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Conjunction Analysis and Collision Avoidance Maneuvers for electrodynamic tether missions

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Alla mia famiglia, che mi ha dato l'opportunità di proseguire gli studi e che mi ha sempre sostenuto in ogni mia scelta.

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ABSTRACT

This Master's thesis presents an astrodynamics study on Conjunction Analysis and Collision Avoidance Maneuvers for electrodynamic tether systems. The main purpose of the thesis is to analyze the risks of impact between satellites and space debris, and study in detail the processes which minimize the possibility that spacecraft collide with other orbiting objects.

The research work is related to space sustainability and Space Situational Awareness (SSA), which concerns the understanding of Space through the tracking of resident space objects (RSO), in other words all those objects that orbit the Earth, with the goal of preventing collisions to protect the space environment and the use of extra-atmospheric resources.

When evaluating the level of impact risk, i.e. estimating the probability that a close approach becomes a collision, some fundamental data related to the two identified objects are considered: size, orbital parameters and position covariance. However, the knowledge of a body's orbital position and its future evolution is influenced by the effect of errors on measurements and by the presence of different perturbative terms, such as atmospheric drag and solar radiation pressure. This thesis aims to create a model using MATLAB to investigate the uncertainties on this type of analysis, the probability of collision, the accuracy and the time tolerance necessary for carrying out the orbital correction maneuvers. In particular, satellites in LEO orbits (Low Earth orbits) equipped with tethered deorbiting devices are considered for the analysis. These electrodynamic tether systems are qualified as green technologies, because they do not require the consumption of propellant to perform the maneuvers, but base their functioning on switching on and off the current inside the tether, through which they can change the rate of descent.

The model and the results of the simulations, carried out at different altitudes and orbital inclinations, represent a tool to ensure the safety of electrodynamic tether missions that can be integrated into solution strategies for the space debris problem.

SOMMARIO

Questa Tesi Magistrale presenta uno studio di astrodinamica sulle Conjunction Analysis, ossia le Analisi delle Congiunzioni tra satelliti e altri oggetti nello spazio, e sulle manovre di correzione orbitale (Collision Avoidance Maneuvers) per sistemi spaziali a filo elettrodinamico. Lo scopo principale della tesi è quello di analizzare i rischi d'impatto fra satelliti e detriti spaziali e studiare in modo dettagliato i processi che riducono al minimo la possibilità che i veicoli spaziali si scontrino con altri oggetti orbitanti.

Il lavoro di tesi si inserisce nell'ambito della sostenibilità spaziale e della Space Situational Awareness (SSA), che riguarda la comprensione dello Spazio attraverso il tracciamento dei resident space objects (RSO), ovvero di tutti quegli oggetti che orbitano attorno alla Terra, con l'obiettivo di prevenire eventuali collisioni per proteggere l'ambiente spaziale e l'uso delle risorse extra-atmosferiche da parte dell'uomo.

Quando si vuole valutare il livello di rischio d'impatto, ossia stimare la probabilità che un approccio ravvicinato diventi una collisione, bisogna considerare alcuni dati fondamentali relativi ai due oggetti identificati: la dimensione, i parametri orbitali e la covarianza. Tuttavia, la conoscenza della posizione orbitale di un corpo in orbita e della sua futura evoluzione è influenzata dall'effetto degli errori sulle misure e dalla presenza di diversi termini perturbativi, come il drag atmosferico e la pressione di radiazione solare. La presente tesi si propone di realizzare un modello tramite il software MATLAB per determinare le incertezze su questo tipo di analisi, la probabilità di collisione, l'accuratezza e la tolleranza temporale necessaria per lo svolgimento delle manovre di correzione. In particolare, sono considerati satelliti in orbite LEO (orbite terrestri basse) dotati di dispositivi per il deorbiting di tipo tethered. Questi sistemi a filo elettrodinamico vengono qualificati come tecnologie green, in quanto non prevedono il consumo di propellente per eseguire le manovre, ma basano il loro funzionamento sullo spegnimento o sulla modulazione della corrente all'interno del tether, attraverso cui possono cambiare il rateo di discesa.

Il modello e i risultati delle simulazioni, svolte a diverse altitudini e inclinazioni orbitali, costituiscono uno strumento per garantire la sicurezza delle missioni con sistemi elettrodinamici a filo che può essere integrato nelle strategie di soluzione per il problema dei detriti spaziali.

Contents

Chapter 1 – Introduction

The First Chapter introduces the context in which this thesis work took place and defines in detail the problem under analysis and the objectives of the thesis. An introduction about the importance of Conjunction Analysis and Collision Avoidance Maneuvers is followed by some insights on the space debris problem. Then, different types of reentry methods are presented, with a detailed description of the functioning of electrodynamic tether systems (EDTs). The most important features, applications, advantages and disadvantages of these deorbiting devices are explained. The last paragraph includes the main goals and the outline of the thesis.

1.1 Background

Satellites surrounding our planet help us deliver communication and navigation services, study climate change, prevent environmental disasters and solve important scientific questions. However, during its operational life, a spacecraft is exposed to risk of collision with other space objects, like orbital debris (Fig.1). This risk is higher in Low Earth Orbit (LEO) where most of the space population is condensed. Accidental collisions can produce dangerous clouds of debris that can damage other satellites with cascading effect, making the most useful orbits no longer available.

Keeping track of these objects has become a really crucial aspect, which leads to the importance of Conjunction Analysis and the definition of Collision Probability. The study of the orbit's evolution traveled by a space object can be used to determine if there is a risk of impact with the satellite of interest and to implement some evasive maneuvers, known as Collision Avoidance Maneuvers (CAM), in order to avoid the impact.

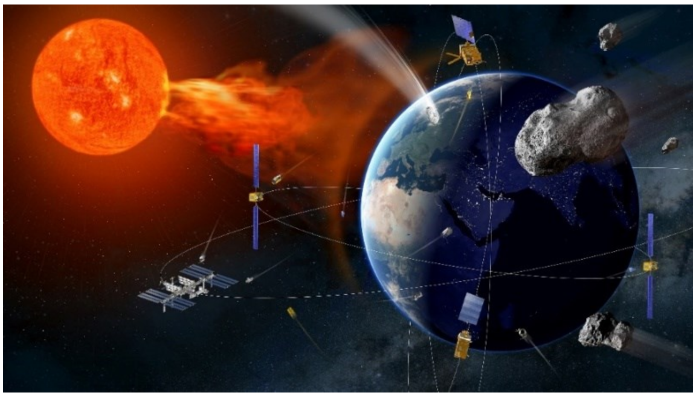


Figure 1: Space Safety & Security (Credits: ESA).

In the last decades, new technologies and strategies have been developed to find a solution for this problem and try to prevent it from worsening. One of the most recent technologies includes the use of electrodynamic tether systems for lowering a satellite's orbit for disposal at the end of life. Although the additional cross section of the deorbiting device may result in higher collision probabilities, it has been demonstrated that electrodynamic tethers are very reliable systems and present a number of advantages [1]. The most important feature which distinguish them from other devices is that they can execute Collision Avoidance actions without the consumption of fuel. Once the impact is estimated, they perform orbital corrections by simply switching on and off the current, thus modifying their orbital dynamics and rate of descent.

The present work of thesis will focus on the study of these systems and their application as part of space debris mitigation strategies.

1.2 Space Debris Problem

Space debris or "space junk" represents a serious threat for today's space activities [2]. In case of collision with operational satellites, it could compromise the functionality of the satellite itself, and consequently the success of the whole mission. According to Inter-Agency Debris Coordination Committee (IADCC), an international association that deals with orbital traffic management, space debris is defined as objects of artificial origin including fragments of satellites which, in orbit around the Earth or reentering through the Earth's atmosphere, are no longer functional. Classified on the basis of average size and mass, about 20,000 have a diameter greater than 10 cm, so they can be tracked by optical measurements performed by space surveillance systems. The rest of them, which constitutes the largest amount, is neither observable nor traceable (Fig.2).

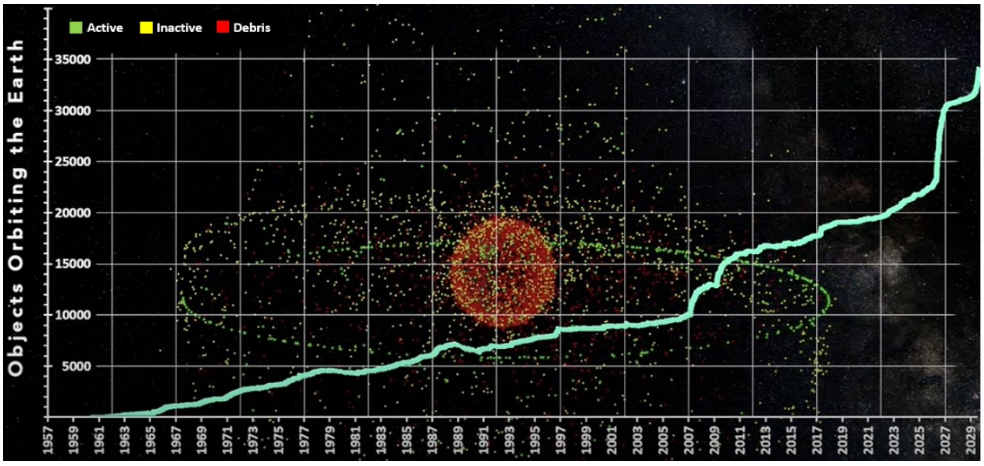


Figure 2: Evolution of the space objects population over the years 1957-2029 (Credits: AGI, an Ansys Company).

Since the launch of Sputnik 1 in 1957 and the beginning of the space age, human beings have increased their activities in space exponentially. Numerous satellites and devices for monitoring the planet, studying the Universe and human life in space still orbit the Earth. However, every object that is thrown into space is destined to become waste. As it happens on Earth, indeed, everything that man uses, once it has lost its usefulness, becomes waste and needs to be disposed of. The increase in space activity also corresponds to a substantial increase in the production of space debris (Fig.3). Artificial space debris is in bigger proportion than objects of natural origin.

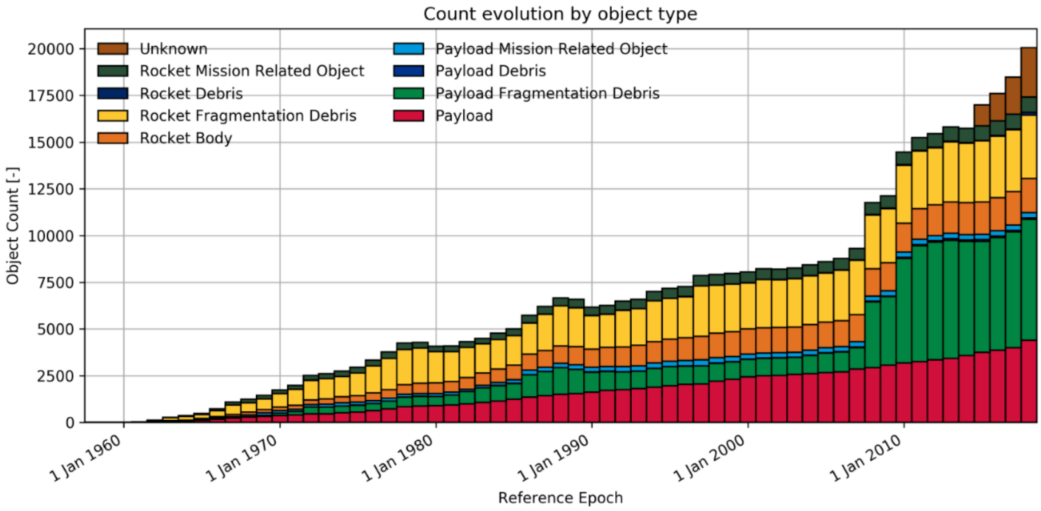


Figure 3: Composition of orbital debris (Credits: ESA).

In the past, a few accidents produced nearly a third of the current debris population in LEO [3]. In 1996, a French satellite was hit and damaged by debris created by a missile that exploded at high altitude a decade earlier. On February 10, 2009, a decommissioned Russian satellite, Cosmos-2251, collided the Iridium-33 satellite, causing the creation of more than two thousand fragments. In 2007, an anti-satellite missile test conducted by China destroyed the Fengyun-1C weather satellite, and added over three thousand fragments to the list.

A solution that is often used is to move the satellite into a geosynchronous orbit, which resides over 35,000 kilometers of altitude. This is usually done at the end of a satellite's operational life, especially for spacecraft in particularly high orbits. During this process, the satellite is passivated, i.e. deprived of all internal energy, in order to prevent explosions or uncontrolled reactivations.
Once the displacement and passivation maneuvers have been completed, the satellite will continue to orbit the Earth in its “cemetery” orbit, even for hundreds of years.

In addition to satellites that end their activity, other categories need to be considered. There are stages of launchers that remain in orbit (or re-enter after a certain period), debris produced by the breakage of equipment in orbit, by collisions or by the action of deterioration due to the highly hostile space environment.
Lately, mega-constellations of satellites like Starlink have been equipped with propulsion systems for controlled reentry into the atmosphere at the end of their operational life. This may be another solution to avoid the problem of overcrowding of the orbits.

In the future, an excessive occupation of the Earth's orbits could determine the so-called “Kessler Syndrome”. In such conditions frequent collisions due to the high volume of uncontrolled debris would trigger a chain reaction that could compromise most of the objects in orbit. If this almost “apocalyptic” scenario were to occur, space activity would be extremely compromised and reduced, if not canceled.

1.3 Reentry Systems

The reentry process into the atmosphere could lead to possible impact of debris on the ground. Our atmosphere is rather dense and allows to disintegrate by friction most of the objects that impact it continuously. However, large objects are not disposable by burning them through friction. If control of orbiting objects were lost, the chances of impact on the ground would increase exponentially, with effects that would be disastrous and extremely dangerous for humans.

Since the first days of space flight, 24,000 objects from the USSTRATCOM catalog have entered the Earth's atmosphere. The objects total mass that returned to Earth until 2016 is about 32,000 tons, while that currently in orbit (2017) amounts to 7,500 tons distributed over about 17,900 large objects.

Generally these are the reentries of objects of small size and mass that do not involve any risk on the ground. There is also space debris of considerable mass whose reentry frequency is about one per week. Comparing the objects put into orbit with those that have returned, the ratio is significantly increasing.

Three different types of reentries can be defined:

- Uncontrolled reentries occur when no maneuver has been performed in orbit affecting the angle of reentry at an altitude of 120 km. In this case, the reentry angle, defined as the angle between the velocity vector and the horizontal plane, will correspond to the angle that the spacecraft assumes due to the decay in the atmosphere. Place and time of return cannot be predicted with precision.
- Controlled reentries are entry events after a deorbit maneuver. They consist in increasing the angle of reentry at 120 km of altitude in order to identify a destination for the object whose mass is considered constant. The drop zone of the reentry break debris is controlled. In this way it is possible to minimize the dangers associated with the reentry of objects, effectively avoiding inhabited areas or areas at risk.
- Semi-controlled reentries are events that characterize objects with particularly elliptical orbits and for which the reentry occurs through the lowering of the perigee within suitable altitudes by the gravitational interaction of the Sun and the Moon.

Although statistically controlled reentries are in a clear minority of the total, they comprise about 47% of the returned mass. This is due to the fact that they generally involve extremely large structures such as space stations, whose uncontrolled impact would be catastrophic.
In the case of uncontrolled or semi-controlled reentry, the forecasts of time and position of reentry are carried out through the temporal propagation of the orbits, determining when the impact with the atmosphere will occur through to the evolution of the orbit itself. The reentry occurs with the progressive loss of energy of the objects orbits. In fact, circular orbits becomes spiraling orbits until the atmospheric friction forces the object to hit the earth's surface (or to disintegrate in the atmosphere).

On the other hand, for controlled reentries, the position and return time are defined according to the needs of the mission. The South Pacific Ocean Uninhabited Area (SPOUA), which is the largest unpopulated ocean space in the globe, is the most used for these kind of reentries. In particular, spacecraft are often landed in one area, called “Point Nemo”, which is one of the farthest places from the mainland and is located in the heart of the Pacific Ocean.

The following image (Fig.4) shows the differences in the reentry process between larger bodies and small objects:

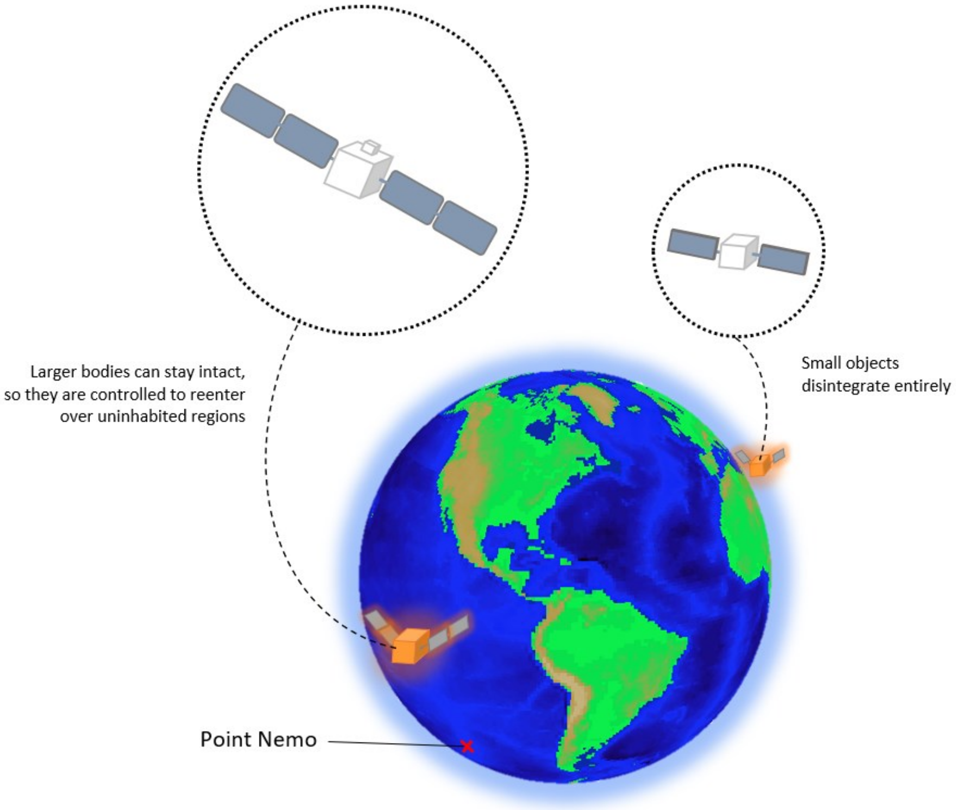


Figure 4: Comparison between larger bodies and small objects reentry process.

