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Mission Analysis and Design of MECSE Nanosatellite

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"Any intelligent fool can make things bigger,
more complex, and more violent. It takes a
touch of genius—and a lot of courage—
to move in the opposite direction."
Albert Einstein

Resumo

Desde o começo da aventura da humanidade no espaço que os problemas associados ao período de blackout de comunicações são uma questão por resolver. Durante este período, o veículo espacial perde toda a comunicação com o centro de controlo ou satélite, incluindo voz, dados de telemetria em tempo real e navegação GNSS. Uma vez que a comunicação contínua é um fator crítico para garantir a segurança e o sucesso de missões espaciais tripuladas e não tripuladas, torna-se essencial encontrar soluções para a mitigação do blackout de comunicações. De facto, estas soluções são de extrema importância e já consideradas um requisito no desenvolvimento de futuros veículos espaciais. Uma solução é a utilização de um campo eletromagnético para manipular a camada de plasma que se forma em volta do veículo.

Nesta tese de mestrado, uma inovadora missão CubeSat para a manipulação do plasma ionosférico é proposta e projetada. MECSE (Experimento de Magneto/Electro hidrodinâmica em Cubesat) tem o objetivo de provar no espaço que a densidade eletrónica da camada de plasma pode ser reduzida através da geração de um campo eletromagnético.

De uma perspetiva de engenharia de sistemas, as fases inicias da missão MECSE são projetadas (fases 0, A e B1 do ciclo de vida da ESA). Começando por uma caracterização da missão, o caso científico é apresentado e a viabilidade da missão é estudada com base em métodos de exploração científica e tecnológica. De seguida, os objetivos de missão, requisitos e figuras de mérito são definidos. A análise de missão é feita considerando uma órbita referência baseada em pesquisa de lançamentos. No fim, um design preliminar do satélite é apresentado incluindo as análises realizadas para os subsistemas, o conceito de operações e a definição dos requisitos de sistema.

Esta tese de mestrado foca-se ainda em estudar a previsão do tempo de vida orbital de um CubeSat. O impacto de usar diferentes modelos recomendados pelas diretrizes standard para a atividade solar e geomagnética é investigado usando STK e DRAMA softwares e comparado com dados históricos de CubeSats que já reentraram. É concluído que ainda existem enormes variações nos resultados de diferentes modelos e que os parâmetros de satélite recomendados pelas directrizes não são adequados para prever o tempo de vida orbital com precisão. O tempo de vida do satélite MECSE é previsto e os efeitos de variações em parâmetros orbitais e de satélite são avaliados.

Palavras-chave

Blackout de Comunicações; Manipulação Electromagnética; Plasma; Re-entrada; Análise de Missão; Design de Missão; Engenharia de Sistemas; CubeSat; Redução da Densidade Eletrónica; Janela Magnética; Análise Orbital; Deisgn Preliminar; Drama; STK; Ciclo de Vida; Satélite

Abstract

Since the moment humankind started venturing into the realms of space, the problems associated with Radio Frequency (RF) blackout period due to plasma sheath interactions with the spacecraft have been an unsolved issue. During this period, the spacecraft loses all the communication with the control center or satellite including voice, real-time data telemetry and GNSS navigation. Considering that continuous communication during atmospheric re-entry is crucial to ensure safety and accomplishment of manned and unmanned space missions, solutions for the mitigation of RF blackout are of high priority and a requirement for the design of future space vehicles. One solution is the use of an electromagnetic field to manipulate the plasma layer surrounding the vehicle.

In this M.Sc. thesis, an innovative CubeSat mission for the manipulation of ionospheric plasma is proposed and designed. MECSE (Magneto/Electro hydrodynamics CubeSat Experiment) aims to confirm in space that the electron density of the plasma layer can be reduced through the generation of an electromagnetic field.

From a systems engineering perspective, the early phases of MECSE mission are fully designed (phases 0, A and B1 of ESA's project lifecycle). Starting with mission characterization, the scientific case is presented and the feasibility of the mission is studied based on tradespace exploration methods. Then, the mission objectives, requirements and figures of merit are defined. The mission analysis is performed considering a reference orbit from a launch survey. In the end, a preliminary design of the spacecraft is presented including the analyses performed for the subsystems, the concept of operations and the definition of system requirements.

This M.Sc. thesis also focusses on the study of orbital lifetime predictions for a CubeSat. The impact of using different solar and geomagnetic activity models proposed by standard guidelines is investigated using STK and DRAMA software and compared against historical data from already decayed CubeSats. It is concluded that there are still large deviations between the results provided by different models and that the satellite parameters recommended by the guidelines are not suitable when predicting accurately the orbital lifetime of a CubeSat. The orbital lifetime of MECSE nanosatellite is predicted and the effects of variations in orbital and satellite parameters are evaluated.

Keywords

Radio Frequency Blackout; Electromagnetic Manipulation; Plasma Layer; Re-entry; Mission Analysis; Mission Design; Systems Engineering; CubeSat; Electron Density Reduction; Magnetic Window; Orbital Lifetime; Project Life Cycle; Preliminary Design; DRAMA; STK; Nanosatellite

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List of Acronyms

AGI	Analytical Graphics Incorporated
AOCS	Attitude and Orbit Control System
AWG	American Wire Gauge
BC	Ballistic Coefficient
CDH	Command and Data Handling
CEiiA	Centre of Engineering and Product Development
C-MAST	Center for Mechanical and Aerospace Science and Technologies
COTS	Commercial Of The Shelf
DLm	DownLink Mode
DRAMA	Debris Risk Assessment and Mitigation Analysis
ECSS	European Cooperation for Space Standardization
EDR	Electron Density Reduction
EHD	ElectroHydroDynamics
EMG	ElectroMagnetic Generator
EPS	Electrical Power System
ESA	European Space Agency
FEMM	Finite Element Method Magnetics
FF	Form Factor
FOCUS-1A	Fast Orbit Computation Utility Software
GEM-T1	Goddard Earth Model
GNSS	Global Navigation Satellite System
GPS	Global Position System
GS	Ground Station
ID	IDentification
ISO	International Organization for Standardization
ISS	International Space Station
KISS	Keep It Simple and Short
LEO	Low Earth Orbit
LEOP	Launch and Early Orbit Phase
LP	Langmuir Probe
LPN	Latest PredictioN
MDR	Mission Design Review
MECSE	Magneto/Electrohydrodynamics CubeSat Experiment
MHD	MagnetoHydroDynamics
mNLP	Multi Needle Langmuir Probe
MO	Mission Objective
MR	Mission Requirement

MSc	Master of Science
MSS	Mechanical System and Structures
NASA	National Aeronautics and Space Administration
OL	Orbital Lifetime
OREX	Orbital Re-entry Experiment
OSCAR	Orbital Spacecraft Active Removal
PDS	Plasma Dynamics Study
PDSm	Plasma Dynamics Study Mode
PL	Payload
PLME	Plasma Layer Mitigation Experiment
PLMEm	Plasma Layer Mitigation Experiment Mode
PRR	Preliminary Requirements Review
RAAN	Right Ascension of the Ascending Node
RAM	Radio Attenuation Measurements
RF	Radio Frequency
RPY	Roll, Pitch and Yaw
S/C	SpaceCraft
SFm	SaFe Mode
SMO	Secondary Mission Objective
SO	Scientific Objective
SRP	Solar Radiation Pressure
SRR	System Requirements Review
SSO	Sun Synchronous Orbit
STEM	Science, Technology, Education and Mathematics
STK	Systems Tool Kit
T	Tesla
TBC	To Be Confirmed
TBD	To Be Determined
TCS	Thermal Control System
TPS	Thermal Protection System
TRL	Technology Readiness Level
TTC	Telemetry Tracking and Control
U	CubeSat Unit
UBI	University of Beira Interior

Nomenclature

a	Semi-major axis
A	Cross Sectional Area of the Satellite
a_r	Acceleration due to Solar Radiation Pressure
A_{SRP}	Area of Solar Radiation Pressure
B	Magnetic Field Intensity
C_D	Drag Coefficient
C_R	Solar Radiation Pressure Coefficient
D	Drag Force
e	Eccentricity
$F_{10.7}$	Solar Radio Flux Index
f_p	Plasma Frequency
f_{radio}	Radio Frequency
G	Gravitational Constant
i	Inclination
I	Current of the EMG
I_c	Current Flow through mNLP
K_n	Knudsen Number
l	Length of mNLP probe
L_c	Characteristic Length
L_{coil}	Length of the EMG
m	Mass of the Satellite
M	Mass of the Earth
N	Numbers of turns
P	Orbital Period
p	Pressure
q	Electron Charge
r	Radius of the mNLP
T	Temperature
V	Voltage of the mNLP
V_s	Satellite Orbital Velocity

Greek letters

β	Beta Angle
η_0	Initial Electron Density of Plasma
$\eta_{critical}$	Critical Electron Density of Plasma
η_e	Final Electron Density of Plasma
λ	Length of the Molecules of a Fluid
μ	Magnetic Permeability
ρ	Atmospheric Density
ω	Right Ascension of the Ascending Node
ν	True Anomaly

Chapter 1

1 Introduction

1.1 Personal Motivation

Science and technology drive the modern world and space is doubtless at the forefront. Ever since humankind has been aware of the broad expanse of the universe, the desire to explore it has stimulated scientists and thinkers alike. In fact, exploration is the most sublime expression of what it is to be human as it is driven by Man's intense desire to satisfy their own curiosity.

Space exploration is a proxy for society's urge to innovate [1]. As a direct result of the immense knowledge that it has already delivered, space technologies have become increasingly integrated into everyday life so profoundly that modern society would not be possible without them. Weather forecasting, telecommunications, navigation, television, remote sensing and national security are only the most visible space technologies that humanity relies on, though spin-offs and technology transfers from space to non-space sectors provide many additional indirect benefits [2]. Thereupon, it is a rock-solid guarantee that investing in space leads to innovations that have far-ranging benefits to society [1].

Innovation and technology are high priority themes on every nation's agenda considering that today's advanced economies rely on the capacity to develop knowledge and on the productivity to drive growth. Therefore, innovation is central to Portugal's future success. To such a degree, space is an innovation driver, since it has no frontiers and remains an exceptionally difficult domain of human endeavor. Space activities are an attempt to reach out for an unreachable goal, the fulfillment of one's dreams and ambitions. Space is about the will to make one's dreams materialize, to measure one's intellect against the final frontier [2], [3].

Moreover, space exploration spurs team-work among experts from different fields of study. This cross-pollination of sciences always stimulates innovation and readily encourages revolutionary discoveries [3]. Few other endeavors combine this interdisciplinary focus nor address the same challenges as space exploration. On that account, space projects are a highway to the progress of knowledge enhancing valuable competencies and increasing the competitiveness in science and technology.

Apart from all those reasons, exploratory space activities have the power to revitalize the latent Portuguese spirit of discovery, search, and pride. Indeed, space has the unique capacity to inspire and motivate a new generation to tackle the tough academic subjects required not just to undertake a robust space program, but to secure the Portuguese future as well [1], [2].

This vision can guide a renewed interest in the academic disciplines of Science, Technology, Engineering, and Mathematics (STEM). Plus, engaging students in these fields becomes essential when preparing the future Portuguese generations to meet the challenges and opportunities of tomorrow which are defined by complexity and multidisciplinarity [2], [3].

In such way, space engineering is deeply connected with STEM education since it demands an interdisciplinary approach to real-world problems [4]. It sharpens technical and personal skills related to the design process, which are directly linked with critical thinking, problem-solving, and teamwork. Also, space hands-on activities have the power of endorsing direct contact with technology, one of the most effective teaching practices [4], [5].

In the light of this matter, the Magnetohydrodynamics / Electrohydrodynamics CubeSat Experiment (MECSE) project endorses these beliefs in exactness. On the one hand, MECSE consists in a CubeSat space mission designed mainly by students, which will develop expertise and inspire future generations to pursue space careers. On the other hand, MECSE aims to innovate and revolutionize the aerospace sector globally by aspiring to help finding the solution for a fundamental problem arising during hypersonic flight and Earth's atmospheric re-entry, the communication blackout.

To achieve this, MECSE will confirm the theory that an electromagnetic field can re-shape the plasma layer surrounding the spacecraft which is the main cause for the communication blackout during the atmospheric re-entry phase [6], [7]. If deemed successful, the outcomes of the project will have high impact in scientific and technological terms [6]-[19], fostering and increasing the competitiveness of the Portugal's knowledge-based economy.

Bearing all that in mind, the author of this M.Sc. thesis aims to, more than just demonstrating the knowledge to design the early phases of an innovative and revolutionary space project, light again a flame in the Portuguese spirit of exploration by triggering the curiosity for space sciences and engineering among the Portuguese youth. By architecting a space mission from the ground up, the author intends to show that space projects, complex as they may seem, are within reach of everyone who is decided to.

1.2 Purpose of MECSE Project

MECSE is a student-driven project with scientific purposes. The project aims to advance the research on the mitigation of Radio Frequency (RF) blackout by designing a nanosatellite based on a standardized modular platform (CubeSat) while giving students the opportunity to enroll in a space project. There are a number of reasons to develop such innovative space.

Firstly, the mitigation of the RF blackout is a crucial requirement in the design of re-entry space vehicles, considering that continuous communications, real-time telemetry, and GNSS signal reception are critical parameters that ensure safety and accomplishment of both manned and unmanned space missions. Therefore, solutions that might solve or attenuate this problem are of high priority in scientific and technological terms [6]–[19].

Secondly, C-MAST, a Center for Mechanical and Aerospace Science and Technologies based at University of Beira Interior (UBI), is developing and validating a Magnetohydrodynamics (MHD) numerical model for assisting in the design of re-entry objects with emphasis on radio blackout mitigation mechanisms and plasma layer manipulation [13], [14]. When validated, the numerical framework will assist in the development of efficient MagnetoHydroDynamics / ElectroHydroDynamics (MHD/EHD) approaches for manipulating the plasma flow. In this perspective, the results of the MECSE experiment will create the basis for a more rigorous study on electromagnetic manipulation of plasma and the possible development of the technology which will eventually allow bypassing the RF blackout completely.

Thirdly, CEiiA, a Centre of Engineering and Product Development, based in Matosinhos, that designs, implements and operates innovative products and systems for technology intensive markets, has recently increased its activity in space-related fields. CEiiA has the vision of establishing Portugal as a reference in the research, development and engineering fields by creating the conditions for a world-class innovation ecosystem. In such way, CEiiA was challenged by the innovative nature and complexity of the MECSE project, partnering with UBI to promote such a unique endeavor. CEiiA has the fundamental role of materializing the mission by creating the bridge between the scientific knowledge and the design of the space system.

Finally, a CubeSat program is a powerful educational tool and technology driver with enormous potential among the commercial market since it allows innovation to occur in a quick manner. Indeed, small spacecraft missions play a compelling role in space-based scientific and engineering programs as they tend to be extremely responsive to new opportunities and technological needs [20]–[22]. Moreover, the CubeSat standard is a true disruptor of the space industry since it is an ideal solution for a cost effective and fast access to space [23]. Concerning this last point of view, MECSE project has the power of fostering the Portuguese space industry by inspiring both institutions to engage in a Cubesat development program.

1.3 Research Objectives and Contributions

The work presented in this master thesis serves two main purposes. Firstly, it aims to perform investigation within space mission analysis and design field of knowledge. Secondly, as a part of MECSE project, it aims to be able to contribute actively for the progress of the project.

The goal is to perform the mission design of MECSE project. That means to prepare the preliminary stages of the project life cycle which includes defining the mission, analyzing it and starting the design of the satellite. Note that the project management tasks such as cost analysis and project planning are not part of this thesis.

The following objectives were defined for this research:

- Investigate the scientific theme of RF Blackout through literature review and formulate the scientific case for the MECSE mission;
- Investigate the feasibility of performing a mission to study the mitigation of RF Blackout within a CubeSat nanosatellite;
- Identify the mission needs and propose alternative mission scenarios for MECSE mission that can be technically feasible within an educational context and valuable for the scientific research being conducted at UBI;
- Perform trade studies to evaluate the feasibility of alternative mission scenarios and select the most suitable one considering technical feasibility and scientific value;
- Define clearly the mission aim, objectives and requirements as well as identify mission parameters that have the most impact for the mission design;
- Perform the mission analysis of MECSE mission which includes trajectory and orbital analyses;
- Investigate the impact of different solar and geomagnetic activity modeling approaches on CubeSat orbital lifetime predictions and validate them against observed orbital lifetimes from former CubeSat missions;
- Evaluate the impact of variations on the satellite and orbital parameters in the orbital lifetime of MECSE satellite and provide a range of possible orbits that could be suitable for MECSE mission;
- Propose a preliminary design of the satellite and develop the concept of operations;
- Propose future work to be developed in the future phases for each subsystem.

Regarding the contributions of this work for the MECSE project, it is expected that in the end the mission must be already in the phase B of the project lifecycle from a systems engineering technical point of view. Therefore, it shall be ready for the Mission Design Review (MDR), Preliminary Requirements Review (PRR) and System Requirements Review (SRR).

1.4 Thesis Outline

This thesis is structured in a coherent and logical manner. The description of each chapter within this document is presented below:

Chapter 1 introduces the author's motivation to design a space mission as well as the purpose and contributions of the project to UBI, CEiiA, the Portuguese Space Program and the overall scientific community. It also presents the research objectives expected to be achieved during this investigation and the new contributions of this work to the MECSE project.

Chapter 2 provides a theoretical introduction of space systems presenting the CubeSat concept and its high importance for the advancements in education, science and industry fields. Afterwards, an investigation about the scientific theme is shown and a revision of state-of-the-art former space missions is presented. In the end, the fundamentals of space mission engineering are explained with focus on the guidelines used for the design of the MECSE space mission. Finally, the space mission engineering process to be used is shown.

Chapter 3 refers to the characterization of MECSE mission. Here, the scientific case is formulated based on the literature review and the scientific research at UBI, the mission needs are identified and alternative mission scenarios are proposed. Then, an evaluation is performed through trade studies to select the most suitable one. In the end, a preliminary feasibility study is carried out based on a point design approach.

In Chapter 4, the mission is defined. This means to define the mission statement, objectives and requirements as well as to identify the figures of merit and the mission parameters. This means the end of phase 0 activities for MECSE project.

Chapter 5 presents the mission analysis of MECSE mission as well as a deep investigation about the impact of different solar activity modeling methods in the orbital lifetime predictions of a triple CubeSat. Firstly, a theoretical background about astrodynamics is presented and the methodologies used for the orbital analyses in this thesis are introduced. Afterwards, trajectory and orbital analyses are carried out to design the mission profile and evaluate the following mission parameters: launch opportunities, orbital lifetime, and access and eclipse times.

In Chapter 6, the author proposes a conceptual design of the space segment. For this purpose, the system architecture and the concept of operations are presented and the system is broken down into subsystems. For each subsystem, a preliminary analysis is performed and the system requirements are defined. This marks the end of phase B1 for MECSE project.

Finally, Chapter 7 presents the conclusions drawn from the mission analysis and the system design of MECSE mission and proposes future work to be performed by the project team.

Chapter 2

2 Literature Review

To better understand the scope of this M.Sc. thesis it is essential to first understand the capabilities of space systems, particularly small satellites, as well as to recognize the importance of systems engineering when designing a space mission. It is also critical to investigate the scientific theme, which is one of the goals of this work, and to be aware of the prominence associated with the RF blackout mitigation.

2.1 The Rise of Small Satellites

2.1.1 Review of Space Systems

In the context of spaceflight, an artificial satellite is usually referred as an object intentionally placed into orbit. The historic launch of Sputnik 1 in 1957 marked the beginning of the space age. Since then, satellite benefits rippled through society and hundreds are now launched every year for a variety of purposes. In fact, satellite applications have become essentially for our daily life activities on Earth [22], [24].

The variety of satellites is extremely ample depending particularly on the function for which it is designed for. Nevertheless, it is important to primarily recognize that the satellite itself is only a part of a larger system. Typically, a space system can be divided into three segments (see Figure 2. 1): the space segment, the launch segment and the ground segment [24].

The launch vehicles transport the spacecraft into orbit. While in orbit, the spacecraft performs the mission objectives and gets in contact with a ground segment. This consists on control and operation centers that need to be able to command the spacecraft as well as store, process and distribute the data for the end users. Concerning the space segment, it can be divided into two modules: the payload that will accomplish the mission objectives, and the service module (or bus) that provides the infrastructure for operating the payload.

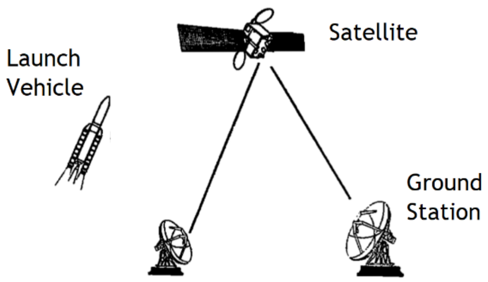


Figure 2. 1 The space system (from [24]).

Given the diversity of satellites, they are often classified by their mission and by their mass. The mission stands for the reason the satellite was designed for, that means its function, which is imposed by the needs of the user. Figure 2. 2 shows the wide range of space missions and applications with some examples of spacecraft. Some missions fall into multiple categories [25], which will be the case of MECSE mission.

	Communications and Navigation	Applications	Science	Education and Training	Exploration	Resource Utilization	Military	Other
Applications	Telephone	Weather	Earth monitoring	CubeSats	Moon	Materials processing	Surveillance	Space tourism
	TV (commercial)	Earth resources	Telescopes (visible, UV, IR, X-Ray, gamma ray, Radio)	Training missions	Planets and Satellites	Solar power satellites	Reconnaissance	Burial in space
	TV (direct broadcast)	Fire detection	Particle detection	Educational satellites	Asteroids	Asteroid mining	Tactical	Space Colonies
	Satellite Radio	Oceanography	Magnetic fields	Student viewing and tracking	Comets	Atmosphere mining	Comm.	Lunar colonization
	Store and Forward	Disaster monitoring	Planetary		Beyond the Solar System	Lunar resource utilization	Space surveillance	Mars colonies
	Space-to-Space	Search and Rescue	SETI		Human and unmanned fly-bys, orbiters,	Mars in-situ propellants	Missile warning	Space Stations
	Relay Satellites	Crop monitoring	Solar		Surface Rovers	He^3 from Lunar Regolith	Nuclear detection	Technology demonstration
	Military Comm	Global warming	Biological		Lunar/Mars sample return		ELINT	Technology tests
	Navigation (air, ships, cars, people)	Warning of space hazards	Materials research				SIGINT	Spacecraft repair, refurbishment,
	Amateur Radio	Commercial cargo/vehicle tracking					IMINT	Tugs, OTVs
		Ice flows					SDI	Tethers
		Orbital debris monitoring					Space-based laser	Prayer Wheel
							ASATs	Interstellar Travel
							Wind measurements	
Examples	IntelSat	LandSat	Hubble	StarShine	Apollo	Industrial Space Facilty	DSP	Sputnik
	Direct TV	SPOT	Chandra	EDUSAT	Galilio	SPS	FLTSATCOM	Celestis
	OrbComm	SeaSat	COBE	AAUSat	Cassini		MILSTAR	ISS
	Iridium	NPOES	IRAS	OUFTI-1	Voyager		GPS	MIR
	GlobalStar	GOES	JWST	PLUME	Mariner		BSTS	SkyLab

Figure 2. 2 - The wide range of space missions (from [21]).

Concerning the mass [24], the different classes are presented in Table 2. 1.

Table 2. 1 - Classification of spacecraft by the mass.

Class	Mass Range (kg)
Conventional large satellites	>1000
Conventional small satellites	500-1000
Minisatellite	100-500
Microsatellite	10-100
Nanosatellite	1-10
Picosatellite	0.1-1
Femtosatellite	<0.1

2.1.2 The CubeSat Concept

Traditionally, the space industry produced only large and complex spacecraft which required significant resources and expertise within the reach of only a few government-backed space agencies such as the National Aeronautics and Space Administration (NASA) and the European Space Agency (ESA) among others [22]. The issue with those missions is that they are associated with very high investments. So, new concepts and ideas are rarely accepted because they would increase significantly the risk of mission failure. This holds back innovation [22], [25].

For this reason, there was the need to develop a new space program which would allow people with little experience in the design of space missions to start with an open mind and incorporate innovative ideas into designs without the fear of failure [25], [26]. In fact, without pushing the boundaries of knowledge, innovation cannot occur [1]. Furthermore, there was the need to resort to the current advances in microelectronics, software, and material science in order to create lower-cost and more responsive systems. In short, combine the modern technology with old-fashioned drive, determination and some willingness to accept risk which would allow doing much more, much faster, with fewer resources [25].

Subsequently, this trend has inspired the rise of small satellites and eventually the development of the CubeSat concept, a standardized subclass of small satellites. The CubeSat standard was created by Stanford and California Polytechnic State Universities in 1999, and it specifies that a standard Form Factor (FF) of 1U unit represents a 10-centimeter cube (10×10×10 cm^3) with a mass of up to 1.33 kg [22]. As it can be seen in Figure 2. 3, a 1U CubeSat could either serve as a standalone satellite or could be combined together to build a larger spacecraft.

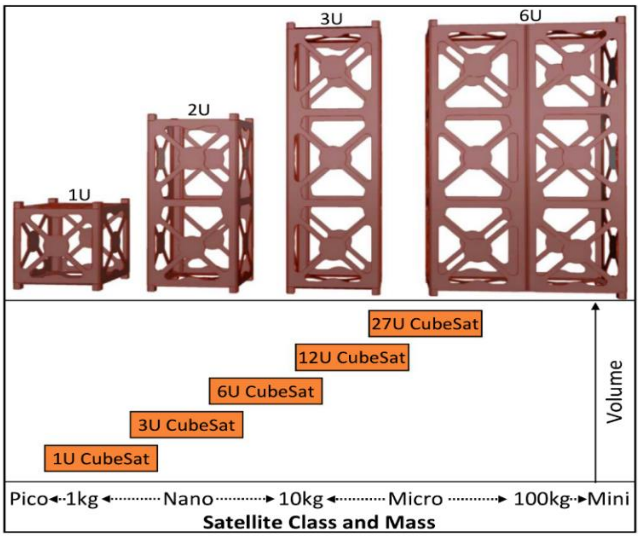


Figure 2. 3 – Small satellite classification with respect to the CubeSat FF standard (from [20]).

The standardization promotes a highly modular, highly integrated system where satellite components are available as “Commercial Off The Shelf” (COTS) products from several different suppliers and can be combined according to the needs of the mission. Moreover, it allows CubeSats to be launched as secondary payloads (piggybacks) within a standardized deployment system. This simplifies the accommodation on the launcher and minimizes flight safety issues, increasing the number of launch opportunities and, thus, decreasing the launch costs. Due to these features, CubeSats can also be readied for flight on a much more rapid basis compared to traditional spacecraft. This accelerated schedule allows students from universities with a CubeSat program to be involved in the complete life cycle of a mission [20], [21].

CubeSats were initially envisioned as educational tools or technology demonstration platforms. However, both the scientific community and the commercial space industry are starting to realize its enormous potential value in terms of high-quality scientific research and economic revenue. Indeed, in the last decade there has been a substantial boom in their development and the future perspectives are to persevere this growing tendency (Figure 2. 4) [22], [27].

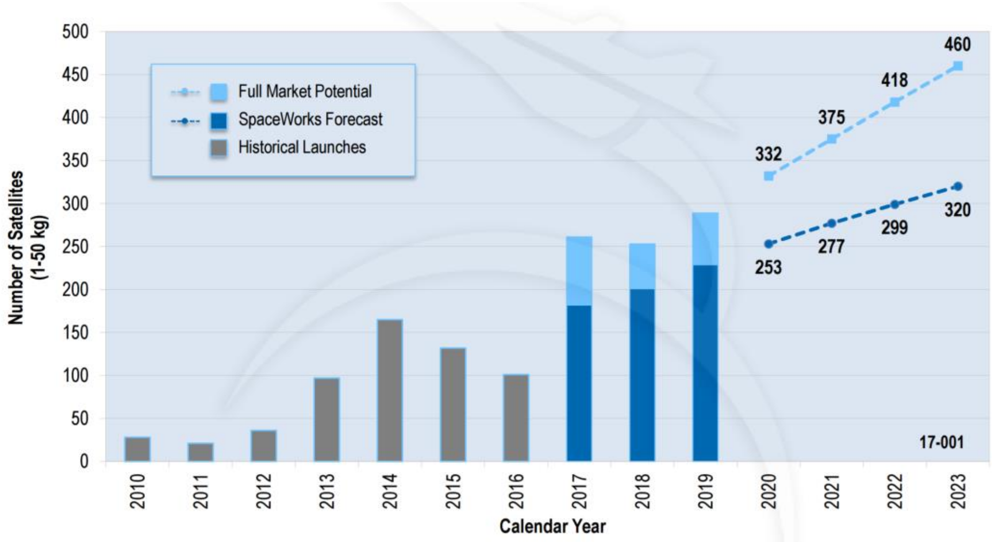


Figure 2. 4 - Nano/microsatellite launch history and forecast (1 - 50 kg) (from [27]).

In a nutshell, CubeSat program will certainly play a vital role in future space activities, providing space access to small countries, educational institutions, and commercial organizations around the world by allowing them to develop and launch their own spacecraft with relatively low-cost budgets. Furthermore, readily available inexpensive COTS components have the capability of enabling large constellations of small spacecraft with a potential to achieve comparable or even greater performance as compared to traditional spacecraft [22]. Moreover, although the CubeSat program still faces many hurdles, its overall success for placing experiments into space and training the next generation of aerospace engineers is undeniable.

2.2 The Scientific Theme

2.2.1 Ionosphere Environment and Plasma Formation

The atmosphere is a huge envelope of gas surrounding the Earth, kept in place by the gravitational field, with density decreasing with height until it becomes negligible [28]. The fact that it changes from the ground up enabled the establishment of five distinct layers: troposphere, stratosphere, mesosphere, thermosphere, and exosphere (see Figure 2. 5). Each is bounded by “pauses” where the greatest changes in thermal characteristics, chemical composition, movement, and density occur [28].

