Development of a SW suite for Space Mission Analysis

Bachelor's Thesis

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Jorge Alonso Rosell: *Development of a SW suite for Space Mission Analysis,*
Bachelor's Thesis, Aerospace Engineering Degree,

«Scientists study the world as it is.
Engineers create the world that never has been.»

— Theodore von Karman —

Dedicated to the unique star of my universe.

Thank you for being always there.

Thank you for being you.

ABSTRACT

The space industry is characterized by long term projects and low adaptability, which usually leads to partial-failures and delays. Therefore, the existence of a tool that minimizes these two issues would become inestimable for the sector.

The objective pursued throughout this bachelor thesis is the development of a useful tool to enhance the initial phases of the current spacecraft mission design, particularly focus on the preliminary mission analysis stage.

The problem was decided to be approached by implementing the traditional models into a software suite looking for developing a fast, effective and easy to handle interface.

The software suite developed has proven to accomplish the objective successfully as demonstrated by noticeable improving the evaluation of possible alternatives available for the *FireSat* mission.

Although the project was forced to be scaled according to the thesis restrictions and hence the implications are in some extent limited, it has demonstrated the potential of a future fully-scaled software development project and the impact that it would have on the sector.

La libertad consiste en poder decir que
dos y dos son cuatro. Admitido eso,
se deduce todo lo demás.

— *1984*[Eric Blair] —

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En primer lugar quisiera dar las gracias a mi familia, cuan larga es: a mis padres, mis hermanas, primos, tíos y todos aquellos que, llevando tanto tiempo a mi lado, los años los han convertido ya en parte de la misma. Gracias por por educarme tal y como lo habéis hecho, por inculcarme los valores que ahora rigen mi vida y en definitiva, por convertirme en lo que soy; y gracias en especial, en relación con la presente tesis, por enseñarme a no amedrentarme ante los desafíos y por enseñarme el valor del trabajo bien hecho.

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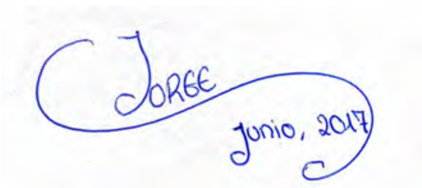

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ACRONYMS AND CONSTANTS

LEO Low Earth Orbit

MEO Medium Earth Orbit

GEO Geosyncronoys Earth Orbit

SMA Space Mission Analysis

SMAD Space Mission Analysis Design

ADCS Attitude Determination and Control Systems

BER Bit Error Rate

ED Energy Density

Part I

STATE OF THE ART

The following chapters introduce to the reader the project behind this thesis. They provide the required background and summarize the Space Mission Analysis process to allow the reader to understand the motivation behind the project and to follow the developments through the chapters; becoming aware of their implications.

1 INTRODUCTION

The following chapter provides the reader with the historical and socioeconomic framework required to be aware of the intrinsic complexity of a space mission design and hence understand the motivation behind this project. Additionally the chapter contains the objectives pursued and the structural organization of the project.

1.1 HISTORICAL BACKGROUND

Space exploration has rightly seized the attention of the contemporary world. On 4 October 1957, the *Sputnik I* was successfully launched and became the first human-made object to orbit the Earth. The age of space exploration had started and the space programs grew rapidly, boosted to a great extent by the Space Race between the superpowers during the Cold War. As a result, dozens of satellites were deployed in different orbits throughout the following decades and their purposes became wider: astronomic observation, telecommunications, atmospheric measurement, etc. In the last thirty years, satellites have undergone an unbelievable evolution and they have become the basis of the Information Age: effortless and instantaneously exchange information with any point of the globe is available to every average user.

At the moment there are officially 4256 satellites orbiting the Earth[1], 1459 of which are currently operating according to the UCS[2]. Every year hundreds of satellites are launched and their purposes vary from Earth observation, scientific investigation or space exploration to communication, navigation or technology development. The satellite mission specifications and its technical characteristics are largely dependent on its purpose and thus the mission life will vary widely from a few months to almost two decades. Even so, the number of satellites orbiting the Earth increases around 4% each year.[1]

As a result of the current satellites characteristics, the sector is mainly divided into four types of institutions that deploy and make use of the satellites:

- Governments (28.3% of the current operational satellites are controlled by them [15]), which will use satellites with numerous purposes through the different ministries and departments to fulfill their needs.

  E.g. SeoSat, belongs to the Spanish government and it will provide ultra-high definition images of the Earth.

1 Index of Objects Launched into Outer Space maintained by United Nations Office for Outer Space Affairs (UNOOSA).[15] [1]
2 Union of Concerned Scientists.[14]

- Companies (40.8%), , basically focus on communication and global positioning satellites.

  E.g. Amazonas 3, operated by Hispasat, it provides civil communications in Brazil.

- Military users (24.3%), which will be interested mainly in communication, observation and global positioning.

  E.g. Skynet 3, which provides strategic communication services to the British Armed Forces

- Civil institutions (6.6%), usually educational centers such as universities or investigation facilities, which take advantages of educational, scientific and space-earth observation satellites.

It is worthy to notice that, due to the nascent development of micro-satellites delivered into low Earth orbits by small launchers[3], the market could be opened to more users in the near future.

1.2 FINANCIAL BACKGROUND

Space missions are highly human, financial and timing consuming projects. Thus, despite nowadays the amount of resources required varies greatly from elemental Cube-Satellites to the most complex military missions; they are not available to the vast majority of the users. Initial missions took almost a decade to be developed from the initial concept to execution, and even though the time was reduced in the following decades, it still takes from several months to few years depending on the mission complexity.

Additionally, apart from the design and manufacturing cost, it is important to highlight the weight that the launch has on the total cost. Analyzing the total price of the launchers offered by the biggest space companies (Arianespace, ULA [4] and SpaceX) it can be roughly estimated that the average cost per ton is around 15 millions of dollars [13]. However there are noticeable variations depending upon the orbit type desired to deploy the satellite and so it can vary from 11m$ to 34m$. As a benchmark, consider that it will cost 25000 $\frac{\$}{kg}$ to put a satellite into a GTO.

Hence it is stated the economic endeavor behind any satellite mission, as well as the enormous influence that the mission specifications and the total weight of the satellite will have on the launch cost, and in turn on the project final cost.

However, there are several costs besides the launch that are frequently overlooked. Firstly, with the technology advancing so fast and the current market variability, the systems installed onboard become outdated rapidly; consequently any delay in the schedule is reducing the potential of the mission, which in turn translates into company economic losses. Moreover, there are particular situations where the schedule can be critical: in general, the schedule is limited and very

3 Mainly led by *startups* like Vector Space Systems
4 United Launch Alliance

tight (certain mission may have only two days each year to be launched); the availability of an orbit changes as new satellites are deployed and the old ones finish their mission life; and the bandwidth available varies identically.

Secondly, there exists an extra cost related with every delay during a satellite development since, according to current financial markets, every missed opportunity represents a profitability loss, a blocked capital that cannot be invested into another project.

1.3 MOTIVATION

Current space mission design follows the next scheme: firstly, the customer requirements are established, from where the mission objectives and the main constraints are derived. Subsequently, a *preliminary analysis* is performed to evaluate different mission alternatives and finally select the most suitable option. Afterwards, the case is analyzed in sufficiently detail to obtain a reliable *preliminary design*. If it fulfills all the requirements and it has the customer approval, the project continues and each satellite system is designed in detail, performing one or several trade-off to accomplish the objectives in the most optimal way and finally enters the manufacturing phase [8] [10]. Any additional modification required by the customer at any point of the process will imply to return to the preliminary analysis and verify that the new design still fulfills the requirements; afterwards, the whole design would be updated. As it can be noticed, it corresponds to a long complex process.

Figure 1: Phases during the space mission design.

However as explained in the previous section, a fast response is a key capacity for any company of the space sector in order to maximize the opportunity cost. It is not only attractive for the customers willing to modify their initial order but it allows the company to respond properly against eventualities.

To achieve this, the *preliminary analysis* is attempted to be accomplished rapidly to carry on with the *preliminary design*, a process that usually takes months, and obtain sooner a design to be presented to the customer for the approval. As a consequence they may overlook a more suitable alternative. The same happens when customers introduce modifications to the initial project in order to reduce the delay associated, which are very frequent considering the duration of the project.

As it can be inferred so far, the existence of a software tool that facilitates the evaluation of the different mission alternatives has a great potential. Obviously, the complex the mathematical models implemented in the tool, the better the accuracy but the higher the computation time. Accordingly, the software tool can be focus on enhancing:

- The *preliminary analysis*, allowing the user to perform fast modifications in the original model and easing the comparison between the alternatives available. The tool must be fast enough to be useful, and thus the mathematical models should not be extremely detailed.

- The *preliminary design*, developing a more detailed analysis that will allow the user to accomplish the alternative selection according to the more accurate estimations.

Although which is the most suitable option will depend on the company preferences, in both cases the software would facilitate enormously the selection of the best alternative and would avoid overlook a potential option as it happens nowadays.

The software proposed to be developed throughout this thesis corresponds to a light software model, focus on the *preliminary analysis* at the very beginning of the space mission design; e.g. during the initial customer meetings. A great advantage considering that it will be possible to provide cost estimations to each of the alternatives and hence the customers can evaluate by themselves the option that better suits their requirements at the very beginning of the process, before establishing the mission objectives and constrains.

As an example: *A customer requires a communication satellite that provides a full coverage to the Scandinavian Peninsula, providing* 100 *channels with a maximum cost of* 50 *millions of dollars. When the company performs a preliminary design they may found that it is impossible to accomplish those requirements with that budget. Therefore the customer must be called to another meeting where the initial contract must be renegotiate, which is not a desirable situation. On the other hand, if the company has this kind of tool, once the customer provides his requirements, it is possible to provide a rough budget at that very moment and, if he does not agree, it will be possible to analyze different alternatives to reduce the total cost. For example, reducing the coverage area or reduce the number of channels. There is a large number of modifications available for the initial plan, and it is possible that the customer, once he notices that by reducing from* 100 *to* 80 *channels or by reducing the coverage the total cost might be reduced to half, it is possible to lay down a much better agreement at the beginning. In the same way, for any modification that the customer may be considered once the process has started (like a better antenna) it is possible to provide a good estimation of in what extent it will modify the initial project, and hence he will be provided with enough information to decide to proceed or not. Notice that, in this way, the delay related with the initial preliminary design could be avoided.*

1.4 OBJECTIVES:

The bachelor thesis presented throughout this document is intended to develop a software suite to provide support during the analysis phase of a space mission.

The software will provide the user with an effective tool to evaluate all the alternatives and to perform a fast trade-off between the different parameters available in order to accomplish the mission requirements in an optimal way.

In order to do so, it is expected that the tool provides:

- A fast response.
- An ergonomic and a clear interface to enhance the fast response.
- Sufficiently precise results.
- Estipulates clearly its limitations regarding the model implemented, in order to be aware in what situation the tool is not longer useful.

1.5 SCOPE OF THE WORK

The bachelor thesis will be structured in the following way:

- Chapter 1: Introduction. The reader is introduced in the Space Mission Analysis design and is provided with the necessary background to understand the motivation behind the thesis.
- Chapter 2: Space Mission Analysis. It determines the current state of the art and every system that plays a main role in the space mission is explained in detail.
- Chapter 3: Tool Design. This section contains a fully detailed compilation of the SW information: the program used, the requirements that must be achieved, motivation for each decision and a fully scope of the SW architecture.
- Chapter 4: Models. The following section includes the mathematical models that were finally implemented in the SW. It collects every assumption and hypothesis that has been considered as well as every reference or tables used. It also depicts the software limitations.
- Chapter 5: Tools Validation. Complete analysis of a real case to support the strength of the mathematical models and the accuracy of the hypothesis considered in the SW tool.
- Chapter 6: Socioeconomic Analysis. It contains an economical study of the thesis considered as an engineering project, evaluating the socio-economic impact on the aerospace industry and the production plan. It also includes a fully detailed project budget.

- Chapter 7: Legal Framework. It contains an analysis of the legislation that rules the implementation of the project (risks, responsibilities, privacy and security). It will include the technical standards and a deep analysis on intellectual property.
- Chapter 8: Conclusions. It summarizes all the ideas developed throughout the thesis, and a detailed analysis of the objectives accomplishment degree. Finally, a complete section has been dedicated to the future work that can be still developed.
- Bibliography and Appendices have been included at the end of the thesis to support every reference, quotation and idea expressed throughout the thesis.

2 SPACE MISSION ANALYSIS

The following chapter establishes the state of the art, hence complementing the background provided in the introduction. The chapter reasserts the importance of the preliminary analysis during a space mission design, where the key parameters are determined and an optimization process is performed to fulfill the customer requirements in the best way. It includes as well an analysis of the principal mission requirements and a revision of the main satellite subsystems.

2.1 THE SMA PROCESS

The aim pursued during the *Space Mission Analysis* is to determine the parameters that characterize the mission and define its objectives, looking for optimizing the time required while minimizing the cost and risk associated.

As it is widely known all the spacecraft missions are long term and highly costly projects. The problem arises from the fact that these two characteristics force the customer to demand for higher reliability, and as a consequence it increases even more the cost, which in turn increases the schedules and thus it reduces the number of missions that are finally developed. This phenomenon is commonly named the *Space Spiral*, and it becomes a significant contributor to the long schedules and cost associated to this sector.

As a consequence of the high reliability demands, all the mission designs consider to a greater or a lesser extent similar technologies, procedures and materials. The only exception occurred when a disruptive technology is developed. It corresponds to a breakthrough design, model or process that leads to a great improvement in the final performance.

On the other hand, the mission parameters would vary widely depending on its purpose: military, commercial, science observation, human-space flight, interplanetary, small satellite launch, i.a. But as it was mentioned, even if the parameters could vary widely, the main stages are common to all the designs and they will be depicted in the following sections.

2.1.1 *Stage I. Define Requirements and Constrains*

During the first stage of a *Space Mission Analysis* the mission objectives are defined; from where, in turn, the constraints that will lead the design process are determined. These constraints could be the maximum cost allowed, the time required, number of ground stations available, the number and types of users, etc.

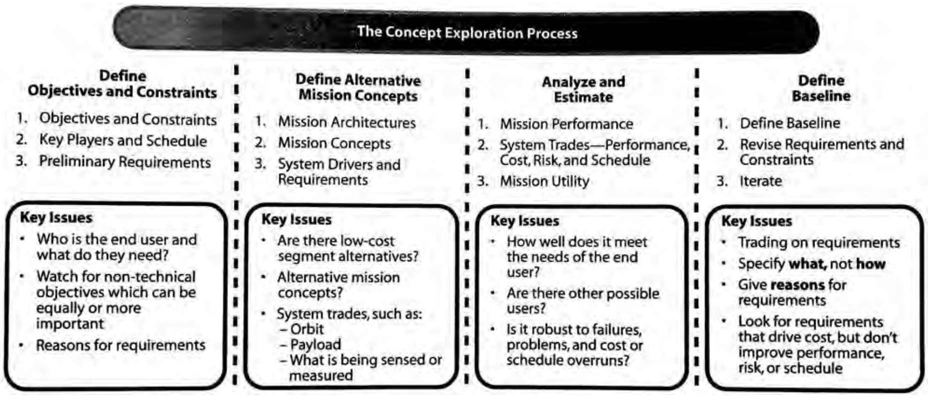


Figure 2: Initial stages during the space mission design process[8].

It is considered that the first stage starts during the first meeting where the customer or the headquarter directors will present the main idea, establish the purpose and provide an initial set of requirements and objectives for the mission. The objectives are usually raised attending to two categories:

- Functional objectives: performance, coverage, timeliness (signal lag, data interpretation, ...).
- Operational objectives: commanding, design life, availability, survivability, data distribution, etc.

Similarly, the mission constrains are obtained once we evaluate the same points according to the objectives provided:

- Functional constraints: determined by in what extent it is desired to fulfill the requirements by improving the performance of each system.
- Operational constraints: determined by how the systems will operate and how users interact with it to meet their specific needs.
- Project constrains: limitations like the maximum cost, schedule or technologies available to develop the project.

Initially the objectives are poorly defined, but they are analyzed in detail throughout the stage, establishing a hierarchy of priorities. Doing so, the impact of each constraint can be evaluated and it is possible to make an initial estimation of how well it is possible to fulfill the objectives. This analysis usually leads to reassess the initial mission and modify the requirements. After the analysis, the design would be ultimately driven by the money, the schedule limitations, the risk, the current regulations, environmental issues, the technology available, etc.

2.1.2 *Stage II. Drivers and Alternatives*

Once the required objectives have been set, the second stage involves finding all the possible alternatives that will fulfill in principle these requirements. Firstly the *space mission architecture* is depicted: each element of the mission is designed in detail and hence the main drivers are found. Secondly, the *mission concept* is performed, where all these systems are put together in practice and evaluated. At this step, iteration is required to solve all the incompatibilities between the satellite systems and optimize the final design. In many cases, the iteration goes back to the first stage where the requirements and the constraints are analyzed again, the hierarchy is redefine and the level at which each requirement must be fulfilled is reviewed.

As it can be appreciated, this second stage becomes one of the most important ones considering the effort required to develop each design and the impracticality to return to a discarded alternative once the process is in an advanced stage, even if later it proves to be more efficient.

The *mission architecture* is determined by the cost, the performance and the system drivers. The drivers represent all those parameters that determine in a great extent the performance of the system. They become very useful to evaluate a possible modification in the total design; it allows focusing the analysis on the drivers' variation and by considering their impact on the mission it is possible to estimate the effect of that modification. The system drivers will depend on the type of mission and the requirements demanded, so they can be the number of satellites, altitude, power, availability, the lifetime, the coverage, the payload size, mass, the signal characteristics, etc. It is important to notice that the drivers are used to enhance the design but ultimately they would determine the cost and the satellite performance, which are in fact the authentic drivers that will drive any project.

The architecture is divided in four steps:

- Design the architectural elements according to preliminary requirements and constrains determined in the first stage and determine which ones are not fully constrained and allow a trade off.
- Analyze the main options for each tradeable element.
- Evaluate which options will be more suitable to work together considering the drivers, study if one option determines the rest and finally analyze which combination is the most suitable one.

The *mission concept* step corresponds to an iteration process that would finally provide a preliminary design of the mission alternative. Subsequently all the systems performances are evaluated as a whole, all the incompatibilities are solved and all the systems are redesigned to finally accomplish the best configuration to fulfill optimally the requirements imposed. The procedure is sketched in figure 3.

Once an optimal preliminary design is obtained the last step is to evaluate the how well this design fulfills the requirements during the *Mission Utility Analysis*, where the selected alternative is evaluated in terms of cost, risk, schedule and performance; then a project baseline is set and the final requirements are finally

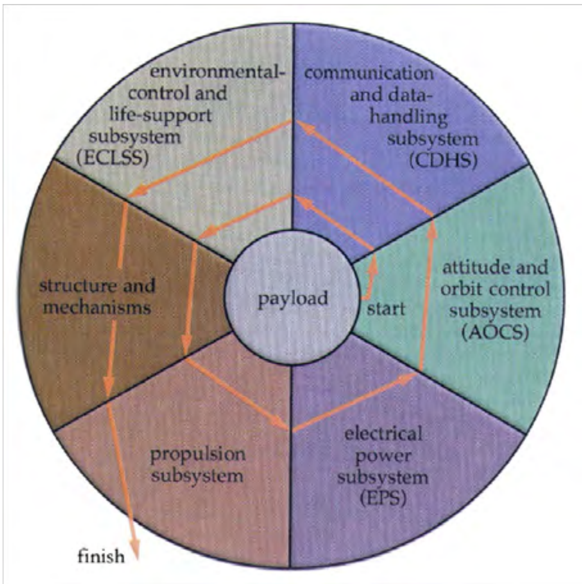


Figure 3: Sketch of the mission concept stage.

determined. After that, the selected alternative is designed in detail and finally enters in the manufacturing stage.

2.2 MOST CRITICAL DRIVERS

As mentioned above, the main drivers would vary depending on the mission characteristics. However they are, in principle, determined by the design decisions of certain subsystems and they will condition in the same extent other mission parameters.

Consequently, although during the *mission architecture* stage a specific analysis of the main drivers and their influence in the particular mission would be performed, it is interesting to evaluate in general terms the most critical drivers, relating them with the main parameters that they determine as well as the main parameters that mostly influence that driver. These results are summarized in figure 4.

2.3 MOST CRITICAL REQUIREMENTS

The mission requirements would be determined by the objectives established during the first meetings. The great majority of the missions share most of the requirements, varying according to the extent and the way in which they are decided to be accomplished.