There are several types of reentry systems. In general, they can be classified as follows:

- high thrust systems, in which the Delta-V is generated by a chemical system. Satellites with this type of propellant are usually landed to Point Nemo. Alternatively, another solution can be sending them into the so-called "graveyard orbits". In this case, spacecraft are not maneuvered until they descend at the predetermined point, but are made to rise to a higher orbit.
- low thrust systems, which include electric thrusters and neutral resistance atmospheric sails. These systems generally must have a mass lower than 1000 kg in order to burn in the atmosphere as the satellite descends spiraling.
- systems which do not require the consumption of fuel, usually satellites equipped with a deorbiting device, such as electrodynamic tethers.

1.4 Electrodynamic tethers

Electrodynamic tethers (EDTs) are long conductive wires which base their functioning on the electromagnetic field. They have many applications, such as debris removal, payload delivery, controlled docking, deployment and recovery of satellites in space. EDTs can be used as deorbiting devices to reenter satellites at the end of life from LEO orbits. They are mounted before the satellite's departure and deployed at the end of the operational mission to bring the satellite back in a few months without consuming propellant.

One of the main advantages of the EDT system compared to passive systems (like atmospheric sails or balloons) is that it is a controllable system: this ability is based on modulating the current to change the satellite descent rate.

The disadvantages are related to reliability of the deployment mechanism at the satellite's end of life and to the length L of the tether (typically in the range from 500 m to 3 km). It is not a compact system, having a fairly high impact area, and is therefore more subject to perturbations and atmospheric drag.

The electrodynamic thrust is the Lorentz force generated by the flow of electric current inside the tether in the geomagnetic field. The force is perpendicular to the tether and its direction depends on the direction of the magnetic field. Electrons are collected from the external plasma by the wire and flowed through the tether generating current until they are emitted back using a cathode into the ambient plasma (Fig.5).

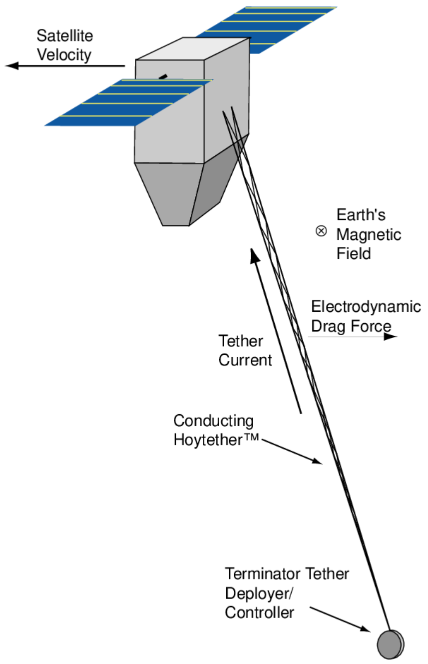


Figure 5: Electrodynamic tether functioning [4].

The Lorentz force created by the flowing current can be used to modify the trajectory of the satellite in a desired way and change the orbital altitude. For example, it is possible to change the direction of the current flow in order to boost the orbital altitude of the satellite. This maneuver is feasible using solar panels to collect energy from the Sun and generate power.

On the other hand, during deorbiting maneuvers, the force is oriented in the opposite direction of the velocity vector, contributing to lower the orbit and slowing down the satellite until it disintegrates in the atmosphere. Such a deorbiting strategy could prevent space debris from accumulating in orbit.

1.5 Thesis Objectives and Outline

The main objectives of this Master's thesis are:

1. to create a reliable model and simulation tool for studying the risk of collision between an electrodynamic tether system and other space objects;
2. to calculate the necessary time tolerance and accuracy to execute a Collision Avoidance Maneuver;
3. to study the effect of the uncertainty parameters on Conjunction Analysis;
4. to validate the model and apply it to realistic situations;
5. to provide the guidelines to improve the model in order to promote further studies.

This thesis is divided into four chapters, excluding the Introduction:

- **Chapter 2** describes the role and the importance of Conjunction Analysis in the space mitigation strategies and how they are useful to prevent possible collisions.
- **Chapter 3** provides the theoretical knowledge and the hypotheses needed for the analysis.
- **Chapter 4** illustrates all the passages necessary to create the model, showing the results obtained by the simulations.
- **Chapter 5** concludes the thesis by presenting some analysis on future application of the model and further investigations that could be conducted.

Chapter 2 – Conjunction Analysis

This chapter provides a comprehensive description of how a Conjunction Analysis Process works, exploring in detail every steps before computing a Collision Avoidance Maneuver. Conjunction Analysis (CA) procedure generally includes: identifying a potential Conjunction, propagating ephemeris and covariances to the time of closest approach (TCA), evaluating the consequences of the satellite eventual maneuver, computing the collision avoidance action.

2.1 State of the Art in Conjunction Analysis

Conjunction Analysis (CA) indicate the collision risk assessment during a satellite conjunction, i.e. a close approach between a spacecraft and another orbiting object. The aim of Conjunction Analysis is to determine if the risk of collision is higher than a specific threshold and therefore requires a collision avoidance action.

An increasing demand for satellite services implies an increase in the number of objects launched on orbit. This phenomenon, along with already existing on-orbit collisions and break-ups, has determined a great density of objects in orbit. Other than the catalogued 20,000 objects with a diameter bigger than 10 cm, it is estimated that there are thousands, possibly millions, of space objects that are not currently trackable and orbit the Earth in an uncontrolled way. As mentioned before, this large population of objects represents a real threat to the operational satellites. In order to keep the active satellites safe, it is therefore necessary to model the population and forecast any potential collisions so that spacecraft can avoid the danger in advance.

Conjunction analysis, collision assessment and avoidance, all have the same goal: determining when and where collisions might occur and analyze the probability and outcome of such a collision.

In general, Collision avoidance processes typically have four stages:

1. Initial screening for potential collisions
2. Manual or automated risk assessment of identified collisions
3. Refinement of the risk assessment
4. Collision avoidance action

In some cases, the later steps are not required when the initial screening produced a so-called false positive alert. This happens when, after a potential collision is identified in the screening process, once the risk assessment is refined, it is considered a non-event and therefore not requiring an avoidance action.

2.2 Conjunction Analysis Methods

The most important steps of Conjunction Analysis methods are summed up in the diagram of Fig.6. Starting from identifying eventual conjunctions, on the basis of Owner/Operator (O/O) ephemeris, or SST catalogues, the process requires positions and covariances to be propagated to the TCA. After that, the computation of collision risk can be determined by two different approaches, the exclusion volume or the calculation of collision probability, which will be discussed later.
Identification of high interest events (HIE) whose risk exceeds the defined thresholds is followed by the optimization of Collision Avoidance Maneuvers, based on the Conjunction Data Messages (CDMs), and the evaluation of the satellite maneuvers impact.

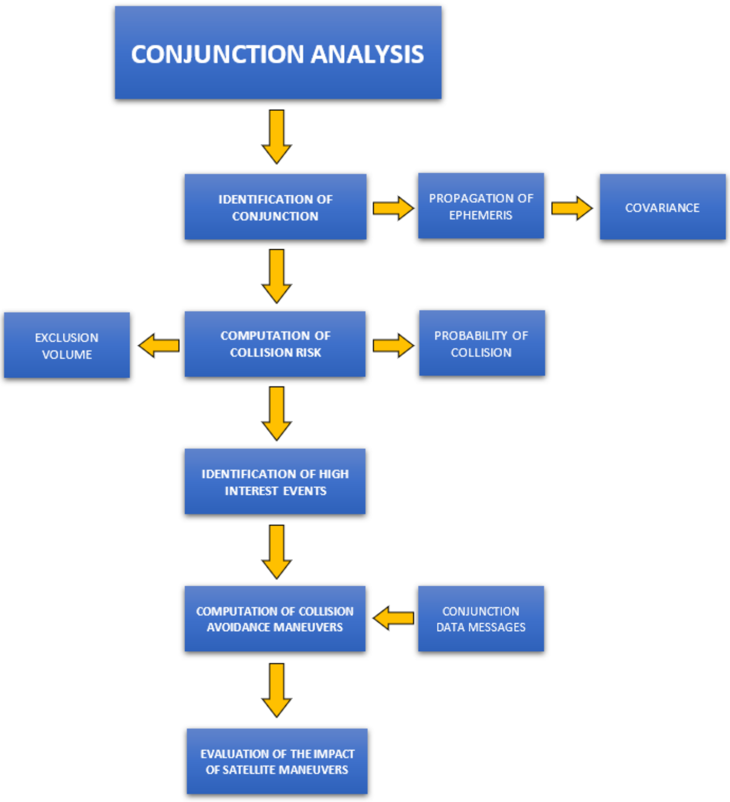


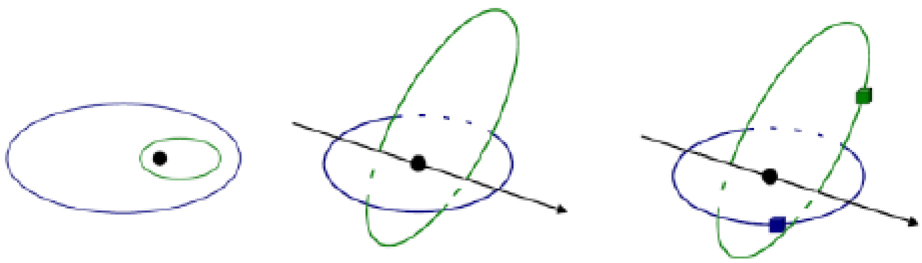
Figure 6: Most important steps in a Conjunction Analysis.

In summary, Conjunction Assessment involves:

- close approach detection, which requires the maintenance of a catalogue of space objects that constitutes the main source to perform screening and identify close approaches for operational satellites. It produces conjunction messages to notify O/O of potential risky conjunctions.
- risk evaluation for collision avoidance decision, which consists of the analysis of all available CDMs describing a conjunction and produces an evaluation of the conjunction risk level in order to detect HIE, alert and recommend avoidance action.
- collision avoidance action. Once the previous steps of Conjunction Assessment are completed, the O/O must evaluate the risk assessment, make the collision avoidance maneuver decision and execute the maneuver.

2.2.1 Identification of Conjunction

The whole CA process starts with the acquisition of data. Conjunction identification includes three steps: firstly, data on space objects' positions is acquired; secondly, that data is propagated to predict the future positions; finally, those positions are screening for potential conjunctions.
Conjunction identification is based on a catalogue of the current population of space objects called ephemeris. The object of interest's future positions are compared to those of objects in the catalogue, in a typically called an all-on-all analysis. This type of analysis is not usually completed because it is very computationally expensive. Instead, the catalogue is filtered to exclude objects that cannot lead to collisions. For example, if the object considered in the analysis is in a LEO orbit, then objects that never enter this region are discarded, such as GEO objects. The most common filtering techniques are depicted in Fig.7:



a) b) c)

Figure 7: Typical filters in the conjunction event search. a) perigee-apogee, b) radial distance and c) phase filter.

a) The apogee-perigee filter is based on orbit altitudes; the two orbits never intersect because the apogee of the lower one is not high enough to reach the perigee of the higher orbit.
b) The radial distance filter is based on the radius of the objects orbit on the line of intersection between the two orbits.
c) The phase filter is used when orbits may overlap but objects do not pass through the orbit crossing point at the same time.

These filtering activities take into account the uncertainties associated to the orbital information, but require a catalogue of objects to be completed. However, there is only one catalogue of orbit ephemerides available to public domain: it is the two-line element (TLE) catalogue produced by the United States Space Command and based on mean orbital elements, according to Simplified Perturbation model (SGP4/SDP4) theory [5]. It currently incorporates all objects larger than ~10cm in LEO and ~1m in GEO, containing a total amount of approximately 20000 objects. The name TLE derives from the format of the data: each object is defined by two lines of data, which includes the object ID, epoch, and orbit information, such as the six Keplerian orbital parameters. Some disadvantages of using this catalogue are that, even though the orbit information is updated daily, the TLE does not provide any information about the orbit accuracy and not contain some crucial data, like the radar cross section (RCS) of the object, its mass, dimensions or materials.

Fortunately, if required, mass, dimensions and RCS can be obtained in a multitude of ways: for example, directly from the O/O, from third parties, or from other catalogues. After obtaining all of the required information for an object, the next step is to propagate its current position from the TLE catalogue forward to compute its future ephemerides, together with those of the other catalogued objects, in order to determine if the orbits of the objects considered ever intersect and find any possible conjunctions.

2.2.2 Propagation of Ephemeris

Several methods are available to propagate the orbit of an object. In general, they can be classified into three categories: numerical, analytical and semi-analytical. All numerical propagators share the principal of adding perturbing forces to an unperturbed model. Each perturbing force generates an acceleration, which is integrated to find the future velocity and position.

Numerical propagators are often the most computationally expensive by nature, but are the simplest in design and are able to produce the most accurate and reliable results.
In analytical propagators, developed before the numerical ones in many shapes and sizes, the results depend only on the formulation of the solution, without providing a single equation for every solution. Nevertheless, all these types of propagators include some form of approximation of the problem in order to provide a simplified version of the equations of motion.
Compared to numerical models, analytical solutions have an advantage in terms of speed but are less accurate because most of them are based on series expansions, whose truncation represents an additional source of error.
Semi-Analytical propagators combines the benefits of both numerical and analytical solutions, with a trade-off between speed and accuracy.

Input errors are the main issues of all propagators, which derive from atmospheric modelling or from radar measurements of mass and dimensions of the object. Estimations in the input data fed into propagators are sources of potentially large errors, leading to inaccurate results.
Improper use is another source of error that propagators face: the timestep and setup must be carefully considered for the analysis. Too large timesteps or too few perturbations will return poor results. This problem can be avoided by validating the model using historical data for well-known orbits.

2.2.3 Computing and Propagating Covariance

Covariance matrix quantifies the uncertainty of the objects estimated state vector and it is normally provided by the ephemeris provider. When this information is not known, such as with TLE data, covariance can be assessed by fitting techniques of historical ephemeris or using look-up tables for different types of orbit published in literature.

The covariance matrix for a satellite position has a 3x3 form:

$$Covariance = \begin{bmatrix} C(R,R) & C(T,R) & C(N,R) \\ C(T,R) & C(T,T) & C(N,T) \\ C(N,R) & C(N,T) & C(N,N) \end{bmatrix}$$

where (R), (N) and (T) stands for radial, normal and tangential component respectively.
The diagonal elements represent the variance in each component [R,T,N] and can be used to define the uncertainty ellipsoid (or error ellipsoid) of an object's position. The off-diagonal entries gives the covariance between each pair of components, the product of the two components' standard deviations and their coefficient. The reference frame for an object's covariance matrix is the object's RTN frame.
The covariance matrix is calculated using the following formula:

$$Covariance = \frac{\sum_{i=1}^{n}(x_i - \bar{x})(y_i - \bar{y})}{n-1}$$

where x and y represent one of the RTN components each.

As the accuracy of a propagated orbit state reduces over time, the covariance matrix itself must be propagated to keep track of the uncertainties. The covariance can be propagated through Monte-Carlo analysis or using numerical or analytical methods as much as the position of the object is propagated. In particular, it can be propagated using the state transition matrix (STM), as follows:

$$Covariance(t) = STM \cdot Covariance(t_0) \cdot STM^T$$

The state transition matrix is formed of the partial derivatives of the current state vector, including position and velocity vectors combined in a 6x1 matrix, with respect to the initial state vector.
Covariance can also be propagated analytically by linearizing the equation of motion through the Clohessy-Wiltshire equations, which provide a transition matrix then used to propagate the covariance.

2.2.4 Computing Collision Risk

When a conjunction is identified, operators can evaluate the event using two different criteria: exclusion volume or Probability of Collision (PoC).

The **exclusion volume approach** consists of defining a region in space around the object of interest, and is based on the principle that, if another object enters this region, then an avoidance maneuver is needed. However, this method often leads to an unnecessarily large number of maneuvers, because it does not allow for risk evaluation and can end up having conservatively large exclusions regions. Exclusion volumes are generally used more in the initial screening step, before the conjunction event is analyzed thoroughly.

