567	0,000	0,000	0,000	0,000	-0,567	0,000	0,000	0,000	0,000	-0,567
Transciever inter-satellite	-4,845	0,000	0,000	0,000	0,000	-4,845	0,000	0,000	0,000	0,000	-4,845
Transciever space to ground	-1,040	0,000	0,000	0,000	0,000	-1,040	0,000	0,000	0,000	0,000	-1,040
Data handling system	-0,945	0,000	0,000	0,000	0,000	-0,945	0,000	0,000	0,000	0,000	-0,945
Variable expenses	-0,756	0,000	0,000	0,000	0,000	-0,756	0,000	0,000	0,000	0,000	-0,756
AOCDS											
Thruster	-9,450	0,000	0,000	0,000	0,000	-9,450	0,000	0,000	0,000	0,000	-9,450
CubeSpace ACDS	-2,835	0,000	0,000	0,000	0,000	-2,835	0,000	0,000	0,000	0,000	-2,835
Communications											
Maintenance GS Canada		-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
Maintenance GS Malvines		-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
Salaries GS Scotland (UK)		-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
Salaries MCC		-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
Total	**-105,47**	**-3,69**	**-3,69**	**-3,69**	**-10,08**	**-115,85**	**-15,52**	**-15,52**	**-15,52**	**-15,52**	-120,93
CASH FLOW	**-109,54**	**31,79**	**34,15**	**36,52**	**32,50**	**-70,91**	**31,79**	**31,79**	**31,79**	**31,79**	**-73,63**
DISC CF	**-109,54**	**29,99**	**30,40**	**30,66**	**25,74**	**-52,99**	**22,41**	**21,14**	**19,94**	**18,81**	**-41,11**
CUM CF	**-109,54**	**-77,75**	**-43,60**	**-7,08**	**25,42**	**-45,49**	**-13,70**	**18,08**	**49,87**	**81,66**	**8,03**
DIS CUM CF	**-109,54**	**-79,55**	**-49,15**	**-18,49**	**7,25**	**-45,74**	**-23,33**	**-2,19**	**17,75**	**36,57**	**-4,55**

Table 11.2.3: Feasibility Study

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

11.3 Conclusions of the financial study

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

11.3.1 Pay Back Time (PBT)

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

$$\frac{25.42 - (-7.08)}{4 - 3} = \frac{25.42 - 0}{4 - x} \tag{11.3.1}$$

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

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

$$\frac{18.08 - (-13.70)}{7 - 6} = \frac{18.08 - 0}{7 - x} \tag{11.3.2}$$

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

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

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

11.3.2 Updated Pay Back Time (UPBT)

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

$$\frac{7.25-(-18.49)}{4-3}=\frac{7.25-0}{4-x} \tag{11.3.3}$$

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

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

$$\frac{17.75-(-2.19)}{8-7}=\frac{17.75-0}{8-x} \tag{11.3.4}$$

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

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

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

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

11.3.3 Break Even Point (BEP)

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

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

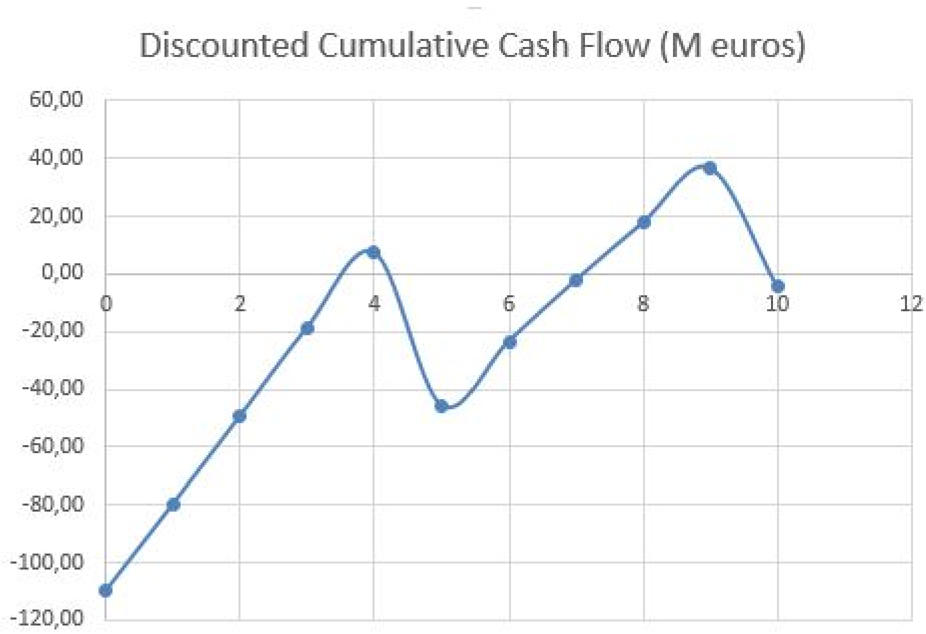
By changing manually the parameter "Number of Mbits hired" of first year, it is found that the Break Even Point is of 36907600 Mbits (with this value, the Cash Flow is approximately 0). This means that under no account there can be less Mbits hired, otherwise, the Cash Flow would be negative and the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