DRIVER	Elements that conditioned the driver	Elements limited by the driver
Size	Shroud size, available weight, aerodynamic drag	Payload Size (frequently antenna diameter or aperture)
Weight	Altitude, inclination, launch vehicle	Payload weight, propellant loads, manufacturing and the design
Power	Size, weight, control	Payload & bus design, sensitivity, mission life, data rate
Propellant	Available mass	Velocity impulse, flexibility, on-orbit responsiveness
Data Rate	Storage, power, antenna sizes, limits of existing systems	Information sent to user, on-board processing
Communications	Coverage, availability of ground stations or relay satellites	Timeliness, ability to command
Pointing	Cost, weight	Resolution, geolocation, overall system accuracy, cost
Number of Spacecraft	Cost	Coverage frequency and overlap
Altitude	Launch vehicle, performance demands, weight	Performance, survivability, coverage, communications, resolution
Coverage (geometry & timing)	Orbit, scheduling, payload field of view and observation time	Data frequency and continuity, maneuver requirements
Scheduling	Timeline and operations, decision making, communications	Coverage, responsiveness, mission utility
Operations	Cost, crew size, communication	Frequently the principal cost driver, principal error source, autonomy

Figure 4: Critical drivers and the relation with respect to mission characteristics, adapted from the "The new SMAD" [8].

On-orbit Lifetime.

As it can be inferred one of the most crucial requirements is the *on-orbit lifetime*. The spacecraft undergoes a constant degradation by operating in the space and requires a constant tracking and orientation corrections. The longer the mission, the higher the probabilities of a system failure and thus the higher the level of redundancy required to guarantee a full operation of the spacecraft subsystems. This redundancy will increase the complexity and the mass of the systems, and thus the cost. Similarly, the amount of degradation conditioned as well the technology and the materials to be implemented, that must guarantee the full operation during the whole life.

Moreover, since the propulsion system corrects the spacecraft position and orientation and the spacecraft requires constantly a power supply, the longer the mission life the higher the power and propulsion budgets required to guarantee full operability. In fact, the mission life conditioned in great extent the size and the technology selected for the solar arrays and the batteries due to the space degradation and the cycle loading that they will undergo.

Coverage or response time.

A huge percentage of space missions are designed to point towards the Earth surface despite having different purposes: Earth observation, communication, global positioning, etc. Therefore, depending on the size and the geographic location of the sector to be covered it, and depending the type of data to be collected by the satellite, it will conditioned the communication architecture, the orbit, the altitude, the inclination, etc.

Furthermore most of the mission will require a constant or a periodical data exchange with a ground station regarding the tracking, the attitude correction, mission modifications, exchange the collected data, etc. Consequently, the availability of a communication channel with a ground station will conditioned as well the number of satellites required, the scheduling and the staffing requirements.

Resolution.

Regarding the quality of the data collected required for the mission it will determine the payload size, cost and operation, but also the accuracy of the altitude and attitude control.

Sensitivity.

Similarly as the resolution, the sensitivity requirements will determine as well the altitude and attitude of the satellite, but it will determine as well the level of complexity of the payload and its size. For a communication satellite, the payload will be the transmitter and the receiver antennas. The sensitivity to the signal received from the user will determine the size, the materials, the quality of the antennas installation and the thermal control (that raise the noise levels of the signal). Furthermore the higher the sensitivity required the larger the data processing associated.

Signal strength.

As it may be expected, the signal strength would determine the payload features; for higher signal strength the larger the size and the power required to accomplish it. The strength of the signal would be reduced by the distance to the receiver. Consequently the altitude position will play a key role when designing this requirement.

Survivability.

Related to the mission life, the survivability of the design would affect several mission parameters. If the satellite is designed to operate at low altitudes, the high level of radiation and particle impacts would force to incorporate extra protective layers in the most critical elements such as the solar arrays. The degree of protection to be installed in the design will determine the weight, the power, the altitude, the component selection to withstand the level of degradation, the redundancy, etc.

Furthermore, in order to guarantee a detailed mission tracking the level of survivability will determine as well the number of ground stations, the communication architecture, the number of satellites, etc.

In general terms, the survivability requirement will drive the design of the space and the ground segments.

The previous analysis has been summarized in the following table:

REQUIREMENT	Elements driven by this requirement
On-board Lifetime	Redundancy, weight, power and propulsion budgets, component selection
Coverage or response time	Number of satellites, altitude, inclination, communications architecture, payload field of view, scheduling, staffing requirements
Resolution	Instrument size, altitude, attitude control
Sensitivity	Payload size and complexity, processing, thermal control, altitude
Signal Strength	Payload size and power, altitude, transmit power
Survivability	Altitude, weight, opwer, component selection, design og space and ground system, number og satellites, number of ground stations, communications architecture, system redundancy

Figure 5: Critical requirements selection [8].

There are many other requirements that can drive a spacecraft mission design, and their relevancy would depend on the mission characteristics. But as it can be observed in the requirements analysis provided, it is highly important to specified all the mission requirements and establish a clear hierarchy between them. Doing so, the design process would be clearer when a modification would alter two requirements.

2.4 MOST RELEVANT SUBSYSTEMS

According to the ideas presented through the previous sections, the specific characteristics of a mission and its purpose will determine the most relevant drivers and requirements. These elements would affect in more or least extent each subsystem while they will be limited at the same time by the design decisions implemented in other subsystems.

Therefore, the relevance of the design of each subsystem and the impact on the final design will be conditioned by the requirements established for that mission and the main drivers. For example a simple satellite mission orbiting the Earth to study the evolution of certain cosmologic event will require an extremely accurate pointing control while the altitude or the propulsion system for orbit maneuver would not be relevant.

On the following subsections the main subsystems would be described together with the usual design models and their complexities. For the sake of simplicity, the deep space missions are not included in the explanation but the same considerations and conclusions can be applicable to them.

2.4.1 *Mission design*

A space mission is characterized in a great extent by the orbit and the maneuvers required. The orbit will determine the frequency and the duration of the eclipses, the signal losses, the accuracy and the quality of the data obtained by the payload, the incidence of the sun flux that in turn will determine the thermal and the power subsystems, the geographical position and the associated scheduling and finally the space environment that highly influence many of the satellite subsystems.

The orbit is corresponds to the traditional keplerian model: a conic section curve with the massive body located in one of the focus and fully defined by six parameters.

- Two determines the shape and the size of the trajectory, usually the *semimajor axis* and the *eccentricity*. For satellite missions, it is very common to describe the geometry providing the satellite height at the perigee and at the apogee, frequently shortened to "perigee" and "apogee", since during the design the distance from the Earth surface will influence several subsystems such as the losses in the communication link.
- Three describe the orbit orientation: [1]
  - The *inclination* i defines the angle between the orbital plane and the equatorial celestial plane [2].
  - The *right ascension of the ascending node* Ω defines the angle in the equatorial plane measured eastward from the vernal equinox to the ascending node of the orbit. The ascending node corresponds to the point where the satellite crosses the equatorial plane going from south to north.
  - The *argument of perigee* ω defines the angle in the orbit plane from the ascending node to the orbit perigee.
- One determines the position of the satellite within the orbit by defining the angle in the orbit plane from the perigee, called the *true anomaly* ν. Since celestial bodies in the solar system rotate counter-clockwise, most of the satellites are launched in the same direction which is said to be in *prograded* orbits. The opposite are called *retrograde* orbits.

1 The following angles are described for Earth-centered orbits. For other missions, they will be referenced towards other significant planes and directions. Additionally, the generic nomenclature will be: inclination, longitude of the ascending node and argument of periapsis.
2 Deep space missions reference towards the ecliptic plane

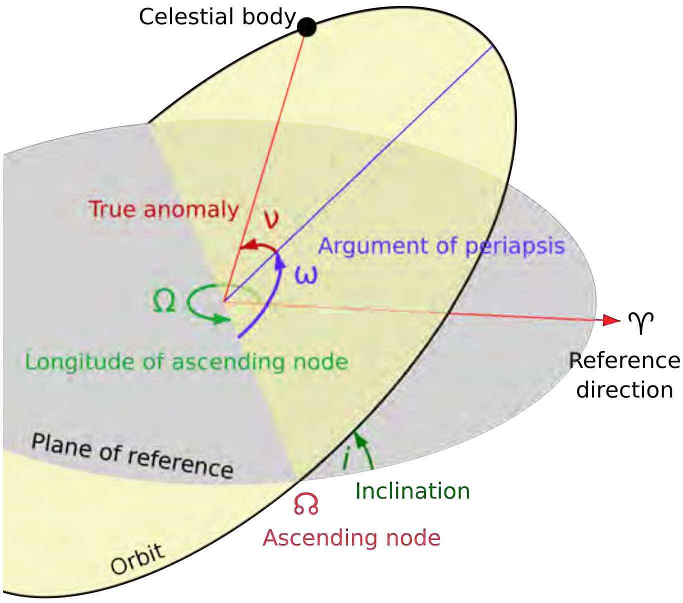


Figure 6: Orbit design parameters.[7]

Traditionally, orbits have been classified according to the size and the shape of the orbits, since they share specific characteristics depending on them:

- Low Earth Orbits (LEO). Geocentric orbits located below 2000 km from the Earth surface. Therefore orbit periods will be 127 min or less travelling at 7.5 $\frac{km}{s}$ approximately. These orbits are characterized by encounter high drag losses and providing high bandwidth and low communication time lag. Furthermore, the velocity impulse required to achieve this orbits is approximately 10 $\frac{km}{s}$; therefore it is possible to place the satellite with the same launch rocket. No propulsion system is required although is very common to incorporate one to finish the maneuver and overcome possible issues. The ISS is located at a $\sim$ 400 km LEO orbit.
- Medium Earth Orbits (MEO). Geocentric orbits located between LEO and the geosynchronous orbits (35.786 km). Since at those altitudes the satellites covers wide surface regions they are usually used for navigation, communication and science observation. The most common altitude is 20.200 km since it yields an orbital period of 12 hours, which simplify the mission tracking and schedule. The *GPS*, *Glonass* and *Galileo* systems are located at that semi-sidereal period orbits for several reasons, i.a they follow the same track over the Earth and the drag losses are minimum at those altitudes.
- Geosynchronous (GSO) are located at 35.786 km from the Earth surface and their key characteristic is that they match the Earth's sidereal rotation period. Those orbit located at the equator are called Geostationary orbits (GEO) and they are characterized by having a fixed position in the Earth sky. The best advantage is that they do not require pointing systems since they can be focus in the same point. One of the most famous GSO is the *tundra* type orbit, a highly elliptical-highly inclination orbit (near 63.4ž) characterized

because the satellite spends most of its time over a chosen area of the Earth. The *molniya* orbit corresponds to a variation of the *tundra* type orbit with a semi-sidereal period (MEO).

- High Earth orbits (HEO) for orbits located above GSO. Since their orbital periods are greater than the sidereal they will have an apparent retrograde motion moving westward. There are slightly used.

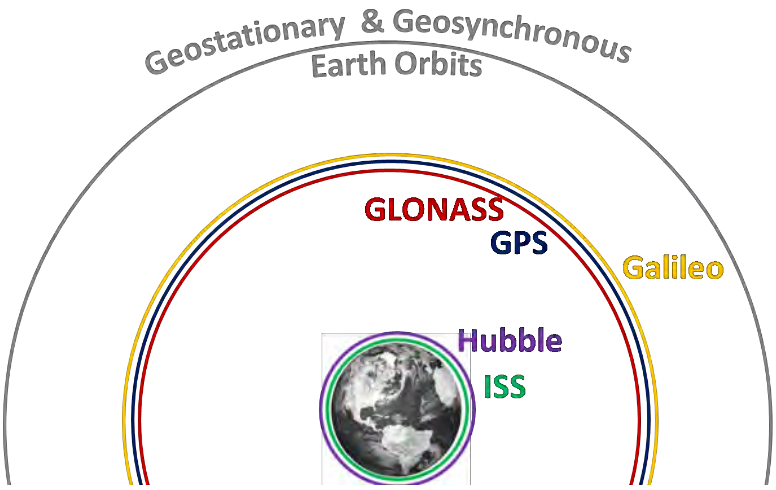


Figure 7: Orbit type locations

The shape, size and orientation of an orbit would determine the relative position of the Sun, the Satellite and other celestial body that may interfere in between them. The duration and the frequency of the eclipses will determine the power subsystem of the satellite, since the batteries installed must be able to supply the required power during those periods.

Eclipses are mainly generated by the Earth position, but there are many other celestial bodies that may interfere like the moon. Although the sun is located at 149 millions of kilometers from the Earth, the size of the Earth generates a conical shadow that prevents the satellite from receiving the solar incidence.

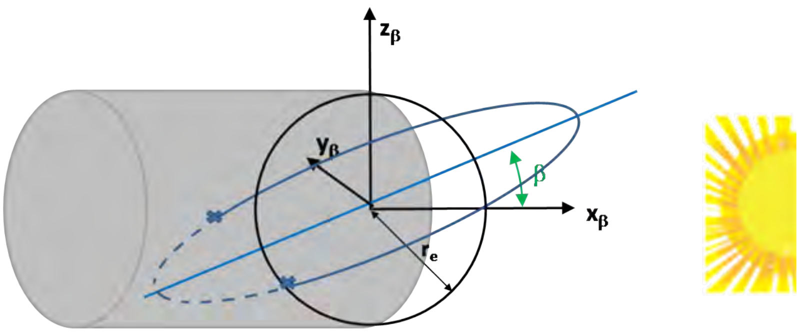


Figure 8: Satellite eclipse diagram

As it can be observed, eclipses occur at the apogee and at the perigee, but larger heights at those points will minimize the time spent in the umbral region until they finally disappear. Although the velocity at the apogee is considerably smaller than at the perigee the orbit segment shadowed at the apogee would be much smaller as well, therefore both eclipses would be similar.

It is important to notice that the angle formed between the ecliptic and the orbit plane β determines significantly the duration of the eclipse. The smaller the angle, the longer the segment of the orbit included in the shadow cone, hence the longest eclipses will be achieved at inclinations of 23.4° ($\beta = 0$). The eclipse does not reduce linearly, but with the cosecant due to the sphericity of the geometry as it can be observed in figure 10. [12]

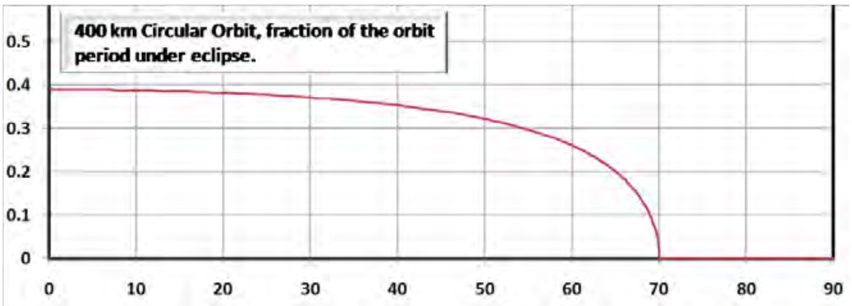


Figure 9: Variation of the eclipse duration with the orbit inclination

However the eclipses vary considerably throughout the year due to two main factors that modify the β angle:

- The seasonal variation of the solar vector (the relative position between the Earth and the Sun is not constant but varies seasonally).
- The perturbation of the orbit, mainly due to the oblateness of the planet.

Apart from the eclipses at the apogee and perigee, the satellite will be eclipsed as well when the Earth translation locates the orbit nodes in the shadow cone, or when other planetary bodies intersect the sun flux.

2.4.2 *Propulsion system*

Satellite maneuvers are performed by the propulsion system. In order to modify the operating orbit high powerful impulse systems are required, while smaller and lighter versions are used to correct the position and the orientation of a satellite. All the propulsion systems are based in the same principle: they generate thrust by exerting a certain amount of stored mass through a nozzle that accelerates the flux to maximize the impulse achieved. The momentum associated to the mass leaving the control volume will generate a force acting towards the thruster, thus impulsing the spacecraft in the opposite direction.

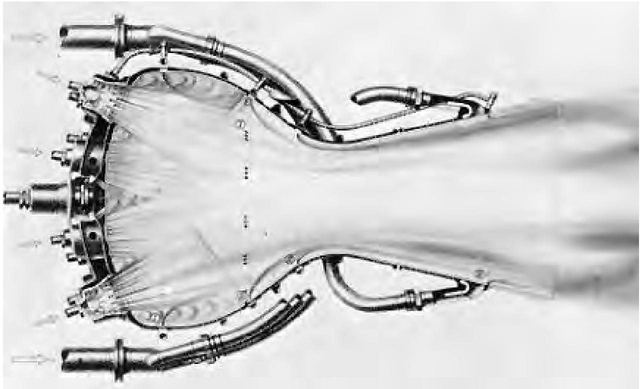


Figure 10: Control volume depicting thrust developed in a rocket

Nowadays there are basically two types of propulsion systems: chemical and electric. Chemical propulsion systems are the traditional ones and they have been widely used in the space sector. There are basically three types of chemical propulsion systems:

- Cold gas thrusters: The most basic ones, exerts the mass storage without any type of combustion, propelled exclusively by the pressure at which it is stored. The most common propellants are GN2 and Xe, they are able to develop thrust around $50mN$ (some models achieve up to $3N$) with a specific impulse of $70s$. Due to their performance characteristics they are mainly used to attitude control and orbit maintenance.
- Liquid rockets engines: Characterized by wide range of nominal thrust (1 to $10.000N$) and medium specific impulse ($200 - 450s$), they can be used for orbit maintenance, maneuvering and orbit insertion. They can use two types of propellants:
  - Monopropellant: Achieve an exothermal decomposition of the propellant using a catalytic bed, resulting in a heated, high pressure gases that are expanded through the nozzle and generate thrust. They are much simpler than the bipropellant thrusters but achieve a worse performance, thus the nominal thrust and the specific impulse are not enough to perform orbit insertion maneuvers.
  - Bipropellant: achieve the exothermic reaction by mixing a fuel and an oxidizer that will react spontaneously (hypergolic systems) or as a result of an ignition. These systems are heavier and more complex than the monopropellant but they provide an improved performance.
- Solid rockets engines: A solid mix is forced into a continuous combustion by an igniter. The main advantage with respect to the liquid rockets is their simplicity; they do not require moving parts, regulators or feed systems. Thus the costs and the mass are noticeable reduced. Furthermore, the higher density allows reducing the total size. They provide a wide thrust range from 1N to 1MN and a specific impulse around 300s, thus they are able to perform orbit insertion maneuvers. However, since the combustion cannot be properly controlled they are not used for orbit maintenance or maneuvering.

The relevance of the electric propulsion systems has increased during the last decades. The main advantage is that, unlike chemical rockets that are limited to intrinsic molecular bond energy, the electric propulsion can develop an specific impulse up to 20.000s, although it is still confined to low thrust applications due to the limitations on the on-board power. The electric propulsion systems are classified in three main categories:

- Electrothermal: where propellants are electrically heated and then expanded through a nozzle to generate thrust. There are several energy deposition methods to accelerate the propellant: the simplest one by using a resistive heater (*resistojet*), or using a sudden electric discharge between two plates (*arcjets*) to more exotic methods like a laser ablative thruster.
- Electrostatic: where electric fields are used to accelerate charged particles according to Lorentz equation. In order to achieve higher thrust, heavier propellants are preferred. The most used electrostatic system is the *Hall effect thruster* that accelerate the plasma particles through several electrically charged gridded plates.
- Electromagnetic: based in the same principles as the electrostatic systems (*Lorentz Law*) but in this case the force on the charged particles is produced by the interaction of the charged particle velocity and a magnetic field. Nowadays electromagnetic thruster are not extensively used but there are several systems available like the *pulsed plasma thruster* and the *magnetoplasmadynamic thruster*.

2.4.3 *Power subsystem*

The power subsystem is characterized by four elements: the power source, the energy storage, the Power Regulation and control and the power distribution.

The most common power source of a satellite is the solar arrays, which provides the required power during the sunlight period by mean of their photovoltaic cells. Solar cells are well-know and reliable, and they have proven to withstand the space environment degradation during the whole mission life. The main drawback related to this technology is the limited power that it can provide; therefore they are only useful for low-power missions (usually less than 15kW). Furthermore, photovoltaic cells are not attractive for interplanetary missions since the solar radiation reduces with the square of the distance. For all these missions there are other technologies available, the most common alternative is nuclear power reactor that it is able to provide a continuously high power supply.

When the power source cannot supply the required power to the satellite, the power is obtained from the energy storage systems. Those missions that relay on solar array power source require a system to store energy for eclipse periods or peak-power demands. They also provide the back-up power and even all the power for low missions (less than 1 week [8]). The energy is stored in chemical batteries, whose size, technology and number will depend on the satellite power requirements and the mission characteristics.

The power regulation, control and distribution systems require a detail design. Satellite components operates at different conditions and therefore it is necessary to control for each component the appropriate voltage range, current, acceptable normal transient time duration and voltage ripple. Many techniques are used to regulate and control the power going into a component, and they will depend on the component, the power source, the energy storage system, etc. For example, it is very common that the output voltage from the power source differs significantly from the battery output one; consequently when leaving the eclipse region and the solar arrays start generating power the system must exchange the power origin system, readapt towards the new voltage and control the transient time that may be associated.