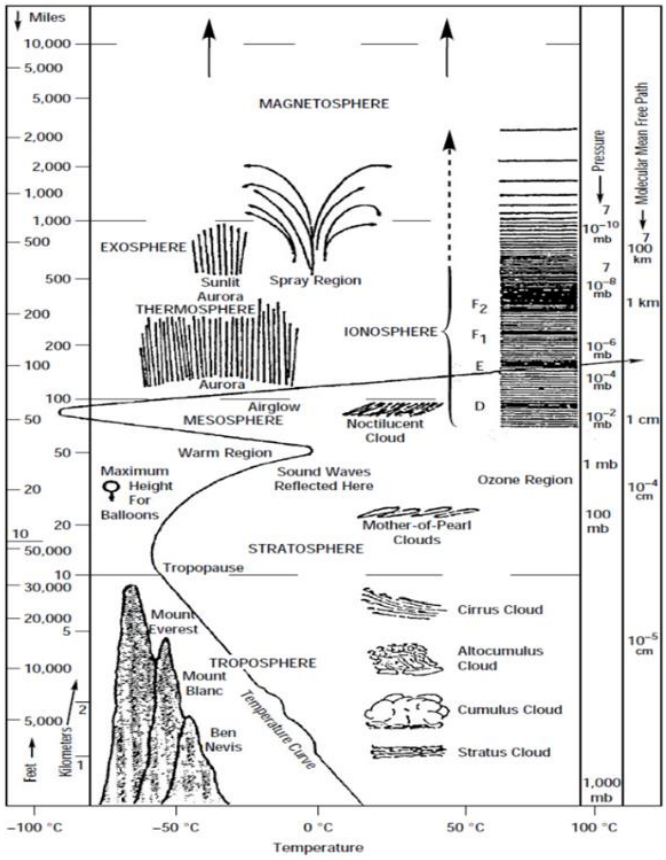


Figure 2. 5 - Layers of the Earth's atmosphere (from [29]).

An interesting layer called the Ionosphere lies in the upper atmosphere, overlapping the middle layers. The Ionosphere is an active part of the atmosphere as it changes with time depending on the energy that it absorbs from the sun. The name comes from the fact that gases in these layers are excited by solar radiation forming a gas of ions and free electrons: the plasma. [28], [30] Plasmas are ionized gases, globally neutral and displaying collective effects, which means that particles within plasma interact with each other through the electric and magnetic field that they have collectively generated. [30] Just as temperatures define the main layers of the atmosphere, electron densities of plasma define the layers of the Ionosphere. Due to the spectral variability of the solar radiation three layers are created: D, E, and F.

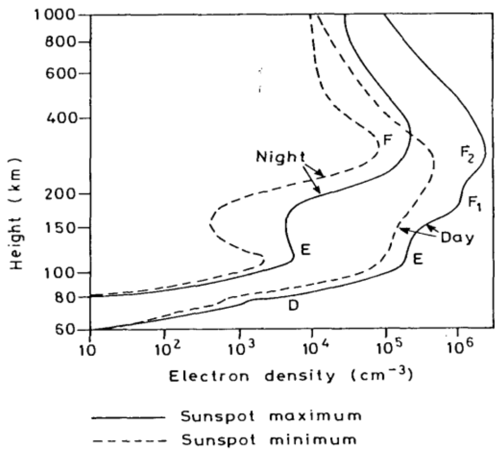


Figure 2. 6 - Typical vertical profiles of electron density in the Ionosphere (from [30]).

Looking at Figure 2. 6, it is possible to conclude that the density of plasma in the ionosphere depends strongly on two variables: the solar irradiance and the altitude. The solar irradiance changes over the time of the day and it depends on the solar activity. Nevertheless, Figure 2. 6 only shows the formation of plasma due to environmental causes, i.e. without being disturbed by a spacecraft.

Concerning the case in which a spacecraft travels through the atmosphere, the electron density would increase as the vehicle travels through it and reaches its maximum during atmospheric re-entry phase which starts around 120 km altitude [6]. The formation of plasma surrounding the vehicle can also depend on the type of flow regime. This can be deduced by the Knudsen Number, K_n, (Figure 2. 7) which is a dimensionless number defined as the ratio between the mean free path length of the molecules of a fluid, λ, and the characteristic length, L_c [31]:

$$K_n = \frac{\lambda}{L_c} \tag{2.1}$$

While in orbit, if $K_n > 10$, a free-molecular flow regime occurs. If $K_n < 0.1$, the vehicle travels in continuum flow and a shock wave is formed in the front of the vehicle causing the creation of a dense plasma layer. In between, there is a transition flow with combined properties.

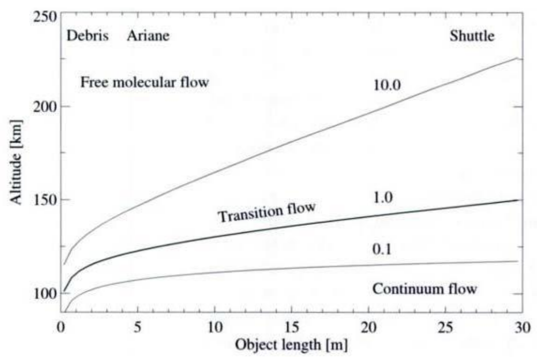


Figure 2. 7 - K_n as a function of the altitude and the object length (from [31]).

2.2.2 Radio Frequency Blackout

During the Earth's atmospheric re-entry, a shock wave is formed in the front of the vehicle, causing air compression and heating (Figure 2. 8). At hypersonic velocities, this heating will be enough to excite the gas molecules' internal energy modes up to the point where dissociation and ionization reactions occur, forming a dissociate plasma layer around the spacecraft. This layer consists of ions and free electrons [8] [6].

The ionized plasma layer causes an important issue known as the RF blackout. At a sufficiently high plasma density, the plasma sheath either reflects or attenuates communications to and from the vehicle causing all communication to be degraded or temporarily disrupted, which includes GNSS navigation, data telemetry, vehicle tracking and voice communication. As a result, the plasma field generated around the vehicle can cause signal attenuation or complete communication interruption [6], [8]-[15].

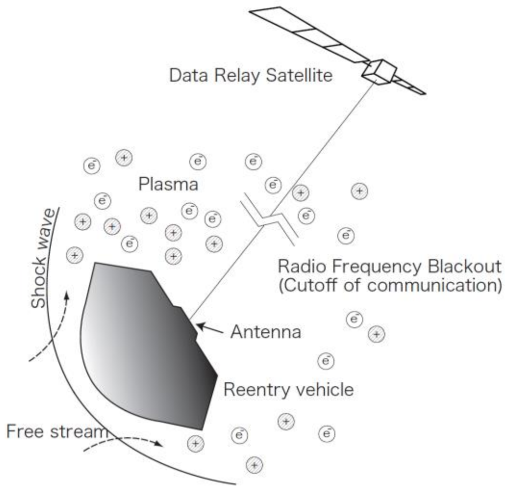


Figure 2. 8 - Schematics of RF blackout during atmospheric re-entry (from [12]).

The degree of severity of the communication blackout problem during Earth's atmosphere re-entry is usually between 4 and 16 minutes depending on the vehicle configuration, flight velocity, angle of re-entry, and different free-stream conditions. [6], [8]. However, entering atmospheres of larger planetary bodies such as Jupiter, this phenomenon may take as long as 30 min [10].

One of the most important parameters when dealing with the RF blackout problem is the plasma frequency which is directly associated with the electron density. For a given electron density, η in m^{-3}, the plasma frequency, in Hz, is expressed as [6], [7]:

$$f_p = 8.985\,\eta^{1/2} \tag{2.2}$$

The communications with the vehicle is completely cut-off when the plasma frequency, f_p, exceeds the transmitting radio wave frequency, f_{radio}, used for communication [6]:

$$f_p > f_{radio} \qquad (2.3)$$

Hence, one can deduce the critical plasma density, $\eta_{critical}$, from equations (2.2) and (2.3), which defines the maximum electron density of the plasma sheath surrounding the hypersonic vehicle in order to properly transmit a radio wave signal in the plasma field [6]:

$$\eta_{critical} = \left(\frac{f_{radio}}{8.985}\right)^2 \qquad (2.4)$$

The critical plasma densities for different radio wave frequencies are presented in Table 2. 2 [18].

Table 2. 2 - Common radio wave frequencies and their critical plasma density.

Frequency [GHz]	Critical Plasma Density [m^{-3}]	Designation
0.30	1.12×10^{15}	Voice Communication
1.55	2.99×10^{16}	GNSS
1.68	3.52×10^{16}	L-band
8.20	8.75×10^{17}	X-band
32.0	1.27×10^{19}	Ka-band

Nonetheless, the plasma layer may attenuate the radio wave even when the electron density is lower than the critical one. Concerning these special cases, radio wave attenuation depends on the transmission frequency, the electron collision frequency, and the plasma frequency [6]. Topics that require further investigation and are not considered in this thesis.

The literature contains an extensive amount of data on the plasma sheath formed by solar radiation in Ionosphere [30] or by the heat generated from vehicles reentering the atmosphere [6], [8], [10]-[13], [16]. Plasma density profiled as a function of several variables such as elapsed time, altitude, and vehicle velocity are available for the re-entry phase [6], [14].

The density of the plasma sheath cited in the literature ranges from 10^9 to 10^{12} m^{-3} in low Ionosphere [30] and from 10^{17} to 10^{20} m^{-3} during re-entry [6], [10]-[12], [15]-[17] when the RF blackout occurs. At such high densities the plasma frequency greatly exceeds the frequency range of conventional S, C, and X band communication signals that range from approximately 1 GHz to just over 10 GHz [10].

2.2.3 The Importance of RF Blackout Mitigation

The RF blackout period has been an issue during hypersonic flight since the dawn of the manned space program [10] and is an especially significant hindrance during the atmospheric re-entry of a spacecraft [6]. The consequences are multiple and stand as a technological obstacle for the development of hypersonic vehicles and advancement in space interplanetary atmospheric entry missions [6], [8], [13], [18].

To understand the science's urge for MECSE mission it is crucial to comprehend the main reasons why the RF blackout problem must be solved. The attenuation of the radio frequency signals during hypersonic flight and re-entry missions can be severe and, in most cases, will be total during a part of the flight [8], [18].

Firstly, to have a more precise idea, hypersonic vehicles could be traveling at velocities up to 26 times the speed of sound $(\approx 8\,km/s)$ [8]. At those velocities, one single minute of RF blackout represents approximately 480 km of vehicle's incapability to send/receive real data telemetry and access to a navigation system (GNSS) which can introduce problems related to vehicle's positioning accuracy. The position error can range from several meters to tens of meters even with little attenuations [32].

In fact, real-time telemetry monitoring becomes especially important at hypersonic velocities, primarily for flight safety reasons. During the RF blackout period, the vehicle loses the capacity of precise guidance and maneuvering initiated by a GNSS satellite or control center which can compromise the mission success [6], [18]. Also, without real-time telemetry, it is extremely difficult to make quick decisions on when to abort a flight [8].

Secondly, current unmanned space missions, as well as future manned missions to Mars and other planets with unfamiliar atmospheres would greatly benefit from a communications blackout solution [6], [10], [12], [17], [18]. As a result of radio blackout, the vehicle loses navigation and mission command, which degrades the landing accuracy and may lead to catastrophes. As an illustration, for the Mars entry vehicle, the RF blackout lasts, approximately, twelve seconds. Future Mars missions demand high precision entry navigation capability, particularly when landing accuracy is needed to land on the scientifically interesting sites surrounded by hazardous terrain. This motivates the need for high accuracy entry navigation system which urges for RF blackout mitigation. [19].

Moreover, many missions to planetary bodies with atmospheres, necessarily require the use of aerodynamic braking maneuvers in which the spacecraft uses atmospheric friction to slow down and transfer itself to a lower orbit minimizing the use of propellant [25]. During this period, the spacecraft will experience the same communications blackout problem [17].

Fourthly, the inability of transmitting telemetry in real-time prevents catastrophe analysis, which is a key factor for understanding and preventing re-entry accidents. Data collected milliseconds prior to a catastrophe could be critical in determining the cause. At hypersonic flight, continuous telemetry is absolutely necessary because the velocities and altitudes involved imply that it is unlikely that onboard recorders would survive a crash or be found if they do survive after a disaster [6], [8].

In addition, mitigation technology will also be valuable for the defense sector. Critical functions of anti-missile defense systems such as tracking and radar identification, missile electronic countermeasures, and mission abort functions are prevented by the communications blackout period [6], [8], [10].

Lastly, it stands to reason that future hypersonic vehicles will also require blackout mitigation technologies since they must have constant radio contact with ground control for communication and navigation [8], [10]. Also, if one has into consideration that a Mach 10 flight allows traveling to anywhere in the world in about 2 h, then there is a strong reason for developing a vehicle capable of achieving such velocities [8], [18].

In summary, the ability to communicate through a plasma layer remains a critical area of research in hypersonic flight and spaceflight. The need for a robust methodology for transmission of vehicle health and trajectory information, as well as scientific data through the ionized plasma sheath, is essential for advancements in hypersonic vehicle design [18].

As mentioned previously, consequences of the RF blackout are severe and can compromise the success of a hypersonic or re-entry mission. Even though it has been continuously investigated, no satisfactory solution has yet been established and the problem has ultimately become an undesirable obstacle [6], [8], [10], [11].

RF blackout is a problem at the forefront of science community technological interest and so is the urgency to find a solution. This issue becomes of the utmost importance regarding the guidance, health monitoring, and data telemetry, particularly, during atmosphere re-entry. [6], [12], [17], [18].

2.2.4 Mitigation of RF Blackout

Several mitigation techniques have been discussed to attenuate the communication blackout period [10], [11]. In general, two methods are suitable for addressing the radio blackout problem: passive and active (Figure 2. 9).

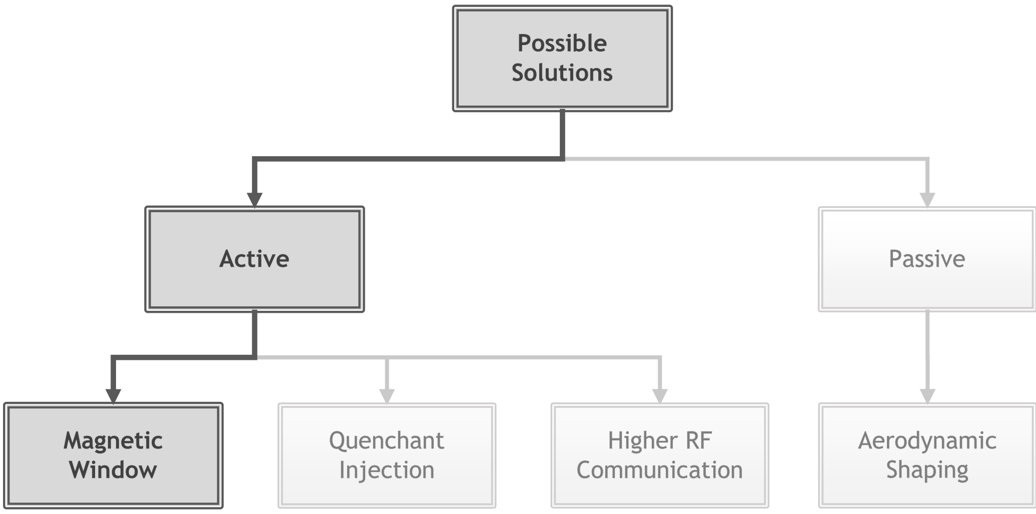


Figure 2. 9 - Possible solutions for RF blackout mitigation.

Concerning the aerodynamic shaping, it includes changing the leading-edge geometries to decrease the plasma density and allow data to be transmitted through the plasma sheath [10]. Sharply pointed re-entry vehicles are surrounded by a much thinner plasma sheath than that surrounding blunted re-entry vehicles. On the downside, a sharply pointed vehicle has a reduced payload capability and increased aerodynamic heating problems compared to a blunted vehicle [10], [15]. Hence, this solution is not adequate for blunted vehicles of generic shape.

Active technologies propose to actively reduce the plasma sheath effects on radio communication attenuation and blackout [8]. The three leading candidate solutions are high frequencies transmission, quenchant injection, and magnetic window [6], [10], [11], [14].

The first one is what would seem the simplest: communicate in higher frequencies, well above the plasma frequencies [8]. The drawback is that those frequencies are not currently used in radio communications because they often suffer huge attenuations in signal caused by rain and other atmospheric phenomena [6].

Quenchant injection of electrophilic liquids or gases into the shock layer will modify the plasma properties in a specified region and allow communication. This process has experimentally shown to restore radio communication for re-entry conditions. However, the amount of quenchant mass needed for scale-up to large vehicles remains an issue [11], [16].

Lastly, the magnetic window method aims to reduce the plasma density in a localized region creating a “channel” for communications [8], [10]. The idea is to manipulate the plasma using a magnetic field [13]. However, for a successful blackout mitigation, the required magnetic field strength is about 1 Tesla (T) [6], which means that the weight of the magnet used would be an issue.

Nevertheless, the magnetic window method can be expanded via the addition of electric fields to increase the plasma density reduction for a given magnetic field. The applied configuration of this method is shown schematically in Figure 2. 10 [6]. As it can be seen, the electromagnetic manipulation system mainly consists of an embedded electromagnet together with electrodes which will create the electric and magnetic fields.

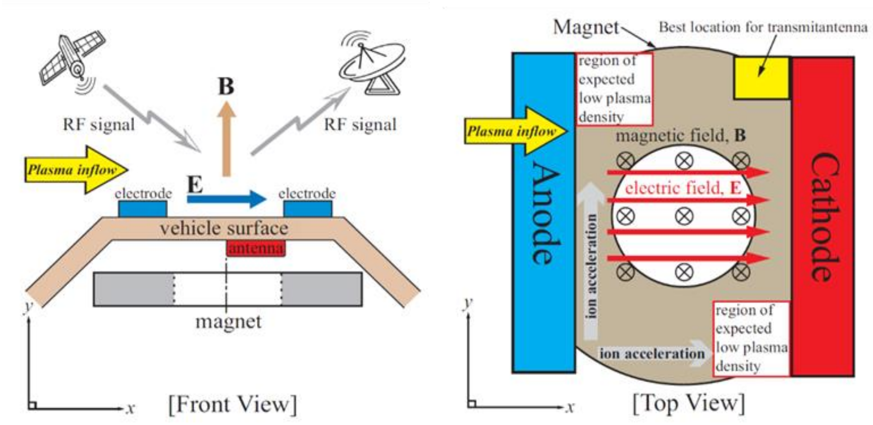


Figure 2. 10 - Schematics of an applied electromagnetic (ExB) layer in two different views(from [6]).

Among the mechanisms of active plasma control that have been studied, the electromagnetic manipulation seems to be the most promising method for the possibility of tailoring the plasma layer [15]. In fact, recent numerical simulations and experimental tests, performed in particular by M. Kim [6], [7], [9], [15], have shown that the application of electromagnetic fields can reduce the plasma density significantly under re-entry plasma conditions.

2.2.5 Electron Density Reduction

Research on the magnetic window method has been carried out primarily via computational modeling [6], [7], [9], [13], [14], but also via experimental test [6]. These efforts have been largely successful, showing that the magnetic window approach should work to mitigate the reentry blackout.

Several simulations have been performed to determine the magnetic field strength required to mitigate the blackout [6], [10].

Studies presented in [10] refer that right-handed polarized waves will propagate along magnetic field lines with a magnitude as low as 0.0357 T and 20 dB improvement in signal reception is expected with a magnetic field of 0.75 T for re-entry plasma conditions.

In [6], the parameter used to characterize the plasma layer manipulation was the Electron Density Reduction (EDR), which measures the amount of plasma density reduced when an electromagnetic field is applied to a plasma layer. Basically, it is the ratio between the final electron density, η_e, and the initial one, η_0 [6]:

$$EDR = \frac{\eta_e}{\eta_0} \tag{2.5}$$

This parameter was used during the numerical simulation performed by Kim [6] for an electromagnetic mitigation scheme over the OREX reentry vehicle in a hypersonic flow. The simulation results for OREX show that by applying an electromagnetic field the plasma density can be reduced [6]. As expected, this depends on the strength of the magnetic and electric fields applied (Figure 2. 11). The initial plasma density used for the study was $10^{17}\ m^{-3}$.

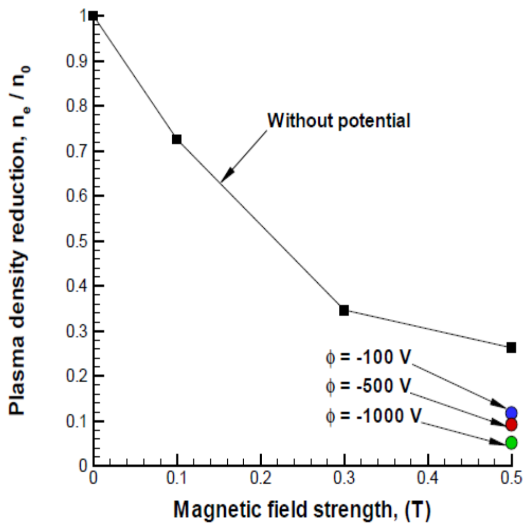


Figure 2. 11 - Electron density reduction for an electromagnetic manipulation scheme (from [6]).

Looking at Figure 2. 11, it can be concluded that the EDR decreases with the increase of the magnetic field strength, which means that the final plasma density will be lower when high magnetic fields are applied.

Also, it can be noticed that there is no need of using electric fields (potential) to manipulate the plasma layer. Although, they may be required to successful mitigate the blackout. For instance, the maximum magnetic field strength from Figure 2. 11 (0.5 T) without potential results in a EDR of 0.3. So, the final plasma density would be of, approximately, $3 \times 10^{16}\ m^{-3}$ (0.3 times lower than the initial one). This value is still higher than the critical plasma densities presented before in Table 2. 2 for voice communication and GNSS. By adding electrical fields, the EDR will increase.

In summary, the density of the plasma layer can be reduced using an electromagnetic field scheme. If the reduction is enough, the RF blackout will be mitigated.

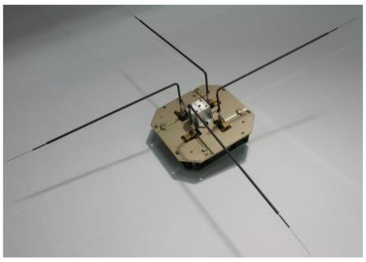
2.3 State-of-the-Art Space Missions

Having defined the scientific theme, it is now important to understand and discuss former spacecraft related missions that can serve as a reference for MECSE mission. This study will allow a comprehension of which scientific related experiments have already been performed as well as to identify system engineering decisions associated with the design of the spacecraft.

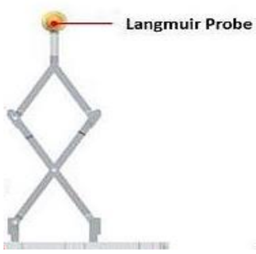
One of the first and more important researches on RF blackout began around 1960 at the NASA Langley Research Center with the Radio Attenuation Measurements (RAM) program [16]. The purpose of this program was to measure several re-entry plasma sheath parameters in order to enhance re-entry plasma simulation on the ground as well as to investigate some mitigation methods [10]. The RAM program flew seven successful blunt-body probes using a multiple electroacoustic diagnostic system which includes sensors such as the Langmuir probes. These sensors were able to measure the plasma density at various distances from the spacecraft surface within the plasma sheath. The experiments yielded data that is still useful today when studying the RF blackout problem [10], [11].

Secondly, the CubeSTAR, which is a student nanosatellite project developed at University of Oslo in Norway [33], [34], also studied the Ionospheric plasma density. This spacecraft was designed using a “2U” CubeSat (see Figure 2. 12 a)). The mission purpose was to perform a technology demonstration of a new scientific instrument: the multi-Needle Langmuir Probe (mNLP) in Figure 2. 12 b)). The instrument was designed to be able to perform plasma density measurements with high spatial resolution. Furthermore, an active potential control system was also developed to mitigate the spacecraft charging which affects the measurements [33].

Thirdly, DICE [35], which consisted of two identical “1.5U” CubeSats launched simultaneously, also addressed the same scientific theme. The purpose was to measure plasma density distributions and electric fields in the Ionosphere. Each spacecraft carries, as scientific payloads, a fixed-bias spherical DC Langmuir Probe (in Figure 2. 12 b)) to measure in-situ ionospheric plasma densities.



a) mNLP (from [34]).



b) DC Langmuir probe (from [35]).

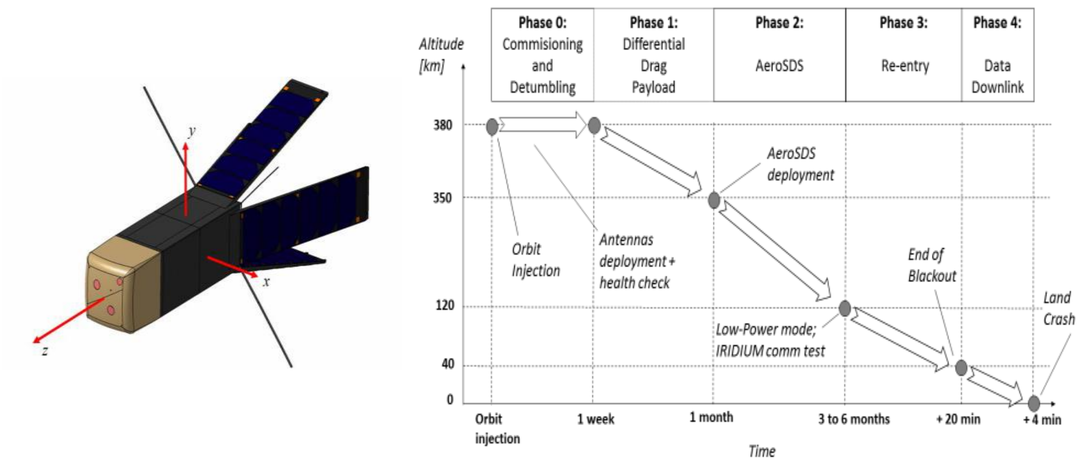
Figure 2. 12 – Types of Langmuir probes used in CubeSTAR and DICE missions.

Finally, QARMAN, which is a triple unit (“3U”) CubeSat mission developed at Von Karman Institute for Fluid Dynamics, in Belgium [36], [37], also targeted a similar mission to MECSE. The main objective was to use a CubeSat platform as an “Atmospheric Entry Demonstrator”, that means it was designed to collect scientific data related with aerothermodynamic phenomena during re-entry. The mission is extremely useful to identify the technical challenges intrinsic to the atmospheric re-entry phase, as well as to understand its mission profile and trajectory which may serve as a baseline for MECSE.

It is important to be conscious of the aggressive environment conditions which the spacecraft is subject to during re-entry. During this phase (Phase 3 in Figure 2. 13 b)), the temperature will rise up to more than 2000 K at the tip and 1000 K at the end of the side panels. Hence, an ablative cork based Thermal Protection System (TPS) was integrated in order to protect the front of QARMAN (see Figure 2. 13 a)). Similarly, the side panels were also thermally insulated with appropriate TPS to prolong the functionality of all subsystems [36].

Still, in order to successfully provide a flight data set for the entry trajectory, QARMAN mission requires an accurate de-orbiting system. Thus, the QARMAN design also incorporates an aerodynamic stability subsystem called the “Aerodynamic Stability and De-Orbiting System” which would be deployed into a dart configuration (see Figure 2. 13 a)). The system must provide aerodynamic stabilization and an increased drag area, progressively reducing the satellite altitude too [36].

Moreover, during re-entry (see phase 3 of Figure 2. 13 b)) QARMAN will experience a communications blackout where no data can be transmitted to mission control. Consequently, during this phase, the acquired data is stored on a flash memory and will be transmitted towards the Iridium constellation once the blackout ends and before crashing [36], [37].



a) QARMAN design. b) QARMAN’s mission profile.

Figure 2. 13 –The QARMAN nanosatellite design and mission profile (from [36]).

2.4 Space Mission Engineering

Understanding the basic principles of space mission engineering is a critical step before moving forward into the development of MECSE's spacecraft. This section intends to give on the basic principles behind the design of space missions providing therefore a context for the work accomplished during the development of this M.Sc. thesis.

2.4.1 Project Life Cycle

One of the major challenges found while developing a space mission is to define and stick to a specific timeline. Lack of experience, financial budgets, system's complexity or low technology readiness levels are some of the aspects that can compromise the timeline of a mission [38].

Therefore, it is necessary to create a project lifecycle which is basically a timeline of the project divided in phases. Each of them can be created to result in deliverables or accomplishments that provide the starting point for the next one. Figure 2. 14 show the examples of the standardized NASA's and ESA's project life cycle which are rather similar [38], [39]. Each triangle in Figure 2. 14 act as a key decision point which basically means that by that time all the required deliverables need to be finished in order to proceed to the next step. For the purpose of this thesis, the ESA project life cycle is considered as reference.

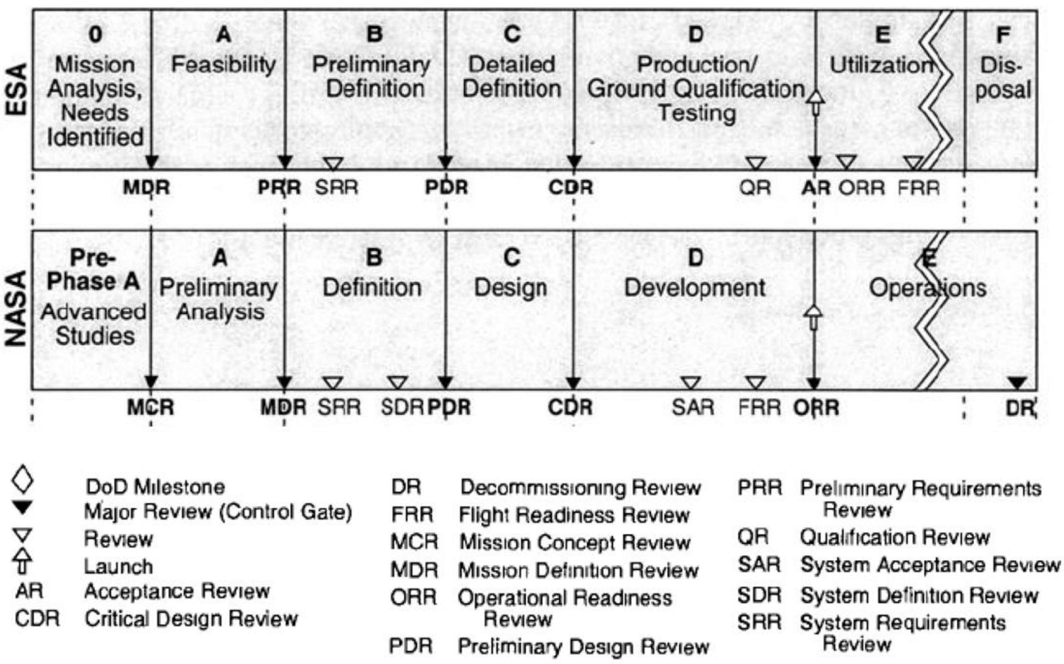


Figure 2. 14 – ESA's and NASA's project life cycles (from [39]).

