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Launch Operations for Direct Ascent Trajectory and Re-entry vehicles

TESI DI LAUREA MAGISTRALE IN
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Abstract

The emergence of anti-satellite (ASAT) weapons and suborbital threats, and the proliferation of launch capabilities have elevated the importance of effective launch detection, trajectory modelling, and re-entry operations. This thesis explores the boundaries of space launch operations monitoring, which is particularly relevant for Space Domain Awareness (SDA) services. The work focuses on the dynamics of three representative classes of launch systems: ballistic missiles (BM), direct-ascent anti-satellite (ASAT) weapons, and orbital launch (OL) vehicles. These systems share common ascent characteristics while serving distinct operational purposes, making their reliable discrimination a key challenge. This project develops a trajectory modelling approach capable of reproducing representative boost phase behaviours for the different launch system classes. By identifying the key vehicle and guidance parameters that govern the ascent profile and systematically varying them, a comprehensive catalogue of launch trajectories is generated, serving as the foundation for subsequent identification, classification, and system-level analyses. A measurement simulation framework is introduced to emulate real-time observations from a sensor network. This enables the evaluation of a developed trajectory classification algorithm under realistic measurement conditions and provides insight into the requirements imposed on sensor capabilities. In addition, these analyses are complemented by a mission-level study of satellite constellation designs. The coverage performance of different HEO–GEO configurations is investigated to connect launch detection considerations with global surveillance objectives and to assess the feasibility of achieving continuous, multi-satellite coverage through appropriate constellation design.

Keywords: Space Domain Awareness, Early warning systems, Ballistic Missiles, Anti-Satellite systems, Orbital Launch vehicles, trajectory classification

Sommario

L'emergere di armi anti-satellite (ASAT) e minacce suborbitali, insieme alla proliferazione delle capacità di lancio hanno aumentato l'importanza di un rilevamento efficace dei lanci, modellazione delle traiettorie e operazioni di rientro. Questa tesi esplora i confini del monitoraggio delle operazioni di lancio spaziale, particolarmente rilevante per i servizi di Space Domain Awareness (SDA). Il lavoro si concentra sulla dinamica di tre classi rappresentative di sistemi di lancio: missili balistici (BM), armi anti-satellite a salita diretta (ASAT) e veicoli di lancio orbitale (OL). Questi sistemi condividono caratteristiche comuni di ascesa pur servendo scopi operativi distinti, rendendo la loro discriminazione affidabile una sfida chiave. Questo progetto sviluppa un approccio di modellazione delle traiettorie in grado di riprodurre comportamenti rappresentativi della fase di spinta per le diverse classi di sistemi di lancio. Identificando i principali parametri del veicolo e di guida che governano il profilo di ascesa e variandoli sistematicamente, viene generato un catalogo completo di traiettorie di lancio, servendo come base per successive analisi di identificazione, classificazione e a livello di sistema. Viene introdotto un framework di simulazione delle misure per emulare le osservazioni in tempo reale da una rete di sensori. Ciò consente la valutazione di un algoritmo di classificazione delle traiettorie sviluppato in condizioni di misura realistiche e fornisce informazioni sui requisiti imposti alle capacità dei sensori. Inoltre, queste analisi sono completate da uno studio a livello di missione dei design delle costellazioni satellitari. Si indaga la performance di copertura di diverse configurazioni HEO–GEO per collegare le considerazioni sul rilevamento dei lanci con gli obiettivi di sorveglianza globale e per valutare la fattibilità di raggiungere una copertura continua multi-satellite attraverso un design appropriato della costellazione.

Parole chiave: Space Domain Awareness, Sistemi di Early Warning, Missili balistici, Sistemi anti-satellite, Veicoli di lancio orbitale, Classificazione delle traiettorie

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1 | Introduction and background

In the contemporary security environment, access to the space domain has become a fundamental dimension of military strategy, global communication, and critical infrastructure. The growing dependence of civilian and military systems on space-based assets has rendered launch operations vital not only for space missions, but also for national defence, strategic dissuasion, and global surveillance. Against this background, the operations associated with direct-ascent trajectory and re-entry vehicles have become essential.

1.1. The changing space environment

Space is no longer a passive environment for terrestrial operations. It has become a contested, congested, and increasingly militarised area. The emergence of anti-satellite (ASAT) weapons, the expansion of ballistic missile capabilities, and the growing number of orbital launch activities have introduced new operational and security challenges. In this context, Space Domain Awareness (SDA) has become critical for detecting, tracking, and identifying objects and events in near-Earth space. States view their space assets as vital force multipliers in weather forecasting, navigation, communications, reconnaissance, and early-warning. Thus, compromising these assets is considered a strategic lever in military operations.

1.2. Launch operations in the defence and security context

A primary challenge within SDA is the early assessment of launch events. During the boost phase, ballistic missiles, ASAT systems, and certain orbital launch vehicles may exhibit comparable kinematic signatures, particularly when observed through limited or noisy measurements. This similarity complicates the rapid identification of hostile launches, especially in the initial stages of flight when decision time is minimal.

Launch operations constitute the initial phase of any space mission, but when considered

in a defence context, they also represent the critical transition from preparation to execution. The boost phase of a launching vehicle is typically characterised by high infrared emissions, rapid acceleration, and evolving trajectory geometry, making it detectable and operationally decisive. Early detection and correct classification during this phase are fundamental for timely response and strategic decision-making.

However, existing early-warning approaches often prioritise detection over discrimination. While identifying that a launch has occurred is essential, distinguishing its nature poses additional technical challenges. The absence of unified and validated modelling frameworks that enable a consistent comparison of these trajectory classes under realistic conditions limits the ability to assess classification strategies and their implications for space-based early-warning architectures.

1.3. Scope of the thesis

This thesis addresses these challenges. It focuses on the modelling, simulation, and identification of direct-ascent trajectories within the context of defence and space security applications. Specifically, the work addresses the boost phase dynamics of three representative classes of systems: Ballistic Missiles (BM), Direct-Ascent Anti-Satellite (ASAT) weapons, and Orbital Launch (OL) vehicles. These systems are selected due to their operational relevance, shared ascent characteristics, and their critical role in Space Domain Awareness (SDA) and early-warning architectures.

The primary objective of this research is to develop a unified framework for generating representative ascent trajectories during the boost phase, and to investigate how their dynamic behaviour can be exploited for classification and identification purposes.

The thesis explores the feasibility of distinguishing between these classes using state information. A data-driven classification approach is adopted, with particular emphasis on the selection of discriminative features and the robustness of the classification process under realistic measurement conditions.

Finally, the project includes a mission analysis to evaluate Earth coverage performance for different satellite constellation configurations. This analysis serves as a complementary study, linking trajectory detection considerations with broader system-level space surveillance objectives.