The **Probability of Collision approach** involves defining an Accepted Collision Probability Level (ACPL) or threshold value of probability (PoC*) for a mission. The Collision Avoidance Maneuver is therefore recommended when this threshold is exceeded.
There are a number of options for computing PoC. In this thesis the calculation of the Collision Probability is based on the Alfriend & Akella algorithm [6], which determines the PoC from the integration of a three-dimensional Gaussian probability density function (Fig.8). The procedure, used by many CA service providers, describes the PoC as a function of the miss distance between two space objects, and will be discussed in the following chapter.
The uncertainties of the two orbits are therefore translated into the uncertainty of the miss distance. In order to evaluate the risk of collision, the probability connected to the miss-distance is integrated over the area projected by the collision volume (A_c). The intersection of A_c with two Gaussian curves produces different probability values for the same collision scenario: the blue curve, with a higher peak, corresponds to a well understood orbit and better quality data, providing a lower probability than that of the red curve, that is associated to a worse-known orbit and a much larger risk.

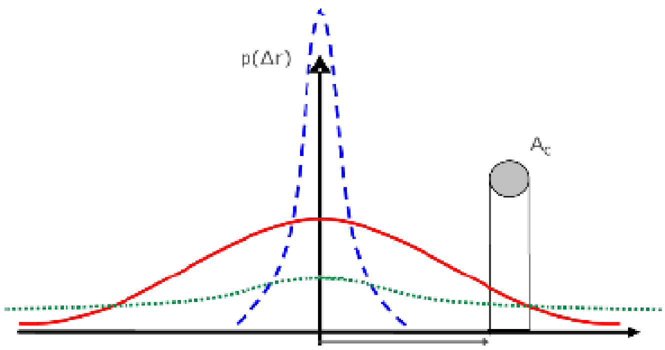


Figure 8: Representation of the Probability function of the miss-distance and integration area [7].

2.2.5 Identification of High Interest Events

High Interest Events are based on the time to closest approach (TCA), the PoC and the distances in the local reference frame (along-track, cross-track and radial miss distances). Typical values for these parameters are: TCA ≈ 3 days, PoC Є [10^{-4}, 10^{-6}]. These can be determined through software tools such as ESA's DRAMA (Debris Risk Assessment and Mitigation Analysis).
An HIE is an identified conjunction whose parameters exceed the threshold required values for PoC and miss-distance. For instance, ESA set a threshold of PoC > 10^{-4} for HIEs.

NASA uses a different approach, which implies the classification of events through multiple threshold values and using colors: events with a PoC > $4.4 \cdot 10^{-4}$ are red events, those with PoC < 10^{-7} are green events, and those between these two thresholds are yellow events. Green events are further investigated if both objects are operational spacecraft, to ensure that any planned maneuvers do not affect to conjunction evolution. Yellow events are examined and monitored, while red events are actively investigated by a team of analysts.

2.2.6 Computing Avoidance Maneuvers

The decision to compute a Collision Avoidance Maneuver (CAM) is made when PoC or exclusion volume criteria exceeds the defined threshold. However, planning a CAM must consider the operational constraints for the satellite together with the risk level of the event. Collision Avoidance action includes change in orbital state, modification of attitude, orbital trajectory correction. For EDTs, a CAM may consist in lowering the orbit by switching off the current inside the tether a certain interval of time before the expected conjunction.

Even though several potential collision avoidance maneuvers could be performed, they are usually chosen from typical approaches (along-track, cross-track), in order to make optimization easier to implement. As mentioned before, a relevant aspect of the CAM analysis is related to the constraints to be applied for computing the most optimum maneuver, which are usually imposed on the trajectory geometry, or also on execution time and direction of the maneuver. For example, if tangential maneuvers are desired, CAM will be parallel to the velocity vector. On the other hand, CAM will be perpendicular to it in case of desired maneuvers not changing the orbital period.

2.2.7 Conjunction Data Messages

Conjunction Messages (CM), either Conjunction Data Messages (CDMs) or Conjunction Summary Messages (CSMs), are standard format message warnings for upcoming conjunctions provided within 72 hours of the TCA by 18th Space Control Squadron (18SCS), previously JSpOC (Joint Space Operation Center), for all active satellites. For LEO events, CDMs are received when there is an overall miss distance of 1 km between two objects and a radial miss distance lower than 200 m; for GEO events the overall miss distance is 10 km.
CDMs are strictly advisory messages only and do not provide recommendation to perform avoidance actions, as they cannot take into consideration neither the operational constraints of the satellite nor the maneuvers it plans to perform.

Every Conjunction Message includes:

- time of closest approach (TCA);
- identification of the two objects;
- relative position and velocity;
- orbital characteristics for each object

In the European Union, Conjunction data is provided through the EU SST (EU Space Surveillance and Tracking Support Framework) web portal. The EU SST is a framework to improve SSA in Europe and offers three main services: the Collision Avoidance, the Fragmentation Analysis, and the Re-entry Analysis Service. The first service provides a risk assessment of collision between space objects, the second one detects in-orbit fragmentations to provide detailed reports, and the third assesses the risk of uncontrolled reentry of space objects into the Earth's atmosphere.

The Conjunction Data Messages data are updated daily thanks to the tracking information. However, waiting until the last moment to have the maximum accuracy to undertake the maneuver is impossible, because any CAM takes time to be planned and executed. For this reason, ESA's Space Debris Office warns the mission control teams three days before the conjunction, while the decision to execute the CAM is taken one day before.

2.2.8 Evaluation of the impact of satellite maneuvers

Prior to executing a Collision Avoidance Maneuver, it is checked that the spacecraft is moved on to a safer trajectory with a lower PoC, and verified the planned maneuver does not result in a new close approach event with a different object.
Another aspect to be taken into account is that these maneuvers are costly in terms of hours spent on the ground monitoring the skies and calculating the collision risk, not to mention missed science and data collected if onboard scientific instruments are turned off while executing the maneuver.

Moreover, the computation of the maneuver is coupled with different considerations:

- for a spacecraft with strict requirements regarding its orbit, the planned CAM must consider the time outside the nominal orbit, which can involve an interruption of a service;
- the satellite could perform a combined station-keeping and avoidance maneuver if there is a planned station-keeping maneuver around the date of the conjunction;
- the optimal maneuver is not recommended as it would add additional threats to the satellite if the post-maneuver orbit may result in additional close encounters with other space objects.

2.3 Available Conjunction Analysis Tools

There are several software tools used in SSA and CA that simulate the debris environment and can be useful to predict conjunctions and collision risk.
MASTER and DRAMA developed by ESA are some of the free available tools which provide information about the latest types and quantities of space objects.

MASTER (Meteoroid and Space Debris Terrestrial Environment) aims to define the debris environment in Earth orbit and produces a model for the prediction of debris and meteoroid fluxes. In particular, predictions can be made from the beginning of the space era to a reference epoch on November 1st 2016, until 30 years into the future. The model includes both natural and artificial (or man-made) debris.
The user is offered the option to choose different size of impacting objects with minimum values of the order of micrometers.

There are three different scenarios that can be implemented:

- Target Orbit scenario: calculate the flux of debris and meteoroid on a chosen Earth target orbit, defined by the six Keplerian orbital parameters.
- Inertial Volume scenario: compute the flux on an inertially fixed target, whose position is known and given in terms of geocentric radius, right ascension and declination.
- Spatial Density scenario: determine the debris spatial density as a function of altitude, declination and time.

The next graph (Fig.9) created with the MASTER software displays the spatial density in LEO orbits (200 km – 2000 km) for objects with a diameter greater than 1 cm.

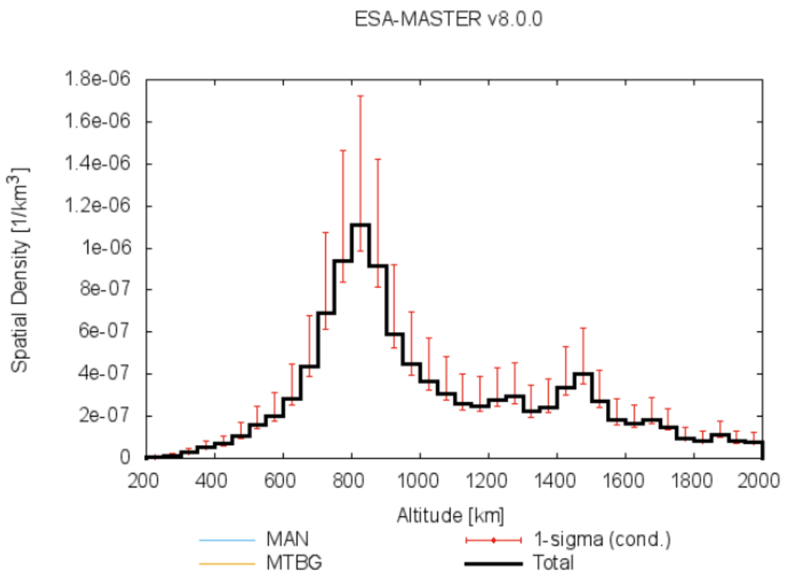


Figure 9: Debris spatial density in LEO vs Altitude (with uncertainty bars).

The most densely populated regions are between 800 km and 1000 km, which are the most critical altitudes for the analysis and for collision risk.

DRAMA (Debris Risk Assessment and Mitigation Analysis) is a tool for assessing compliance of space programs with space debris mitigation requirements. For a given mission, DRAMA enables the user to evaluate debris and meteoroid impact flux levels, analyze Collision Avoidance Maneuver frequencies, and forecast reentry survival predictions of a given object.

Chapter 3 – Theoretical Model

The Third Chapter describes all the background theoretical knowledge and the assumptions necessary to create the model, from the orbital parameters and the reference frames used in the analysis, to the environment models implemented in the orbits propagator and the functioning of a Neural Network through MATLAB Deep Learning Toolbox. In particular, the environmental models include the perturbative contribution of atmospheric drag, that of the ionosphere and of the Earth's magnetic field. Furthermore, the theoretical passages for computing the Probability of Collision (PoC) are explained thoroughly.

3.1 Orbital Parameters

All the simulations implemented to create the model are based on a Fortran orbital propagator for tethered systems, called FLEXSIM, that has been developed at the University of Padova.

Fig.10 shows the most important orbital elements considered for the analysis:

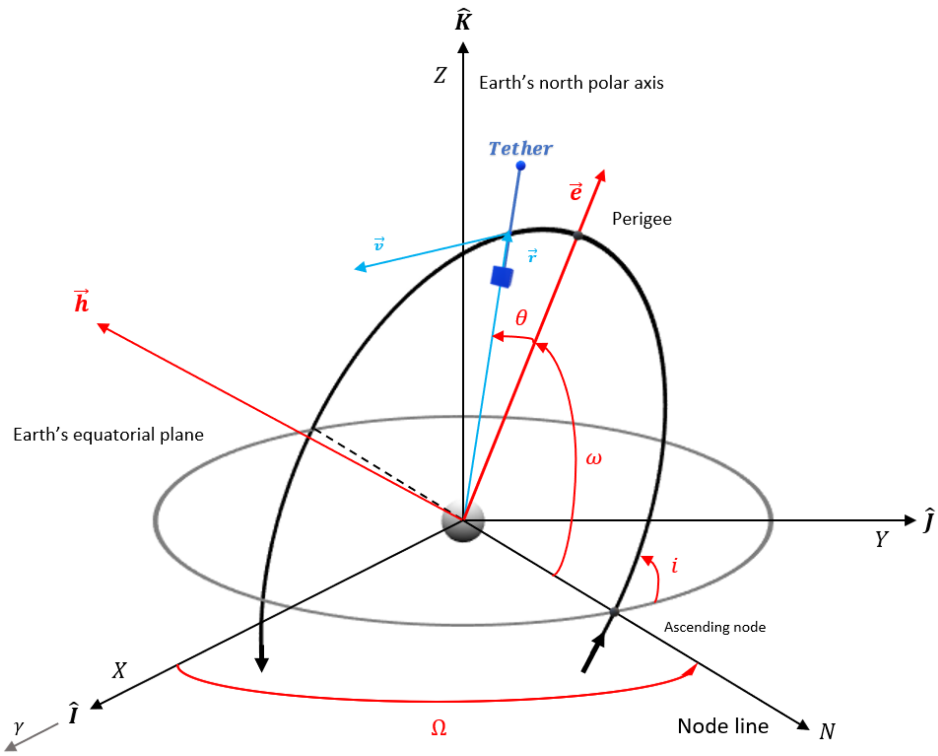


Figure 10: Orbital Parameters considered in the simulations.

The intersection between the orbital plane and the equatorial plane XY generates the node line, identified by the vector ***N***. The point located on the node line where the orbit intersects the equatorial plane from below is called the ascending node.
The perigee is located at the intersection of the eccentricity vector with the orbital path.

The six orbital parameters are:

- a semimajor axis
- i inclination
- Ω right ascension of the ascending node
- e eccentricity
- ω argument of perigee
- ϑ true anomaly

The semimajor axis is the distance which corresponds to half of the length of the orbit's major axis. The angle between the orbital and the equatorial plan is the inclination, which is also defined as the angle between the Z-axis and the normal to the plane of the orbit. The right ascension of the ascending node is therefore the angle between the X-axis and the node line, and is a positive number with values from 0 to 360 degrees. Eccentricity is the vector pointing toward the perigee. The argument of perigee is the angle between ***N*** and *e,* assuming the same values as Ω, while the true anomaly represents the angle formed by the position vector **r** and the eccentricity vector [8].

The orbital propagator aims to compute the position vector **r** and the velocity vector **v** of the tether in the geocentric system at any given time in a five-day interval, starting from the conditions of an initial circular orbit ($e = 0$) with Ω, ω and ϑ equal to zero, and varying the altitude and the inclination.

3.2 Reference Frames

The diagram in Fig.11 shows the two reference systems considered for the calculation of the collision probability and how the tether is oriented:

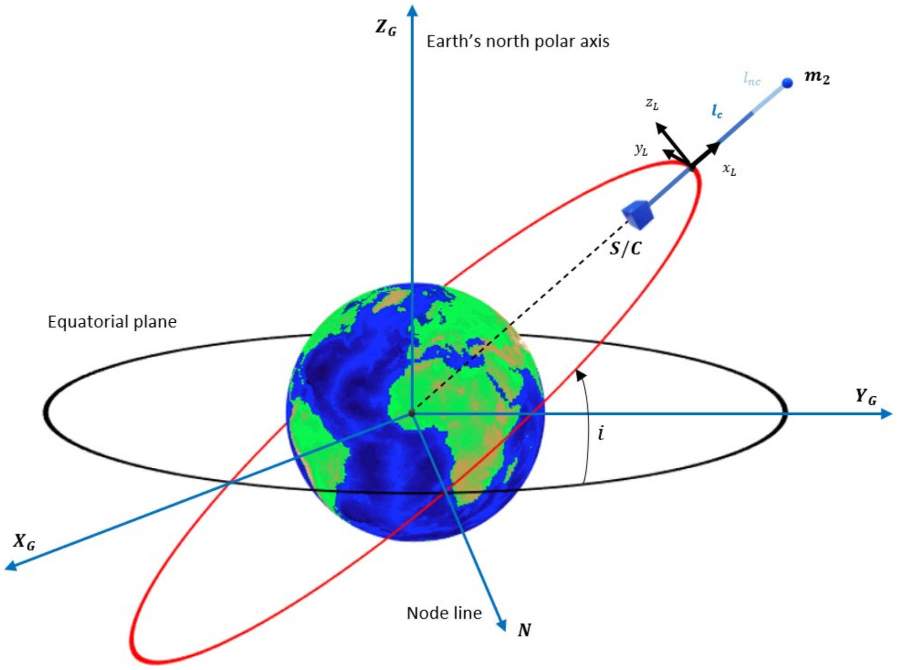


Figure 11: Reference frames and tether orientation.

The geocentric reference system, defined by the triad (X_G, Y_G, Z_G), with origin in the Earth's center of mass, has the Z_G axis coinciding with the terrestrial rotation axis, and the X_GY_G plane corresponding to the equatorial plane. The X_G axis also points towards the vernal equinox.

The body system, on the other hand, originates in the center of mass of the tethered system, located along the tether but closer to the spacecraft, as it has a greater mass. The wire that links the main mass of the spacecraft with that of the de-orbiting kit (m_2) is oriented along the local vertical which connects the center of the Earth with the satellite and corresponds to the x_L direction of the body reference system. It can be distinguished the length of the conductive part (l_c) in aluminum, and that of the insulating part (l_{nc}) in PEEK, to dampen the oscillations that would make the system unstable. The local y_L axis is tangent to the orbit and directed in the direction of velocity, while the z_L axis is normal to the orbit and parallel to the angular momentum.

The satellite in LEO orbit has a nadir-pointing configuration and it is placed in an altitude range between 600 km and 900 km, with orbital inclinations between 0 and 90 degrees.

3.3 Reference Environment Models

The simulations include three different perturbative contributions: non-sphericity of the Earth's gravitational field, atmospheric friction and solar radiation pressure. A condition of maximum solar activity is assumed, thus considering the most critical condition for the analysis. The following models are integrated in the MATLAB code:

- NRLMSISE-00 Atmosphere Model [9];
- International Reference Ionosphere IRI-95 [10];
- International Geomagnetic Reference Field – IGRF [11].

NRLMSISE-00 is the international atmospheric reference model of the Earth from ground to the exobase, the lower boundary of the exosphere. It is an upgrade of the previous MSIS-86 and MSISE-90 models, because it includes new components, such as the anomalous oxygen, whose contribution influences the drag forces at high altitudes.
This model is used to define temperatures and densities of the atmosphere's components, and to help predict the orbital decay of satellites caused by atmospheric drag.