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

11.3.4 Net Present Value (NPV)

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

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



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

11.3.5 Internal Rate of Return (IRR)

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

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

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

Chapter 12

Marketing Plan

12.1 Executive Summary

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

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

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

12.2 Target Customers

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

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

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

12.3 Unique Selling Proposition

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

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

12.4 Pricing & Positioning Strategy

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

12.5 Distribution Plan

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

12.6 Marketing Materials

The marketing materials we count on are:

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

12.7 Online Marketing Strategy

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

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

12.8 Conversion Strategy

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

12.9 Joint Ventures & Partnerships

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

Chapter 13

Environmental Impact Study

13.1 Introduction

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

13.2 Ground Stations

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

13.3 Satellites

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

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

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

13.4 Launch system

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

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

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

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

The document also evaluates the following possible risks:

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

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

- **Environmental effect of the displacement of fishing activities.** For the demersal fish and mobile invertebrate community, marine mammals and seabirds, the effects of fishing displacement would be low because these populations could also be impacted in the areas to which fishing is displaced. In the eastern jettison zone there is less fishing activity so the consequences of fishing displacement on the seabed community, demersal fish and mobile invertebrates, marine mammals and seabirds are negligible, reaching minor impacts after 1000 or more launches.

- **Effect of the provision of hard substrates.** Another potential positive outcome for seafloor biota requiring hard substrates is that the jettisoned materials would provide further attachment sites. However, even after 10,000 launches this would provide only about 50 ha of additional attachment surface, leading to a moderate benefit at most.

- **Disturbance to marine fauna.** Noise and disturbance to marine fauna above and below water is a potential consequence of the jettisoned materials falling into the jettison zone. The chance of repeated disturbance to the same individuals or groups of marine mammals or seabirds increases with the number of launches. This was assessed as a low risk for up to 100 launches over two years, a moderate risk for up to 1000 launches over almost 20 years, and a high risk for up to 10,000 launches over almost 200 years.

- **Risk of direct strikes causing mortality to components of the ecosystem.** Direct strikes causing mortality are a low risk for all components of the ecosystem up to 1000 launches over an almost 20 year period. Direct strikes reach moderate levels of risk for the benthic invertebrate community, sensitive benthic environments, and a rare threatened species, the magenta petrel, after 10,000 launches over a period of almost 200 years.

- **Risk of smothering of sea floor organisms.** Smothering the feeding or respiratory structures of sea floor organisms by jettisoned materials was assessed as a low risk for all levels of launches up to 1000 launches and a moderate risk by 10,000 launches. This is likely to be a factor principally in areas of hard substrate where the jettisoned materials are unlikely to become buried in sediment so will be important principally on the Bounty Platform.

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

Chapter 14

Legislation

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

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

Another requirement of the UNOOSA is that space objects have to be registered, so all the member states have to establish their own national registries and provide this information to the United Nations Register. UNOOSA updates all the registrations through its website and through the United Nations Official Document System so that they are publicly available to anyone. For example, in the United States of America a license is required to operate a space system. This license is given by the National Oceanic and Atmospheric Administration (NOAA). Its register is the U.S. Registry of Objects Launched into Outer Space. It also has an agency that identifies and tracks satellites, the Joint Space Operations Center, which provides some recommendations on how to develop, launch and operate a satellite. These are the regulatory requirements that the CubeSat developers recommend to fulfill because they are based in California, United States.

However, in the case of the Astrea constellation, since the company is based in Spain (a party of the Space Law), the current legislation is the *Real Decreto 278/1995* of 24 February 1995. According to this Royal Decree [**?**], the objects launched from

Spain or whose launch has been promoted by Spain, should be registered in the *Registro Español de Objetos Espaciales Lanzados al Espacio Ultraterrestre* (Spanish Registry of Objects Launched into Outer Space). The necessary data to register the satellite must be provided to the *Dirección General de Tecnología Industrial del Ministerio de Industria y Energía* (Department of Industrial Technology of the Ministry of Industry and Energy). This department will notificate the registry to the Secretary-General of the United Nations.