1.4. Structure of the thesis

The structure of the document is defined as follows:

Chapter 1: Introduction and background. This chapter provides the defence and space security context that motivates this research. It outlines the evolving space environment and the role of launch operations within security frameworks, while defining the scope and objectives of the work.

Chapter 2: State of the art. A comprehensive review of direct-ascent systems is presented, covering ballistic missiles, ASAT systems, and orbital launch vehicles from historical, technological, and operational perspectives. This chapter establishes the necessary background to justify the modelling assumptions adopted in subsequent sections.

Chapter 3: Trajectory modelling and simulation. This chapter develops the numerical framework for trajectory simulation. The equations of motion, guidance laws, and simplifying assumptions are detailed for each vehicle class. The implementation of these models within a custom simulation environment is described, followed by validation against representative case studies and the generation of a large trajectory database.

Chapter 4: Trajectory identification and classification. The generated database is analysed to extract discriminative features, and a K-Nearest Neighbours (KNN) classification algorithm is implemented and optimised. The analysis is extended to a realistic scenario by introducing a simplified sensor network model and assessing the impact of measurement uncertainty on classification performance.

Chapter 5: Mission analysis. This chapter evaluates Earth surface coverage as a function of satellite constellation design. Different orbital configurations are analysed to assess coverage efficiency, providing a system-level perspective that complements the trajectory and detection studies.

Chapter 6: Conclusions and future work. The thesis concludes with a summary of the main findings and suggestions for future research directions.

2 | State of the art

In this chapter, an overview of the current state of the art concerning vehicles whose trajectories are most relevant to direct-ascent and re-entry dynamics is presented. The analysis covers three major categories: ballistic missiles, anti-satellite (ASAT) systems, and orbital launch vehicles. These systems constitute the primary real-world examples of platforms that initiate their missions through steep or near-vertical ascent profiles and, therefore, provide essential insight into the operational, technological, and historical foundations of ascent dynamics. By examining their evolution, classification, representative models, testing campaigns, and launch infrastructures, this chapter establishes the basis for the trajectory modelling, simulation, and detection analyses developed in the subsequent sections of this thesis.

2.1. Ballistic Missiles

The trajectory of a ballistic missile is commonly divided into three phases: the boost phase, the ballistic phase, and the re-entry phase [1]. The boost phase extends from launch until the final burnout, after which the missile follows an unpowered ballistic trajectory toward its target. Ballistic missiles are conventionally classified according to their maximum range [2] as follows:

- Battlefield Range Ballistic Missiles (BRBM): range less than 100 km.
- Tactical Ballistic Missiles: range between 150 and 300 km.
- Theatre Ballistic Missiles (TBM): range between 300 and 3000 km.
 - Short-Range Ballistic Missiles (SRBM): range less than 1000 km.
 - Medium-Range Ballistic Missiles (MRBM): range between 1000 and 3000 km.
- Intermediate-Range Ballistic Missiles (IRBM): range between 3000 and 5500 km.
- Intercontinental Ballistic Missiles (ICBM): range higher than 5500 km.

The term ICBM is typically used for missiles launched from ground, while SLBM (Submarine-

Launched Ballistic Missile) refers to ballistic missiles launched from submarines. However, it should be noted that all modern SLBMs exhibit intercontinental ranges.

In the context of this thesis, the systems of primary interest are the SRBMs, MRBMs, IRBMs, and ICBMs, since their ascent profiles are directly relevant to direct-ascent modelling and detection. A schematic illustration of the classification of ballistic missiles by range is provided in Figure 2.1.

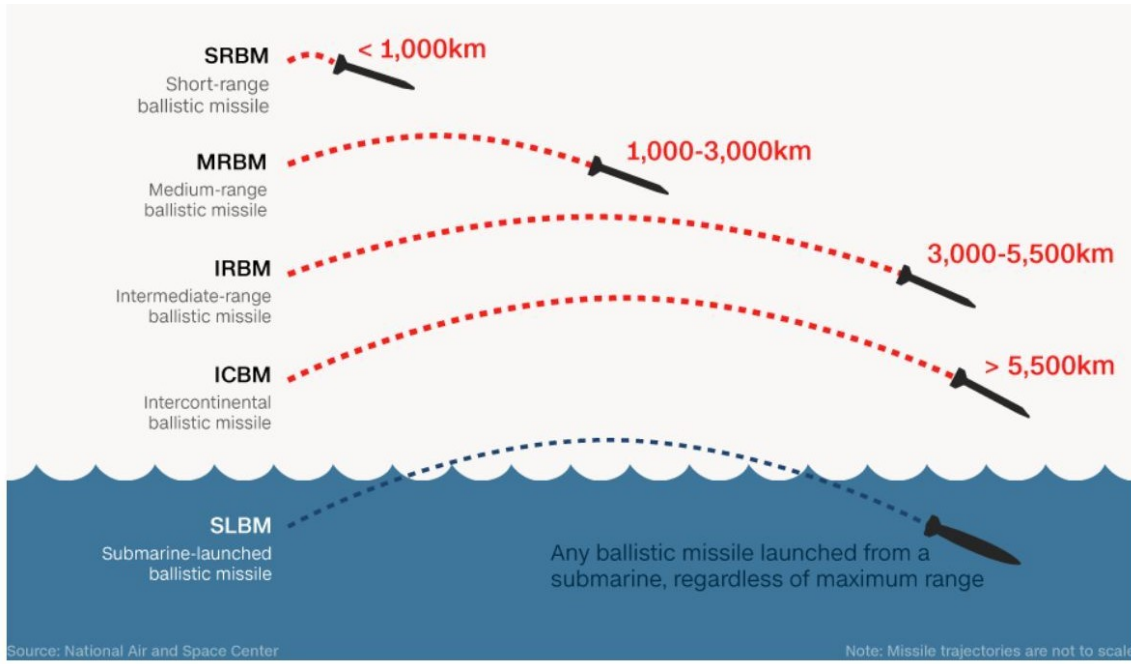


Figure 2.1: Ballistic Missile classification by range. (Source: National Air and Space Center.)

2.1.1. History of Ballistic Missiles

The first true precursor of the ballistic missile was the German *V-2* (see Figure 2.2), developed under Wernher von Braun and first launched in 1942. As the world's first long-range, liquid-propellant ballistic missile, the *V-2* demonstrated key technological elements, such as vertical launch, turbo-pumps, gyroscopic guidance, and a high-arc ballistic trajectory. It had a weight at launch of 13000 kg, a range of 300 km with a highest point of altitude of 90 km and an impact velocity of about 1100 m/s, carrying a 738 kg warhead. Up to 6000 *V-2* rockets were built, of which approximately 3000 were launched, especially against the cities of London and Antwerp, resulting in the deaths of about 9000 civilians and military personnel [3]. The *V-2* established the conceptual foundation for all subsequent missile and space-launch programmes.

