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Study of the ground - to -Very Low Earth Orbit (VLEO) satellite communication link

BACHELOR'S DEGREE THESIS

Bachelor's degree in Aerospace Vehicles Engineering

DOCUMENT 1. REPORT

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Equipped with his five senses, man explores the universe
around him and calls the adventure Science.

Edwin Hubble, The Nature of Science, 1954

Acknowledgements

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On the other hand, I would like to express my infinite gratitude to my parents, Anna and Paco, and my brother Biel. They have been supporting me as far back as I can remember and they have not let me down this time neither. I would like to thank Gurinder, too. His patience and immense support helps me every day to look forward.

I would not like to finish without thanking my friends. Shearing this experience with them has done it much bearable.

Thank you all very much for being with me through this research and let me accomplish my dreams.

Summary

This study is divided in three documents and one DVD which are described below.

Document 1. Report
The report corresponds to this document and it presents the work that has been done during the development of the study. It is divided into the following parts:

- **Chapter 1**. **Introduction**: A summary of the aim, scope and objectives is presented beside the justification and the planning of the study.
- **Chapter 2**. **State of the art**: An explanation of the satellites and types of existing orbits is presented.
- **Chapter 3**. **Communication link geometry**: The geometry problem between a satellite and a ground station is exposed, together with the equation which calculates the visibility windows time. Also, the Matlab code implemented is explained.
- **Chapter 4**. **Communications architecture**: A brief introduction of the process the information collected by the satellites has to follow in order to safely be transmitted to the ground station is presented.
- **Chapter 5**. **Results**: The different results obtained with the code are exposed.
- **Chapter 6**. **Conclusions**: With all the information of the thesis, some conclusions about the visibility windows and the implemented code are detailed.
- **Future work**: The possible improvements and the future continuity of the thesis are presented.
- **Environmental and security impact**: An insight of the environmental impact and the security measures of this thesis are commented.
- **Bibliography**: All the references that have been used while developing this work are attached.

Document 2. Annexes

It contains all the implemented Matlab code.

Document 3. Budget

In this document, the budget of the implementation of the thesis in a real business is attached.

DVD

All the files of the study have been burned into a DVD. It includes the electronic documents in PDF format and the images that have been used. Also the Matlab code file has been included.

Contents

List of Figures

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List of abbreviations

BER	Bit error rate
BPSK	Binary phase shift keying
C	Centre of the Earth
ECF or ECEF	Earth-centred fixed reference frame
ECI	Earth-centred inertial reference frame
EO	Earth Observation
ESA	European Space Agency
FSK	Frequency shift keying
GEO	Geosynchronous Orbit
GMST	Greenwich mean sideral time
GS	Ground station
HEO	High Earth Orbit
HP	Horizontal plane
IRP	International reference pole
IRM	International reference meridian
ISS	International Space Station
ITU	International Telecommunications Union
JD	Julian Date
LEO	Low Earth Orbit
MEO	Medium Earth Orbit
RJD	Reduced Julian Date
RTS	Rise time satellite
SAT	Satellite
SSP	Subsatellite point
STS	Set time satellite
TLE	Two line elements
TT&C	Tracking Telemetry & Command
UT0	Universal time
UT1	Standard universal time
UTC	Coordinated Universal time
VLEO	Very Low Earth Orbit
VWT	Visibility windows time
WARC	World Administrative Radio Conference

List of symbols

$\mathbf{a}$	Semi-major axis
$\mathbf{Az}$	Azimuth angle
$\mathbf{d}$	Slant range
E_b	Energy received per bit
$\mathbf{e}$	Eccentricity
$\mathbf{H}$	Satellite altitude
$\mathbf{i}$	Inclination angle
N_o	Noise spectral density
$\mathbf{r}$	Radius of satellite'sorbit
$\mathbf{r_E}$	Radius of the Earth
$\mathbf{v}$	True anomaly
t_a	Satellite's rise time
t_b	Satellite's set time
t_m	Time when elevation angle is maximum
x_m	Satellite's position when elevation angle is maximum
γ	Central angle
γ_c	Central angle for minimum elevation angle
θ	Elevation angle
θ_c	Elevation angle for minimum angle of elevation
$\lambda_\mathbf{0}$	Initial position's longitude
$\lambda_\mathbf{s}$	Latitude of subsatellite point
$\lambda_\mathbf{gs}$	longitude of ground station
τ	Visibility window duration
$\phi_\mathbf{0}$	Initial position's latitude
$\phi_\mathbf{gs}$	Latitude of ground station
$\phi_\mathbf{s}$	Latitude of subsatellite point
Ω	Right ascension (or longitude) of the ascending node
ω	Argument of periapsis
$\omega_\mathbf{E}$	Angular velocity of Earth's rotation ECF frame
$\omega_\mathbf{F}$	Angular velocity of satellite in ECF frame
$\omega_\mathbf{I}$	Angular velocity of satellite in ECI frame

Chapter 1

Introduction

1.1 Aim

The aim of this study is to analyse the problem of the reduced windows communication between Very Low Earth Orbit (VLEO) and Low Earth Orbit (LEO) satellites and ground stations. The objective is to parametrize all elements involved in the data transmission and try to compare the results with an existing program.

1.2 Scope

The scope for this study contains the following points:

- Study the geometrical problem of the reduced visibility window for VLEO and LEO satellites.
- Parametrize what elements affect the communication time and how these elements influence to it.
- Write a Matlab code which predict the visibility windows duration for satellites orbiting at VLEO and LEO.
- With the previously mentioned code, also the data rate of design should be obtained.
- Understand how the data is transmitted form the satellite to the ground station.

Furthermore, in order to successfully complete all the study, some specific deliverables must be done:

- Report: document that would include all the relevant information about the study development, emphasizing the conclusions and the applicability of the study itself.

- Annexes: The code implemented has to be completely written in document which is not the report. In the present study, it is presented in the Annexes.

- Budget: it is going to be budgeted also the resources used to develop the study, in terms of hours of dedication, and in consequence in terms of human costs.

1.3 Requirements

Analysing the scope of this project, it can be extracted some common requirements exposed next:

- The state of the art has to provide a general introduction of what satellites are and their uses. Specifically, CubeSats are needed to be explained in detail. What are they, and their advantages and disadvantages must be exposed.

- For the analytical part of the study, all kind of satellites can be studied in order to understand how windows communications work.

- The CubeSats orbit will be a VLEO: orbiting from 160 km to 450 km above the Earth's surface. [1]

- Only the communication between the satellite and the ground station will be studied. The communication between satellites is out of the scope.

- Earth is considered a perfect sphere.

- The orbits studied have to be circular.

1.4 Justification

1.4.1 Identification of the need

Satellite technology is developing fast, and the applications for it are increasing all the time. Not only can satellites be used for radio communications, but they are also used for astronomy, weather forecasting, broadcasting, mapping and many more applications.

Nowadays the satellites located near the Earth’s surface are gaining importance. Because they are closer to the Earth, they can reduce their dimensions and take information with the same quality than satellites orbiting much further. Therefore, their construction and launch are much cheaper. With this smaller satellites, new technology can be tested in space more quickly. But not all are benefits. Being so close to the Earth, makes their visibility windows shorter. This is an important handicap because the information collected can not be sent to the ground station, and therefore can not be analysed and processed to extract conclusions of it.

1.4.2 Usefulness of the study

Studying the visibility windows is an essential part of every space mission. At any given location on Earth, a LEO satellite is visible only for a short duration of time. During this brief period, the ground station and the satellite have to get in contact with each other in order to stablish a communication link. Being this time so short, makes valuable every second of connection, so knowing the in-view period becomes essential. Therefore, predicting the visibility of VLEO and LEO satellites at a ground station is valuable for a number of reasons.

Firstly, knowing the visibility windows duration would permit the engineers of the space mission to plan the uplinks and downlinks of information. This would save time and no information would get lost. Secondly, ground stations and satellites could conserve power by going into a power conserving mode till the next rise time of the satellite.

On the other hand, during the visibility windows time, the geometry between the satellite and the ground station changes. The communication link between the objects is better when the satellite is passing as closer as possible to the ground station, so knowing when this occurs would be interesting to increase the data rate. However, for the present thesis, this improvement is not implemented since the data rate is considered constant during all the visibility windows.

Finally, the acquired knowledge while carrying out this study can be divulged so as to other engineering students can learn about the communication link issues between satellites and ground stations.

1.5 Study's approach

The objective of the thesis is to predict the visibility windows duration of a satellite orbiting in a VLEO or LEO. Orbits will be circulars and the Earth is going to be supposed as a perfect sphere.

To approach this goal certain tasks has to be done. The study will start with an information research about satellites and their different orbits around the Earth. Once this introduction part is completed, the elements that affect the visibility windows are going to be studied and a method to predict the visibility time is going to be exposed.

After that, a code will be implemented and with it, the visibility window time and the design data rate for a pair of one satellite with one ground station will be obtained. On the other hand, a brief summary of how the information is treated in order to be sent from the satellite to the ground station, is going to be written.

Finally, with the results obtained, some conclusions will be extracted. Also an environmental and security impact, together with the annexes and the budget documents will be performed.

1.6 Planning

In order to achieve the goal of this thesis, the work that has to be completed has been divided into different work packages. The tasks explained in the scope section have been broken down into subtasks and coded respectively.

Table 1.1: Interdependency relationship among tasks

Code of task	Task identification	Preceding task
0	**Information research**	
1	**Theoretical Part**	
1.1	Describe satellites, in particular CubeSats	0
1.2	Identify types of orbits	0
1.3	Identify and describe what elements affect visibility window time	0 - 1.2
1.4	Parametrize and analyse how the elements that affect visibility window time	0 - 1.3
1.5	Introduction to communication architecture	0 - 1.3
2	**Practical Part**	
2.1	Develop a code to calculate visibility window time and design data rate	0 - 1.3
2.2	Compare and validate the results with qbapp app	0 - 2.1
3	**Deliverables**	
3.1	Project Charter	0 - 1.1
3.2	Follow-up 1	
3.3	Follow-up 2	
3.4	Follow-up 3	
3.5	Delivery	
3.5.1	Report	1 - 2
3.5.2	Annexes and budget	1 - 2 - 3.5.1
3.6	Presentation	
3.6.1	Presentation realization	1 - 2 - 3.5.1 - 3.5.2
3.6.2	Presentation preparation	3.6.1

Gantt Diagram

Once identified the work packages and their tasks, the calendar has been organized to complete all of them. For a better representation of the time, a Gantt diagram has been developed.

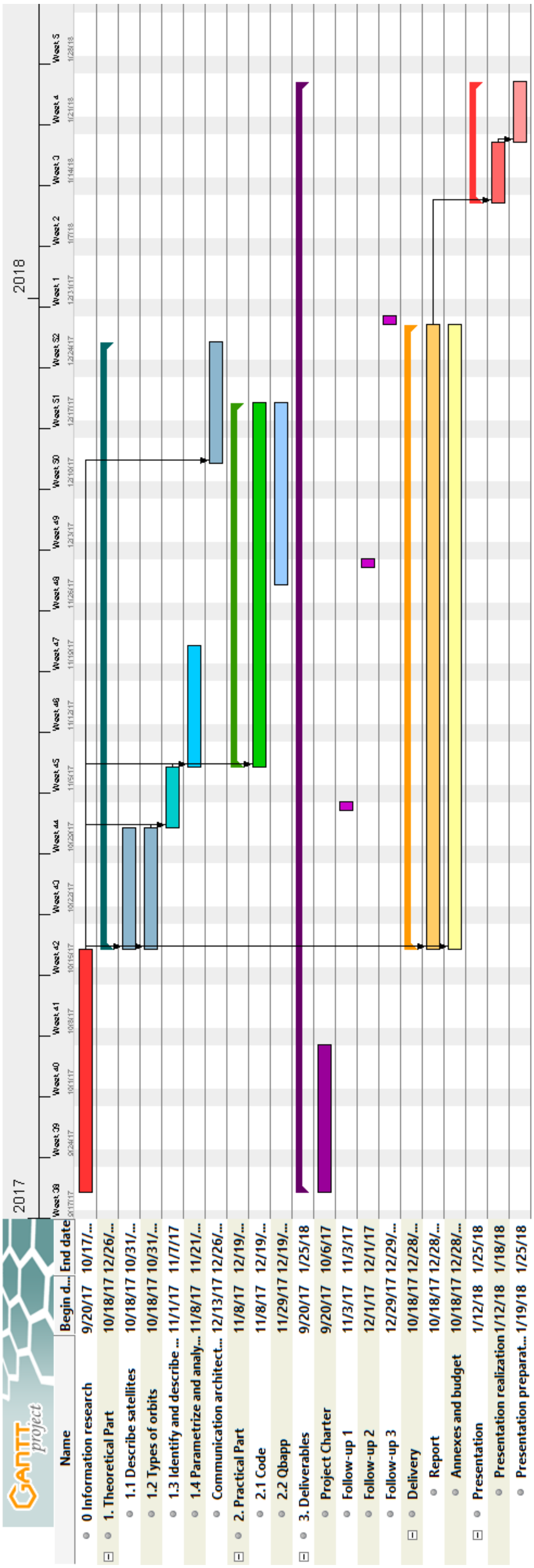


Figure 1.1: Gantt diagram

Chapter 2

Satellites and Orbits

In this chapter, a brief summary of what is a satellite and the current tendencies in space technologies are presented. Also, basic orbital mechanics concepts are defined defined, paying a special attention to the special characteristics of Low Earth Orbit and Very Low Earth Orbit.

2.1 Satellites

A satellite is a celestial body or machine which orbits a planet or star. It can be natural, as Moon to the Earth, or artificial, i.e., man-made object which has been sent into space.

Every celestial body with an identified orbit around a planet, has been considered a natural satellite. The ambiguity of this definition shows up when studying the rings of Saturn, for example. Countless small particles ranging from μm to km in size orbit around Saturn. With the aforementioned definition, these particles would be considered satellites but they are not. If the object is ten times smaller than the planet or star it orbits, it is considered a moonlet.

On the other hand, there are the man-made satellites, being the ones considered in the present study. Humans have produced them in order to orbit around the Earth but also, to send them on missions through the space. Depending on their purposes, they are built differently. Even though, there are parts which all of them contain; the antenna, the power source, the payload, the altitude subsystem, among others.

Firstly, the antenna is indispensable for communicating the satellite with the ground stations located at the Earth's surface. Without it, all the data collected is useless, and it is also impossible to send instructions to the satellite. Secondly, a power source is needed

for different purposes such as establishing the satellite in the correct orbit or path that it will make through the space, or to maintain the payload machines working properly.
Payload is not an essential part for keeping the satellite active, but it is the cause of the satellite construction. There exist multiples types of payload; intended for communications, imaging, navigation and also to produce science, such as biological and technological tests. With them it is possible to observe the Earth from the space, to communicate by mobile phone, to surf the internet or to study the growing of a plant in microgravity conditions, among many others examples.

Artificial satellites can be classified by different categories:

- By height of orbit: Low Earth Orbit, Medium Earth Orbit (MEO), High Earth Orbit (HEO)
- By application: Telecommunications, Observation, Weather, Military, Scientific etc.
- By size: Nano-sats, Micro-sats, Small, Medium, Large
- By power consumption: measured in kilowatts
- By country of origin and region of operation: SPAINSAT is a Spanish military geostationary communications satellite or EUTELSAT satellites which are owned by the European Telecommunications Satellite Organization.

Focusing on the classification by the satellite size, the weight difference between a large satellite and a smaller one can be around 700 times. In Figure 2.1 it can be seen a comparison of weights between satellites and more common objects.

Figure 2.1: Satellite size comparison
[2]

Classification by mass is useful because it has a direct bearing on the launcher vs. cost tradeoff. The Table 2.1 below illustrates a scheme for classifying satellites in terms of deployed mass. Within this scheme the term ”small satellite” is used to cover all spacecraft with in-orbit masses of less than 500 kg.

Table 2.1: Satellite mass classification

Category	**Mass range (Kg)**
Large satellite	>1.000
Medium satellite	500-1.000
Minisatellite	100-500
Microsatellite	10-100
Nanosatellite	1-10
Picosatellite	0.1-1
Femtosatellite	<0.1

Satellites have evolved very fast during the last century. The first satellite that humans put in orbit was the Russian Sputnik 1. The Soviet Union launched it into an elliptical Low Earth orbit on 4 October 1957. It had a mass of 83.6 kilograms (Minisatellite) and the pressurized sphere made of aluminium alloy had five primary scientific objectives: Test the method of placing an artificial satellite into Earth orbit; provide information on the density of the atmosphere by calculating its lifetime in orbit; test radio and optical methods of orbital tracking; determine the effects of radio wave propagation though the atmosphere; and, check principles of pressurization used on the satellites [3].

Since then, there have been many satellites put in orbit, with multiple purposes. The first Human spaceflight was also launched by the Soviet Union on 12 April 1961 with the cosmonaut Yuri Gagarin aboard. Now, the largest satellite orbiting around the Earth is the International Space Station, intended to be a laboratory, observatory and factory in low Earth orbit. It is also planned that it provides transportation, maintenance, and act as a staging base for possible future missions to the Moon, Mars and asteroids [4].

Nowadays, the tendency to build large satellites is changing and this is because the design of a mission, which cost millions of dollars, is not always profitable. One of the objectives of the new satellites is to try groundbreaking technologies, that might work or not. For that, expensive in time and money missions are only affordable for governments and private space companies with a large trajectory in the business.
Smaller satellites are an opportunity for universities and research centres. They do not have large budgets and all the necessary infrastructures in order to build a big satellite is out of their grasp. One of the most known type of small satellites are CubeSats. Next subsection gets deep in what they are and their possibilities.

2.1.1 CubeSats

CubeSats are a class of nanosatellites which use a standardized size and form. A one unit CubeSat, 1U, measures 10x10x11 cm and can be extended to 2U, 3U, 4U, 6U, 8U and the biggest one, 12U. They were developed in 1998 by California Polytechnic State University at San Luis Obispo and Stanford University in order to provide a platform for education and space exploration. Specialised infrastructures or large budgets are not needed to construct a CubeSat, so universities around the world have developed their own ones with wide different proposes; Earth Observation (EO), science investigations, new technology demonstrations and advanced mission concepts [5]. According to the World's largest database of nanosatellites up to July 2017, 764 CubeSats had been launched together with 829 nanosatellites, satellites from 1 kg to 10 kg, and most of these launches have occurred in the last 5 years as it can be seen in Figure 2.2

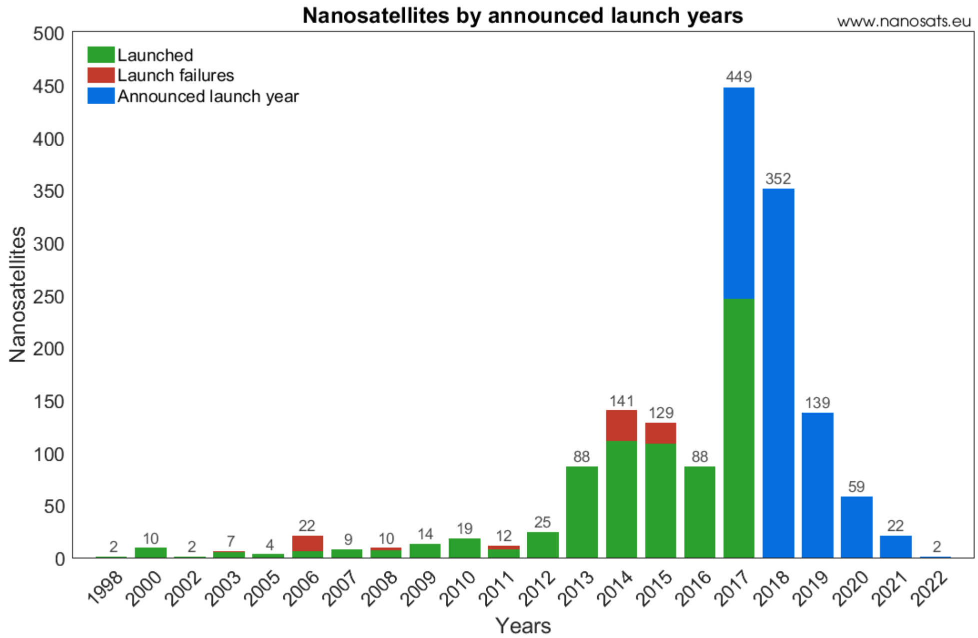


Figure 2.2: Number of nanosatellites by announced launch year [6]

Hence, thanks to their easy construction and launch, space experts identified a niche of market and began to create their own companies. These companies can be classified in three main groups; CubeSat manufacture, information data sale and resulting downstream applications [7].

First group, CubeSat manufacture, encompasses product design and construction, together with systems and components development. Some companies sell the complete CubeSat without payload, that has to be added depending on the mission objectives. An example of this kind of companies is EnduroSat [8]. Others such as Clyde Space [9] of Pumpkin Space Systems [10] sell CubeSats by pieces (structures, solar panels, NSL Globalstar Communication Systems, etc). Furthermore, some of them offer ground segment solutions and launch services in order to complete all the needs of a nanosatellite mission (Surrey Satellites Technology Ltd [11]).

On the other hand, there is the business of information data sale. Conventional satellites are big, heavy and costs millions to build and launch. Most of the companies dedicated to capture information from the space were governmental or private, with an extensive background of experience in space missions and also important investment budgets. Thus and because they did not have lot of competition, the product sold was expensive and not affordable to all clients. CubeSats orbiting at VLEO or LEO provide a cost-effective platform. They have the opportunity to offer the same or better product but much

cheaper. The two main reasons are: construction is easier and launch can be done as a secondary payload of a big satellite, thus reducing costs.
Finally, all information data obtained with CubeSats can be used for a wide range of applications. A list of the most relevant ones is exposed next:

- Agriculture Health Monitoring
- Humanitarian Aid
- Insurance Modelling
- Oil Storage Monitoring
- Natural Disaster Response
- Oil and Gas Infrastructure Monitoring
- Financial Trading Intelligence
- Mining Operations Monitoring
- Carbon Monitoring
- Maritime Monitoring

Also, Figure 2.3 shows the tends by purpose of nanosatellites and CubeSats. The first graph, shows the data between 2009 and 2013, being the percentage for EO only a 12%. By contrast, the projection *Spaceworks* made for 2014-2016 showed that this percentage would dramatically increase to 52%. Other purposes such as trying new technology or biologic experiments were expected to reduce their presence.

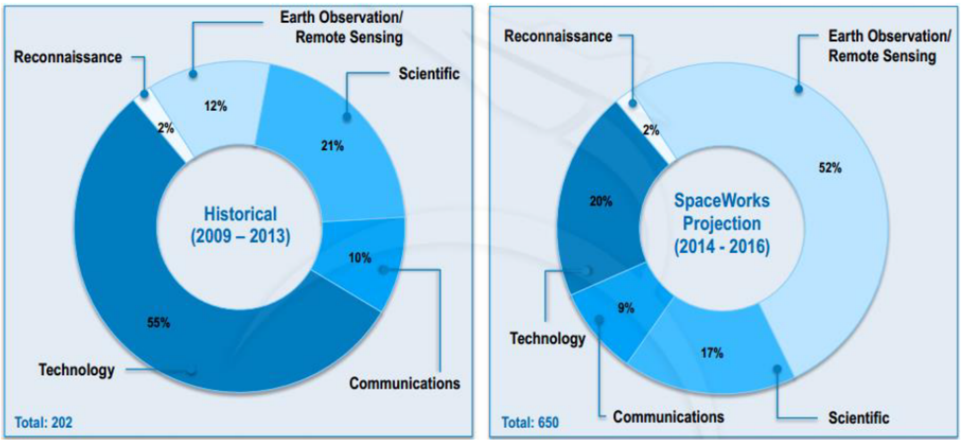


Figure 2.3: Nano/Micro-Satellite Trends by Purpose
[7]

Therefore, SpaceWorks and Euroconsult have come to the conclusion that the most likely application to be disrupted by the small mission technology trend is EO, which is also demonstrated by the success of high-profile Silicon Valley based start-ups like Planet Labs and Skybox Imaging. If all this data could be accessible in real time, the applications would boost greatly. [7]

2.2 Orbits

As mentioned above, satellites move in an orbit: a regular, repeating path that one object in space takes around another one.