At a top level, the space mission life cycle goes through four broad phases [25], [38], [40]:

- **Concept Exploration:** The preliminary study phases, where the mission needs to be designed and analyzed. The result is a broad definition of the mission architecture and its components, cost and overall schedule.
- **Detailed Development:** The formal design phase which results in a detailed definition of the system components and, in some cases, technology development.
- **Production and Qualification:** The development of the required hardware and software. It also ensures that all components integrated into the spacecraft and launchers are fit for purpose over the entire lifetime of a mission.
- **Operations and Disposal:** The operation and utilization of the space system, its maintenance and support and finally its deorbiting and end of the mission.

The aim of this thesis is to focus on the concept exploration, that means, to go through the early project phases: phase 0 (mission definition and analysis), A (feasibility) and B1 (preliminary design up to SRR) [24] according to ESA's life cycle. Those phases are usually inter-connected when designing small spacecraft projects such as CubeSats.

Concept exploration plays a huge role when designing a system because it determines most of the total development cost. In fact, decisions performed in this phase define up to 80 percent of the total cost [41].

However, in MECSE's project case, the requirements for the system are not yet fully defined. Therefore, it will be necessary to design the mission from the ground up, which will require identifying mission needs (how the mission can be helpful), clarifying mission objectives, and defining requirements and constraints before one can start the design of the system itself. This can only be done by allying systems architecting with systems engineering practices, which ally creativity with critical thinking and problem solving [24], [35].

The next section describes the fundamentals of both disciplines within the concept exploration phase and expound their importance for the design of space systems.

2.4.2 Systems Architecting and Systems Engineering

To distinguish architecture from engineering, Rechtin [42] (in 2020) discusses that engineering is a deductive process since it deals almost entirely with measurable elements using analytic tools derived from mathematics. Whereas, architecting is an inductive process as it deals mostly with unmeasurable factors using non-quantitative tools and guidelines based on experience. Thereupon, *"in a sense engineering is more of a science and architecting more of an art"* [42].

Briefly, architecting deals with ill-structured situations where the requirements for the system have not yet been stated. In this way, the architect engages in a joint exploration of requirements and design, looking for satisfactory and feasible problem-solution pairs. This contrasts with the classic engineering approach of seeking an optimal design solution to a clearly defined set of objectives. When dealing with complex systems both processes are interconnected and it is rarely necessary to draw a sharp line between them [42].

It is essential to understand the key concepts of systems thinking and its relevance for the design of advanced spacecraft. To begin with, a system can be defined as a construct or collection of different elements that together produce results unachievable by themselves alone [43]. These results represent the final function of a system, that is the value added by the system as a whole which is primarily created by the interconnections among the parts [42].

The final function of a system is called emergence. Emergence arises from the interface of multiple system elements. Nonetheless, one must keep in mind that these elements are not only the technical parts, but also all the things required to produce system-level results, which include other components such as people, hardware, software, facilities, policies/regulations and documentation [43]. Therefore, systems engineering is about how to manage those interfaces as well as, how to coordinate all the people involved working on diverse subsystems, which requires both technical and management skills [44].

Emergence implies complicated interactions between system elements meaning that in order to have high-performance emergent functions, one has to pay in complexity. Although, the more complex a system, the more difficult it is to design, build and use, and, intuitively, the more expensive it is. So, it is crucial to understand how to manage complexity [42]. There are basically three approaches to doing so: decomposition, hierarchy, and abstraction [43].

Firstly, a system can be broken down into subsystems which themselves can be divided into parts, components and so forth. The idea is to decompose the system in atomic parts which are more manageable, easier to understand and may be worked on sub-teams. Also, it is possible to structure this decomposition by conferring hierarchic values to the subclasses which must focus on relevant information to the designer. Finally, abstractions are often used by engineers to characterize system elements in their functional and existential attributes. They are a way of understanding a complex system in a compact manner since they allow the engineer to detach himself/herself from the physical form of a system and only look at the action being delivered, the intrinsic function [43]. An example of a common abstraction is the use of "black boxes" which replace parts of a system or even entire subsystems.

However, systems architecture is a heuristic art and, thus, it relies on well-chosen methods and guidelines (heuristics). Citing Rechtin [42]: *"The art in architecting lies not in the wisdom of the heuristics, but in the wisdom of knowing which heuristics apply to the current project"*.

Therefore, it is crucial for a systems architect to define its own principles before architecting a system. The heuristics used by the author in this thesis are based on Golkar's principles [43] (2016) which were already discussed by Rechtin [42] and Crawley [41] in previous work.

Golkar's methodology [43] is based on three principles: elegance, traceability and Occam's razor. Elegance means to find the simplest solution to complex problems which implies minimizing functions, elements, and interfaces while still satisfying stakeholders needs. Additionally, functions need to be traceable to stakeholder needs, that means that any function embedded in the system must be connected with a specific need. Otherwise, the system contains non-required functions. Finally, Occam's razor declares: *"do not include plurality if it is not necessary"* which means that good concepts are simple, lean and elegant without the need to include unnecessary complexity which often lowers the overall performance.

Those principles are related with the KISS-heuristic, which variably stands for *"Keep it simple and short"* or *"Keep it simple, stupid"* which is widely accepted in engineering. The idea is that the simpler the system, the easier it is to design, implement and maintain [41].

Additionally, both systems engineering and architecting seek a safe and balanced design in the face of opposing interests and multiple and, sometimes, conflicting constraints. This means that trade-offs and compromises will be demanded, not only to ensure that the stakeholders get the design right (meeting the requirements), but that they get the right design [38], [42].

Systems engineering plays a key role in leading the development of the system architecture, defining and allocating requirements, evaluating trade-offs and balancing technical risks between systems by assessing interfaces [38]. Whereas systems architecting deals more with the innovative part of the design. The architect needs to come up with alternative scenarios and innovate concepts in search for optimization [42].

Space mission engineering combines both systems architecting and systems engineering allying both disciplines which is central when trying to reduce the cost and risk of a space mission, without compromising its performance. It can be settled as *"the definition of mission parameters and refinement of requirements so as to meet the broad and often poorly defined objectives of a space mission in a timely manner at a minimum cost and risk"* [25].

2.4.3 The Space Mission Engineering Process

Designing space missions is an inherently iterative process, gradually refining system requirements and concepts which can turn out to be complex and time-consuming [25]. Therefore, it becomes critical to delineate a procedure to be followed.

The process begins with the exploration of a mission purpose, which is the reason to perform the mission. In this case, it is the scientific theme. Then, it is required to clearly characterize the mission by identifying the needs and exploring different mission scenarios that can meet those needs with satisfactory performance. Afterwards, the mission scenarios must be evaluated which implies to identify preliminary requirements and constraints for each one.

Having selected the mission scenario, the mission shall be defined. The first step is to clearly state the mission aim and objectives. This will allow to understand what are the most relevant parameters to be analyzed (figures of merit) through the project, as well as to define the mission requirements which are important for the system design.

Afterwards, the mission analysis and the conceptual design of the system will be performed in parallel. The goal here is to analyze the figures of merit and understand their impact for the progress of the mission.

The space mission engineering process adopted in this thesis for the concept exploration phase of MECSE project is presented in Figure 2. 15 . Note that, it uses the methods endorsed by Wertz in “Space Mission Engineering: The New SMAD” [24] as guidelines.

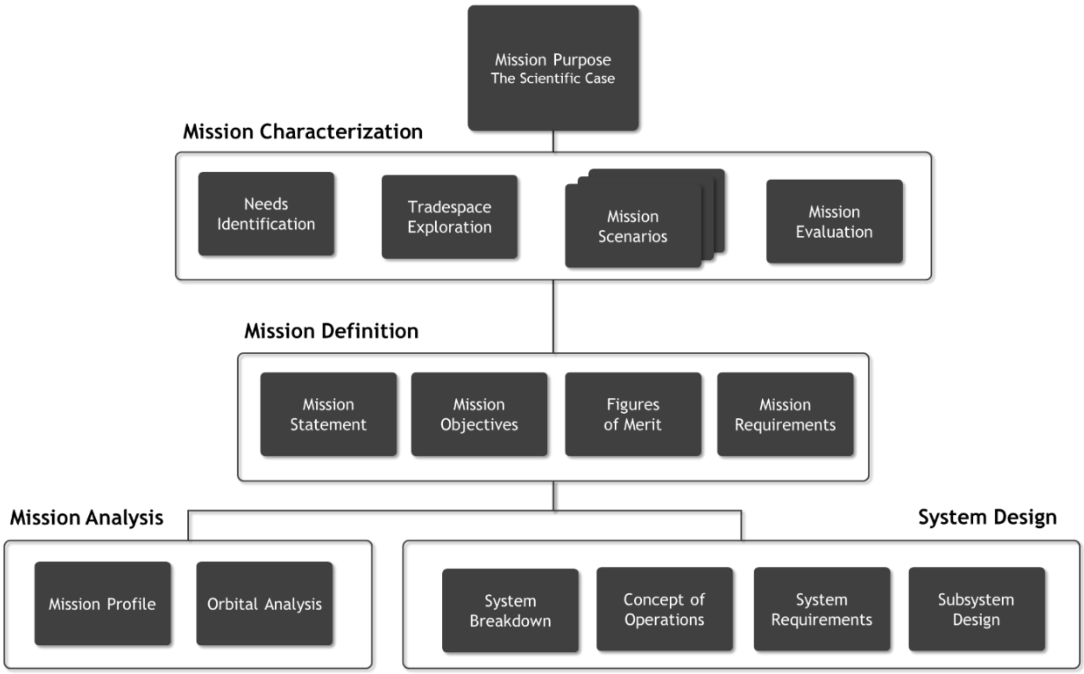


Figure 2. 15 - The space mission engineering process for the mission design of MECSE.

Having defined the space mission engineering process to be used, the mission design can start. The first step will be to formulate the scientific case based on the literature review and the research being conduct at UBI about the scientific theme.

Chapter 3

3 Mission Characterization

Mission characterization is the process of architecting the mission, which requires to first identify the stakeholders needs and consider all the possible mission scenarios capable of fulfilling those needs in a satisfactory manner. Afterwards, it is important to assess preliminarily the feasibility of each scenario by identifying requirements and constraints. In the end, the most suitable scenario must be selected.

3.1 Mission Purpose

3.1.1 The Scientific Research at UBI

The RF blackout is an emergent research topic within the Space Science and Technology field at the forefront of space exploration. There is an ever-increasing scientific need to mitigate this problem.

Currently, C-MAST at University of Beira Interior, the scientific stakeholder of this project, is developing and validating an MHD numerical model [13], [14] for assisting in the design of re-entry objects, with an emphasis on radio blackout mitigation mechanisms and plasma layer manipulation.

When validated by experimental data, this numerical framework could assist in the development of efficient MHD approaches for manipulating the plasma flow. The final goal is to start designing three-dimensional magnetic control systems for plasma layer under hypersonic flow conditions, which will assist directly the design of hypersonic vehicles.

Therefore, the ultimate scientific purpose of the hereby envisioned CubeSat mission – MECSE – would be to validate the numerical model being developed at UBI which would imply to test the theory that an electromagnetic field can re-shape the layer of plasma surrounding the spacecraft and therefore allow communication during the atmospheric re-entry blackout phase.

However, the high complexity associated with this scientific aim has resulted in challenging scientific goals which shall meet the technical feasibility of the system. Thus, it is crucial to find an agreement between both parts by trying to reduce the complexity. This will allow to find a mission that can be achievable within a CubeSat concept and at the same time serve the scientific purpose.

3.1.2 The Scientific Case

From the scientific theme, the scientific case can be formulated. To do this, the investigation presented in the scientific theme is combined with the scientific research at UBI.

From the scientific theme, it was concluded that RF Blackout is caused by a highly dense plasma layer that forms around the vehicle during atmospheric re-entry. At those altitudes and velocities, the conditions for the plasma formation are ideal because the atmospheric density is high and a shock wave is formed in the front of the vehicle. It has also been shown that the plasma formation still happens in Low Earth Orbits (LEOs) (within the Ionospheric layers). In this last case, the plasma layer is thicker, so the electron density is less than during re-entry.

Furthermore, the formation of plasma may vary depending on the type of flow regime. For this thesis, it is assumed that the type of plasma found during free-molecular flow and continuum flow are the same.

Investigations performed in UBI [14] have already shown that the use of magnetic fields has effects on the distance of the shock wave to the vehicle during re-entry, impacting the plasma layer. However, it was necessary to find a way of associating that study with a measurable quantity of the plasma layer. It was found in the literature [6] that the use of electromagnetic fields would decrease the electron density of the plasma layer, which is a parameter that can be measured using Langmuir probes.

In conclusion, the manipulation of plasma using electromagnetic fields can be confirmed by assessing the EDR when an electromagnetic scheme is applied to the plasma layer. This can be performed through the generation of a magnetic field without the need to use electrodes. Also, it was found (see section 2.2.1) that this study can be performed during re-entry or in the Ionosphere because plasma formation happens in both situations.

On the other hand, in order to mitigate the RF Blackout, it is necessary to manipulate the re-entry plasma flow in such a way that the plasma density is decreased to a value lower than the critical plasma density for communications. For this case, the EDR required shall be high enough, which implies to use high magnetic field strengths combined with electrical fields.

3.1.3 Needs Identification

The first step in the space mission engineering process is to define the broad mission objectives that the system must achieve to be productive for the end users. Nevertheless, similarly, to many other scientific satellite missions, the mission required clarification of its objectives. The clarification starts with the identification of the stakeholders and its needs.

For the project, it was decided to divide the stakeholders in three main categories: education, science and engineering. Firstly, MECSE aims to provide hands-on experience to university students on space projects. Secondly, the project must be relevant to the scientific research at UBI. Thirdly, the project aims to develop more competences within the space field, particularly on the design of satellites and space missions.

Having identified the stakeholders, it is now possible to clarify the stakeholder needs. From an education and technical point of view, the space system shall be as simple and economical as possible in order to be able to be designed in majority by students.

Likewise, the scientific needs must be clarified from the scientific case. To simplify it, they were divided in scientific studies to be performed in space that can benefit the research being conducted on the ground. The scientific studies are listed in Table 3. 1 in the form of objectives.

Table 3. 1 –Scientific studies and objectives.

Scientific Study	Scientific Objective	Description	Scientific Value [1-3]
Plasma Dynamics Study (PDS)	SO1	Study the formation of the plasma layer in LEO by collecting data for different altitudes	1
	SO2	Assess the effects of the spacecraft attitude motion on the plasma layer	2
Plasma Layer Mitigation Experiment (PLME)	SO3	Study the effects of an electromagnetic field on the plasma layer	3
Re-entry	SO4	Study the formation of the plasma layer surrounding the spacecraft during re-entry	2

The success of a space mission depends heavily on whether the mission is able to comply with the main user needs [41]. Thus, it was important to quantify the importance of each scientific objective for the end users (scientific players) in order to measure the mission utility. Each scientific need has been ranking from 1 to 3 representing its scientific value (see Table 3. 1).

The scientific values were chosen through concurrent engineering sessions which promote communication between engineer and scientific players as well as through a deep literature research which has allowed to understand the value of such studies within the scientific community. These sessions are part of a communication link created between UBI and CEIIA and are fundamental when making decisions that affect the mission performance.

In conclusion, the PLME is the scientific study with the most importance for the advancements in RF blackout mitigation [6], [9]–[15]. Therefore, it must be considered as the priority. Also, the PDS was established as a necessary requirement to guarantee the scientific success of the mission. This means that it must be performed independently of the mission selected.

To sum up, in this section, the scientific case was formulated which has allowed to identify the stakeholders' needs and transform them in a set of scientific objectives (SO). Each objective was also quantified regarding its importance for the scientific theme. In the next section, different mission scenarios will be formulated based on these objectives.

3.2 Mission Scenarios

As seen previously, one of the major roles of the system architect is to find the feasible problem-solution pairs which satisfy the stakeholder's needs [42]: the mission scenarios. In order to do that, there are quantitative tools named tradespace exploration methods which can support architecture selection and formulation [43].

3.2.1 Tradespace Exploration

The first step is to understand what are the possible missions that could satisfy in some way the stakeholder needs. In other words, the ones that could contribute with useful information to the scientific studies (see Table 3. 1).

Four mission scenarios were created. Each scenario will be associated with a specific scientific value and level of complexity. The four scenarios are represented in Table 3. 2.

Table 3. 2 – Alternative mission scenarios proposed for MECSE mission.

Mission Scenario	Plasma Layer Mitigation Experiment	Re-entry
A	No	No
B	Yes	No
C	No	Yes
D	Yes	Yes

Looking at Table 3. 2, one can deduce that scenario A would be the simplest to design but it would also be the one with the least scientific value associated (very similar to CubeSTAR mission shown in section 2.2.5), whereas scenario D would be the most complex, but it would also be the most innovative and scientific valuable.

The next step before evaluating the mission scenarios is to understand the basics of how the different mission scenarios will perform, which means to study the architecture of the mission [43]. This will help to analyze the feasibility associated with each scenario.

For that purpose, it is first required to identify what are the subjects of the mission namely the parameters that the spacecraft must sense to achieve the objectives [25]. The subjects will directly drive the payload which will then influence the design of the spacecraft subsystems. In Table 3. 3, the mission subjects were identified.

Table 3. 3 – Mission subjects and respective payloads.

Subject	Parameters	Payload	ID	Description
Environment	T, p, ρ Solar Irradiance	**PL01**	**ENVISENSE**	Environmental Sensors
Plasma Layer	Electron Density	**PL02**	**LP**	Langmuir Probes
Electromagnetic Field	Magnetic Field Intensity	**PL03**	**EMG**	ElectroMagnetic Generator

In addition, the systems architect must identify the differences between the alternative scenarios. One major difference would be regarding to the phases of the mission. As it can be seen in Table 3. 2, the scenarios A and B are not required to survive to an atmospheric re-entry. The high complexity associated with re-entry survival is something that will have huge consequences in spacecraft design and operations.

Firstly, as seen in section 2.2.5 (QARMAN example), in order to survive to an atmospheric re-entry, the satellite must consider using a TPS in order to protect the spacecraft from the aggressive environment conditions which is subjected to. Secondly, during re-entry communications blackout phase the spacecraft will not be able to transmit the data in real time. Therefore, the spacecraft must include a survival capsule capable of surviving to the re-entry phase and transmit the data before crashing. Finally, the risk of not surviving to the re-entry is too high due to the unpredictability associated with this phase.

The other main difference concerns the use of the ElectroMagnetic Generator (EMG – PL03) which is associated with a higher risk of failure due to its high complexity. In fact, the technology readiness level associated [45] is still low (TRL 3) which compromises the mission success. Also, the power required to generate the necessary magnetic field is still unclear but will certainly drive the mass and cost of the satellite.

In Table 3. 4, the differences associated with each mission scenario were estimated in the form of broad design requirements and challenges to overcome.

Table 3. 4 – Tradespace exploration of mission scenarios.

	Mission Scenario	A	B	C	D
Scientific Objectives	S01	•	•	•	•
	S02	•	•	•	•
	S03		•		•
	S04			•	•
Mission Phases	Disintegration in early Re-entry	•	•		
	Re-entry and RF Blackout			•	•
	Data Downlink after RF Blackout			•	•
Payload	PL01 - ENVISENSE	•	•	•	•
	PL02 – LP	•	•	•	•
	PL03 - EMG		•		•
Challenges	EMG Required Power		•		•
	EMG's Mass		•		•
	High Risk – Low TRL (EMG)		•		•
	High Risk – Surviving Re-entry			•	•
	Short Window for Communication			•	•
Spacecraft Design*	Size and Mass	< 3U	~3U	~3U	> 3U
	Thermal Protection System			•	•
	Survival Capsule			•	•

*Considering former space missions as a reference.

From Table 3. 4, it is possible to assess preliminarily the feasibility of the mission. This strategy is used to establish whether a particular mission is achievable and to place broad limits on its level of complexity [25]. Through a simple comparison with former systems, it is possible to conclude that at least scenario A and C are feasible because similar missions have already been performed with existing technology. It is also possible to estimate the mass and size of the spacecraft by comparing with existing systems.

Moreover, it can be concluded that mission scenario D is the one with more technological constraints associated with. Despite having the most scientific value, it is improbable that this scenario would be feasible because of its high complexity and high probability of failure.

In summary, alternative mission scenarios were considered for the possibility of becoming the definitive mission. A preliminary feasibility of the mission was performed by identifying the main challenges and design constraints of each one. Furthermore, the mission subjects and payloads were identified which has allowed to understand how the system will operate.

3.3 Mission Evaluation

3.3.1 Trade-off Parameters

Mission evaluation is the process of examining as many reasonable alternatives as possible to understand how the system behaves as a function of the principal design features. Consequently, trade studies consist of selecting and analyzing mission parameters which largely determine mission performance, cost, risk and schedule. These parameters define a mission scenario and can then be used to conduct performance and utility analysis [25].

The first step is to identify and enumerate the key trade-off parameters or the system drivers [43]. Five parameters were identified: scientific value, cost, risk, total mass/size of the system, required power (associated with the PL03) and systems overall design complexity (number of subsystems and interfaces between them).

Having defined the drivers, it is necessary to quantify the weight factor of each one, that means the level of importance to the mission design. The weight factors were, once more, defined through concurrent engineering sessions in concordance with the multiple stakeholder's needs.

As the mission's purpose is mainly scientific, the highest weight factor goes for the scientific value (40%). This driver serves as a measurement of mission performance allowing to check whether the scientific objectives are fulfilled. The cost (22%) and risk (17%) are also two important parameters when designing a space mission [39], [40]. Although, it was decided that the cost should have a larger influence in the mission design than the risk due to the strict funding limitations which the mission is subjected to as well as due to the fact that the risk of failure should not hold back innovative experiments [21], [22]. Moreover, the total mass (9%) and required power (9%) are two parameters deeply related with the system design and therefore they were ranked with the same weight factor. For example, the use of a TPS will increase the mass, but the use of an EMG will require a larger amount of power. Lastly, the overall system's design complexity (3%) is taken into little consideration because, even though, the mission is supposed to be designed by university students which have little experience in the space mission design, there is the eagerness and motivation to learn.

The next step will be to evaluate each mission scenario as a function of system drivers through trade studies.

3.3.2 Trade Studies

Trade studies, are formal tools used within decision analysis since they are helpful in ranking viable solutions by their satisfaction level to key trade-off parameters [43]. In Table 3. 5, a trade-off study was performed in order to identify the most suitable mission scenario.

A trade study is essentially a comparative study between alternative solutions. Thus, a scale ranging from 1 to 5 was defined which allows to obtain a simpler and more objective comparison among the multiple trade-off parameters. It was adopted a conventional scale where 1 stands for a deficient performance and 5 for a fantastic performance. In this way, we were able to quantitatively compare the four mission scenarios.

Table 3. 5 - Trade-off study between the alternative mission scenarios.

	Mission Scenario	A	B	C	D	Weight Factor
Trade-off Parameter	Scientific Value	1	4	2	5	40 %
	Cost	5	3	4	1	22 %
	Risk	5	4	3	1	17 %
	Total Mass	5	3	4	1	9 %
	Required Power	5	2	4	1	9 %
	Complexity	5	2	4	1	3 %
Total score		3,404	3,451	2,937	2,596	
Ranking		**2**	**1**	**3**	**4**	

Firstly, the scientific value was defined taking into consideration what were the scientific studies (Table 3. 1) and scientific objectives (Table 3. 4) achieved in each scenario. For example, the scenario A is only able to complete the PDS, whereas the scenario B is able to complete the PDS and PLME studies which raise its scientific value.

Secondly, the cost and risk associated with each of the scenario were estimated based on literature review. The cost of mission B was ranked bigger than the cost of mission C given the price associated with the development of the PL03 technology. Furthermore, it was defined that the risk associated with the atmospheric re-entry is greater than the risk of developing the PL03 since the unpredictability of the atmospheric re-entry makes difficult to guarantee that the spacecraft will survive and be able to transmit the acquired data after the blackout. Finally, the PDS also requires an accurate control of the spacecraft attitude which becomes complicated to achieve when doing a re-entry. This compromises the quality of the data acquired.

Thirdly, likewise, literature review allows to compare the mass and required power between different scenarios (Table 3. 4). The energy required to supply the PL03 is currently unknown and it can be a critical key parameter for the design of the spacecraft. In this field, mission A and C are a better option.

Finally, in terms of the complexity associated with the design of the space segment: the simpler the system is, the easier is to design [41], [43]. Therefore, mission scenario A is the less complex and D the most complex one. Due to the need for developing the technology (PL03), the complexity of the system B is considered greater than system C.

In the end, a tool based on analytical hierarchy process was used to conduct the trade study (Table 3. 5). The conclusion that arises from the mission evaluation is that, despite the cost, risk and complexity associated with, the scenario B was the chosen one given its prominent scientific value. However, the difference between the scenario A and B is narrow, which means that scenario A could also be a proper choice for a fast and low-cost educational mission, but with very low scientific value.

In summary, the alternative mission scenarios were evaluated considering different trade-off parameters. This has allowed to select the most suitable mission which combines technical feasibility with scientific value.

3.4 Feasibility Analysis

To evaluate the feasibility of a system, a point design can be used. The point design serves two main purposes: it demonstrates that the mission is feasible up to a certain point, and it can be used as a baseline architecture open to upgrades [25], [38]. Also, it behaves as a back-up plan, which means that the system engineer can return to this point if the upgrades are not feasible.

Mission scenario A can be considered as the first point design since it presents some similarities with mission scenario B (see Table 3. 6). It is known that this scenario is feasible because it is not a novel concept. It only aims to study the ionospheric plasma in LEO, which has already been done before in the past, for example in [34], [35], [46]. In this view, its scientific value may not be sufficient to meet the scientific needs.

Thus, from this first point design forward, all the efforts will be made to optimize the system with focus on the maximization of its performance. This means to try to achieve the mission scenario B. To do this, this thesis will propose an ingenious design of the space system, presenting a novel concept of operations strategy (see chapter 6). Meanwhile, the project team will work in parallel on the PL03 design to proof its viability. For now, the author of this thesis will consider PL03 as a feasible subsystem under development (“black box”).

Table 3. 6 – Feasibility analysis based on a point design approach.

	A - Point Design	B - MECSE Mission
Performance	SO1, SO2	SO1, SO2, SO3
Trajectory	LEO Orbital Decay Disintegration	LEO Orbital Decay Disintegration
Payload	PL01, PL02	PL01, PL02, PL03
Uncertainties	Scientific Value	PL03 (“black-box”) Power, Mass, Size System Interfaces
Design Assumptions	CubeSat Standard 2U	CubeSat Standard 2U (BUS) + 1U (Payload)

In conclusion, in this chapter the MECSE mission was selected by exploring several mission scenarios and evaluate them regarding the technical feasibility and the scientific value. It was concluded that the mission selected seems feasible due its similarities with former ones. Although, the PL03 design is still an uncertainty and further work is required in this area. In the next chapter, the design of MECSE mission starts with a clear definition of the mission.

Chapter 4

4 Mission Definition

Having characterized the mission by selecting the most suitable mission scenario, it is now time to define it precisely by clarifying the mission aim as well as its objectives and requirements.

4.1 Mission Statement

The mission statement is the mission's aim or function. It describes what the spacecraft aims to achieve during its operation. MECSE mission statement is:

Table 4. 1 - Mission statement.

Mission Statement
MECSE is a student-driven project aiming to study the plasma dynamics surrounding the spacecraft when traveling in Low Ionosphere and create a benchmark for the validation of the theory that an electromagnetic field can manipulate the plasma layer. To be successful, MECSE shall orbit the Earth (LEO) gathering data on the plasma layer while using an electromagnetic generator.

By coupling low-cost flight experiments with low-cost numerical simulations, the research on blackout mitigation can fast-forward. The results of this experiment can help the development of the technology that will allow bypassing the blackout in the near future.

It seems important to clarify that MECSE nanosatellite will not perform atmospheric re-entry for now. This scenario has been evaluated as too expensive, risky and complex since it would imply to test a new concept while trying to survive a very harsh environment.

The nanosatellite's main goal is to perform a proof of concept by confirming in orbit that the manipulation of plasma is possible with electromagnetic control. MECSE will prove it within LEO where there is enough ionospheric plasma to be manipulated. By doing so, this will create a tool that could then be improved to mitigate the blackout in a future phase.

4.2 Mission Objectives

Mission objectives are specific statements that characterize the mission performance. Therefore, mission performance is the ability of achieving the mission objectives. As seen previously, MECSE mission has scientific and educational purposes which must be fulfilled. The mission objectives were clarified and are presented in Table 4. 2.

Table 4. 2 - Mission objectives.

Primary Mission Objectives		
Education	MO1	Provide hands-on experience to university students on space projects
Science	MO2	Study the formation of plasma surrounding the S/C when travelling in LEO
	MO3	Assess the effects of the S/C attitude motion on the plasma layer
	MO4	Study the effects of an electromagnetic field on the plasma layer
Secondary Mission Objectives		
Technology	SMO1	Develop a MHD/EHD device for plasma layer manipulation
	SMO2	Develop a modular structure for a CubeSat to be used in future space missions

Likewise other university CubeSat projects [22], the main mission goal is to actually provide experience to university students on space projects, which would not be possible by just reading books or assisting to lectures. The bottom line here is that the learning factor is much greater when one actually design a space mission [4]. Therefore, the first objective is educational and, for the students participating in it, the mission will already be considered a success when the spacecraft have been built and launched.

On the other hand, as a scientific mission, it is expected that the mission will be able to perform the proposed scientific studies in orbit in order to be useful for the scientific community.

4.3 Traceability Tree

As already mentioned, functions need to be traceable to stakeholder needs. Therefore, any payload embedded in the system must be connected with a specific scientific need or requirement. The traceability tree presented in Figure 3. 1 has the ability to link payload functions to scientific requirements.