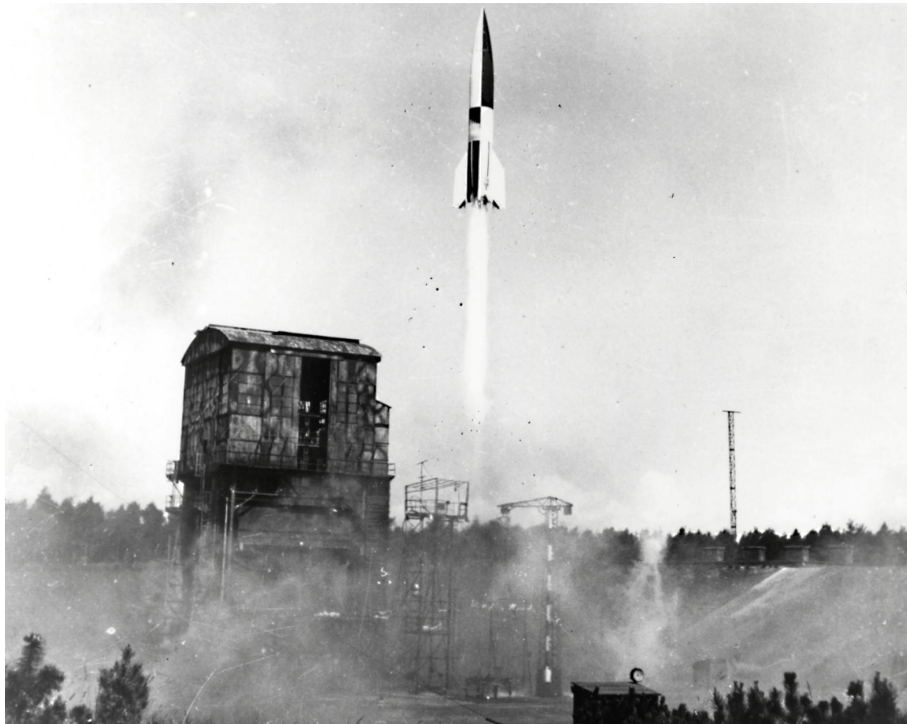


Figure 2.2: Launch of a V-2 rocket. (Source: Wikimedia Commons.)

At the end of World War II, the United States (U.S.) and the Soviet Union (U.S.S.R.) both acquired V-2 hardware and personnel, accelerating their respective missile programmes that would rapidly expand under the strategic pressures of the emerging Cold War. By the late 1950s, the Soviet *R-7 Semyorka* became the world's first operational ICBM and simultaneously the launcher of *Sputnik-1*, the first artificial satellite, in 1957 [4]. The U.S. soon followed with the *Atlas*, *Redstone*, *Titan*, and *Proton* missiles, which also constituted also the basis of early space launch systems. It was the *Minuteman* series that introduced solid-propellant ICBMs, significantly reducing launch preparation time and improving reliability [5].

During this period, ballistic missiles evolved from experimental prototypes into central components of nuclear deterrence. The introduction of Multiple Independently Targetable Reentry Vehicle (MIRVs) in the late 1960s further multiplied the strategic significance of ICBMs, enabling a single missile to deliver several warheads to different targets [6].

Up to 1970, the number of ICBMs and SLBMs was continuously increasing both in the U.S. and the U.S.S.R. Due to their extreme range, high speed, and demanding performance requirements, ICBMs are highly complex strategic weapons, developed and maintained almost exclusively for the delivery of nuclear warheads to distant and well-defended targets. For instance, in 1967 the U.S. fielded 1054 ICBMs and 656 SLBMs [3]. The cost of

this deployment was huge and the number of missiles was more than sufficient to ensure a mutual assured destruction. As a consequence, several treaties were pursued to reduce the number of deployed missiles.

The first major effort began with the 1972 *Strategic Arms Limitation Talks* (SALT) treaty [7], which froze the number of ICBM launchers in both the U.S. and the U.S.S.R. and allowed the deployment of new SLBM launchers only if an equivalent number of land-based ICBM launchers were dismantled. A second round of negotiations, SALT II [8], conducted between 1972 and 1979, introduced additional limits that reduced the number of deployed nuclear warheads.

A significant step forward came with the 1991 *Strategic Arms Reduction Treaty* (START), which capped each side at 6000 nuclear warheads and a total of 1600 strategic delivery systems, including ICBMs, SLBMs, and heavy bombers. Its full implementation in 2001 led to the elimination of roughly 80 percent of all existing strategic nuclear weapons.

START was followed in 2003 by the *Strategic Offensive Reductions Treaty* (SORT), under which both nations agreed to reduce their operationally deployed warheads to between 1700 and 2200.

The current arms-control framework is defined by the New START treaty (*Measures for the Further Reduction and Limitation of Strategic Offensive Arms*) [9], in force since 2011. It limits each party to 1,550 deployed strategic warheads and sets a ceiling of 800 deployed and non-deployed ICBMs, SLBMs, and nuclear-capable heavy bombers, with no more than 700 of them allowed to be deployed.

Other nations' ballistic missile programmes

While the early development of ballistic missiles was dominated by the United States and the Soviet Union, other countries gradually pursued strategic and tactical missile programmes since the 70s.

- France developed its *M*-series missiles, and nowadays only has SLBMs in operation (M45 and M51).
- The United Kingdom fielded the American *Trident II* SLBM aboard its *Vanguard*-class submarines, integrating with its nuclear deterrent strategy.
- Israel initiated missile development with tactical and medium-range systems, focusing on regional deterrence. *Jericho* is a general designation given to the Israeli ballistic missiles.

- China established a robust programme spanning short-range to intercontinental missiles, including the *Dong Feng* (DF) series. Among these, the *Dong Feng-41* (DF-41) missile is one of the world's longest-range missiles, with an estimated range of 12000 - 14000 km, capable of covering any position of the planet.
- India and Pakistan have developed medium- to long-range ballistic missiles, such as India's *Agni* series and Pakistan's *Shaheen* series, with both nations conducting multiple tests in recent decades.
- North Korea has pursued increasingly capable systems, including *Hwasong* and *Tae-podong* missiles, some of which demonstrate intercontinental potential.
- Iran has focused on short- and medium-range ballistic missiles, developing systems such as the *Shahab* and *Sejjil* families.

A list of the principal SRBMs, MR/IRBMs, and ICBMs in each country, as reported by [2], is presented in Appendix A.