IRI-95 is the global recognized standard model for the Ionosphere and describes how the plasma parameters vary in Earth's Ionosphere. The Committee on Space Research (COSPAR) and the International Union of Radio Science (URSI) are two organizations involved in constantly improving the IRI model. It provides information about monthly values of electron temperature, electron density, ion temperature and composition, and other parameters at different altitudes, from 60 to 1500 km.

IGRF defines Earth's internal magnetic field through a set of spherical harmonic coefficients, from epochs 1900 A.D. to the present. It is produced and developed by the International Association of Geomagnetism and Aeronomy (IAGA), which revises the model every five years because of continuous changes of the Earth's core field. The IGRF has been derived from satellites observations, ground observatories and magnetic surveys, and is used for monitoring Earth's magnetic field anomalies, space weather and electromagnetic induction.

3.4 Method for determining Collision Probability

There are several methods for computing the Probability of Collision. In the simulations, Levin's approach is used [12]. Considering Fig.12, the following assumptions are made for the calculation:

- the electrodynamic tether is aligned with the local vertical and stabilized by the gravity gradient;
- the debris object B has a spherical shape with a diameter $\lambda = 2$ m;
- the object and the satellite are propagated to the point of closest approach A;
- the distance of closest approach $z_0 > 0$ between B and the tether line is relatively small compared to the tether length L = 500 m;
- the standard deviations of the position error σ_k is larger than the size of B but smaller than L, i.e. $\lambda \ll \sigma_k \ll L$.

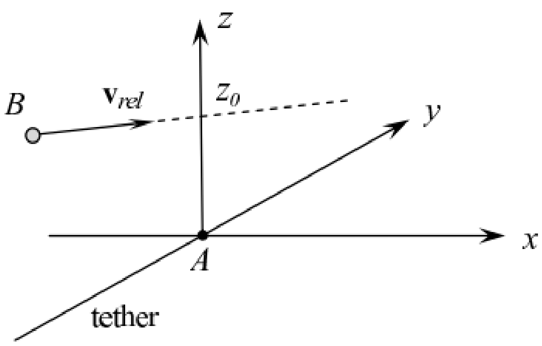


Figure 12: Conjunction with a tether.

It must also be taken into account that the closest approach between two compact objects is different if compared with the closest approach with a tether: the first is measured between the objects' centers of mass, while the second is relative to the tether line and can be far from the center of mass of the tether system.

A_{xyz} is the reference frame used to analyze the conjunction geometry, where x points in the radial direction, y is directed as the relative velocity of the tether, and z is perpendicular to the other two axis. Moreover, the plane A_{xy} is parallel to object B velocity, while the axis z points to the position of B at the time of the closest approach (TCA).

The three-dimensional probability density of the combined position error at TCA is defined by the function:

$$f = f(x, y, z - z_0)$$

As the conjunction is nearly linear, the error distribution is assumed to vary as:

$$f = f(x - v_{rx}(t - t_0), y - v_{ry}(t - t_0), z - z_0)$$

where v_{rx} and v_{ry} are the components of object B's relative velocity $\boldsymbol{v_{rel}}$.

From the last assumption discussed before, the probability of collision P_c (or PoC) can therefore be approximated as follows:

$$P_c \approx \lambda \int_{-\infty}^{\infty} \int_{-\infty}^{\infty} f(x, y, z_0)\, dx dy$$

When the probability density can be represented as

$$f = f_1(x, y)\, f_2(z - z_0)$$

the collision probability formula is simplified into the following expression:

$$P_c \approx \lambda f_2(z - z_0),$$

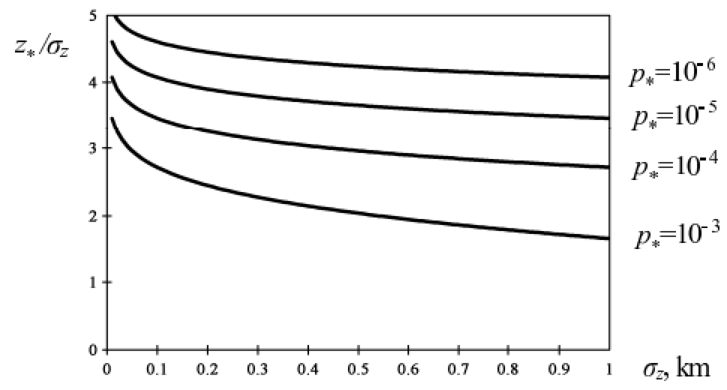
$$P_c \approx \frac{\lambda}{\sigma_z \sqrt{2\pi}} \exp\left(-\frac{z_0^2}{2\sigma_z^2}\right)$$

for a Gaussian distribution f_2, with σ_z corresponding to the standard deviation of the position error in the z-direction.

From the last formula it can be seen that the probability of collision does not depend on the tether length L, but is only a function of the size of the object, the distance of closest approach, and the relative position error.

The following graphs (Fig.13) show the relationship between different P_c* threshold value (PoC*) and the threshold distance z* for three typical size λ of the object B:

a)



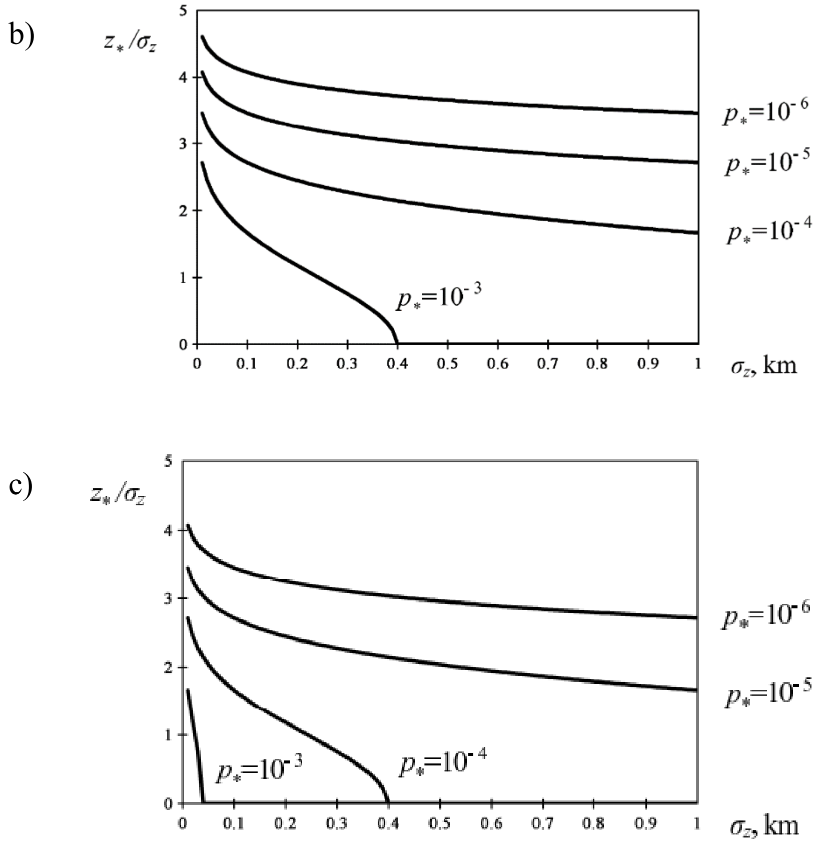


Figure 13: Threshold approach distances for a) λ = 10 m, b) λ = 1 m, c) λ = 0.1 m [13].

The ratio between z* and σ_z is calculated as

$$\frac{z_*}{\sigma_z} = \begin{cases} \sqrt{2\ln(\sigma_*/\sigma_z)}, & \text{with } \sigma_z < \sigma_* \\ 0, & \text{with } \sigma_z \geqslant \sigma_* \end{cases}$$

where

$$\sigma_* = \frac{\lambda}{p_*\sqrt{2\pi}}$$

is the threshold standard deviation of the position error.

3.5 Covariance and error ellipsoid

Since it is not possible to know the position of the two bodies at risk of collision with extreme accuracy, statistical approaches are used: for each of them a probability ellipsoid is defined, i.e. a region of space within which the object is located certainly. These ellipsoids are propagated over time by simulating how the uncertainty about the position of the two bodies will evolve, up to the moment of their closest approach. If at TCA the two probability ellipsoids overlap, this means that there is a possibility of collision (Fig.14).

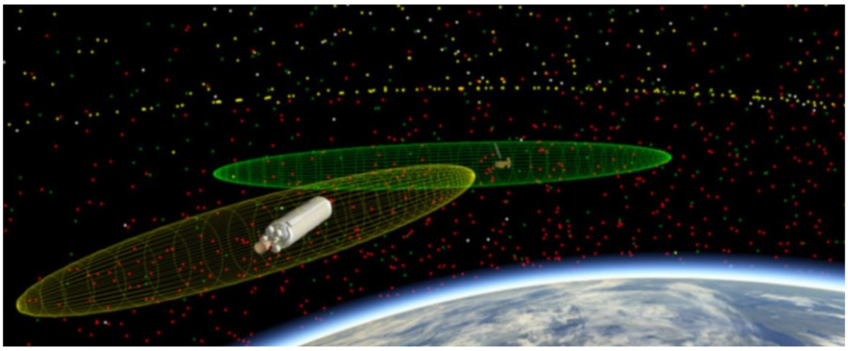


Figure 14: The overlap of the probability ellipsoids of two bodies indicates the probability of a collision (Credits: FreeFlyer).

When estimating the probability of collision, covariance linked with orbit parameters is therefore the key factor. It has been demonstrated that the combined error covariance matrix can be obtained by adding the error covariance matrices of the two objects (Fig.14), and is associated with a three-dimensional probability density function that represents the uncertainty in relative position between the colliding objects.

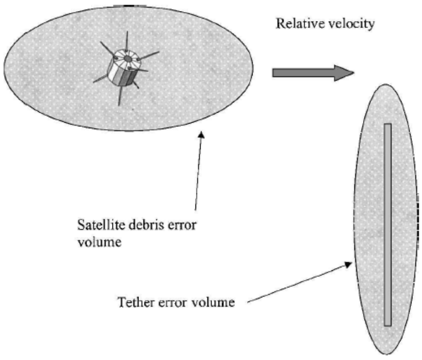


Figure 14: Notional satellite debris and tether encounter geometry with error volumes [16].

The three-dimensional Gaussian probability function is given by:

$$p(x) = \frac{1}{\sqrt{(2\pi)^3}\,\sigma_x\,\sigma_y\,\sigma_z}\, e^{-\frac{x^2}{2\sigma_x^2}-\frac{y^2}{2\sigma_y^2}-\frac{z^2}{2\sigma_z^2}}$$

where σ_x, σ_y and σ_z are the combined uncertainties.

3.6 Neural Networks

An artificial neural network is an interconnection of a group of nodes called neurons in a layered structure. Its functioning is inspired by how neurons signal to each other in the human brain: every node of each layer processes the external signals received and transmits the result to the subsequent nodes, so that the outputs of all nodes in a previous layer become inputs of the next one. Neural networks have many applications, such as recognizing patterns, classifying data and making predictions on future events. In general, they are used for modelling non-linear data structures and can simulate complex relationships between inputs and outputs which conventional algorithms cannot represent. This capability of neural networks allow them to be a convenient candidate. The network consists of at least three layers: the input layer, one or more hidden layers, and the output layer (Fig.15).

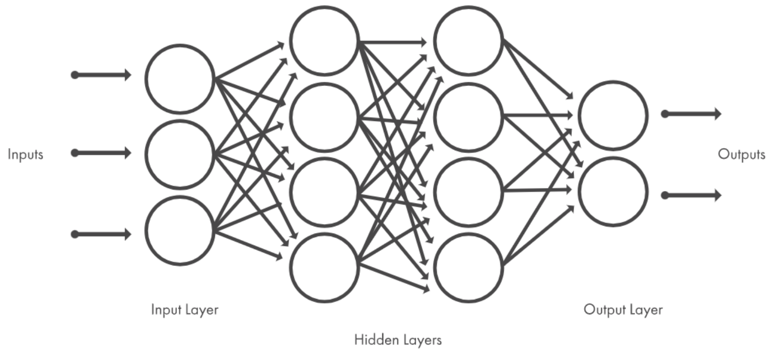


Figure 15: Typical neural network architecture (Credits: MathWorks).

Generally, neural networks involves mathematical operations that calculate a non-linear weighted sum of the inputs:

$$f(x) = k\left(\sum_i w_i g_i(x)\right)$$

where $g_i(x)$ is the vector of the input functions, w_i the weights, and $f(x)$ the output function.

Each node is associated with adjustable weights and the neural network learns (or "is trained") by adjusting these weights to minimize the error and obtain better results. In other words, neural networks elaborate data, containing a known input and a target output, to determine the difference between the prediction (i.e. the output processed by the network) and the target output. This difference represent the error which is progressively reduced adjusting the weights associated to the input, until the neural network produces an output increasingly similar to the target.

MATLAB implements neural networks through the Deep Learning Toolbox (DLT). For this thesis work, the Neural Net Pattern Recognition app available in the DLT is used to validate the model and forecast new results. In particular, it helps solve pattern recognition and data classification problems using a two-layer feed-forward network with sigmoid output neurons.

The structure of the neural network architecture inside the pattern recognition app is shown in Fig.16:

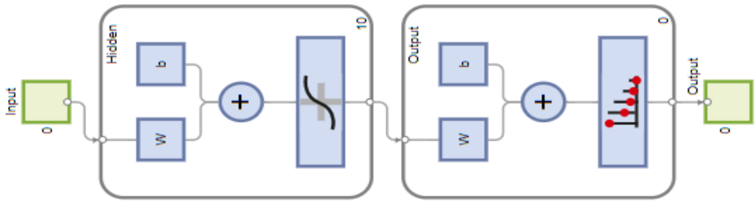


Figure 16: Diagram of a neural network Architecture.

After importing the data from MATLAB Workspace, they are split into training, validation and test sets, and a neural network is defined and trained. Using the *plotconfusion* MATLAB function, it is possible to conveniently visualize the performance of the implemented algorithm. A confusion matrix of predictions will be generated, showing the average accuracy in the lower right cell (Fig.17):

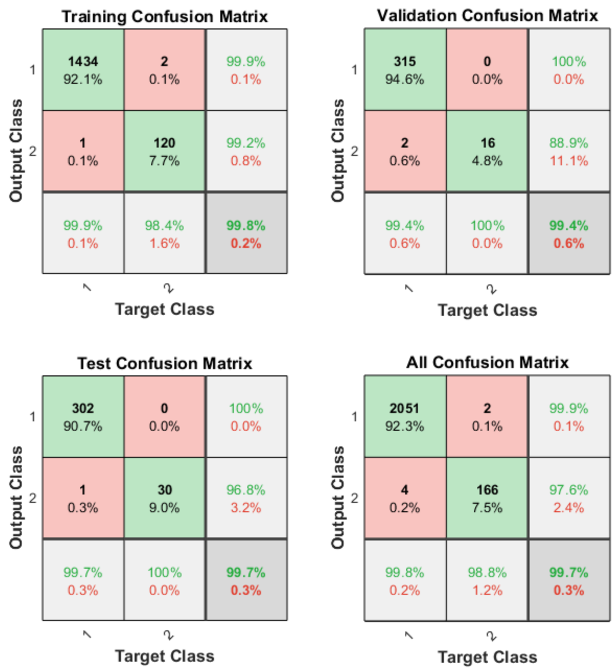


Figure 17: Example of Confusion Matrix showing individual classes accuracy and average accuracy.

The rows of the matrix represent the predicted class (Output Class) and the columns the true class (Target Class). Furthermore, diagonal cells (in green) show correct predictions, while off-diagonal cells (in red) correspond to wrong predictions. The number of observations and its percentage with respect to the total observations are displayed in each cell. Overall, the model in the example matrix has 99.7% accuracy, while 0.3% of the predictions are wrong.

Chapter 4 – MATLAB Simulations and Results

In this chapter the structure of the MATLAB Codes is described and the results of the simulations are discussed, with the presentation of the graphs of Probability of Collision as a function of altitude and orbital inclination. The last part of this section includes completing and generalizing the model through neural networks, which are used to make predictions about the results not obtained from simulations. After choosing some of the most relevant altitudes and inclinations, the simulations with the orbital propagator and the calculation of probabilities are carried out again to compare the results obtained with those predicted by the neural network and validate the model.

4.1 MATLAB Code Structure

The main idea while writing MATLAB codes and algorithms was to create a model as general as possible to evaluate different possible conjunction scenarios for an electrodynamic tether system, and then apply it to some realistic cases by simulating a conjunction with real debris currently in orbit.

Before computing the probability of collision, FLEXSIM orbit propagator was used to create a series of orbits traveled by the satellite at various altitudes and inclinations.
Choosing a five days time interval for every simulation, a conjunction is assumed to occur at the end of each simulation. Sensitivity analysis were conducted with the goal to evaluate how long before the potential impact the current must be turned off to reach a probability of conjunction lower than a threshold value (PoC* = 10^{-6}), and determine the minimum time in which this conjunction can be avoided. The Collision Avoidance maneuver and the orbital correction trajectory are a consequence of the instant the current is switched off.

The simulations consist of one set where the current is maintained always on, and other sets where it is switched off up to 24 hours before the expected Conjunction with a 3-hours step. The altitudes vary in a range from 600 km to 900 km, using a 25-km step, while the inclinations increases from 0 to 90 degrees with a 5-degrees step. A total of 2223 simulations were launched.