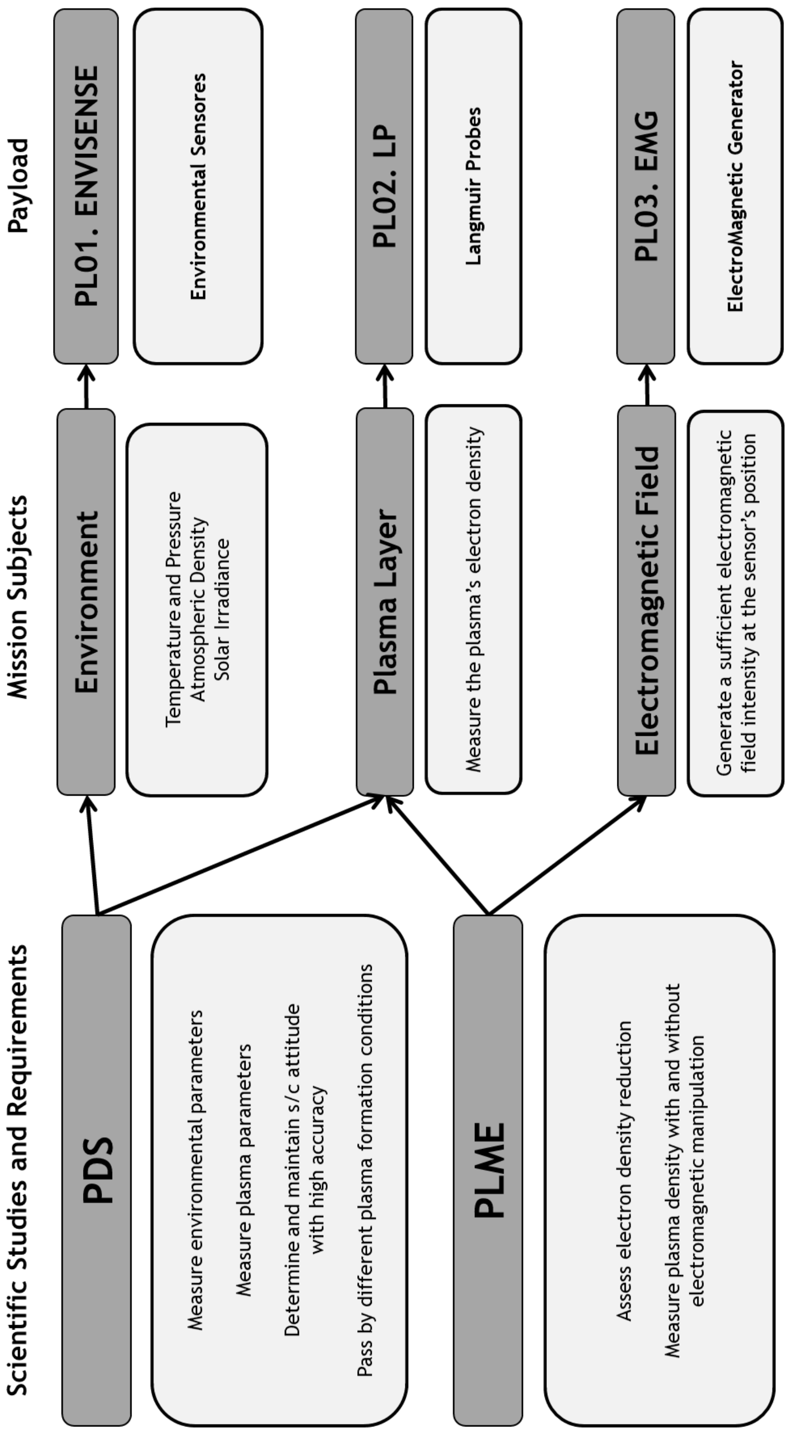


Figure 3. 1- Traceability tree from scientific needs to payloads.

4.4 Figures of Merit

According to [25], [38], a figure of merit is a quantity used to characterize the performance of a system relative to its alternatives. For MECSE project, the figures of merit represent key parameters which characterize the system design in a subsystem level and will impact directly the mission performance.

These parameters will be important when performing subsystem design decisions. Therefore, it is of the systems engineer responsibility to identify them in the beginning of the project. The figures of merit were identified and are described in Table 4. 3.

Table 4. 3 – Figures of merit.

Figure of Merit	Details	Justification
Scientific Payload	PL02 - LP PL03 - EMG	The payload is the most important subsystem since it influences the system overall design. All the other subsystems are designed to support it. PL02 and PL03 will drive the system drivers such as power, mass and attitude.
Power	Peak Power Duration of the PDS and PLME	The PLME will require the generation of an electromagnetic field during a short period of time. The required peak power and energy will drive the Electric Power System (EPS) design. Thus, it is important to find solutions that can minimize the peak power. Also, the duration of the scientific studies will impact the EPS design.
Attitude	Pointing Errors S/C precession Velocity-Vector Stabilization	Pointing errors and precession of the S/C will cause static and dynamic disturbances and deviations of the plasma layer affecting the measurements. Solutions that can reduce the amplitude of the S/C deviations from the direction of orbital velocity vector while minimizing the power consumption and mass must be identified. Meanwhile, it is assumed that the scientific experiment shall be aligned with the velocity vector to minimize the error associated with the plasma measurements.

Mass and Volume	CubeSat 3U Configuration	The system's overall mass and volume are constraint by the CubeSat configuration. Thus, all the subsystem's mass and volume must be minimized to be fitted in a 3U configuration.
Orbital Lifetime	System Lifetime Scientific Requirement	Due to scientific reasons, it is required to perform the experiment in lower altitudes where the ionospheric plasma reaches higher densities. Also, the system lifetime should not be longer than 1.5 years because of the degradation of the components. Therefore, the orbital lifetime shall be decreased and solutions for a faster de-orbit must be studied.
Scientific Data	Data Storage Data Rate Communication Time	As a scientific mission, MECSE will collect a considerable amount of scientific data. The system must be able to store all the data and transmit in a regular basis. Failure in transmitting it to the end user will compromise the mission success.
Risk	PL03 - EMG Flight Heritage	The EMG as well as the structure will be designed in-house which increases the risk of failure. Also, there is the risk of not being able to fit the experiment in a 3U configuration. Subsystems shall consider COTS and flight heritage components, as well as try to decrease the interfaces with other subsystems.
Cost	Financial Budget	The project has limited funding. Therefore, the project must consider educational opportunities and low-cost launch as well decrease the subsystems costs. Also, the project shall consider finding partners and investors.

4.5 Mission Requirements

Having defined clearly the objectives that the MECSE mission is to achieve, it is now possible to draw up the mission requirements. These are the translation of the objectives into verifiable and quantitative statements [47]. Note that requirements must be open to continuous iteration during the architecting phase since they must not constrain creative thinking [42].

The first estimate of mission requirements usually comes from the mission objectives combined with a preliminary feasibility analysis already performed in the previous section [25]. Here the requirements are divided in two types: constraints which are the limitations imposed on the system; and mission high-level requirements, mostly scientific, which define the system performance and operation. The constraints can be found in Table 4. 4, whereas the high-level requirements are presented in Table 4. 5.

To note that these will define the mission in a general way and they will be refined iteratively as the project progresses. In fact, most of the times they are just preliminary assumptions which are required to start the design of the SpaceCraft (S/C). Therefore, it is common to use TBC (to be confirmed) or TBD (to be determined) terminology when referring to parameters that are expected to change or are still unknown.

Table 4. 4 – System Constraints.

Constraint	Justification
CubeSat 3U Standard **ECSS**	CubeSat 3U configuration constraints both the mass and the volume. Therefore, the payload shall be designed with respect to those specifications. Also, all the ECSS (European Cooperation for Space Standardization) requirements must be taken into account.
Piggyback Launch	MECSE nanosatellite will be launched as secondary payload, therefore it is not able to choose the orbital elements neither the launch date precisely.
Budget	The budget is essential to the project. However, it is expected that the scientific and educational value of the mission could attract interesting collaborations and investments which would make this constraint less compelling.

Table 4. 5 - Mission high-level requirements.

# ID	Mission Requirement	Rationale
MR-01	The project shall open new positions for undergraduate students every semester.	MO1
MR-02	The S/C shall be able to study different plasma formation conditions when traveling in low Ionosphere.	MO2
MR-03	The S/C shall be able to collect data about the plasma layer and environmental characteristics.	MO2, MO4
MR-04	The S/C attitude shall be known with high accuracy [TBD].	MO3, MO4
MR-05	The S/C shall consider velocity-vector stabilization and orient the payload in the direction of motion.	MO3, MO4
MR-06	The S/C shall be able to measure the electron density reduction.	MO3, MO4
MR-07	The S/C shall be able to perform the PLME study at least twice per orbit with the duration of 1 second [TBC] per experiment.	Assumption
MR-08	The S/C shall be able to store the data and transmit it on a regular basis until disintegration.	Maximum data for the end user
MR-09	Low orbits altitudes [TBD] shall be reached before the end of life.	MO2, MO3, MO4
MR-10	The S/C shall be designed for a maximum of 1.5-year mission in space.	Components Lifetime
MR-11	The mission shall consider opportunities on low-cost launch.	Budget Constraint
MR-12	Commercial off-the-shelf (COTS) components, custom solutions and flight heritage will be preferred.	Risk Constraint
MR-13	The S/C shall be operational within 4 years.	Project Management

4.6 Concluding Remarks

In this chapter, MECSE mission has been successfully clarified in a concise manner which allows to be easily understood by all the current project team elements, as well as the future ones.

This means that from this point forward, MECSE is a mission with well-defined scientific aim and objectives which were evaluated as technically feasible through a preliminary feasibility analysis. This definition is of the most importance because it allows the project team to start divide efforts for the design of the mission and the space system.

At this phase, the following tasks have already been performed [25], [38]:

- Identify mission needs;
- Identify and involve users and stakeholders;
- Identify alternative mission scenarios and architectures;
- Perform preliminary evaluation of possible mission scenarios;
- Identify and perform trade studies and preliminary analyses;
- Define the mission baseline;
- Perform the preliminary feasibility study of the mission;
- Define mission statement and objectives;
- Define mission parameters to be analyzed in more detail in the future phases;
- Identify mission requirements, which include science, functionalities and constraints.

Therefore, from a systems engineering technical point of view, the mission is already ready for the MDR, which marks the end of the phase 0 of the ESA's project lifecycle and the beginning of phase A (see Figure 2. 14). This work was one of the main goals of this MSc thesis and it represents a huge contribution for MECSE project.

Chapter 5

5 Mission Analysis

Mission analysis is the process of analyzing the mission parameters and the resulting performance to guarantee that the mission requirements are fulfilled [24]. Orbital analysis is one of the most important tasks to be performed in the early phases of the project because it allows to evaluate the orbital lifetime, the attitude of the spacecraft, and the eclipse and access times. These parameters will have impact in the system design.

5.1 Astrodynamics

"Astrodynamics is the study of the motion of man-made objects in space, subject to both natural and artificially induced forces" [48]. It combines knowledge from orbital mechanics, which studies the motion of orbiting bodies, with attitude dynamics, which deals with the orientation of an object in space [48]. It is important to understand the fundamentals of astrodynamics in order to evaluate the orbital effects on the motion of the spacecraft [25].

5.1.1 Orbital Elements

Beginning with the foundations, celestial mechanics define a Keplerian orbit as one on which gravity is the only force acting on the space body. This orbit obeys to the three laws of planetary motion defined by Kepler and latterly justified by Newton's gravitational theory. Here, these laws are detailed in the form derived by Newton [25] [49]:

First Law: *"If two objects in space act gravitationally, each will describe an orbit that is a conic section with the center of mass at one focus. If the bodies are permanently associated, their orbits will be ellipses; if not, their orbits will be hyperbolas."*

Second Law: *"If two objects in space interact gravitationally (whether or not they move in closed elliptical orbits), a line joining them sweeps out equal areas in equal intervals of time."*

Third Law: *"If two objects in space revolve around each other due to their mutual gravitational attraction, the sum of their masses multiplied by the square of their period of mutual revolution is proportional to the cube of the mean distance between them, that is:"*

$$(m+M)P^2 = \frac{4\pi^2}{G}a^3 \tag{5.1}$$

where P is their mutual period of revolution, a is the mean distance between them, m and M are the masses of each body, and G is Newton's gravitational constant. These laws may as well be applied to the motion of a spacecraft (with mass m) around a planet (with mass M). Additionally, the problem can be simplified by considering the spacecraft as a point of mass without dimensions. Bearing that in mind, it is possible to characterize the motion of a satellite along its orbit through six orbital elements (see Figure 5. 1): two to describe the size and shape, three to describe the orientation and one to describe the satellite location [48], [50].

Semi-major Axis (a): The semi-major axis describes the size of the orbit. For a circular orbit, the semi-major axis is the radius of the Earth plus the altitude (h) of the spacecraft, whereas for elliptical orbits, the semi-major axis is half of the major axis diameter.

Eccentricity (e): The eccentricity describes the shape of the orbit. It is the distance from the center of an ellipse to the focus divided by the semi-major axis. For a circular orbit, the eccentricity is 0.

Inclination (i): The inclination describes the orientation of the orbit in space. It is defined as the angle between the orbit plane and the equatorial plane.

Right Ascension of the Ascending Node (Ω or $RAAN$): The RAAN describes the orientation of the spacecraft. It is the angle between the vernal equinox and the ascending node which is the point where the spacecraft crosses the equatorial plane traveling from south to north.

Argument of Perigee (ω): The argument of perigee defines the orientation of the ellipse in the orbital plane. It is the angle measured from the ascending node to the perigee.

True Anomaly (v): The true anomaly describes the location of the satellite within the orbit. It is the angle measured in the direction of motion from the perigee to the satellite's position at the given time.

These orbital parameters are also dependent upon a reference date, referred as **Epoch**. This information is required because the orbit and spacecraft position will change over time.

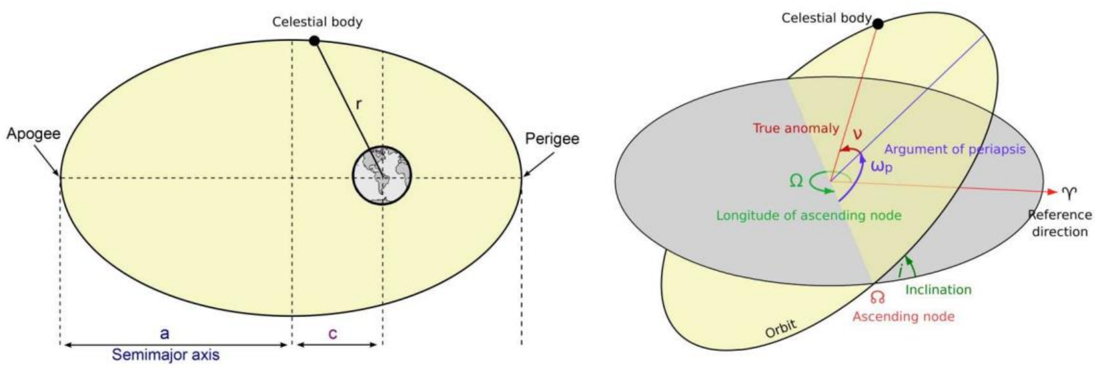


Figure 5. 1 - Classical orbital elements (from [49]).

5.1.2 Orbit Perturbations

Even though Keplerian orbits provide a convenient analytic approximation to a true orbit, they are based on a spherically symmetric mass distribution and do not take into account non-gravitational forces or the gravity of other bodies. Consequently, real orbits never follow Kepler's Laws precisely. In this section, some perturbations regarding Earth orbits are discussed which will impact the orbital elements and the attitude of the spacecraft [25].

Non-spherical Mass Distribution

Earth's gravitational force is often modelled with the assumption that it is an inert sphere of symmetric mass distribution. In reality, the Earth is an oblate spheroid with an equatorial bulge and flattening at the poles. This oblateness is caused particularly due to the Earth's rotation rate. In addition, there are minor mass anomalies in Earth's topography, such as continents and mountain ranges, that need also to be considered [25], [51]. The geoid, which is the surface of equal gravitational potential of a hypothetical ocean at rest, is represented on Figure 5. 2

To take into account all these aspects, it is possible to model the Earth using a geopotential model which is a set of coefficients in the spherical harmonic expansion. For example, common Goddard Earth model 10B is a 21x21 matrix of coefficients. Earth's geopotential causes periodic variations in all orbital elements but these are dominated by the secular variations in RAAN and argument of perigee due to the J_2 coefficient, which is the largest of the geopotential terms. This coefficient is often called "Earth oblateness" since it represents the mass distribution of the equatorial bulge [25], [49], [50].

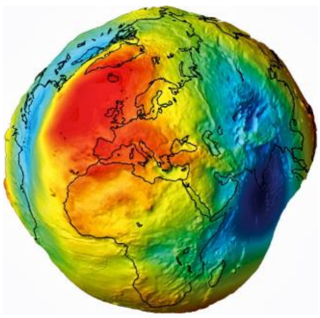


Figure 5. 2 – The Earth geoid in an exaggerated scale (from [50]).

Third-body Perturbations

The third-body perturbations are dominated by the gravitational forces of the Sun and the Moon (referred as luni-solar). These forces cause small periodic variations in all the orbital elements but the RAAN and argument of perigee experience secular variations. The effect is generally similar to the Earth's equatorial bulge described above and it is extremely small in LEO [25].

Solar Radiation Pressure

Solar Radiation Pressure (SRP) causes periodic variations in all the orbit elements. However, it is predictable and consistent, so it does not require sophisticated numerical models [52]. The effect is stronger for satellites with large area to mass ratios. The magnitude of the acceleration a_r in m/s^2 due to solar radiation pressure is approximately [25]:

$$a_r = -4.5 * 10^{-6} \frac{A_{SRP}}{m} \tag{5.2}$$

where A_{SRP} is the area of solar radiation pressure in m^2 and m is the mass of the spacecraft in kg. For satellites in LEO the effects are negligible [50].

Atmospheric Drag

Atmospheric drag is the principal non-gravitational force acting on most satellites in LEO. Drag acts opposite to the direction of the velocity vector, thus slowing the satellite and removing energy from the orbit [25], [48]. This loss of energy causes a positive feedback effect (Figure 5. 3): the more the orbit decays, the lower the altitude and the faster the decay. This is because the drag is higher in lower orbits where the atmosphere is denser. Eventually, the altitude becomes so small that the satellite reenters the atmosphere [49].

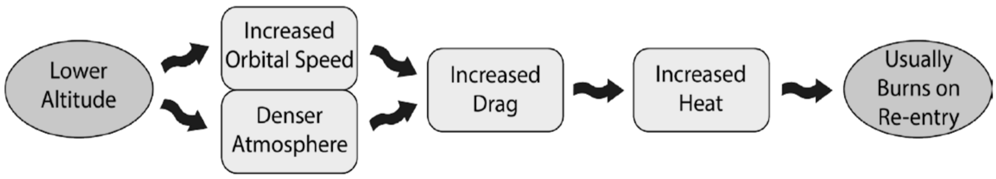


Figure 5. 3 – Positive feedback effect during orbital decay of a satellite (from [49]).

For circular orbits, drag will act continuously, and the orbit will spiral downward. In case of elliptic orbit, the drag acts mainly at the perigee lowering the altitude of the apogee. Consequently, the semi-major axis is reduced and the orbit leans towards becoming circular [25]. The drag force D on the spacecraft is given by the equation:

$$D = \frac{1}{2}\rho V^2 C_D A \tag{5.3}$$

where ρ is the atmospheric density, V the sattelite velocity, C_D the drag coefficient and A the cross-sectional area perpendicular to the direction of motion.

Atmospheric drag plays a critical role when analyzing the orbital lifetime of a satellite in LEO, since it is the primary cause of orbital decay. The effects of drag for orbital lifetime predictions will be further discussed in section 5.4.

5.1.3 Coordinate Frames and Attitude Dynamics

To define a coordinate system for space applications, one must first specify two characteristics [25]: the location and motion of the origin and what the coordinate system is fixed with respect to. Table 5. 1 lists the most common coordinate systems used in space mission engineering and their applications. Also, they are illustrated in Figure 5. 4.

Table 5. 1 – Common coordinate systems used in space applications (adapted from [25]).

Coordinate System	Fixed with respect to	Center	Z Axis	X Axis	Applications
Inertial	Inertial Space	Earth or S/C	Celestial Pole	Vernal Equinox	Orbit Analysis, Inertial Motion
Earth-Fixed	Earth	Earth	Earth Pole	Greenwich Meridian	Geolocation
Spacecraft-Fixed	S/C	Engineering Drawings	S/C axis toward nadir	S/C axis in direction of velocity vector	Positions and orientation of S/C instruments
Local Horizontal (RPY)	Orbit	S/C	Nadir	Toward velocity vector	Attitude Maneuvers

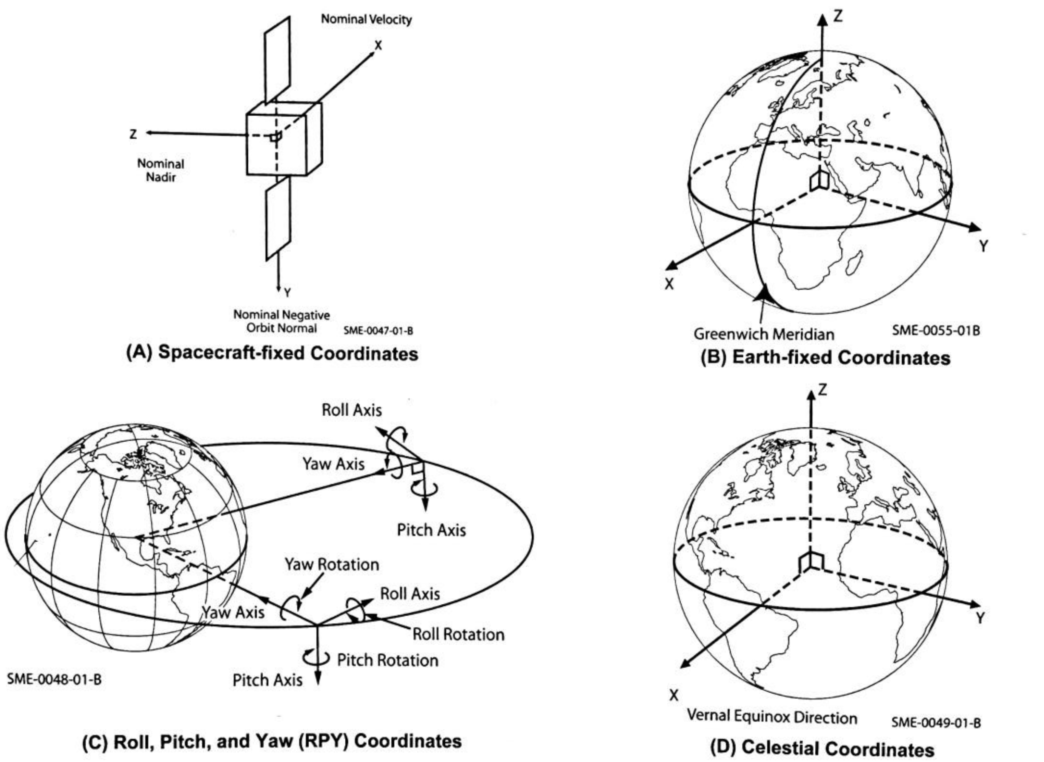


Figure 5. 4 – Coordinate systems used in space mission engineering (from [25]).

The attitude of a satellite can be described with respect to reference frames. The two reference frames used for spacecraft attitude dynamics through this work were the orbit reference frame (RPY coordinate system) and the body reference frame (spacecraft-fixed coordinate system).

The orbit reference frame is fixed to the orbit position with the X_O and Z_O axes within the orbital plane. The Z_O axis points to nadir, X_O points in the direction of the orbital velocity of the spacecraft, and the Y_O axis points in the orbit anti-normal direction, completing the right-hand set. The body reference frame is fixed to the spacecraft body, with its origin at the center of mass. The X_B, Y_B and Z_B axis need to be perpendicular to each other and should be popping out of the different faces of the satellite [53].

The body frame is usually chosen to match with the orbit reference frame when the satellite is in its normal attitude. However, as the attitude of the satellite changes, the two frames will misalign. This feature is useful when analyzing the attitude of the spacecraft. Figure 5. 5 illustrates the differences between the orbit and the reference frame are illustrated.

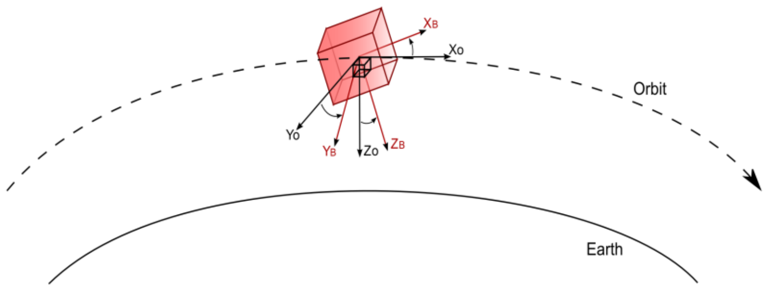


Figure 5. 5 – Orbit (O) and Body (B) reference frames (from [53]).

As MECSE Nanosatellite follows the 3U CubeSat standardization, the body reference frame shall comply with the CubeSat Design Specification [54]. Therefore, to be compliant with the MR-05 requirement, the reference attitude of the spacecraft is determined when Z_B axis (of the satellite body frame) is aligned with the X_O axis, which is the velocity direction. Figure 5. 6 illustrates the orbit reference frame used for attitude analysis.

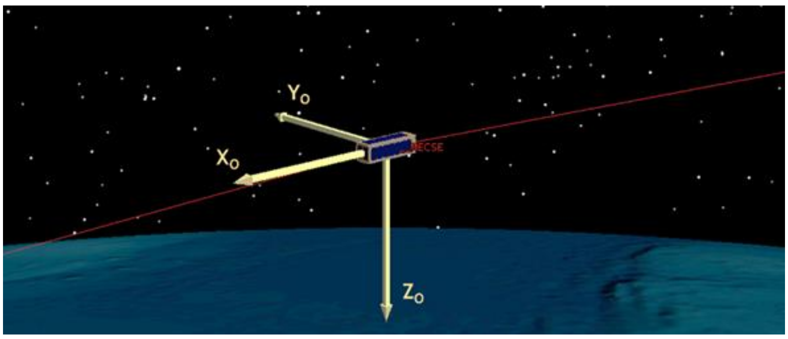


Figure 5. 6 – MECSE's orbit reference frame considered for attitude analyses.

5.2 Models and Tools for Simulations

Having understood the theoretical background about astrodynamics, it is now essential to select the tools used for the analyses. In this thesis, the Systems Tool Kit (STK) from Analytical Graphics Incorporated (AGI) is used for the major part of the orbital simulations [55]. Additionally, the OSCAR (Orbital Spacecraft Active Removal) tool from DRAMA (Debris Risk Assessment and Mitigation Analysis) software provided by ESA, is also used for orbital lifetime analysis [56]. This allowed to compare the results between two different tools.

5.2.1 Orbit Propagation

Propagation concerns the determination of the motion of a body over time, which depends on its initial state and the forces that act upon it [55]. The propagator is the tool used to solve the equations of motion of the satellite in orbit. As a result, the accuracy of the results will depend on the selected propagator. High fidelity propagators use a numerical integration approach and attempt to include all significant force models acting on the body. On the other hand, low fidelity propagators tend to analytically approximate the effects of some forces while completely disregarding others [55].

For the STK analysis, the High-Precision Orbit Propagator (HPOP) is selected. It uses a numerical integration of the differential equations of motion. The orbit propagator used in OSCAR is a semi-analytical propagator known as FOCUS-1A (Fast Orbit Computation Utility Software).

Several models for the main perturbations can be included as inputs in those propagators such as gravitational field model (based upon spherical harmonics), third-body gravity, atmospheric drag and solar radiation pressure.

5.2.2 Geopotential and Third-Body Perturbations Model

Both of the propagators were used with the GEM-T1 (Goddard Earth Model) gravity field model [55]–[57] which considers geopotential coefficients J_2 through J_5, including J_2 short-periodic variations and luni-solar gravity attraction .

5.2.3 Atmospheric Density Model

As mentioned in section 5.1.2, the atmospheric drag plays a key role in the orbital analysis of LEO satellites. Thus, for a suitable analysis, an appropriate atmospheric model needs to be chosen. For the orbital analysis presented in this thesis NRLMSISE-00 model was chosen. This model has been largely used in literature and shows accurate results. Also, it is one of the last update models available having more atmospheric drag data incorporated [58], [59].

5.2.4 Solar and Geomagnetic Activity Model

The solar and geomagnetic activity is one of the most important parameters to be taken into account during orbital analysis since it directly influences the atmospheric density [25], [60]. However, predicting this activity is complex and difficult to perform with reasonable accuracy. The solar activity is defined by a 11-year cycle during with very large month-to-month variations. Also, the solar cycles are not constant, which means that some solar cycles levels are much higher than other solar cycles [25] (see Figure 5. 7).

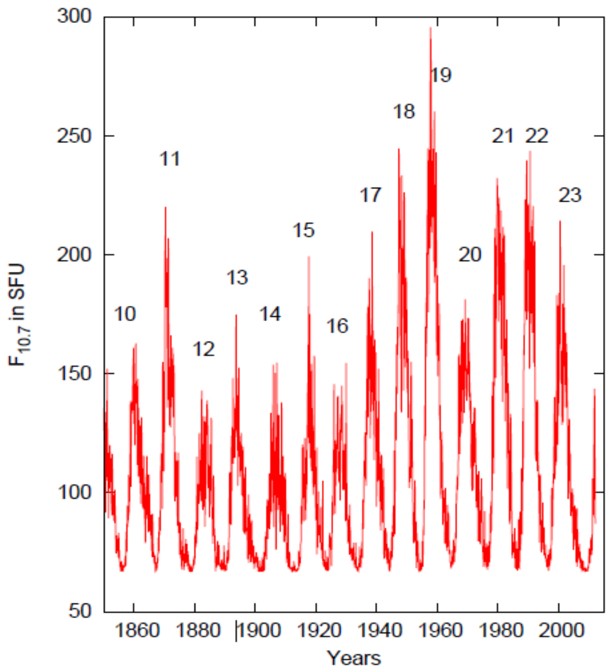


Figure 5. 7 - Mean solar activity from 1850-2012 divided in solar cycles (from [57]).

OSCAR tool provides different methods for the forecast of solar and geomagnetic activity based on recommendations from ECSS (European Cooperation for Space Standardization) and ISO (International Organization for Standardization) [56], [57], [60]. For the purpose of this thesis, the Latest PredictioN (LPN) method and the ECSS method are used within OSCAR. The LPN is recommended by ISO [61] and uses latest available data of the current solar cycle to predict the future evolution of the solar and geomagnetic indices. The ECSS [62] method describe the future predictions based on a repetition of the 23rd solar cycle. The results are then compared with the STK CSSI files which are used by the Lifetime Tool in STK software to predict values of the monthly mean solar flux and geomagnetic index [55].