2.2.1 Characterization of an orbit

As it has been said, the path followed by a satellite is referred to as its orbit. There can be multiple orbits around a celestial object, so an standard way of specify an orbit, is knowing what is called the orbital elements. The traditional ones are the six Keplerian elements; three for spatial dimensions and the others three, for velocities.

In the Figure 2.4, the abovementioned elements are represented together with the orbital plane. This plane intersects with the celestial equatorial plane in a line called the line of nodes and it is known as the geometrical plane in which the orbit of the satellite lies. On the other hand, the ascending node is defined as the point on the celestial equator at which the satellite moves form the southern to the northern hemisphere. All these concepts are important in order to understand the six independent Keplerian elements.

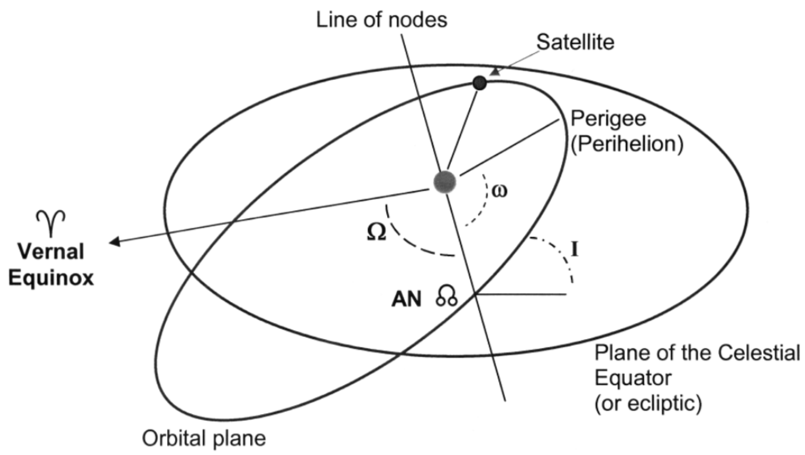


Figure 2.4: Keplerian elements
[12]

- **Eccentricity (e) [-] :** refers to the orbit's shape. A satellite with a low eccentricity orbit (e=0) moves in a circle around the Earth. On the other hand, an eccentric orbit is elliptical ($0 \leq e \leq 1$), with the satellite's distance from Earth change depending on where it is in its orbit.
- **Semi-major axis (a) [m] :** is the sum of the periapsis and apoapsis distances divided by two. These two distances are the closest and farthest points of the

satellite orbit to the Earth, respectively. For circular orbits, the semi-major axis is the distance between the centres of the bodies and for parabolas or hyperboles, it is infinite. The semi-major axis refers to the orbit's size.

- **Inclination (i) [°]:** is the angle of the orbit in relation to Earth's equator. A satellite that orbits directly above the equator has zero inclination. If a satellite orbits from the north pole (geographic, not magnetic) to the south pole, its inclination is 90 degrees.
- **Right ascension (or longitude) of the ascending node (Ω) [° or rad] :** horizontally orients the ascending node of the ellipse (where the orbit passes upward through the reference plane) with respect to the reference frame's vernal point
- **Argument of periapsis (ω) [° or rad] :** defines the orientation of the ellipse in the orbital plane, as an angle measured in the direction of the motion between the line of nodes to the vector **e**, which points to the periapsis.
- **True anomaly (v) [° or rad] at epoch (M_0) [s]:** defines the position of the orbiting body along the ellipse at a specific time.

An advantage of the classical orbital elements is that they can be interpreted intuitively and all except the true anomaly, are constant for motion for the two-body problem.

2.3 Types of orbits

For human-made satellites which orbit around the Earth, there are lots of different classifications but one of the most important is grouping by the altitude from the Earth's surface, i.e., the radius orbit taking the centre of the circumference the Earth's core.

There are four main groups in the altitude orbit classification.

Table 2.2: Orbits: Altitude classification

[13]

Orbit type	Mission	Altitude	Period	Tilt	Shape
LEO					
· Polar sun-synchronous · Inclined nonpolar · Polar non-sun-synchronous	Remote sensing/weather International Space Station Earth observing scientific	150-900 km 340 km 450-600 km	98-104 min 91 min 90-101 min	98° 51.6° 80-94°	circular circular circular
MEO					
Semisynchronous	Navigation, communications, space environment	20.100 km	12 hours	55°	circular
GEO					
· Geosynchronous · Geostationary	Communication, early warning, nuclear detection, weather	35.786km	24 h (23h 56' 04s)	0°	circular
HEO					
Molninya	Communications	Varies from 495 km -39.587	12h (11h 58m)	63.4°	long ellipse

The first one to be explained is Geosynchronous Orbit (GEO). It stays approximately at 35,786 km above the equator and complete one revolution around Earth precisely every 24 hours. In the GEO it can be found communications and weather satellites which try to avoid the rotation of the ground stations placed in the Earth's surface. GEO is included in the group of High Earth orbits having altitudes which vary from approximately 500 to 39.500 km.

Secondly, lowering the altitude, there is the Mid or Medium Earth Orbit. Historically, the MEO has been less used than GEO or LEO and the main reason is because the presence of the Van Allen radiation belts (a zone of charged particles that extends to a distance of 60.000 km from the Earth's surface) which occupies part of this zone. Usually, navigation satellites occupy MEO.

One of the other orbits in this classification is the LEO, which is where the International Space Station stays. LEO is the first 161 to 2.000 km of space and is the easiest orbit to get and to and stay in. Those are two of the reasons why most scientific satellites and many weather ones are placed in these orbits. Also because one complete orbit in LEO lasts between 88 and 127 minutes depending on the altitude and thus, much information per day from a specific place can be taken [14]. Any object below this altitude will suffer from orbital decay and will rapidly descend into the atmosphere, either burning up or crashing on the surface.

Figure 2.5: Different orbits by altitude.
[15]

The height of the orbit, or distance between the satellite and Earth's surface, determines how quickly the satellite moves around the Earth. An Earth-orbiting satellite's motion is mostly controlled by Earth's gravity. As satellites get closer to Earth, the pull of gravity gets stronger, and the satellite moves more quickly.

GEO and LEO are quite saturate of satellites adding also, the problem of space debris. Approximately 95% of the objects orbiting around the Earth are orbital debris, i.e., not functional satellites. In Figure 2.6 it can be seen a larger population of objects over the northern hemisphere due mostly to Russian objects in high-inclination, high-eccentricity orbits.

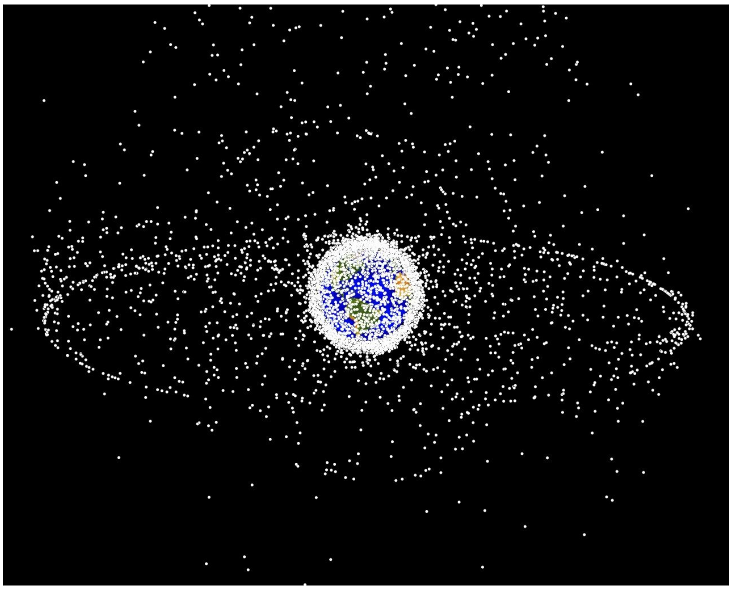


Figure 2.6: Computer generated image of objects in Earth orbit currently being tracked [16]

Objects which are in a LEO are subject to atmospheric drag since they are still within the upper layers of Earth's atmosphere; thermosphere (from 80 km to 500 km), thermopause (from 500 km to 1.000 km), and the exosphere (1.000 km and beyond). The higher the object's orbit, the lower the atmospheric density and drag, so it is a point to take into consideration.

Because human curiosity never stops, the doubt whether if it was more convenient to orbit at the very beginning of the LEO border started to born in some space experts. It arosed the concept of VLEO which, as a first approximation, was defined as those orbits with a mean altitude below 450 km. Generally, at 450 km, the presence of a denser atmosphere and hence a strong aerodynamic drag, make a spacecraft decay in less than 5 years, requiring significant changes in traditional spacecraft mission designs.

VLEO opens the opportunity to improve the existing applications e.g. EO or communications, but also to create new ones yet to come. Josep Virgili Llop together with some experts of The University of Manchester have written a paper called "Very Low Earth

Orbit mission concepts for Earth Observation. Benefits and challenges" that summaries the potential of VLEO orbits [1].

2.3.1 Challenges and opportunities of VLEO

Nowadays, most satellites put in a VLEO are destined to EO business. This is because their proximity to the Earth's surface that makes easier getting pictures with high resolution having smaller cameras. In other words, EO satellites can increase their competitiveness by decreasing the operative height in which they work.

The main benefits to VLEO are described below with some of them being already identified above:

- **Resolution of optical payloads:** The ground resolution is one of the most important parameters for this kind of payloads. There is a significant difference between an image with 3-5m of resolution or another with 10m or 0.5m, being the last option the one with more information but also the hardest one to obtain.
  The angular resolution limit of a telescope is determined by the Rayleigh criterion, and assuming visible spectrum the ground resolution is described by the

  $$res = 6,71 \cdot 10^{-9} \frac{H}{D} \tag{2.1}$$

  where:
  res = ground resolution
  H = spacecraft altitude
  D = aperture diameter of the telescope

  To increase the ground resolution there are two main options. The first one is to increase the aperture size of the optics, meaning bigger and heavier cameras. Second option consists on reducing the operating altitude without modifying the optics, a solution which implies less economic impact.

- **Increase of the Effective Surveillance Footprint Size:** If only a minimum useful resolution is required for the platform, then the lower the spacecraft flies the bigger will be the area that complies with this requirement. That is due to the shorter path length to the target. This effect causes the timeliness of revisit to also improve.

- **No-Deorbit required:** The average satellite lifetime which orbits 600-700 km above the Earth's surface is between 15 and 78 years. European Space Agency

(ESA) guidelines state that an inactive satellite should be de-orbited within 25 years. Satellites orbiting at 400 km are operable for 3 years and this is because the atmospheric drag. Being at such low altitude, the density is much higher and particles of the atmosphere impact with the satellite which renter at such a high speed that burns before touching the Earth's surface. This can be seen as something negative since it is very rooted in the society that satellites are really expensive and difficult to built so they have to last for many years. A change of outlook is proposed by others scientist who think this short lifetime is an opportunity which enables new technology enter to the space every three years more or less.

- **Lower Risk of Collision with Space Debris:** As explained before, the region for VLEO is not crowed with satellites if it is compared with the higher LEO, because is a region less explored. Also because the rapid decay of the satellites orbiting meaning the VLEOs clear itself from debris.

VLEO missions also have some challenges needed to be solved to achieve a good spacecraft performance. The most relevant challenges are exposed next, and a brief description of each is added:

- **Aerodynamic forces:** As VLEO is the set of orbits closer to the Earth, it means the satellites orbiting in them, are affected by the density of the atmosphere. Therefore, the drag turns out to be an important aerodynamic force, which cause the decay of the satellite. The spacecraft design in order to adopt low drag configurations and a power source are necessary to limit and correct the altitude of the satellite, respectively.

- **Atomic oxygen erosion:** Atomic oxygen is one of the main atmospheric constituents of the thermosphere. It is highly reactive and can damage the satellite surface, together with the optical/thermal coatings and sensors surfaces degrading their performance.

- Windows communications: Orbiting at a lower altitude reduces the communications windows over a certain ground station. The data needed to be downloaded and the orders which the satellite has to receive in order to properly operate, have to be transmitted in a short period of time.

EO market has been traditionally providing services mainly to Military and Defence market, which requires a very high ground resolution but do not demand a high revisit time. This market has been monopolised by traditional space agencies which have enough resources to design and perform expensive missions. Their product has the best quality but is also pricey, being not affordable for research groups or companies other than governmental.

The demand of EO products has grown and more enterprises are trying to stablish themselves in the market. New private start-ups and governments with own space programs have began to commercialize their own solutions, making available a wide range of EO products. Most of these new initiatives are developing low cost technologies to drop down EO satellite missions's cost and to make them more competitive against the already established solutions. The Military and Defence market is still unreachable for these new enterprises due to the strong technological barriers imposed by the organizations established in this market but other markets are open such as maritime, natural resources monitoring, location based services, disaster management, energy and environment monitoring.

Chapter 3

Communication link geometry

The present chapter, introduces the theoretical base necessary to predict when a satellite is visible for a determinate ground station. Using this background, a Matlab code has been implemented to put into practice the studied concepts. The code is presented at the second part of the chapter, highlighting the difficulties founded while writing it.

3.1 Visibility window

The time interval over which a satellite is visible from a given ground station on the Earth is called the duration of visibility or visibility window. This period of time is the one that the satellite has in order to stablish an uplink or downlink communication with the Earth. It varies for each satellite's orbit and, also, every time the satellite passes over the ground station, so predicting it, enables the user to better design the space mission.

Before going deeper in discussing the theory to predict the visibility window, it is essential to have notions of the orbital motion. For that, a brief introduction of the laws of the astrodynamics is presented next.

Orbit motion

Back in the XVII century, Johannes Kepler was the first person to describe the motion of planets around the Sun with three laws.

His first law, published in 1606, exposes that planets are orbiting the Sun in a ellipse path with the Sun at one of the two ellipse focus. He reached this conclusion after studying the Mars's orbit. Law of equal areas has his second law (1619) and it states that the line joining the planet to the sun sweeps out equal areas in equal times. Based on his observation of Mars, Kepler reasoned that a planet's speed depended on its distance from the sun.

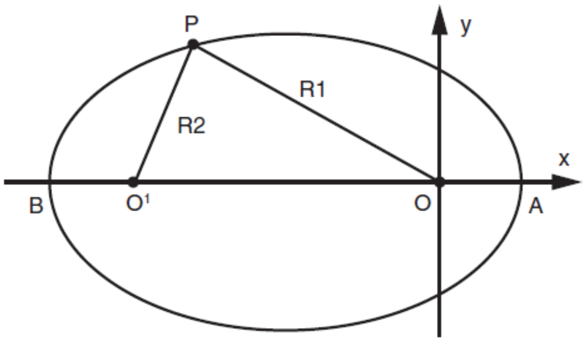


Figure 3.1: Kepler's first law [17]

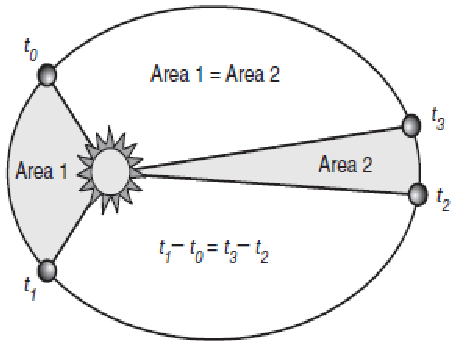


Figure 3.2: Kepler's second law [17]

Finally, Kepler's third law says that the square of the orbital period of a planet is proportional to the cube of the mean distance from the sun. By this law, the altitude of a circular orbit uniquely determines how long it will take to travel around the earth and vice versa.

$$\frac{P^2}{a^3} = \frac{4\pi^2}{G(M+m)} \approx \frac{4\pi^2}{GM} \tag{3.1}$$

where:

P = period of the planet [s]

a = mean value between the max. and min. distances between the planet and sun [m]

G = $6.674 \cdot 10^-11$ $[m^3 \cdot kg^-1 \cdot s^-2]$, universal gravitational constant

M = mass of the sun [kg]

m = mass of the planet [kg]

The laws Kepler developed, describe the observed motions of the planets, but they made no attempt to describe the forces behind those laws. The laws regarding those forces are the key to understand the physicals behind the movement of an object around another. This work was formulated by Isaac Newton.

The law of inertia is Newton's first law. It states that every body continues in a state of rest, or of uniform motion in a straight line, unless it is compelled to change that state by a force imposed upon it.

To reach the definition of force, first, the concept momentum has to be explained. Momentum is a vector quantity defined as the product of an object's mass and its relative velocity,Equation 3.2 . So, assuming the mass of an object remains constant, if there is a change in momentum there must be a change in velocity of the object as well, Equation 3.3. Now, force can be defined as the time rate of change of an object's momentum.

As acceleration is defined as the change in velocity over time, the concept force is completely defined, Equation 3.4.

$$p = mv \tag{3.2}$$

$$\Delta p = m\Delta v \tag{3.3}$$

$$F = \frac{\Delta p}{\Delta t} = \frac{m\Delta v}{\Delta t} = ma \tag{3.4}$$

Therefore, the statement of the second law of Newton is when a force is applied to a body, the time rate of change of momentum is proportional to the applied force and in the direction of it. Then it comes the action-reaction Newton's law is his third one. It exposes that for every action there is a reaction that is equal in magnitude but opposite in direction to the action. This leads to the conservation of momentum.

Finally, there is the Newton's law of universal gravitation which says that every particle in the universe attracts every other particle with a force that is proportional to the product of the masses and inversely proportional to the square of the distance between the particles.

$$F_g = G\left(\frac{Mm}{R^2}\right) \tag{3.5}$$

Therefore, taking into consideration all Newton’s laws at the planet-satellite problem, it can be said that the planet applies a force of attraction to the satellite. For its part, the satellite applies the same force to the planet but at the opposite direction. The reactions of both objects to the forces applied to them are different because their masses differ. While the planet has a much higher mass, the force produced by the satellite does not implies a perceptible effect, whereas the satellite being much lighter, the force produced by the planet, oblige the satellite to stay in an orbit.

Earth’s rotation

After the general overview of the orbit motion its time to begin to focus on the problem of the communication link geometry. To start with, lets pretend a nonrotating Earth with a circular orbiting satellite. The ground track the satellite would drawn over the Earth surface would be a circle passing over the same terrestrial points every orbit. But because the Earth does rotate, by the time the satellite returns to the same place in its orbit after one revolution, the Earth has rotated eastward at a rate of 15° per hour. The ground track, therefore, looks like it has moved westward on the Earth's surface. The amount of equatorial orbital shift is proportional to the time it takes for one orbit (the period). For instance, the lower the altitude of a satellite, the shorter the time it takes to travel around the Earth, i.e., the satellite has a faster period. This means that in a day duration, for example, the satellite will trace more orbits around the Earth, being them closer between each other.

On the other hand, farther out satellites, travel slower. They will have long periods meaning their passes over a ground station will last longer. So the satellites which have the most important constrain in communicating with the ground stations are the ones orbiting in a LEO and, even more, those in a VLEO. It is for that reason that the study is focused in predicting the visibility of this type of obits.

To represent the previous explanation, Figure 3.3 and Figure 3.4, show the ground track of two satellites, which have different altitudes thus, their equatorial orbital shifts vary. [18].

Table 3.1: ERBS details

ERBS	**Altitude: *608,9 km***
Inclination: $57,13°$	Period: $96,85min$
Time span shown: $720,0min = 0,50day$	Equat. orbital shift: $2.732,1km(24.5°)$

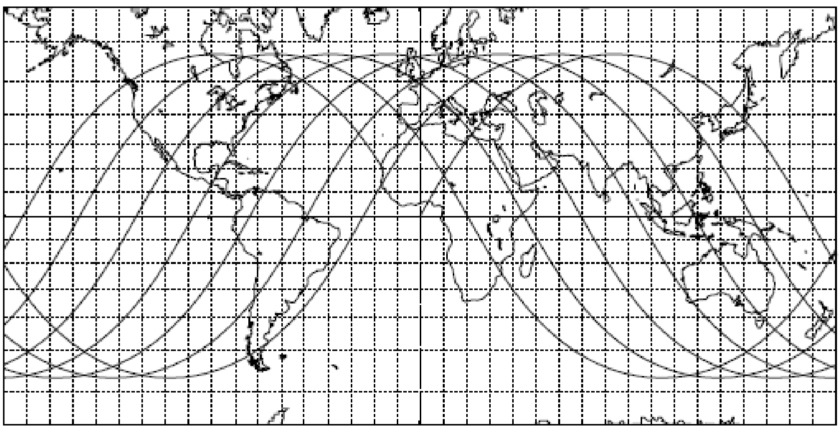


Figure 3.3: ERBS Ground track for one day
[18]

Table 3.2: Meteor-3-07 details

Meteor-3-07	**Altitude: 1.194,6 km**
Inclination: $82,56°$	Period: $109,42min$
Time span shown: $109,42min = 0,08day$	Equat. orbital shift: $3.059,5km$

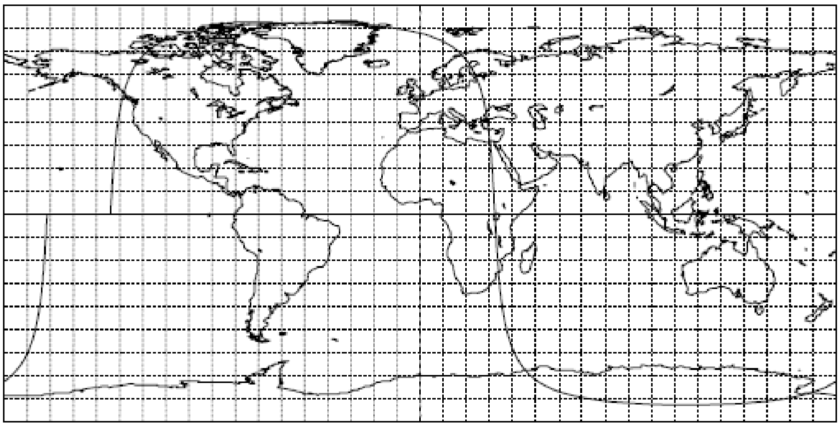


Figure 3.4: Meteor-3-07 Ground track for a period
[18]

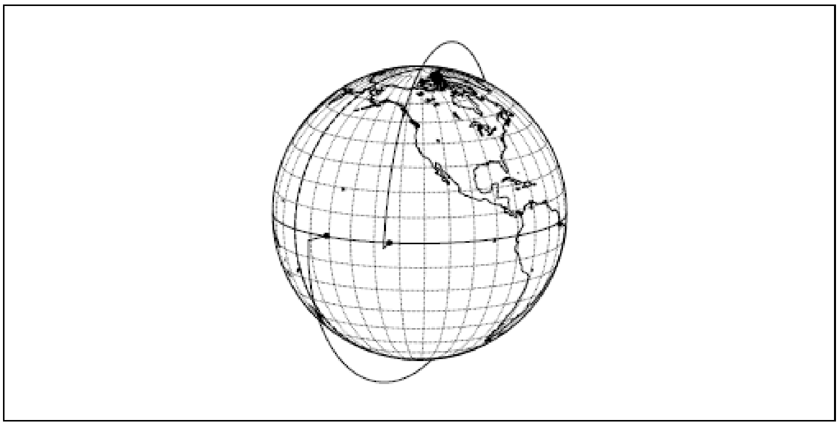


Figure 3.5: Equat. orbital shift fer period Meteor-3-07, 3D view
[18]

It is important to bear in mind this concept because while the satellite is orbiting, the Earth has a rotating movement. This introduces another fundamental issue which has to be consider; the frame of reference.