Regarding the solar array power sources there are two main power bus control techniques: the peak-power tracker and the direct-energy-transfer. The main difference is in the way they deal with the solar input and output power.

2.4.4 *Communication system*

The communication system is one of the most important systems since every single mission would have implemented a communication function to transfer data to a ground station in more or least extent. Satellites are tracked from the ground to control their position and correct possible deviations. To guarantee the tracking process, the satellite is provided with a telemetry, tracking & command subsystem (TT&C) that collects the current satellite mission data and radiometric tracking and transmits towards the required station through a downlink channel. The mission data includes both the spacecraft engineering data (called *housekeeping data*) and the sensor/instrument data generated by the payload. It also allows the spacecraft to receive commands from the ground station via the uplink channel. The principal components that size the TT&C subsystem are the spacecraft power amplifier and high gain antenna; those elements would determine the bus power, mass data return and link margins of the subsystem.

The TT&C subsystem components are sized to overcome the losses between the satellite and the receiver and to guarantee that the signal received fulfill the mission communication requirements: the data rate and the maximum bit error rate (BER) allowed. The communication losses depend on every component that takes place in the process: the transmitter losses, atmospheric losses, space losses and receiver losses; but also the transmitter and the receiver antenna incorporate losses by mean of the working temperature at which they operate (those losses are called *Temperature noise*). The range between the ground station and the satellite is set by the nature of the mission and the technical characteristics of the ground station are generally fixed and do not allow any modification. Therefore, as mentioned before, the communication link would overcome the losses if the satellite antenna and the transmission power are properly designed. The antenna includes a number of components such as feeds and reflectors that concentrate the energy in a given direction and therefore amplify the signal. This improvement is given by the antenna *Gain* that includes both the amplification factor and the antenna internal losses.

For communication satellite missions, the spacecraft has other communication systems apart from the one required for the TT&C. These systems are intended to exchange continuously data between different ground stations, different users or different relay satellites. This configuration is called the *communication architecture* and it will determine together with the mission requirements the communication subsystem design. The communication systems are designed in the same way as the TT&C but considering higher data rates and powerful antennas.

2.4.5 *Orbit perturbation and ADCS systems*

During a mission design process, it is essential to assess an orbital perturbation analysis to determine the variation of the orbit with respect to the nominal one since it will determine the number and the size of different control components required to correct the orientation or the position perturbations. Perturbations are generated by external forces that act on the satellite apart from the uniformly centered force considered during the keplerian orbit design. Depending on its nature, these perturbations can be considered as:

- *Secular variations*: Linear variation in the model.
- *Short-period variations*: Periodic variations with a period smaller than the orbital one.
- *Long-period variations*: When the period is greater than the orbital one.

Some of these perturbations are generated by the gravitational forces of the Sun and the Moon; and to a lesser extent by other celestial bodies. They are also generated due to non-spherical Earth, the atmospheric drag, the solar radiation, the variable Earth's magnetic field and the orbital debris that orbit the Earth space specially at LEO heights.

Based on the classical fundamental equations of motion for rotational dynamics, these perturbation forces exerted on the spacecraft will generate a torque on the spacecraft. This disturbance torque can be split in three torque contributions on each of the main directions of the reference frame attached to the satellite: velocity (*roll*) moment, Y (*pitch*) moment and nadir (*yaw*) moment. The estimation of these torques often requires the use of geometrical averaging, i.e. to compute the *centroid* of the main spacecraft elements so the perturbation force could be modeled to be exerted at those points. Doing so, the perturbation problem will be simplified to compute the torques considering the distance from their *centroids* to the center of mass of the satellite.

The major disturbances will depend on the orbit location. Solar and Moon perturbation effects can be negligible in LEO missions where the gravity and drag losses are highly noticeable. On the contrary, at higher altitudes the drag contribution is almost inexistent while other planetary disturbances start to affect the orbit. Usual Earth orbiting missions are affected by four main perturbation sources: The solar radiation pressure, the atmospheric drag, the magnetic field and the gravity-gradient.

Once the magnitude and the periodicity of the main torques have been modeled, the satellite design is incorporated with Attitude and Control systems to counteract undesirable spacecraft behavior. There are several control methods implemented nowadays in space missions:

- Passive control techniques, like the gravity-gradient control uses the inertia properties of the vehicle elements to keep it pointed towards the Earth based on the fact that objects tend to align its longitudinal axis through the Earth's center.

- Spin control techniques, like the spin stabilization where the satellite rotates so keeping fixed its angular momentum vector in inertial space.

- 3-axis control techniques, able to stabilized the satellite in the three axes providing more stable and accurate maneuver depending on the sensors and the actuators installed. Nowadays is one of the most used techniques, although it raises considerably the cost and the complexity of the ADCS system.

In order to control and modify the effect of these control methods two types of elements must be incorporated to the design: sensors and actuators. Currently there are numerous types of these elements and the selection process will depend on the mission characteristics. Regarding the sensors, the most used models are gyroscopes, sun/star/earth sensors and magnetometers. On the other hand, the most used actuators to modify the spacecraft behavior are the momentum wheels, the electromagnets and the thrusters.

2.4.6 *Launch system*

The launch vehicle is one of the most expensive elements of a space mission; it represents approximately one third of the total cost. According to the *Tsiolkovsky equation*, the total velocity impulse that must be developed by the launch vehicle will only depend on the satellite mass and the specific impulse (an intrinsic parameter of the vehicle design).

Nowadays the development of a launching system is not longer part of the space mission design but the current market provides several launch vehicles available for a wide variety of missions. The selection process is focus on balancing the performance, availability, risk and cost of each model.

It is important to notice that current launch vehicles are not capable to deliver satellites in MEO or GSO orbits. The best models only develop enough impulse to reach high LEO altitudes and then deliver the payload into a Geosynchronous Transfer Orbit, thus any space mission apart from LEO mission will require a propulsion system to end the maneuver.

2.4.7 *Thermal system*

Every component installed on the satellite has a maximum temperature range at which it may operate. Many elements performance are subjected to progressive temperature deterioration and in other cases, their properties are significantly reduced at high or very low temperatures. Consequently, the spacecraft thermal analysis becomes crucial to determine the deterioration rate of the different components and therefore the on-orbit mission life.

The thermal problem corresponds to a radiant energy heat balance in space. The spacecraft receive an incident radiation from the sun and the radiation reflected and irradiated by the Earth. This incident heat is added to the internal heat generated by the internal satellite components during their operations. At the same time, the satellite irradiates energy to the space. The difference between the heat extracted and the heat received would raise or decrease the total satellite temperature.

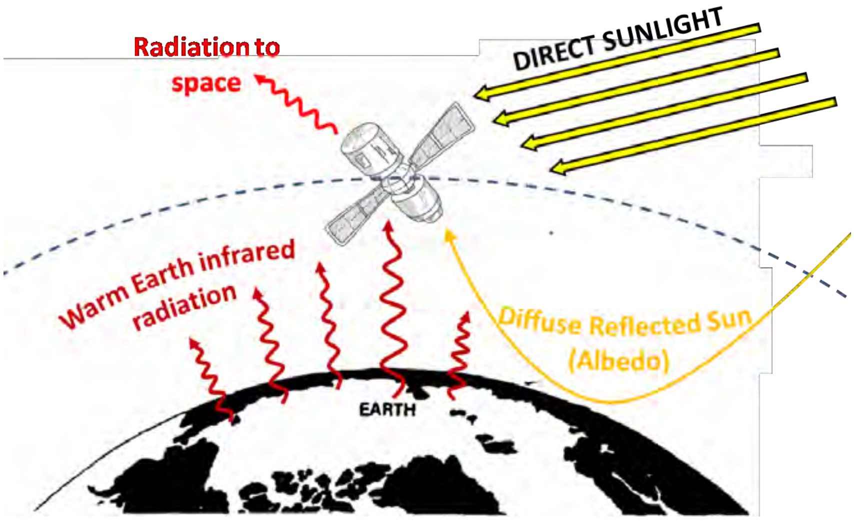


Figure 11: Incident energy on spacecraft surfaces

The incidence heat would depend on the total effective area pointed towards the heat source as well as the thermal properties of this surface. Meanwhile the total heat irradiated by the satellite would depend on the total irradiation area, its thermal properties and its current temperature.

Notice that these conditions vary widely throughout the mission. Firstly, the solar radiation flux varies seasonally, depending on the distance between the Sun and the Earth and the thermal conditions at the sun surface (which varies periodically). Secondly, the energy flux reflected by the Earth is highly influenced by the clouds and other meteorological conditions of the areas located below the satellite. Thirdly, the satellite power operation may be variable in some missions, thus the amount of internal heat generated would be variable as well. Finally, the incidence heat will vary significantly according to the eclipse intervals. When the satellite enters the eclipsed region the incidence heat is suddenly reduced and the temperature starts to decrease fast, reaching the minimum conditions right after leaving the eclipse region. When the satellite receives again the sunlight the incidence heat rises suddenly and therefore the satellite temperature starts to increase. The maximum temperature conditions are reached right before entering again in the eclipse region.

Consequently, by analyzing the lower and the maximum conditions the operating temperature range can be estimated. Based on this estimation, the satellite design may have certain thermal control elements incorporated to modify the temperature range if needed like surface finishes to modify the thermal properties, insulation layers that shield certain components and conduction isolators, radiators, heaters, louvers and heat pipes to provide a wider control over the working temperature.

2.4.8 *Satellite Structure*

The structure of a spacecraft is one of the most complex design processes. The structure must contain all the satellite subsystems and protect the most critical elements from the hazards and space environment. Moreover it must allow the relative movement of certain subsystems preventing from damage other elements as well as withstand all the loads developed during the mission, particularly during the launch. In general terms, it must guarantee the nominal operation of every subsystem during the whole life of the mission.

Therefore the satellite structure is highly conditioned by the rest of the subsystems designs and the components that they require. Consequently, once all the main subsystems have been designed it is possible to establish the structural requirements and constrains necessary. They are related with the loads and the environment, the stiffness, the design and strength criteria, performance, accommodation and mass properties.

Mission performance features like the pointing and stability, grounding, range identification and the orbit parameters will influence in a great extent the final structure required. For example, the mission life would determine the number of cycles that must withstand the structural elements; while the orbit will determine the environmental hazards that the structure must withstand.

The environment characteristics would determine several structural requirements according to the concerns about the thermal, the radiation and the launch subsystems.

- Thermal environmental requirements: The number of components and their position will determine in turn the size and the position of the radiators. Furthermore the operational temperature range may induce to the structural elements thermally stresses, stability issues and thermal expansion/contraction. They may require to select special materials in certain elements or to provide certain areas with a thermal shielding.
- Radiation requirements: The amount of radiation dictates the amount of radiation shielding required to protect sensible electronic components.
- Launch vehicle: As commented, the most critical loads are achieved during the launch due to the local accelerations, the acoustic resonances and the shock associated o the separations. The structural elements must be designed accordingly.

Regarding the interfaces between the subsystems it is possible to find numerous structural constrains related to the alignment between items and systems deployments.

Moreover, apart from the main structure that creates the principal frame and all the secondary structures required to hold, deploy and displace the different objects, the structural design must take into account all the mechanical requirements associated to the mounting. Details like the area, the location, the fasteners, interface loads, stability, grounding, material compatibility, mass, moment of inertia, calibration, etc.

The final structure must be designed to fulfill all these requirements by performing dynamic analysis and trying to optimize the final mass.

Part II

THE SW SUITE DEVELOPMENT

The following chapters comprise all the information regarding the software development. They include a justification of the design, an explanation of the mathematical models implemented and finally a software tool testing by modeling an already successful satellite mission, analyzing the divergence of the predicted results with respect to the real ones.

3 TOOL DESIGN

This chapter contains a fully detailed compilation of the information about the software developed throughout the thesis. It is divided in three sections: the requirements that must be achieved, discussion of the decisions taken during the development and finally a scope of the software architecture.

3.1 DESIGN REQUIREMENTS

The main objective of the present thesis is the development of an efficient software suite that supports the evaluation decisions taken during a mission analysis.

As mentioned in the introduction chapter, it was decided to create a visual and easy-handle interface, able to produce fast computations in order to be feasible the use during a customer meeting. Accordingly, a set of fundamental requirements were listed and it served as a useful guide during the software development and facilitated subsequent revisions. The software suite was intended to fulfill the following requirements:

- Easy-handle interface, to allow an ergonomic trade-off between the different alternatives. It should include a graphical visualization of the total budgets and the positive or negative clearance, to orient the trade-off during the design of the different subsystems.
- Fast response, as explained above. Thus the mathematical model implemented cannot be too complex to avoid being an excessive time-consuming process.
- High degree of flexibility, to allow the user to decide whether include certain reference computations, introduce any extra values to roughly evaluate the presence of unusual elements, change any parameter set by default and have the choice to select among many different satellite architecture. The more flexible, the highest the number of missions where the software can be used.
- Great versatility, avoiding excessively specific configurations. In relation to the previous points, it would increase the range of situation where the software tool could be used and it would guarantee the simplicity of the interface to be useful in a meeting.
- Reliable results, the mathematical models must be sufficiently good to become a feasible tool. However, as it will be detailed in the next section, the accuracy of the results would depend in a great extent on the user inputs.
- Broad portability, to guarantee a high user-friendly tool. The tool is intended to be launched in the largest possible number of OS platforms (Windows,

Mac, Linux,. . .) and they should not required to install any specific program like Matlab.

- Easy Data Handles. The user must have access to the entire configuration that defines the mission; but he should also have all the relevant information summarized in a file to provide a straightforward access to the mission characteristics.

3.2 DESIGN JUSTIFICATION

Space Mission Analysis is a wide field of study. There are hundreds of possibilities to be covered in each subsystem and the number of parameters involved makes the process extremely complex. Nowadays there is an extensive literature that covers in detail numerous procedures, elements and the advantages of each configuration. Thus the scale of the project would vary widely depending on the level of detail.

Accordingly, the present project has been sized taking into account both the requirements exposed in the previous section and the time available to develop the project.

3.2.1 *General considerations*

In order to guarantee a fast performance during a meeting, the software has been loaded with a fully operative LEO mission by default. The main advantage is that the software would be able to compute a solution at any moment even if the user, consciously or not, does not design some subsystems. Otherwise he would be forced to fulfill all the input requirements or the software suite would break with an error; an impractical process if it is necessary to search constantly the missing inputs.

Furthermore, it also allows the user to consider the default mission as an initial state of the project during the customer meeting and the modifications are subsequently introduced.

The default state is doubly useful since every time the user introduces an incorrect input, the software returns to the default value and display an error message depending on the type of input:

- "*Only a number input is allowed*", when the input contains non-numeric characters.
- "*<Parameter> should be positive*", "*<Parameter> should be <range>*", etc; when the input is out of the expected range. It would depend on the type of parameter that it is being modeled: efficiency, radius, extra mass, etc.

Secondly, in order to prevent the software to hang in an endless loop, a hierarchical structure has been imposed to the opened windows; thus the user must close the present subsystem design window in order to return to the main suite interface or before opening the next one.

3.2.2 *Budgets modeling*

The number of budgets to be modeled was reduced to three: mass, cost and power. The time constraint made unprofitable to analyze budgets such as the risk, the launch windows, the schedule or the manufacturing time due to their complexity and the minor relevance compare to other budgets.

Every system has an impact in terms of money and mass. The total mass of the satellite is one of the most relevant budgets during a design; it would determine the launch system required for the mission, thus a great percentage of the final cost.

On the other hand, although the total cost of the project is ultimately the most relevant parameter, the hermeticism of the space industry made impossible an assessment model and nearly all the costs must be roughly estimated from outdated systems. Consequently, the accuracy of the results is not sufficiently good and besides it has been finally implemented, the user must be aware that the result should be considered as a roughly estimation.

Moreover, since many systems are driven by the total power and considering the weight that the power subsystem has on the satellite mass, it was considered to include a power budget analysis to enhance the trade-off between the subsystems.

Notwithstanding the fact that the satellite size driver was considered as relevant as the mass, it could not be ultimately implemented due to the impossibility to include a section in the software to visually positioned each element and generate a final structure. It could not just add the individual volumes of each element since the complex geometry, the initial position and the deployment after the launch is a key procedure during the satellite design and must be positioned carefully.

3.2.3 *General mission design*

Despite the wide variety of space missions, the software suite was designed to be focus on satellite missions orbiting the earth, since those represent most of the sector: communication, navigation, earth observation and many military and scientific satellites.

Despite the similarity of these missions with a satellite orbiting other space objects, specially the planets nearest to the Earth, the configuration adds an extra complexity that it was impossible to be implemented in time. The main differences were the communication link (that would depend on the relative position between the planets) and the transference orbit to reach the target.

On the contrary, deep-space missions have many differences with respect to the earth orbiting satellites: variable communication distance, constant tracking position, strictly schedule and the propulsion and the orientation systems play fundamental roles. The implementation with the orbital-satellites became too complex to be developed in time. Additionally, the commercial information about the technology implemented in these kinds of missions is much less available and so the accuracy of the results would not be as good as earth orbiting satellites ones.

Accordingly, it was more efficient to focus on Earth-orbiting satellites taking into account the time required implementing the other two cases and the low percentage of space missions that they would be able to model.

3.2.4 *Subsystems design*

The satellite subsystems have been designed carefully since they were particularly affected by the requirements: the way they are modeled plays a fundamental role to provide flexibility, speed, accuracy and a generic nature. They also would determine the user friendly level of the software suite.

The decisions adopted in the previous subsections determine in a great extent the subsystems to be modeled. Certain subsystem drives some budgets more than others; consequently, for the present project, the most relevant ones are those that influence the mass, the cost and the power. Similarly, the role that a subsystem plays will depend on the type of mission: e.g. the propulsion system is essential during a deep-space mission to correct the course and perform maneuvers, while for a satellite orbiting the earth it is only used to correct the orbit perturbations (or to transfer the satellite in/out the orbit at the beginning/end of life).

Finally, six subsystems were implemented: Power, Communication, Structure, ADCS Thermal and Launch Systems. Other subsystems like the Propulsion and the Telemetry Subsystems were implemented inside the Launch and the Communication subsystems respectively since they are highly related and their impact is less relevant. In this way, the software interface becomes clearer, enhancing the trade-off process.

Unfortunately, the thermal and structural subsystems were finally implemented following very simple models due to the fact both are highly related with the geometry and relative position of the satellite elements and, in relation with the discussion of the size budget, their modeling became highly limited.

There were another two subsystems that in principle were going to be included but ultimately the complexity added to the interface made it impossible:

- Ground Segment. The ground coverage becomes one of the main drivers during the communication and the orbit design. However, due to the extra complexity that requires the computation of the coverage and the variable communication distance the segment was replaced by the following assumption: the satellite is always inside one ground station coverage range and the communication distance is fixed and corresponds to the orbit height at the apogee.

- The perturbation modeling. The perturbation modeling determines the importance that the propulsion and the attitude control systems will have during the mission, thus the final mass and power consumption that will be associated to them. However, it was finally not implemented considering the relative low impact that it has in earth orbiting satellites and the complexity that it involves.

On the contrary, the constellation design was discarded at the very beginning: having an enormous influence on the communication and the ground segment design, the complexity added to the project was unmanageable. In this way, the cost associated to a constellation design would not be properly estimated by the software suite; but it will be still useful to analyze one of those satellites individually.

Finally, the payload would be included as if it were another subsystem that contributes in terms of mass, cost and power to the satellite design.

3.2.5 *Development environment*

Considering the nature of the project, it was possible to be developed in several environments: Visual Basic or similar C-based programs, Matlab, Java or Python. At the beginning of the project, all the options were considered. The most interesting ones were those that provide an easy way to program an interface and an effortless way to refresh the results.

In principle, Java should be the best choice: it is easier to deal with the different classes and it would generate a lighter program. However, Matlab and VisualB provide more intuitive software to develop interfaces. In addition, Python was a suitable option as well due to the similarities with Matlab. Finally, it was decided to develop the space software suite in Matlab, considering that the student was already deeply familiar with it.

3.3 ARCHITECTURAL DESIGN

The software architecture had to be planned in advance since it would determine in a large extent the flexibility and particularly the handling of the software suite. The input concepts must be clearly established, the design processes must be easy to carry on and the general interface of the software must help the user to proceed with the trade-off process in a fast and easy way.

The best way to fulfill all these conditions is by implementing a central architecture; i.e. the tool has a principal window, the *Main Suite*, from where the user calls the rest of the functions to define the satellite subsystems and returns once he finished. A sketch of the software architecture is provided in figure 12.