4.1.1 Computing the orbits of the Tether

Through the "user_input" file, the solver receives input data in order to generate the different orbits. More in details, it contains the initial orbital elements of the satellite, whose orbit is assumed to be circular, the tether parameters, the epoch time of the simulation, its duration and what time the current is switched off.

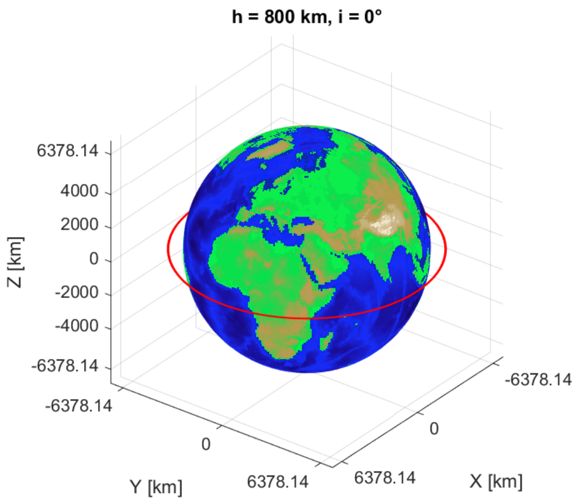
The tether parameters include:

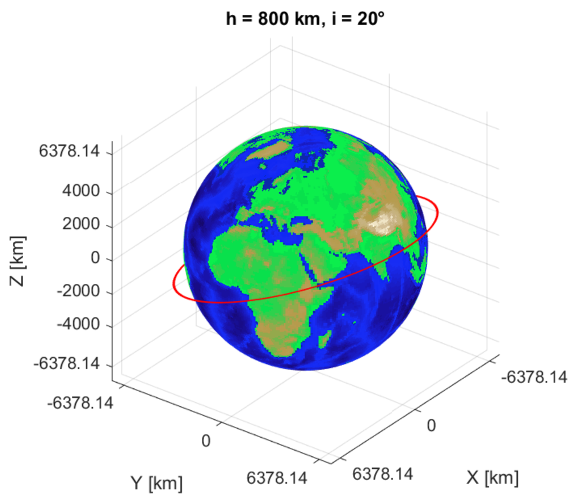
- the masses m_1 of the main spacecraft and m_2 of the deorbiting kit;
- the length and thickness of the conductive part of the wire;
- the length and thickness of the non-conductive part of the wire;
- the maximum value I_{max} of the current that can be generated independently of the external magnetic field;
- the cathode voltage, which allows electrons in the current to return to the external plasma.

The "Run_simulation" file is used to launch the simulations. After that, the simulator creates a DATA folder with binary files that are processed by the "MatlabPostProcessingData" file, while the results are saved in "DataComplete.mat". The code also includes a BETS file that corresponds to the executable and .asc and .DAT files that are necessary for the environmental models (magnetic field, atmosphere, etc.).

An additional Matlab script called "external loop" has been added to automate the simulations. In particular, this allows to start consecutively simulations of orbits with different altitudes.

The orbits obtained as a result of the simulations are presented in the following figures (Fig.18). By way of example, only a few orbits are shown, with an altitude of 800 km and different inclinations:





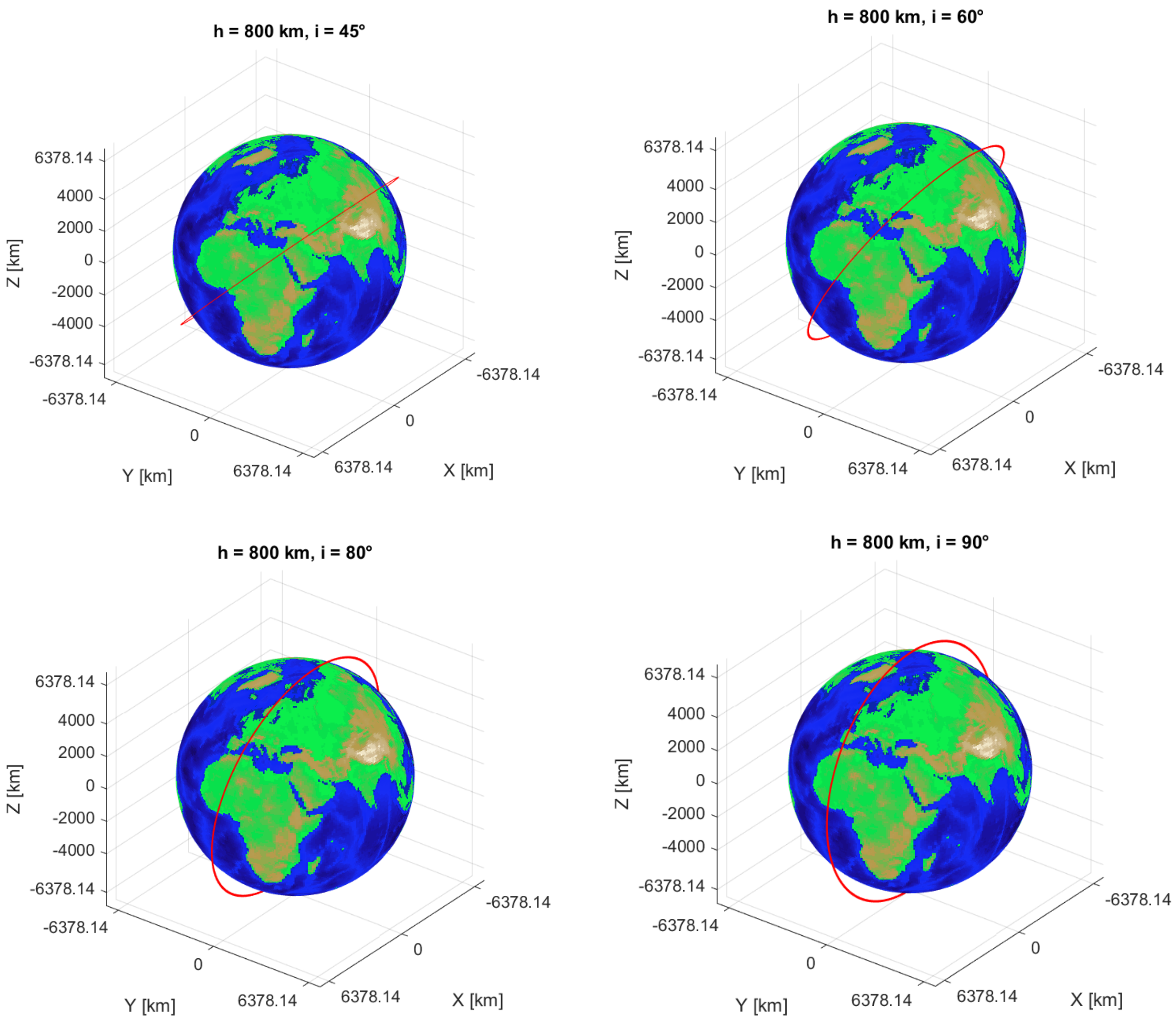


Figure 18: Examples of orbits obtained by the simulations.

4.1.2 Uncertainties Analysis and Error Ellipsoid

Monte Carlo simulations were conducted to evaluate the contribution of the uncertainties in the measurements and compute the covariance matrix, using the multivariate distribution function. The lower the collision risk, the larger the number of Monte Carlo runs required for confidence in the solution.

The uncertainty on the position of the debris is added to the one on the satellite to obtain the combined uncertainties.

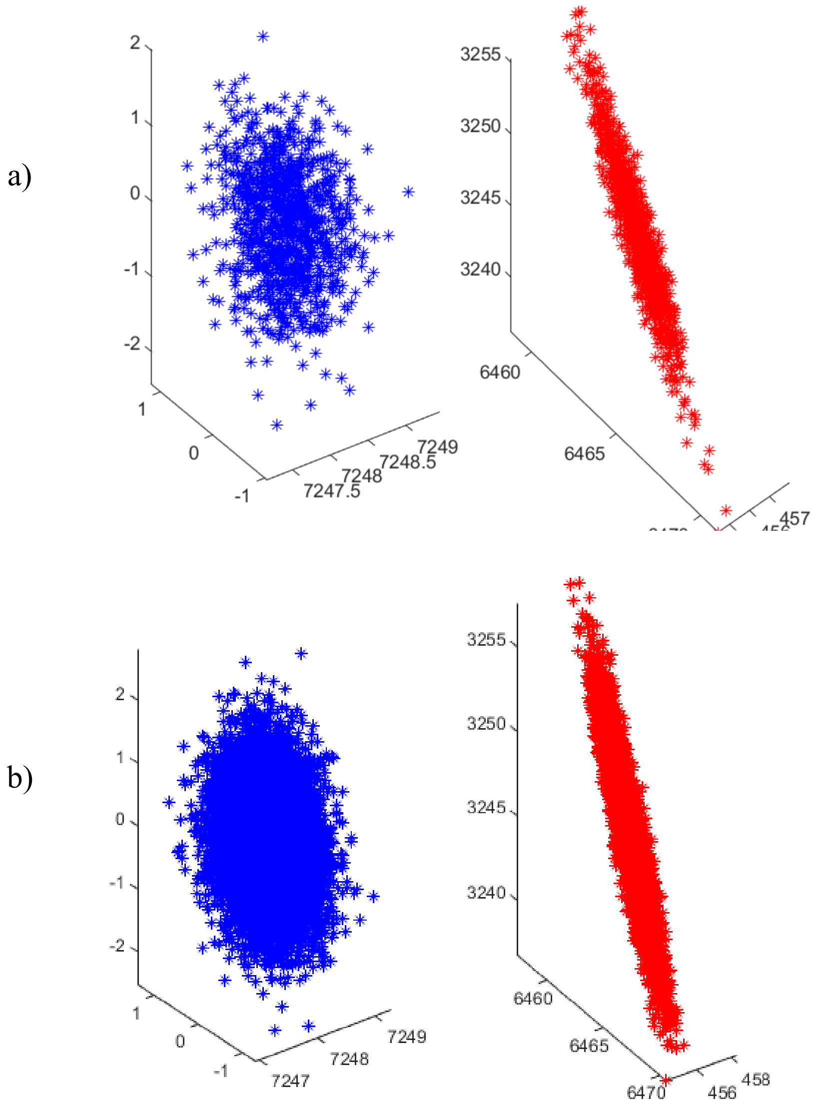
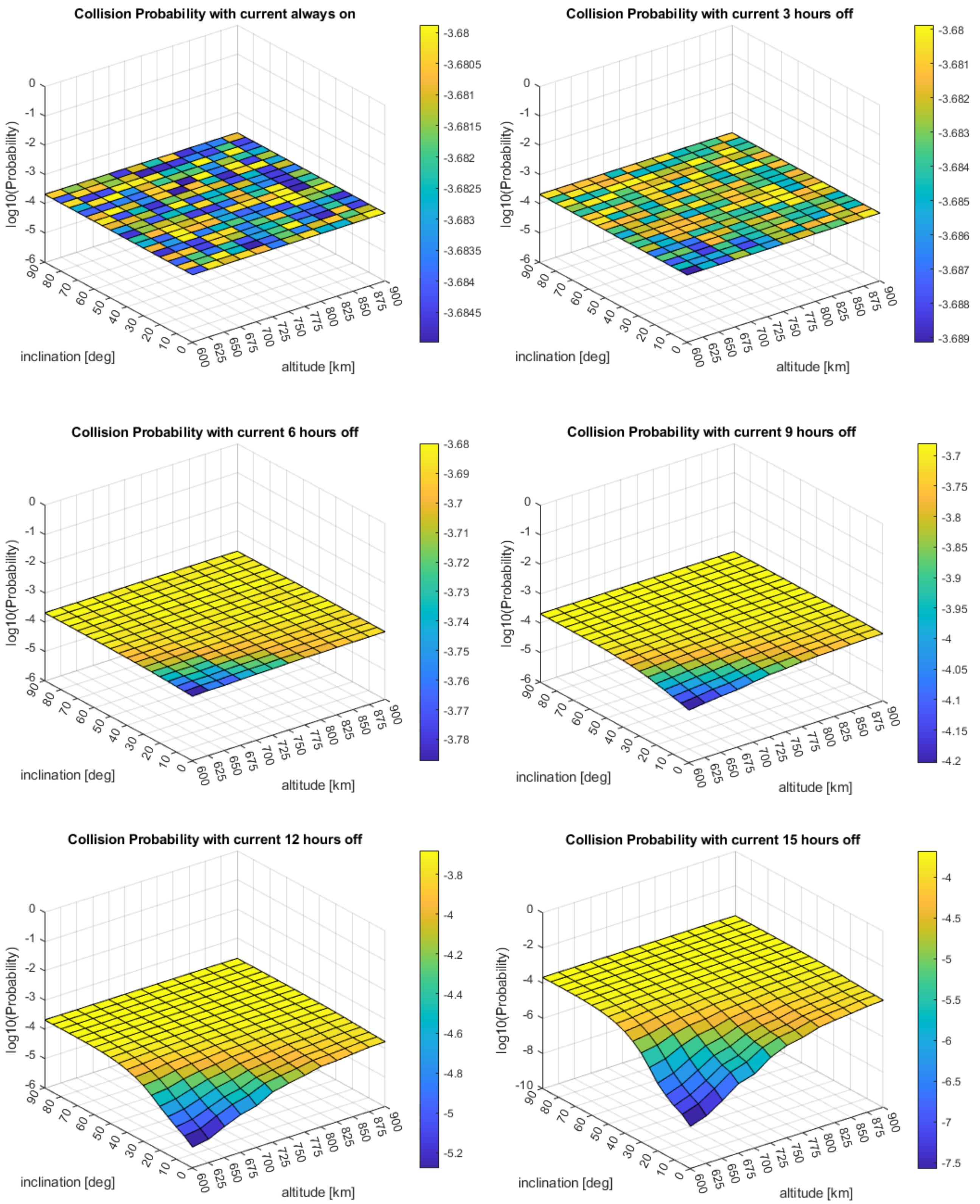


Figure 19: Initial (blue) and final (red) position distribution using a) 1000 and b) 10000 data. The axis unit is km.

The corresponding ellipsoid on the left is very flattened, while the one on the right, after orbital propagation, is more elongated. The points stretch along the direction of the velocity, so transversely the two sigma corresponding to the uncertainties are smaller.

4.2 Results of the Simulations

The results obtained in the simulations are shown in Figure 20:



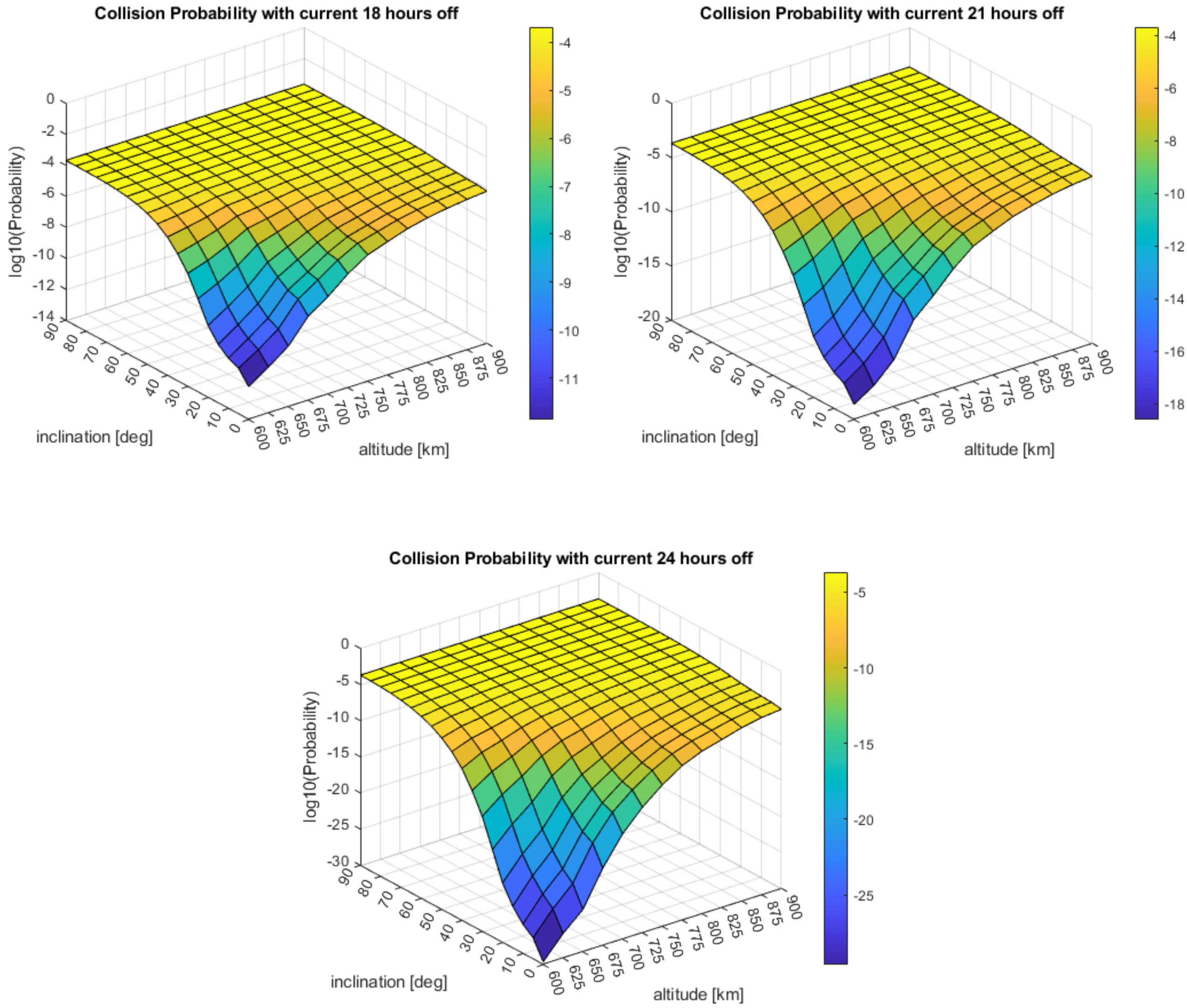
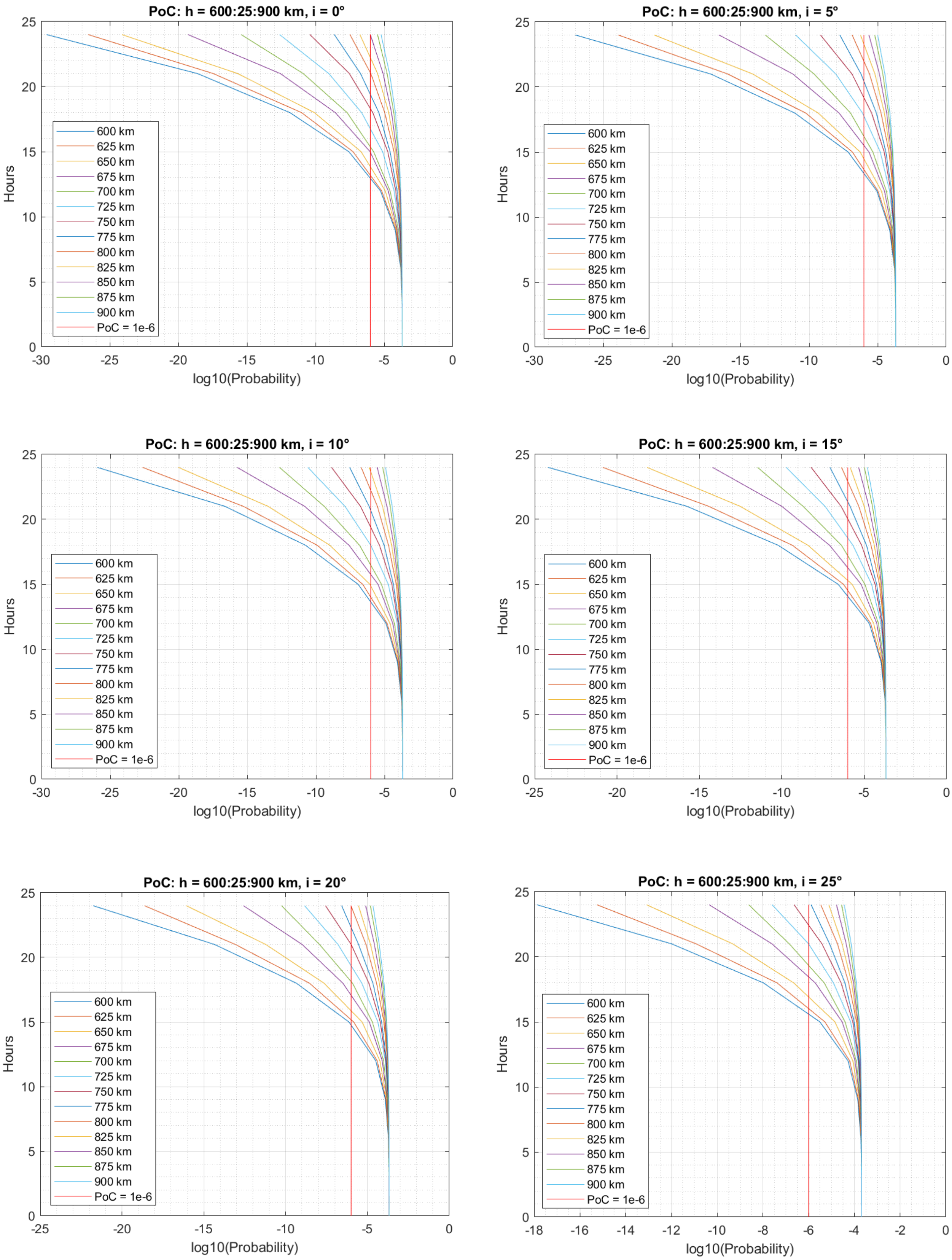


Figure 20: 3D graphs of the Probability in logarithmic scale as a function of inclination and altitude.

The graphs show the evolution of the probability in relation to different altitudes and orbital inclinations for nine cases under consideration: current switched off 3 to 24 hours before the expected conjunction. Each graph is in three dimensions, with the altitude in km on the abscissa axis, the inclination in degrees on the ordinates, and the probability, represented in logarithmic scale, on the z axis. It can be seen that for higher altitudes the generated current is very low and this causes a minor variation in terms of probability. With regards to the inclinations, on the other hand, for values close to 90°, and therefore for polar orbits, the probability remains close to the maximum value, while for equatorial orbits it rapidly decreases. The sooner the current is switched off, the lower the calculated probability is.

The following 2D graphs (Fig.21) displays how the probability varies at different altitudes for a fixed inclination.



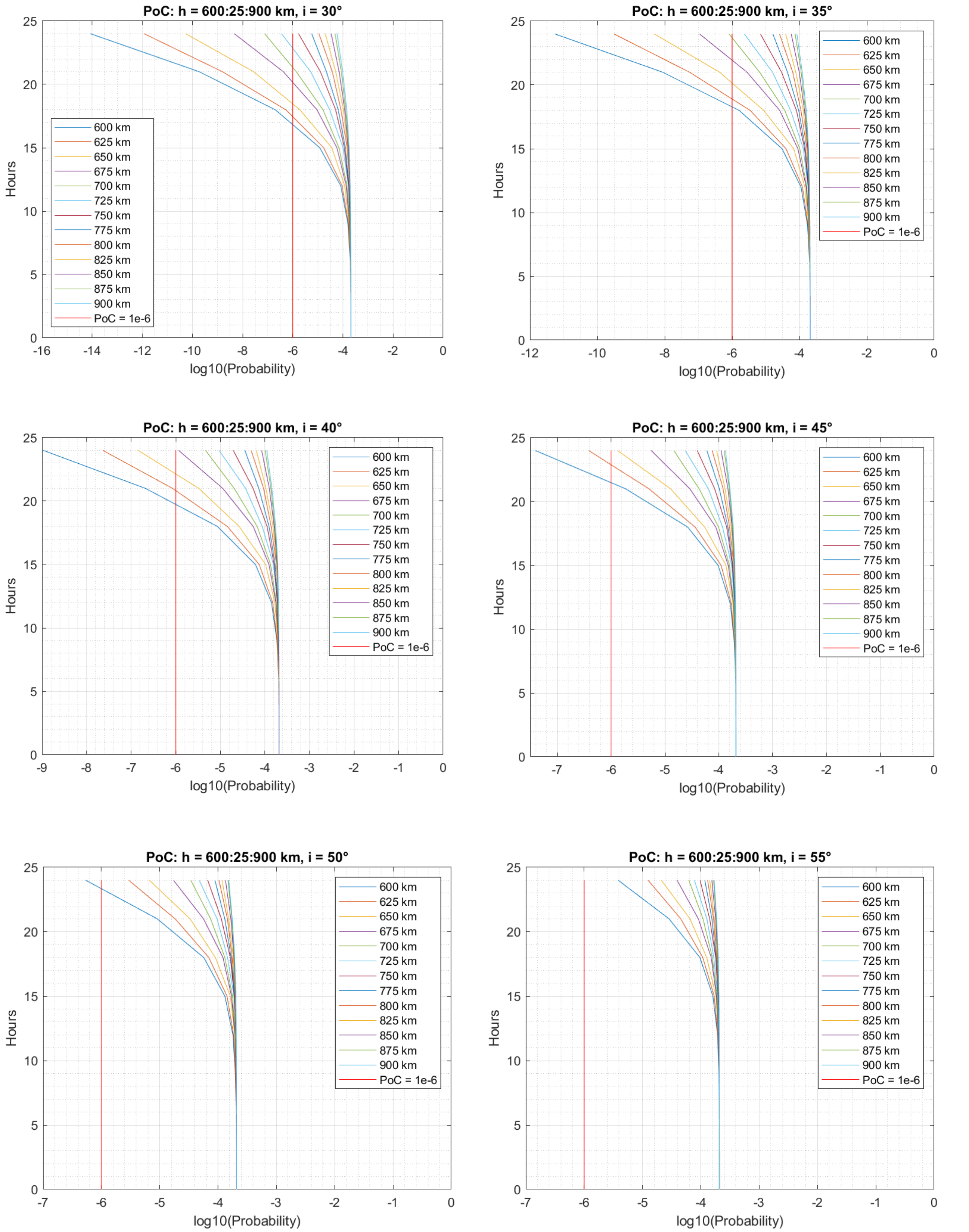
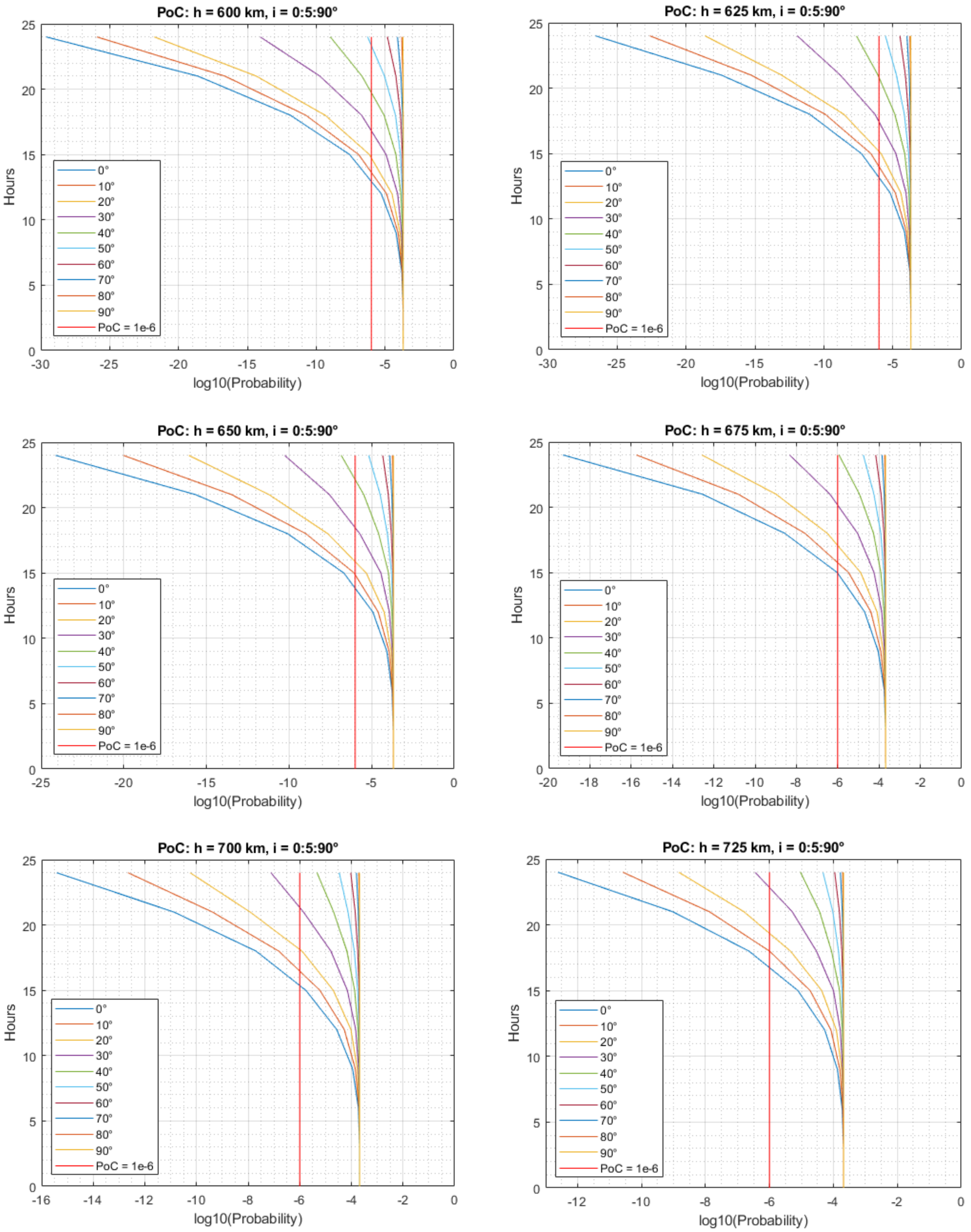
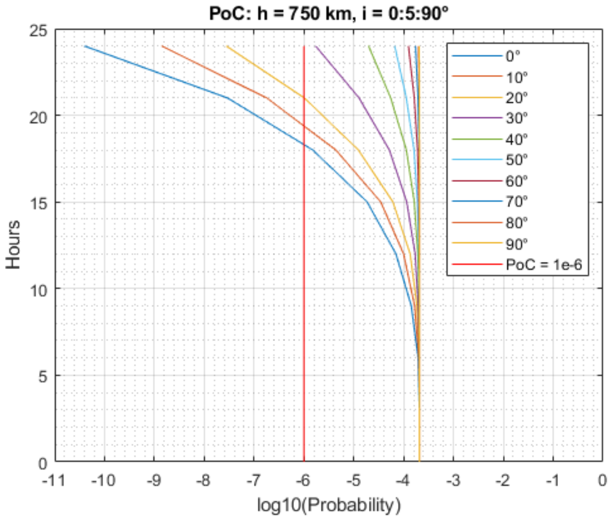


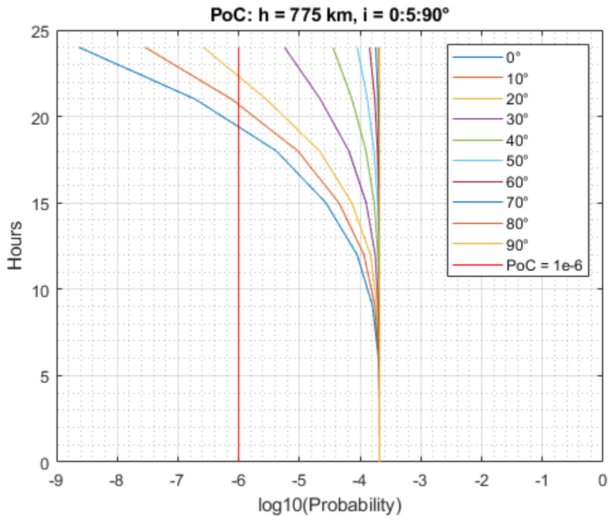
Figure 21: 2D graphs of the Probability in logarithmic scale as a function of altitude at fixed inclinations. The vertical red line identifies the probability threshold value.

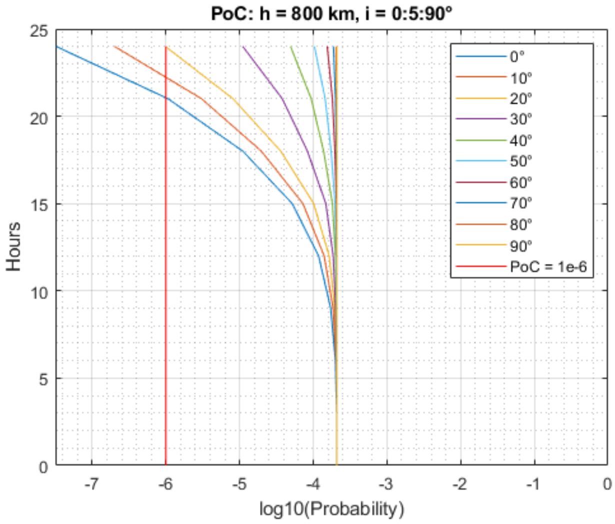
It can be observed that for high altitude and inclinations the probability of collision is higher than the permitted threshold value.