The reference frame is an abstract coordinate system and a set of physical reference points that uniquely define (locate and orient) the coordinate system and standardize measurements. The relative movement between the Earth and any satellite which orbits around it, can be studied in diverse reference frames. Earth-centred fixed and Earth-centred inertial are two of them, both usually used.
The Earth-centred fixed reference frame (ECF or ECEF) is a cartesian coordinate system and the point (0,0,0) is situated at the centre of mass of the Earth. Its axes are aligned with the international reference pole (IRP) and international reference meridian (IRM) (prime meridian), that are fixed with respect to the surface of the Earth. Consequently, while the Earth is rotating, the reference frame does it too, at the same time. By contrast,

the Earth-centred inertial reference frame (ECI), remain fixed relative to the stars. It also has the centre of the coordinate system at (0,0,0) but its axis X is fixed to the vernal equinox direction, axis Z is the Earth's rotation direction and the third axis, Y, is the remaining one to complete the coordinate system with three perpendicular axis.

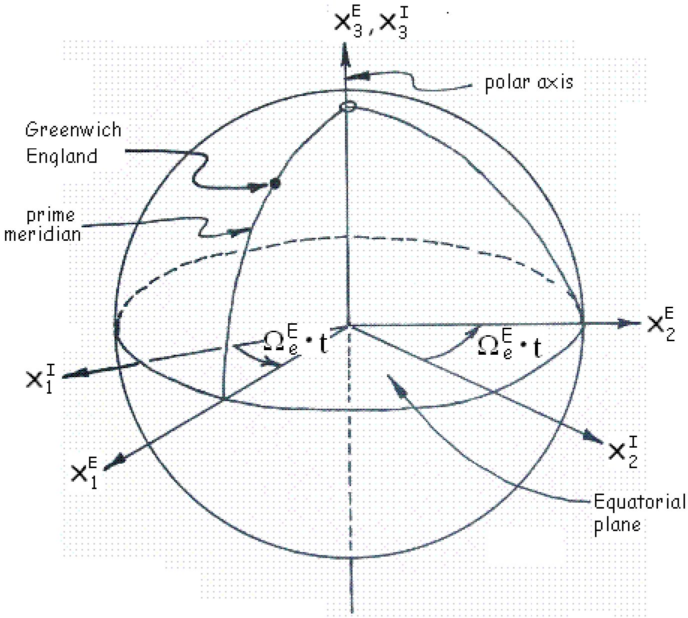


Figure 3.6: Earth-centred fixed and Earth-centred inertial reference frames [19]

Using the ECF reference frame, it is easy to situate a ground station on the surface of the Earth. Its position will be defined by two angles, the longitude, λ, and the latitude, ϕ, and also a distance, the radius of the Earth. Where the ground station is installed, a plane tangent to the Earth's surface can be traced. This plane is called *Horizon Plane, hp*.

3.1.1 Horizon plane

As it has been previously mentioned, the horizon plane is considered a tangent plane to the surface of the Earth at the observer's position, in this case of study, a ground station. For this observer, the position of the satellite within its orbit considered, can be defined by azimuth and elevation angles.

The azimuth, Az, is the angle of the direction of the satellite, measured in the horizon plane from geographical north in clockwise direction. Its range goes from 0° to 360°. On the other hand, the elevation angle, El or θ, is the angle between a satellite and the ground station's horizontal plane, and it can measure from 0° to 90°. When the satellite is located on the horizon plane, the elevation angle is 0°. Immediately after that moment, when the satellite crosses it, the angle increases up to its maximum ($\theta_{\max}$) and later, decreases until it is 0° again. When the satellite is behind the plane, it is not visible for the ground station, so the communication link is not possible.

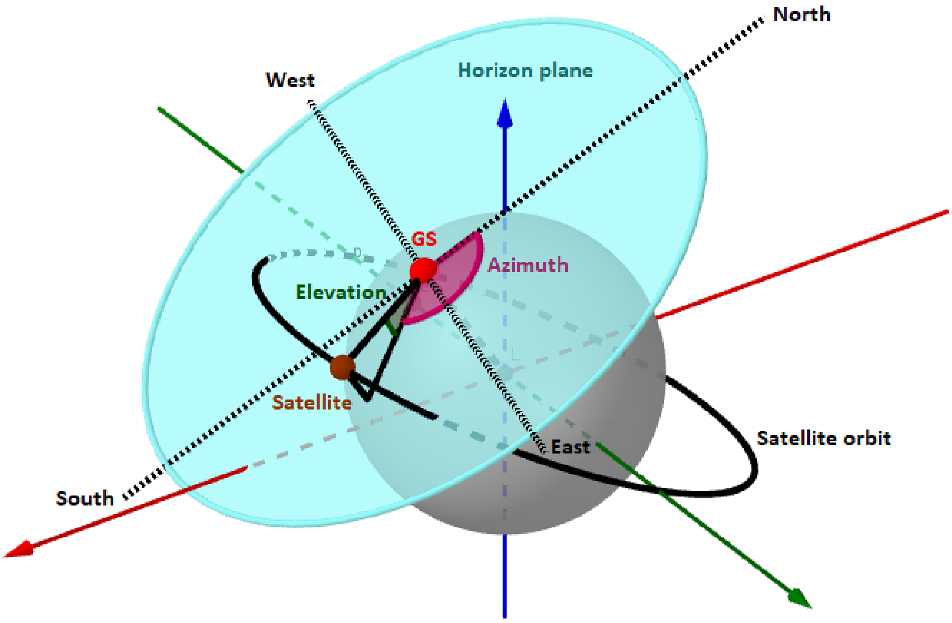


Figure 3.7: Azimuth, elevation and horizon plane

Therefore, the horizon plane is the circle with north on the top ($Az = 0°$), then East ($Az = 90°$), South ($Az = 180°$) and West ($Az = 270°$) which all its perimeter has an elevation angle of 0°. Figure 3.8 represents the horizontal plane seen from above. The three concentric circles, mean different elevations; 0°, 30° and 60°. At the centre, where the ground station is located, the elevation is $\theta = 90°$.

In fact, an horizontal plane like the one described above is not perfect. Around the ground stations there can be different natural or artificial obstacles like mountains or high buildings that suppose a difficulty for the connection link between the satellite and the ground station. Sometimes, the horizontal plane is designed with an offset angle of elevation, transforming it to a cone where the ground station is the vertex. In other cases, an study is done in order to analyse the obstacles and finally obtain the practical horizontal plane, Figure 3.9. However, for the present study, an ideal horizon plane has been considered in order to obtain more general results.

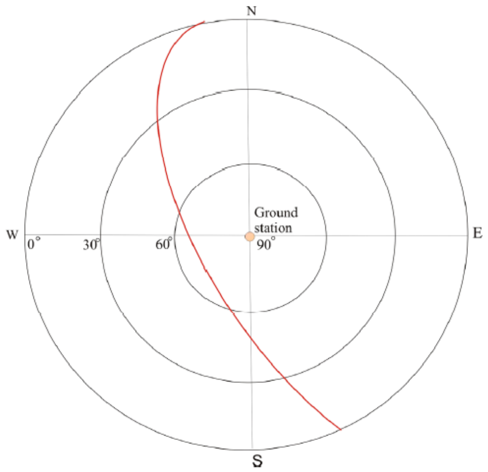


Figure 3.8: A figure
[20]

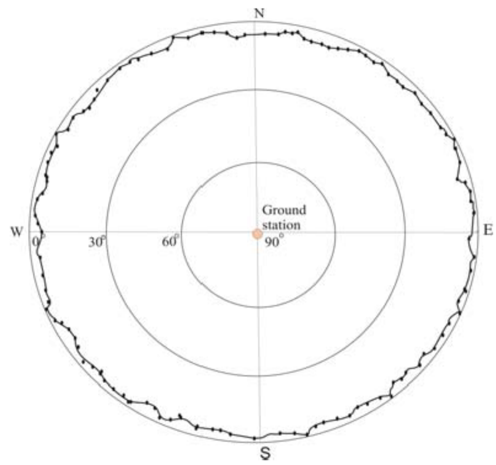


Figure 3.9: Another figure
[20]

In Figure 3.8, the ground track of the satellite is represented with a red line. When the satellite first enters to the horizon plane is named the rise time of the satellite (RTS). It is the beginning of the visibility windows which will end at the set time of the satellite (STS). The time comprised between the RST and STS is the visibility window time (VWT). From the ground track example exposed in the figure, it can be said that the RST occurs at $El = 0°$ and $Az \approx 170°$ and the STS at $El = 0°$ and $Az \approx 350°$. The maximum elevation angle is $\theta_{max} \approx 70°$.

3.1.2 Constants and initial parameters

In order to analyse the communication link geometry, it is indispensable the use of certain computational and physical constants, as well as the orbital parameters which describe the satellite path. Also, an initial reference location and time has to be defined because the results obtained are going to be referred to this space-time coordinates.

Constants

There are multiple types of space missions. One can consist of only one satellite which communicates with only one terminal, whereas another can be the satellite communicating with multiple ground stations. For missions with LEO and VLEO satellites, it is frequent to find a constellation configuration. This means that multiple satellites are orbiting in different orbits and they communicate with different grounds stations or in some cases, between them. For that reason, two of the computational constants are the number of satellites and ground stations the space mission is going to have.

For the present study, only the communication between the ground station and the satellite has been considered. The communication link between satellites is out of the scope.

Regarding the physical constants, the radius and the mass of the Earth are necessary, together with the gravitational constant.

Orbital parameters

As it has been mentioned in subsection 2.2.1, an orbit is defined with its orbital parameters. They can be obtained for a certain satellite by different methods, but the most common one is the two-line element set (TLE). A TLE is a data format encoding a list of orbital elements of an Earth-orbiting object for a given point in time, the epoch. The data is fitted into 70 columns and 3 rows. Next is presented an explanation on how to interpret the TLE set with the example of the International Space Station (ISS) orbit.

The TLE of the ISS at a certain time is:

ISS (ZARYA)
1 25544U 98067A 08264.51782528 -.00002182 00000-0 -11606-4 0 2927
2 25544 51.6416 247.4627 0006703 130.5360 325.0288 15.72125391563537

The first line is the title line, where the orbit name is presented together with the satellite name. After it, the next two lines have all the information required to precisely know the orbital parameters and also the satellite position at a certain time.

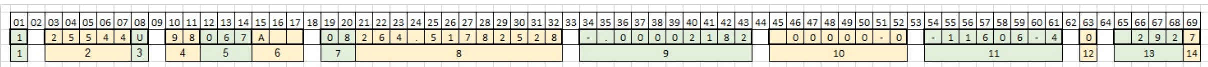


Figure 3.10: TLE Line 1

Table 3.3: Parameters of the first TLE line

Field	Columns	Content	Example
1	01-01	Line number	1
2	03-07	Satellite number	25544
3	08-08	Classification	U
4	10-11	International Designator (Last two digits of launch year)	98
5	12-14	International Designator (Launch number of the year)	067
6	15-17	International Designator (piece of the launch)	A
7	19-20	Epoch Year (last two digits of year)	08
8	21-32	Epoch (day of the year and fractional portion of the day)	264.51782528
9	34-43	First Time Derivative of the Mean Motion (MM) divided by two	-.00002182
10	45-52	Second Time Derivative of MM divided by six (decimal point assumed)	00000-0
11	54-61	BSTAR drag term (decimal point assumed	-11606-4
12	63-63	The number 0 (originally this should have been ”Ephemeris type”)	0
13	65-68	Element set number. Increased every TLE is generated for this object	292
14	69-69	Checksum (modulo 10)	7

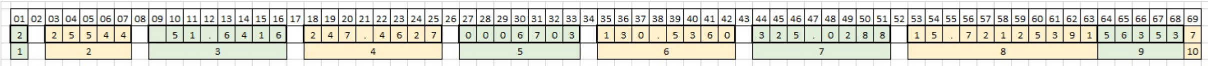


Figure 3.11: TLE Line 2

Table 3.4: Parameters of the second TLE line

Field	Columns	Content	Example
1	01-01	Line number	2
2	03-07	Satellite nimber	25544
3	09-16	Inclination (degrees)	51.6416
4	18-25	Right ascension of the ascending node (degrees)	247.4627
5	27-33	Eccentricity (decimal point assumed)	0006703
6	35-42	Argument of perigee (degrees)	130.5360
7	44-51	Mean Anomaly (degrees)	325.0288
8	53-63	Mean Motion (revolutions per day)	15.72125391
9	64-68	Revolution number at epoch (revolutions)	56353
10	69-69	Checksum (modulo 10)	7

All the parameters necessary for the development of the present study are highlighted at the previous tables. From the first line, the most important parameters are the 7th and 8th, Epoch Year and Epoch respectively. From them it can be extracted the date when the satellite was at certain point of its orbit.

For the Epoch Year, only the last two digits are written. This might change in the near future, when the exploration of the space celebrates a century because these numbers would be the same for the year 1957 and 2057 for example. Meanwhile, it works so the number 08 refers to 2008.

It is more complicated for the Epoch which indicates the day of the year because it also includes the fractional portion of the day. Taking the example of the ISS, the Epoch is 264.51782528, being 264 the number of the day. The procedure to extract the hours, minutes and seconds is showed next:

$$0,51782528 * 24 = 12,42780672hh$$

$$0,42780672 * 60 = 25,6684032min$$

$$0,6684032 * 60 = 40,104192seg$$

Regarding the second line of the TLE, the position of the satellite is obtained with the inclination, the right ascension of the ascending node, the argument of perigee and the mean anomaly. Some of the other parameters such as eccentricity would be important if the study analysed elliptical orbits too.

Initial position

As it has been said before, with the data provided by the TLE, the satellite initial position, latitude, ϕ_0, and longitude, λ_0, can be calculated at the given time epoch, t_0'. However, in order to make the calculations easier, it is required that the initial time epoch, the satellite be at the equator, i.e, ϕ_0=0.

Imposing the latitude means that the time epoch for it has to be calculated. To do so, the first step is to compute the Δt_0 it took the satellite to propagate form the equator to its current phase in the circular orbit.

$$\Delta t_0 = \frac{(\alpha + M_0)T_{\mathrm{s}}}{2\pi} \tag{3.6}$$

The corresponding initial time epoch is:

$$t_0 = t_0{}' - \Delta t_0 \tag{3.7}$$

The longitude of the subsatellite point at the equator can be obtained from the right ascension of the ascending node, Ω, parameter.

$$\lambda_0 = \Omega - GMST(t_0) \tag{3.8}$$

Greenwich Mean Sidereal Time (GMST) or also called Greenwich Mean Time (GMT) is a timekeeping system that astronomers use to locate celestial objects and it is based on the rotation of the Earth with respect to the fixed stars in the sky.

The Greenwich mean sidereal time at a concrete time t_0 is not simple to get. This is because the time can be standardized in multiple ways, so sometimes it is difficult understand how the time has been standardized but also is hard to find the conversions from one standard to another. There are various recognized time standards but all of them can be classified in two big groups. The first one is based on astronomy and is called *Astronomical Time*. The other, is based on the frequency of atomic oscillations and is called *Atomic Time* [21].

Astronomical Time is based on the repetition of astronomical events for setting frequency standards. In the past, people used a device that tells the time of day by the apparent position of the Sun in the sky, a sundial. This time standard was referred to as *Apparent Solar Time*. However, as the Earth revolves around the Sun in an elliptical orbit, and due to the inclination of the Earth's rotation axis to the orbital plane, the speed of the apparent motion of the Sun varies through the year and *Apparent Solar Time* is not a uniform standard. To solve that, a more uniform time standard called *Mean Solar Time* was characterized. It is defined by the uniform motion of a fictitious Sun on the celestial sphere that agrees with the averaged *Apparent Solar Time*. The *Universal Time* (UT0), is the *Mean Solar Time* at the Greenwich meridian, also known as *Greenwich Mean Time*.

By using better clocks, some discrepancies in UT0 were measured in different places of the Earth. It was discovered that the reason behind it was the the rotation axis of the Earth wobbles. After different measurements, the discrepancies were corrected defining a new time standard, UT1.

On the other hand, research on atomic frequency standards revealed that the frequency of atomic oscillations of caesium-133 atom was very stable. For that, a definition of the "second" was adopted based on such oscillations. In 1967, the Thirteenth General Conference of Weights and Measures adopted a resolution to replace the definition of the "second" based on astronomical observation by that of the atomic oscillation of caesium-133 atom. This decision was taken because the Earth's rotation is slowing down gradually and as a result, the "second" defined by astronomical time is slightly longer than that defined by atomic time.

Since the beginning of 1972, to reconcile the two time scales, the Coordinated Universal Time (UTC) was adopted. It primarily follows the atomic time, but with a leap second introduced when necessary such that the difference between UTC and the astronomical time is kept to less than 0.9 seconds.

After this brief summary of the time standards history, the conclusion obtained after an extensive research of this topic is that Greenwich mean sidereal time is the hour angle of the average position of the vernal equinox. In section 3.2 a detailed method to calculate the value of $GMST(t_0)$ is developed.

3.1.3 Satellite - Ground station geometry

Once the initial parameters of the satellite are obtained from the TLE, its geometry configuration with the ground station can be calculated. This subsection starts explaining the basic geometry concepts between both objects. Then, it goes through a more profound study of the geometry to finally obtain the visibility window time.

Slant range

The slant range, $d(t)$ is the distance between the satellite and the ground station. This range changes over time while the satellite is orbiting around the Earth.

In Figure 3.12 there is the Earth represented as if it was cut in half at the plane formed by the ground station (GS), the centre of the Earth (C) and the satellite (SAT). It can be seen that the slat range is also represented, together with the elevation angle, θ, and the central angle, γ, for this precise instant of time. The subsatellite point (SPP) is the point of the $SAT - C$ line which intersects with the Earth's surface .

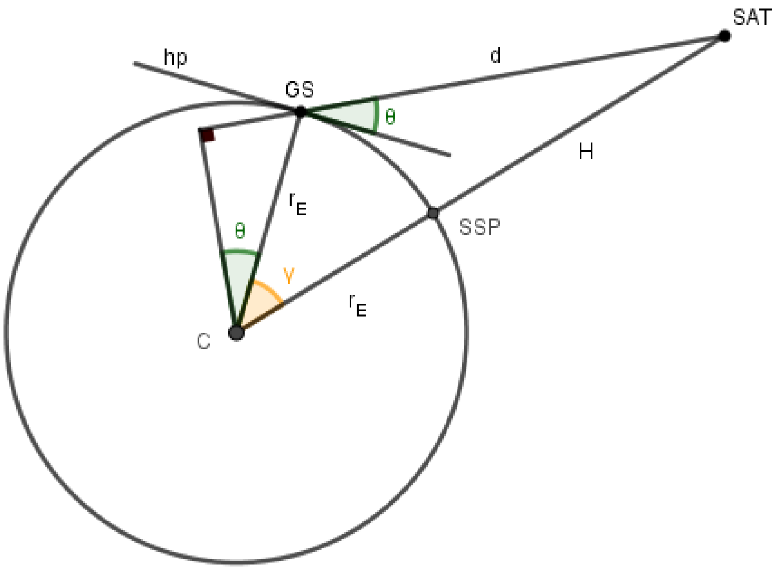


Figure 3.12: Ground station - Satellite basic geometry

Using basic notions of trigonometry, from the triangle $GS - SAT - C$ it can be extracted the equation to calculate the slat range. First, the cosine law is applied:

$$d^2 = r^2 + r_E^2 - 2rr_E cos(\gamma) \tag{3.9}$$

where:

$r \quad = H + r_E$

$H \quad =$ Distance between SSP and SAT

$r_E =$ Radius of the Earth

Therefore, the slant range equation, as a function of the central angle, is obtained.

$$d(t) = \sqrt{r^2 + r_E^2 - 2rr_E cos\gamma(t)} \tag{3.10}$$

The aforementioned central angle, $\gamma(t)$, is defined as the angle formed between the satellite and the ground station and it also changes over the time. Using again Figure 3.12 the following relation can be found.

$$rcos(\gamma(t) + \theta(t)) = r_E cos\theta(t) \tag{3.11}$$

Arranging it, an equation for the central angle as a function of time and elevation angle, is obtained.

$$\gamma(t) = acos\left(\frac{re}{r}cos\theta(t)\right) - \theta \tag{3.12}$$

It is important to highlight that when the elevation angle is minimum, the central angle at this exact instant is characteristic, γ_c.

$$\gamma_c = acos\left(\frac{re}{r}cos\theta_c\right) - \theta_c \tag{3.13}$$

Coverage zone

Not every satellite is visible for all the existing ground stations. Analyzing the satellite coverage region is indispensable in order to know with how many ground stations the satellite might have a communication link.

Most of the satellites orbit on inclined orbits. Because they have a certain inclination, not all the latitudes of the Earth are visible. Therefore, an inclined orbit is visible only within a latitude range on the Earth's surface. This latitude range, also named coverage zone, depends on the central angle apart from the inclination. The coverage region expressed in terms of degrees latitude is given by:

$$coverage = [max(-90, -i - \gamma_c), min(90, i + \gamma_c)] \tag{3.14}$$

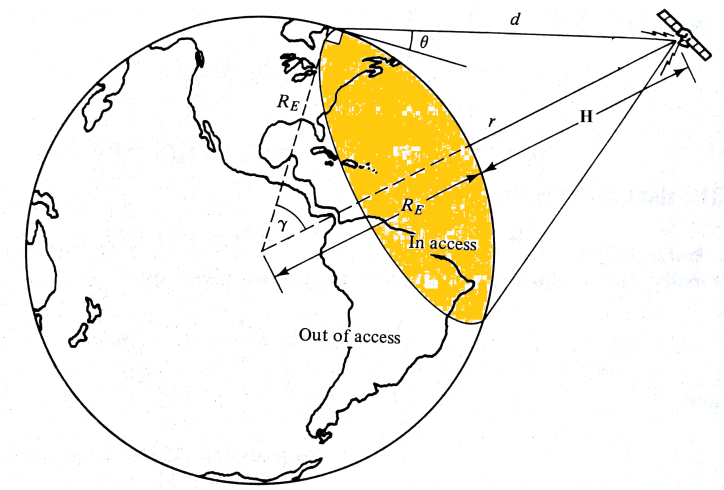


Figure 3.13: Earth coverage by a satellite at altitude H
[22]

If the ground station is not located in the coverage region, the satellite will never be visible to it, so the connection link will not be possible.

Once it has been verified that the ground station is in the coverage zone of the satellite, it also has to be checked if the satellite ground trace will intersect or not with the terminal's latitude.