Even though there are many existing models to forecast solar and geomagnetic activity, all of them are still based on simplifying assumptions as the underlying physics of the solar and geomagnetic interaction are not well understood [59]. Thus, in this thesis, the influence of different modeling approaches on orbital analyses shall be investigated.

5.3 Trajectory Analysis

Having understood the models and tools to be used for simulations, one can start designing the trajectory of the mission. Trajectory analysis is often a crucial part of mission analysis due to its highly importance in the mission performance. The choice of the initial orbit is a major step in every space mission since it strongly influences the system's design. However, MECSE does not have any launch contract fixed yet and therefore does not have knowledge about the orbit it will be inserted in. In this view, the mission analysis must contemplate a wide range of possible orbits. In this section, the trajectory of the nanosatellite is designed. Also, a launch survey is performed in order to assume the launch vehicle and initial orbit. These parameters are crucial inputs for the orbital analyses and system conceptual design.

5.3.1 Mission Profile

MECSE trajectory can be divided in 5 broad mission phases, starting from the launch to end of life. The mission profile of MECSE is summarized in Figure 5. 8.

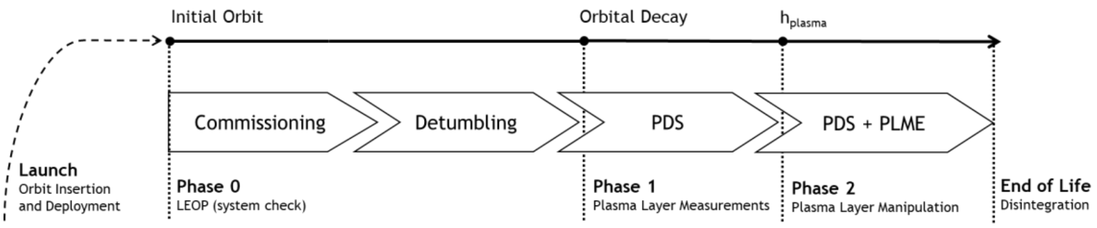


Figure 5. 8 – MECSE mission profile.

The launch phase includes the launch sequence and the deployment of the CubeSat from the orbital deployer (P-POD) into the initial LEO orbit. After that, the Launch and Early Orbit Phase (LEOP) starts. The vital systems are booted and the antennae are deployed in order to establish contact with the ground station which will allow to check the systems' condition. In this phase, the satellite will also be detumbled and stabilized from the initial tumbling rates. The Z axis of the satellite (Z_B) shall be aligned with the velocity vector and the attitude shall be maintained.

Afterwards, the scientific studies start. Firstly, the spacecraft will only perform plasma dynamics studies collecting data about the environment and the plasma around the vehicle. During this phase, the satellite is decaying due to atmospheric drag. Depending on the initial orbit (defined by the primary payload on the space vehicle), this phase duration will vary. Also, at high altitudes the plasma layer density will be too low to be manipulated. Therefore, the spacecraft will have to decay the faster as possible to a lower altitude (h_{plasma}) where the plasma density is high enough to perform plasma manipulation studies. This altitude will depend on several factors such as the solar activity. Once there, the scientific studies will be combined until the end of life, which corresponds to the disintegration of the satellite in the atmosphere.

5.3.2 Launch Survey

Given that the launch of MECSE project has not yet been stablished, a launch survey is carried out in order to identify launch vehicles and future launch opportunities, as well as to understand what types of orbits are more common among different launch services. The goal is to select a launch vehicle and an initial orbit to be used as preliminary references.

Even though that there are already companies that can provide launch services, it is still difficult to find precise information about future launch campaigns and rideshare opportunities. As MECSE is an educational mission, one solution would be to apply for educational programs to come such as ESA's "Fly your satellite!" [63] or QB50 similar projects [46], which offer the possibility of designing, testing and launching the satellite at very low cost.

For the mission analysis, initial considerations about the choice of the launch date, launch vehicle and initial orbit are required. The first step is to identify available launch vehicles and their launch frequency which will allow to consider different launch scenarios. Table 5. 2 shows some examples of launch vehicles that have already launched small satellites to LEO for educational purposes.

Table 5. 2 - Launch vehicles already used in educational space programs.

Launch Vehicle	PSLV	Falcon 9	Minotaur	Vega	Dnepr
Operator	ISRO	Space X	Orbital ATK	Arianespace	ISC Kosmotras
Country	India	USA	USA	Europe	Russia and Ukraine
Launch Frequency	~2 a year	~2 a year	~2/3 a year	~1/2 a year	~3 a year

From Table 5. 2, the Vega launcher was the one considered due to the primary reason that is European and is often used for ESA educational space programs [63].

Additionally, it is important to search for future launch services that could also suit the mission in order to understand what are the most common orbits used, as well as the frequency which are reached. Thus, data about future launch services was gathered and is presented in Table 5. 3. Notice that only the services that could suit the mission are shown. For this purpose, all the LEO orbits with a perigee altitude equal or higher than 500 km were not considered since they will require an active deorbiting system (which add unnecessary complexity to the system design) to fulfill MR-08 and MR-09. The International Space Station (ISS) orbit was also taken into consideration due to the fact that the ISS is often used to deploy small satellites [51], [63].

Table 5. 3 – Future launch opportunities survey (H - Half; Q - Quarter; SSO – Sun Synchronous Orbit).

<table>
<tr><th>Launch Service Provider</th><th>Orbit Type</th><th colspan="2">Launch Dates</th><th>Launches [2017-20]</th></tr>
<tr><td rowspan="4">Spaceflight</td><td>220 - 420 km
52.6°</td><td>Q4 2017
2x Q1 2018
2x Q2 2018
Q4 2018</td><td>Q1 2019
Q2 2019
2x Q4 2019</td><td>10</td></tr>
<tr><td>460 km 45°</td><td colspan="2">Q2 2018</td><td>1</td></tr>
<tr><td>450 - 500 km
52.6°</td><td>Q2 2018
Q4 2018</td><td>Q3 2019
Q4 2019</td><td>4</td></tr>
<tr><td>400 - 500 km
SSO</td><td colspan="2">Q2 2018
Q3 2018
Q4 2019</td><td>3</td></tr>
<tr><td rowspan="2">ISIS</td><td>400 - 500 km
SSO</td><td>H2 2017
2 x Q4 2017
Q1 2018
H1 2018</td><td>Q2 2018
2x H2 2018
H2 2019
Q4 2019</td><td>10</td></tr>
<tr><td>450 - 500 km 45°</td><td colspan="2">H2 2018</td><td>1</td></tr>
<tr><td>PLD</td><td>400 km 116°-160°</td><td colspan="2">Q3 2021</td><td>1</td></tr>
<tr><td>ISS</td><td>~400 km 51.6°</td><td colspan="2">N/A</td><td>N/A</td></tr>
</table>

From Table 5. 3, it is concluded that there are several launch opportunities per year to LEO with very low altitudes at 52.6° inclination. These altitudes would be advantageous for MECSE.

5.3.3 Initial Orbit Selection

Considering the launch survey performed, the initial orbit must be selected. The perigee altitude and inclination are chosen based on the high launch frequency offered by spaceflight to 52.6° inclination (Table 5. 3). Regarding the other orbital elements, some assumptions are necessary.

The first assumption is that the orbit is circular, thus the altitude of apogee and perigee are the same and the eccentricity is 0. Secondly, the argument of perigee, RAAN and true anomaly were quantified arbitrarily as 0. Thirdly, the epoch time assumed was the 1st of January 2020, which represents the launch date.

Table 5. 4 summarizes the orbital elements for the initial selected reference orbit. Figure 5. 9 shows a tridimensional view of the orbit and Figure 5. 10 shows a typical orbit ground track.

Table 5. 4 -Orbital details of MECSE's initial reference orbit.

Epoch	1-Jan-2020
Orbit Type	LEO
Altitude of Apogee / Perigee	350 km
Eccentricity	0
Inclination	52.6°
Argument of Perigee	0°
RAAN	0°
True Anomaly	0°
Orbital Period	1.52 h
Orbital Velocity	7.7 km/s

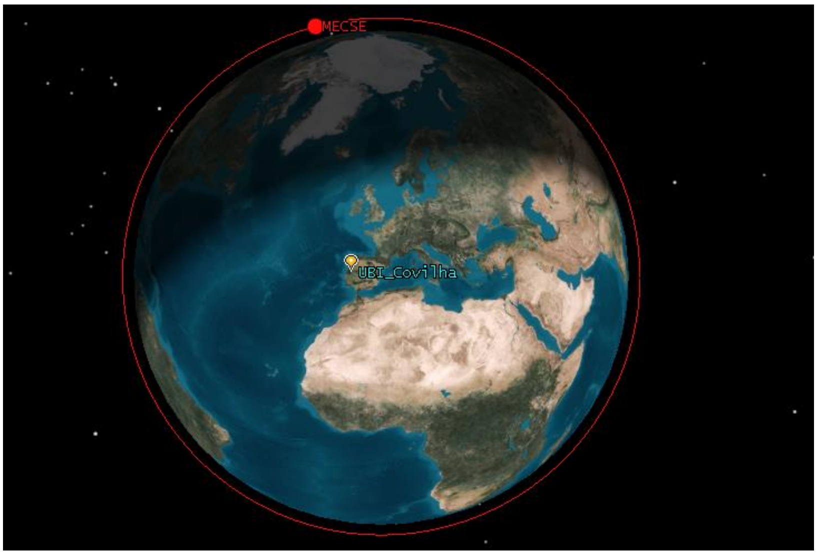


Figure 5. 9 – MECSE's initial orbit.

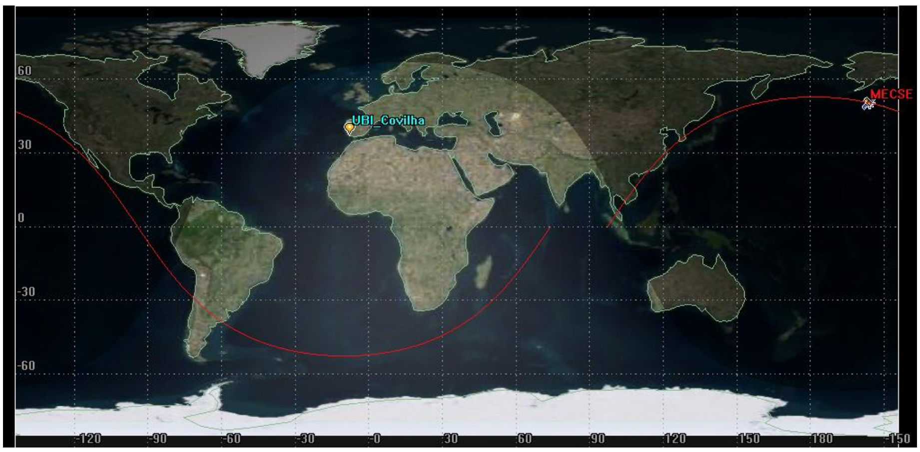


Figure 5. 10 - MECSE's typical ground track.

5.4 Orbital Lifetime

5.4.1 Overview

Orbital Lifetime (OL) estimates the amount of time a low Earth orbiting satellite can be expected to remain in orbit before the drag of the atmosphere causes it to re-entry [25]. For scientific reasons, MECSE's orbital lifetime will directly impact the mission performance. Therefore, an analysis of the satellite decay is necessary [58].

As discussed in section 5.1.2, the drag caused by the residual atmosphere at low orbital altitudes is the main cause of the orbital decay [51]. Thus, an accurate determination of the aerodynamic force is key when deriving the lifetime of an object [52]. However, even though the physics of atmospheric drag are very well understood, drag is almost impossible to predict with reasonable precision [25]. This difficultly arises because it depends on three main set of parameters: the satellite characteristics, the initial orbital elements and the atmosphere density, which varies itself with the solar and geomagnetic activity.

Regarding the satellite parameters, there are three that particularly influence the orbital decay: mass, drag coefficient, and cross-sectional area. More information about those parameters is given in the following section. For the orbital elements, ISO [61] states that for non-Sun-Synchronous orbits the lifetime results are not sensitive to the three angular orbital elements (RAAN, argument of perigee and true anomaly). Therefore, only the semi-major axis, eccentricity and inclination are considered for the simulations. Lastly, *"the atmosphere density varies by as much as two orders of magnitude depending upon the solar activity level"* [25]. Thus, special attention must be given to the solar and geomagnetic models used, as well as to the epoch time which is correlated with the solar flux level (see section 5.2.4).

In this section, the effects of the three set of parameters in orbital lifetime results are investigated (see Figure 5. 11). The approach used to conduct the investigation is divided in small studies performed in DRAMA and STK software. Each study is conducted using the three solar and geomagnetic activity models (see section 5.2.4): LPN, ECSS and STK CSSI files.

Firstly, it is necessary to validate the method and the models chosen. This can be done by simulating satellites that have already decayed. As the decay date is known, one can analyze the accuracy of the models by comparing the simulation results with historical data.

Secondly, a sensitivity study concerning MECSE'S spacecraft parameters is performed by fixing the orbital elements. MECSE's initial reference orbit and epoch are assumed. The study will contemplate a range of cross-sectional areas and drag coefficients.

Thirdly, a sensitivity study regarding the orbital elements is performed by fixing the satellite characteristics and epoch. Assuming circular orbits, the orbital lifetime for different altitudes and inclinations will be investigated. For simplicity reasons, only circular orbits are considered.

Finally, a sensitivity study regarding the epoch will be performed by fixing the orbital elements and the satellite parameters. Once more, MECSE's initial reference orbit is assumed.

It must be emphasized that although lifetime computations are based on sophisticated orbital theory and accurate environment models, the result is still an approximation. Due to the large variations in atmospheric density and because of the difficulty in accurately modeling solar activity, satellite lifetimes cannot be determined with accuracy higher than 10% [55]. Furthermore, assumptions and approximations made through the process introduce an additional degree of uncertainty in the final result.

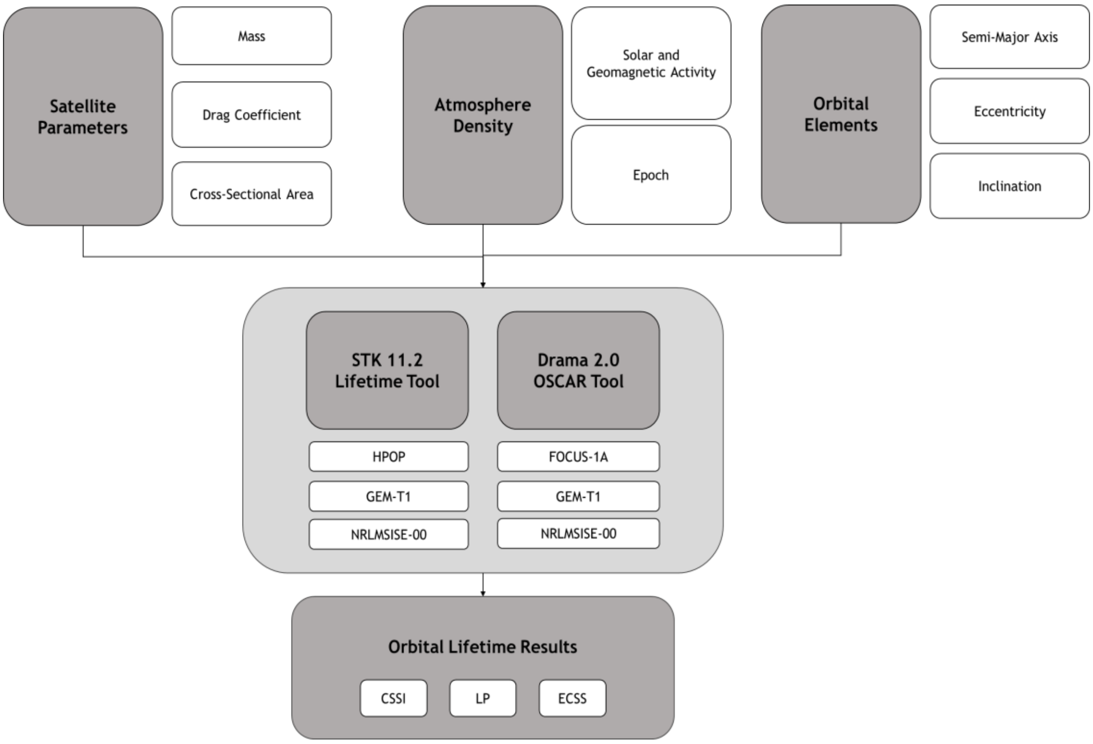


Figure 5. 11 – Set of parameters and models considered that can impact orbital lifetime prediction.

In summary, given that one of the main goals of this thesis is to investigate the impact of different solar and geomagnetic activity modeling in orbital lifetime predictions, the results using three different models are always provided through the studies. Therefore, all the studies will result on an interval of orbital lifetime predictions that can include the three models.

5.4.2 Satellite Parameters

The satellite parameters must be known in order to predict atmospheric drag. This includes the mass, geometry and surface characteristics, as well as the satellite attitude when travelling within the atmosphere.

Beginning by the basics of aerodynamics, the acceleration due to drag, a_D, is defined as [25]:

$$a_D = -\frac{1}{2}\rho\, V_s^2 \left(\frac{C_D\, A}{m}\right) \tag{5.4}$$

where ρ is the atmospheric density, V_s the satellite velocity, C_D the drag coefficient, m the satellite mass and A the cross-sectional area perpendicular to the direction of motion.

Mass (m)

Satellite mass is probably the easiest parameter to set in calculations because it is often known and it is constant when there are no thrusting capability on the satellite [59]. For CubeSats the mass is often fixed by the standardized FF. The mass of MECSE is set to be 4 kg.

Cross-Sectional Area (A)

The cross-sectional area is the area perpendicular to the direction of motion, which depends on the satellite's attitude. For the typical cases which the attitude of the spacecraft cannot be anticipated, the user shall compute a mean cross-sectional area by assuming that the attitude will vary uniformly relatively to the velocity direction [58], [61].

ISO guidelines [61] recommend to use a composite flat-plate model. Using this model, the mean cross-sectional area of a CubeSat spacecraft (parallelepiped-shaped) can be calculated as:

$$A_{mean} = \frac{S_1 + S_2 + S_3}{2} \tag{5.5}$$

where S_1, S_2 and S_3 are three visible surfaces. Note that, assuming this model, the cross-sectional area of a CubeSat becomes standardized for a given FF regardless its attitude. This model has been shown to be accurate to within 20% for tracked objects [58], [61].

On the other hand, if the attitude can be anticipated the analyst shall try to predict the cross-sectional area. As one of the objectives of the sensitivity analyses is to study the impact of the cross-sectional area in orbital lifetime, different cross-sectional areas will be considered.

Drag Coefficient (C_d)

The drag coefficient is a dimensionless parameter used to quantify the object's resistance behavior in a fluid environment [52].

For this thesis, it is used when modeling aerodynamic drag across LEO's free molecular flow regime. Unfortunately, it is a complex parameter to predict since it depends primarily on four factors: shape, attitude, surface condition and multi-collision effects (form drag) [25], [59]. Thus, to estimate this value one must recur to the literature.

ISO guidelines for orbital estimation [61] suggest that 2.2 is a reasonable value of the drag coefficient for typical spacecraft. This information is supported by former orbital lifetime analyses such as [51], [56] and [58]. Although, both J. Wertz [25] and Vallado [59] suggest that it is in the range of 2-4 depending on the above characteristics and the altitude (Figure 5. 12). Furthermore, both declare that it must be higher than 2.2 for flat plates.

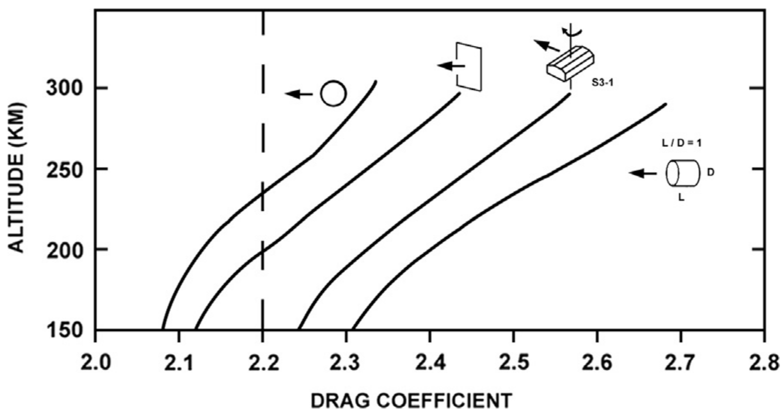


Figure 5. 12 - Drag coefficient values for different shapes and altitudes (from [59]).

Once more complying with the objectives, the impact of the drag coefficient in orbital lifetime is aimed to be studied. Four different values of drag coefficient will be considered for the simulations: 2.2, 2.5, 3 and 4.

Ballistic Coefficient (BC)

The previous individual parameters can also be combined in a single one called the ballistic coefficient. It can be defined as:

$$BC = \frac{m}{C_D\, A} \tag{5.6}$$

where m is the mass of the satellite, C_d the drag coefficient, and A the cross-sectional area.

Solar Radiation Pressure Area (A_{SRP}) and Coefficient (Cr)

The effect of SRP is negligible in LEO and it does not affect lifetime predictions. However, both tools require the input of A_{SRP} and Cr. It is assumed $A_{SRP} = A$ [51] and $Cr = 1.3$ [56].

5.4.3 Validation Study

The validation study aims to compare the orbital lifetime results given by the computer simulations with historical data of orbital decay. Two CubeSats were selected to conduct the study: AeroCube-3 and GeneSat-1. Information about the two nanosatellites can be found at eoPortal Directory [64] and is summarized in Table 5. 5.

Table 5. 5 – Historical data about the CubeSat study cases.

Parameter	AeroCube-3	GeneSat-1
Form Factor (FF)	1U	3U
Mass [kg]	1.1	4.1
Dimensions [m^3]	0.1 x 0.1 x 0.1	0.1 x 0.1 x 0.3
A_{mean} [m^2]	0.015	0.035
Orbit	433 x 473 km 40.5°	460 km 40.5°
Launch Date	19-May-2009	16-Dec-2006
Decay Date	6-Jan-2011	4-Aug-2010
OL Observed [yrs]	1.64	3.64

Both nanosatellites use a simple passive magnetic system for attitude control, so it is reasonable to assume the value of their mean cross-sectional area. By doing this, only the drag coefficient remains unknown which allows to study the effects of the drag coefficient value on the orbital lifetime, as well as to evaluate the accuracy of different modeling approaches by comparing it with the observed orbital lifetimes. The simulation results are shown in Table 5. 6.

The error was calculated by comparing the simulated Orbital Lifetime (OL_{sim}) with the observed one (OL_{obs}) using the following relationship:

$$error = \frac{|OL_{sim} - OL_{obs}|}{OL_{obs}} \times 100 \qquad (5.7)$$

As expected, it is observed that the satellite decays faster for higher drag coefficients. Moreover, the results confirm that there are large deviations between different modeling approaches even using the same parameters. While for STK CSSI case the most accurate results are outputted with drag coefficient values near 2.5, the LPN suggest using values near 4. ECSS does not show a conclusive relationship since it shows different accuracy relationships for the two satellites. This also shows that different models require different values of drag coefficient.

Table 5. 6 – Error between simulated and observed orbital lifetimes.

S/C	C_D	BC [kg/m^2]	OL Simulated [yrs]			Error [%]		
			CSSI	LPN	ECSS	CSSI	LPN	ECSS
AeroCube-3	2.2	33.33	1.80	2.53	1.59	10.05	54.68	2.79
	2.5	29.33	1.60	2.42	1.46	2.18	47.96	10.74
	3	24.44	1.50	2.20	1.36	8.29	34.51	16.85
	4	18.33	1.20	1.84	1.14	26.63	12.50	30.30
GeneSat-1	2.2	53.25	4.00	5.28	4.93	10.02	45.23	35.60
	2.5	46.86	3.80	5.04	4.77	4.52	38.63	31.20
	3	39.05	3.40	4.74	4.46	6.48	30.38	22.68
	4	29.29	2.90	3.94	3.97	20.23	8.37	9.20

From Table 5. 6, it can be concluded that CSSI is the most accurate model given that it provides an error lower than 5% for a drag coefficient of 2.5 for both nanosatellites, which shows that the model is also consistent. Therefore, even though the other models are always presented in the following sections for comparison reasons, the results given by this specific model will be considered as the more reliable ones. Furthermore, this study allowed to estimate MECSE drag coefficient. The value of 2.5 will be used given its very good accuracy and consistency through the validation study. The simulations of orbital decay for the lowest error are in Appendix A.

In conclusion, the results presented confirm that orbital analysis cannot rely only on standard guidelines for accurate orbital lifetime predictions. For instance, ISO [61] proposes to use a drag coefficient value of 2.2 which does not provide accurate results. For GeneSat-1 the error is always higher than 10% for any model. This reinforces the huge importance of performing sensitivity studies within orbital analyses in order to estimate values and analyze results. This work is performed in the following sections.

5.4.4 Sensitivity Study of Satellite Parameters

The first step for the sensitivity study is to estimate the satellite parameters and understand their impact for orbital lifetime estimations. As MECSE is a triple CubeSat, this study focuses on the effects of attitude and drag coefficient. Table 5. 7 summarizes the parameters used for the simulations. Also, the initial reference orbit (see Table 5. 4) is assumed.

Table 5. 7 – MECSE Parameters for the simulation.

MECSE Parameters	
Form Factor (FF)	3U
Mass [kg]	4.0
Dimensions [m^3]	0.1 x 0.1 x 0.3
Orbit	350 km 52.6°
Epoch	1-Jan-2020

Simulations were performed to evaluate the effect of variations of the satellite parameters on orbital lifetime. The numerical results are shown in Table 5. 8.

Table 5. 8 - Orbital lifetime predictions for different combinations of MECSE parameters.

MECSE Parameters			Orbital Lifetime Predictions [yrs]		
A [m^2]	**CD**	**BC [kg/m^2]**	**CSSI**	**LPN**	**ECSS**
0.01	2.2	181.80	1.70	1.76	1.90
	2.5	160.00	1.50	1.61	1.79
	3.0	133.30	1.30	1.39	1.60
	4.0	100.00	0.97	1.12	1.30
0.02	2.2	90.91	0.89	1.03	1.21
	2.5	80.00	0.80	0.92	1.08
	3.0	66.67	0.68	0.80	0.91
	4.0	50.00	0.50	0.62	0.71
0.03	2.2	60.61	0.62	0.74	0.84
	2.5	53.33	0.53	0.66	0.75
	3.0	44.44	0.43	0.54	0.63
	4.0	33.33	0.33	0.39	0.45
0.035	2.2	51.95	0.52	0.64	0.74
	2.5	45.71	0.45	0.56	0.65
	3.0	38.10	0.37	0.45	0.53
	4.0	28.57	0.28	0.34	0.38

Regarding attitude, MECSE aims to achieve velocity-vector stabilization (see Figure 5. 6), which means that, most of the time, the area perpendicular to the direction of motion will be smaller than the mean one. So, a range of cross-sectional areas varying from the minimum value to the mean one was analyzed. Also, the drag coefficients were evaluated together with the different modeling approaches.

For the discussion of the results, the ballistic coefficient value becomes useful since it combines the satellite parameters. Thus, the orbital lifetime is presented as a function of the ballistic coefficient in Figure 5. 13.

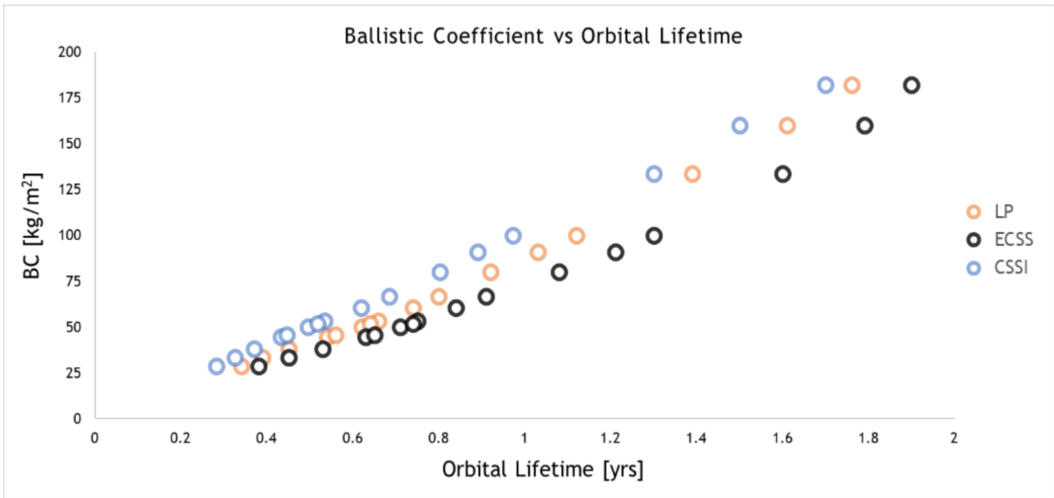


Figure 5. 13 – Effects of the ballistic coefficient on orbital lifetime prediction for the initial orbit.

It can be observed that the orbital lifetime decreases as a function of the ballistic coefficient. In fact, the smaller the BC, the faster the satellites decays. This means that, besides the drag coefficient, the cross-sectional area has also a significant impact in orbital lifetime predictions.

In conclusion, assuming that MECSE mission relies on a velocity-vector stabilization attitude, low values of cross-sectional area are expected, which means high values of BC. This represents an issue given that the orbital lifetime will be longer in those cases. Considering the maximum value of BC, the worst-case scenario would be an orbital lifetime between 1.7 (CSSI) and 1.9 (ECSS) years, which is too much for MECSE mission. However, the cross-sectional area will not always be 0.01 m^2 because the spacecraft will experience pointing errors in the attitude. Therefore, one can consider 0.02 m^2 of cross-sectional area for a more precise worst-case scenario. Assuming the drag coefficient as 2.5, the orbital lifetime is expected to be between 0.8 (CSSI) and 1.08 (ECSS) years. These results are very positive for a worst-case scenario.

Nevertheless, these results are obtained for the assumption of the initial orbital parameters, which will be evaluated in the following section.

5.4.5 Sensitivity Study of Orbital Elements

The second step of the sensitivity analysis is to understand the impact of the orbital elements in the lifetime prediction. Circular LEOs are assumed, so only altitude and inclination are analyzed. Moreover, the satellite parameters are fixed.

As explained in the previous section, MECSE mission aims for a velocity-vector stabilization so it is reasonable to consider a value of cross-sectional area smaller than the mean one recommended by ISO [58]. Regarding the drag coefficient, some authors [25], [59] recommend to use higher values which was corroborated by the validation study already performed. Table 5. 9 lists the parameters selected for the study.