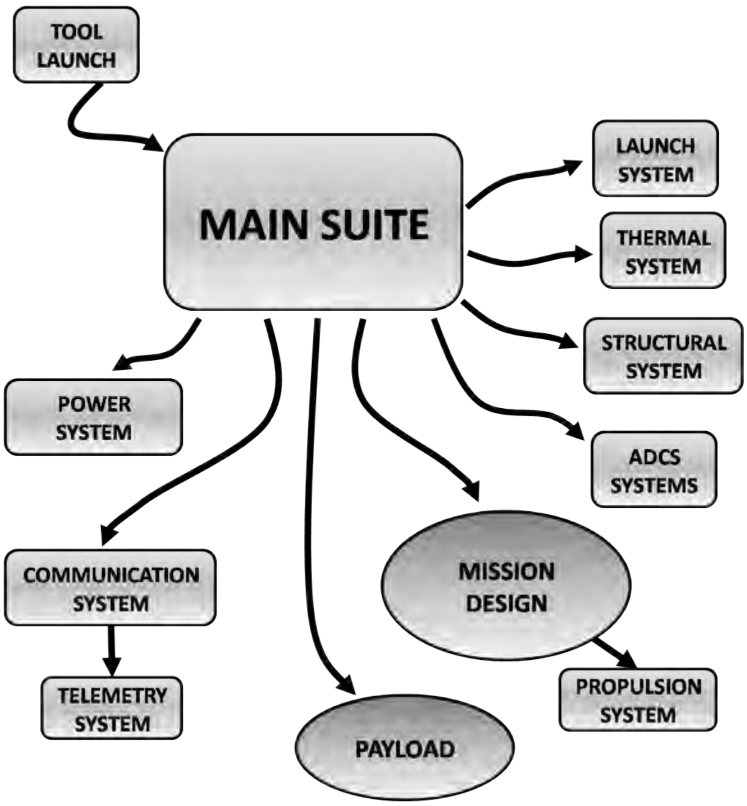


Figure 12: Tool arquitecture sketch.

3.3.1 *Launching the tool*

Firstly, in order to fulfill the tool portability and handling requirements, the program would be launched by an executable file. Thus it could be used in any windows environment even without a Matlab License installed.

Secondly, to avoid the user entering straightforward to the main suite window, it was decided to include a previous window to introduce the software. In this way, a nice and clear interface is obtained and the user experience is improved. As it can be observed in figure 13, the window introduces the function of the tool, the author, the date and important warnings that the user must be aware of. It also allows the user to start a *new mission* (from the default state) or a previous one by loading a file.

Figure 13: Initial tool window.

3.3.2 *Main Suite*

Similarly, to achieve a user-friendly interface, the main software suite was planned following the design sketched below:

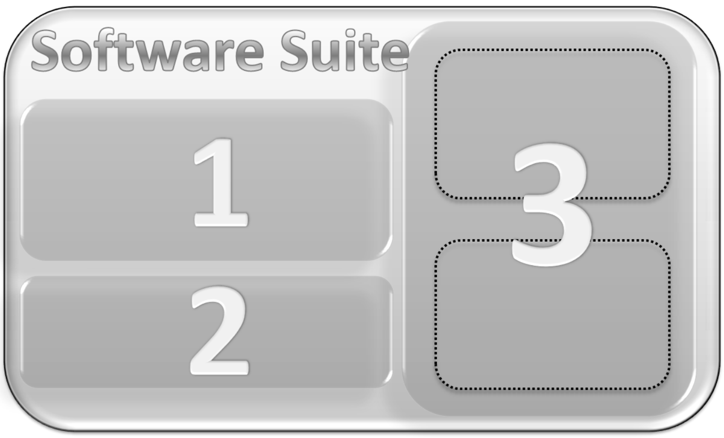


Figure 14: Main suite arquitecture design.

As it can be notice, the main window was divided in three sections:

- The first one would contain all the satellite subsystems that could be modeled. They would be presented as buttons that would lead the user to a secondary window where the subsystem would be modeled in detail. At any moment, the user would be able to return to the main suite to model another subsystem or check the budget and graphics.
- The second subsection would enclose the graphical settings. It would include options to change the budget displayed, the type of graphics and it would also include a section to introduce the project budgets as an input that would be displayed as a graphical reference in the plots.
- The third section would display the different graphics, according to the inputs introduced in the other sections.

It is worthy to notice the importance of having a full visibility of the budgets graphics during the subsystem design, since it would facilitate the trade-off process if it is used as a visual reference. Consequently, the additional windows that would be opened must be constrained in the left hand side of the suite, leaving the third section completely uncovered.

The software would be able to refresh automatically the graphics displayed every time a subsystem window is closed or a graphical setting is modified; however a *Refresh* button was finally included in order to allow the user to force the software to compute the actual design state.

Moreover, as specified in the easy data access requirement, it would include a *Save button* that would create two files, a *Data file* containing all the relevant information regarding to the mission design and a *Configuration file* that would include the entire configuration. The user would be able to select the directory to save these files.

Figure 15: Main suite final design.

The final design of the main suite interface actually meets the initial idea, as it can be observed in figure 15. It allows the user to determine the mission, the payload, the six subsystems and the three relevant budgets discussed in the previous section. It also includes the Refresh and Save buttons, as well as the graphical settings to display the budgets.

The satellite budgets are displayed as a stacked bar plots, where the user is able to evaluate the different subsystems that contribute to determine the budget. The colored legend allows the user to check the total value of each subsystem. The stacked plot includes as well the limit imposed by the project budget by plotting a horizontal red line; and in case the satellite estimations overpass the limit, it turns red the portion of the stacked plot located above and displays a warning message containing the clearance.

Furthermore, the graphical options can be set to display one or two budgets in two different plots.

Figure 16: Main suite final design, double display configuration.

3.3.3 *Mission design*

The mission design is the most relevant stage since it would greatly influence the process to determine the following ones.

Although it is not a subsystem per se the general parameters of the mission would be introduced through a secondary window similar to the satellite subsystems; thus, for the sake of simplicity and henceforth, interpret the term mission subsystem accordingly.

In order to enhance the mission design, the subsystem window has been provided with a 3D graphical representation of the orbit, and the design parameters have been divided in four sections: *Orbit determination*, *Longest associated eclipse*, *Transfer from initial orbit* and *Mission parameters*.

The architecture described can be observed in the following picture:

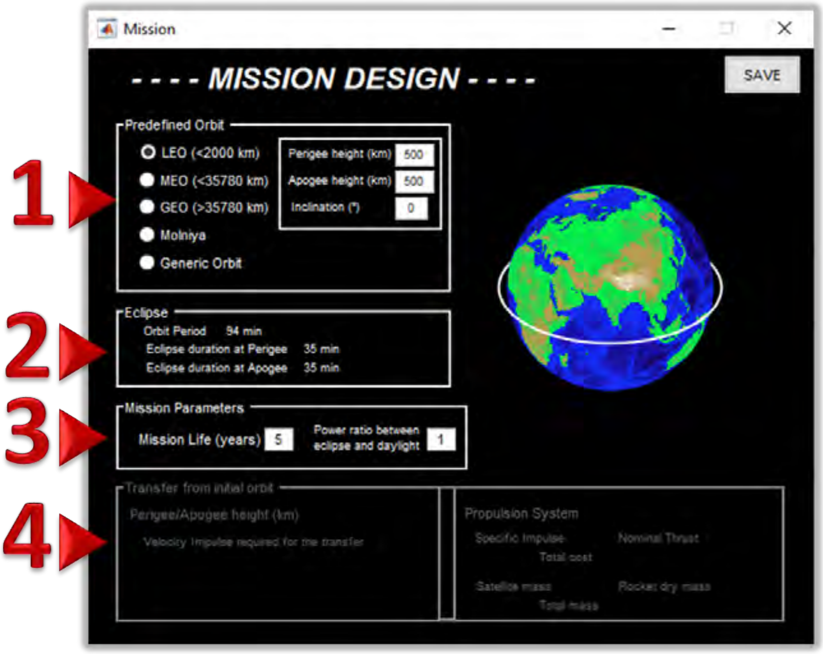


Figure 17: Mission design final architecture.

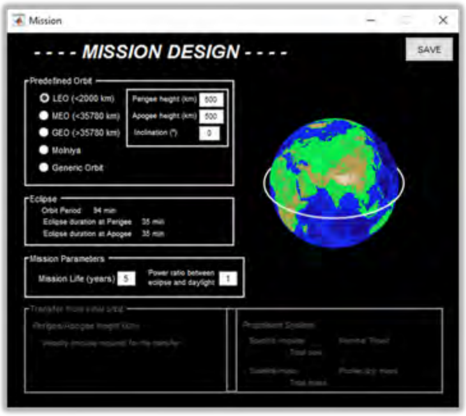
1. Firstly, the user selects the most suitable orbit type: LEO, MEO, GEO, Molniya or directly design a generic orbit in detail [1]. Then he would determine the orbit by introducing the perigee and the apogee height and the inclination angle.

2. The eclipse section would display the orbit period and the eclipse period at the apogee-perigee or at the nodes (check *Chapter IV: Models* for more details). The values are actualized every time the user modifies the orbit.

3. The user will introduce the expected mission life and the power ratio between operating during the eclipse and the daylight.

4. In the transfer orbit section the user design the transfer maneuver from the initial orbit where the rocket launcher leaves the satellite towards the final desired orbit. The user can introduce the apogee and perigee of the initial orbit, and the software would compute automatically the total velocity impulse required. It would also display the points of the initial and final orbit at which optimal transfer orbit is realized. Moreover, the user can include extra impulse to confront extra losses that he might be considering like atmospheric drag or transfer perturbations. This section is not available for LEO orbit (check *Chapter II: SMA* for more details).

1 Although this differentiation is not strictly necessary, it is a common practice in space mission analysis and it would help the user to familiarize with the design



LEO selection

MEO selection

GEO selection

Molniya selection

Figure 18: Orbit selection during the mission design

On the other hand, the user is allowed to design an orbit in detail by selecting the Generic Orbit option. Doing so, two additional buttons will appear:

- An Edit button that will open a third window to fully design the orbit.
- A check box that will allow the user to decide if its orbit requires a transfer maneuver from an initial one.

Figure 19: Extra options when *Generic Orbit* option is selected.

The third window opened when the Edit button is pressed provides the user with a strong interface to design the desired orbit. It provides with a large 3D orbit display and many useful graphical tools to design easily the orbit:

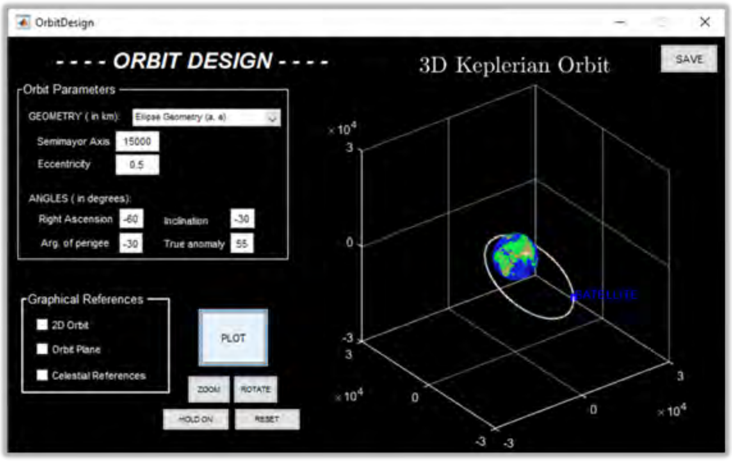


Figure 20: Orbit design window when *Edit Generic Orbit* button is pressed.

Firstly, the geometry of the orbit could be introduced in four different ways:

- As an ellipse: semimajor axis and the eccentricity.
- Introducing a pair of parameters: the perigee radius and eccentricity.
- Introducing a pair of parameters: the perigee and apogee radius.
- Introducing a pair of parameters: the perigee and apogee height.

  Then the four angles that determine the orbit and the satellite position are introduced.

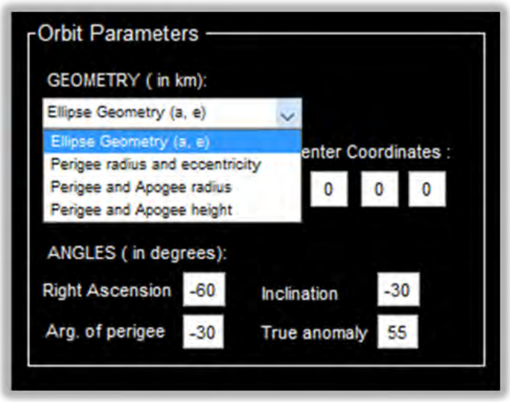
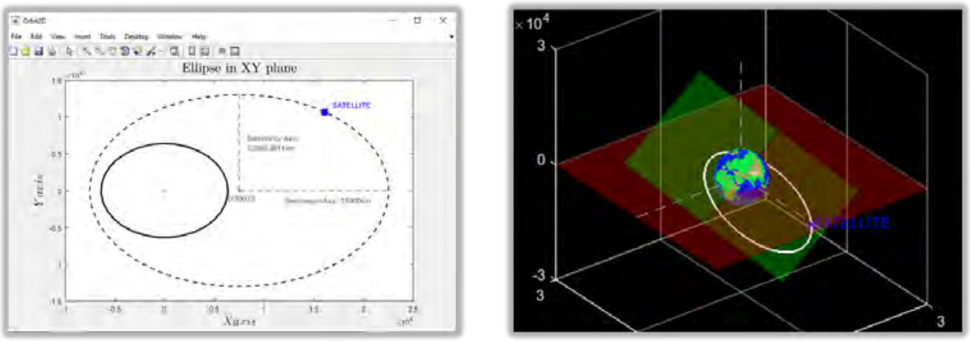


Figure 21: Geometry options available to define the orbit in *Generic Orbit* design window

Secondly, it is possible to active three graphical tools to help during the orbit design:

- 2D Orbit. It displays in an external figure a 2D sketch of the Earth and the satellite orbit. It becomes very useful to check if the relative position of the actual orbit and the Earth is correct.
- Orbit Plane. It displays in the main plot the plane were the orbit is contained. It becomes extremely useful to evaluate the 3D orientation of the orbit, particularly when the Argument of Perigee is introduced.
- Celestial References. It displays in the main plot the Celestial coordinate system (formed by the Vernal, the Polar and the right-hand proper direction). It also displays the celestial equator plane. It becomes a useful reference during the 3D orbit design.



(a) 2D Orbit (b) Plane references

Figure 22: Graphical tools during the Generic Orbit design

Finally, the interface provides a set of buttons that helps to visualize the orbit design. The user can rotate and zoom in; reset the view and also hold the actual design to be displayed in the next plot and used as a reference.

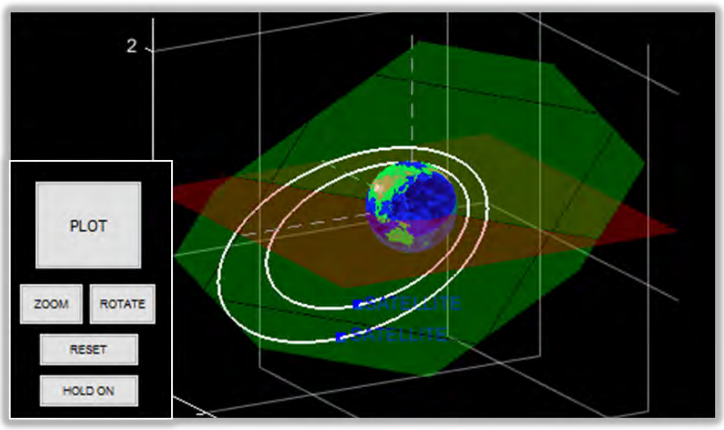


Figure 23: Visualization options available during the *Generic Orbit* design

Regarding the transfer maneuver, it was decided to include only solid and liquid rocket motors as the possible available propulsion systems. This decision was based considering that most of the orbit insertion maneuvers are performed by high impulse motors. Moreover, the current alternatives available such as certain electric propulsion systems require a continuous low impulse maneuver computations that add extra complexity to the user interface [6]. Once the velocity impulse has been computed, the user will select a propulsion system and the total cost and mass will be obtained.

Figure 24: Propulsion system design

The cost would be computed according to the specific characteristics of the propulsion system selected by the user and the total mass would be the wet mass of the satellite, hence including the propellant required to perform the impulse maneuver.

Considering that communication satellites hardly ever modify their operating orbit, the software only considers the propellant required for the transfer maneuver. However it would be possible to model roughly a future impulse maneuver by the extra velocity impulse option.

3.3.4 *Payload Design*

The payload is implemented as an extra subsystem that contributes to the total power consumption, mass and cost of the satellite mission. It was decided not to include the payload cost in the mission budget but only the cost associated to its installation.

Additionally, the user is able to include and customize an onboard data manager computer, for those missions that require a huge data handling and therefore the size and power consumption of the computer is no longer negligible.

3.3.5 *Power subsystem*

The left hand side of the power subsystem interface has been divided in three sections to design each of the main elements individually: *the solar panels*, the *secondary batteries* and the *Power transfer system*. The user is able to choose among different options (technologies, materials, etc) the one that suits better with the mission. Every option set certain values for the parameters but they are not fixed so the user will be able to modify any of them.

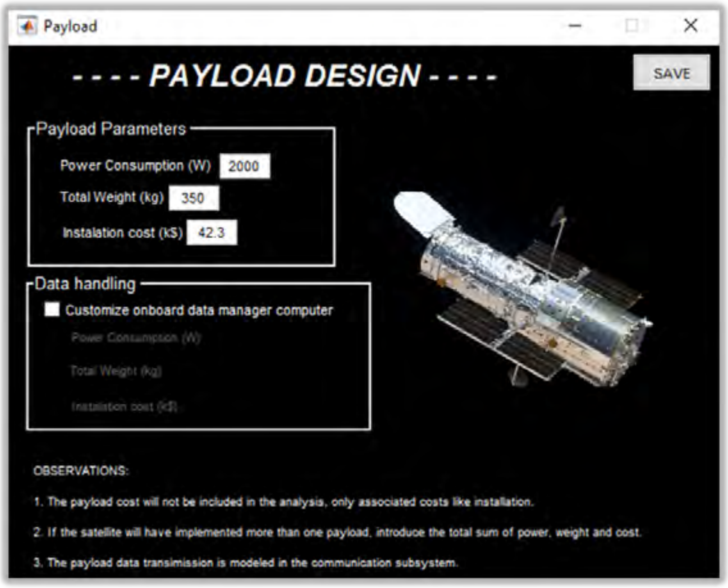


Figure 25: Payload design window

The results are displayed on the right hand side. The numeric values of the solar array area, the total weight and the total cost are shown at the top. Below them, it is possible to analyze the distribution of the total weight/cost between the four main fields that characterize the power subsystem: the power source, the energy storage, the power regulation and control and the power distribution.

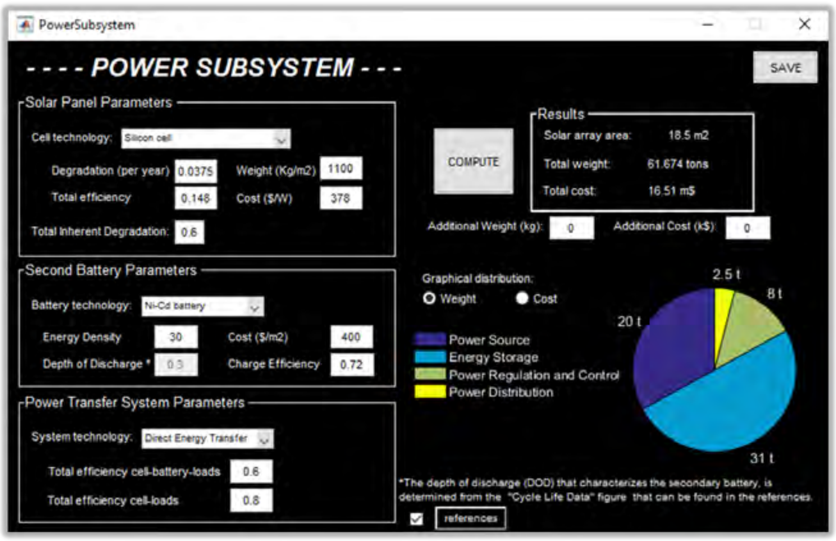


Figure 26: Power Subsystem final architecture

The user is able to introduce additional weight and cost in order to model elements or extra losses that are not included. Furthermore, by selecting the appropriate option, he can select which option, the mass or the cost distribution, will be displayed.

Finally, the user is able to check or dismiss the references used to determine the *depth of discharge* with the options provided below the distribution display.

Similarly as in the propulsion system, the only power source that has been included in the software is the solar arrays since it is the only source used in the majority of the present-day communication satellites orbiting the Earth; while other primary sources such as the *fuel cells*, *Radioisotope thermoelectric generators*, *nuclear dynamics* or *solar heat systems* are implemented only in few occasions [8].