To do so, the ground station latitude has to be within the range $[-i, i]$. If it does happen (Satellite 2 of Figure 3.14), the latitude of the intersection point will be the terminal's one. In case it does not occur (Satellite 1 of Figure 3.14), the location of the satellite closer to the terminal's latitude is when satellite's latitude is equal to the inclination of the orbit. In this case, for the calculation procedure, the intersection of the ground trace with the latitude's ground station will be computed either $+i$ or $-i$ degrees latitude depending on whether the terminal is located in the northern or the southern hemisphere, respectively.

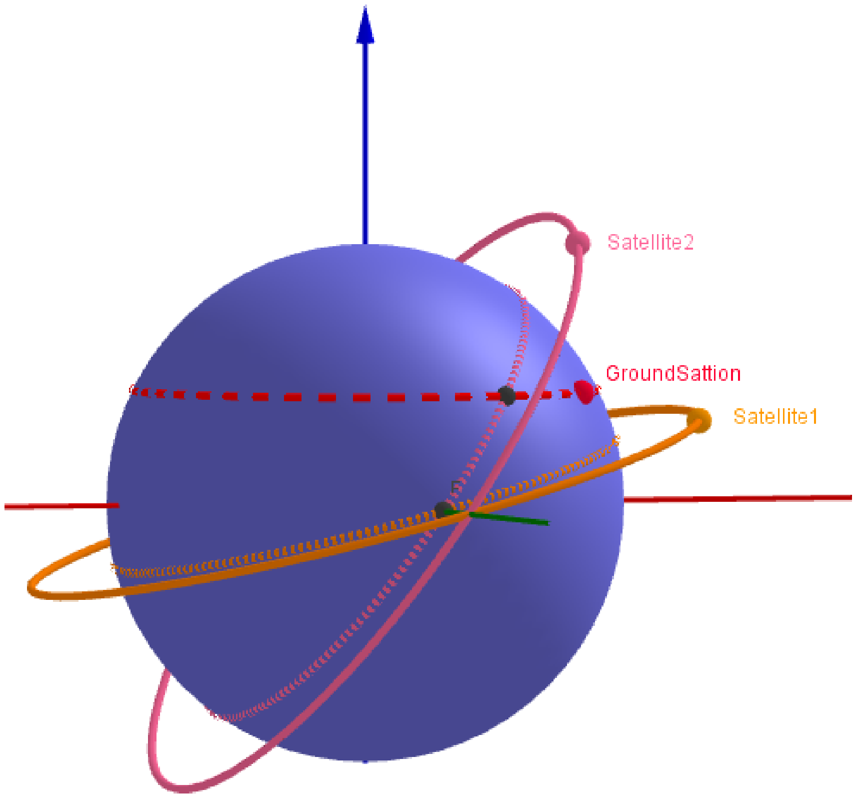


Figure 3.14: Intersection of the ground station latitude with different satellites

Ground track

Since the Earth is considered a perfect sphere, the spherical trigonometry has to be used in order to get the ground track that the satellite leaves on the Earth's surface.

The basic laws of the spherical trigonometry are the cosine and sine laws.

$$\frac{sinA}{sina} = \frac{sinB}{sinb} = \frac{sinC}{sinc} \quad (3.15)$$

$$cosc = cosa \cdot cosb + sina \cdot sinb \cdot cosC \quad (3.16)$$

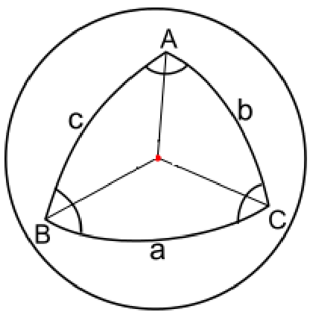


Figure 3.15: Spherica39:online [23]

The spherical geometry laws are applied to the spherical triangle $A - SPP - B$ shown in Figure 3.16. This triangle starts at the point A, at the time instant $t = t_0$, which has a longitude of $\lambda(t_0)$ and a latitude of 0°. As the satellite moves through its orbit, it draws a ground track on the Earth's surface that gets to SSP point. Using the ECI coordinate system, the distance covered is denoted $w_s t$. At SSP, longitude and latitude have changed so they are $\lambda_I(t)$ and $\phi(t)$, respectively. Finally, replacing the variables of the study to Equation 3.15 and Equation 3.16, the equations for the ground track as a function of time, are obtained.

$$\frac{sin(i)}{sin\phi(t)} = \frac{sin(90)}{sin(w_s t)} \tag{3.17}$$

$$\begin{aligned} cos(w_s t) = cos\lambda_I(t)cos(\phi(t)) + sin\lambda_I(t)sin(\phi(t))cos(90) = \\ = cos\lambda_I(t)cos(\phi(t)) \end{aligned} \tag{3.18}$$

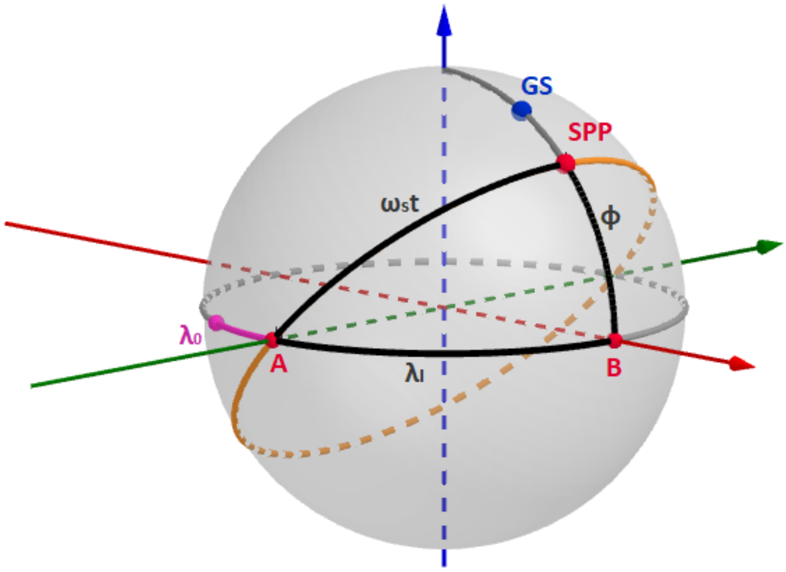


Figure 3.16: Spherical triangle A - SPP - B

To compute the subsatellite point's longitude in the ECF frame, it has to be compensated the Earth's rotation.

$$\lambda(t) = \lambda_I + \lambda_0 - w_E t \tag{3.19}$$

Therefore, the equations that describe ground track of a satellite which orbits around the Earth with a circular orbit are:

$$sin\phi(t) = sin(i)sin(w_s t) \tag{3.20}$$

$$cos(w_s t) = cos(\lambda(t) - \lambda_0 + w_E t)cos(\phi(t)) \tag{3.21}$$

Now, the basic equations needed to calculate the geometrical issue between the satellite and the ground station have been deduced. Knowing the TLE set parameters of the satellite, it can be checked if it is visible for a certain ground station. Also, the satellite's ground track can be calculated and represented on a world map if desired.

3.1.4 Visibility window duration

The analysis developed in order to obtain the equation which gives the time of the visibility windows is performed as seen from the terminal's location, i.e., in ECF coordinate frame. In an ECF frame, the satellite's orbit it is not a great-circle, condition necessary in order to apply trigonometric formulas for spherical triangles. However, as the period of visibility is really short compared with the whole orbit, the assumption that during the visibility windows the satellite traces a great circle, is made.

Therefore, a satellite with such an orbit that the ground station is at inside its latitude coverage region is taken in order to deduce the equation for the visibility windows duration.

When the satellite is visible for the ground station the geometrical configuration is shown in Figure 3.17. As it has been said, the analysis is done using ECF frame, but considering the ground track a great circle.

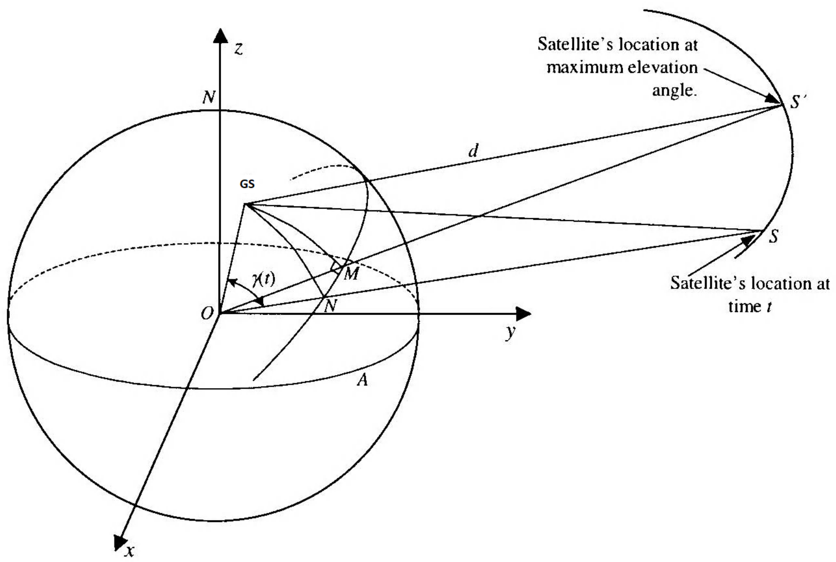


Figure 3.17: Satellite geometry duration visibility windor at location GS [24]

Therefore, the law of cosine is applied to the spherical triangle formed by the points $P - M - N$ (amplified in Figure 3.18). This triangle is formed as the satellite moves through its orbit. P denotes the position of the ground station and M is the subsatellite point where the elevation angle is maximum, at $t = t_m$. At M the central angle is has a precise value, $\gamma(t_m)$, and then changes as the satellites moves to the subsatellite point N, where the central angle is $\gamma(t)$. The side of the triangle $\psi(t) - \psi(t_m))$, denotes the angular distance measured on the surface of the Earth along the ground trace during the time t.

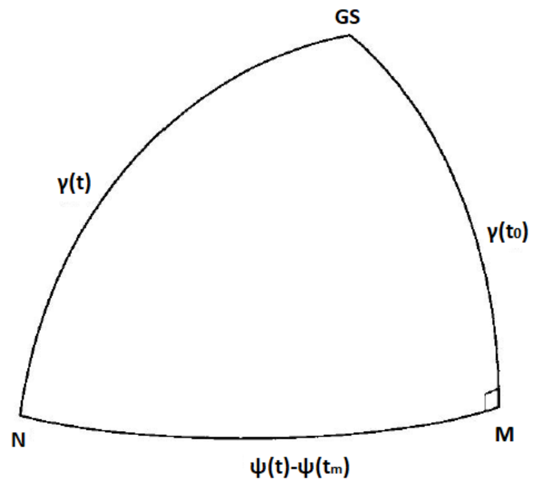


Figure 3.18: Spherical triangle GS - N - M
[24]

When the law of cosine is applied to the $P - M - N$ spherical triangle, the following expression is obtained:

$$\begin{aligned} cos\gamma(t) = cos(\psi(t) - \psi(t_m))cos\gamma(t_m) + sin(\psi(t) - \psi(t_m))sin\gamma(t_0)cos(90°) = \\ = cos(\psi(t) - \psi(t_m))cos\gamma(t_m) \end{aligned} \tag{3.22}$$

The time t_m, has been defined as the time when the elevation angle is maximum. Therefore, applying this condition to the Equation 3.12, an expression for the $\gamma(t_m)$ is obtained.

$$\gamma(t_m) = acos\left(\frac{re}{r}cos\theta_{max}\right) - \theta_{max} \tag{3.23}$$

On the other hand, t_c denote the time when the satellite just becomes visible to the ground station, i.e., the rise of the satellite. At this time, the angle of elevation to the satellite is θ_c, called also, the minimum elevation angle for visibility. So, using Equation 3.22 for the period between the rise of the satellite and the time it has the maximum elevation angle:

$$cos\gamma(t_c) = cos(\psi(t_c) - \psi(t_m))cos\gamma(t_m) \tag{3.24}$$

Arranging last equation, the angular distance drawn by the ground trace over the Earth's surface is:

$$\psi(t_c) - \psi(t_m) = \omega_F(\phi)(t_c - t_m) = cos^{-1}\left(\frac{cos\gamma(t_c)}{cos\gamma(t_m)}\right) \tag{3.25}$$

The angular velocity of the satellite in the ECF frame is denoted by w_F. Because it is measured in this frame, it depends on the latitude due to the Earth's rotation. By contrary, the satellite's angular velocity in the ECI frame, w_I, is constant. If in the ECF frame the satellite's velocity could be considered constant, the calculations would be much simple.

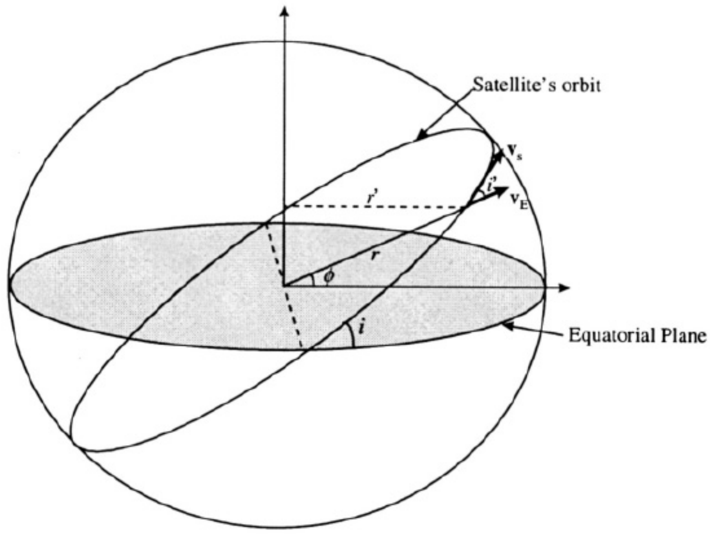


Figure 3.19: Satellite's tangential velocity in the ECI frame
[24]

Therefore, using Figure 3.19, i denotes the inclination of the orbit. Moreover, the corresponding tangential velocity of the satellite in the ECI frame is denoted by $\overrightarrow{v_s}$. Vector $\overrightarrow{v_E}(\phi)$ denotes the velocity of the subsatellite point due to Earth's rotation at latitude ϕ degrees projected up to the satellite's altitude and vector $\overrightarrow{v_F}$ denotes the corresponding velocity of the satellite in the ECF frame and its given by the resultant:

$$\overrightarrow{v_F}(\phi) = \overrightarrow{v_s} - \overrightarrow{v_E}(\phi) \tag{3.26}$$

Using the triangle law of cosines and the relation $cosi' = cosi/cos\phi$

$$\mid v_F(\phi) \mid^2 = \mid v_s \mid^2 + r^2\omega_E^2 cos^2(\phi) - 2r\omega_E \mid v_s \mid cosi \tag{3.27}$$

It has been proved by [25] that for low to medium orbit altitudes, the magnitude of the tangential velocity of the satellites in the ECF shows small variation and can be approximated by a constant.

$$| v_F | \approx | v_s | - r\omega_E cosi \tag{3.28}$$

$$\omega_F \approx \omega_s - \omega_E cosi \tag{3.29}$$

The total period of the visibility windows is $2 \mid t_v - t_0 \mid$ since it is considered that the moment of the maximum elevation angle occurs at half time of it. Therefore, using the mentioned angular velocity approximation, the total visibility window duration of the satellite at the ground station, $\tau(\theta_{max})$, is:

$$\tau(\theta_{max}) \approx \frac{2}{\omega_s - \omega_E cosi} \cdot cos^{-1}\left(\frac{cos(cos^{-1}\left(\frac{r_E}{r} cos\theta_c\right) - \theta_c)}{cos(cos^{-1}\left(\frac{r_E}{r} cos\theta_{max}\right) - \theta_{max})}\right) \tag{3.30}$$

As the altitude of the satellite is increased, the approximation of the angular velocity stars deferring from being constant and the results obtained do not correspond with the reality. With the results extracted from the code chapter 5, it will be analysed until what altitude it can be considered that this assumption is correct.

Maximum elevation angle

From Equation 3.30 the only parameter that is still pending to calculate is the maximum elevation angle. In order to do so, firstly,the rise and set times of the satellite are calculated.

Equation 3.15 is used to calculate when the ground trace of the satellite intersects with the ground station's latitude. This equation has two solutions; one for the rise time and another for the set time.

$$\omega_{\mathrm{s}} t_{\mathrm{a}} = sin^{-1}\left(\frac{sin(\phi_{\mathrm{gs}})}{sin(i)}\right) = \eta \tag{3.31}$$

Arcsine function only returns values in the range $[-\pi/2, +\pi/2]$, therefore the two solutions given by $sin^{-1}(\theta)$ are η and $\pi - \eta)$, both comprised in the range $[0, 2\pi]$.

From the two solutions, the longitude is also calculated with Equation 3.16. The intersection of the ground trace which is closer to the ground station's longitude is the one chosen, obtaining finally the vector x_a composed by $[\lambda_a \ \phi_a \ r]$

$$\lambda_{\mathrm{a}} = cos^{-1}\left(\frac{cos(\omega_{\mathrm{s}} t_{\mathrm{m}})}{sin(\phi_{\mathrm{gs}})}\right) - \omega_{\mathrm{E}} t_{\mathrm{a}} + \lambda_0 \tag{3.32}$$

With the solution chosen, another vector of the ground track is calculated at the time

$t_b = t_a + \Delta t$. With equations 3.20 and 3.21 the longitude and latitude of the vector x_b are calculated resulting $[\lambda_b\ \phi_b\ r]$. With both vectors, x_a and x_b form a plane that results a great circle approximation at the Earth surface. The unitary vector perpendicular to it n is obtained.

$$n = \frac{x_a \times x_b}{\mid x_a \times x_b \mid} \tag{3.33}$$

The perpendicular distance of the ground station from the plane is $p = x_t \cdot n$ and the central angle between the terminal and the point on the great circle arc approximation closest to the terminal is given by

$$\gamma = sin^{-1}\left(\frac{p}{r_E}\right) \tag{3.34}$$

Therefore, the location of the subsatellite point at the time instant of the maximum elevation angle observation, denoted by x_m:

$$x_m = \frac{1}{cos(\gamma)}(x_t - pn) = [\lambda_m\ \ \phi_m\ \ r] \tag{3.35}$$

Note that $[\lambda_m, \phi_m, r]$ is an approximation of the subsatellite point at the instant of maximum elevation angle observation based on the great circle arc approximation of the ground trace. This point might not lie on the actual ground trace, i.e., $[\lambda_m, \phi_m, r]$ not satisfy equations 3.19, 3.31 and 3.32 simultaneously.

An iterative process begins here in order to approximate as much as possible the great circle to the real ground trace. With the ϕ_m obtained, the time t_m is calculated again using from Equation 3.31. The following instant is computed and for both times, their coordinates are obtained. The perpendicular unitary vector from the plane created by both vectors is calculated again and also the minimum distance between the plane and the ground station. A new vector x_m is obtained and the procedure is repeated since the difference between the previous x_m and the actual one, hardly differ.

It is necessary to recheck if the satellite is visible to the terminal in order to ensure the correct procedure of calculus. The condition on γ for visibility is $-\gamma_c \leq \gamma \leq \gamma_c$. If this condition is not met, the satellite is not visible during the current rotation.
If the condition is met, at the point of closest approach the ground trace to the ground station, the central angle between the terminal and the ground trace is minimum. At this instant, the elevation angle from the terminal to the satellite is maximum. The maximum

elevation angle $\theta_{\max}$ is given by trigonometry using Figure 3.20

$$\varepsilon = acos\frac{p}{r_E} \tag{3.36}$$

$$d^2 = r_E^2 + (H + r_E)^2 - 2r_E(H + r_E)cos\gamma \tag{3.37}$$

$$A = acos\left(\frac{(r_E + H)^2 - (r_E^2 + d^2)}{2r_E^2 d^2}\right) \tag{3.38}$$

$$\theta = A - \varepsilon - \gamma \tag{3.39}$$

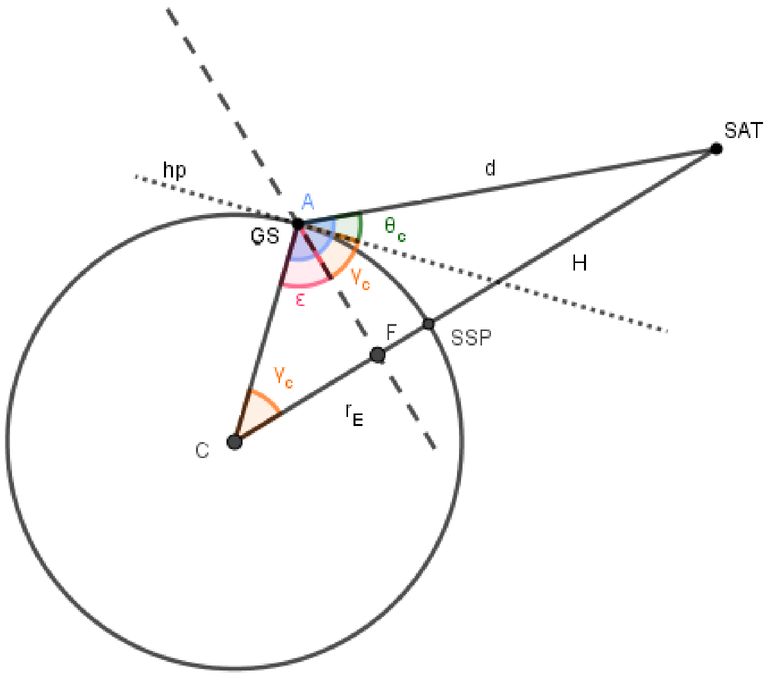


Figure 3.20: Ground station - Satellite geometry

Finally, the maximum elevation angle has been obtained, therefore the visibility window time can be calculated.

Doppler Characterization

It is important to point out that for satellite communications in LEO, ground stations observe a significant Doppler effect, which has to be estimated and compensated to enable reliable communications[25].

The Doppler effect is the change in frequency or wavelength of a wave for an observer who is moving relative to the wave source. For the present study, the movement of the satellite produces a Doppler shift of the downlink transmissions as seen by the ground station, and a corresponding shift for the uplink signals as seen by the satellite. This Doppler shift has to be estimated and compensated to enable reliable communication.

The change in frequency due to the movement of the satellite is given by:

$$\Delta f = \frac{\Delta v f}{c} \tag{3.40}$$

where:
Δv = relative radial velocity between the LEO satellite and the ground station[m/s]
f = Carrier frequency [Hz]
c = Speed of light [m/s]

The Doppler shift is captured in terms of normalized Doppler shift:

$$\frac{\Delta f}{f} = \frac{\Delta v f}{cf} = \frac{\Delta v}{c} \tag{3.41}$$

To solve the Equation 3.41, the relative radial velocity between the LEO satellite and the ground station has to be calculated.