2.1.2. Operational aspects of Ballistic Missiles

Trajectory phases

As previously introduced, the flight path of a ballistic missile can be divided into three main phases: the boost phase, the ballistic (or midcourse) phase, and the re-entry phase (see Figure 2.3). Each phase is governed by different regimes and operational considerations, and together they determine the final impact point with high sensitivity.

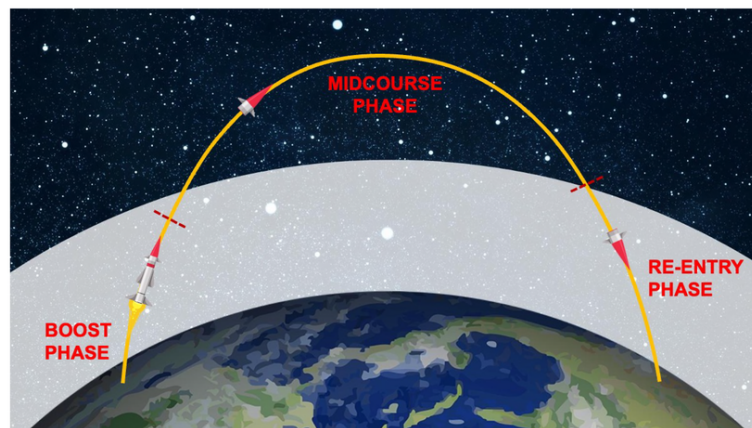


Figure 2.3: Trajectory of a Ballistic Missile. (Source: [10])

1. Boost phase:

During this phase, all propulsion stages ignite sequentially until burnout. Depending on the missile's range and propulsion configuration, this burnout may occur either within the atmosphere or well above it. The missile typically begins in a near-vertical attitude (launch pitch angle of approximately $80\text{-}90^\circ$) and performs a pitch-over manoeuvre to a lower final flight-path angle.

The purpose of this manoeuvre is to reach a final injection flight-angle suitable for the target range, while trying to keep a low angle of attack in order to avoid lateral forces on the missile.

The conditions at the end of the boost phase, i.e., position, velocity and flight-path angle, fully determine the subsequent ballistic trajectory and thus the eventual impact point.

2. Ballistic (midcourse) phase:

Once the final boost stage has burned out and separated, the missile enters the ballistic or midcourse phase. At this point, the vehicle typically consists of a guidance module and one or more re-entry vehicles (RVs).

The guidance module is responsible for orienting the RVs so that they re-enter the atmosphere with the appropriate flight-path angle to reach the specified impact point. In some missile architectures, this module includes a small engine that allows limited trajectory shaping or range extension. Therefore, in systems equipped with MIRVs (Multiple Independently Targetable Reentry Vehicle), these small manoeuvres allow the post-boost vehicle (PBV) to release several warheads on different trajectories, enabling impacts separated by hundreds of kilometres using a single missile, as illustrated in Figure 2.4.

3. Re-entry phase:

The re-entry phase begins once the RV (or RVs) separates from the guidance module. The RV typically possesses a controlled spin to ensure attitude stability during atmospheric entry.

The release of each RV is usually accompanied by the release of decoys in order to make the identification of the warhead by possible anti-missile systems more difficult.

The RVs follow individual ballistic trajectories, re-enter the Earth's atmosphere, arm, and detonate according to the planned target profile.

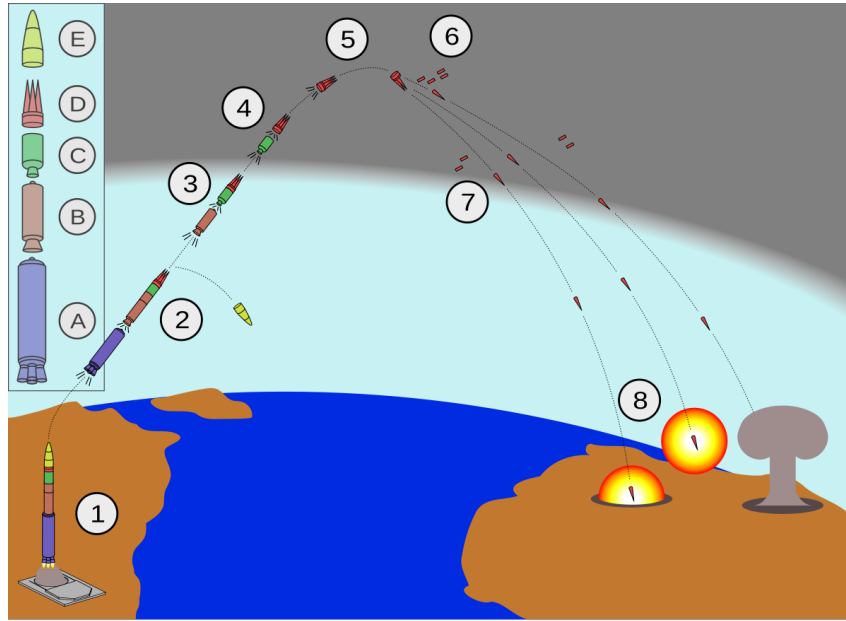


Figure 2.4: *Minuteman III* MIRVs flight path. (Source: Wikimedia Commons.)

Launch platforms

Ballistic missiles can be deployed from a wide variety of launch platforms. The most common categories include silo-based, road-mobile, rail-mobile, submarine-launched, and ship-based systems [2]. Silo-based missiles rely on fixed, hardened underground structures that provide high readiness but limited mobility. Road-mobile systems, on the other hand, typically employ Transporter Erector Launchers (TELs) offering significant survivability by enabling frequent relocation, and represent the dominant launch mode for many modern SRBMs and MRBMs. Ship-based launchers and SLBMs extend operational reach by enabling deployment far from national territory.

These launch platforms also determine the initial launch geometry. Mobile ground-based systems such as TELs or ship-based launchers allow the missile to be oriented prior to ignition, enabling initial elevation angles slightly below the vertical. Therefore, this operational flexibility justifies the typical initial launch pitch angle of $80\text{-}90^\circ$ indicated in the previous section.

As illustrative examples, a TEL-mounted missile during launch is depicted in Figure 2.5a, while the launch of a SLBM is shown in Figure 2.5b.



(a) *Hwasong-17* launch from a TEL.



(b) Russian *Bulava* SLBM launch. (Source: [2])

Figure 2.5: Illustrative examples of launch platforms for ballistic missiles.

Propulsion system

The propulsion configuration of the ballistic missiles varies significantly depending on the range, mission profile, and technological development level. One of the most distinctive design parameters is the number of stages. SRBMs, MRBMs, and IRBMs typically employ single- or two-stage architectures. Many SRBMs are built as single-stage vehicles that remain structurally intact until warhead detonation [2]. In contrast, long-range systems require more sophisticated staging strategies to achieve the necessary velocity.