3.3.6 *Communication Subsystem*

The communication subsystem is divided in two subsequent windows in order to make the design process clearer.

3.3.6.1 *First Window - Communication architecture selection*

In the first window, the user selects the communication architecture that he wants to be implemented. They have very similar interfaces indeed, but by preselecting the architecture, the interface of the next window would be significantly easy to handle.

There are six options available:

- Satellite transmits data to one or more ground stations.
- Satellite transmits data to one or more satellites.
- Satellite transmits data to one or more ground stations by two different types of links.
- Satellite transmits data to one or more satellite by two different types of links.
- Satellite exchanges data with one or more ground stations and with one or more satellites.
- Satellites exchanges data with ground stations and satellites by four different types of links.

Additionally, the interface has two buttons, the *Communication design* and the *Telemetry design* buttons, that will open another window to design the architecture selected or the telemetry system, respectively. In fact, the telemetry interface is the same as the first communication architecture but handling a lower data rate.

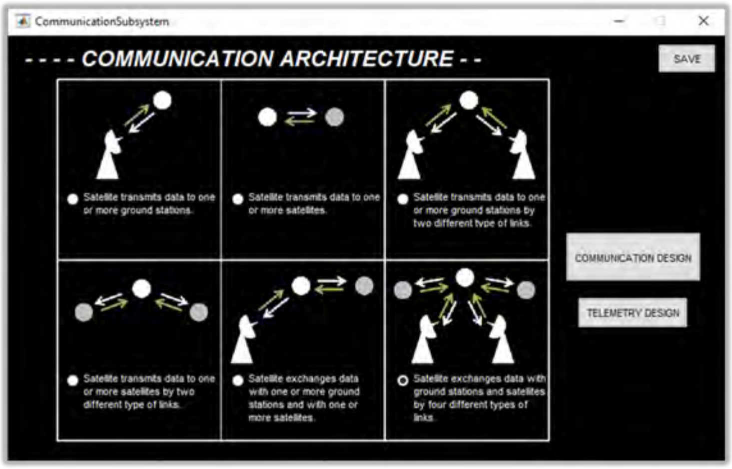


Figure 27: Communication Subsystem - architecture selection

3.3.6.2 *Second Window - Communication architecture design*

The main difference between the architectures is the number and the type of links: satellite-satellite or satellite-ground station. Each link considered in the communication architecture will be designed through a link design window (described later in another subsection); and furthermore, the user can introduce the number of times that a link is repeated, for those missions where the satellite exchange information with similar stations..

The left hand side of the interface is identical for all of them. It includes:

- The parameter to be determined: Power or Mass. Depending on the point of the trade off process, it is possible to have the power already constrained and the antenna must be size accordingly or vice versa.
- The number of times that each link is repeated.
- A sketch of the communication architecture selected.
- Extra mass/cost/losses to model certain elements that were not included.
- The Compute button to calculate the parameter desired and display the mass, cost and power results.
- Reference settings, to check or dismiss the references used to fix the Bit-energy to noise-spectral-density ratio, the link distance, the space and the atmospheric losses.

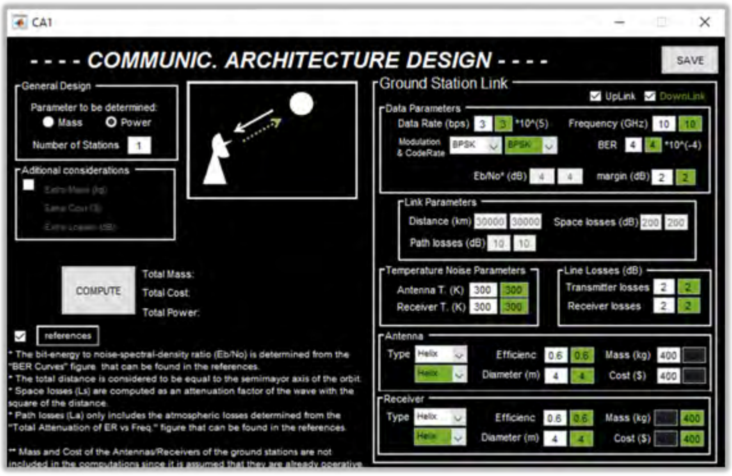


Figure 28: Communication Subsystem - architecture design

3.3.6.3 *Link design interface*

Each link is designed using the same interface, with few differences between the satellite-satellite and the satellite-ground station link.

Firstly, the user can enable or disable both the uplink and the downlink for each type of link in order to adapt the communication architecture to the mission requirements. The uplink and downlink are computed identically, except for the cost associated to those elements installed in the external station as it will be commented below.

The interface to compute each link is divided in six sections according to the most significant elements that play a role in the link balance:

- Data parameters, which will determine the bit-energy to noise-spectral-density ratio.
- The link parameters, that compute the losses associated to the path between the antenna and the receiver.
- The temperature noise parameters, that will determine the intrinsic losses associated to the use of the antenna and the receiver.
- The antenna. The user selects the type of antenna and thus he sets the antenna characteristics in order to compute the gain. Those parameters are not fixed and can be modified in all the cases. The antenna cost and mass are not considered in the downlink since it is not installed in the satellite, as commented above.
- The receiver. Similarly to the antenna, but in this case it is the receiver in the uplink the one that is not considered in terms of mass and cost.

Figure 29: Ground Link and Satellite Link interface

As it can be notice, both links are very similar. The only difference is that the satellite-ground station link has the link distance fixed, determined by the orbit geometry, and it includes atmospheric losses since the link crosses the most critical zone located near the Earth surface.

3.3.7 *ADCS Subsystem*

The *Attitude Determination and Control Systems* window allows the user to include different ADCS systems to counter the torques that are exerted on the satellite due to the Solar Radiation Pressure, the Aerodynamic Drag, the Magnetic Field and the Gravity Gradient.

The magnitude of these torques are estimated from the mission characteristics determined in other subsystems and displayed at the top of the window. The estimations can be adjusted by modifying the model parameters included on the left hand side box.

Furthermore, the user must include the maximum pointing error that the mission will have, and that will be used in fact to determine other subsystems operation.

Finally, the total mass, cost and power consumption of the subsystem would be computed according to the number of each ADCS systems included.

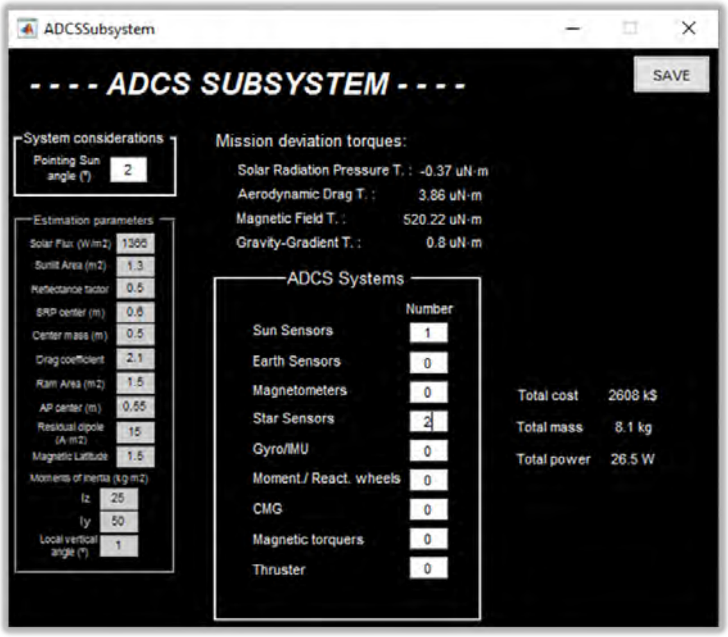


Figure 30: ADCS design window

3.3.8 *Launch Subsystem*

The launch subsystem is provided with a list of the most used launch stations throughout the globe. The user is able to select a specific location or introduce arbitrary coordinates if none of the locations suggested suit with the mission.

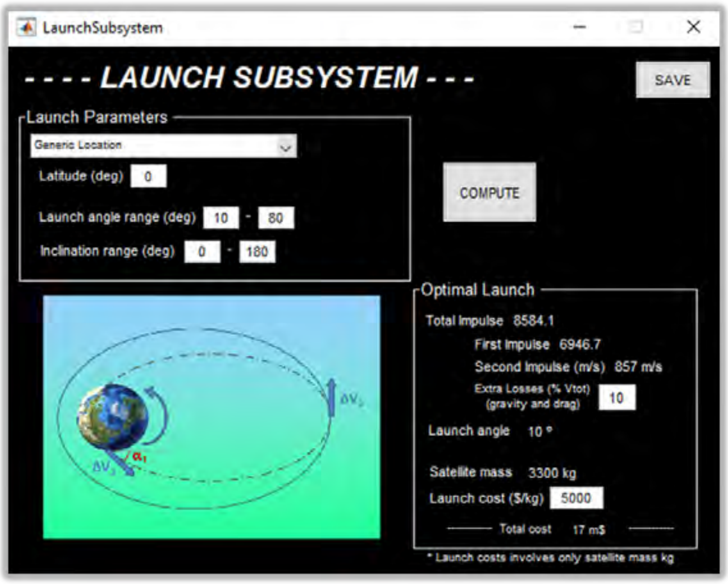


Figure 31: Launch subsystem final architectur

It is worthy to remember that the location of the launch station would influence the initial velocity of the rocket as well as the orbit inclinations attainable.

The window displays a warning message if the orbit inclination is out of the recommended range, nevertheless the user can dismiss these considerations. It is possible to modify as well the launch angles available for the launch.

By pressing the *Compute button*, the software will provide the total velocity impulse required to put the satellite into the initial orbit (determined during the mission design) or directly into the final orbit in case of a LEO mission. It also provides the initial and the final impulses required to take off from the ground or to modify the satellite path to enter into the desired final orbit, respectively. Moreover, the software indicates the orbit position in which the second impulse is performed and the launch angle and the user can introduce the extra impulse required to sustain extra losses that he might be considering.

Finally, the subsystem displays the total mass of the satellite that has been designed and it is used to compute the launch cost.

3.3.9 *Thermal and Structural Subsystem*

As explained in the previous section, the implementation of these two subsystems was strictly limited since the position of each element cannot be modeled. Consequently, since they contribute noticeable to the total mass and cost, the initial models were simplified to be able to include these subsystems and estimate their mass, cost and power consumption contributions.

The thermal subsystem was modeled similar to the ADCS subsystem. The user determines the satellite thermal parameters and the selects one of the thermal models to compute the minimum and maximum operation temperatures during the mission. In order to do so, the user must consider the configurations corresponding to those mission operations were the minimum and maximum temperatures are achieved.

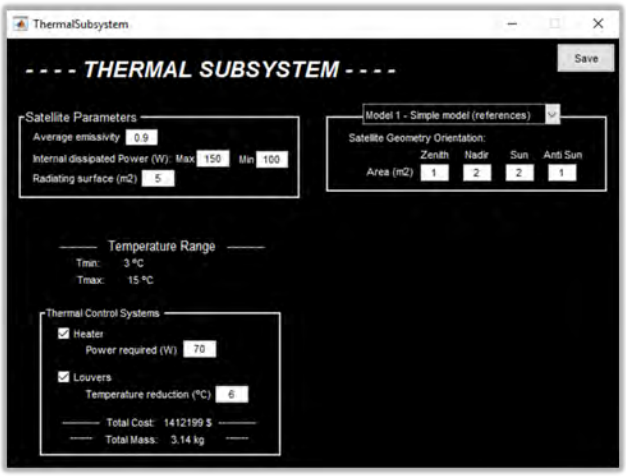


Figure 32: Thermal subsystem design window

Subsequently, the user is able to include thermal control systems to modify the initial temperature range considering the working temperatures of the elements that will be installed in the satellite. The window will compute the total mass, cost and power consumption that these control systems will add to the satellite.

On the other hand, the structural subsystem allows the user to include the percentage of the total mass that the different structural elements will represent. Thus the user must include the contribution of the primary structure, the secondary structures and the associated fasteners. Additionally, the user can include an extra amount of mass to represent any extra component that may not be considered during the design.

Subsequently, a computational cost model would be defined to compute the cost associated to the structural subsystem considering its mass.

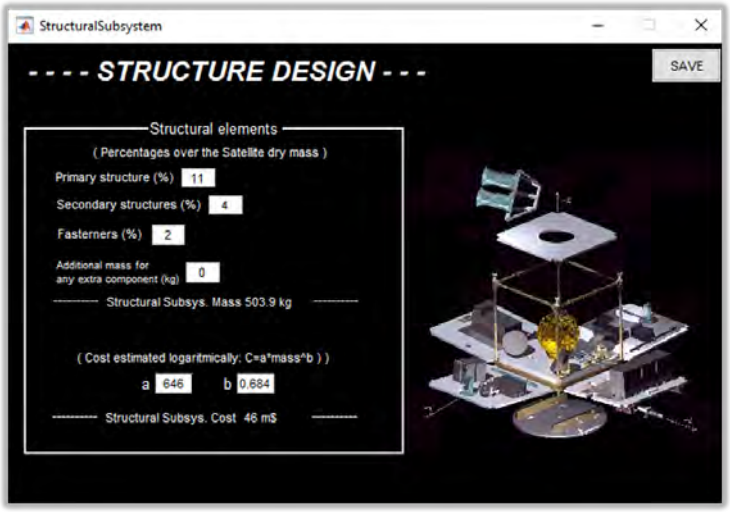


Figure 33: Structural subsystem design window

4 MODELS

The mathematical models behind the software computations are explained throughout the following sections. They collect every assumption and hypothesis that has been considered as well as every reference or tables used. It also depicts the software limitations. The chapter is divided in eight sections, each of one describing one of the subsystems that were finally implemented.

For the sake of simplicity, all the coefficients used for the mass and cost computations in each system have been summarized in the Appendix A at the end of the thesis.

4.1 MISSION DESIGN

The mission window is implemented with three mathematical models: one that defines the orbit in 3D, another one that computes the period of the orbit and the eclipses and the last one that evaluates the propulsion system required to transfer the satellite from the initial orbit where the launcher will deliver it towards the final orbit where the satellite will operate.

4.1.1 *Orbit definition model*

The orbit size and shape are evaluated as a two dimensional ellipse; hence a pair of geometric parameters will determine the full geometry, basing on the trigonometry relations that can be found in a wide literature [10]. The most relevant equations are included below:

$$2a = r_p + r_a; \qquad r_a = a(1+e); \qquad r_p = a(1-e);$$
$$f = \sqrt{a^2 - b^2}; \qquad b = a\sqrt{1-e^2}; \qquad r_{a/p} = h_{a/p} + R_E;$$

Where a is the semimajor axis; r_p is the radius of perigee; r_a is the radius of apogee; e is the eccentricity; h_p is the height at perigee; h_a is the height at apogee; f is the foci distance and b is the semiminor axis.

The orbit will be parameterized to facilitate the following orientation calculations:

$$\begin{cases} X = f + a \cdot \cos(\theta) \\ Y = b \cdot \sin(\theta) \qquad \text{where } \theta \in [0 \;\; 2\pi] \\ Z = 0 \end{cases}$$

Thus the points that determine the ellipse will be included in a 3D matrix:

$$E = \begin{bmatrix} X & Y & Z \end{bmatrix}$$

Secondly, the orbit orientation will be determined by the set of angles described in *Chapter II*. In order to perform the three subsequently reference system rotations, the rotation matrices according to the equation below [9] defined by the vector of the axis of rotation and by the rotation angle.

$$\mathbf{R} = \begin{bmatrix} \frac{u^2+(v^2+w^2)\cos\theta}{u^2+v^2+w^2} & \frac{uv(1-\cos\theta)-w\sqrt{u^2+v^2+w^2}\sin\theta}{u^2+v^2+w^2} & \frac{uw(1-\cos\theta)+v\sqrt{u^2+v^2+w^2}\sin\theta}{u^2+v^2+w^2} \\ \frac{uv(1-\cos\theta)+w\sqrt{u^2+v^2+w^2}\sin\theta}{u^2+v^2+w^2} & \frac{v^2+(u^2+w^2)\cos\theta}{u^2+v^2+w^2} & \frac{vw(1-\cos\theta)-u\sqrt{u^2+v^2+w^2}\sin\theta}{u^2+v^2+w^2} \\ \frac{uw(1-\cos\theta)-v\sqrt{u^2+v^2+w^2}\sin\theta}{u^2+v^2+w^2} & \frac{vw(1-\cos\theta)+u\sqrt{u^2+v^2+w^2}\sin\theta}{u^2+v^2+w^2} & \frac{w^2+(u^2+v^2)\cos\theta}{u^2+v^2+w^2} \end{bmatrix}$$

Where $\vec{v}$ = (u,v,w) is the axis vector and θ is the rotation angle.

On the contrary, the satellite position on the orbit is determined by the *true anomaly* ν, imposed before the rotations, as an angle between the satellite position vector and the initial X axis.

$$\vec{s} = \begin{bmatrix} f + a \cdot \cos(\nu) & b \cdot \sin(\nu) & 0 \end{bmatrix}$$
$$\vec{S} = R \cdot \vec{s}$$

As commented in the previous chapter, the orbit is maintained invariant since no perturbations have been considered.

4.1.2 *Eclipse computation model*

Firstly, the period of the orbit is computed according to the classic keplerian equation:

$$\tau = 2\pi\sqrt{\frac{a^3}{\mu}}$$

Where μ is the Earth gravitational constant.

Secondly, the maximum eclipse duration at the apogee and at the perigee are estimated considering the angular radius of the Earth at these points and the inclination of the orbit plane. Considering the geometry in Figure 34, the angular radius can be computed as:

$$\sin\rho = \frac{R_E}{R_E + h}$$

As it can be observed in figure 34, the eclipse duration varies widely depending on the angle β between the orbit and the ecliptic plane. The eclipse duration reduces with the angle as follows:

$$\cos\frac{\phi}{2} = \frac{\cos\rho}{\sin 90 - \beta}$$

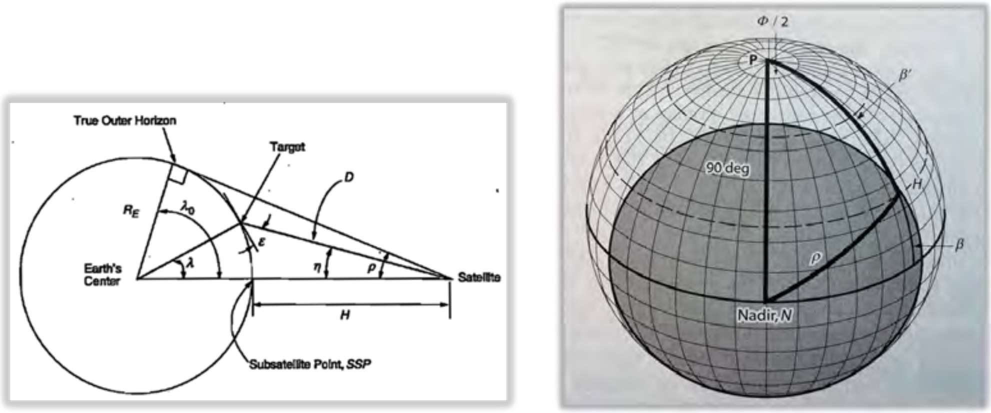


Figure 34: Eclipse trigonometric relations [8]

Once the rotation angle covered by the sun as it passes behind the disk of the Earth is computed (ϕ), it is possible to obtain the fraction of the orbit period spent in each eclipse:

$$E = \frac{\phi}{360} \cdot \frac{R_E + h}{a}$$

Finally the duration of the eclipses would be:

$$\tau_{apogee} = E_{apogee} \cdot \tau_{orbit}$$
$$\tau_{perigee} = E_{perigee} \cdot \tau_{orbit}$$

4.1.3 *Propulsion system evaluation model*

The propulsion system is evaluated taking into account the initial and the last orbit geometries defined by the user, then the total velocity impulse required to perform the Hohmann transfer maneuver between the orbits is computed.

The initial orbit is modeled as a circular one with the same orientation as the operating orbit, since it corresponds to a transition stage where the launcher leaves the satellites, thus a detailed description is not practical for this purpose. The Hohmann transfer would be constituted by two impulses, even though for GEO missions the ratio between the orbits is sufficiently large to consider a three impulse maneuver. The transfer maneuver can be observed in Figure 35.