As MECSE case has a higher ballistic coefficient, the orbital lifetime would be higher than for ISO case. In this way, MECSE parameters would also serve as the worst-case scenario between both options (see Appendix B). Thus, if the orbital lifetime predicted for MECSE parameters comply with the mission requirements, which demand a fast satellite decay, the predictions for ISO would also comply.

Table 5. 9 – Comparison between MECSE parameters and the ones recommended by ISO standard.

Parameter	ISO	MECSE
Form Factor (FF)	3U	3U
Mass [kg]	4.0	4.0
A [m^2]	0.035	0.02
C_D	2.2	2.5
BC [kg/m2]	51.95	80.00
Epoch	N/A	01-Jan-2020

For the orbital altitude sensitivity study, the circular altitude was varied from 250 km to 500 km with a step of 25 km. Three modeling approaches were used and the study was performed for ISO and MECSE cases (see Appendix B). Using the MECSE Parameters, the orbital lifetime as a function of altitude is illustrated in Figure 5. 14.

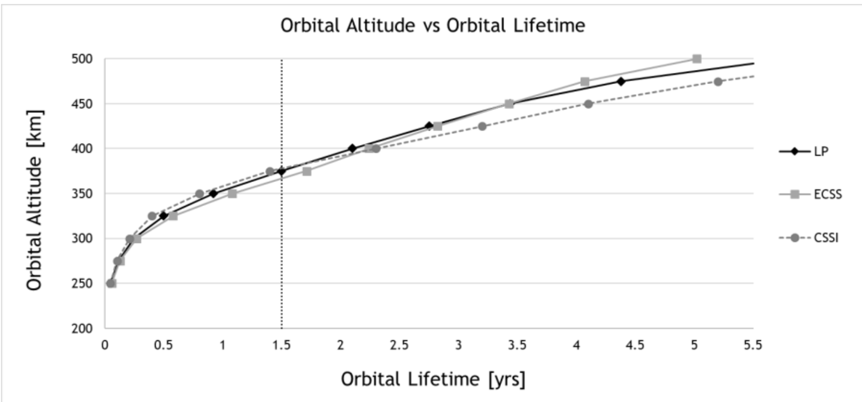


Figure 5. 14 - Effects of orbital altitude on orbital lifetime for 52.6° inclination circular orbit.

It is observed that the orbital lifetime does not increase linearly with the altitude and it increases more for higher altitudes, where the density of the atmosphere is lower. To understand the influence of the altitude on orbital predictions, one can consider two orbital altitudes: 375 and 400 km. Considering the first, the orbital lifetime for MECSE parameters and epoch is within 1.40 (CSSI) to 1.71 (ECSS) years. Considering the second, the orbital lifetime for the same assumptions is within 2.10 (LPN) to 2.30 (CSSI) years. By considering the most accurate model (CSSI), a difference of only 25 km of altitude leads to 10.8 months of difference.

Therefore, for altitudes above 375 km, the orbital lifetime exceeds 1.5 years which does not comply with the mission requirement (MR-10). Considering MECSE parameters and the epoch time, it is recommended not to accept launch opportunities above these altitudes. Therefore, 350-km orbit is a reasonable consideration.

Regarding the inclination sensitivity study, the inclinations were varied from 0º to 180º with an interval of 15º.Given that three modeling approaches were used again and the study was performed for the two cases, a total of 78 simulations were performed (Appendix A). For MECSE parameters, its effect on orbital lifetime predictions are shown in Figure 5. 15.

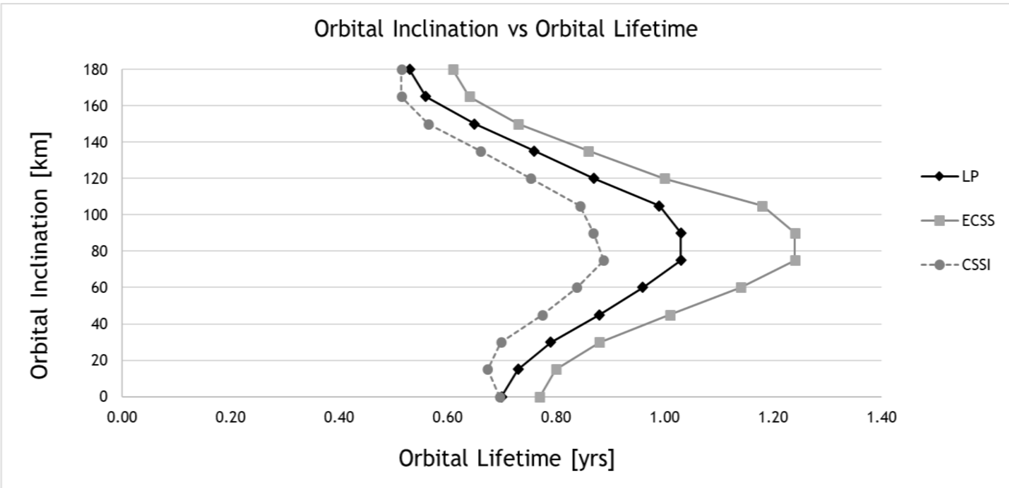


Figure 5. 15 - Effects of orbital inclination on orbital lifetime for 350 km circular orbit.

As it can be seen, the satellite lifetime increases significantly as the orbit plane inclination increases toward 90º. This difference can be up to 3 months in accordance with CSSI. From an orbital lifetime point of view, it would be benefic for MECSE to be inserted in non-inclined orbits, ideally prograde orbits below 20º or retrograde orbits above 140º inclination. However, assuming the MECSE parameters and epoch and considering the worst-case scenario of a 375-km polar orbit (90º of inclination), the orbital lifetime interval would be within 0.87 and 1.24, which are still positive results. Also, comparing this results with the ones estimated in the last section (for a 350-km orbit with 52.6º inclination) the difference is less than 1 month, which is almost negligible in this phase. Therefore, 52.6º inclination is a reasonable consideration.

In summary, a total of 144 simulations (Appendix B) were performed to evaluate the effects of orbital inclination and altitude on the orbital lifetime prediction. It was concluded that both elements have influence in the results, but the altitude of the orbit has the major impact. Also, considering MECSE parameters and the epoch time, the project must consider launch opportunities for low altitude orbits, ideally less than 375 km and non-inclined. This is considered feasible considering the launch survey from section 5.3.2.

Nevertheless, these results are obtained for the considerations of the epoch time, which will be evaluated in the following section

5.4.6 Sensitivity Study of Epoch

The last sensitivity study was performed concerning the epoch. This is a crucial parameter since it is directly connected with the level of solar and geomagnetic activity (see Figure 5. 7) which strongly affects the atmosphere density and, therefore, the atmospheric drag.

Because of the large density variations between a solar maximum and a solar minimum, satellites will decay far more rapidly during periods of solar maximum and more slowly during periods of solar minimum [25]. To better understand these variations, simulations for different epoch times are performed assuming the MECSE parameters and the reference orbit. The epoch is varied from 2020 to 2027 with one year of step. The results are shown in Figure 5. 16.

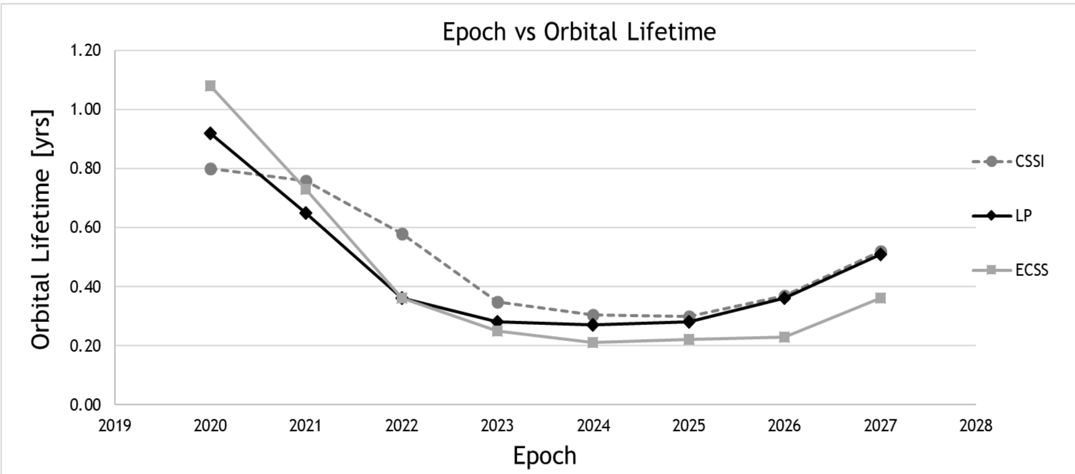


Figure 5. 16 - Effects of epoch on orbital lifetime for the initial reference orbit and MECSE parameters.

Looking at Figure 5. 16, one can notice the big difference between the reference orbital lifetime, at 2020, and the remaining ones. In fact, from the data obtained, the 1-Jan-2020 epoch is the worst-case scenario, since it has the highest orbital lifetime prediction.

The results for this scenario are within 0.80 (CSSI) and 1.08 (ECSS) years, whereas the results for the best-case scenario (1-Jan-2024) are within 0.21 (ECSS) and 0.27 (CSSI) years. Considering the most accurate model CSSI, the difference is up to 6 months. Therefore, from an orbital lifetime point of view, the project should reconsider the launch date. Launching during a solar maximum would be preferable.

The reason for the dissimilarities between different epoch and models can be understood by analyzing Figure 5. 17, where the solar activity cycles are represented for the three models. Firstly, it is observed that the solar maximum happens around 2024 for all the models. So, the satellite will have a low orbital lifetime in this period, which is confirmed by Figure 5. 16. Secondly, it is observed that each model uses a different solar flux. For instance, at 1-Jan-2024, ECSS uses the highest level of solar flux (approximately 170 sfu) compared with CSSI (approximately 130 sfu) and ECSS (approximately 150 sfu). Thus, the decay is faster when using the ECSS method.

This feature is responsible for the significant deviations in lifetime results that observed through all the analyses performed in this section. One can conclude that, each model uses different values of solar activity does providing different results in orbital predictions.

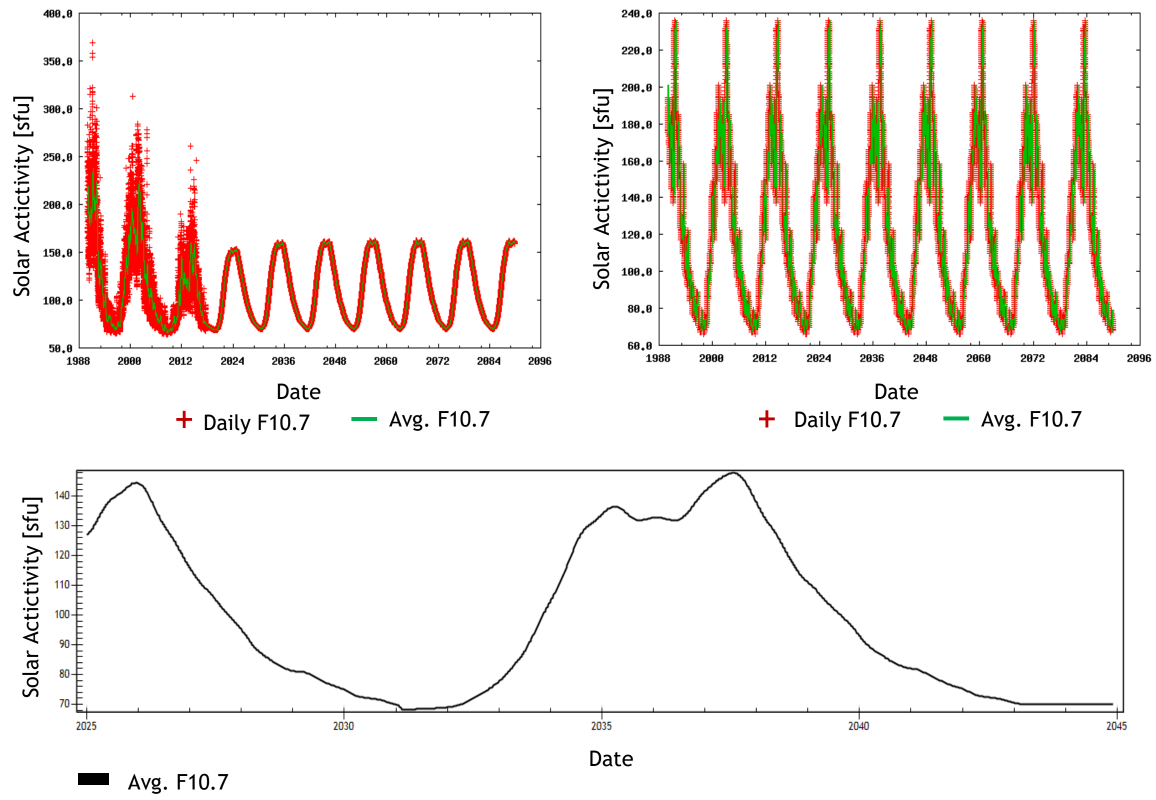


Figure 5. 17 – Solar activity by different models: LPN (top left); ECSS (top right); CSII (bottom).

In short, this study has allowed to evaluate the effects of epoch on the orbital lifetime, as well as to understand the reason why different solar and geomagnetic models provide different results from each other. It can be concluded that the time of the launch date will be of the most importance for the orbital analysis since it will define the level of solar activity. This will impact directly the atmospheric density and the orbital lifetime of the satellite. It has also been concluded that, the project would benefit from a delay in the launch day. For instance, being launched in the interval from 2022 to 2025 would help to have a faster decay.

5.4.7 The Lifetime of MECSE

Considering the initial reference orbit (see Table 5. 4), MECSE's parameters (see Table 5. 7) and a launch date on the 1st of January 2020, the lifetime of the nanosatellite can be predicted.

Figure 5. 18 shows the orbital decay of MECSE simulated at STK lifetime tool using the CSSI files. According to the simulation, MECSE will decay for 293 days and will start the re-entry on 20th October 2020. This will be the lifetime used for the remaining analysis, due to the fact that they can only be performed in STK. The lifetimes computed in DRAMA using ECSS and LPN models are shown in Figure 5. 19, respectively.

Observe that the time in an orbit below 250 km will be around one month and a half. This period of time shall be saved as a reference because it will be nearly constant for a triple CubeSat independently of the set of parameters used. This is because the solar activity does not have a big influence for altitudes lower than 200 km [25].

In summary, the orbital lifetime of MECSE has been computed for the three different modeling methods recommended by the standards. The large deviations on the decay date are evident. It is critical to note that some considerations were necessary and the analyses must be performed again in a future phase of the project were the parameters are more defined.

To conclude, for the analysis performed with the considerations already referred, the orbital lifetime prediction is within the interval of 0.80 (CSSI) and 1.08 (ECSS) years. These results are very positive given that the mission requirements were fulfilled. Also, recall that these results are obtained for worst-case scenarios for both the MECSE parameters (high ballistic coefficient) and the epoch (during a solar minimum), which means that the orbital lifetime would probably be even lower than the predicted ones.

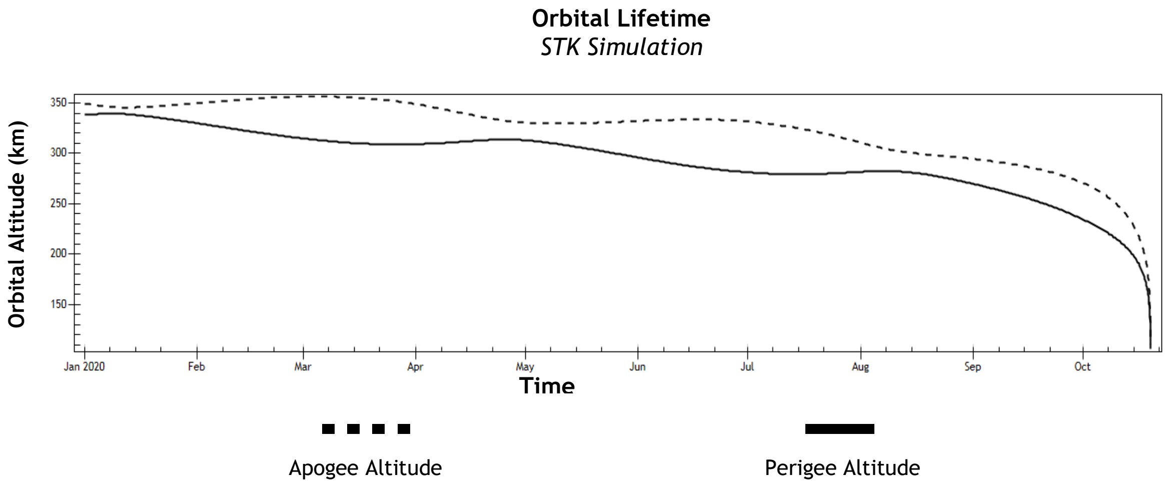


Figure 5. 18 - The orbital lifetime of MECSE Nanosatellite by STK with CSSI.

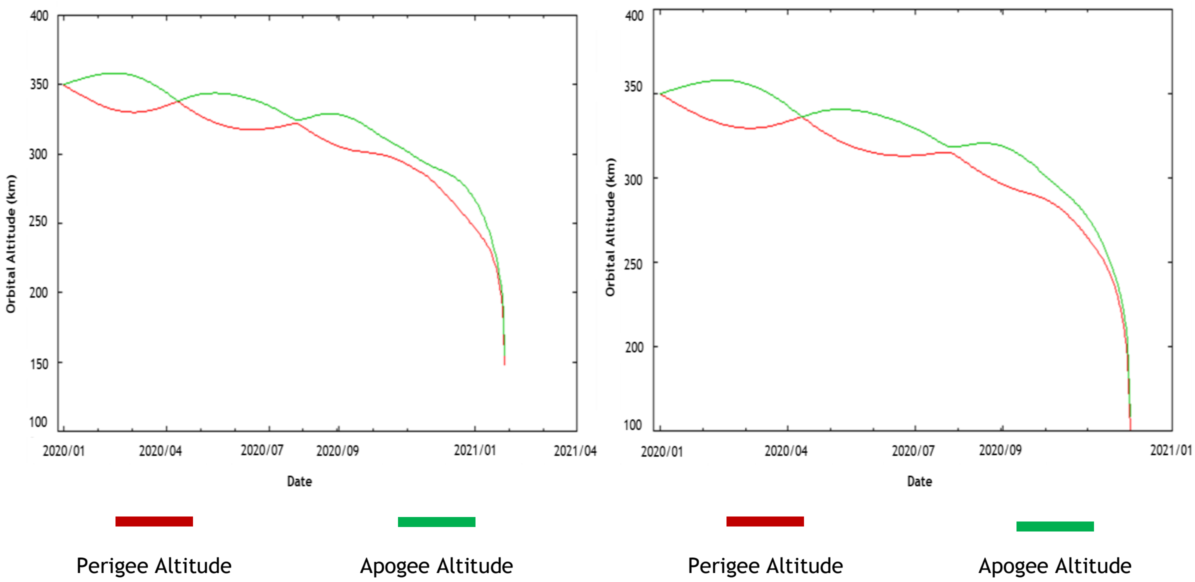


Figure 5. 19 - Orbital lifetime by DRAMA with ECSS (on the left) and LPN (on the right).

5.5 Communication

5.5.1 Access Time

Access times are another important parameter to be evaluated within mission analysis since they drive the communication system definition. STK allows the user to determine the times periods which the spacecraft can access the ground segment.

As MECSE project does not have a selected operational Ground Station (GS) neither has a ground station contract established yet, the Santa Maria ground station in Azores was chosen for the access time analysis. This assumption was made based only on the fact that Santa Maria Station is Portuguese and is part of the ESA cooperative network which means that the data about the station is accessible to the public [65]. Some information can be found in Table 5. 10

Table 5. 10 - Information about Santa Maria Ground Station in Azores.

Santa Maria GS – Azores	
Longitude	25º 08' 08.60'' W
Latitude	36º 59' 50.10'' N
Altitude	275 m

In addition, one must consider a minimum elevation angle of the satellite with respect to the ground station in order to take into account possible obstacles around the ground station. This constraint was set to be 10º above the horizon.

Figure 5. 20 shows all the communication periods between the satellite and the ground station during the mission lifetime. The access time are represented by segments with different thicknesses depending on their duration.

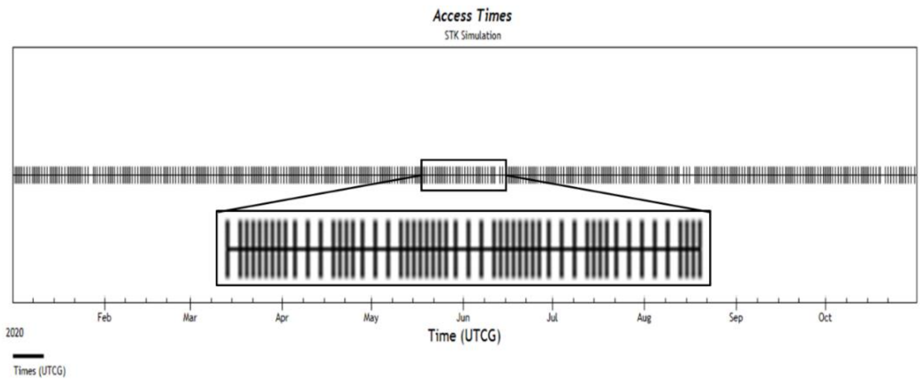


Figure 5. 20 – Ground station access times during the mission lifetime with the zoom for a small period.

Furthermore, the duration of the accesses was computed and the results are presented in Table 5. 11. Also, it was observed that there are at least 3 accesses per day which ensure a daily communication between Santa Maria ground station and MECSE nanosatellite.

Table 5. 11 - Access global statistics.

Ground Station Access	Duration [s]
Minimum	5.32
Maximum	384.84
Mean	271.04
Total	326330.83

Since the mean duration is about 4.52 minutes, the satellite will have a communication period of 13.55 min/day on the average. This is expected to be more than enough to transmit the mission data on a regular basis, therefore meeting the requirement MR-07. Besides, one of the advantages of LEO is that it allows higher data rates because of the short communication distances involved.

5.5.2 Mission Data

Concerning the data generated during the mission that will be communicated to and from the ground station, there are two distinct types: scientific data and housekeeping data [24].

Scientific data is the data gathered by the payloads and necessary to meet the scientific objectives (see the mission subjects in Table 3. 3). It was decided that all the mission raw data is important and shall be transmitted to the ground station. Then, it will be processed by the end user on the ground. This decision of not processing the data in space also simplifies the space system and reduces the associated cost.

On the other hand, the housekeeping data is the information used to support the mission itself which indicates the status and condition of all spacecraft subsystems. The mission operation center uses this data to monitor and control the satellite.

To summarize, both the two types of data must be transmitted during the access times with the ground station. Considering Santa-Maria GS, the results are very positive and it seems feasible to transmit all the mission data on a regular basis.

5.6 Eclipse Time

Sunlight periods are of the most importance for power and thermal subsystems' design because they are the satellite source of energy and temperature. Although, there are some periods of eclipses where the satellite will not be illuminated. Figure 5. 21 shows the two types of eclipses: penumbra and umbra.

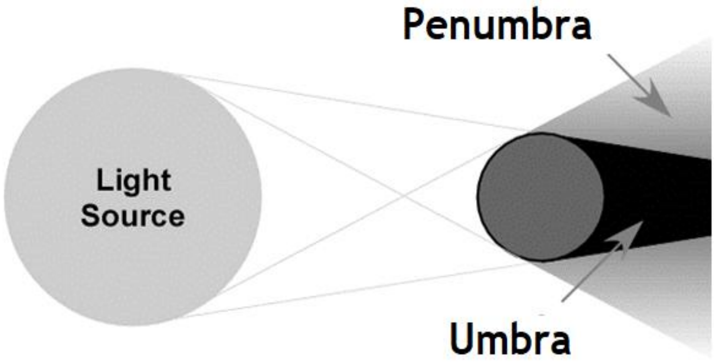


Figure 5. 21 - Scheme of umbra and penumbra eclipses.

The sunlight and eclipse durations were computed in STK for the mission lifetime. The results in percentage are presented in Figure 5. 22. As MECSE will be inserted in a LEO, high frequency of eclipses are expected due to the short orbital period. Indeed, MECSE will perform around 15.8 revolutions per day with a period of 1.52 h each (calculated for the initial reference orbit). During this time MECSE will be 64.4% in sunlight and 35.4% in eclipse.

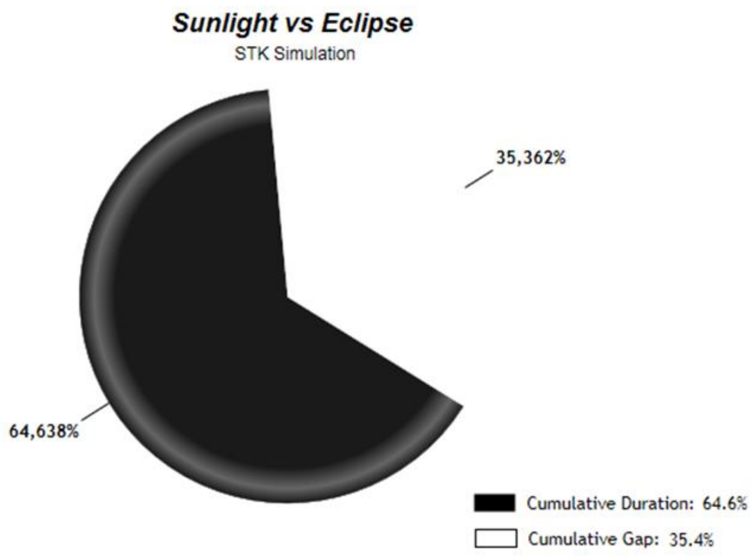


Figure 5. 22 - Percentage of sunlight and eclipse times for the mission lifetime.

The minimum, maximum and total duration of umbra periods are shown in Table 5. 12. Due to the low altitude of MECSE nanosatellite, penumbra times are negligible since they represent less than 1% of the total eclipse time. The average duration of an umbra is 32.91 min. This means that during this period the spacecraft cannot rely on solar panels for the generation of energy. Also, it must guarantee that the temperature of the equipment will not drop below the operational range.

Table 5. 12 - Global statistics of umbra times.

Umbra Time	Duration [min]
Minimum	2.82
Maximum	36.51
Mean	32.91
Total	5438.80

The time of direct sunlight can also be deduced directly from the Beta angle (β) which is defined as the smaller angle between the spacecraft's orbital plane and the Sun vector. The variation of the beta angle is presented in Figure 5. 23. This angle tells exactly from which direction the Sun is shining from which is particularly helpful for thermal and power subsystems because it allows to deduce the number of faces illuminated at specific times and durations. The simulations have also shown that there is a period of few days (at the end of July) that the satellite is free of eclipse, which means that during that period the satellite receives constant light all the time. This happens because the orbital plane is approximately perpendicular to the sunlight direction.

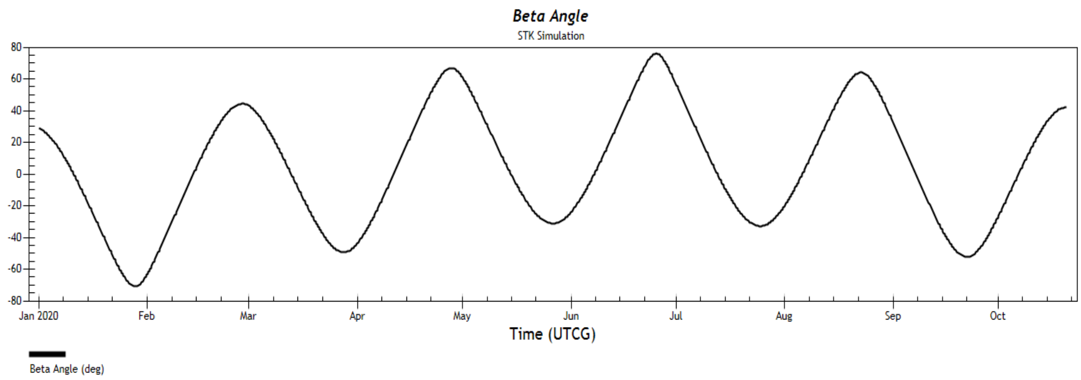


Figure 5. 23 - Variation of beta angle during the mission lifetime.

In short, the eclipse times were analyzed and the results will serve as input for power subsystem design. These results are not expected to vary much with future analyses.

5.7 Concluding Remarks

In this chapter, the mission analysis of MECSE project been performed which has allowed to design the satellite trajectory as well as to examine several parameters that affect the system's design such as eclipse time, access time, and orbital lifetime.

In summary, the motion of a spacecraft in orbit is disturbed by diverse perturbation effects that need to be considered through time. These effects can be modeled and used as data inputs by an orbit propagator which will simulate the motion of the satellite in orbit over a certain period of time. There are several propagators and models available but they offer different levels of accuracy. Thus, to be able to perform reliable orbital analyses, it was necessary to first select the correct tools used for simulation. This was performed through a literature research. It was found that, when predicting orbital lifetime, solar and geomagnetic activity modeling is particularly associated with high uncertainties and different standards do not agree on the modeling approach to be used. In fact, it has already been shown in past studies, such as [60], that the use of different models in DRAMA result in significant deviations in orbital lifetime predictions. No study was found that compared the CSSI models used by STK with the ECSS and LPN models used by DRAMA.

Special attention was paid to the orbital lifetime research given the facts that it is an important figure of merit and it was one of the main objectives of this thesis. However, the lifetime is remarkably difficult to predict since it will depend on detailed design parameters, the details of the actual orbit and the launch date relative to the solar cycle. All of these parameters are still unknown at a conceptual phase. Therefore, the lifetime of the satellite could not be predicted without making some considerations. If it is considered a 350-km initial orbit with 52.6° inclination and a BC of 80.00 kg/m^2, the MECSE lifetime is expected to be around 0.8 years as predicted by CSSI. This complies with the mission requirements.

Nevertheless, as this result is still associated with uncertainties given by the assumptions, it was possible to identify the set of characteristics that can influence the lifetime of MECSE by understanding the physics behind orbital decay predictions of a Cubesat. In short, satellites decay very little during solar minimum and then rapidly during solar maximum. The decay also depends on the BC. With a low BC, the CubeSat will respond quickly to the atmosphere and decay promptly. Finally, the decay date predicted varies depending on the method used to model the solar and geomagnetic activity.