All existing ICBMs have 2 or 3 stages. The use of only 2 stages can be associated with and older generation of missiles, while the newest generation of ICBMs is, in all countries, based on 3-stage rockets.

In addition to staging, ballistic missiles may rely on either solid- or liquid-propellant engines. While both technologies remain operational worldwide, the modern trend favours solid propulsion due to its operational simplicity and reduced logistical requirements. Liquid-propellant systems, on the other hand, continue to be used primarily by some less developed nations [2].

2.2. Anti-Satellite (ASAT) systems

Unlike terrestrial or atmospheric vehicles, satellites are optimised for mass and volume efficiency rather than structural robustness. Once deployed into orbit, spacecraft operate in a near-vacuum, experiencing no external forces beyond minor orbital perturbations. Consequently, satellite components are typically lightweight, highly specialised, and not designed to withstand physical aggression. This fundamental vulnerability makes the disruption of a satellite's function comparatively simple from an offensive perspective. The primary challenge, therefore, lies not in damaging the spacecraft, but in successfully reaching and engaging it within the space domain.

Anti-Satellite (ASAT) weapons can be systematically classified according to a multidimensional framework, highlighting the diversity of strategies and the corresponding defensive measures required to mitigate them. A comprehensive analysis typically considers three primary dimensions: starting location, mechanism of attack, and intended operational effect.

The first layer of classification distinguishes between ground-based systems, co-orbital (or space-based) vehicles, and non-material cyber-physical attacks [11]:

- Ground-based ASATs: These systems, which include direct-ascent interceptors, are geographically fixed and operate within limited engagement windows dictated by orbital mechanics. Their launch origin is easily attributable based on the region of operation.
- Co-orbital ASATs: In contrast, these vehicles are placed into orbit prior to engagement, allowing them to manoeuvre to rendezvous with or shadow targets. This capability enables ambiguous attribution and establishes a persistent threat potential.
- Non-material Attacks: This category includes cyber-physical attacks which do not involve a physical vehicle. These operations can target any layer of the satellite system (ground, link, or space segment) from virtually any attack vector.

A second classification layer considers the physical or non-physical means by which damage or disruption is inflicted, often categorized as follows [12]:

- Kinetic-Energy Weapons (KEW): These systems, also known as Kinetic Kill Systems (KKS), rely on the physical impact of a projectile to destroy the target, leveraging the high relative velocity in orbit to maximise destructive force. This category includes direct-ascent interceptors (DA-ASATs, or "pop-up", the system of interest

in this document) and co-orbital impactors (space mines). KEW are favoured for the certainty of their destructive effect but are associated with the high political and environmental cost of creating significant, long-lived orbital debris.

- Directed-Energy Weapons (DEW): DEW mechanisms apply focused, non-kinetic energy to disrupt, degrade, or destroy the target. They are generally sub-divided based on the specific type of energy deployed into: (i) High-Energy Physical Systems (High-Energy Lasers (HEL) and Neutral-Particle Beams (NPL), which seek to overwhelm the satellite's structure, sensitive components, or optical systems) and (ii) Radio-Frequency (RF) Systems (designed to overload and permanently damage onboard electronics).
- Isotropic Nuclear Weapons (INW): INW represent the most destructive class of ASAT. Detonation scenarios, whether ground-based or space-based, would release isotropic energy, primarily in the form of X-rays and an electromagnetic pulse (EMP), capable of destroying or severely degrading multiple satellites across vast orbital regions.
- Electronic Warfare (EW): EW measures focus on the non-physical disruption of communication and data links. These active measures include techniques such as jamming (overwhelming communication frequencies), spoofing (inserting false data), and general RF-based denial.
- Cyber operations: They target software, communication protocols, ground-control infrastructure, or command-and-control (C2) links to hijack control, inject false commands, or render the system inoperable without ever physically engaging the satellite or its signal.

The classification of these damage mechanisms is summarized with some examples in Table 2.1.

Finally, ASAT capability is often evaluated in terms of the intended operational outcome, which can range from subtle interference to full elimination. These desired effects may include deception, temporary disruption, reversible denial, long-term degradation, or full destruction of the satellite [13].

Table 2.1: Classification of ASAT weapons by physical mechanism.

Kinetic-Energy Weapons (KEW):
• Ground-based – Direct-ascent (“pop-up”) interceptor • Space-based – Co-orbital interceptor (“space mines”) – Non co-orbital interceptor
Directed-Energy Weapons (DEW):
• Ground-based – High-power radio-frequency (HPRF) and active electronic countermeasures (ECM) – High-energy laser (HEL) and active electro-optical countermeasures (E-OCM) • Space-based – High-power radio-frequency (HPRF) and active electronic countermeasures (ECM) – High-energy laser (HEL) and active electro-optical countermeasures (E-OCM) – Neutral-particle beam (NPB)
Isotropic Nuclear Weapons (INW):
• Ground-based – Direct-ascent (“pop-up”) interceptor • Space-based – Co-orbital interceptor (nuclear ‘space mines’)
Electronic Warfare (EW):
• Engagement Layers – Uplink/Downlink Jamming (RF-based denial) – Spoofing – Interference with tracking and telemetry systems (TT&C)
Cyber Operations:
• Target Segments – Ground segment attacks (targeting control software and C2 infrastructure) – Link segment attacks (exploiting communication protocols) – Space segment attacks (targeting onboard software/firmware)

2.2.1. History of Anti-Satellite (ASAT) systems

The development of Anti-Satellite (ASAT) systems is linked to the strategic competition and technological progress of the Cold War. The initial anti-satellite capabilities came as final tests of already existing Anti-Ballistic Missile (ABM) weapons, with the objective of establishing a countermeasure intended to destroy reconnaissance, navigation, telecommunications and weather satellites in case of war [14].

The first ASAT technologies emerged in the late 1950s. The Soviet Union pioneered the development of the first successful kinetic ASAT system, the *Istrebitel Sputnikov (IS)*. This co-orbital weapon was designed to rendezvous its target over several orbits before detonating a fragmentation warhead nearby to disable the spacecraft. The first successful test of this system was achieved in 1968, and it was declared operational shortly thereafter.

In parallel, early U.S. efforts followed two main paths: the adaption of air-launched missiles (such as *Bold Orion*) and the use of high-altitude nuclear detonations. The latter was considered a viable ASAT mechanism due to the devastating effects of the resulting electromagnetic pulse and artificial radiation belts, as demonstrated by the *Startfish Prime* test in 1962, which unintentionally damaged several satellites. By the mid-1960s, the U.S. deployed the *Thor*-based *Program 437* ASAT, relying on a nuclear-tipped missile.