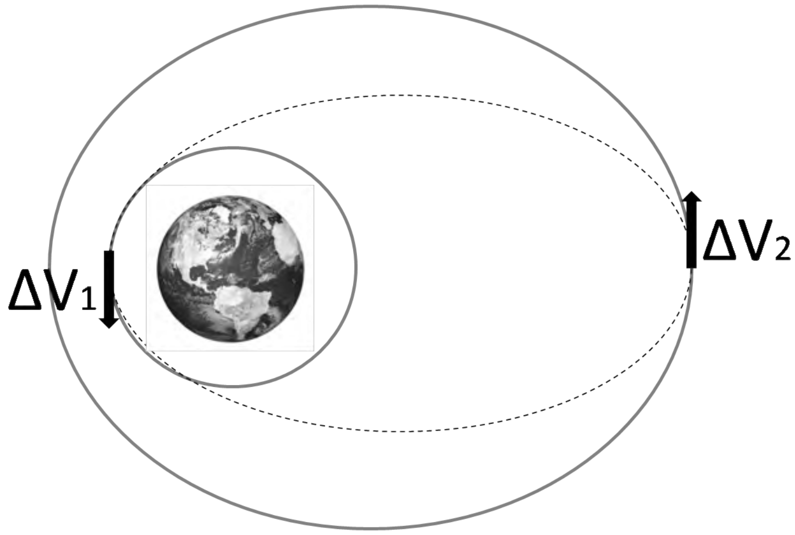


Figure 35: Hohnman transfer maneuver sketch

The initial velocity corresponds to the solution to the *vis-viva equation* for a circular orbit:

$$v_1 = \sqrt{\frac{\mu}{R_1}}$$

For an elliptic orbit, the velocities at the perigee and apogee can be computed as well from the *vis-viva equation* as follows:

$$v_{perigee} = \sqrt{\frac{\mu(1+e)}{a(1-e)}}; \qquad v_{apogee} = \sqrt{\frac{\mu(1-e)}{a(1+e)}}$$

The transfer orbit is fully defined considering that the perigee and the apogee radius coincide with the circular radius of the initial orbit and the apogee orbit of the final orbit, respectively. Thus, each impulse would be computed as the absolute different between the transfer orbit velocity and the initial/final orbit velocity at the intersection points.

$$\Delta V_1 = \left|V_1 - V_{transfer,perigee}\right| \Delta V_2 = \left|V_2 - V_{transfer,apogee}\right|$$

The signs indicate if the satellite must be accelerated or decelerated but in terms of the propellant required this difference is irrelevant; thus the signs are ignored.

Finally, the total impulse would be:

$$\Delta V_{tot} = (1 + p_{extra})(\Delta V_1 + \Delta V_2)$$

Where p_{extra} is the percentage of the ideal total impulse that represents an additional impulse required for corrections, drag losses, perturbations, etc.

By selecting a propulsion system, the specific impulse I_{sp} and the dry mass of the engine m_e are included; hence it is possible to compute the total wet mass of the satellite by applying the *Tsiolkovsky equation*:

$$m_0 = m_f \cdot e^{\frac{\Delta V_{tot}}{I_{sp} \cdot g_0}}$$

Where m_f is the final mass after the maneuver; it is considered that at the end there is no remaining propellant and thus m_f = m_e+ $m_{satellite}$(without the propulsion system).

The propulsion system cost is computed straightforward from the net thrust of total impulse (N·s) associated to the engine. Thus:

$$\text{Cost} = C_e \cdot I_{sp} \cdot F$$

Where $C_e = 0.034\ \frac{\$}{N \cdot s}$ for solid rocket motors and $C_e = 0.09\ \frac{\$}{N \cdot s}$ for liquid motors [Indicar paper]

As it was commented during the previous chapter, no other propulsion systems have been included to perform an orbit insertion maneuver.

4.2 PAYLOAD DESIGN

The payload window does not require any model. The user introduces directly the mass, cost and power consumption associated to the payload and they are directly included in the system.

In case the user desires to include a customized onboard computer, the software will proceed identically; i.e. adding the power, mass and cost contributions to the total payload parameters. There must be noticed that this computer is not a specific computer for the payload, but corresponds to general computer for handling and process all the satellite data. In some simple missions this computer is nearly a small processor board and therefore their contributions can be neglected. However, in more complex mission the mission design or the payload generated data may require a more powerful computer. Since the payload plays a fundamental role in determining the type of computer required, it was decided to include the onboard computer modeling in the same design window.

4.3 POWER SUBSYTEM

The power subsystem computes the size of the solar arrays needed to supply the power required by the satellite systems, as well as the secondary batteries required to supply the loads during the eclipse operation. It estimates as well the control, distribution and regulation power systems.

4.3.1 *Primary power source model*

The size process followed corresponds to the one suggested in "The new SMAD" from the literature [8]. Firstly, it loads the total power P_d required by the communication, the ADCS, the telemetry and the thermal subsystems, and the power required by the payload. Then, it computes the total power required during the eclipse period from the operation power rate Π provided in the *mission design*:

$$P_e = \Pi \cdot P_d$$

Therefore, the total power required would be computed as:

$$P_{required} = \left(\frac{\frac{P_e \cdot T_e}{X_e} + \frac{P_d \cdot T_d}{X_d}}{T_d} \right)$$

Where X_e and X_d are the efficiency cell-battery-load and the efficiency cell-load respectively, and they are determined by the Power Transfer System installed.

Secondly, the power produced per area would be computed. It will depend on the type of cell used and the orientation angle θ between the solar arrays and the sun incidence flux.

$$P_0 = \eta \cdot 1368 \frac{W}{m^2}$$
$$P_{BOL} = P_0 \cdot I_d \cdot \cos\theta$$

Where η is the cell efficiency and I_d is the Total inherent degradation. The orientation angle θ will be determined in the ADCS design window when modeling the pointing losses.

In the last step, the degradation during the life of the mission is considered and the solar arrays are size accordingly to provide full power until the end.

$$L_d = (1 - D)^L$$
$$P_{EOL} = P_{BOL} \cdot L_d$$
$$A_{solararray} = \frac{P_{required}}{P_{EOL}}$$

The mass and cost associated to the solar array sizing would be computed directly according to the mass-to-Watt and cost-to-Watt introduced by the user.

As commented in previous chapter, no other primary sources have been included.

4.3.2 *Secondary power source model*

The secondary battery is modeled considering the energy that must be storage to supply the satellite during the eclipse, and the charge efficiency and the depth of discharge that will be determined the battery technology.

The first step is to load the total power required during the eclipse and the eclipse duration. In order to be conservative, the eclipse duration considered to size the batteries corresponds to the maximum orbit eclipse: apogee or perigee.

The second step is to determine the maximum *Depth of Discharge* (DOD) that can be applied to the battery in order to last until the end of the mission. For each technology the maximum allowed DOD will depend on the number of cycles that the battery will undergo. Therefore:

$$N_{eclipse/year} = n \cdot \frac{365 \cdot 24 \cdot 60}{\tau_{orbit}}$$
$$N_{cycles} = L \cdot N_{eclipses/year}$$

Where n is the number of eclipses per orbit, that will depend on the mission configuration; and L is the mission life in years. τ_{orbit} is in min

Based on the references provided in figure 36, the DOD can be directly obtained. Finally the energy required to be storage by the battery would be:

$$E = \frac{P_e \cdot T_e}{DOD \cdot \eta}$$

The total mass and cost associated to the battery can be directly related to the energy required to be stored.

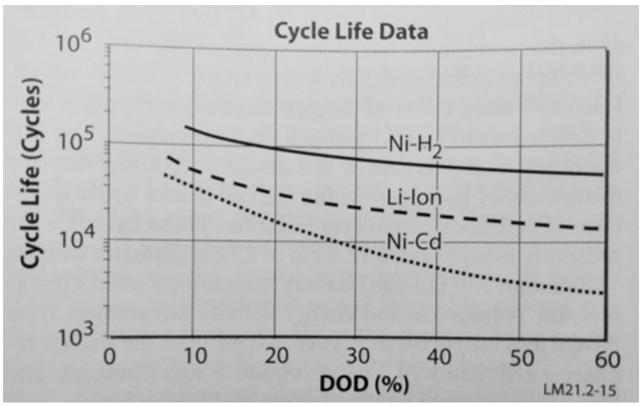


Figure 36: Depth of Discharge vs Cycle Life curves for each battery technology

4.3.3 *Power distribution, regulation and control systems*

These systems are simply estimated considering that they usually represent around the 17% of the total mass of the power subsystem and the 21% of the total cost.

4.4 COMMUNICATION SUBSYSTEM

The objective of the communication design window is to obtain the solution to the communication link basic equation. Taking into account the mission parameters and the losses associated to the elements involved in the communication, it is possible to compute the minimal transmitter power required by the satellite to make possible the communication with the ground station with a certain BER.

Similarly, the communication link equation can be solved to obtain the minimum size of the satellite receiver antenna to recognize properly the signal from the ground [10].

The communication link is generally formulated as:

$$\frac{E_b}{N_0} = \frac{P \cdot L_t \cdot G_t \cdot L_s \cdot L_a \cdot G_r \cdot L_r \cdot L_e}{k \cdot T_s \cdot R}$$

Where E_b/N_0 is the rate of received energy-per-bit to noise-density, P is the transmitter power, L_t is the transmitter to antenna line loss, G_t is the transmitter gain, L_s is the space loss, L_a is the transmission path loss, G_r is the receiving antenna gain, L_r is the antenna to receiver line loss, L_e represent any other extra losses, k is the Boltzmann's constant, T_s is the system noise temperature in K and R is the data Rate in bps.

Firstly, in order to avoid an overloaded interface, it was decided to consider a constant link distance equal to the semimajor axis of the orbit. Thus, the free space losses would be computed as:

$$L_s = \frac{4 \cdot \pi \cdot a}{\lambda} = \frac{4 \cdot \pi \cdot a \cdot f}{c}$$

Where f is the frequency at which the data is transmitted determined by the available bandwidth and the mission requirements.

In the same way, the atmospheric losses would depend mainly on the transmitted data frequency. The total atmospheric losses would be estimated considering the model sketched in figure 37. [8] [2] [5]

Secondly, the antennas were modeled as uniformly illuminated antennas and their gains were modeled as the ratio of their effective aperture area A_r and the effective area of a hypothetical isotropic antenna, leading to:

$$G = \frac{A_r \eta}{A_{r,iso}} = \frac{\frac{\pi \cdot D_r^2}{4}}{\frac{\lambda^2}{4\pi}} \eta = \eta \left(\frac{\pi \cdot D_r}{\lambda} \right)^2$$

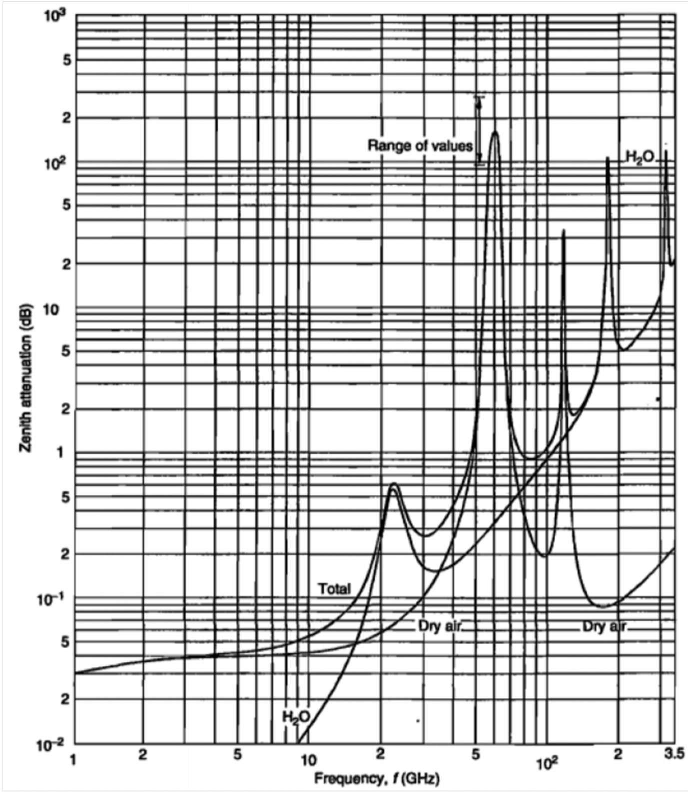


Figure 37: Atmospheric attenuation as a function of the signal frequency [8]

The next step is to find the system temperature noise contribution to the line losses the user must consider the noise figure of each antenna, the transmitter and the receiver, and the line loss associated. Thus, the final noise temperature will be computed as:

$$T_s = T_{ant} + \frac{T_0 \cdot (1 - L_r)}{L_r} + \frac{T_0 \cdot (F - 1)}{L_r}$$
$$where \qquad F = 1 + \frac{T_r}{T_0}$$

Finally, the minimum E_b/N_0 ratio to make the mission feasible is determined by two mission data requirements: the Bit Error Rate (BER) and the modulation method implemented. There are various method to determine the E_b/N_0 ratio, and the one finally implemented was a graphical reference of BER vs E_b/N_0 curves provided in SMAD where the modulations BPSK, QPSK, 8PSK and 16PSK were included.

The line losses L_l and L_r are directly user inputs that would depend on the transmitter and receiver installations and the data rate R would be determined directly according to the mission requirements. The extra losses L_e are included by user considerations.

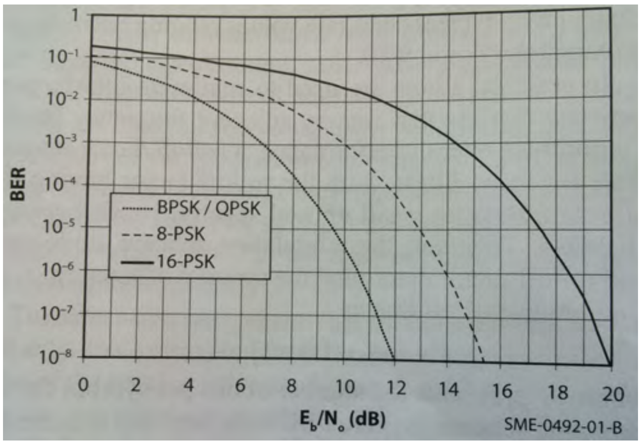


Figure 38: BER vs E_b/N_0 curves

There are two different types of links: satellite-ground station and satellite-satellite link, the only difference is that the atmospheric losses of the last one are negligible. Each communication-architecture would require a different number of links to be solved. All the total power, mass and cost of each link are added up and determined the communication subsystem characteristics.

In order to enhance the computations, it was decided to express the link equation in decibels. Therefore, by isolating the power term:

$$P = \left(\frac{E_b}{N_0} + m\right) + (L_l + L_s + L_a + L_r + L_e) - G_t - G_r - 228.6 + 10 \cdot \log T_s + 10 \cdot \log R \tag{1}$$

In this last equation, the possible margin m has been included; and the losses are gathered together and introduced without the negative sign.

Similarly, when the transmitted power is known and the receiver antenna of the satellite must be determined, the link equation is solved to obtain the receiver gain, from where the antenna size will be finally obtained.

4.5 ADCS SUBSYSTEM

The Attitude Determination and Control Systems contribution to the mass, cost and power required do not require any model; the software sum up the total mass and power associated to the systems implemented through the design window. Finally the total cost would be computed proportionally to the total mass obtained.

On the other hand, the design window provides to the user with an estimation of the main torques acting over the satellite and disturbing its position: the solar radiation pressure, the aerodynamic drag, the magnetic field and the gravity gradient. The magnitudes of these four torques are estimating according to the model provided by the "The new SMAD" from the literature [8] and summarized as follows:

Solar Radiation Pressure Torque:

$$T_s = \frac{\sigma}{c} \cdot A_s \cdot (1+q) \cdot L_{cp} \cdot \cos\theta$$

Where σ is the solar constant flux (W/m^2), A_s is the sunlit area (m^2), q is the reflectance factor, $L_c p$ is the distance between the center of mass and the center of pressure of the solar radiation and θ is the pointing angle.

Aerodynamic Drag Torque:

$$T_a = \frac{1}{2} \cdot \rho \cdot C_d \cdot A_r \cdot V^2 \cdot L_{ca}$$

Where ρ is the density, C_d is the drag coefficient, A_r is the ram area, V is the velocity and $L_c a$ is the distance from the aerodynamic center to the center of mass. Taking into account that the model must be valid for a wide range of orbit sizes, it was decided to determine the density basing on a graphical simplification of the NRLMSISE-00 model. [4]

Magnetic Field Torque:

$$T_m = D \cdot \frac{M \cdot \lambda}{R^3}$$

Where D is the spacecraft residual dipole in $A \cdot m^2$, B is the magnetic field strength in T and λ which is a unit less function of the magnetic latitude.

Gravity-Gradient Torque:

$$T_g = \frac{3\mu}{2R^3} \cdot \left|I_z - I_y\right| \cdot \cos 2\phi$$

Where I_z and I_y are the inertial moments in $kg \cdot m^2$ and ϕ the angle between the vertical and the Z principal axis.

4.6 LAUNCH SUBSYSTEM

The launch subsystem is modeled as a partial Hohmann transfer orbit, determined by four parameters: the launch angle, the initial and final velocities and the final radius.

The launch angle would depend on the launching trajectory design and so it is provided by the user. The final radius corresponds to the initial orbit defined during the mission design or the orbit itself for LEO missions. Furthermore, since the rocket performs a Hohmann transfer, the angle between the velocities at the insertion point is zero. The geometric configuration can be checked in figure 39.

Taking into account the *conservation of angular momentum* and the *conservation of the mechanical energy* that characterize the elliptical orbits the following system is obtained:

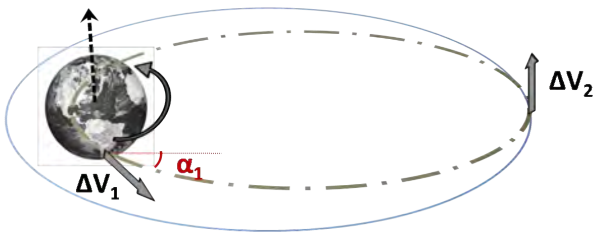


Figure 39: Launch geometry sketch

$$\begin{cases} \frac{V_1^2}{2} - \frac{\mu}{R_E} = \frac{V_{a,2}^2}{2} - \frac{\mu}{R_2} \\ R_E \cdot V_1 \cdot \cos\alpha_1 = R_2 \cdot V_2 \cdot \cos\alpha_2 \end{cases}$$

Where α_1 is the launch angle, V_1 and $V_{a,2}$ the initial and final velocities respectively, R_2 is the final radius and α_2 is the final angle.

By solving the system the last two parameters are obtained: V_1 and $V_{a,2}$. Therefore, it is possible to compute the two impulses required to perform the launch identically as it was done in the propulsion system model:

$$\Delta V_1 = |V_1 - V_0|$$
$$\Delta V_2 = \left|V_{2,apogee} - V_{transfer,apogee}\right|$$
$$\Delta V_{tot} = (1 + p_{extra})(\Delta V_1 + \Delta V_2)$$

Once more, the user can include extra impulse required to deal with extra losses due to gravity, drag, perturbations, etc.

On the other hand, in this case the rocket is launched from the ground at rest but the Earth rotation provides an initial velocity that would not be aligned with the computed V_1. Therefore the first impulse must be computed as the magnitude of a resultant vector. The initial velocity would depend on the latitude radius, being maxima at the equator and zero at the poles.

$$V_0 = R_E \cdot \omega_E \cdot \cos(Latitude)$$
$$\text{thus} \quad \Delta V_1 = V_1^2 + V_0^2 - 2 \cdot V_1 \cdot V_0 \cdot \cos\alpha_1$$

The total launch cost will be computed directly, by introducing a unitary cost per kilo that will depend upon the launch rocket used as it was discussed during the introduction chapter. It is important to remember that only the mass of the satellite will be considered when computing the launch costs.

4.7 THERMAL SUBSYSTEM

Taking into account the consideration discussed in previous chapters, the thermal subsystem has finally two models implemented to compute the temperature range at which the satellite operates. In order to modify the range, the user can install a radiator or a louver. These thermal control systems will have a cost and a mass associated that will be computed directly considering the power input, for the case of the radiator, and the temperature reduction, for the case of the louvers.

4.7.1 *Model I*

The user introduced the area of the satellite oriented towards the main directions (Zenith, Nadir, Sun and AntiSun) and the model interpolates the *Solar*, *Albedo* and *Irradiated Fluxes* that reach at each surface according to the orbit radius and inclination for the hot case and the cold case. The interpolation model was based on the data provided on the Table 22-11 of "The new SMAD" of the literature [8]. Therefore the heat energy that reaches the satellite is:

$$Q_{environment} = A_{zenith} \cdot F_z + A_{Nadir} \cdot F_n + A_{sun} \cdot F_s + A_{antisun} \cdot F_{as}$$

4.7.2 *Model II*

The second model requires the user to introduce eight design parameters to compute the heat energy as follows:

$$Q_{environment} = \alpha \cdot S \cdot (A_p + R \cdot A_R) + \epsilon \cdot IR \cdot A_{IR}$$

Where α and ϵ are the surface absorptivity and emissivity for a solar source and infrared source respectively, S is the solar irradiance, R the percentage of the solar irradiance reflected from the planet, IR is the irradiance of infrared energy from the planet, A_p is the projected area towards the sun, A_R towards the reflected planet surface and A_{IR} towards the infrared energy.