Considering a ECF coordinate system, the ground station, which is located at the Earth's surface, does not move. By contrast, the satellite does move on its orbit. In subsection 3.1.3, the equation which describe the change over time of the slant range has been deduced as a function of the central angle, Equation 3.10. Therefore, the radial relative velocity between the ground station and the satellite is obtained by the derivative of it.

$$\dot{d}(t) = \frac{1}{2}[r_E^2 + r^2 - 2r_E cos\gamma(t)]^{\frac{1}{2}} \cdot \frac{dcos\gamma(t)}{dt}(-2r_E r) \tag{3.42}$$

The term $\frac{dcos\gamma(t)}{dt}$ can be obtained from the derivative of the Equation 3.12. Substituted in Equation 3.42, gives the relative radial velocity between the satellite and the ground station as a function of time and angular velocity of the satellite, Equation 3.44

$$\frac{dcos\gamma(t)}{dt} = -sin(\psi(t) - \psi(t_m)\dot{\psi} cos\gamma(t_m) \tag{3.43}$$

$$\begin{aligned} \dot{d}(t) &= \frac{1}{2}\frac{1}{\sqrt{r_E^2 + r^2 - r_E r cos\gamma(t)}}(2r_E r)(-sin(\psi(t) - \psi(t_m))\dot{\psi} cos\gamma(t_m)) = \\ &= \frac{r_E r sin(\psi - \psi(t_m)) cos\gamma(t_m) \cdot \dot{\psi}(t)}{\sqrt{r_E^2 + r^2 - 2r_E r cos(\psi - \psi(t_m)) cos(\psi(t) - \psi(t_m)) cos\gamma(t_m)}} \end{aligned} \tag{3.44}$$

Now, $\dot{\psi}$ is the angular velocity of the satellite in the ECF frame; hence, $\dot{\psi} = \omega_F(t)$.

So, finally, introducing Equation 3.23 to the Equation 3.44 and noting that normalized Doppler ($\Delta f/f$) is given by $-\dot{d}(t))/c$, next expression is obtained.

$$\frac{\Delta f}{f} = -\frac{1}{c}\frac{r_E r sin(\psi - \psi(t_m))cos(cos^{-1}(\frac{r_E}{r}cos\theta_{\max}) - \theta_{\max}) \cdot \omega_F(t)}{\sqrt{r_E^2 + r^2 - 2r_E r cos(\psi - \psi(t_m))cos(cos^{-1}(\frac{r_E}{r}cos\theta_{\max}) - \theta_{\max})}} \tag{3.45}$$

Consequently, the normalized Doppler is a function of the maximum elevation angle and the angular velocity of the satellite in ECF frame.

A positive Doppler frequency shift implies that the satellite is rising above the horizon plane, whereas a negative Doppler frequency shift implies that the satellite is descending towards the horizon plane. The zero Doppler instant is the time during the visibility window at which the elevation angle form the ground station to the satellite is at its maximum value and the satellite is at its closest approach to the ground station.

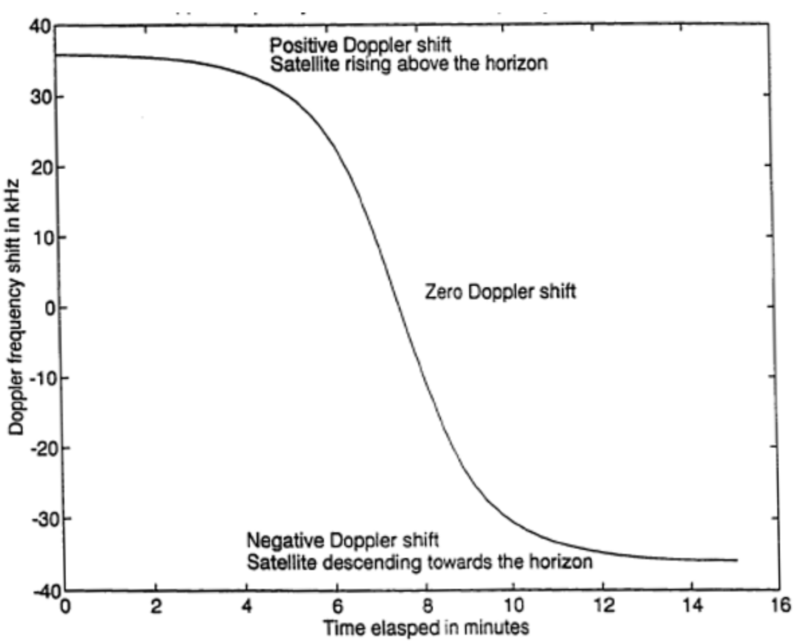


Figure 3.21: Doppler frequency shift of a Iridium satellite with a carrier frequency $fc = 1.618GHz$

[26]

As it can be seen in Figure 3.21, for LEO satellites the Doppler frequency at the ground station exhibits a well-behaved variation with time. The parametrization of this curve is Equation 3.45. As the maximum elevation angle descends, the curve is comprised in a shorter interval of time and has a tendency to seem more a straight line [25].

3.2 Predicting the visibility window: Matlab code

In order to put into practice all the theory developed, a Matlab code has been written. This code determine the visibility window time for a circular LEO and VLEO satellites from a ground station on the Earth's surface.

Therefore, in the present subsection, an overview of the code developed is shown, highlighting all the difficulties encountered while writing it. The complete code can be found in *Document 2. ANNEXES*.

3.2.1 Constants and initial parameters

All the necessary physical and geometrical constants are defined at the beginning of the code. Here, the user must specify the minimum elevation angle of visibility that he or she wants to consider for all the study. In addition, the user has to enter the data generation rate of the satellite, ($mbps_{sat}$). This parameter depends on factors such as the type of information the satellite payload is recollecting (optical, radar, etc.), and how fast it is capable to take it.

```
%% Constants
re=6371000;              % (Km)     Earth's radius
we=(2*pi)/86400;         % (rad/s)  Earth's rotation velocity
theta0=0;                % (deg)      Minimum angle of elevation
G=6.673e-11;             % (N*m^2/kg^2) Gravitional constant
M_Earth= 5.977e24;       % (kg)     Earth's Mass
mbps_sat=7;              % (mbps)   Satellite data generation
deltat=20;                    % (s)     Step time
```

Regarding the initial parameters, this code has been conceived to be able to work with multiples satellites and grounds stations at the same time. For this reason, the amount of data which define the satellites's orbit and the position of the ground stations is quite a lot.

Matlab software has a command that asks the user for data (*x=input (' ')*). It is used later during the code, but in order to recollect the information of the satellites and grounds stations, an Excel document has been chosen because it is faster for the user to enter all the parameters.

	A	B	C	D	E	F	G	H	I
1	Satellite	Inclination (°)	Right ascension of the asending node (°)	Argument of the perigee (°)	Mean anomaly(°)	Altitude (m)	Epoch year	Epoch	nº orbits
2	1	51,64	232,1171	226,7446	133,3436	404000	2017	345,957	5
3	2								
4	3								

Figure 3.22: Excel table to collect satellites information

	A	B	C
1	Ground station	Longitude (°)	Latitude (°)
2	1	2,009	41,563
3	2		
4	3		

Figure 3.23: Excel table to collect ground station information

As it can be seen, the Excel document, which is named *'Data.xlsx'*, only asks for the ground station longitude and latitude angles, i.e., its position. Regarding satellites only the necessary TLE parameters have to be entered. As many satellites and ground stations as the user wants can be added only making bigger the above tables.

Therefore, when the Matlab code is compiling, it enters to *'Data.xlsx'* and converts every column in a distinct vector.

```
GroundStation=xlsread('Data.xlsx');
Satellite=xlsread('Data.xlsx',2);

N=1;                          % Number of satellites
phi_s=zeros(1,N);
in=Satellite(:,2);            % Satellite orbit inclination (deg)
Omega=Satellite(:,3);         % R.a. of the ascending node (deg)
alpha=Satellite(:,4);         % Argument of perigee (deg)
Mo=Satellite(:,5);            % Mean anormaly at time epoch(deg)
H=Satellite(:,6);             % Satellite altitude (m)
year=Satellite(:,7);          % Field 7 of the Two Line element set
day_frac=Satellite(:,8);      % Field 8 of the Two Line element set

M=1;                                % Number of ground stations
phi_gs=GroundStation(:,3).';         % Latitude groud station (deg)
lambda_gs=GroundStation(:,2).';      % Longitude groud station (deg)
```

It is important to mention that M denotes the number of ground station and N the number of satellites. For example, if *'Data.xlsx'* has information for 3 ground stations and 2 satellites, M and N would be 3 and 2, respectively.

3.2.2 Calculation of the visibility window duration

The code starts analysing the first ground station ($M = 1$). It computes where the terminal is located whereas the north or south hemisphere.

Once the program knows where the ground station is, it takes the parameters of the first satellite ($N = 1$). With them, its longitude and latitude, at the indicated time epoch, can be calculated. However, as said in subsection 3.1.2, in order to have a faster algorithm, it is required that the initial time epoch be when the satellite is at the equator plane. This means that time epoch which is extracted form the TLE set is not the one used.

The procedure to calculate the initial time has already been exposed in subsection 3.1.2, using the Equation 3.7. Regarding the initial longitude, which is given by the Equation 3.8, first the term $GMST(t_0)$ has to be calculated. At subsection 3.1.2, this term has only been defined but in order to write the code, it is essential to know how it is calculated.

First of all, it has to be explained what Julian Date (JD) is. Julian Date is simply a way to name moments, that is simpler than the customary date-plus-time system. Normally, to specify a date and time requires six different numbers - year, month, day, hour, minute, and seconds - and comparing two dates takes a terrible amount of math, with some extra complications as there are months with 30 days, others with 31 days, for example. Julian Dates assign a simple floating-point number to each date-plus-time on the calendar, and are much easier to do math with.

JD counts the number of days since Jan 1, 4713 BC 12:00 UTC in the Julian calendar. Sometimes, that is a bit hard to work with since it is so long ago. For that, Reduced Julian Date (RJD) exists. It is another official standard, related by the Equation 3.47. This counts the number of days since 2000 Jan. 1 12h UT1. [27], [28]

To calculate JD, the following equation is used:

$$JD = 2451544,5 + 365Y + 0,25Y - 0,01Y + 0,0025Y + A + D + \frac{1}{24}H + \frac{1}{1441}M + \frac{1}{86400}S \tag{3.46}$$

where:

Y = Year
A = is a correction factor. If the year is a leap year, it is -1, otherwise it is zero.
D = Days
H = Hours
M = Minutes
S = Seconds

$$RJD = JD - 2451545.0 \tag{3.47}$$

Once the RJD has been calculated, next equation is applied in order to calculate the applied:

$$GMST[s] = 24110.54841 + 8640184.812866T + 0.093104T^2 - 0.0000062T^3 \tag{3.48}$$

where:
T = julian centuries from 2000 Jan. 1 12h UT1, $T = RJD/36525$

The last step is to transform seconds to degrees, being the conversion:

$$1second = 0.0002777778degrees \tag{3.49}$$

The part of the code which implies this calculus is presented next:

```
day(n)=fix(day_frac(n));                            % Day [day]
hour_frac(n)=(day_frac(n)-day(n))*24;
hour(n)=fix(hour_frac(n));                          % Hour [hour]
min_frac(n)=(hour_frac(n)-hour(n))*60;
minu(n)=fix(min_frac(n));                           % Minute [min]
seg(n)=(min_frac(n)-minu(n))*60;                    % segonds [s]
ly = leapyear(year(n));                             % Leap year
     if ly==1
          A=-1;
     else
          A=0;
     end

JD(n)=2451544.5+365*year(n)+0.25*year(n)-0.01*year(n)+
     0.0025*year(n)-A+day(n)+(1/24)*hour(n)+(1/1440)*minu(n)
d(n)= JD(n) - 2451545.0;
T(n) = d(n) / 36525;
```

```
GMST_sec(n)=24110.54841+8640184.812866*T(n)+0.093104*T(n).^2
      -0.000006210*T(n).^3;
 GMST(n)=0.0002777778* GMST_sec(n)            % seconds to degrees

lambda_s(n)=Omega(n)-GMST(n);                          % (deg)
```

Matlab offers a predetermined function, $jd = juliandate([y, mo, d, h, mi, s])$ which can also calculate the Julian Date. It has not been used because the time epoch is provided by the TLE in such a way that it is complicated to extract the month and the day of the month, as the function asks. Because of that, the resource previously mentioned which is easier to implement, has been used, Equation 3.46.

All the previous steps are initial calculations. From now on, the code starts the analysis of the geometry configuration between the satellite and the ground station.

First of all, depending on where the ground station is situated, it has to be checked if it is placed in the latitude coverage zone of the satellite or not. In case it is not, the code returns a message which says the ground station will never be visible to the satellite. That way the user is aware that communication between the satellite and the terminal is impossible.

```
%% Determinate if the terminal is within the satellite's coverage
%latitudes

cov=[max(-90,-in(n)-gammac(n)) min(90,in(n)+gammac(n))];
      if phi_gs(m)>max(-90,-in(n)-gammac(n)) &&
         phi_gs(m)<min(90,in(n)+gammac(n))
           cov_gs=1;
      else
           continue
           disp (['Ground Station ',num2str(m), ' is not in the
           coverage region of the satellite. There is no possible
           communication between them. '])
           disp(' ');
      end
```

In case the ground station is in the latitude coverage zone of the satellite, next step is to check if the intersection between ground trace of the satellite with the ground station's latitude occurs. If there were a case which this circumstance did not occur, as the theory says, the inclination of the satellite is computed as the latitude intersection.

On the other hand, as said, the code has been implemented to be able to work with as many ground stations and satellites the user wants. Also, the number of orbits the satellite performs, can be configured by the user. For that reason, K is the numerical variable the user introduces to the program in order to determine how many orbits the code will reproduce. So, the next step in the development of the code is to calculate the ground trace as a function of time for all the orbits the satellite does.

To do so, the time is divided in intervals of time which last *delat*. A loop command is implemented to calculate the longitude ($lambda_{ss}(j,1)$) and latitude ($phi_{ss}(j,1)$) of the satellite for every interval of time as well as the information it recompile ($mbps_{ss}(j,1)$). For example, if $delat = 10$ and $mbps_{sat} = 7$, the code calculates the position of satellite at times $0s$ and $10s$, and the information captured during the interval is $mbps_{ss} = 7 \cdot 10 = 20$ mb.

```
%% For every orbit
t_in(1,1)=0;                % Initial time for the orbit k
t_fin(1,1)=Ts(n);           % Final time for the orbit k
t=t_in:deltat:Ts(n)*K;      % Vector of initial times of the
                              % intervals
time=0;                     % Initial time for the  whole simulation
tfin=K*Ts(n);               % Final time for the whole simulation

% Track of the subsatellite point
while time<=tfin
phi_ss(j,1)=asind(sind(in(n))*sind((ws(n)*(360/(2*pi)))*time));
     lambda_l(j,1)=acosd((cosd((ws(n)*(360/(2*pi)))*time))/
                   (cosd(phi_ss(j,1))));
lambda_ss(j,1)=lambda_o(1)-we*(360/(2*pi))*time;

mbps_ss(j,1)=mbps_sat*deltat;
time=time+deltat;
```

As it can be seen, to calculate the longitude for every step time, the terms, initial longitude ($lambda_0$) and Earth's rotation ($we*(360/(2*pi))*time$) have been taken into account. Without the last term, the initial longitude for each orbit would be the same. Figure 3.24 presents an orbit of a satellite which the rotation of the Earth has not been considered, whereas in Figure 3.25 it has been taken into account. The strange circular shape it has, is explained afterwards.

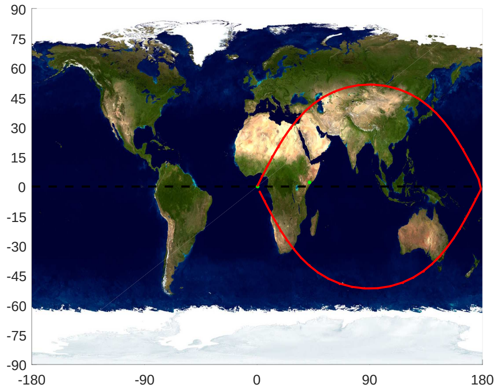


Figure 3.24: Ground track without Earth's rotation

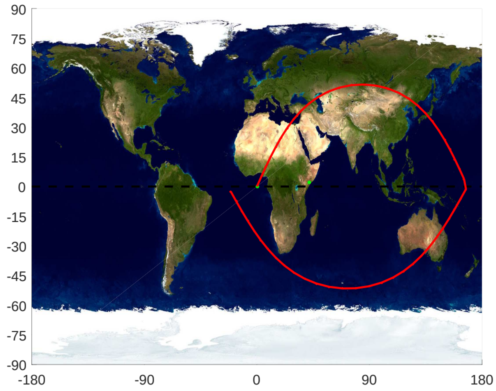


Figure 3.25: Ground track with Earth's rotation

The longitude given by the code until now, gives values between 0° to 180°. This means, for example, that while the latitude is positive, the corresponding longitude values goes from 0° to 180°, and when the latitude is negative, the longitude goes from 180° to 0°. As a result, the strange circular shape of the previous figures is obtained.

As the Earth is considered a sphere, the longitude can be defined within the interval $[0°, 360°]$ or $[-180°, 180°]$. In order to represent the obits in the habitual world map, it is convenient to have the longitude within the interval $[-180°, 180°]$. At the *Document 2. ANNEXES* a correction on the longitude previously calculated can be observed. Next, a brief summary of this correction is exposed.

The correction begins creating a vector which stores the difference of longitude between the current *time* with the previous one, *time-deltat*.

```
subtraction(j,1)=abs(lambda_l(j,1)-lambda_l(j-1,1));
```

When the code detects that for the same orbit, the longitude is being repeated, it recalculates the longitude taking the previous one and adding to it the corresponding difference stored in the vector *subtraction*. With this first step of the correction, the longitude interval of representation has changed from [0°, 180°] to [0°, 360°], as it can be seen in the following Figure 3.26. The Earth surface has not been represented in this case.

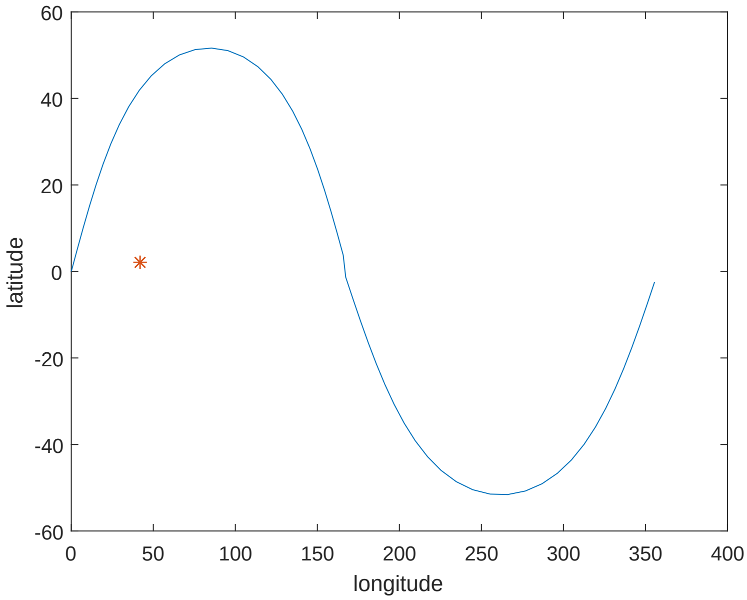


Figure 3.26: Ground track after the first step of the longitude correction

The second and last step of this correction is to put all the longitudes between $[180°, 360°]$ and $[-180°, 0°]$. In that way, the representation of the orbit is fitted in the interval $[-180°, 180°]$.

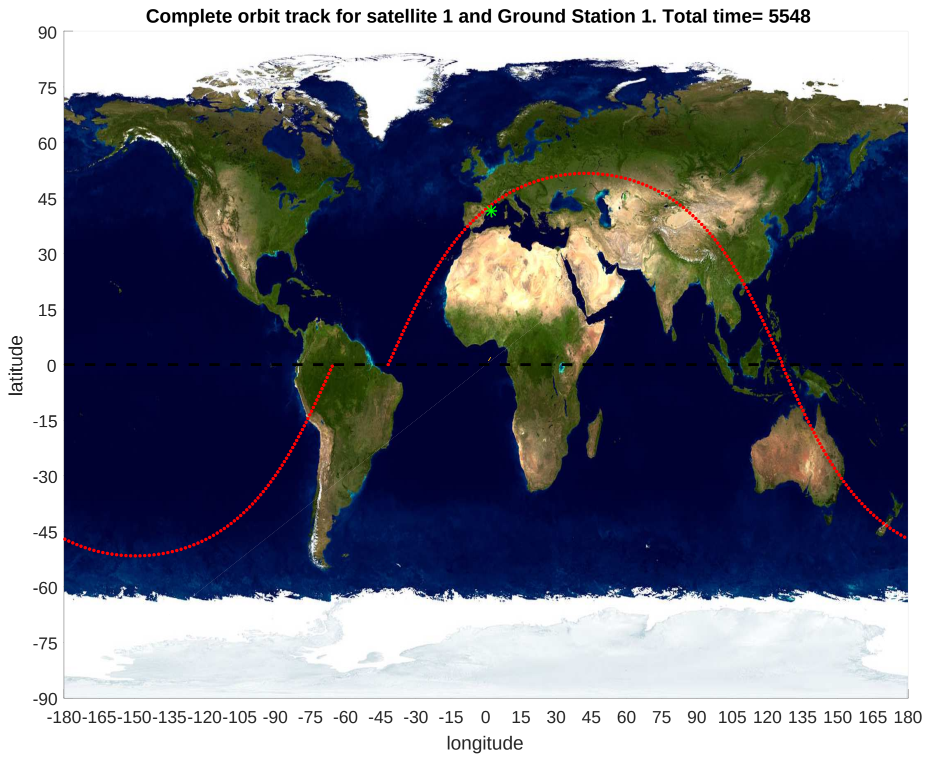


Figure 3.27: Ground track for one satellite

This correction has to be extrapolated to all the orbits the code is simulating. So far, it has been explained for only one orbit, but when there are various of them, after the first step of the correction, the longitude extends within the interval $[0°,\, k \cdot 360°]$.

Until now, the code works with one vector for the longitude and another for the corresponding latitudes of all the orbits Figure 3.28a. After the correction of the longitude, the program defines two matrix in which the columns are the orbits and the rows the longitude and the latitude, respectively Figure 3.28b. This step is important because the aim of the code is to predict the visibility window for each orbit for separated. The sum of all the visibility window times is going to provide the total time of visualization.

	lambda_ss	phi_ss
	....	....
ORBIT 1	158,2126	-7,5199
	158,1293	-6,5072
	158,0460	-5,4931
	157,9626	-4,4779
	157,8793	-3,4619
	157,7960	-2,4451
	157,7126	-1,4279
	157,6293	-0,4104
ORBIT 2	157,5460	0,6072
	157,4626	1,6247
	157,3793	2,6418
	157,2960	3,6584
	157,2126	4,6743
	157,1293	5,6893
	157,0460	6,7031
	....	....

(a) Data in vectors

LONGITUDE (lambda_ssorbit)	
ORBIT1	ORBIT2
....	157,5460
....	157,4626
158,2126	157,3793
158,1293	157,2960
158,0460	157,2126
157,9626	157,1293
157,8793	157,0460
157,7960	....
157,7126	....
157,6293	....

LATITUDE (phi_ssorbit)	
ORBIT1	ORBIT2
....	0,6072
....	1,6247
-7,5199	2,6418
-6,5072	3,6584
-5,4931	4,6743
-4,4779	5,6893
-3,4619	6,7031
-2,4451	....
-1,4279	....
-0,4104	....

(b) Data in matrix

Figure 3.28: Data in vectors moved to matrix

Following the theory previously developed, the code continues by calculating the rise and set time of the satellite and also in which position both occur. The event which has the longitude closer to the ground station's longitude, is the one chosen to start the iteration. The objective is to find the position of the satellite which has the maximum elevation angle.