In this view, different methods were analyzed. In STK software, CSSI files are used together with a numerical propagator (HPOP). Whereas, in DRAMA software, LPN and ECSS are used as inputs by a semi-analytic propagator (FOCUS-1A). The models for atmospheric density, geopotential and third-body perturbations are identical for the two software analyses and remained the same for all the simulations performed.

The results have shown that there are significant deviations between them. Even though that ISO [61] and the ECSS [62] tried to standardize the process of estimating the lifetime, the parameters and models proposed by the standards lead to very different results. It seems prudent to stablish just one standard model. The models were also compared against observed orbital lifetimes which allowed to conclude that the satellite parameters recommended by ISO and ECSS are not suitable when simulating CubeSat lifetimes. Therefore, it is of the most important to estimate parameters and analyze carefully the results by performing sensitivity studies and refine them as the project progresses. Additionally, the validation study also shows that the CSSI modeling in STK presents the most accurate and consistent results. However, this may be because it uses a more accurate orbit propagation method than DRAMA. Therefore, it can be concluded that the user should use this methodology when predicting orbital lifetime for a CubeSat.

More important, it was possible to predict a range of possible orbits for MECSE project considering the mission requirements. It is known that, due to the mission requirements, MECSE nanosatellite needs to achieve low orbits in about a year. If we consider the launch date at 2020 as targeted by the project management, the orbital altitude shall be constraint to a maximum of 375 km. This does not represent an issue because the launched survey performed has shown that there are several launch opportunities able to meet this constraint. In any case, a system for fast deorbit may as well be considered such as dart configuration deployables. This system can increase the cross-sectional area of the satellite lowering the ballistic coefficient. Otherwise, if the project delays, MECSE can extend the launch date and benefit from a solar maximum period, which allows the satellite to go into higher orbital altitudes.

To sum up, regarding the orbital lifetime analysis for MECSE satellite, the study is considered a success. The impact of several parameters that influence the orbital lifetime was investigated. The tools used were validated against observed orbital lifetime scenarios showing good accuracy, particularly the CSSI model used with STK software. The parameters estimated provided positive results even for the worst-case scenarios which would allow to widen the range of launch opportunities when more accurate parameters are analyzed in the future phases.

Chapter 6

6 System Design

This chapter purposes a preliminary study of the whole spacecraft using a system thinking approach. This means that different options for the design will be explored. However, the studies are not detailed to values and do not focus on the selection of the hardware. The aim is to understand the design drivers of each subsystem and to define requirements to be followed in the latter phases. In the end, a summary of each system development state is presented and future tasks for the next stages are proposed.

6.1 System Architecture

6.1.1 System Breakdown

Decomposition is a crucial step when designing a space system. By partitioning the space system, the work can then take place in parallel with the systems engineer ensuring that each subsystem lead stays connected with the overall system concept [24].

In this way, the interfaces between subsystems are minimized and the design complexity is reduced. MECSE system is broken down in the following hierarchic order: system, segment, module, subsystem, part and component.

The product breakdown structure of MECSE is presented in Figure 6. 1 from the whole system to part level. The work presented in this chapter will focus on the space segment. Therefore, before moving on to the design of the subsystems, the concept of operations of the spacecraft must be defined.

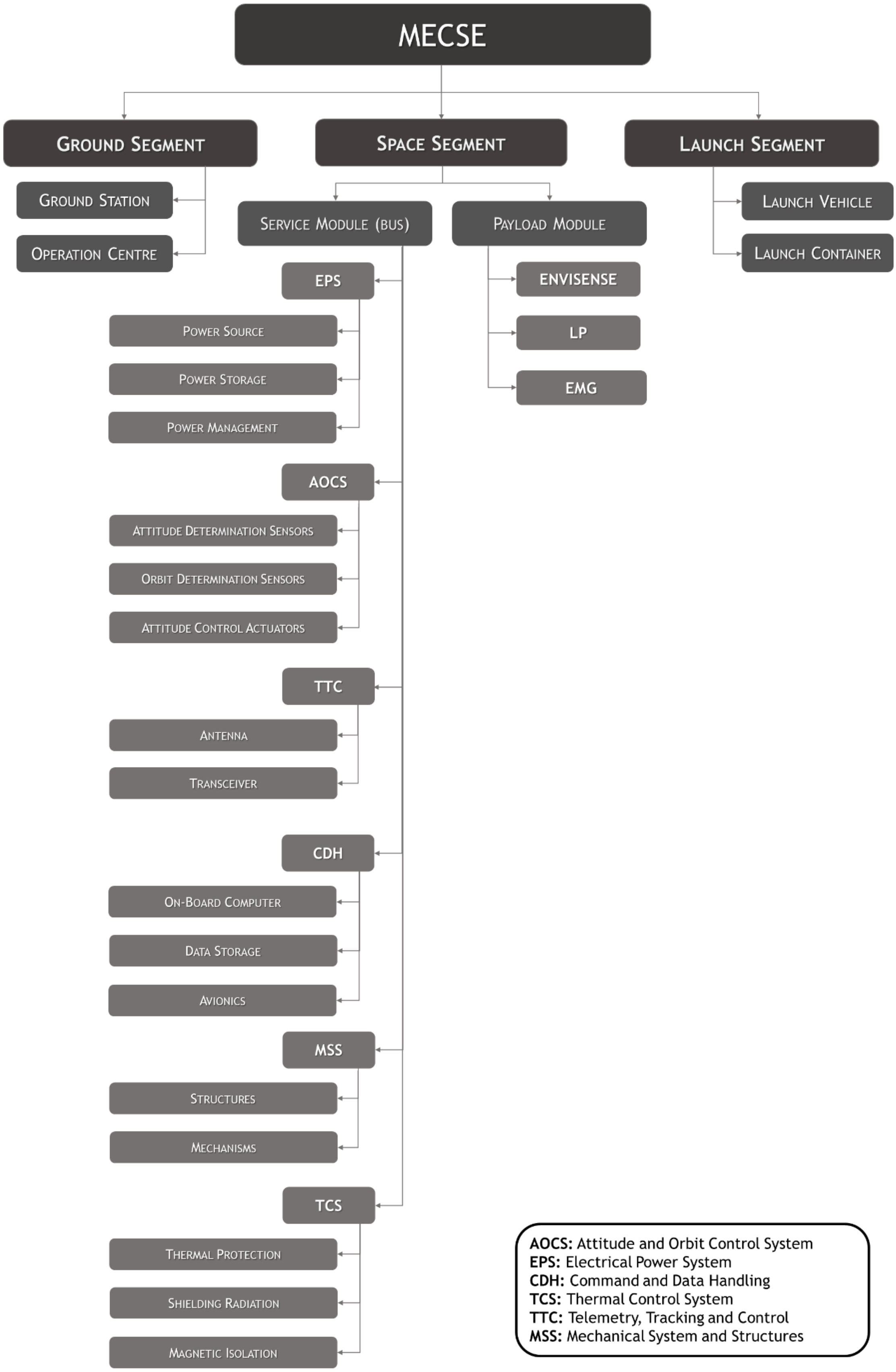


Figure 6. 1 - Product breakdown structure of MECSE.

6.1.2 Concept of Operations

The concept of operations defines how the satellite will operate while in orbit. We aim to be able of perform the scientific studies as well as to transmit the data to the ground. Four modes of operation were identified: safe, downlink, PDS and PLME.

Safe mode is only used for essential functions. The spacecraft needs only to be able of keeping its orientation and collecting data about the subsystem status. The TTC is on a standby mode, meaning that the spacecraft awaits commands from its mission control center. The safe mode is the nominal state of the spacecraft. On the other hand, the downlink mode uses all the subsystems of the safe mode plus the TTC for the downlink of scientific data. The scientific modes, PDSm and PLMEm, will use the payloads required to complete the scientific study. Table 6. 1 details the subsystems used during each operation mode.

Table 6. 1 - Subsystems switched on during each operation mode.

SaFe Mode (SFm)	DownLink Mode (DLm)	PDS Mode (PDSm)	PLME Mode (PLMEm)
EPS	SFm	SFm	PDSm
CDH	TTC	ENVISENSE – PL01	EMG- PL03
AOCS		LPN -PL02	
TTC (standby)			

The scientific modes were also studied in more detail since they represent the main challenge of the mission. A deep examination was necessary to come up with a novel concept capable of achieving the scientific objectives while being technologically feasible within a CubeSat. This means that the mass and power of the subsystem must minimize. The idea consists on using a time-varying magnetic field instead of a constant magnetic field (too heavy). It makes use of a pulsed current flowing through an insulated conductor surrounded by plasma which will generate a variable magnetic field magnetizing solely the electrons. These electrons are expelled from the field decreasing the electron number density of the plasma layer. By measuring the plasma layer density before, during and after the generation of the electromagnetic field, the system will be able to determine the electron density reduction (EDR) and prove that the plasma layer can be manipulated. Also, by using short pulses of current from time to time, the overall power consumption is decreased and the system has time to recharge for the next experiment. The concept is illustrated in Figure 6. 2.

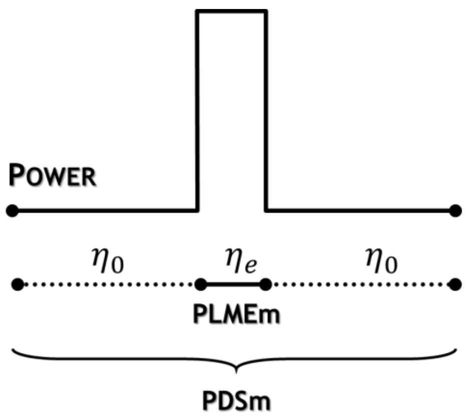


Figure 6. 2 – Concept of operations for the scientific studies.

6.1.3 Conceptual Design

Having defined the concept of operations, the space segment can now start to be designed. The first step is to visualize the spacecraft as an entire system. For this purpose, a preliminary configuration is presented in Figure 6. 3. The design is divided in two modules. The payload module is allocated on the first unit (on the top) and the bus occupies the remaining space.

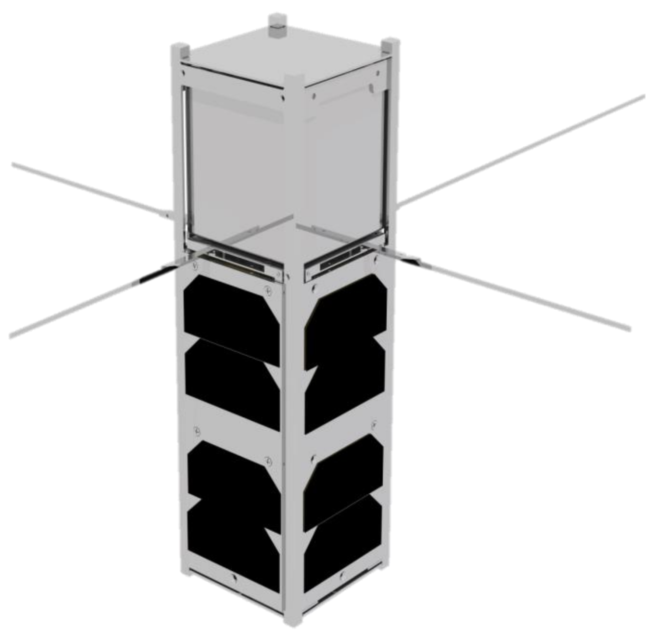


Figure 6. 3 – Conceptual design proposed for MECSE nanosatellite.

Figure 6. 3 presents a first proposal design of the system which is going to be evaluated and detailed in the next sections.

6.2 Payload Module

The design of the satellite will start with the discussion of the payload module. MECSE has three scientific payloads able of fostering the research on plasma layer manipulation in low Ionosphere through electromagnetic control. For this purpose, it is required to collect data about the environment and the plasma layer with and without the electromagnetic control and determine the EDR. The payload requirements are presented in Table 6. 2.

Table 6. 2 – Payload module requirements.

# ID	Payload Requirements	Rationale
PL-01	The payload shall fit on a 1U.	Design Constraint
PL-02	The S/C shall be able to collect environmental data.	MR-02, MR-03
PL-03	The LP shall be able to measure the density of the plasma layer before, during and after the operation of the EMG.	MR-03, MR-06
PL-04	The EMG shall have switch on/off capability.	Functional
PL-05	An electromagnetic field shall be generated with a magnetic field intensity of 0.0375 T [TBC] at the LP position [TBD].	EMG Design Assumptions

6.2.1 Environmental Sensors - ENVISENSE (PL01)

The first payload consists of environmental sensors to measure atmospheric conditions that could affect the plasma layer formation in low Ionosphere. The parameters identified to be helpful were: atmospheric temperature, pressure and density, and solar irradiance.

Concerning the temperature, the main consideration for choosing the sensor was the range of temperatures measurable by the sensor. Temperatures in orbit can reach extreme highs and lows during the sunlight and eclipses respectively. Type K thermocouples [66], already used in QARMAN mission [36], were identified as a suitable choice for measuring temperature in low Ionosphere. However, it was recently found that thermocouples are very susceptible to the effects of electromagnetic fields [67], so the use of these sensors is not advised.

Atmospheric pressure decreases as altitude increases. Therefore, in the upper levels of the atmosphere, air pressure would be much lower than closer to the surface. The pressure sensor selected for MECSE must be able to measure pressures close to 0 kPa. The NPC-1220 [68], manufactured by Amphenol, was used in QARMAN mission [36], and could be a suitable choice.

No solution could be found which directly measures air density. However, it would be possible to estimate it using temperature and pressure measurements or through the altitude.

Solar irradiance is the measure of the sunlight power contacting a particular area in W/m^2 (SI units). The device typically employed to measure this quantity is a spectrometer such as the AvaSpec-Mini manufactured by Avantes [69], which was designed originally to fit within 1U. However, it has high mass (174g) and power consumption (3.75W). Another option would be to use the solar arrays to also measure solar irradiance. This could be done by tracking the amount of power received by area of solar array. Further studies are required on this topic to understand the feasibility of measuring solar irradiance in such way.

In conclusion, the scientific prerequisite of measuring all the identified parameters should be reviewed. Firstly, because they are not critical for the fulfillment of the mission objectives which are focused on the plasma layer. Secondly, because all the parameters can be estimated with reasonable accuracy using indirect techniques. For instance, by determining the altitude, the temperature, pressure and air density can be deduced using atmospheric models. Also, there is the possibility of estimating solar irradiance with solar arrays.

6.2.2 Langmuir Probes – LP (PL02)

The second payload, LP, is meant to measure the electron density of the plasma layer surrounding the CubeSat and determining how it changes with the generation of an electromagnetic field. For this purpose, it is required to measure the EDR (see section 2.2.5). The device able of measuring this parametr is called Langmuir probe.

Several types of Langmuir probes were analyzed [34], [35]. It was concluded that, in previous missions, these probes are always attached to long deployable booms. This is done to reduce the errors in measurements caused by the spacecraft floating potential, which is the voltage on the surface of the CubeSat as it moves through plasma. For MECSE case, this is an issue to be solved since it is required that the sensors are placed the closer possible to the CubeSat surface because their location will affect the design of the EMG (see section 6.2.3).

The mNLP was the selected instrument. It is a technology recently developed at the University of Oslo [34] and it was successfully used in several QB50 missions [46] and sounding rockets [70], [71], so it has already flight heritage.

This new Langmuir probe concept was invented for the in-situ investigation and has the capability of measuring absolute electron density at a sufficient resolution to resolve the finest conceivable structure in an ionospheric plasma [70]. In fact, it has already proven to be able to measure structures down to the scale of one electron gyro radius [71]. Thus, it provides high-quality measurements of electron density at any desired resolution [70].

The mNLP consists of four cylindrical probes set at a positive fixed-bias. Each probe is maintained at a constant voltage different from the other three. Note that each of these voltages must be greater than the spacecraft's floating potential. When inserted into a plasma, current flows through each probe. There is a linear relationship between the square of this current (I_c) and the probe voltage (V), as illustrated in Figure 6. 4.

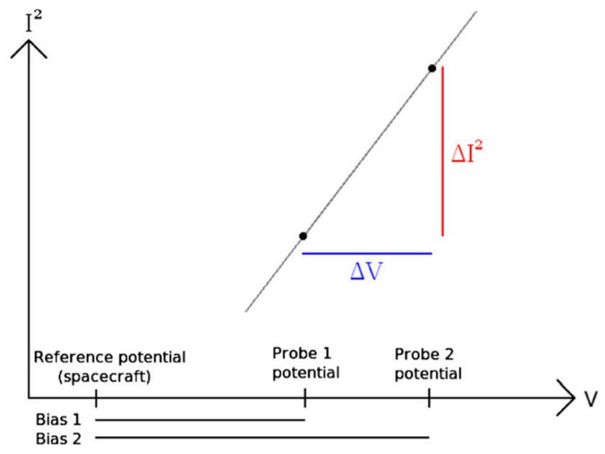


Figure 6. 4 - Example of measurements by two fixed-bias probes (from [70]).

From two measurements given by two probes, one can deduce the electron density, η_e, by [70]:

$$\eta_e = \sqrt{\frac{m_e}{2q(q2rl)^2}\frac{\Delta(I_c^2)}{\Delta V}} \tag{6.1}$$

where m_e is the mass of an electron, q is the charge of an electron, r is the radius of each probe, and l is the length of each probe. While only two probes are needed for the linear relationship in the equation, four are required to ensure accuracy and for redundancy. Furthermore, using this setup, the value of the reference potential becomes irrelevant as long as the probes potential are sufficiently above the plasma potential. The only concern is the difference in potential between the probes [70].

In conclusion, mNLP seems to be the most suitable instrument for highly-quality ionospheric plasma measurements. The issue is that the spacecraft floating potential may affect the bias potential of closer probes, skewing measurements. However, having the probes biased to a high enough voltage relative to the CubeSat floating potential should be sufficient to mitigate this effect. For future work, experiments must be conducted to confirm this hypothesis.

6.2.3 Electromagnetic Field Generator - EMG (PL03)

The third payload, EMG, is the MHD/EHD control device used to generate the electromagnetic field. Given that no technology with the required specifications was found until now, it was decided that the EMG should be developed in-house.

Table 6. 3 -EMG design drivers for MECSE.

Design Drivers	Driven By	Impact
Magnetic Field Intensity	Numerical Simulations	Power, Mass
Location of the Sensor	LP, Numerical Simulations	Power, Mass
Type of Materials	Magnetic Field Intensity	Power, Mass

The EMG is an electromagnet which is a device capable of generating a uniform magnetic field by circulating electric current in a solenoid coil (see Figure 6. 5). A current through the wire creates a magnetic field which is highly intense in the interior and weak in the exterior. By using a core material with magnetic properties, the magnetic field intensity can be increased.

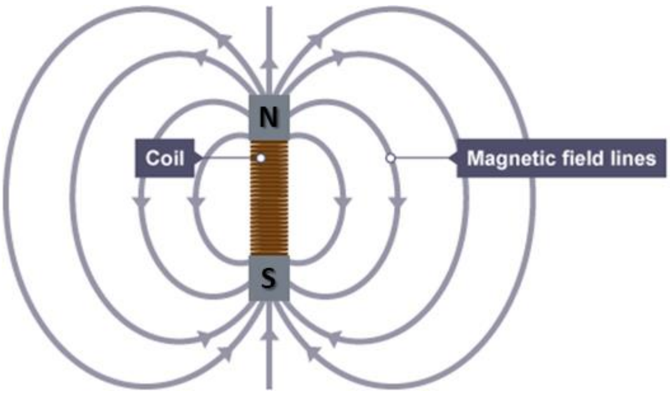


Figure 6. 5 - Electromagnet composed by a solenoid coil and magnetic core (adapted from [72]).

In practice, an electromagnet is a conductor wire of finite length rolled helically with a determined number of turns. The magnetic field intensity, B, in the interior is given by:

$$B = \frac{\mu N I}{L_{coil}} \tag{6.2}$$

where μ is the magnetic permeability of the core material, N the number of turns, I the electric current and L_{coil} the coil length. As shown, an electromagnet requires a continuous supply of current to maintain the magnetic field. Also, if the volume is a constraint (PL-01 requirement), N and L will be constrained and the magnetic field intensity will depend mostly on the applied current which will impact the power consumption.

The main goal of the EMG is to generate a sufficient magnetic field intensity to reduce the plasma density in the location of the Langmuir probe. Thus, to start the EMG's preliminary design, some assumptions were required. Firstly, a magnetic field intensity value of 0.0375 T at the LP position was assumed as a requirement. This assumption was based on the statement that 0.0357 T is already sufficient to modify the plasma characteristics within re-entry conditions (see section 2.2.4) plus 5% of safety margin. Secondly, it was assumed that the mNLP could be used without booms. Given that each probe has 25 mm length, the distance from the center of the core material to the probe was defined as 75 mm. Thirdly, the electromagnet radius was constrained to a maximum of 90 mm to be able to fit into a 1U and the mass was constrained to a maximum of 1.2 kg. Finally, a horizontal dipole configuration was chosen based on the studies performed in [14]. The schematics are presented in Figure 6. 6.

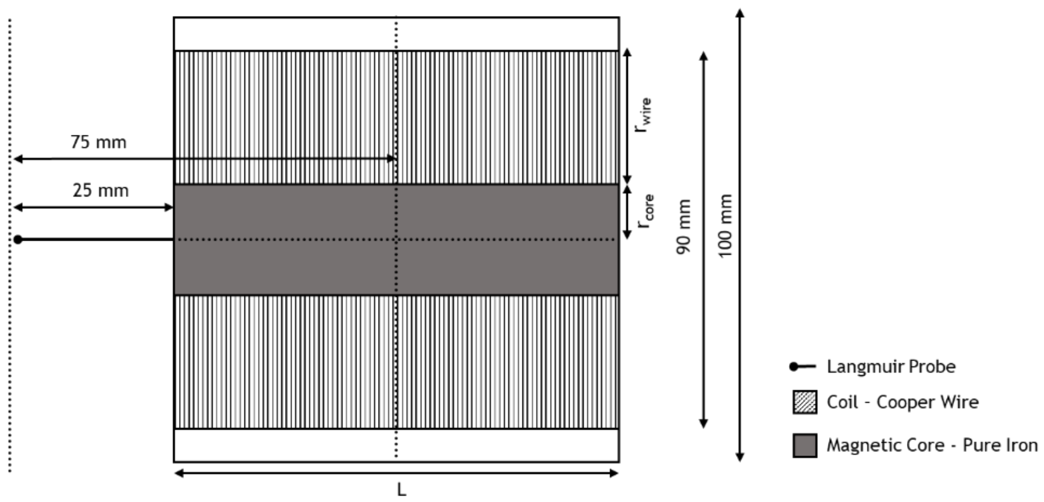


Figure 6. 6 – Schematics of the EMG setup together with the LP.

Several preliminary analyses were performed to understand the impact of different designs choices on the subsystem power consumption and mass, as well as to evaluate the magnetic field decay with the distance from the center. This included simulations using Finite Element Method Magnetics (FEMM) which is a finite element package for solving 2D planar and axisymmetric problems in low frequency magnetics and electrostatics. For the simulation, a simple configuration of horizontal dipole was tested assuming that the wire surrounds the total length of the core. Pure iron was chosen for the core material due to its high magnetic permeability and typical usage, but the radius of the core was varied. For the solenoid coil, cooper wire was selected but different AWG (American Wire Gauge) wires were evaluated. The diameter, resistance and maximum current varies between them. Different lengths (L) of the electromagnet were also studied.

From this study, it was possible to create a database regarding the mass, power requirements and magnetic field intensity for different positions relative to the center of the magnetic core. The results were successful and the requirements were able to be fulfilled. Although, some rough assumptions were made and further work is required on this topic. Firstly, the numerical studies shall be validated with experimental data. Different configurations and materials shall be analyzed to optimize power consumption and mass. Also, 3D simulations shall be performed.

Having defined the payload, the design will move forward to the service module (BUS).

6.3 Service Module (Bus)

This section intents to introduce the BUS subsystems and discuss design drivers and requirements. The requirements serve as guidelines for the subsystems design. Thus, they can be subjected to modificationws as the project advances if the systems engineer approves it.

6.3.1 Electrical Power Subsystem (EPS)

Power is a crucial figure of merit due to the high-power requirements for the PLME and it can be decisive for the mission progress. Thus, special attention was paid to the design of the EPS. The EPS needs to provide all the power generation, storage and distribution for the spacecraft [22], [25]. The power design drivers for MECSE are presented in Table 6. 4.

Table 6. 4 - Power subsystem design drivers for MECSE.

Design Drivers	Driven By	Impact
Power Consumption	Payload, EMG Peak Power	Solar Array, Power Storage
Power Distribution	EMG Peak Power	Power Management Board
Bus Voltage	S/C Design	LP Bias, Power Electronics
Payload Duty Cycle	Modes of Operation	Solar Array, Power Storage

It can be concluded that power consumption and power distribution are the main challenges for the design of the space segment. This is because the Power Management Board (PMB) needs to be able to handle a huge peak flow and to deliver the power to the EMG during PLMEm. Also, the components used to store the power need to be able to supply a huge amount of power.

Given that power is directly dependent on the EMG design, which is being developed in parallel, it is a parameter highly subjected to iteration. Therefore, the research focused on the development of a power budget tool which allowed to be updated. The power budget is needed to ensure that there is enough power to sustain all the operations for the lifetime of the mission. Conclusions on components’ choices can be made using the spreadsheet through the design process and can be updated as other subsystems change.

Regarding the design, it must follow the power requirements presented in Table 6. 5. The ones marked with TBC or TBD require further analyses. The values are provided based on former space missions and literature review.

Table 6. 5 – Power system requirements.

# ID	Power Requirements	Rationale
EPS-01	The EPS shall provide sufficient power at the appropriate voltage, to meet the power requirements of all subsystems in all modes of operation.	Functional
EPS-02	The EPS shall be able to supply at least 140 W [TBC] of peak power.	PLME, Assumption
EPS-03	The EPS shall be designed for a typical LEO eclipse fraction of 35 %.	Piggy-back to LEO
EPS-04	The S/C shall be able to perform the PLME at least twice per orbit [TBC].	MR-07
EPS-05	The PLME shall have the duration of 1 second [TBC] per experiment.	MR-07 PLME Duty Cycle
EPS-06	The EPS shall be able to support at least 4 [TBC] PLME without recharging.	Assumption
EPS-07	The PDS shall be switched on at least 10 [TBD] seconds before the PLME and switched off at least 10 [TBD] seconds after it.	Operation Modes Assumption
EPS-08	The downlink mode must be operational for a maximum of 15% of the orbit.	GS Access Time

Concerning the sizing, solar panel sizing (the power source) is achieved by using the average power used during nominal activity periods, whereas the power storage is usually sized by using the average peak power requirements. Although, it shall also be able to provide the absolute peak power, which is the main issue for MECSE nanosatellite given the high demand of the EMG.

To solve this issue, two strategies were adopted: minimize the duty cycle of the operation modes and explore alternative options for high power storage. For the cycling of the CubeSat, the four modes are cycled through to provide time for data collection, data transmission and idle periods used to ensure that stability is maintained. An example of cycling during the sunlight of the reference orbit can be seen in Figure 6. 7, where the four operation modes are easily identified. The longest one represents the SFm which is followed by the DLm with the duration of the mean communication access time (see Table 5. 11). Succeeding, the PDSm and PLMEm operation modes are represented with the duration established in the power requirements (see Table 6. 5). As expected, the maximum power to be used for a given moment of time is during PLMEm when all systems are running and there is a burst of power from the EMG.

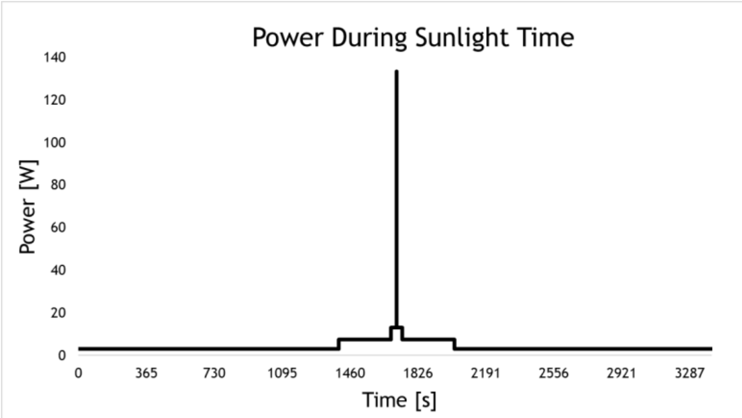


Figure 6. 7 - Power cycle example during the sunlight time of the orbit for the four operation modes.

For power storage, two options were analyzed. The first option is using the batteries for all the peak operation modes, whereas the second option is using supercapacitors to supply the PLMEm and batteries for the remaining modes. Supercapacitors [73] store energy quickly and then release that energy in bursts. They have an extremely high specific power and a low specific energy as seen in Figure 6. 8. Because of this, they are ideal for power burst operations such as turning on the EMG for one second.

Function	Supercapacitor	Lithium-ion (general)
Charge time	1–10 seconds	10–60 minutes
Cycle life	1 million or 30,000h	500 and higher
Cell voltage	2.3 to 2.75V	3.6V nominal
Specific energy (Wh/kg)	5 (typical)	120–240
Specific power (W/kg)	Up to 10,000	1,000–3,000
Cost per kWh	$10,000 (typical)	$250–$1,000 (large system)
Service life (industrial)	10-15 years	5 to 10 years
Charge temperature	–40 to 65°C (–40 to 149°F)	0 to 45°C (32°to 113°F)
Discharge temperature	–40 to 65°C (–40 to 149°F)	–20 to 60°C (–4 to 140°F)

Figure 6. 8 - Comparing supercapacitors and li-ion batteries (from [73]).

In conclusion and following preliminary analyses, supercapacitors seem to make it possible to use smaller batteries to deliver large amounts of power for a short period of time. This saves on mass and volume which may aid other systems that need more available space. Therefore, they are considered for now as the best option for power storage.

For future work, it is recommended that the supercapacitors are tested to determine the charge time and power consumption. Additionally, it is necessary to research for a PMB that can handle the high peak power that is needed by the EMG. Based on market research, this can be an issue and the team shall consider developing the PMB in-house. Lastly, it is critical to solidify the selection of the other subsystem components which will allow to update the power budget tool and correctly size the EPS.

6.3.2 Attitude and Orbit Control Subsystem (AOCS)

Attitude is another important figure of merit to be analyzed. The attitude of a spacecraft concerns its orientation and stabilization in space with respect to a given reference frame. The AOCS has the responsibility of determining and controlling the orientation of the spacecraft. Table 6. 6 shows the major drivers of the AOCS design for MECSE.