The 1980s marked a shift towards faster non-nuclear interceptors. The U.S. developed the *Vought ASM-135 ASAT*, an air-launched missile carried by a modified F-15 fighter jet (see Figure 2.6). This missile had an estimated operational range of 648 km, a flight ceiling of 563 km, and speed of over 24000 km/h. The only successful kinetic interception against a satellite target, the *Solwind P78-1*, occurred on 13 September 1985. Debris from the destruction persisted on orbit until 9 May 2004. Although successful, the programme was cancelled in 1988 due to a mix of budgetary, technical, and political concerns [15].



Figure 2.6: *F-15A Eagle* launching the *ASM-135*.

(Source: Wikimedia Commons.)

Although primarily designed for ballistic missile defence, the U.S. Strategic Defense Initiative (SDI) significantly accelerated research into technologies with inherent dual-use potential for ASAT applications. This included advanced concepts for space-based kinetic interceptors (such as *Brilliant Pebbles*, a satellite constellation of 4600 kinetic interceptors in Low Earth Orbit (LEO) [16]) and high-power Directed-Energy Weapons (DEW). In parallel, the Soviet Union also adapted its defensive research, expanding its DEW projects. Although many of these ambitious programmes were later cancelled, this period matured the technological foundations for rapid ASAT capability development in subsequent decades.

These technological advances unfolded within the context of several strategic arms-control agreements. Instruments such as the *Anti-Ballistic Missile (ABM) Treaty* and the *Strategic Arms Limitation Talks* (SALT I and SALT II) did not explicitly ban ASAT weapons, but imposed constraints by limiting certain missile-defence architectures and space-based interceptor deployments [12]. As a result, ASAT progress during the late Cold War evolved in tension between technological ambition and treaty-driven limitation.

As a summary, known and suspected US Direct-Ascent ASAT (DA-ASAT) testing activity conducted from the early 1960s through the 21st century is compiled in Appendix A.

In the 21st century, three other nations (China, India and Russia) successfully demonstrated kinetic DA-ASAT capabilities, significantly intensifying concerns over space security and the creation of long-lived orbital debris [17]. In contrast, the United States has since stopped the testing of DA-ASAT missiles, having outlawed the practice in 2022 [18].

2.2.2. Main DA-ASAT demonstrations

China

The most notable event demonstrating the expanded threat was China's test on 11 January 2007. Using a ground-launched kinetic interceptor derived from a ballistic missile (identified as the *SC-19*), China successfully destroyed its defunct weather satellite *Fengyun-1C* orbiting at an altitude of approximately 865 km. This high-altitude intercept was highly controversial, as it generated the largest and most persistent artificial debris field in history, consisting of more than 3000 pieces of trackable space debris (larger than 10 cm in size) that pose a long-term threat to LEO assets, including the International Space Station (ISS) [19].

United States

The U.S. most recent official kinetic demonstration was Operation Burnt Frost in 2008. In this test, a ship-launched *RIM-161 Standard Missile 3 (SM-3)*, originally designed for naval missile defence, was used to intercept the malfunctioning US reconnaissance satellite *USA-193*. The intercept occurred at an altitude of approximately 247 km, resulting in the successful destruction of the satellite. The stated reason for the action was the need to mitigate the risk posed by the approximately 450 kg of toxic hydrazine fuel on board the descending spacecraft [20].

India

India became the fourth nation to join this group on 27 March 2019, with the successful execution of Mission Shakti. The test involved launching a modified interceptor missile developed under India's Ballistic Missile Defence (BMD) Programme, the *PDV Mark-II*, to destroy its own low-orbit satellite *Microsat-R*. The intercept was performed at a comparatively low altitude of 284 km, within 168 seconds of the launch, at a closing relative speed of almost 10 km/s. The destruction created fragments along its polar retrograde orbit, inclined 96.7° with respect to the Earth's equator, as the ASAT missile strikes the target with an upwards angle of about 48° with the horizontal. This low-altitude choice was a measure explicitly stated by the Indian government to ensure that

the resulting debris would rapidly decay and re-enter the Earth's atmosphere within a few weeks, thereby minimising the long-term debris hazard [21]. In Figure 2.7, a graphical representation of missile trajectory is presented.

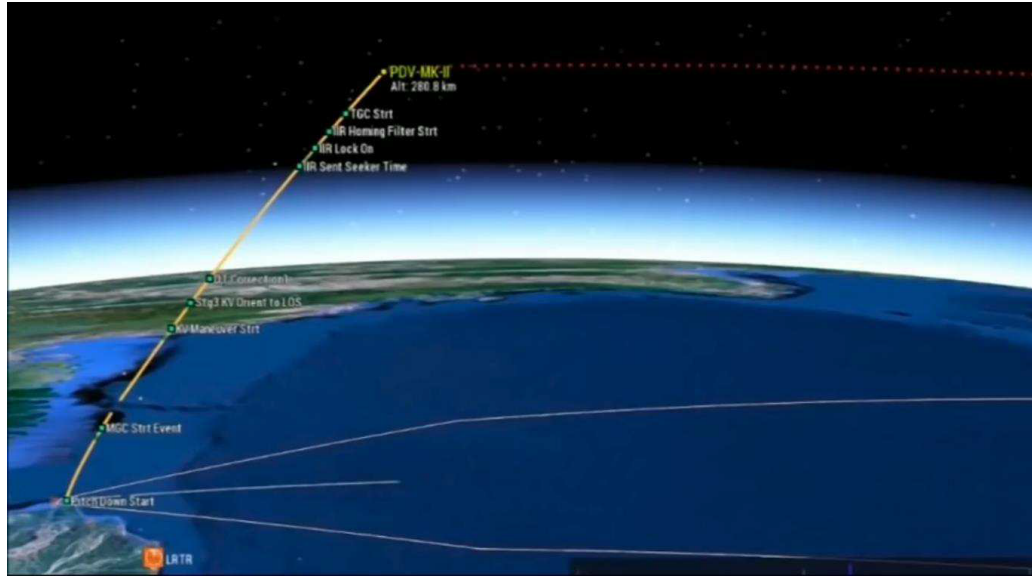


Figure 2.7: Representation of missile trajectory in Mission Shakti. (Source: [21])

Russia

Russia demonstrated its own DA-ASAT capability in November 2021 by destroying its defunct satellite *Kosmos 1408* with the *A-235 PL-19 Nudol* missile. Although the test targeted a satellite in a relatively lower orbit of approximately 465 km of altitude, the event generated thousands of pieces of debris, briefly forcing the crew of the ISS to enter safe haven procedures. This test highlighted the persistent risk posed by DA-ASAT weapons to LEO operations [22].