The registration has to contain the following data:

a) Name of launching State or States;

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

c) Date and territory or location of launch;

d) Basic orbital parameters, including:

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

e) General function of the space object.

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

a) Name of launching State or States: Spain

b) Designator of the space object or its registration number: AstreaSAT 1

c) Date and territory or location of launch: 22 February 2018, New Zealand

d) Basic orbital parameters: Low Earth Orbit

   I) Nodal period: 95,4815 minutes
   II) Inclination: 72 degrees
   III) Apogee: 6.913,0 km
   IV) Perigee: 6.913,0 km

e) General function of the space object: CubeSat 3U, part of the communications constellation Astrea

Usually the launching state and the state of the launch are different. In this case the states have to determine which of them will register the launching object. It is common that the object is registered in the register of the launching state, the state that promotes the launch.

Chapter 15

Bibliography

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

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

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

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

[5] Carlos Jorge and Rodrigues Capela. Protocol of Communications for VORSAT Satellite - Link Budget. (April), 2012.

[6] Application Note. Tutorial on Basic Link Budget Analysis. *Intersil*, (June 1998):1–8, 1998.

[7] Behrouz a. Forouzan. *Data Communications and Networking - Global Edition.* 2012.

[8] CCSDS. TC Space Data Link Protocol. (September), 2010.

[9] CCSDS. TM Synchronization and Channel Coding—Summary of Concept and Rationale. *CCSDS Green Book*, (November 2012), 2012.

[10] CCSDS. *Recommendation for Space Data System Standards - Space Packet Protocol.* Number CCSDS 133.0-B-1. 2003.

[11] Information Sciences Institute University of Southern California 4676 Admiralty Way and California 90291 Marina del Rey. *Internet Protocol Specification.* 1981.

[12] S Deering and R Hinden. *Internet Protocol, Version 6 (IPv6) Specification.* 1998.

[13] CCSDS. *Recommendation for Space Data System Standards - Encapsulation Service.* Number 133.1-B-2. 2009.

[14] Space Assigned Number Authority (SANA) Registry: Packet Version Number. http://sanaregistry.org/r/packet_version_number/packet_version_number.html.

[15] Space Assigned Number Authority (SANA) Registry: Application Identifier. http://sanaregistry.org/r/space_packet_protocol_application_process_id/space_packet_protocol_a

[16] Space Assigned Number Authority (SANA) Registry: Protocol Identifier. http://sanaregistry.org/r/protocol_id/protocol_id.html.

[17] CCSDS. *Recommendation for Space Data System Standards - IP over CCSDS Space Links.* Number CCSDS 702.1-B-1. 2012.

[18] Space Assigned Number Authority (SANA) Registry: IP Extension header. http://sanaregistry.org/r/ipe_header/ipe_header.html.

[19] J Postel. Internet Control Message Protocol. pages 1–21, 1981.

[20] S Kent and K Seo. Security Architecture for the Internet Protocol. pages 1–101, 2005.

[21] B Fenner, M Handley, H Holbrook, I Kouvelas, R Parekh, Z Zhang, and L Zheng. *Protocol Independent Multicast - Sparse Mode (PIM-SM): Protocol Specification (Revised).* 2016.

[22] A Adams, J Nicholas, and W Siadak. *Protocol Independent Multicast - Dense Mode (PIM-DM): Protocol Specification (Revised).* 2005.

[23] D Savage, J Ng, S Moore, D Slice, P Paluch, and R White. *Cisco's Enhanced Interior Gateway Routing Protocol (EIGRP).* 2016.

[24] J Moy. *OSPF Version 2 Status.* 1998.

[25] R Coltun, D Ferguson, J Moy, and A Lindem. *OSPF for IPv6.* 2008.

[26] G Malkin. RIP Version 2. pages 1–39, 1998.

[27] G Malkin and R Minnear. RIPng for IPv6. pages 1–19, 1997.

[28] Internet Assigned Number Authority (IANA) Registry: Protocol Numbers. http://www.iana.org/assignments/protocol-numbers/protocol-numbers.xhtml.

[29] Energy Star. US Energy Use Intensity by Property Type, 2016.

[30] Endesa. Precios de Tarifas Reguladas Luz y Gas.

[31] Precios de contratos de mantenimiento en Madrid | Fojenet.

[32] LimpiezasSIL. Como Calcular un Presupuesto de Limpieza.

[33] OVO Energy. Average electricity prices around the world.

[34] CalPoly. Cubesat design specification (CDS). page 42, 2014.

[35] Robert Burt. Distributed Electrical Power System in Cubesat Applications. pages 2–3, 2011.

[36] IADC Space Debris Mitigation Guidelines. 2007.

[37] 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.

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