To ensure the code work properly, the coordinates of the analysed points need to be changed from spheric to cartesian coordinates. As a reminder, the equations from the conversion are written next.

$$x = r_E sin(90 - \phi)cos(\lambda) \quad (3.50)$$

$$y = r_E sin(90 - \phi)sin(\lambda) \quad (3.51)$$

$$z = r_E cos(90 - \phi) \quad (3.52)$$

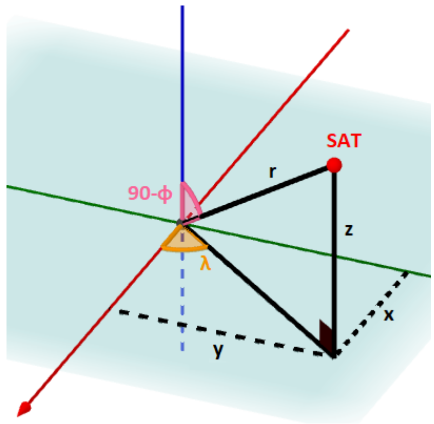


Figure 3.29: Spherical-Cartesian coordinates

Now, the rise or the set position of the satellite has been taken to start the iteration. The vector which describes this position is called x_a. Immediately after, the code enters to a loop section where first, it computes another position of the satellite needed in order to perform the iteration, x_b. This new position is the result of calculating the longitude and latitude at time $tb = ta + \Delta t$.

Therefore, this two vectors create the plane where the orbit of the satellite is located. The unit vector perpendicular to the plane is computed, and with it, the distance between the plane and the position of the ground station, p, is calculated. Also, the central angle can be obtained applying Equation 3.34. Finally, taking the central angle and the subtraction of the vector, np, from the ground station position vector, x_t, the first approximation of the satellite Cartesian coordinates (x_m) for which the elevation angle is maximum, are obtained.

```
xa_cartesianes=[x_a y_a z_a];
xb_cartesianes=[x_b y_b z_b];

nxs=cross(xa_cartesianes,xb_cartesianes);
nxi=sqrt(nxs(1,1)^2+nxs(1,2)^2+nxs(1,3)^2);
nx=nxs/nxi;

p=dot(xt_cartesianes,nx);
gamma(n,k)=asind(p/re);
xm_cartesianes=(1/cosd(gamma(n,k)))*(xt_cartesianes-p*nx);
```

From this position, a new position is computed and the procedure is repeated since there is no variation in the central angle. When this occurs, it means the code has reached the position of the satellite where the elevation angle is maximum. In order to calculate it, equations 3.36, 3.37, 3.38 and 3.39 refered to Figure 3.20 are used.

Therefore, since now, the code has evaluated all the geometry configuration of the satellite. The position, longitude and latitude, as a function of time have been obtained, together with the maximum elevation angle for each orbit the satellite performs. In order to obtain the visibility windows time Equation 3.30 is applied.

Leaving aside the geometrical issues which has already been solved, now the code begins the part where analyses how much data the satellite can download to the ground station. Obviously, it depends directly to the visibility window time. Actually, it is the reason it has been previously calculated. The amount of information gathered by the satellite during all the simulation is stored in a variable named *data_collected* [*mb*]. Separately, another variable contains the total visibility time of the satellite, *visibility_total s*. Like this, when *data_collected* is divided by *visibility_total* the data rate of the downlink, *data_rate* [*mbps*] can be obtained.

Taking the visibility window for each orbit, it can be computed the data transferred to the ground station for this period of time. If the visibility windows is longer, more data can be downloaded.

To end the code, the representation of all the information generated is implemented. Firstly, for the ground track representation, the user can choose between seeing the final result directly of seeing how the orbits are developed through time. The red line represents the ground track of the satellite and green points are the ground stations. Secondly, a graphic which represents the memory status of the satellite is also obtained. An example of the graphics generated by the code are presented next.

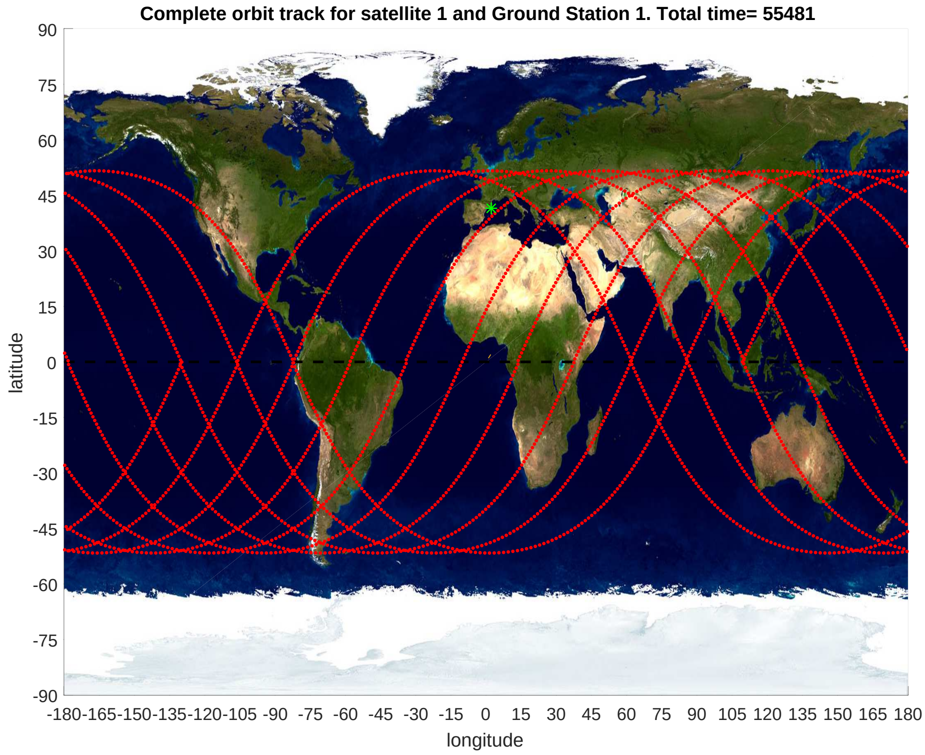


Figure 3.30: Satellite ground track, 10 orbits. Ground station situated in Catalunya

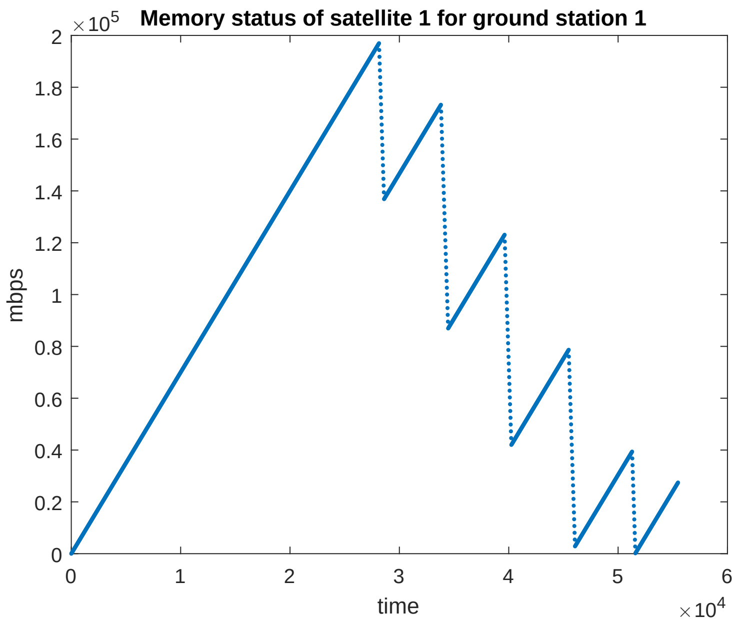


Figure 3.31: Satellite memory status while simulating the scenario of Figure 3.30

To graphically summarize the explained program, a code flow diagram has been done using *draw.io*, which is a free online diagram software for making flowcharts, process diagrams, org charts, UML, ER and network diagrams.

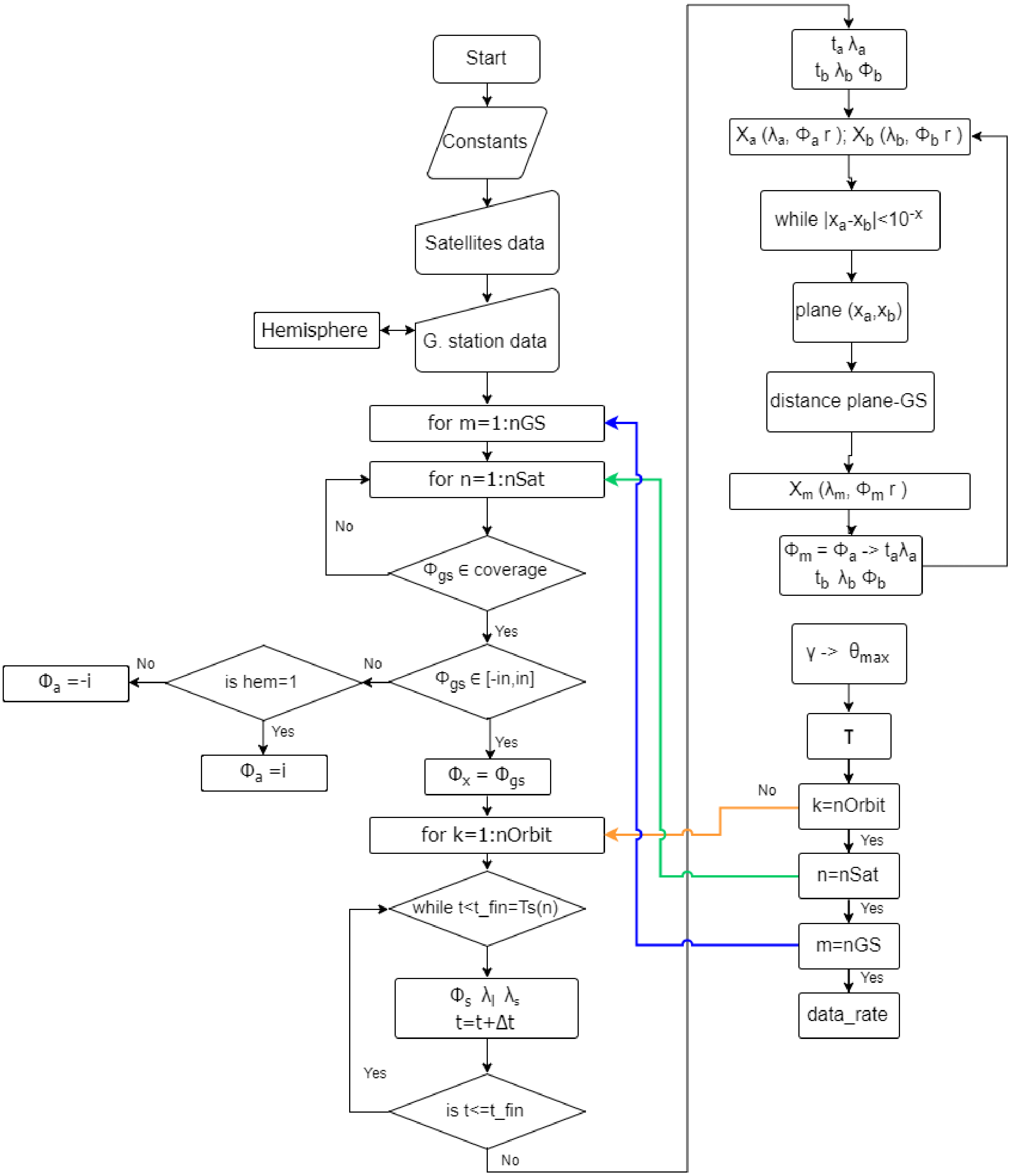


Figure 3.32: Code flow diagram

Chapter 4

Communications architecture

A communications architecture is the configuration of satellites and ground stations in a space system, and the network of communication links that transfers information between them. This chapter gives a brief summary of this links' arrangement and its effect on system design.

To design a communication architecture for a specific space mission, it is important to be clear about the purpose of it. Depending on it, the mission will be characterised differently. Even so, a data flow diagram based on mission requirements and targets is done for every type of mission. Next, it is presented a three parts diagram that every space mission follows.

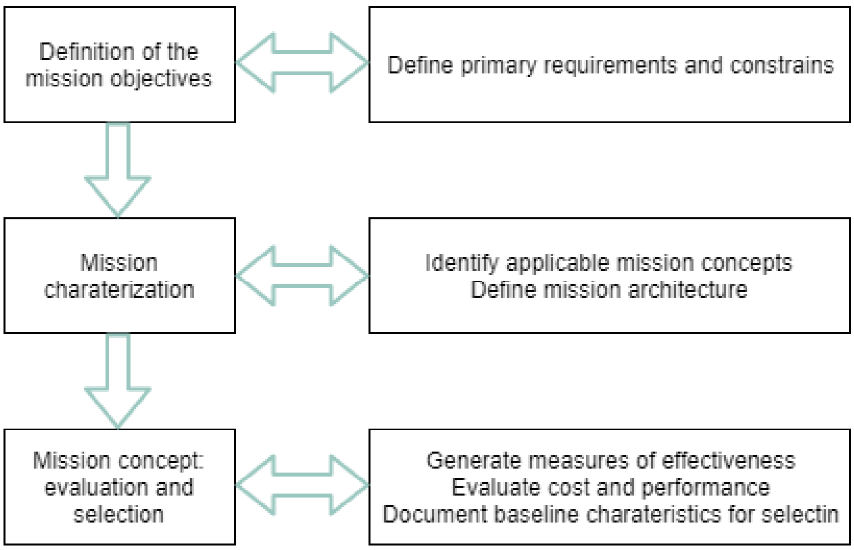


Figure 4.1: Mission diagram flow

The first step is to analyse mission's objectives and requirements in enough detail. Data sources, quantity of information per unit time, access time and transmission delay, among others, are some of the important parameters to be defined. With them, a study is carried out in order to identify the best communication architecture and also to perform a comparison of the possible alternatives. There are multiple interconnections for a network of satellites and ground stations, so depending on the needs for the mission, the links it contains have to be designed separately. The factors which most affect to the design of the mentioned links are the availability of a radio frequency spectrum, the coverage area of the satellite antenna and the path lenght between the satellite and the ground station. When they are defined, the antenna size and the transmitter power can be determined.

Communications architectures can be classified by its satellite to ground geometry or by its function.
The classification by its satellite to terminal geometry gives five different types. First communication architecture is *store and forward*. Satellites communicate with ground stations only during a period of time of its whole orbit. From one terminal, they receive the data and stores it in its payload. When it has communication with another ground station, it downloads the information if it is required. *Geostationary orbit* is another kind of communication architecture. Satellites are placed 35.786 km over the Earth's surface and they orbit at the same velocity that Earth rotates. Because of that, if a terminal is placed in a suitable location, the satellite can communicate constantly with the ground station.

On the other hand, *Molinya orbit* it is an architecture used by the Russians. The satellites orbit in a elliptical orbit with a high inclination, that way the northern regions can be tracked. Its main disadvantage is the antenna pointing. Because of the orbit's shape, the antenna needs to change its direction while satellite is moving over it. This happens also with the *Low altitude* architecture. This kind of satellite to ground station geometry is based on the satellites orbiting near the Earth's surface, almost in a circular orbit. Because of their short visibility windows time, they usually are connected between them in order to send information to the satellite which is communicating with a ground station. Therefore, this type of architecture requires high level of synchronisation between satellites.

If the communication architectures are classified by its function, there are three different types; tracking telemetry and command, data collection and data relay. First, transmits Tracking Telemetry & Command (TT&C) data between the satellite and ground station, either directly of by a relay satellite. This relay satellite can provide broadcast TT&C service to multiple satellites too. Data collection is based on satellite sensor which collect data and transmit it to single or multiple ground stations, either directly or by a relay satellite. Finally, data relay satellites relay data originating on ground or on another satellite to single or multiple ground stations.

In order to decide the best communication architecture for a specific space mission, there are different factors which affect the decision. The most important are listed next, with a brief description of each.

- **Orbit**: It basically determines the visibility window time for uplink and downlink connections. Also, transmitter power and antenna size depend on the distance between the satellites and the ground stations.
- **RF Spectrum**: The RF carrier frequency affects the satellite and ground station transmitter power, antenna size and beamwidth, and requirements for satellite stabilization. Therefore, these factors affect satellite size, mass, and complexity. It is also important to point out that it is necessary to apply for and receive permission to use an assigned frequency from a regulatory agency such as the International Telecommunication Union, the Federal Communications Commission, or the Department of Defense's Interdepartmental Radio Advisory Committee, and every nation's regulatory agency.
- **Data Rate**: The data rate is proportional to the quantity of information per unit time transferred between the satellite and ground station. The higher the data rate, the larger the transmitter power and antenna size required. Processing the space generated data on board the satellite reduces the data rate without losing essential information, but makes the satellite more complex.
- **Link Availability**: it is the time the link is available to the user divided by the total time that it theoretically could be available. It depends on equipment reliability, use of redundant equipment, time required to repair equipment, outages caused by rain, and use of alternate links.
- **Link Access Tune**: The maximum allowable link access time, or time users have to wait before they get their link, depends on the mission. It depends strongly on orbit selection, which determines when a satellite is in view of the ground station.

Even there are different types of communication architecture, all are composed by a network of communication links. There are four types; the ground station-to-satellite uplinks, the satellite-to-ground station downlinks, and satellite-to-satellite crosslinks, or intersatellite links, all represented Figure 4.2. The same communication architecture can have different communication links, all depends on the purpose and of the mission.

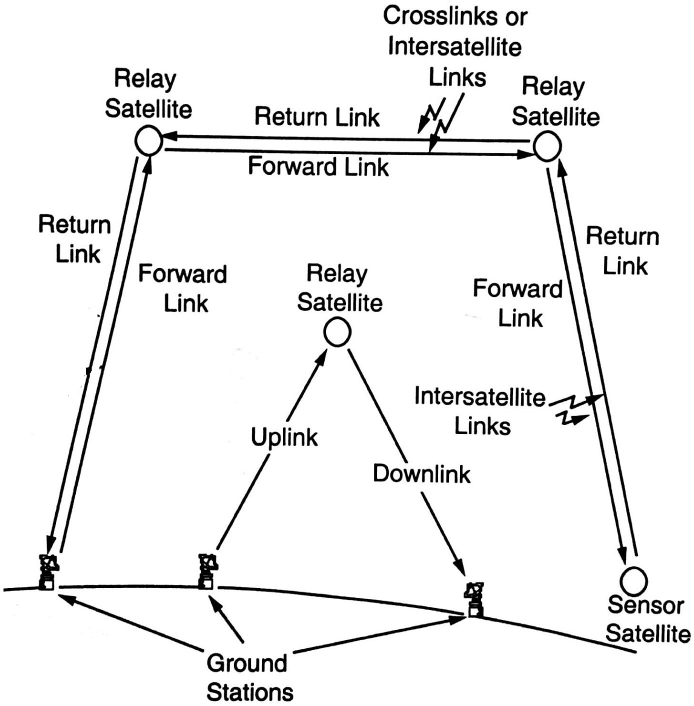


Figure 4.2: Types of communication link
[29]

For the present project, only the downlink communication has been studied. Therefore, it is a small part of all this topic, which is complex because it also has a big part of telecommunications theory. Next section's objective is to provide a brief summary of the process the information follows since it is captured by the satellite, until it gets to the ground station.

Nowadays, digital communications are used instead of analogue because they are less susceptible to distortion and interferences, therefore, they can transmit the data more precisely. Also they can be easily regenerated so that noise and disturbances do not accumulate in transmission through communication relays. In addition, digital links have low error rates and high fidelity, and security communication-links are easy to implement. [29]

4.1 Data process

The information a satellite wants to transmit to the ground station is diverse depending on the objectives of the space mission. The payload carried by the satellite is responsible for taking the data; photos, temperature, chemical properties, among others. This information is stored in the satellite until there is contact with any ground station in order to download it. Other option is to sent the information to another satellite because it will have a better and/or longer visibility window.

For any of the options the satellite has to download the information it collects, the data has to pass a process of modification to ensure the correct sending and receiving. Next are explained the main actions applied to data.

Modulation

Modulation is the process by which an input signal varies the characteristics of a radio frequency carrier, usually a sine wave. There are multiple ways to create this modulation depending on the characteristic which is being modified:

- The modulator can vary the amplitude of the sine wave, known as amplitude modulation.
- The modulator can vary the frequency of the sine wave, known as frequency modulation.
- The modulator can vary the phase of the sine wave, known as phase modulation.

Phase and frequency modulation techniques are the ones more used by satellite systems. For example the *Binary phase shift keying (BPSK)* consist of setting the carrier phase at 0 deg to transmit a binary 0 and setting the phase at 180 deg to transmit a binary 1, as it is represented in next figure.

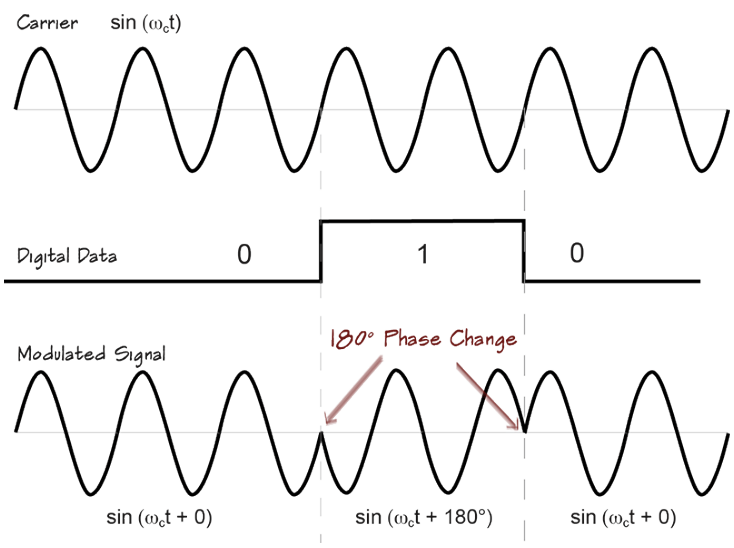


Figure 4.3: BPSK modulation technique
[30]

On the other hand, the *Frequency shift keying (FSK)* sets the carrier frequency at F1 to transmit a binary 0, and at F2 transmit a binary 1. Both, BPSK and FSK, are basic modulation techniques. Satellite's systems use much more sophisticated techniques to communicate large amounts of data in a small amount of spectrum over very long distances. Many satellite systems use phase modulation or even a combination of phase and amplitude modulation.

Demodulation

As the name indicates, the demodulation step receive the modulated carrier, it measures the variations in its characteristics and extract the modulating waveform to recover the information that was transmitted. It is not a easy process to carry out because there exist several factors which corrupt the received signal.

The first factor is noise, which is several natural and man-made unwanted sources of attenuations of the signal send by the satellite. Any natural absorbing medium in the atmosphere which interacts with a radiowave will not only produce a signal amplitude reduction (attenuation), but will also be a source of thermal noise power radiation. The noise associated with these sources, referred to as radio noise, or sky noise, will directly add to the system noise through an increase in the antenna temperature of the receiver. Next figure shows the major natural sources of radio noise which can be present in a downlink.