In summary, the modern history of ASAT testing is defined by five destructive DA-ASAT demonstrations that have fundamentally altered the risk landscape in LEO. These tests are presented in Table 2.2.

Table 2.2: List of destructive Direct-Ascent ASAT (DA-ASAT) tests.

Date	Country	Interceptor	Target	Altitude (km)
13 Sep. 1985	United States	ASM-135 ASAT	Solwind P78-1	555
11 Jan. 2007	China	SC-19	FY-1C	865
21 Feb. 2008	United States	SM-3	USA-193	247
27 Mar. 2019	India	PDV Mark-II	Microsat-R	284
15 Nov. 2021	Russia	A-235 PL-19 Nudol	Kosmos 1408	465

2.3. Orbital Launch vehicles

Orbital launch vehicles, commonly referred to as launch systems or launchers, represent the third primary class of vertical-ascent vehicles considered in this study. Unlike ballistic missiles or DA-ASAT interceptors, the fundamental purpose of a launcher is to deliver a payload to a sustained, stable orbit, which requires both sufficient altitude and the necessary orbital velocity (approximately 7.8 km/s for LEO). This requirement dictates a distinct set of design, propulsion, and operational constraints, thereby differentiating orbital launch vehicles from sub-orbital vehicles.

Launchers may be classified according to several criteria based on their design, operational concept, and performance capability. These classification schemes enable a clear differentiation between system types with distinct mission profiles.

The most fundamental classification relates to the number of stages used to reach orbital velocity. According to the Tsiolkovsky rocket equation, shedding structural mass (empty tanks, engines, interstages) during ascent increases the final achievable velocity [23]. In this context, launchers are divided into:

- Multi-Stage to Orbit (MSTO). The vast majority of operational launchers use multiple stages to increase efficiency and maximise payload delivery.
- Single-Stage to orbit (SSTO). While the conceptual goal of many advanced programs, no fully functional SSTO launch vehicle has yet reached orbit.

A second classification criterion is reusability, which distinguishes launch vehicles based on the recovery and restoration of high-value components:

- Expendable Launch Vehicles (ELV). These systems represent the traditional architecture, where all stages are discarded after a single use, either through atmospheric re-entry or ocean impact.
- Reusable Launch Vehicles (RLV). These modern systems (e.g., *SpaceX Falcon* family) are designed to recover and reuse components, typically the first stage or the entire vehicle. This approach reduces launch costs and significantly increases launch cadence [24].

Finally, launchers are classified by their payload capacity delivered to a reference orbit, generally Low Earth Orbit (LEO) or Geosynchronous Transfer Orbit (GTO). Although exact thresholds may vary between space agencies, the following ranges are commonly used [25]:

- Small-lift: Up to 2000 kg to LEO.
- Medium-lift: Between 2000 kg and 20000 kg to LEO.
- Heavy-lift: Between 20000 kg and 50000 kg to LEO.
- Super-Heavy-lift: Above 50000 kg to LEO.

A vast array of orbital launch vehicle designs has been developed worldwide. Although modern launchers differ considerably in size and configuration, they generally follow a cylindrical, multi-stage architecture. These vehicles typically incorporate two or more liquid- or solid-propellant stages, as mentioned before, with the first stage frequently supplemented by solid boosters to increase lift-off thrust. The payload is positioned at the apex, protected during ascent by an aerodynamic fairing. A representative configuration is shown in Figure 2.8.

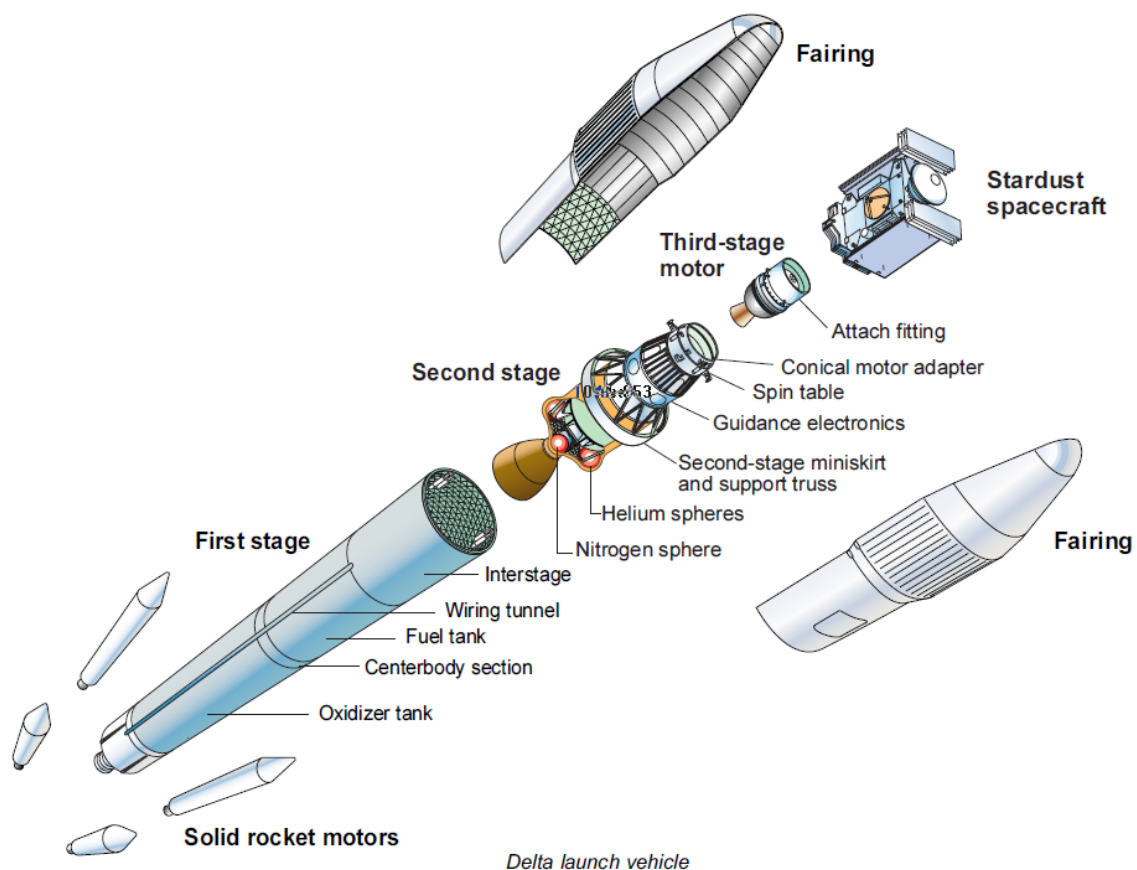


Figure 2.8: Launch vehicle assembly diagram for the *Stardust* spacecraft. (Source: Wikimedia Commons.)