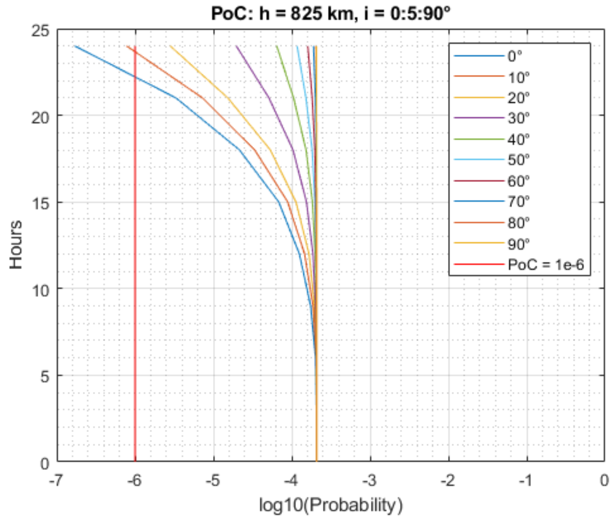
The following 2D graphs (Fig.22) shows how the probability varies at different inclinations, for a fixed altitude:

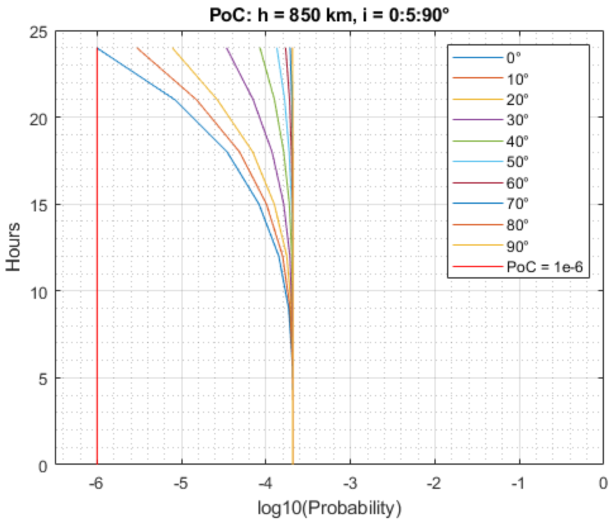


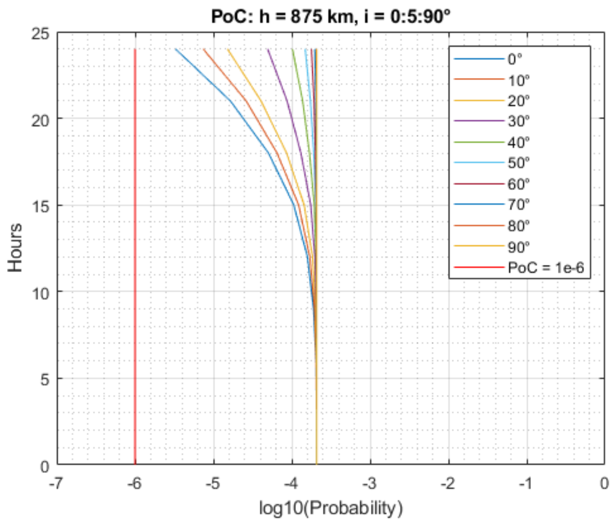

PoC: h = 750 km, i = 0:5:90°
Hours
log10(Probability)
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
PoC = 1e-6


PoC: h = 775 km, i = 0:5:90°
Hours
log10(Probability)
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
PoC = 1e-6


PoC: h = 800 km, i = 0:5:90°
Hours
log10(Probability)
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
PoC = 1e-6


PoC: h = 825 km, i = 0:5:90°
Hours
log10(Probability)
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
PoC = 1e-6


PoC: h = 850 km, i = 0:5:90°
Hours
log10(Probability)
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
PoC = 1e-6


PoC: h = 875 km, i = 0:5:90°
Hours
log10(Probability)
0°
10°
20°
30°
40°
50°
60°
70°
80°
90°
PoC = 1e-6

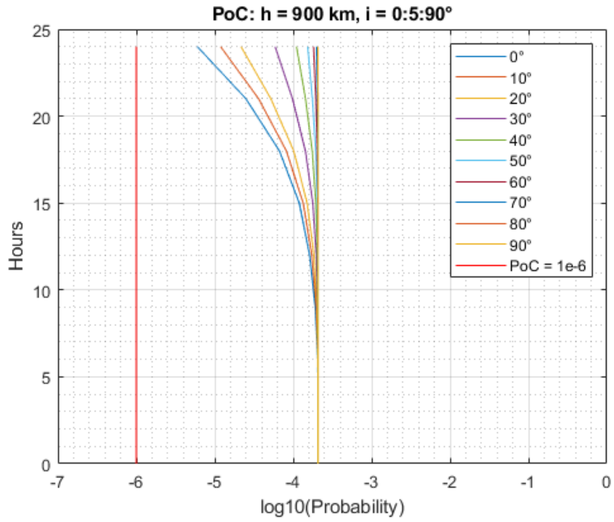
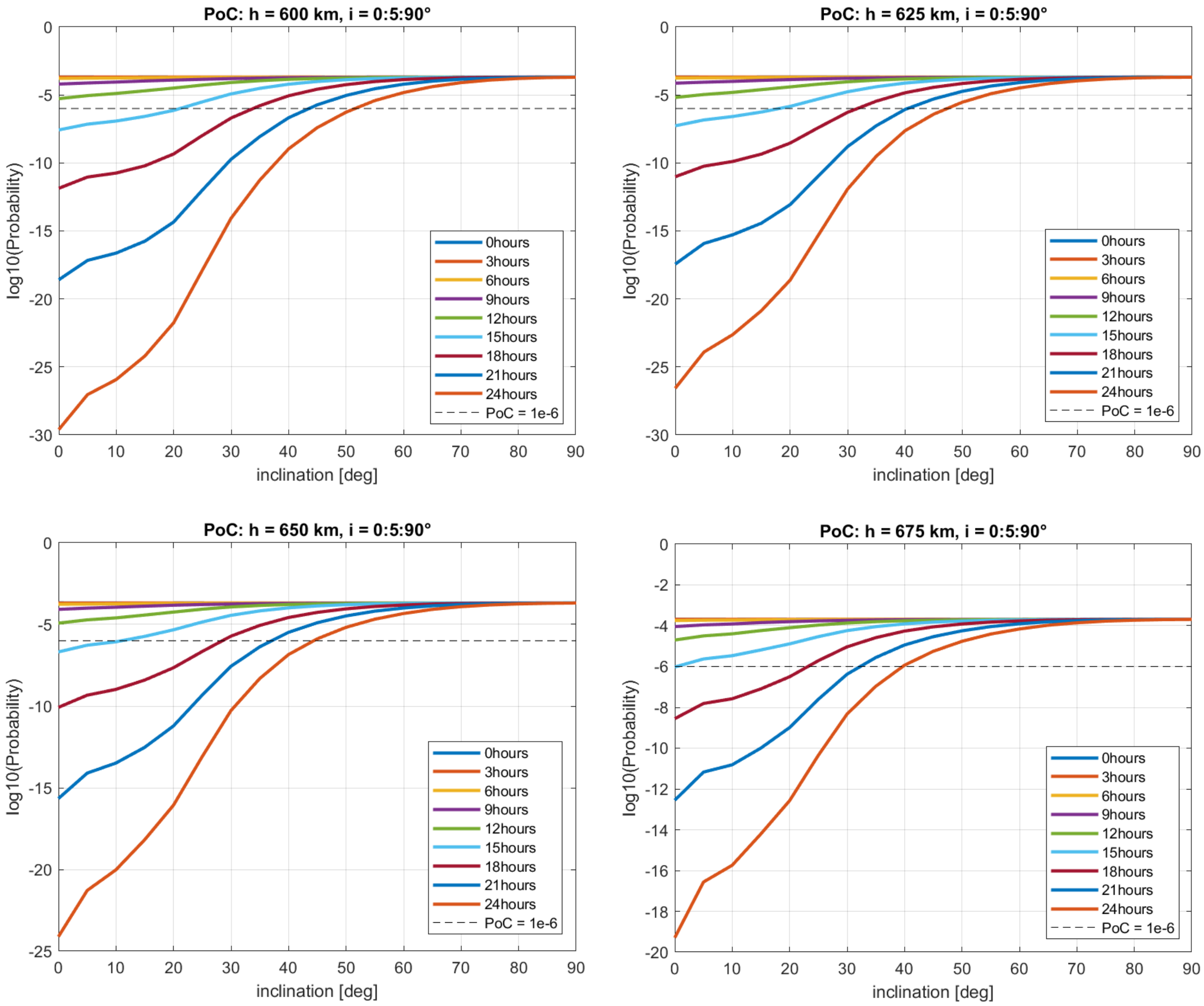
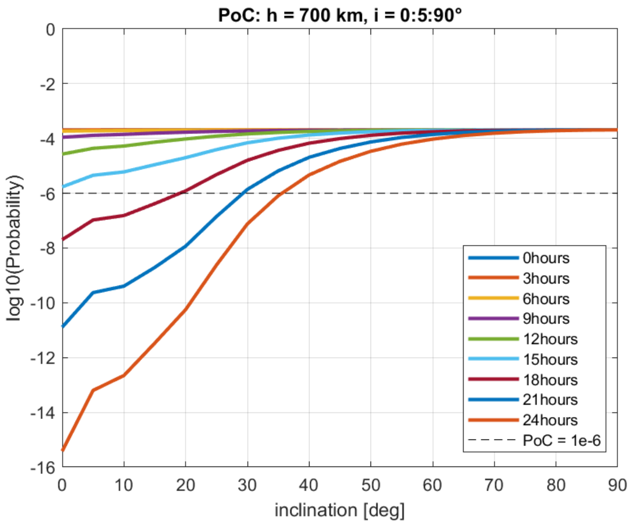


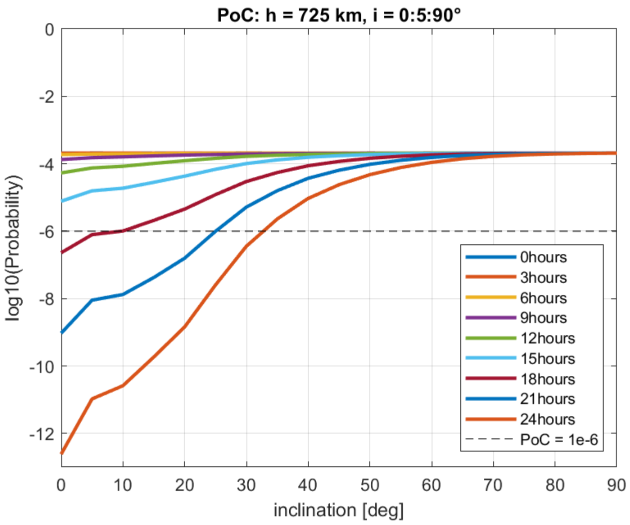
Figure 22: 2D graphs of the Probability in logarithmic scale as a function of inclination at fixed altitudes. The vertical red line identifies the probability threshold value.

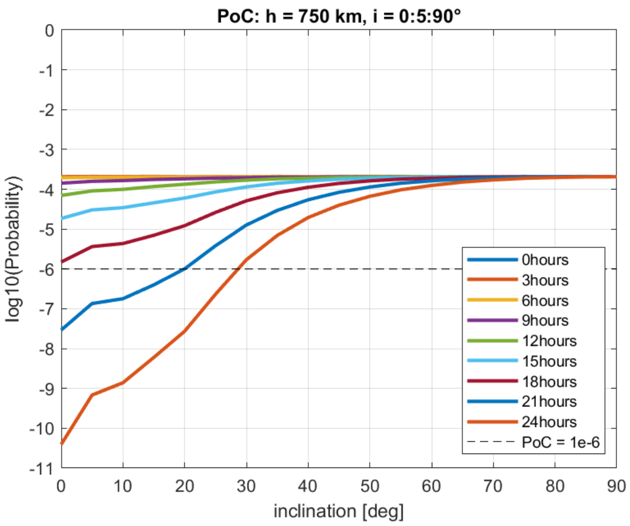
In the same way as before, higher altitudes and inclination implies a probability value over the permitted threshold.

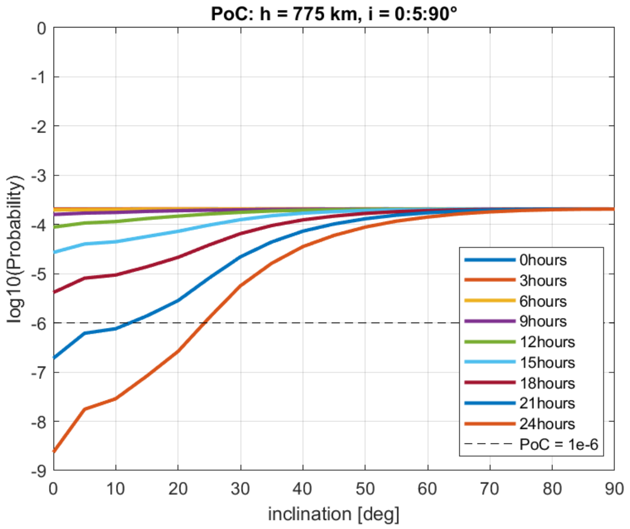
Finally, the last graphs (Fig.23) indicate how long before the impact the current should be switched off in order for the probability to be lower than PoC*.

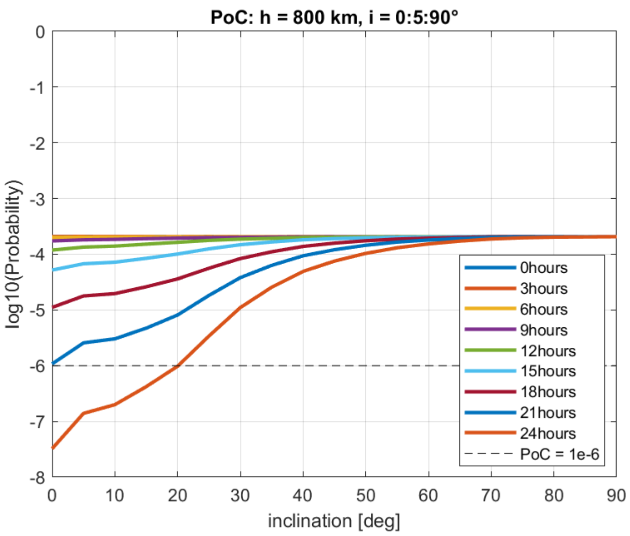


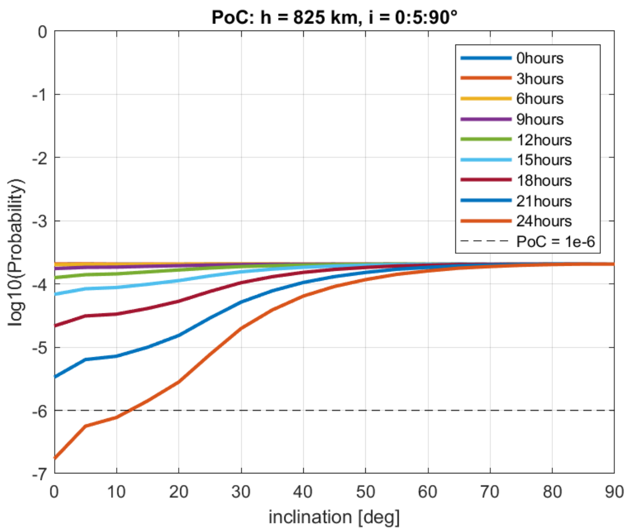

PoC: h = 700 km, i = 0:5:90°
log10(Probability)
inclination [deg]
0hours
3hours
6hours
9hours
12hours
15hours
18hours
21hours
24hours
PoC = 1e-6


PoC: h = 725 km, i = 0:5:90°
log10(Probability)
inclination [deg]
0hours
3hours
6hours
9hours
12hours
15hours
18hours
21hours
24hours
PoC = 1e-6


PoC: h = 750 km, i = 0:5:90°
log10(Probability)
inclination [deg]
0hours
3hours
6hours
9hours
12hours
15hours
18hours
21hours
24hours
PoC = 1e-6


PoC: h = 775 km, i = 0:5:90°
log10(Probability)
inclination [deg]
0hours
3hours
6hours
9hours
12hours
15hours
18hours
21hours
24hours
PoC = 1e-6


PoC: h = 800 km, i = 0:5:90°
log10(Probability)
inclination [deg]
0hours
3hours
6hours
9hours
12hours
15hours
18hours
21hours
24hours
PoC = 1e-6


PoC: h = 825 km, i = 0:5:90°
log10(Probability)
inclination [deg]
0hours
3hours
6hours
9hours
12hours
15hours
18hours
21hours
24hours
PoC = 1e-6

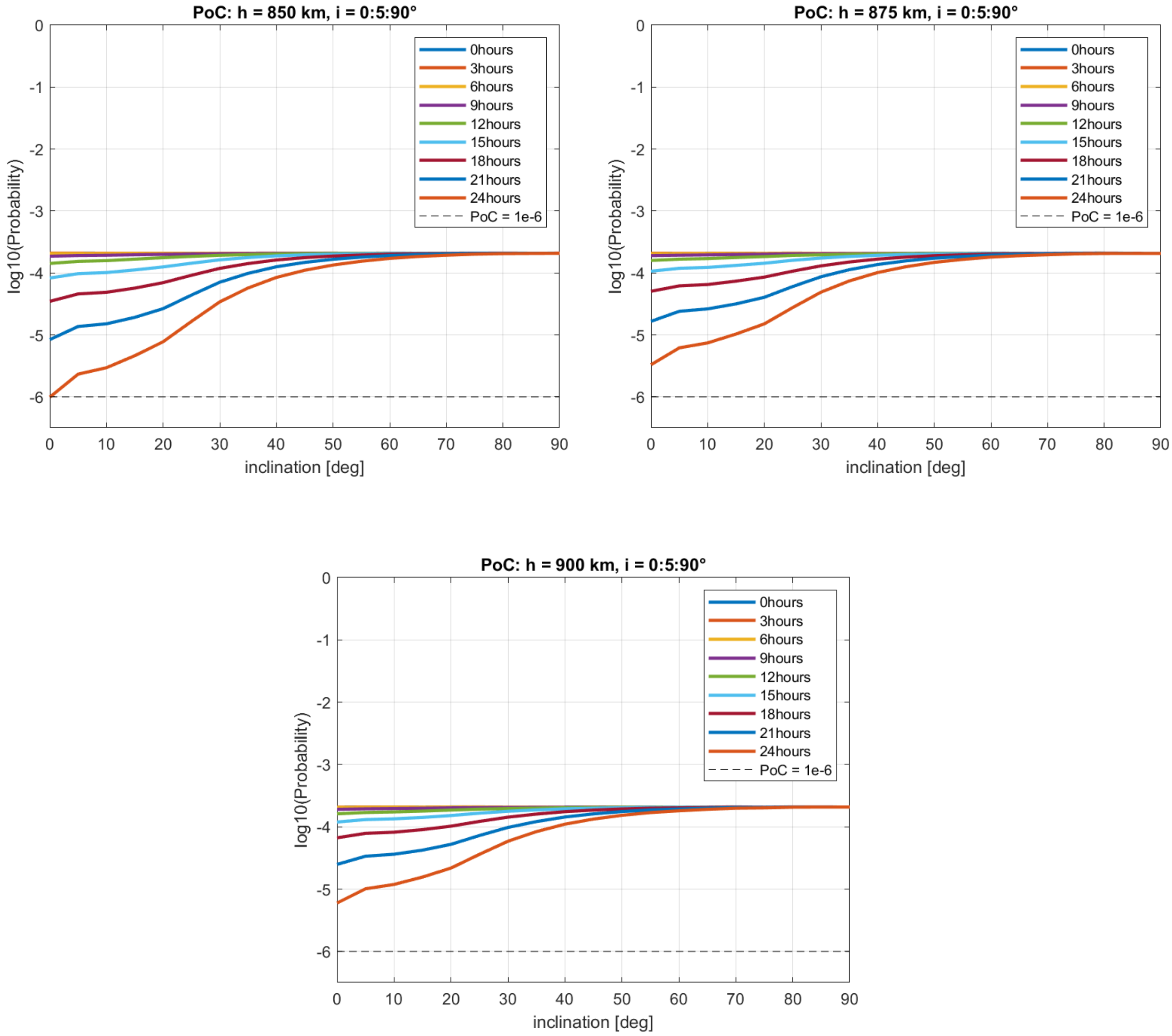


Figure 23: Probability values when the current is switched off 0 to 24 hours before the expected conjunction. The dash line represents the probability threshold value.