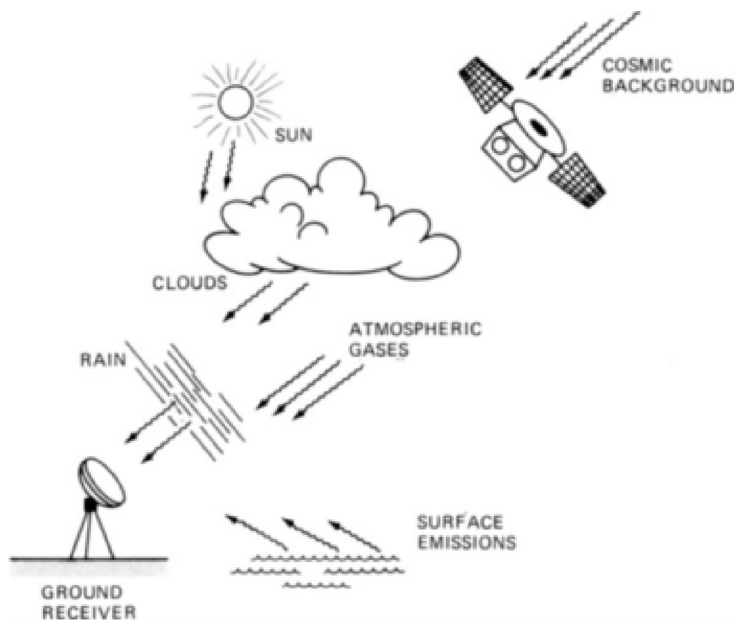


Figure 4.4: Natural sources of radio noise on the downlink of a space communications. [31]

Another source of noise is the electronic systems. The difference of voltage between the resistor's terminals caused by the thermal motion of the electrons generates a which is named thermal noise.
Interferences from other satellites and the Doppler effect because of the satellite movement, corrupt the signal which is trying to reach the ground.

To demodulate a digital bit reliably, the amount of energy received for that bit, E_b, must exceed the noise spectral density, N_o, by a specified amount. Communication theory derives the ratio of received energy-per-bit to noise -density, E_b/N_o needed to achieve a required Bit Error Rate, (BER), at the receiver output. The BER gives the probability of receiving an erroneous bit. For example, a BER of 10^{-5} means that, on the average, only one bit will be in error for every 10^5 bits received. As a signal degrades (a lower signal to noise ratio), the number of bit errors increases.

Coding

Channel coding is also known as forward error correction. Extra bits, called parity bits, are inserted into the data stream at the transmitter. These bits enable the receiver to detect and correct for a limited number of bit errors which might occur in transmission because of noise or interferences. Therefore, the modulator performs the channel coding which adds redundant information to the transmitted data, reducing significantly E_b/N_o. This reduction is known by the name Coding Gain.

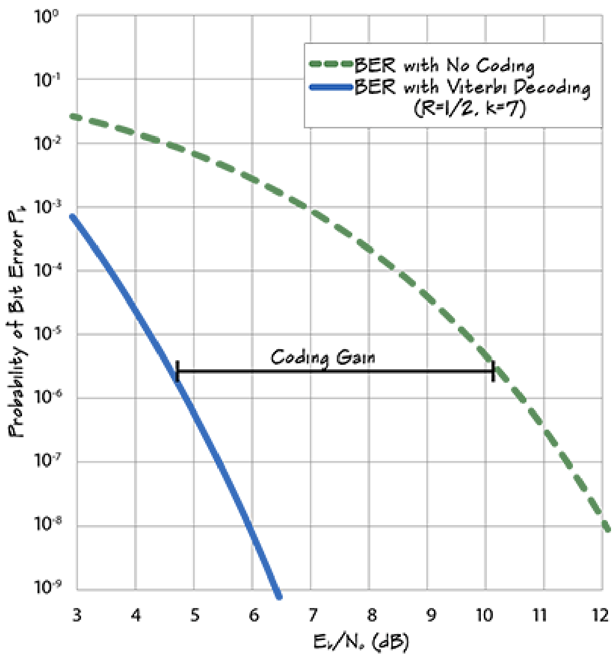


Figure 4.5: Bit Error Probability as a function of E_b/N_o
[30]

At Figure 4.5 is shown the probability of bit error as a function of E_b/N_o. It can be seen that for a signal which has not been codified, the probability of a bit error is higher that for a signal which has been codified for the same ratio of energy-per-bit to noise-density level.

Frequency

For selecting the frequency band there are regulatory constraints, as well as selecting transmission bandwidth and power flux density. There are international agreements which have allocated frequency bands for space communications. These agreements originated with the International Telecommunications Union (ITU) and the World Administrative Radio Conference (WARC). When a satellite is put in orbit, the engineers must apply for and receive permission from the appropriate agency to operate at a specified frequency with the specified orbit and ground locations in order to avoid interferences. Next, are presented the different frequencies that exist and the ones used by satellites are briefly described.

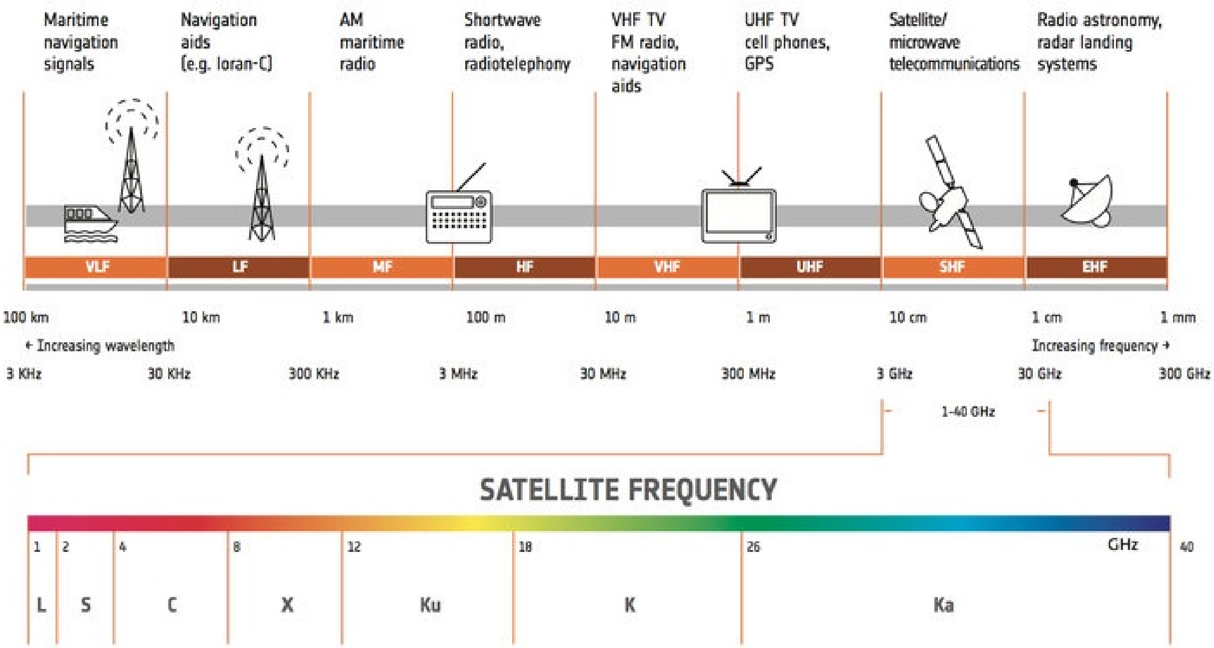


Figure 4.6: Radio frequencies
[32]

L-Band (1-2 GHz)

Global Positioning System (GPS) carriers and also satellite mobile phones, such as Iridium; Inmarsat providing communications at sea, land and air; WorldSpace satellite radio.

S-band (2-4 GHz)

Weather radar, surface ship radar, and some communications satellites, especially those of NASA for communication with ISS and Space Shuttle.

C-band (4-8 GHz)

Primarily used for satellite communications, for full-time satellite TV networks or raw satellite feeds. Commonly used in areas that are subject to tropical rainfall, since it is less susceptible to rain-fade than Ku band.

X-band (8-12 GHz)
Primarily used by the military. Used in radar applications including continuous-wave, pulsed, single-polarisation, dual- polarisation, synthetic aperture radar and phased arrays. X-band radar frequency sub-bands are used in civil, military and government institutions for weather monitoring, air traffic control, maritime vessel traffic control, defence tracking and vehicle speed detection for law enforcement.

Ku-band (12-18 GHz)
Used for satellite communications. In Europe, Ku-band downlink is used from 10.7 GHz to 12.75 GHz for direct broadcast satellite services, such as Astra.

Ka-band (26-40 GHz)
Communications satellites, uplink in either the 27.5 GHz and 31 GHz bands, and high-resolution, close-range targeting radars on military aircraft.

The process that a signal has to go through in order to achieve its final destination is complex and can have lots of variations depending on the nature of the signal and where it has to reach. This has been a brief introduction of some of the steps that signal passes. For a space mission there are aeronautical and specially telecommunication engineers in charge of this section of the mission; satellite system communications.

Chapter 5

Results

The code developed gives the user the opportunity to study different configurations between satellites and ground stations. Also, it enables to extract the data rate necessary in order to download all the information the satellite's payload stores throughout its orbit. This chapter shows the results that can be extracted from the code and how they can be used.

5.1 Visibility window variation and design data rate

In the present section, the altitude and inclination values have been analysed to vary and see how they affect the visibility window time. Also, the design data rate has been calculated in order to see the order of magnitude it has depending on the main mission characteristics.

Regarding the ground station, the Terrassa Ground Station has been selected because it is a university terminal which has been designed and constructed by professors and students form the Universitat Politècnica de Catalunya. Its longitude and latitude are $\lambda_{gs} = 2,00887469999997°$ and $\phi_{gs} = 41,563211°$, respectively. In addition, the satellite has been designed include an optical payload which takes a picture of the Earth's surface every second weighting *7 Mb/picture*. There is no processing on board, therefore all the information recollected has to be transmitted to the ground station. It has also been considered that since the first contact between the satellite and the ground station, the connection is homogeneous. This means the downlink data rate is constant for all the visibility window time.

The influence of the altitude is the first to be tested. In this case, the inclination has been kept constant, taking as a reference the inclination of the ISS obit which is $i = 51,64°$.

As explained in section 3.2, the code gives two graphs; one represents the ground track of the satellite over the Earth's surface and the second one shows the memory status of the satellite.

Therefore, starting with an initial altitude of $H = 404km$, the program calculates the total visibility window time for a 10 orbit simulation.

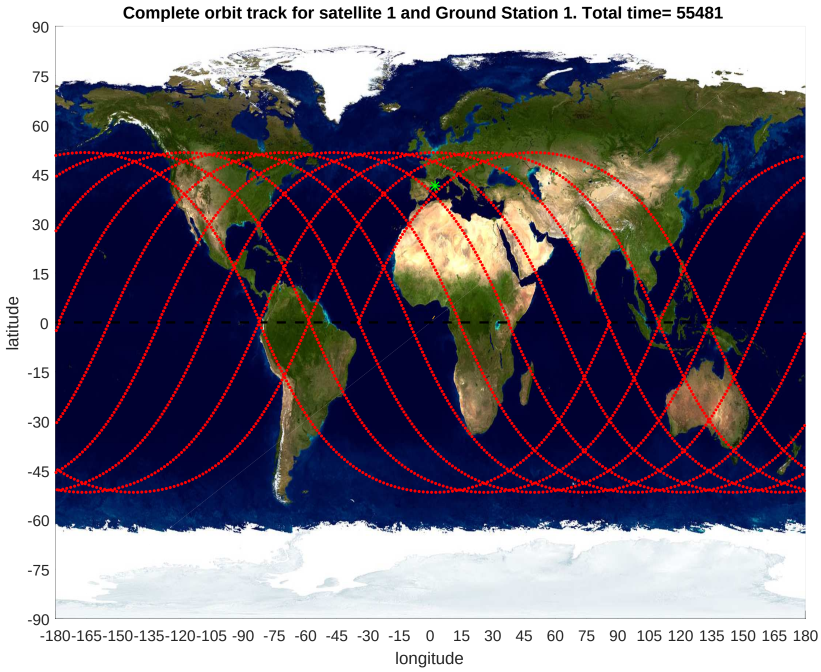


Figure 5.1: Satellite's ground track, *H=404 km*, i=$51,64°$ and *Epoch year=2017*, *Epoch*=$346°$

At the right top of the Figure 5.1 it can be seen the total time of the simulation. The satellite needs *55.481 s*, which is 0,64 days, to perform the 10 orbits. In this case, the visibility windows time is *3.564,2 s* so only the 6% of the time is available to transmit information form the satellite to the ground station. From this data, the minimum data rate to be able to download all the information is *2.183 Mbps*. The obtained data rate is high if it is compared for example with the satellite FireSat. This satellite has an orbit altitude of *700 km* and a data rate of *85 Mbps* according to the calculations of the book Space Mission and Design, [29]. It has to be pointed out that the actual technology is capable to interconnect satellites. With it, a satellite which has a lot of information and can not communicate with a ground station, can send the data to another satellite which has a wider visibility windows, therefore reducing the necessary data rate of the first satellite. On the other hand, nowadays satellites are equipped with a system to process data. It is used to delete data wrong captured.

Therefore, the design data rate obtained is the one necessary if the satellite is working out of any constellation of satellites and without any data process system in its payload. In Figure 5.2, the memory status of the satellite while performing the present simulation is shown.

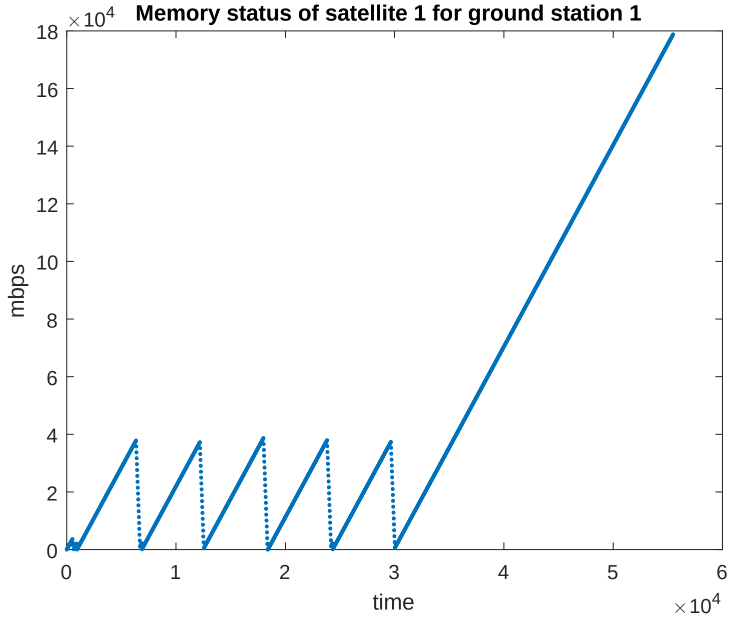
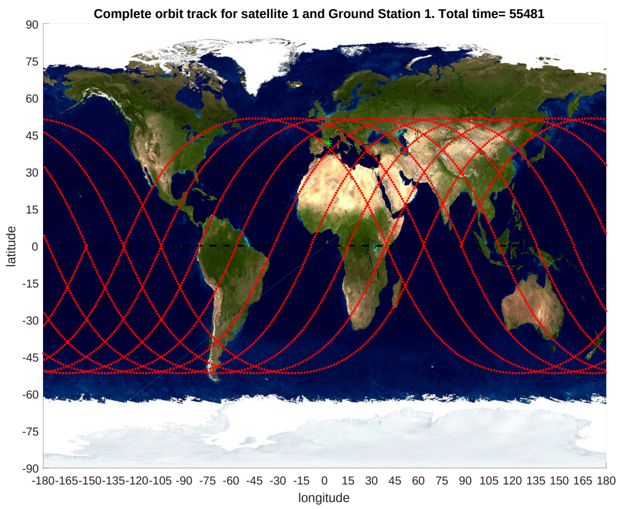
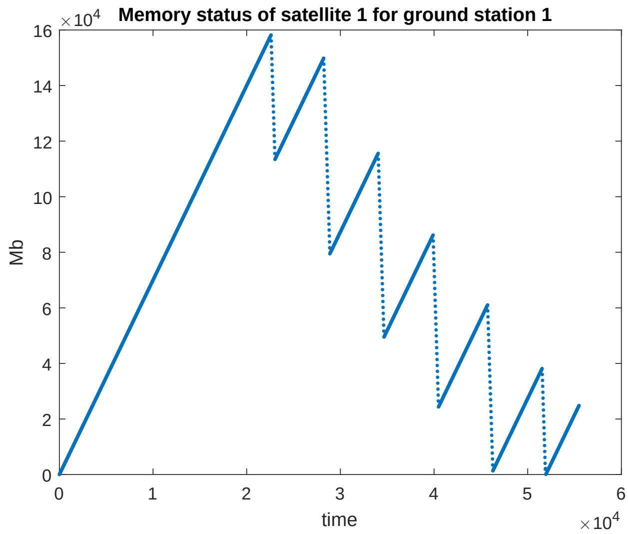


Figure 5.2: Satellite's memory status, *H=404 km*, i=51,64° and *Epoch year=2017*, *Epoch*=346°

As already said, *2.183 Mbps* is the data rate necessary to download all the information captured within *10* orbits. Even so, in Figure 5.2 it can be seen that the memory status of the satellite ends being *180.000 Mb*. This is because the visibility windows are placed at the beginning of the simulation, when the satellite has not recollected all the information yet. If the simulation started at another longitude and therefore, the visibility windows were placed later in the simulation, the memory status at the end of the 10 orbits would be lower.



(a)



(b)

Figure 5.3: Satellite's ground track and memory status graphs, H=*404 km*, i=$51,64°$ and *Epoch year=2008*, *Epoch*=$5°$

For the following case, the altitude has been increased until $H = 3.500km$ above the Earth's surface. As explained in section 3.1, the equatorial orbital shift expands while the altitude grows.

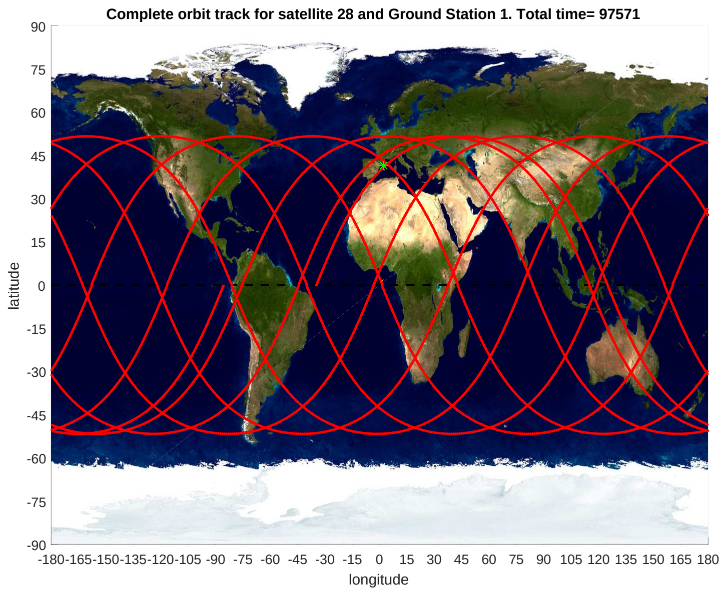


Figure 5.4: Satellite's ground track H=*3.500 km*, i=$51,64°$ and *Epoch year=2017*, *Epoch*=$346°$

In this case, the simulation time is *97.571 s* and the visibility window time is *20.600 s*. As the satellite is more distant from the Earth surface, the trajectory over the ground station is longer, therefore the visibility window times last more.

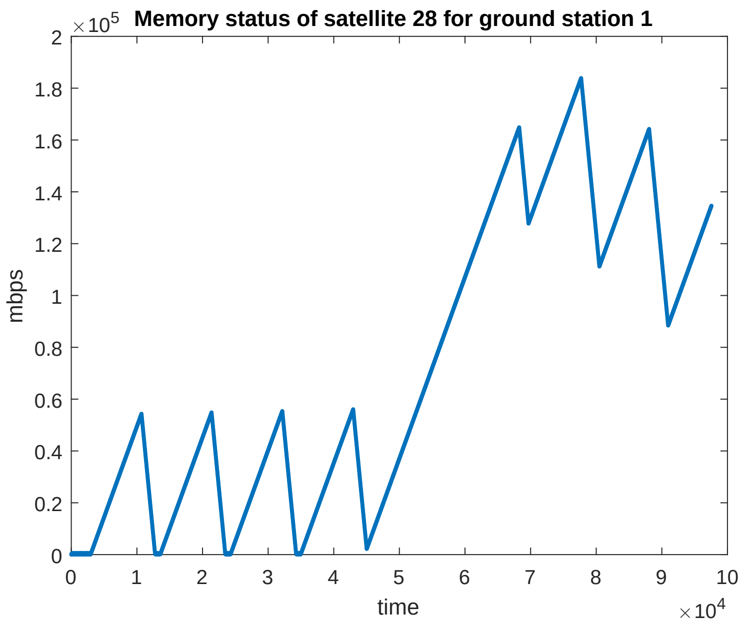


Figure 5.5: Satellite's memory status, *H=3.500 km*, *i*=51, 64° and *Epoch year=2017*, *Epoch*=346°, *Epoch*=5°

The memory status for this simulation shows that there are groups of orbits that make the satellite visible for the ground station. Approximately, the orbits that take place between *3.000s* and *40.000 s* , and from *65.000 s* to *90.000 s* the satellite can communicate with the terminal. Therefore, the orbits that occur between *40.000 s* and *65.000 s*, only capture information, but they are not able to download it.

The evolution of the visibility window time while increasing the altitude for the example exposed (*i=51,64*, *Epoch year=2017* and *Epoch*=346°) is shown at Figure 5.6. As it can be seen, the visibility windows duration increases with the altitude.
With the total visibility window time obtained, the evolution of the design data rate evolution as a function of the altitude can also be obtained. As it can be seen, it decreases when altitude increases thus the longer periods of visibility between the satellite and the ground station.