Table 6. 6 – Attitude determination and control design drivers for MECSE.

Design Drivers	Driven By	Impact
Pointing Control Accuracy	Payload Requirements	Sensors and Actuators
Pointing Knowledge	Payload Requirements	Sensors
S/C Momentum of Inertia	S/C Configuration	Actuators
External Torques	S/C Configuration, Orbit, PLME	Momentum storage

The first step to identify the most appropriate AOCS system is to quantify the disturbance torques acting on the satellite. In a conceptual design phase, they are difficult to estimate because they depend on several yet unknown parameters. However, it is possible to characterize the major environmental disturbances torques acting on the satellite in LEO, which are: gravity gradient torque, aerodynamic drag and magnetic field torque [25] [51].

For the first, it is known that any non-symmetrical object of finite dimensions in orbit is subjected to a gravitational torque because of the variation in the Earth's gravitational force around the object. As MECSE orbits in low altitudes, the aerodynamic drag becomes the largest disturbance. Lastly, the magnetic disturbance torque results from the interaction between the satellite's residual magnetic field and the geomagnetic field. For MECSE case, a strong electromagnetic field will be generated inside the spacecraft. It is expected that this electromagnetic field will interact with the geomagnetic field, creating an enormous disturbance torque. Notwithstanding, this torque can be reduced by turning on the experiment on equatorial regions where the magnetic field strength of the Earth is smaller or by trying to align the magnetic field produced by the EMG with the Earth's magnetic field. Further study on this topic is required as well as it is necessary to evaluate the impact of the experiment's duration on the created torque. Recall that, the EMG will only be turned on for a short period of time, which is expected to be less than 1 second.

Having characterized the disturbances, the requirements for the attitude subsystem were derived and are listed in Table 6. 7 - Attitude system requirements.Table 6. 7. Once again, the values are provided based on former space missions, particularly QARMAN [36].

Table 6. 7 - Attitude system requirements.

# ID	Attitude Requirements	Rationale
AOCS-01	The S/C shall be able to be de-tumbled and stabilized within few days [TBD].	Functional
AOCS-02	The AOCS shall be able to determine the orbit with 1 km [TBD] of accuracy.	Functional
AOCS-03	The S/C attitude shall be known within 2º [TBC] of uncertainty.	MR-05
AOCS-04	The long axis of the S/C shall be aligned with the velocity vector so as to point the payload in the direction of motion with 5º [TBC] of pointing error accuracy.	MR-05
AOCS-05	The PLME shall be turned on when the Earth's Magnetic Field is aligned with the S/C long axis to prevent magnetic torque disturbances.	Minimize Actuators
AOCS-06	The system shall keep track of the magnetic field torque generated by the EMG.	Functional

Before moving on for the selection of the AOCS components, it is to note that given the high magnetic noise caused by the payload, it was decided to reduce the use as much as possible of magnetic systems. This include to not rely on magnetic sensors neither try to use the magnetic field of the Earth for attitude control. By doing it, one is constraining the AOCS selection since these systems are frequently used in LEO due to its low mass and power requirements.

Concerning attitude control, it was concluded that pointing errors should be minimized toward the velocity vector. Thus, pitch and yaw angles (Figure 5. 6) should be constrained, but there is no restriction for roll angle which means that the spacecraft is free to rotate around the X_0 axis. To provide the orientation different attitude control techniques were studied and are summarized in Table 6. 8. They include three-axis stabilization, momentum-bias and aerodynamic stabilization.

Three-axis stabilization can be achieved [25] by using actuators such as reaction wheels or thrusters, one for each axis to be controlled. Reaction wheels are the best option here in terms of simplicity and cost. However due to parasitic external torques they need to be periodically desaturated (momentum management) using magnetorquers [25], [74]. From a market research, integrated units combining different components into a single package can be acquired for precise 3-axis control [22], [74]. For instance, Blue Canyon Technologies [75] offers an integrated package (XACT) with a stated spacecraft pointing accuracy of better than 0.007° for the 3 axes, which occupies a 0.5 U. However, these mechanisms have major drawbacks due to the power consumption of the system during actuation and have a big impact on the mass and volume budget. Also, it is expected that the aerodynamic disturbances due to the low altitudes will be too large to couteract with a reasonable sized system.

Momentum-bias systems often have just one wheel with its spin axis mounted along the pitch axis, normal to the orbital plane. The angular momentum of the wheel results in gyroscopic stiffness effect about the orthogonal body axes (roll/yaw). The gyroscopic stiffness contributes to maintaining pointing accuracy with respect to external disturbance inputs. However, it will also need to be desaturated from time to time.

On the other hand, aerodynamic stabilization is a passive control system based on the aerodynamic forces [36]. The idea is to use a dart configuration with deployable side panels Figure 2. 13 a)) such that the center of pressure is behind the center of mass. This will provide an aerodynamic restoring torque allowing passive pitch and yaw stabilization [76]. The system can then be coupled with a small momentum-biased pitch wheel offering yaw and roll stabilization [72], [73]. The concept has already been tested [33], [73] and it is said to provide less than 5 degrees of pointing accuracy.

Table 6. 8 – Comparing different attitude control techniques.

Attitude Orientation	Subsystem Selection	Main Advantages	Main Disadvantages
3-Axis Stabilization	**Reaction Wheels (3x)**	High Accuracy (<0.1°) COTS with Flight Heritage	Power, Mass, Volume Momentum Management
	Thrusters	No Momentum Management	Cost, Complexity, Fuel
Momentum Bias	**Momentum Wheel (1x)**	Simplicity	Pitch Accuracy Momentum Management
Aerodynamic Stabilization	**Dart Configuration**	Passive System Low Power Consumption Increase Solar Array Area Faster De-Orbit	Deployables Actuators for Deployment

From a conceptual analysis point of view, the best strategy seems to be the combination of aerodynamic stabilization with momentum bias. It allows to optimize the power budget while providing a proper pointing accuracy. Also, it can be used as a faster de-orbit device which would allow to widen the range of possible launch opportunities.

The main challenge here is to detumble the satellite during the LEOP phase and to perform momentum management. Magnetic torquers may be considered for both phases. However, a future analysis is needed to comprehend if it is possible to do it without interfering with the payload. One option is to only use the magnetic torquers for momentum management when the spacecraft is on safe mode.

For attitude determination, sensors must be selected based on the required orientation and required accuracy. For inertial pointing, sun sensors, horizon sensors, and gyroscopes can be used together to acquire the vehicle orientation [25], [74]. One combination could be to use six fine sun sensors from New Space Systems (0.1° accuracy [74]), one static horizon sensor from Aerospace Servo (0.25° accuracy [74]) to be used during eclipse times and MEMS gyros to provide tri-axial rotation measurements. Star sensor can also be used if there is the need for more accuracy (74 arcsec [74]) but they drive cost, mass and power. Moreover, a GPS (Global Position System) must be used to precisely determine the position of the spacecraft.

In conclusion, the aerodynamic stabilization combined with the momentum bias was the strategy selected for attitude control. It allows to perform a 3-axis stabilization without the need for high power consumptions. The combination of 6 fine sun sensors, 1 horizon sensor and 1 MEMS gyroscope is used for attitude determination as well as 1 GPS for orbit determination.

For future work, a careful study should be done to determine the final number of sensors by performing a trade between accuracy, mass and power requirements. Also, a deeper study about the attitude control is required, particularly to understand the magnitude of the aerodynamic external torques which will allow to size the wheel and to choose the angle of the deployable mechanism.

6.3.3 Telemetry, Tracking and Command (TTC)

The telemetry, tracking and command is the system responsible for the communications between the satellite and the ground station. For MECSE case, it provides the radio link allowing the spacecraft to downlink the mission data (scientific and housekeeping) to the operation center as well as to receive operator commands (telecommand) from the ground station [22]. Table 6. 9 lists the major drivers for the TTC design.

Table 6. 9 – Telemetry, tracking and command design drivers for MECSE.

Design Drivers	Driven By	Impact
Data Rate	Payload (ENVISENSE and LP), Modes of Operation	Hardware Choice
Radio Frequency Band	Data Rate, Access Time, GS	Hardware Choice

Given that the TTC will be a COTS, the design drivers will only impact the choice of the hardware. The first aspect needed is the data rate of the payload which is determined by the amount of data collected by the sensors during operation and the time it will be operating. Secondly, it is important to choose the frequency radio band to use for the Cubesat, which is based on availability as well as the ground station capability and the data rate required. If

Santa-Maria GS is considered, S-band needs to be selected for the frequency. However, as it was only used as an assumption for the computation of the access times and there is no contract yet stablished, it was decided to not constrain the system design at this point.

For the choice of frequency, UHF/VHF and S-band communication systems have the strongest flight heritage [74]. The first is the most used given its low cost and power consumption, but it offers lower data rates when compared with S-band [22]. Looking at former similar scientific missions such as DICE [35] and QARMAN [36] and knowing that the operation time of the payloads is so short, the UHF/VHF seems to be sufficient. Notwithstanding, a further detailed analysis is required to understand if S-band will be required to downlink the scientific data.

Having defined the frequencies, it is now possible to focus on the subsystem parts. Given that MECSE is orbiting LEO, omni antennas will be able to support the needed data rate. Therefore, no antenna-pointing is required [25]. Also, a transceiver was selected which is a device that both receives and transmits data. In this stage, ISIS deployable antenna [77] and UHF uplink/VHF downlink transceiver [78] were selected given the low mass, power and cost.

6.3.4 Command and Data Handling (CDH)

The Command and Data Handling subsystem (CDH) represents all the system's on-board processing. It receives commands from the ground via the communications system, passes them to the appropriated components and payloads, collects and stores telemetry from across the system, collects and stores science data, and forwards it to the ground via the communications system [25]. Table 6. 10 lists the major drivers for the CDH design.

Table 6. 10 – Command and data handling design drivers for MECSE.

Design Drivers	Driven By	Impact
Processing Requirements	Payload (ENVISENSE and LP), Modes of Operation	Hardware Choice
Data Storage	Payload, Modes of Operation	Hardware Choice

The choice of the hardware will vary depending on the type of processing and the amount of data that is required to be stored. Many commercial vendors are providing complete integrated avionics systems on a PC/104 board, incorporating computer processor, memory, and engineering development systems. It is recommended to buy it as a COTS with flight heritage, for instance "Nanosatellite On-Board Computer" [79] from Clyde Space which provides a highly integrated robust computing platform with memory storage and low power consumption.

6.3.5 Mechanical System and Structures (MSS)

The structure is the primary chassis of the spacecraft. It mechanically supports all spacecraft subsystems as well as it might serve as thermal and radiation shielding for sensitive components [22]. It supports the launch loads and provides a stable platform for on-orbit operations [25]. One of the mission goals (SMO2) is to develop a CubeSat modular structure in-house and validate it in space. Therefore, special attention was paid to the development of the MECSE structure. The main goal is to design a 3U structure capable of being used in the future for other missions.

At this point, the preliminary design of the structure is already done assuming the 1U as empty space for the payload. Also, preliminary static linear analysis for the launch loads were performed assuming Vega launch vehicle. The work on this subsystem is being performed in parallel to this research and therefore it is not going to be detailed in this thesis.

6.3.6 Thermal Control System (TCS)

The thermal subsystem keeps all spacecraft's components within operating temperature ranges during normal operations and within survival limits under all circumstances [25]. Thermal control is important because the spacecraft experiences extreme temperature fluctuations over short time periods while in orbit [22].

At this point, preliminary static linear analyses of the spacecraft internal, environmental, and launch temperatures were already performed assuming optical properties for the surface. As a future task, TCS must also consider magnetic isolation solutions to reduce magnetic interferences coming from the payload. Once again, work on TCS is being done in parallel and therefore is not presented in this thesis.

6.4 Systems Engineering

6.4.1 Mass Budget Allocation

Having defined preliminarily the subsystems, the next step is the allocation of resources. One of the main constraints is the mass. Even though it is not possible to perform a mass budget without having the components selected, the mass limit must be stablished. Table 6. 11 presents the mass budget allocation where the final and optimal masses are outputted. Each subsystem lead shall target to the mass optimal in an attempt to optimize the system.

The idea of the budget allocation is to iterate it during the design process. For an initial stage, the mass percentage is guessed based on literature review and experience. Also, some margins were required. Firstly, 10% of a system contingency is applied to the overall system. Also, 25% of margin is given to components in development, 15% to subsystems not yet defined and 5% to COTS [25], [39]. The idea is to decrease gradually the subsystems margins by defining them.

Table 6. 11 – Mass budget allocation per subsystem considering margins.

Subsystem	Mass [%]	Mass Limit [kg]	Margin [%]	Mass Optimal [kg]	Rationale
System Contingency	N/A	4	10 %	3.6	Management
Spacecraft	100 %	3.6	N/A	2.95	Management
Payload	41 %	1.48	25 %	1.11	In Development
EPS	15 %	0.54	15 %	0.46	In Study
AOCS	15 %	0.54	15 %	0.46	In Study
TTC	5 %	0.18	5 %	0.17	COTS
CDH	5 %	0.18	5 %	0.17	COTS
MSS	15 %	0.54	15 %	0.46	In Study
TCS	2 %	0.07	15 %	0.06	In study
Interfaces	2 %	0.07	15 %	0.06	In Study

In conclusion, each subsystem must not exceed the mass limit and shall try to achieve the mass optimal. There is always a remaining system contingency for unplanned events.

6.4.2 Risk Analysis

Finally, it becomes crucial to summarize the decisions performed which will allow to evaluate the state of development of each subsystem and the risk involved (see Table 6. 12)

Table 6. 12 – Summary of technical development of subsystems.

Reference	Technical Summary	Rationale	Further Work	Risks and Impacts
ENVISENSE (PL01)	No need for environmental sensors. Use only GPS.	Manage power, mass and volume.	Estimate p, T and ρ using altitude and solar irradiance using solar arrays.	Accuracy of atmospheric data.
LP (PL02)	mNLP needs modifications. Decrease the size of the booms. Determine the bias voltage.	Flight heritage. High-quality measurements of electron density.	Study the relationship between probe bias voltage and boom size. Analyze the errors in measurements due to LP bias and variations in the attitude.0	mNLP without booms is assumed. The booms impact the design of EMG.

EMG (PL03)	Develop the EMG in-house.	Not available in the market. The design shall be optimized.	3D simulations and experimental studies. Analyze the impact of the sensor location. Optimization studies for power and mass.	EMG drives the EPS and the mass and size of the system. Interfaces can be an issue.
EPS	Power Source: Solar Arrays. Power Storage: Ultracapacitors, Battery. Power Management: PMB.	Further studies are required to define the subsystem. EPS is driven by EMG.	Finalize the power budget. Determine number of ultracapacitors and select the components. Evaluate the necessity of developing a PMB. Test the ultracapacitors.	Some components may not be able to handle the power flow coming from the EMG.
AOCS	Attitude and Orbit Determination: 6x fine sun sensors, 1x horizon sensor, 1x MEMS gyroscope, 1x GPS. Attitude Control: aerodynamic stabilization with momentum wheel.	Further studies are required. AOCS is driven by the EMG design and aerodynamic torques.	Perform trade studies. Model the dynamics of the spacecraft. Evaluate the attitude of the system through time. Test different dart configurations. Size the momentum wheel. Study the detumble mode.	Dart configuration impact the MSS. Deployables panels with actuators must be used.
TTC	Antenna: ISIS Transceiver: ISIS	Flight heritage COTS	Define final GS and data rate. Develop a link budget.	If high data rate, a S-band transmitter shall be chosen.
CDH	On-Board Computer: Clyde Space	Flight heritage COTS	Define data flow diagram and the total amount of data.	More memory may be required.
MSS	Develop a modular structure. (1U for payload and 2U for BUS)	Preliminary design of the structure. Static linear analysis for the launch loads.	Transient analysis for the launch loads. Detail the design of the payload unit.	Structural joints and some parts may need to be re-designed.
TCS	Coatings Surface Painting	Preliminary static linear analysis of the spacecraft internal, environmental, and launch temperatures.	Select the final materials for structure. Transient analysis of the temperature in orbit. Consider magnetic isolation materials for EMG.	Optical properties were assumed for the surface.

6.5 Concluding Remarks

In this chapter, a preliminary analysis of the MECSE satellite has been performed, including the definition of the subsystem requirements and main drivers. More important, the operation of the payload was deeply investigated which has allowed to develop an innovative concept of operations capable of minimizing power consumption and mass of the system. Furthermore, by analyzing the different subsystems, particularly the payload, power and attitude, the feasibility of the mission has been analyzed in more detail. It is concluded that the mission seems feasible under certain minor assumptions that still require a further analyze.

At this stage, the following tasks have already been performed [25], [38]:

- Refine mission requirements and constraints;
- Perform the orbital analysis;
- Develop the system architecture;
- Develop the concept of operations;
- Define and document system requirements;
- Demonstrate the feasibility of the system design;
- Prepare a technology development plan;
- Identify future technical work to be performed.
- Perform system risk analyses;

Therefore, regarding the technical part of the project, MECSE has already passed the phase A and is now ready for the SRR review which marks the end of the phase B1 of the ESA's project (see Figure 2. 14). This work was the final goal of this MSc thesis and it represents a huge contribution for MECSE project.

Chapter 7

7 Conclusion

RF blackout mitigation is of the utmost importance for the design of future space entry vehicles considering that communication is critical for the accomplishment of manned and unmanned space atmospheric missions. One solution is to manipulate the plasma layer using an electromagnetic field which will reduce the electron density in a specific region.

This thesis proposed an innovative and low-cost nanosatellite mission capable of validating that theory in space. The mission can pave the way for the development of a tool for the manipulation of the plasma layer and fast-forward the research on RF blackout mitigation. The work presented here consisted in the mission design of the early phases (0, A and B1) focusing on the systems engineering activities that are required during the project life cycle of a space project.

Starting with the feasibility analysis, it was concluded that the mission is feasible, but the EMG needs to be developed. The mission was defined and analyzed and a preliminary design of the system was proposed. Analyses for each subsystem were performed and the system drivers and requirements were defined.

During the analysis, the power, attitude and orbital lifetime were identified as the parameters with the most influence for the success of the mission and were thoroughly investigated. For the power, a novel concept of operations was proposed capable of minimizing the power consumption. Regarding the attitude, aerodynamic stabilization was the solution found to achieve a velocity-vector stabilization while decreasing the mass of the subsystem. For the orbital lifetime, considering a launch date of 2020, MECSE shall be inserted in orbital altitudes lower than 375-km. The results show that for a 350-km initial orbit with 52.6º inclination and a satellite BC of 80.00 kg/m^2, MECSE lifetime is 0.8 years.

Furthermore, an investigation was performed regarding the orbital lifetime prediction of a CubeSat in LEO. The impact of different solar and geomagnetic activity modeling approaches on the final results was assessed and the parameters that are recommended by the standard guidelines to predict orbital lifetime were evaluated.

The results have shown that are significant deviations in the orbital lifetime predictions given by different methods used to model solar and geomagnetic activity. Also, the comparison with historic data from already decayed nanosatellites has allowed to conclude that the satellite parameters recommended by the standards to perform such simulations can lead to erroneous results. Hence, this study has demonstrated the importance of performing a careful evaluation of the methodologies and parameters used in the simulations before trying to predict orbital lifetime of a CubeSat.

As a final conclusion, it was possible to conclude that the objectives stipulated for this project were successfully accomplished.

7.1 Achievements

All the research objectives proposed at the beginning were fully addressed, which has resulted in several contributions for the MECSE project.

The results of this work are of high importance for the advancements of the mission within the project life cycle. The required technical activities in the reviews were successfully completed which allowed the project to go from an initial scientific theme (beginning of phase 0) to a completely defined mission with the preliminary design of the satellite and the respective concept of operations, as well as the definition of the system requirements (phase B1).

More important, this work has become the stepping stone of MECSE's project since it has identified the most important mission aspects to be analyzed in latter phases.

7.2 Difficulties

Although the mission analysis of MECSE nanosatellite was one of the main goals of this M.Sc. thesis, this was also the phase in which most of the difficulties arose.

The first challenge appeared when trying to predict the orbital lifetime. It was found that the standard guidelines proposed different solar and geomagnetic activity modeling approaches. When performing simulations, it was found that the results were very different from each other. Thus, it was decided to investigate further this issue by comparing different models against historical data from already decayed CubeSats.

The second challenge appeared when searching for these already decayed CubeSats. It turned out that most of the satellite missions do not provide information about the orbital lifetime but only about the mission lifetime, which are different parameters. The mission lifetime refers to the end of satellite operations, that means that the satellite has not decayed necessarily. Also, the observed orbital lifetimes have some uncertainties associated that need to be taken into account. For instance, no information was found about the decay altitude, which was assumed as the same for both satellites and for the simulated results.

Defining the satellite parameters was the third challenge encountered. The standard guidelines recommend using a drag coefficient of 2.2 and the mean cross-sectional area. However, when compared against the historical data, it was found that the use of these parameters leads to erroneous results. Thus, it was decided to investigate the effects of variations of satellite parameters on the orbital lifetime prediction.

Concerning the system design, the main challenge was that no technology for the EMG was found available in the market. This had consequences at a system design level due to interdependencies between different subsystems.

7.3 Future Work

As mentioned throughout this M.Sc. thesis, there are several studies that still need to be performed in different areas of the project. A summary is provided below concerning the most important tasks from a systems engineering point of view. For a more detailed description in a specific area, it is recommended to refer to the respective chapter or section.

Firstly, regarding the scientific case, a further study about the relationship between the formation of plasma and the type of flow, given by the Knudsen number, is required, particularly to understand what are the differences between manipulating ionospheric plasma and re-entry plasma. Recall that ionospheric plasma is caused due to solar irradiance ionization whereas re-entry plasma is caused due to the formation of a shock wave at the front of the vehicle. Given that the numerical model being developed at UBI is for re-entry plasma conditions, the results coming from the simulations may have some degree of uncertainty that needs to be evaluated.

Secondly, the mission analysis needs to be performed again when more information about the launch, orbit and the ground station are available. Also, to accurately predict the orbital lifetime, the attitude of the satellite over time must be studied to determine the variations of cross-sectional area. The drag coefficient of the satellite must also be analyzed in more detail. Moreover, given that aerodynamic stabilization was chosen for attitude control, it is also proposed to study the orbital lifetime in function of the angle of the deployables.

Thirdly, several analyses are required concerning the figures of merit. The external torques on the spacecraft over time must be analyzed which will allow to choose definitively the AOCS subsystem. Then, the impact of the attitude motion on the plasma layer measurements must be evaluated. This can be done by combining an examination of the attitude dynamics of the spacecraft with plasma layer simulations. For power, the EPS must be designed in agreement with the EMG. So, the two subsystems must work together to find the best solution.

Fourthly, for the system design, there are still lots of work to be done. The selection of the final hardware for all subsystems must be finished. However, most of the work depends on the payload design. There are two main studies to be performed regarding the payloads. The EMG must be tested experimentally assuming the location of the LP. This will allow to proof its feasibility and provide a preliminary value of power and current. Meanwhile, a study about the relationship between the size of booms and probe bias is required to determine the final location of the sensors with respect to the satellite surface. Regarding the other subsystems, for more details about the future work please refer to 6.4.2.

Finally, a work breakdown structure must be defined in order to plan and prioritize the next technical tasks to be performed by each subsystem lead. Also, recall that in this thesis the project management tasks are not included and are being done in parallel. Thus, it is essential to meet with the project management lead to review the already performed work and plan the future phases of the project. In the end, a concurrent engineering meeting shall be scheduled.

For the orbital lifetime predictions, it would be useful to analyze in more detail the drag coefficient of an already decayed CubeSat and compare it with the values obtained from the validation study. This would allow concluding more about the accuracy of the methods used to predict the lifetime. Also, the step used for orbital and satellite parameters in future sensitivity studies must be reduced.

7.4 Publications and Conferences

The work developed in this thesis regarding the mission characterization and definition has already been presented in an international conference with the title: “Mission Definition and Conceptual Design of MECE Nanosatellite”. The 10th Pico and Nano Satellite Workshop on "Technologies for Small Satellite Research" occurred at Wurzburg University, in Germany [80].

Also, the methodology used for the mission analysis in this thesis were already used before during the design of Snow Water Equivalent with Altimetry mission during an ESA workshop. This work has been presented at the International Astronautical Congress (2017) and it will be published in the conference proceedings with the title: “Snow Water Equivalent Altimetry Mission: Enabling Direct Measurement of SWE on Sea Ice and Landing in the Cryosphere” [81].

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Appendix A

A Simulations of Orbital Decay

The simulations for orbital decay presented in Figure A. 1 and Figure A. 2 were performed in STK software using the CSSI model for solar and geomagnetic activity. The mean cross-sectional area and a drag coefficient of 2.5 were considered.

A.1 Orbital Decay of AeroCube-3

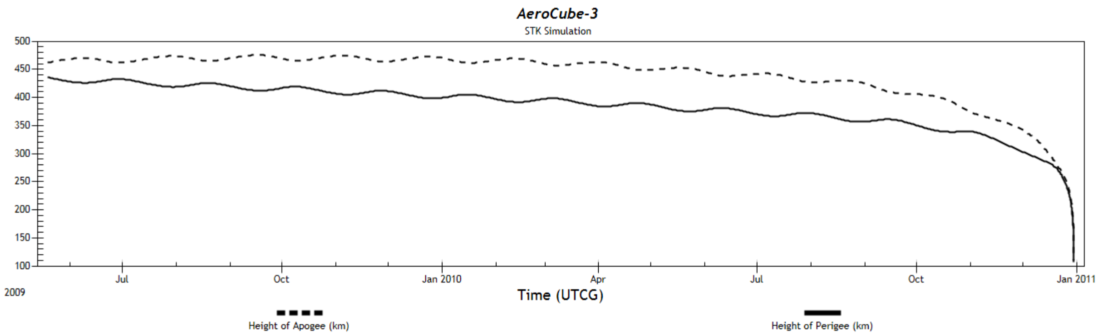

Figure A. 1 – Simulation of AeroCube-3 orbital decay considering a C_d of 2.5 and the A_{mean}.

The error calculated between the simulated orbital lifetime of AeroCube-3 (1U CubeSat) and the observed one was 2.18 %.

A.2 Orbital Lifetime of GeneSat-1

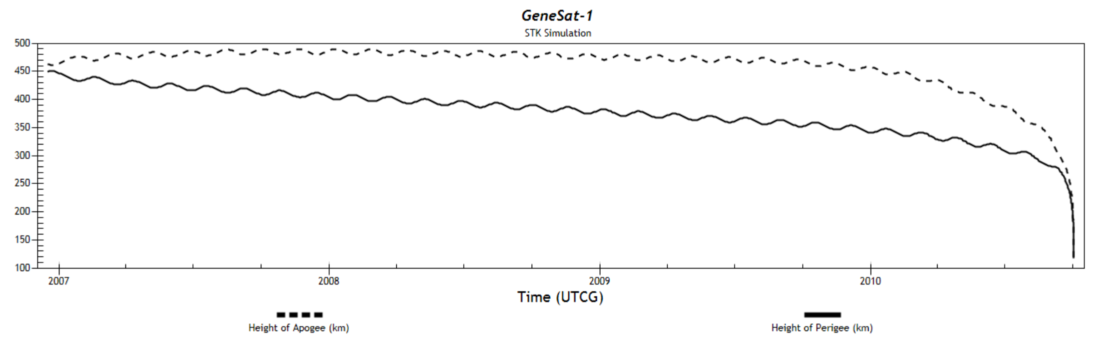

Figure A. 2 - Simulation of GeneSat-1 orbital decay considering a C_d of 2.5 and the A_{mean}.

The error calculated between the simulated orbital lifetime of GeneSat-1 (3U CubeSat) and the observed one was 4.52 %.

Appendix B

B Comparison of Orbital Lifetime Predictions

B.1 Sensitivity Study of Orbital Altitude

Table B. 1 - Orbital lifetime prediction in function of altitude using MECSE and ISO parameters.

h [km]	ISO - Orbital Lifetime [yrs]			MECSE - Orbital Lifetime [yrs]		
	CSSI	LP	ECSS	CSSI	LP	ECSS
250	0.03	0.04	0.04	0.05	0.05	0.06
275	0.07	0.08	0.08	0.10	0.12	0.13
300	0.14	0.17	0.18	0.21	0.25	0.27
325	0.27	0.32	0.36	0.40	0.50	0.58
350	0.52	0.64	0.74	0.80	0.92	1.08
375	0.95	1.08	1.28	1.40	1.50	1.71
400	1.70	1.66	1.83	2.30	2.10	2.24
425	2.60	2.20	2.31	3.20	2.75	2.82
450	3.40	2.81	2.87	4.10	3.44	3.43
475	4.20	3.47	3.44	5.20	4.38	4.07
500	5.30	4.35	4.02	6.60	5.79	5.02

From Table B. 1, the main differences between ISO and MECSE orbital lifetime are evident. For instance, for the 350-km case with 52.6° inclination, the CSSI prediction is 0.28 years different (approximately 3 months). Also, using the MECSE parameters the longest time is predicted. This confirms that by using MECSE parameters the worst case is being studied.

B.2 Sensitivity Study of Orbital Inclination

Table B. 2 - Orbital lifetime prediction in function of inclination using MECSE and ISO parameters.

i [°]	ISO - Orbital Lifetime [yrs]			MECSE - Orbital Lifetime [yrs]		
	CSSI	LPN	ECSS	CSSI	LPN	ECSS
0	0.42	0.43	0.49	0.70	0.70	0.77
15	0.41	0.45	0.51	0.67	0.73	0.80
30	0.43	0.51	0.58	0.70	0.79	0.88
45	0.51	0.6	0.7	0.78	0.88	1.01
60	0.56	0.67	0.77	0.84	0.96	1.14
75	0.59	0.73	0.84	0.89	1.03	1.24
90	0.56	0.73	0.84	0.87	1.03	1.24
105	0.55	0.7	0.8	0.84	0.99	1.18
120	0.47	0.59	0.64	0.75	0.87	1.00
135	0.41	0.48	0.57	0.66	0.76	0.86
150	0.35	0.4	0.45	0.56	0.65	0.73
165	0.33	0.35	0.39	0.52	0.56	0.64
180	0.32	0.33	0.37	0.52	0.53	0.61

From Table B. 2, the main differences between ISO and MECSE orbital lifetime are once more evident. For instance, for the 90º inclination case at 350-km altitude, the CSSI prediction is 0.31 years different (approximately 4 months). Also, using the MECSE parameters the longest time is predicted. This confirms that by using MECSE parameters the worst case is being studied.