4.3 Neural Network Results

Estimations on the probability values can be made using a neural network model. In Table 1, for some selected altitude and inclination values, such predictions are presented, and the hours before the conjunction the current should be switched off in order to execute a safe maneuver are reported:

Altitude [km]	Inclination [deg]	Hours the current is off at PoC* (from graphs)	% with NN input 1hours after PoC*	% with NN input 0.5hours after PoC*	% with NN input at PoC*	% with NN input 0.5hours before PoC*	% with NN input 1hours before PoC*
600	0	13	71.27	96.26	99.61	99.96	99.99
600	20	15	9.14	55.48	92.63	98.99	99.83
600	45	21.5	9.50	31.67	64.00	85.52	94.49
600	60	More than 24	100% PoC < PoC* when current is switched off 32.2 hours before				
700	0	15.5	0.02	1.78	13.24	61.46	95.13
700	20	18	0.02	1.76	13.57	57.57	91.53
700	45	More than 24	100% PoC < PoC* when current is switched off 30.5 hours before				
700	60	More than 24	100% PoC < PoC* when current is switched off 41 hours before				
800	0	21	0.01	0.06	27.41	83.73	98.67
800	20	24	0.01	0.09	36.31	74.02	92.85
800	45	More than 24	100% PoC < PoC* when current is switched off 43 hours before				
900	0	More than 24	100% PoC < PoC* when current is switched off 27.5 hours before				
900	20	More than 24	100% PoC < PoC* when current is switched off 34.1 hours before				
900	45	More than 24	100% PoC < PoC* when current is switched off 46 hours before				

Table 1: Percentage predicted by the neural model for which the probability is less than the threshold value.

In the confusion matrix, samples are divided into two classes, one with a probability greater than the threshold value, and the other with PoC < PoC*.
Looking at the results in the table, it is evident that the percentages predicted increase if the maneuver is performed with a larger time window.

Table 2 compares the data obtained with the neural model with those from the orbit simulator, showing good agreement between the two of them.

Altitude [km]	**Inclination [deg]**	**Confusion Matrix [hours predicted]**	**FLEXSIM [hours predicted]**
600	0	14.1	13
600	20	16.9	15
600	45	23.4	21
600	60	32.2	30
700	0	16.8	15.5
700	20	19.6	18
700	45	30.5	30
700	60	41	42
800	0	22	21
800	20	25.6	24
800	45	43	45
900	0	27.5	27
900	20	34.1	33
900	45	46	48

Table 2: Confusion matrix results against FLEXSIM results.

The third and fourth columns indicate the hours before Conjunction the Manuever should be executed to avoid the impact.

It was first applied the neural network model, trying to predict the cases in which the current is turned off more than a day before the conjunction. The calculation is then repeated with FLEXSIM, demonstrating that the results are compatible with each other.

4.4 Conjunction with real debris

In the final part of the simulations, some Conjunctions with real debris are studied. Four space debris were chosen for the analysis, and their characteristics are reported in the table below (Table 3):

	SL-8	**MOZ.5/SAFIR/ RUBIN 5/SL-8**	**IRIDIUM 28**	**ATLAS CENTAUR**
Object type	Rocket body	Payload	Payload	Rocket Body
Altitude [km]	766.55	691.16	772.31	655.19
Inclination [deg]	74.05	98.15	86.39	35.00
RCS [m^2]	4.35	6.834	4.08	13.11
Miss distance [km]	0.185	0.093	0.205	0.357
Impact_time	1 d 6 hours	1 d 1 hour	1 d 1h	1 d 17 hours

Table 3: Main features of real space debris

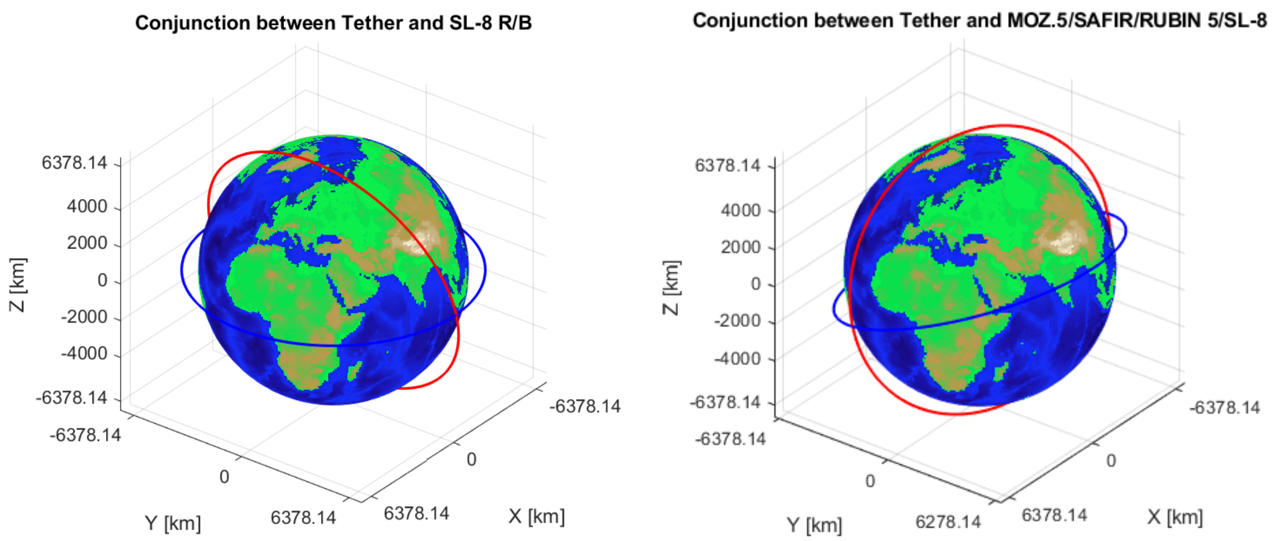
The last three rows represent the radar cross section (RCS), the distance tether-debris (or miss distance), and the impact time, which corresponds to the time from the beginning of the simulation to the impact.

In table 4 the data of the tether was chosen to make it impact with the debris:

	Tether1	**Tether2**	**Tether3**	**Tether4**
Altitude [km]	766.35	702.90	773.81	636.19
Inclination [deg]	0	20	65	30

Table 4: Altitude and inclination of the tether in four different cases

Orbits of the tether and the debris are represented in Fig.24:



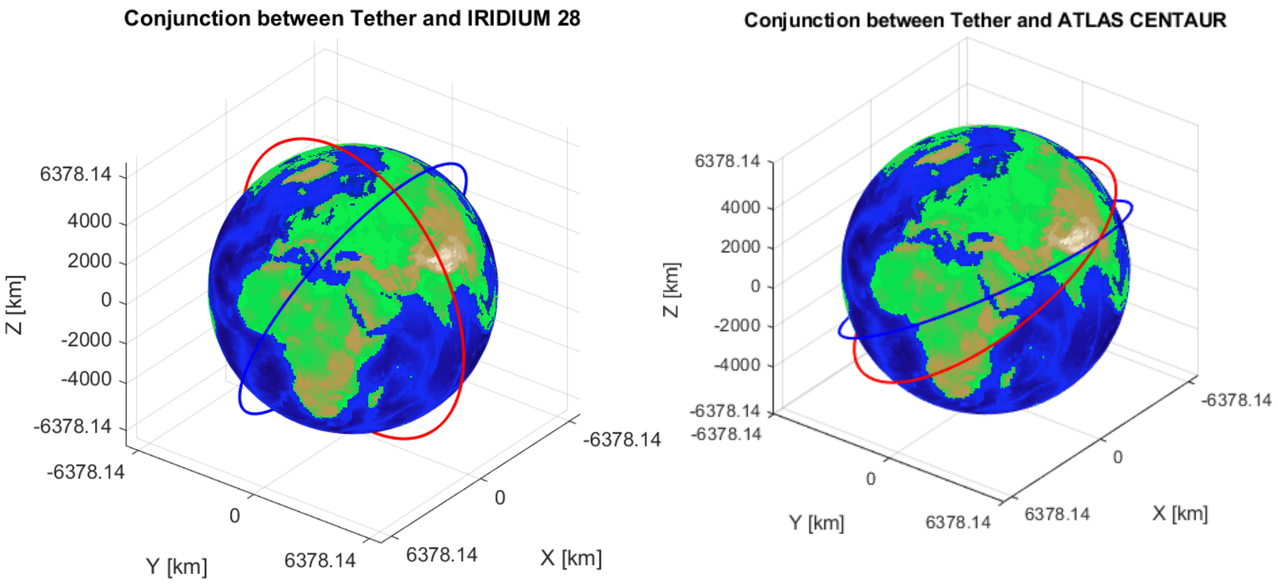


Figure 24: Conjunctions between tether (blue) and four different space debris (red).

After determining through the neural networks the temporal tolerance to perform the maneuver, simulations in FLEXSIM were repeated to test whether that tolerance produces a probability lower than the expected threshold (see Table 5):

	Conjunction 1	**Conjunction 2**	**Conjunction 4**
NN time tolerance [hours]	22	21.6	23
FLEXSIM Probability	$1.33 \cdot 10^{-125}$	$4.59 \cdot 10^{-57}$	$3.67 \cdot 10^{-9}$

Table 5: Probability calculated by the simulations on FLEXSIM starting from the time tolerance predicted through the neural network.

Three out of four cases investigated have demonstrated the validity of the model.
One of them (the third object) did not produce solutions in the range 0-24h. This is attributable to a combination of the altitude and inclination values of the tether's orbit and that of the impacting object. Another reason can be linked to the small size of the tether (only 500 m long) compared to its mass (500 kg).

Chapter 5 – Conclusions and Future Developments

The final Chapter includes the conclusions and explains which further investigations should be conducted to improve the model.
Lastly, it presents the future developments in Conjunction Analysis and what is the room for improvement in this field.

5.1 Conclusions

This Master's thesis had three main goals: to develop a valuable model to assess the probability of collision between an electrodynamic tether and other space objects, to determine how much before the impact a Collision Avoidance Maneuver must be executed, to demonstrate the validity of the model by applying it to real scenarios.
The model was implemented in the MATLAB environment exploiting a simulation tool developed at the University of Padova to propagate the orbits of the satellite and analyze different Conjunction events. Once defined the state vector of the tether system, the miss distance between the spacecraft and an hypothetic dangerous object was calculated to subsequently compute the probability of collision. The calculation of the probability includes errors due to the uncertainties of the simulated tether system and that of the object's positions.
PoC is fundamental in the risk assessment process and it is considered as reference for computing an eventual Collision Avoidance Maneuver. Moreover, it allows to determine the minimum time to execute this Maneuver and, consequently, improve its accuracy.
A possible CAM for tethered systems is based on the principle of switching on and off the current inside the tether. Different sets of simulations have been generated to evaluate this process, considering different altitude and orbit inclinations. In particular, it was observed that high altitudes and inclinations requires the Maneuver to be performed within a larger time window.
In order to verify the reliability of the developed model, a realistic representation of the Conjunction scenario was provided by simulating the potential impact of the satellite with real orbiting debris. Through neural networks, it was possible to make predictions on conjunction events by calculating in what percentage the probability of collision was lower than the threshold value, thus validating the model developed. A good compatibility was observed between the outcome of the forecasts and the results of the simulations.

5.2 Future Work

Whether it is assessing the risk of impact between an active satellite and space debris, evaluating the reentry into the Earth's atmosphere of a satellite at the end of its operational life, planning an avoidance action or supporting the tracking activities of space objects, orbital prediction plays a fundamental role. However, accurately determining the position that an object will occupy along its path in orbit is not a simple challenge, because of the presence of different perturbative contributions that influence the motion of the satellite.

The current model developed in this Master's thesis has led to good results in achieving its goal to predict the probability of collision between two space objects, and could be used as a tool for space debris remediation strategies involving electrodynamic tethers.
Despite this, it still has a good room for improvement. Further investigations may be conducted to evaluate the effect of perturbative terms on the analysis and the uncertainty on the atmospheric models. An interesting development could be studying more accurately the impact of the contribution of solar activity, in order to determine how much it influences the position of the satellite.

Some simulations have already been generated to study this phenomenon.
In the time interval 2000 - 2009, for a simulation of 5 days with current always on (0 hours), or off 24 hours before the expected conjunction, the intensity of solar activity has been evaluated, obtaining the following result:

$$\left\|\left(r_{January2000\_0hours} - r_{January2009\_0hours}\right)\right\| = 10.81\ km$$

$$\left\|\left(r_{January2000\_24hours} - r_{January2009\_24hours}\right)\right\| = 10.34\ km$$

where the $\boldsymbol{r_{GM}}$ vectors represent the position of the satellite at the end of the simulation for an orbit with altitude equal to 870 km and inclination of 82 degrees, i.e. the one that has been identified to be one of the most critical orbits, according to a debris flux analysis performed with MASTER. In the considered 5-day interval, comparing the two final positions, it can be observed that solar activity has such an influence that at the end of the simulation the two objects are about 10 km distant. In other words, solar activity delays the satellite's position by ~10 km considering two different time intervals.

Taking into consideration a shorter time span, equal to 2 months, a less significant result is obtained:

$$\left\|\left(r_{\,January2003\_0hours} - r_{\,March2003\_0hours}\right)\right\| = 0.51\ km$$

$$\left\|\left(r_{\,January2003\_24hours} - r_{\,March2003\_24hours}\right)\right\| = 0.48\ km$$

With regards to the contribution related to the intensity of the atmospheric density, it is obtained that the satellite only changes its position by 5 m in the same time interval considered. It is therefore deduced that in the deorbiting phase using an electrodynamic tether system the contribution associated to the uncertainty related to solar activity is much more intense than atmospheric drag.

Further analysis can be conducted in this field to evaluate how the final position of the tethered system changes by considering solar activity and other perturbative contributions and thus evaluating the effects of uncertainty on atmospheric models.

5.3 Future developments in Conjunction Analysis

In the recent years, Space Situational Awareness and Space Traffic Management, which deal with the control and management of traffic in orbit, have become priorities in order to face the problem of space debris [14]. One of the main issues is that the control of space traffic is not regulated by internationally valid protocols and standards. Each satellite operator adopts their own criteria to assess the risks of a potential collision compared to the costs of an avoidance maneuver. Coordination between the various operators is therefore difficult or non-existent.

Another reason for the absence of common legislation is that each operator uses different data to make its assessments. These data are provided by government agencies or they are obtained by the same operators, who have no interest in making public such information they consider sensitive.

Surely, in order to define a common regulation valid at international level, it would be necessary for satellite operators to start sharing such data to facilitate coordination between the various players. Collective awareness and improvements in the relationships between the different parties involved are essential, with the aim of optimizing what is becoming an overcrowded space.

Another important aspect to take into account is that the Collision Avoidance process is completely manual and the procedures described are specific for each different situation. With the exponential increase in the number of satellites in orbit, it will be impossible to continue following this approach. The presence of automated mechanisms that autonomously process the ever-increasing number of possible collisions will become necessary. For this reason, the idea of exploiting Artificial Intelligence and Machine Learning is gaining ground, in order to automate the operations and decision-making processes involved in the Conjunction Analysis.
The use of artificial intelligence would speed up the processes of collecting data and risk assessment, from the initial warning of a potential conjunction to the final avoidance maneuver.

Space organizations should therefore invest in technologies that are able to process collision warnings and send commands to spacecraft entirely automatically, coordinating maneuvers with other operators to ensure the benefits of space resources and continue to guarantee access to space in the years to come.

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