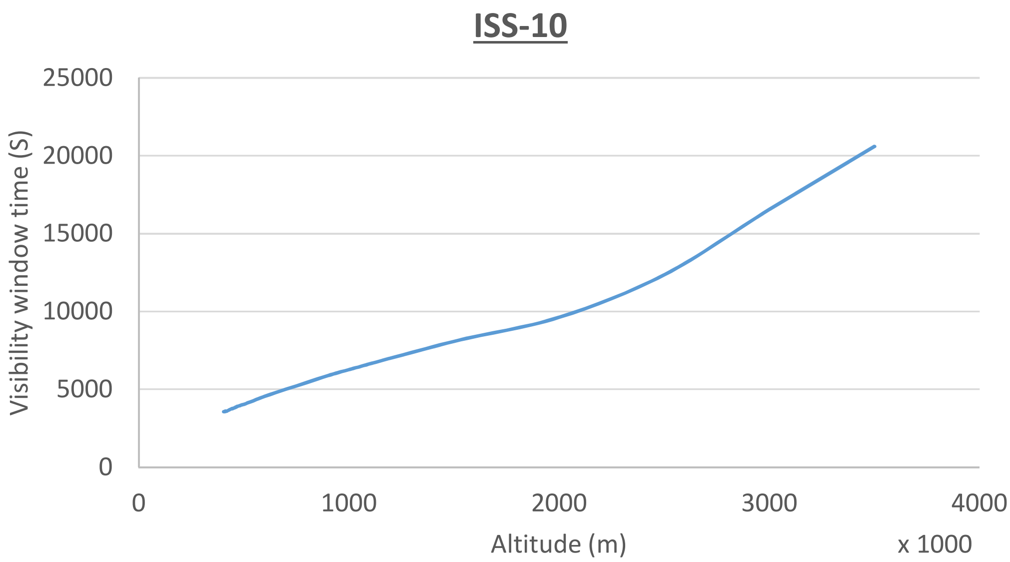


Figure 5.6: Visibility window time - Altitude, ISS orbit

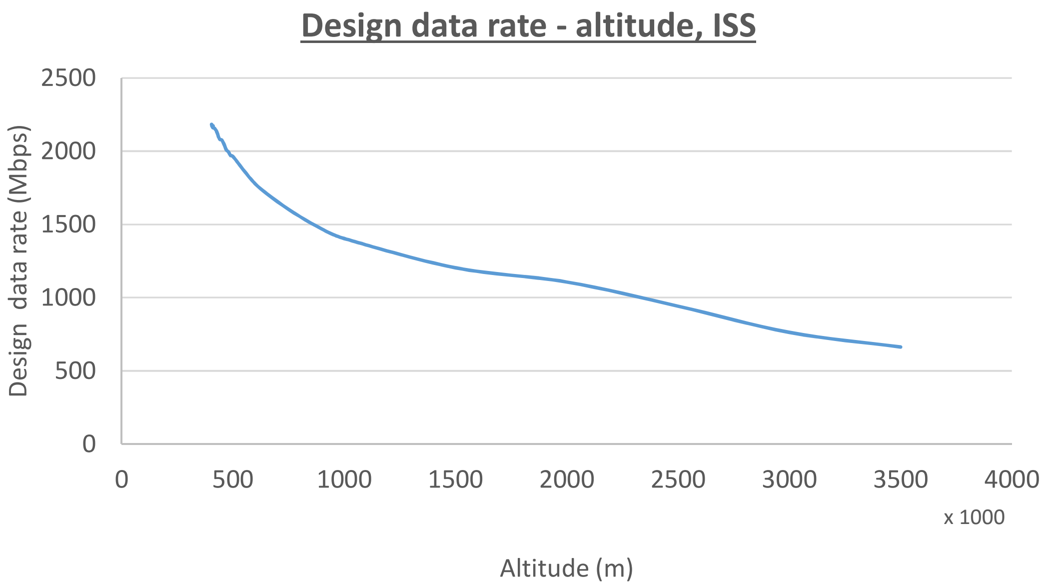


Figure 5.7: Design data rate - Altitude, ISS orbit

The same altitude study has been done with another inclination. In this case the orbit selected has been a SSO. It is a nearly polar orbit and its main characteristic is that the satellite passes over any given point of the planet's surface at the same local mean solar time every orbit. It is wide used for imaging, forecasting and also for security purposes.

Next are presented the figures which correspond to the satellite's ground track and memory status for the altitudes $H=404$ *km* with an inclination of $i=90°$. The time epoch has been kept, *Epoch year=2017* and $Epoch=346°$.

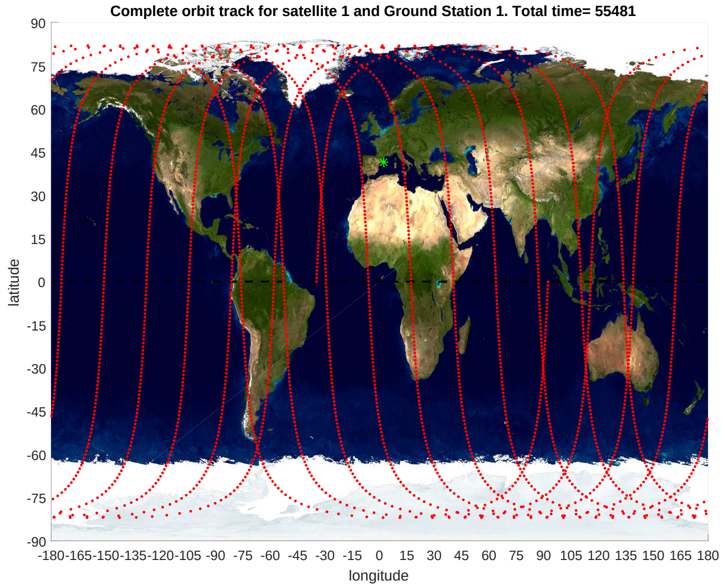


Figure 5.8: Satellite's ground track, *H=404 km*, *i*=90° and *Epoch year=2017*, *Epoch*=346°

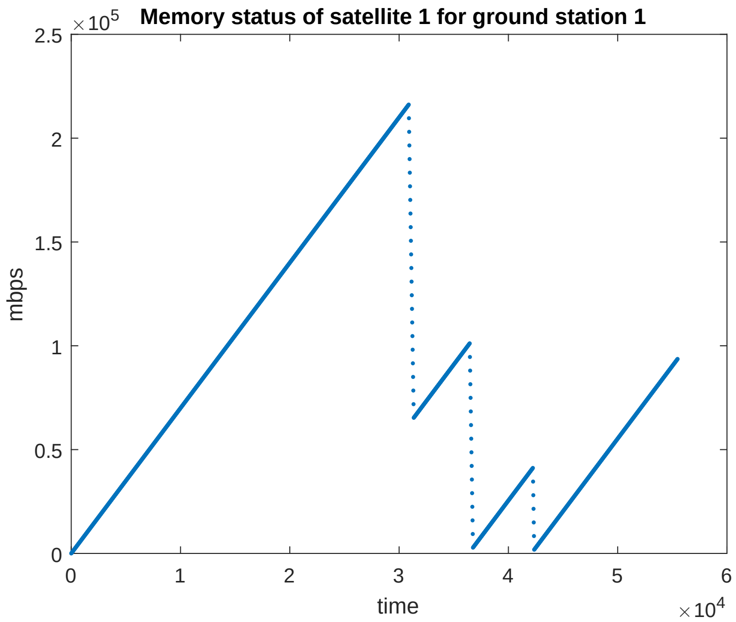


Figure 5.9: Satellite's memory status, *H=3.500 km*, *i*=90° and *Epoch year=2017*, *Epoch*=346°, *Epoch*=5°

Also, the visibility windows time and the design data rate graphs have been obtained from the code. It is observable that the tendency when the orbit was ISS, is maintained when the orbit is SSO. The order of magnitude of the visibility window time does not change for both cases. By contrary, the design data rate for SSO orbit is higher than for ISS orbit, at least when the altitude is low.

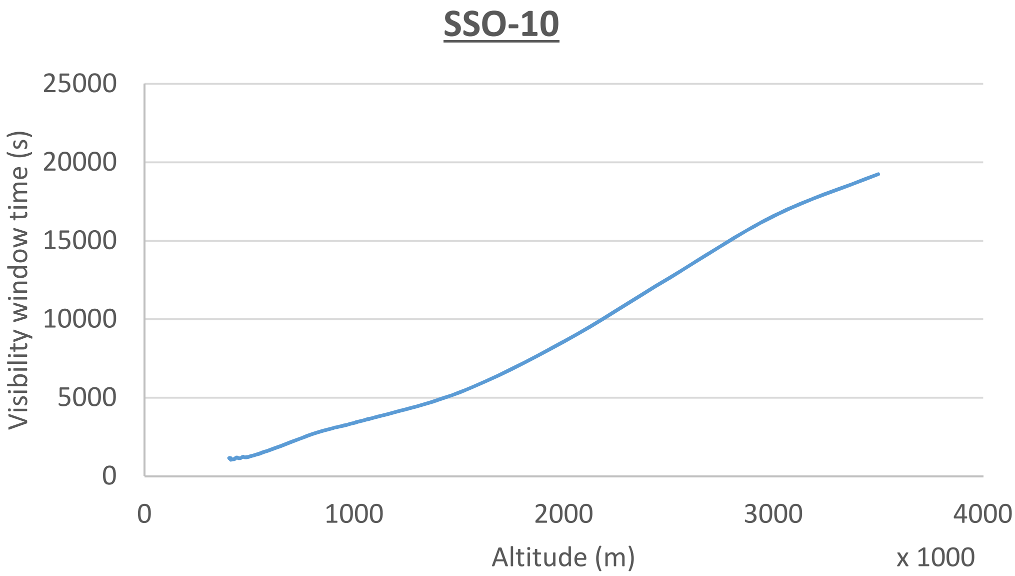


Figure 5.10: Visibility window time - Altitude, SSO orbit

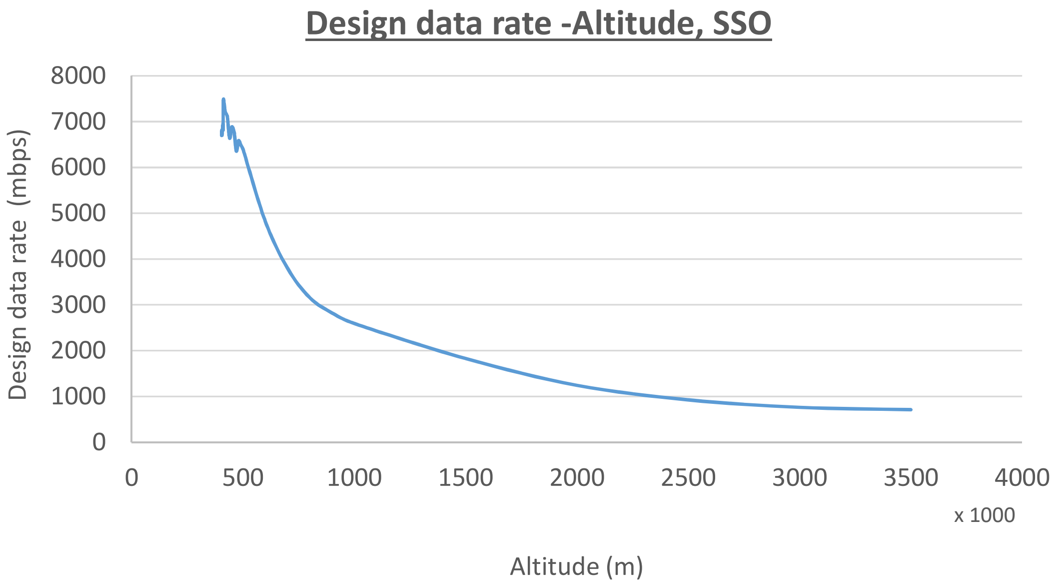


Figure 5.11: Design data rate - Altitude, SSO orbit

This study is important because the altitude directly affects the information that can be collected. As the altitude increases, the period of the orbit is longer so more information can be captured, but also visibility window times last more. The inclination does not affect to the information gathered but it does influence to the coverage latitudes of the satellite, therefore being essential for the possible communication between the satellite and the ground station.

5.2 Comparison with Qbapp

To finalize this work, a comparison between the obtained results and the ones from a reliable source has been considered.

Open Cosmos a privately funded start-up willing to use nano-satellites to provide simple and affordable access to space to organisations ranging from SMEs and research institutions to space agencies in developing countries. One of the services this company offers is a one-stop-shop service that covers all aspects of nanosatellite space missions. The costumer only has to design the payload the satellite will carry to the space. Therefore, OpenCosmos's service includes:

- Full mission simulation
- Spacecraft design
- Integration
- Testing
- Launch procurement
- Frequency allocation
- Insurance
- Operations.

In order to design the payload, the customer receives which is called a qbkid. It is an electronic kit which emulates the mechanical and electrical interfaces of a nanosatellite. Contracting this service, also gives access to a cloud-based web application developed by Open Cosmos called Qbapp. This application allows customers to perform mission analysis studies based on the payload characteristics that the user defines. It also enables the interaction with qbkit in a Hardware-in-the-loop environment where qbkit responds according to a previously defined scenario.

The Universitat Polièccnica de Catalunya has bought two qbkids, having a user for the Qbapp application. On this platform, it has been tried to reproduce the same conditions entered to the code developed, but a proper comparison has not been possible.

To begin with, the code self-developed has some restrictions, like the satellite's latitude at the initial time must be 0°. For that reason the time epoch introduced to the code and the one in the qbapp do not match. Still, the shape of the ground track can be compared in order to see if at least they resemble. Figure 5.12 is the ground track obtained with Qbapp for a period of 1 day. If compared with Figure 5.1 it can be seen that the ground trace is similar.

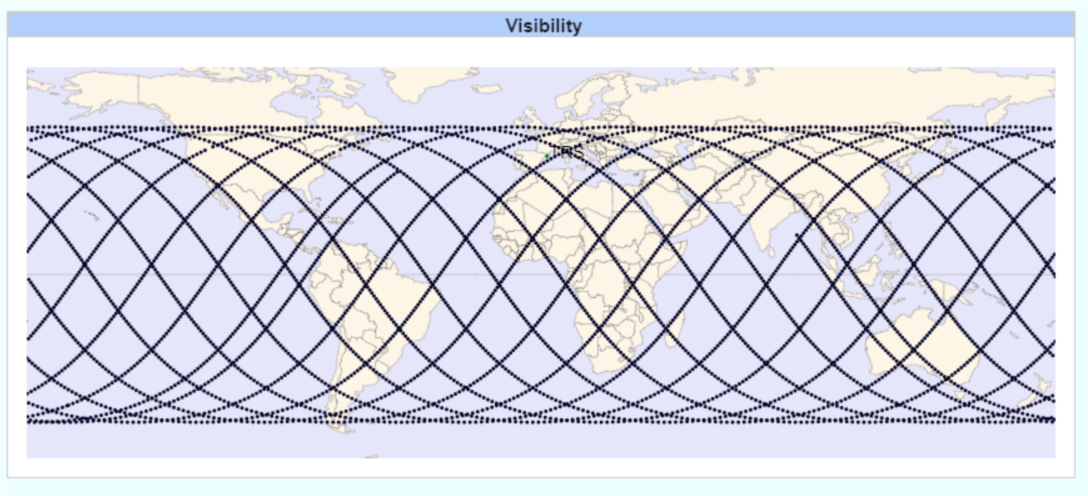


Figure 5.12: Qbapp Ground track

Figure 5.12 is overlay on Figure 5.1 to see that with the same inclination and orbit's altitude, the ground track the satellite draws over the Earth's surface coincide.

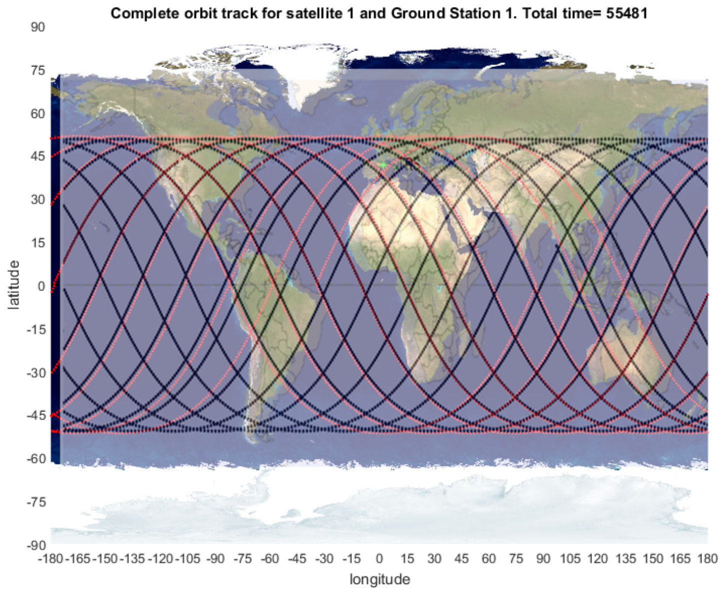


Figure 5.13: Figure 5.12 and Figure 5.1 overlay

Regarding the visibility window time, it is not possible to compare the one obtained with the code developed in the present study with the one given by Qbapp. As this online application is much more advanced, it takes into account the real time of visibility. This means that not all the path that geometrically is suitable for a communication link, can be used. The code explained in section 3.2 does take the first instant of possible connection as a valid communication link, sending all the data possible. This is not accurate to the reality, as it has been said in chapter 4. In addition, the code has been implemented taking as a minimum elevation angle 0° and for the Qbapp, this datum is not given and neither put as a parameter the user can change.

About the memory status, Qbapp considers data generation form the payload (PL) and from the platform (PF). The sum of both results the spacecraft generation data (SC), and it is plotted too.

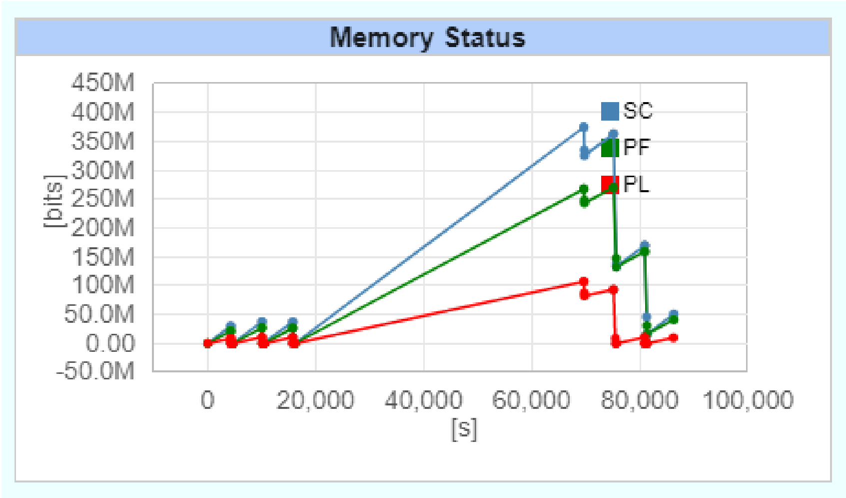


Figure 5.14: Qbapp satellite memory status

The data rate generation of the payload can be introduced by the user, because depending the purpose of it, the data can vary its magnitude. Qbapp sets a range between 1-999.999 bites per second. This online application does not show to the user the data rate transmitted to the ground station while a downlink communication. This part is let to the engineers of Qbapp who design and develop the ground segment technology together with the communications systems of the space segment. However, the concept of the memory status of the satellite has been understood and the plots obtained by the code, coincide with the ones given by Qbapp.

Qbapp is still under development. The technical engineers are working on features for a good visualization of the Cubesat. Depending on the requirements of the payload, a different size of Cubesat is needed.

It is understandable that they do not show all the technical parts of the software and the parameters they use in order to calculate information such as data rate of download or periods of visibility. Even so, this has been an impediment for the present study because the results obtained could not be fully contrasted.

Chapter 6

Conclusions

After the development of the project, some conclusions can be extracted of all the work carried out. This chapter tries to summarize what has been learned while making the present thesis and the conclusions achieved.

Every space mission has its objectives and to perform them, a hard background work is done by all the engineers involved in the project. All the work would be useless if the space segment of the mission can not communicate with the ground segment. All the information gathered would be lost, making the data processing impossible.

The communication link between satellites and grounds stations can be divided into two big groups; the geometry existing between both segments and the process data has to pass in order to be sent safely.

Starting with the communication link geometry, it has been proved that, all the parameters needed for further calculations can be extracted from TLE set. How this set is read and especially how the Epoch time can be calculated has been explained carefully. The decision to establish the initial time of the simulation when latitude is $0°$ makes the first calculations harder, specially because of the different standards which count the time. However, the calculations that come after, are easier to carry out.

Continuing with the geometry issue, an equation to calculate the visibility windows time as a function of the maximum elevation angle and satellite velocity has been obtained. It is remarkable that this function comes from the existing Doppler effect which appears because of the fast satellite velocity over the Earth's surface. As the satellite is paced farther, this velocity decreases and so the Doppler effect, so the equation stops being accurate to the reality. Therefore, the theory developed and the Matlab code implemented can be used for very low and low earth orbit satellites. With the results in hand, the code has given satisfactory results up to orbits with 3500 km of altitude.

The visibility windows duration obtained for each pass depends on the maximum elevation angle. As the elevation angle increases, visibility windows time does it to. Therefore, not for all orbits the in-view time is the same. For the orbits with good geometrical conditions (high elevation angle between the satellite and ground station), the visibility windows time obtained is around 6 min for an altitude of 404 km. This value increases as the altitude gets higher, achieving 25 min at 3.000 km.

Regarding the design data rate, the values obtained are in fact, a first approximation. As the interlink between satellites has not been considered, the amount of data that has to be transmitted while the visibility window is happening, is a lot. In addition, in this study, the satellite is considered to capture a photo every second, it does not matter what it is photographing. In reality, the satellite is programmed to only take photos of specific places that interest the costumers which are going to buy the information. It also should be added that nowadays, most of satellites carry a system which process the data collected in order to eliminate the defective information. It is a preprocessing step that avoids to send useless information to the Earth. All these factors have not been taken into account to determinate the design data rate because it would mean an extensive study that is out of the scope of the present project. Therefore, the design data rate obtained, is the first approximation, that is useful to have an idea of the quantity of information the satellite is gathering.

Turning now into the process that the data has to pass in order to be sent, chapter 4 has been a brief introduction of this immense topic. Images, temperature or density measurements among other information that can be captured in the space, can not be sent to the Earth without being modulated and codified.

The modulation is used to transform the carrier signal so that it contains the information being sent. With it, the information can be passed to the digital world, which is much reliable than the analogue one. Coding ensures all data is transferred safely, adding repeated information. Noise and interferences complicate the communication link, therefore the coding becomes essential.

The present study provides an initial knowledge of the communication link problem. Much further study can be carried out in order to obtain accurate results, which would be useful to safe energy and plan the communication link between the satellite and the ground station when it occurs.

This study has been carry out with hard work and constance. Even the difficulties encountered and the approximate results obtained, this project can be used as a first approach to predict the visibility window time and the design data rate necessary to download all information gathered. With this study, I have learned from basic notions about orbits to complicated concepts about communication links. I am proud of the worked done because the knowledge I have acquired while carrying it out, will serve me to my future as an aeronautical engineering. It will also be useful to all the readers in order to have an initial idea of what communication links between satellites and ground stations are, and how they are produced.

Future work

The present project is an introduction to the reader to understand the issue in visibility window for LEO and VLEO satellites. Even so, this thesis can be developed further. Next, are listed some of the improvements it can be implemented both theoretical and computational level.

- Make a satellite capable to connect with every ground station introduced to the code in order to obtain the data rate necessary to download all the information collected.
- Introduce the interconnection between satellites.
- As explained, not all the visibility window time is useful. It would be a big improvement if depending on the elevation, the data rate of information send by the satellite was calculated.
- Extend the code to be used with elliptical orbits.
- Extend the code to be used at any altitude.
- Carry out a more extensive study of the communication architectures and the modulation and coding procedures.

To perform all these improvements, an aeronautical engineer with experience in space topic and also with skills in computation, would be a good candidate to continue the present project.

Environmental impact and security impact

An environmental study is based on an interdisciplinary subject examining the interplay between the social, legal, management, and scientific aspects of environmental issues.

At the present project does not have a direct impact in the environment since it is a theoretical study about predicting the visibility windows and the design data rate of satellites orbiting at VLEO and LEO. Even so, it is important to point out that satellites have a significant impact to the environment. Throughout their construction, launch and mission control, engineers have to be aware of the environmental impact of their decisions.

Regarding the security impact, the main things to be considered are related with a healthy use of the computer. The engineer has to bear in mind the consequences that entail inappropriate habits. He or she should take into account to work in a correct position, with a straight spine posture and with the look to the front. The illumination is also a crucial point because of the eyes care and it is also recommended doing breaks from time to time.

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