By applying an energy balance with respect to the total energy radiated out, it is possible to compute the maxima and minima temperature at which the satellite will be operating:

$$Q_{in} = Q_{env} + Q_{intern} = Q_{out} = A_r \cdot \epsilon \cdot \sigma \cdot T^4$$
$$T = \sqrt[4]{\frac{Qenv + Qinter}{\sigma \cdot A_r \cdot \epsilon}}$$

Where $Q_{internal}$ represents the electrical power dissipated by the satellite systems. The heat of the radiator will be added to $Q_{internal}$ while the louver temperature reduction will be directly applied to the range.

It has been considered that the louver mechanism do not contribute to the power consumption. The system would only require power for the adjustment actuator, when modifying the angular position of the louver blades.

4.8 STRUCTURAL SUBSYSTEM

The model implemented in the structural design window is very basic considering the limitations discussed in previous chapters. The user introduces the percentage of the primary, secondary structure and fasteners with respect to the total satellite mass and therefore the final mass would be computed as:

$$m_{satellite} = \frac{m_{restofsubsystems}}{1 - p_{primary} - p_{secondary} - p_{fasteners}} + m_{extra}$$

The model proposed in "The new SMAD" [8] to model the costs corresponds to a potential relation with respect to the total mass of both the thermal and the structural subsystems. Consequently, by summing up the mass contribution of both systems the total costs are computed as follows:

$$cost = a \cdot (m_{struct} + m_{thermal})^b$$

Where a = 642 k$ and b = 0.684

These parameters are selected to predict the total cost of each subsystem, including design, manufacturing and testing costs, for a unique satellite production. The user must be aware that when producing similar satellites the costs are not accumulative since they complement each other; thus the cost per satellite would decrease.

In order to model each subsystem individually, the following approximation was implemented:

$$C_{tot} = C_{struct} + C_{thermal} = c \cdot (a \cdot m_{struct}{}^b + a \cdot m_{thermal}{}^b)$$

Where c = 0.8214 leads to a residual errors of R = 0.99934 when compared with the original expression.

5 TOOLS VALIDATION

The following chapter includes a full analysis of a real space mission using the software tool developed throughout the thesis. The predictions will be compared with the mission results provided in the literature, analyzing the divergences according to the hypothesis included in the model.

5.1 MISSION SELECTION

As a consequence of the hermetism that characterizes the space industry, it is not possible to found a detailed database to model all the subsystems of a real space mission. Although there are numerous researches and analyses available providing relevant information for the most significant missions, they are neither complete nor reliable. Therefore, it was decided to test the rigor of the mathematical models implemented by modeling a hypothetical space mission called *FireSat*. Despite the fact that it is a theoretical mission that has never been funded and built, their performance has been widely studied since it was proposed in 1991 to illustrate the space mission design process, becoming a standard example mission throughout the astronautics community. Consequently, the engineering parameters are available to model every subsystem as in the literature; a fact that eliminates the subsystem design error and thus guarantees an accurate analysis of the discrepancies between the literature database and the SW Suite results.

5.2 FIRESAT II MISSION

The *FireSat* mission version to be modeled corresponds to the one described in the 3th edition of "The New SMAD" [8] called *FireSat II*. The *FireSat* statement was proposed as follows:

> *Because forest fires pose an ever-increasing threat to lives and property, have a significant impact on recreation and commerce, and also have an even higher public visibility [. . .] the United States needs a more effective system to identify and monitor them. In addition it would be desirable to monitor forest fires for other nations; collect statistical data on fire outbreaks, spread, speed and duration; and provide other forest management data. This must be done at low cost to make the system affordable to the Forest Service and not give the perception of wasting money that could be better spent on fire-fighting equipment or personnel. [...]*

Following the SMA design process, the first step is to identify the objectives of the mission:

- The primary objective is to detect, report and monitor forest fires in the US in near real time and at low cost.
- Other secondary objectives (not required) would be collect statistical data, provide coverage in other countries and collect other forest management data.

Once the objectives have been established, the next step is to identify the mission requirements associated to those objectives and fixed the project constrains based on the customer preferences. For the sake of simplicity, all the requirements have been summarized in Appendix B. The engineering parameters introduced as *inputs* during the mission modeling derive from these requirements. Ultimately the most relevant ones are the *mission design life*, 8 years, and the *total cost*, non-recurring 10 millions of dollars.

5.3 SW SUITE ACCURACY EVALUATION

In order to enhance the presentation of the results and make the discrepancies analysis clearer the validation process is proposed in the following way: each subsystem will be modeled individually and compared with the expected results provided by the literature; then the mission estimated by the SW Suite will be evaluated as a whole to check the discrepancies of the mathematical model implemented in the Software Suite.

5.3.1 *General Mission*

The *FireSat* satellite is intended to be operating in a circular orbit at 700km, with an inclination of 55°. This orbit will be characterized by a period of 98.9 min and an eclipse of 35.4 min. The SW Suite estimates a period of 99 min and an eclipse of 33 min.

The difference on the period is indeed inexistent since the SW Suite rounds the results displayed to enhance the visibility of the interface. By checking the value used during the computations, it can be observed that it is in fact 98.774 min.

However, the eclipse duration is estimated to be 33 min, which implies a deviation of 6.7%. Analyzing other mission with different orbits it can be observed that there is always a deviation in the eclipse duration, but the relative error does not remain constant neither in magnitude nor in sign. By plotting both functions together it can be notice that both sinusoidal curves are identically, but the software one has a negative phase shift. Analyzing the model implemented, the error was finally found during the β angle computation, where the software do not model properly the angle yearly variation and therefore the eclipse displayed do not correspond to the maximum one.

5.3.2 *Propulsion system*

The satellite is designed to be delivered by the launch vehicle into a circular parking orbit at 200 km. Therefore the total impulse required to reach the operational orbit would be 280 $\frac{m}{s}$ split into two identical impulses. Furthermore, it is estimated that the mission would require an extra impulse of 10 $\frac{m}{s}$ for orbit maintenance and a last impulse of 187 $\frac{m}{s}$ for its disposal.

The SW Suite estimates as well a 280 $\frac{m}{s}$ impulse for the orbit transfer maneuver, split as well in two identical impulses. On the other hand, the maintenance and the disposal requirements must be modeled directly as an extra impulse increment; in this case 36.5% of extra impulse is required.

5.3.3 *Payload*

According to the literature, the payload installed in the FireSat satellite will consume 65W and will add 20kg to the satellite mass. However, it was decided to carry a 30% margin to compensate possible extra elements, resulting in 26kg allocation.

Furthermore, the mission will require an onboard processor to manage all the payload information. This processor is estimated to consume 17W and add 4kg to the payload mass.

As explained in *Chapter IV: Models*, those parameters will be introduced directly in the payload design window to be used in subsequent design windows.

5.3.4 *Power subsystem*

During the preliminary design, the *FireSat* mission is estimated to consume 141 W in both daylight and eclipse working conditions. The daylight power will be supplied by Silicon cell solar arrays sized to $2,4m^2$ and weighing 10.8 kg. On the other hand, the eclipse power will be supplied by Li-Ion batteries with 92% of charging efficiency and a depth of discharge ranged between $20-40\%$ (30% was finally selected). Finally the secondary supply system weight 2.2kg.

However, the SW Suite sizes the solar array to $2.5m^2$ and estimates a total weight of 9.6kg. These deviations, 4.1% and 11%, would probably be caused because the solar arrays considered in the literature must be provided with extra layers that guarantee a higher performance and a better protection (a common practice in the space sector), but they make the elements heavier. The Silicon cells considered by the software have an efficiency of 14.5% and weight $2.3\frac{kg}{m^2}$; extrapolating from the literature results, they will probably consider a 16% and $2.5\frac{kg}{m^2}$ solar cells.

Regarding the secondary batteries, the SW Suite estimates a maximum allowed DOD of 22%, which coincides with the literature predictions (although references have been dismissed to include a 30% DOD) and a total mass of 2.15kg. The mass estimation is deviated a 2.3%. It would probably with related with the fact that the literature split the battery mass into three equal batteries units; thus it may have included extra losses associated to the power distribution.

5.3.5 *Communicaion subsystem*

The *FireSat* communication architecture is designed as a unique satellite-ground station downlink together with the telemetry system. Unfortunately, not all the communication subsystem parameters are provided by the literature and therefore it some component losses must be inferred from the average values that they usually have.

The *FireSat* telemetry system was designed to transmit 5Mbps, at 2.2GHz, with a QPSK modulation and a maximum BER of $5 \cdot 10^{-5}$. This conditions leads to a E_b/N_o of 5.5dB. A margin of 6dB was required.

Furthermore, the link range was estimated to be 2560km resulting in 167.5dB of space losses but negligible atmospheric losses. The ground station is considered to have a receiver antenna with a 45dBic gain. The power required by the satellite to transmit the data is 4W, and the antenna was sized to have a gain of 4dB.

On the other hand, the main communication link was designed to operate under the same conditions but transmitting 100Mbps at 11GHz and requiring a E_b/N_o of 15.6dB. The power required to accomplish the communication link will be 13W.

According to the parameters provided and estimating the missing data from similar missions, the SW Suite estimates that the total power required by the transmitter antenna in the satellite will be 3W for the telemetry system and 13W for the communication downlink. The total deviation would be 5.9%; although the results are not conclusive considering the inaccuracy of the parameters that models the link.

5.3.6 *ADC System*

The *FireSat* literature model estimates the torques of the four main disturbances as follows: *Solar Radiation*, 0.96 μNm. *Atmospheric Drag*, 0.37 μNm. *Magnetic Field*, 21 μNm. *Gravity Gradient*, 1.6 μNm.

Based on these estimations, the following sensors and actuators are incorporated to the design: 1 mid-size momentum wheel, 3 electromagnets, 6 wide-angle sun sensors, 2 scanning type horizon sensors, 3 MEMS gyroscopes and a 3-axis magnetometer. These elements add an extra mass of 5kg and consume 14W.

On the contrary, the SW Suite estimations differ significantly from the literature. The torques associated to the four main disturbances are estimated as: *Solar Radiation*, 0.88 μNm. *Atmospheric Drag*, 0.09 μNm. *Magnetic Field*, 176 μNm. *Gravity Gradient*, 1.27 μNm.

The respective deviations are 9%, 75%, 88% and 8.1%. The atmospheric drag discrepancies may be originated from the atmospheric model selected that will conditioned the value of the density and will influence significantly on the final result: literature estimates the density at 700km around 10^{-13} while the SW Suite model estimated it one order of magnitude smaller, $3.2 \cdot 10^{-14}$. On the other hand the magnetic field deviation may be related with the same bad-inclination model issues considered in the eclipses results. The Solar Pressure and the Gravity-Gradient deviations are more reasonable and must be related with the parameters introduced during the torques modeling.

Since the SW Suite was not provided with an interface to model the sensors and actuators, only standard elements can be selected. In this case, we have included the appropriate elements to guarantee the mass and power contributions established by the literature model.

5.3.7 *Thermal subsystem*

The literature model estimates the working temperature range of the FireSat satellite from $-5°C$ up to 24°C without thermal control elements. According to the SW Suite, this range should be from $-6°C$ to 26°C. The temperature range have been increased , 2°C the upper limit and 1°C the lower one; but considering that these deviations force the design to be more conservative the implications are less relevant.

5.3.8 *Structure*

The mass of the *FireSat* satellite structure is considered to be 23kg, including the primary, secondary and assembly components.

As explained in *Chapter IV: Models*, the SW Suite estimates the mass as a percentage of the total mass defined by the rest of the satellite subsystems:

- The propulsion system selected was the $S-400-12$ since it was the most feasible one to model the 3kg generic system considered in the literature. It weights 3.6kg.
- The payload introduced directly in the design window together with the onboard processing ($26+4$kg).
- The power system, as explained before, weights 11.75kg.
- The communication subsystem mass was estimated to be 2.1kg, both the downlink and the telemetry systems. The literature estimates that the to-

tal mass will be 2kg, a deviation of 5% (related with the same conclusions described during the analysis of the communication subsystem).

- ADC and thermal subsystems mass contribution, 7kg.
- Additionally, the literature adds an extra mass of 3kg to model other satellite systems that have not been considered.

Accordingly, the total mass of the satellite structure will be 21.9kg. The total deviation with respect to the literature estimation will be 4.7%, that will be related with the different mass contributions deviations that have been mentioned.

5.3.9 *General Analysis*

Evaluating the results provided by the SW Suite it can be notice that, dismissing the ADCS model, the average error is 2.61% which fulfills the accurate requirements established at the beginning of the thesis. However, according to the results two points must be raised:

- For certain subsystems, the results are not conclusive. It would be necessary to obtain fully detailed examples to ensure the level of error obtained from this analysis.
- The deviation error varies significantly between the satellite subsystems. Thus, in order to improve the SW Suite, certain models must be reviewed and corrected to increase their accuracy and obtain a more reliable software tool.

The results obtained during the validation process have been summarized in figure 40, where it can be check that the average accuracy of the software models is good enough for a preliminary design tool but there are noticeable differences between the subsystems.

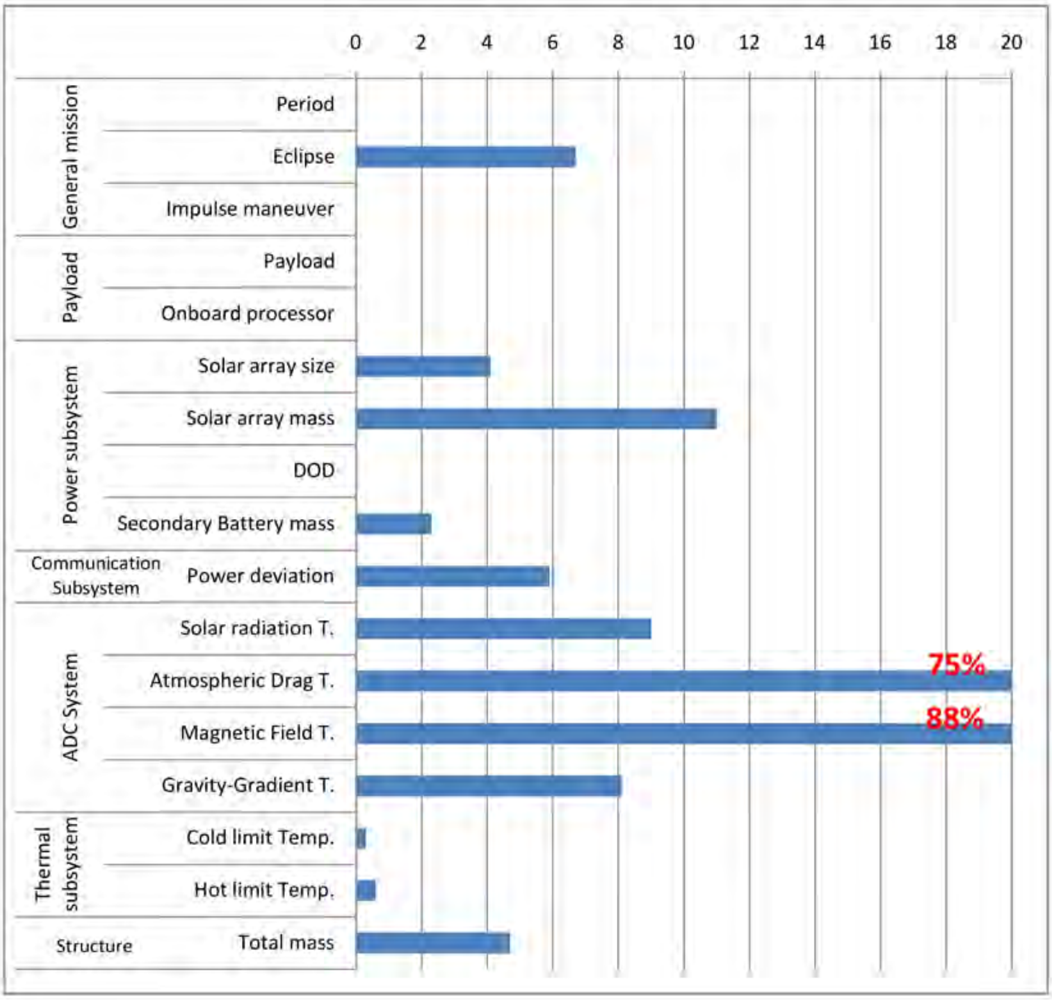


Figure 40: Desviation error distribution of the FireSat analysis.

Part III

THE CONCLUSIONS

The summary and the conclusions of the thesis are gathered through the following sections; including a future plan to improve the current software suite, a socioeconomic analysis associated to the impact of launching the application and finally a legal framework analysis regarding this launching process.

6 SOCIO-ECONOMIC ANALYSIS OF THE PROJECT

In the following chapter the engineering project behind the thesis would be studied. The analysis will be divided in two sections: a brief socio-economic analysis of the impact that the application of the project would have in the sector; and a budget report where the main information about the project is summarized.

6.1 SOCIO-ECONOMIC IMPACT IN THE SECTOR

As it can be inferred from the previous chapters, the software tool developed throughout the thesis has a real application in the aerospace sector as an ergonomic support during the customer meetings.

The impact of the use of this tool must be analyzed under two different points of view:

- The economical and human resources expended in the analysis of the discarded options.
- The enhancement of the business relationship.

In order to provide an estimation of the total cost of a preliminary design it will be taken as a reference a satellite mission project designed in the Astrium dependences in Airbus D&S Factory in Barajas, like the *Express AMU* 1. Considering that the preliminary analysis would be performed in each department by two senior engineers (average salary of 55000 €[1]), it would take 10 hours to develop an initial design and another 30 hours due to the iteration between the different departments. This amount, coupled to the computation resources that those analysis required (the use of super-computer-based tools and mainframe computers to implement models in heavy programs like Ansys Fluent and Nastran), the preliminary design would cost approximately 32.000 €.

Taking into account this estimation, it can be notice that the cost associated to a rejected preliminary design are significant and considering that usually the customer does not approve all the modifications, this cost will depend on the number of modifications approved by the customer.

It is important to recall from the introduction chapter the importance of the opportunity costs associated with the human and time resources involved and that represents a real cost in the situation that it is being considered.

1 Extracted from "Tablas salariales 2017, Barajas" that can be foundin the SIPA webpage

Furthermore, although the economical advantage of an enhancement in the business relationship is difficult to be measured, its importance cannot be overlooked. It must be considered that the customer experience is directly related with future profitability in terms of new projects or recommendations; so any improvement will be always desired as it will represent future wealth.

Additionally, a second application can be considered: in cases where a customer desires to include any modification to the initial project once it has already started. Those modifications would require another preliminary analysis to determine the viability, but they can be rejected later on depending on the results.

Summarizing, the SW suite has a great potential for a space company due to several reasons. Firstly, due to the noticeable economical resources saved. Secondly, due to the workflow improvement that allows the company to optimize their resources. And finally due to the enhancement of the business relation with the customer.

However, current companies in the space sector have already developed operational techniques to deal with these problems and reduce the associated losses. Consequently the initial attraction for the software implementation has been dismissed; they will consider the incorporation of the SW suite as an enhancement of the actual process. Thus, it is crucial to provide the tool with highly attractive features to enchant the customers as for example a user friendly interface, that guarantees an easy and fast trade-off process.

It is worthy to highlight the fact that there are already programs available for the analysis of a space mission: *System Tool Kit* (developed by AGI), *FreeFlyer* (by a.i.solutions), *ALMASim* (by SITAEL), *General Mission Analysis Tool* (by NASA), *Celestia* (by grupo CA Celestia), etc. There are also numerous software resources to design each subsystem individually. The main advantage of the SW suite developed in this thesis lies in the fact that those software tools are mainly focus on modeling the performance and the operational life of a satellite, but they are not useful to design all the subsystems and perform a fast trade-off between them. This characteristic is indeed the feature that differentiates the SW suite from those that are currently in the market and accordingly it has leaded the software interface design.

The best way to engage the customers would be by developing a sales promotion plan focused on a personalized proposal for each customer. It implies that the representative of SW suite team will contact with the space department of each company and offers them a customized demonstration to show the potential of the tool. During this meeting, the representative would be able to gather information about which are the specific problems of the client, in what extent this tool would improve the process, the interest on acquiring the product and additionally he would ask the client for any extra features that he desires. This information would be used to improve software features, find the strong attractive points and therefore customize the product for each customer. Furthermore it will determine the final price of the tool.

Companies of the space sector that can be interested in the mentioned tool will be both the biggest leading companies like *SpaceX*, *Airbus Defense* & *Space*, *Lookheed Martin*, *Boeing Defense*, *Space* & *Security*, etc but also high specialized companies of the space sector such as *Orbital ATK*, *Virgin Galactic*, *XCOR Aerospace*, *RSC Energia*, etc. Additionally, considering the interest of the governments to develop good space programs, the space agencies (NASA, ESA, CNES, INTRA, etc) may be interested as well.

On the other hand, considering the final characteristics of the SW suite it has become as well an attractive tool for educational purposes. Although it was not designed with that purpose the SW suite can be used in the universities to learn the basis of a SMA and its complexity. For this case it would be possible to provide a light non-customized version for a lower price.

6.2 GENERAL REPORT

The present thesis can be evaluated as a software development project. The budget corresponds to a well-grounded estimation of the economic necessities required to accomplish the project. It indicates the amount of resources that must be provided by the investors and it obliges to think rigorously on the consequences of the planning of the different activities in the schedule. It also provides a good reference to control the incomings and the costs and it becomes a useful mean for the accountability and economic transparency.

Regarding the socio-economic framework exposed, all the relevant information related with the project development will be summarized in order to propose a budget. The principal elements that contribute to the project budget are specified below:

- **Duration:** The total duration of the project development is 6 months. It includes the previous investigation, the software development, the writing of the report and the subsequent project revisions.
- **Engineer team:** Formed by a *junior* aerospace engineer with an estimated salary of 29928 €[2] working in full-time and a senior engineering supervisor with an estimated cost of 26.44€/h, working an average of 2 hours each week.
- **Hardware:** The project requires a sufficiently good computer able to support a Matlab GUIDE interface and run the mathematical models.
- **Software:** The project requires an Academic Matlab License with the GUIDE interface implemented in order to develop the software suite. Although this thesis has been elaborated using Windows 10 and Microsoft Office 2016 licenses, they would be considered as extra costs since there are currently free software able to provide the same utility.

2 Extracted from "Tablas salariales 2017, Barajas" that can be found in SIPA webpage

- **Legal expenditure:** Where the costs associated with patent register or legal rights are considered. It includes the official rate as well as the labor fees of the lawyer consultancy.
- **Project expenditure:** It includes all secondary expenses that the project may require: displacements, electricity, the use of academic facilities during the initial investigation, etc.
- **Unforeseen expenses:** It is reasonable to oversize the budget in order to be able to overcome possible costs that have been not considered at first instance.

Element	Tipo	Amount	Unitary Cost	Total Cost
Junior Engineer	Engineering Team	6 months	29.928€/year	14.964 €
Senior Engineer Supervisor	Engineering Team	0,4h / day	26,44€/h	1.269 €
Computer amortization	Hardware	1	10€/month	60 €
Matlab Academic License	Software	1	500 €	500 €
Copyright fee	Legal expenditure	1	50 €	50 €
Project expenditure	Proyect expenditure	-	2% of the total cost	345 €
Unforeseen expenses	Unforeseen expenses	-	10% of the total cost	1.890 €
TOTAL				19.078 €

Figure 41: Project General Budget

The project was divided in three phases according to the nature of the tasks that were planned to be accomplished in those periods:

I Gather information phase (weeks 1-8). The junior engineering gather information and collect data related with the project. Basing on this searching, he will determine the scale of the project by planning and selecting the subsystems and the drivers to be implemented.

II Software development phase (weeks 9-18). The mathematical models are implemented into software, focusing on improving the user experience and enhancing the trade-off process.

III Report writing-Revision phase (weeks 19-24). Once an operative software suite is obtained, all the ideas are included in the report. Meanwhile, new options are included in the software, improving some points and revising the operability. The report chapters are weekly being revised.

The tasks scheduled in each stage were included in the project Gantt diagram shown in figure 42 located in the next page.

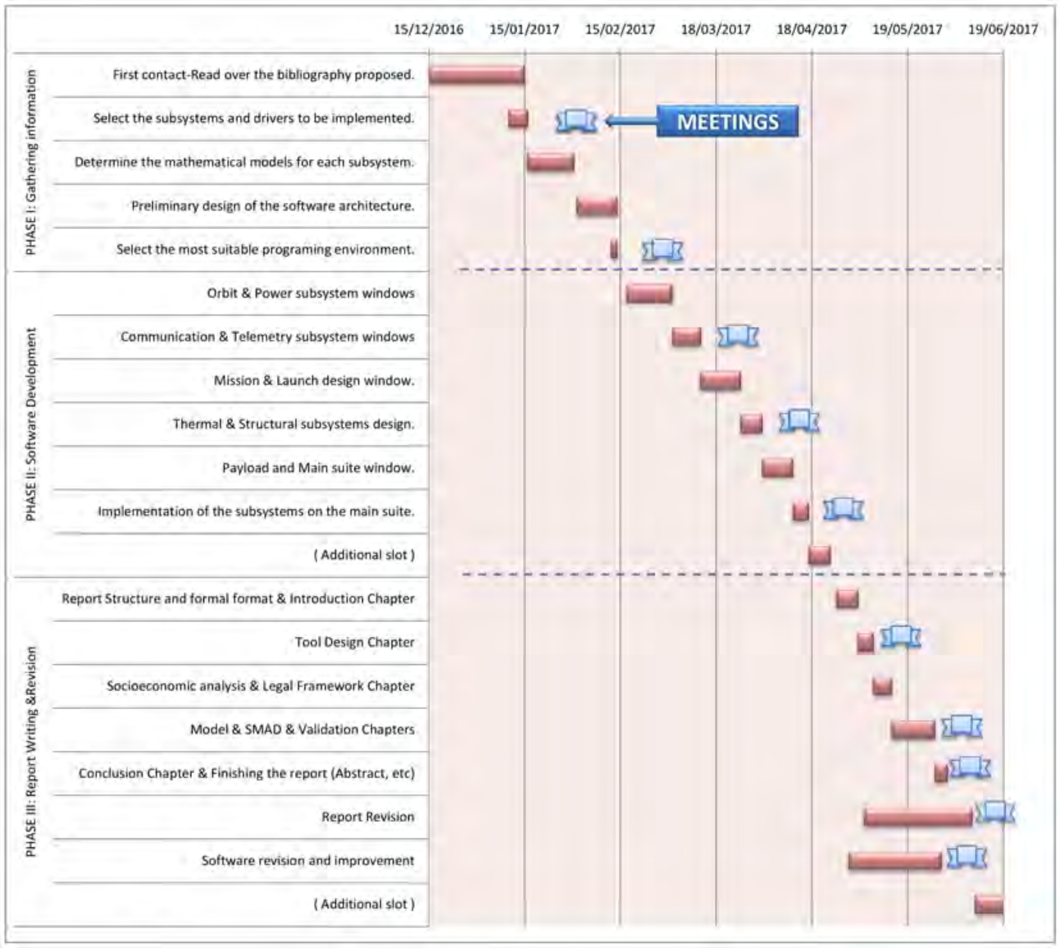


Figure 42: Project planning summarized in an Gantt Diagram

Regarding the economic budget, it is important to notice that costs are not split equally throughout the weeks. Certain elements such as the Licenses must be paid at the first stage while the Legal expenditure is paid at the end. In the same way the junior engineering works the same amount of hours each day while the senior engineering supervision will become more intense through the time. These considerations have been summarized in the budget stage distribution shown in figure 43.

Element	Phase 1	Phase 2	Phase 3	Total Cost
Junior Engineer	4.988 €	6.235 €	3.741 €	14.964 €
Senior Engineer Supervisor	264 €	423 €	582 €	1.269 €
Computer amortization	20 €	25 €	15 €	60 €
Matlab Academic License	500 €	0 €	0 €	500 €
Copyright fee	0 €	0 €	50 €	50 €
Project expenditure	115 €	115 €	115 €	345 €
Unforeseen expenses	630 €	788 €	472 €	1.890 €
TOTAL	6.517,20 €	7.585,70 €	4.975,00 €	**19.078 €**

Figure 43: Budget distribution throughout the project stages

7 LEGAL FRAMEWORK

The following chapter outlines the legal framework that currently rules the intellectually property protection as well as a detailed explanation of which is the most suitable option regarding the present project.

7.1 INTELLECTUALLY PROPERTY PROTECTION

The software development is one of the most active markets nowadays. Every year, hundreds of applications and industrial programs are launched in order to cover the actual necessities in each the sectors: security, financial, automotive, online-shopping, manufacturing, etc. For this reason, it has become a highly competitive market, continuously changing, that requires to optimize the resources and to evaluate the situation carefully in order to pursue with the most profitable option.

Consequently, in order to preserve the advantage in the market, companies take legal actions to protect the underlying ideas behind their projects. Depending on the nature and the commercial value of those ideas, the patent or the copyright protection will be taken.

In order to obtain a patent registration the application must fulfill several formal and substantive requirements, frequently technical and legally demanding; so they have noticeable costs associated. Furthermore, the application must be presented and approved in each country where it is aimed to obtain the protection. This protection has an average life between 20 and 30 years [11].

As a result of the current market situation outlined above, the patent is frequently dismissed in many projects due primarily to its complexity and the financial resources required to obtain and enforce it.

In these cases, the copyright protection becomes the most suitable option. Firstly, it does not depend on legal formalities regarding the register or the copies deposit, but it is substituted instead by the usual copyrights protection associated to any creation. Secondly, the protection is worldwide accepted. And thirdly, it lasts 50 years after the author's life. The main drawback is that the protection is limited to the expression or the software itself, so the idea is no longer protected [11].

Considering the above-mentioned points, the patent registration is not the best option for the current project for the following reasons:

- There is already similar software available in the market. Although these programs are not designed to cover the same industrial necessity, it is not an original concept but a newly perspective.

- The economical resources are strictly limited. In order to support that scale of the project it would be necessary to complement the thesis with a fully enterprise plan that would include, among other things, a complete market research, an implementation planning, obtain potential investors, etc.
- The cost associated to the patent registration can reduce the profitability expected in such amount that the project implementation may be unfeasible [3].
- The information, references and models obtained from the bibliography may represent a legal impediment to the patent acquisition. Thus, a legal study previous to the application must be mandatory.

Accordingly, the copyright will be the most suitable option to protect the intellectual property developed in the thesis.

7.2 LAW AND POLICY CONSIDERATIONS

The space activities are currently rule by strictly legislations both national and international. Once the engineer teams ultimate the technical solutions, lawyers spent the following months resolving the legal questions and complying with the applicable law. It concerns with the ownership, the salvage rights, liability, compensations, insurances, registration, communication law, intellectual property, remote sensing, environmental concerns, etc. [8]

Part of these issues are ruled by an International Space Law accepted by most of the countries through subsequent convections: *Outer Space Treaty*, *Liability Convention*, *International Cooperation*, *Registration*, *Rescue and Return Agreement*, ...

Nevertheless, the software developed throughout this thesis does not provide a fully detailed design, but an estimation to enhance the preliminary mission analysis. Therefore the software license should not have to deal with any of these legislations but only with those related with software development market.

8 CONCLUSION

The following chapter summarizes the ideas exposed throughout the present thesis and constitutes the end of this dissertation. The chapter was provided with an additional section containing the most relevant improvements to the software that can be accomplished in the future.

8.1 CONCLUSION

The space sector constitutes a technologically leader market, characterize by highly human, timing and economical consuming projects. Nowadays, the space industry is attracting once more the interest of the world on account of the latest achievements: *Mars exploration*, *first commercial space flights*, etc; therefore governments and private companies are making serious efforts to improve and optimize the invested resources.

As it has been proven, the alternative analysis is one of the most critical phases during a mission design since a proper selection would avoid future modifications and thus the associated cost and delays. Consequently, improvements on this initial stage like the software tool developed during this thesis would be well regarded by space companies.

Considering the scale of the SMA, the approach of the project has been a constant issue that required frequently to be reassessed. In fact, the initial decision to focus the SW Suite towards Earth orbiting satellite mission has proven to be crucial to constrain the scope of the project. Even though, as more subsystems were implemented, the new complexity leads to impose more constrains to initial concept. It proves that it would have been more efficient to spend more time at the beginning of the project establishing the scope of the work and evaluating the degree of complexity that the implementation of new subsystems will add to the already functional software.

Regarding those requirements established beforehand, it is important to recall the fact that the strength of the final design of the SW Suite interface lies on their easy-to-handle interface that facilitates in a great extent the trade off process among the subsystems. The software tool has proven to be able to deal with a wide range of missions, therefore fulfilling two of the initial requirements: *high level of flexibility* and *great versatility*. The weakest point has been demonstrated to be the level of accuracy of the results, which is detrimental for the reliability required for this type of software missions. The reason for that may be found on the fact that the software tool has been driven mainly by the easy-to-handle requirement, looking forward to guarantee the speed and the efficiency required for this type of software.

Consequently, the improvement of the models accuracy constitutes one of the most relevant working lines for the future.

Regarding the project stages, it is important to valuate positively the time spent on gathering information, which became extremely useful during the reassessment of the project approach to determine if the new element relevancy deserved the time required to implement it.

On the other hand, the process to validate the software tool was not performed in the desired extent. As explained during the validation chapter, the lack of information available to model a space mission prevent from executing the validation process as it was intended; forcing instead to come up with a new method to obtain the results. As it can be inferred, the validation process requires an exhaustive search and data collection beforehand, hence having the engineering data required to perform the analysis properly.

In summary, the work behind the thesis project has culminated in the development of an excellent SW Suite, whose implementation on the space industry market is a feasible possibility, expecting to save a significant amount of company resources. The attached report collects as well all the knowledge achieved during the project development.

8.2 FUTURE IMPROVEMENTS

In spite of the great effort made developing the software and the successful of the suite environment, there are still numerous fields where the SW Suite could be improved. Therefore the following lines depict the upgrading plan to be accomplished in the future. Most of these modifications are related with software limitations mentioned throughout the chapters.

I The most significant enhancement in the SW Suite can be performed in the Structure design window. By implementing a simple 2D or 3D allocation method, that model each component as simple geometrical objects, it would be possible to obtain an estimation of the dimensions of the satellite. Based on this estimation, it would be possible to model the satellite orientation, estimate the area pointing toward the Sun, the Earth, etc; the center of mass, the moments of inertia and any other relevant parameters. Doing so, the subsystems that provide the worst interfaces and the less accurate results will be greatly improved, as described below.

II The ADCS design window offers a wide margin for improvement. Firstly, instead of providing standard sensors and actuators it would be more attractive to be able to size each component according to the mission. Secondly, if the structural design window is implemented with the 3D location method described above, the accuracy of the magnitudes of the torques would be improved and it would be possible to analyze the dynamic problem in each axis.

III Similar to the ADCS case, the thermal subsystem design would be greatly improved with the new structural design window. It would be possible to locate the thermal control components in the most critical areas and therefore accomplish a much accurate design.

IV One of the most interesting enhancements is the implementation of an orbit perturbation design window. In this new window, the user may be able to model the effect of all the possible external forces that may disturb the nominal orbit. Based on this estimations the user will be able to size the propulsion system accordingly to the maneuvers required to maintain the orbit.

V Incorporate an specific propulsion system design window where the user can model mission maneuvers (reorientation, orbit change,. . .) and develop fully detailed budget of the mass propellant, including not only those maneuvers, but also the transfer impulse from the parking orbit, the orbit maintenance propellant (estimated in the Orbit Perturbation design window) and add extra propellant to cope with eventualities. Additionally the design window may allow modeling a generic propulsion system, introducing the engine mass and the total impulse achieved. The design window may be provided with the possibility to model low-impulse maneuvers performed with non-chemical thrusters.

VI Incorporate an analysis of the eclipse evolution throughout the year, providing a visual graphic of the beta angle variation and allowing the user to select the duration of the eclipse.

VII Regarding the power subsystem there are two main improvements to be incorporated. Firstly, it will be possible to complete the design window by including the possibility to select other type of power sources (nuclear, fuel cells, etc). Secondly, the power subsystem could be modeled with the option of fixing the size of the power sources and thus obtaining the total power available. This option would be extremely useful to control the maximum mass of the power subsystem. VIII. The communication subsystem will be improved by allowing the user to select the type of antenna and determine their parameters.

These are the most relevant modifications to be accomplished in a future software version. Notice that the enhancements provided are not ordered considering their relevance, since their implementation would depend mainly on the future interests and the space mission fields intended to be improved.

Part IV

APPENDIX

A

APPENDIX A

- Mass, cost and power computation parameters -

	Propulsion System	Payload	Power Subsystem			Communication Subsystem	Telemetry Subsystem	Launch Subsystem
			Solar Array	Secondary battery	DR&C Power systems			
mass	Tsiolkovsky equation	- U.i.-	$m = C_m * P$ $C_m(kg/W)$ = U.i.	$m = ED * P$ $ED(Wh/kg)$ = U.i.	$m = 0{,}17 \cdot m_{tot}$	- U.i.-*	- U.i.-*	-
cost	$c = C_e \cdot I_{sp} \cdot F$ SRM: $C_e = 0{,}034$ \$/N·s LM: $C_e = 0{,}09$ \$/N·s	- U.i.-	$c = C_c * P$ $C_c(\$/W)$ = U.i.	$c = C_p * P$ $C_p(\$/Wh)$ = U.i.	$c = 0{,}21 \cdot m_{tot}$	- U.i.-*	- U.i.-*	$c = C_m * m_{sat}$ $C_m(\$/kg)$ = U.i.
power	-	- U.i.-	-	-	-	Link Balance equation	Link Balance equation	-
Allows extra m/cost/power	- / - / -	- / - / -	Yes / Yes / -			Yes / Yes / Yes	Yes / Yes / Yes	- / - / -

	ADCS Subsystem							
	Sun Sensors	Earth Sensors	Magnetometers	Star Sensors	Magnetometers	Star Sensors	Star Sensors	Gyro / IMU
mass (kg/u.)	1,05	2	0,8	3,5	7,5	11	10	25,2
power (W/u.)	1,5	2,7	0,8	12,5	100	55	120	8,3
cost (\$/kg)	$c = C_m \cdot m_{tot}$; where C_m = 340 \$/kg							
Allows extra m/cost/power	- / -							

	Thermal Subsystem		Structural Subsystem		
	Radiator	Louvers	Primary	Secondary	Fasteners
mass	$m = C_m * P$ $C_m(kg/W) = 0{,}035$	$m = C_t \cdot (T-8)+1$ $C_t = 0{,}15625$ kg/ºC	$m = C_p \cdot m_{tot}$ $C_p(\%)$ = U.i.	$m = C_s \cdot m_{tot}$ $C_s(\%)$ = U.i.	$m = C_f \cdot m_{tot}$ $C_f(\%)$ = U.i.
power	- U.i.-	-	-	-	-
mass	$c = a*(m_{thermal}+m_{struct})$^b where a = 646; b = 0,684				
Allows extra m/cost/power	- / - / -		Yes / - / -		

*For every link
U.i. = User Input
"Power" refers to power consumption

B
APPENDIX B

Requirement	FireSat II	Factor that conditioned the Requirement
Functional Requirements		
Performance	User needs, payload aperture, orbit	Work through light clouds 50m resolution 1 km geolocation accuracy
Coverage	Orbit altitude and inclination, swath width, scheduling	Coverage of specified foest areas within the US at least twice daily
Interpretation	Cloud cover, image quality	Identify an emerging forest fire withing 8 hours; with less than 10% of false positives
Timeliness	Interpretation, communications, processing, operations	Interpret data to end user within 5 min
Secondary Missions	As above	Monitor changes in mean forest temp. to 2 ºC
Operational Requirements		
Commanding	Who will do commanding, tasking from the field, need for real-time schedule changes	Commandable within 3 min of events, download units of stored coverage areas
Mission Design Life	Duration of need, level of redundancy	8 years
System Availability	Level of redundancy, where processing and interpretation occur	95% excluding weather 24 hours maximum downtime
Survivability	Orbit, hardening	Natural environment; not in radiaton belts
Data Distribution	User needs, communications architecture, ancillary communication channels	Up to 500 fire-monitoring offices 2000 rangers worldwide (max 100 simultaneous users)
Data Content, Form and Format	User equipment, available bandwidth, level and place of processing	Location and extent in lat/long for local plotting, avg. temp. for each 40m2
User Equipment	Mass, size, power, existing equipment, user interface	10x20 cm display with zoom and touch controls, built-in GPS quality map
Constrains		
Cost	Manned flight, size, complexity, orbit	Non-recurring <10m$ Recurring < 3m$/year
Schedule	Financing, technical readiness	Operational within 3 years
Risk	Primary and secondary customers, schedule, cost	Probability of success >90%
Regulations	Law, policy	Orbital debris, civil program regulations
Political	Sponsor, wheter international program, debris removal	Responsive to public demand for action
Environment	Orbit, lifetime	Natural
Interfaces	Level of user and operator infrastructure	Interoperable through NOAA ground stations

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-End of the thesis-

The software developed throughout the thesis can be found in the CD attached to the back cover of the printed edition. It will be also available in the Uc3m Library webpage by searching the title or the author in the e-Archivo: http://e-archivo.uc3m.es/handle/10016/15439

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