2.3.1. Most relevant launch vehicles and launch sites

The operational viability of an orbital launch is strongly constrained by both the available ground infrastructure and the geographical characteristics of the launch site. Most launch vehicles are compatible with only a limited number of dedicated spaceports, which must provide the necessary infrastructure, such as mission-specific control facilities, integration and processing buildings, transport systems, launch pad, and fuelling capabilities [26].

The geographical position of the launch site plays an important role. The latitude defines the range of accessible orbital inclinations: equatorial sites benefit from Earth's rotational velocity, favouring low-inclination orbits, while higher-latitude locations provide more efficient access to polar and Sun-synchronous orbits (SSO). In parallel, range-safety regulations restrict the available launch azimuths to ensure that ascent trajectories and stage-disposal paths avoid populated or foreign territory.

In summary, global launch capability is shaped by geography and the scale of national infrastructural investment.

This section details the current and near-future launch vehicle landscape, focusing on the most relevant operational systems grouped by their country of origin.

United States

The United States (U.S.) has historically fielded the widest development and diversity of orbital launch vehicles, characterised by a transition from legacy expendable systems to next-generation reusable and super-heavy-lift architectures. The current landscape is dominated by commercially operated launchers such as the reusable *Falcon 9* and *Falcon Heavy* (SpaceX), both using two liquid-propellant stages (RP-1/LOX), as well as the legacy *Atlas V* (United Launch Alliance, ULA) and its successor, the *Vulcan*, developed to support national security missions. Additionally, the *Space Launch System (SLS)* and the developmental *Starship* represent the country's ambitions towards crewed deep-space explorations and super-heavy-lift operations [27].

The launchers mentioned are detailed in Table 2.3, together with other systems currently in use.

Table 2.3: List of U.S. launchers. (Source: [26])

Launcher Name	Version	Stages	Lift-off mass (t)	Payload (kg)
Antares	230+	2	298	LEO: 3800
Atlas V	551	2 + 4 boosters	500	LEO: 18800 GTO: 8900
Electron	-	3	12.5	LEO: 300 GTO: 200
Falcon 9	Block 5	2	549	LEO: 22800 GTO: 8300
	Heavy	2 + 2 boosters	1420	LEO: 63800 GTO: 26700
Firefly Alpha	-	2	54	LEO: 1030 SSO ¹ : 630
Minotaur	IV	4	86.3	LEO: 1735
SLS	Block 1	2 + 2 boosters	2610	LEO: 95000 TLI ² : >27000
Vulcan	VC6	2	546.7	LEO: 27200 GTO: 14500
Pegasus	XL	3	23.13	LEO: 450
New Glenn	7x2	2	1815	LEO: 45000 GTO: 13600

¹ Sun-Synchronous Orbit (SSO).

² Trans-Lunar Injection (LTI).

U.S. orbital launch activity is supported by several launch sites, the principal of which are:

- Air Force Eastern Test Range (AFETR). Located in Florida, the Kennedy Space Center (KSC) and the Cape Canaveral Space Force Station (CCSFS) are currently the launch sites for *Falcon 9*, *Falcon Heavy*, *Atlas V*, *Vulcan*, and the *Space Launch System*. Its geographic position is favourable for low-inclination orbits, taking advantage of Earth's rotational speed for eastward launches over the Atlantic Ocean. Historically, this range supported the *Apollo* and *Space Shuttle* programmes, and has more recently re-enabled polar launches routing flights southward along the coast.
- Air Force Western Test Range (AFWTR). Centred on the Vandenberg Space Force Base in California, this range is mainly used for high-inclination and polar orbits due to its safe launch path over the Pacific Ocean. AFWTR is also used for lower inclinations and even some interplanetary missions have been launched from there.

Additional active sites include Wallops Island Flight Facility (WFF) and the Pacific Spaceport Complex - Alaska (PSCA), both of which mainly serve small-lift launchers. The Starbase facility in Texas remains dedicated exclusively to development and testing operations for the *Starship* programme.

Table 2.4 details the coordinates of the mentioned launch sites, and Figure 2.9 presents the location of the principal U.S. orbital launch complexes.

Table 2.4: List of launch sites in U.S. (Source: [26])

Launch Site Name	Latitude	Longitude
Air Force Eastern Test Range (AFETR)	28 ^o 36' 29.52" N	80 ^o 36' 14.99" W
Air Force Western Test Range (AFWTR)	34 ^o 37' 55.31" N	120 ^o 36' 38.16" W
Pacific Spaceport Complex - Alaska (PSCA)	57 ^o 25' 48.53" N	152 ^o 21' 22.15" W
Wallops Island Flight Facility (WFF)	37 ^o 50' 1.86" N	75 ^o 29' 15.80" W



Figure 2.9: Map of the launch sites in U.S.

China

China currently ranks as the second nation worldwide in terms of launch frequency and annual total mass delivered to orbit, with its national space programme, led primarily by the China National Space Administration (CNSA). This programme is centred around the *Chang Zheng* (CZ) family of vehicles, also known as the *Long March* series. However,

in recent years several private companies have emerged developing their own launchers [28].

The CZ series constitute the core of the Chinese space programme, spanning a wide range of capabilities and employing mainly liquid propulsion, with some variants incorporating solid stages. In parallel, the commercial sector has introduced a growing catalogue of launchers, most of them solid-propellant vehicles (e.g., *Kuaizhou*, *Hyperbola-1*, *Jielong 3* and *Kinetica 1*), typically capable for payloads of maximum two tonnes into LEO or SSO.

The launchers discussed are detailed in Table 2.5.

China has a diverse network of six active launch sites, combining inland facilities, coastal bases, and movable sea platforms, which together provide access to a wide range of orbital inclinations.

- Jiuquan Satellite Launch Centre (JSC). Located in the northern steppes, JSC serves various CZ rockets, including the human-rated CZ-2F, and is mainly used for LEO, Medium Earth Orbit (MEO), and SSO launches towards the south.
- Taiyuan Satellite Launch Centre (TSC). Located in northeastern China, TSC commonly supports SSO and LEO missions.
- Xichang Satellite Launch Centre (XSC). Located in a mountainous southern region, XSC has traditionally launched older CZ rockets, often targeting LEO, MEO, and Geosynchronous Orbit (GEO).
- Wenchang Satellite Launch Centre (WSC). Located on Hainan Island in southern China, WSC is the country's primary modern coastal facility. It has two main launch pads, one for heavy-lift vehicles (CZ-5) and another for the CZ-7 and CZ-8 families. It is optimised for achieving LEO, SSO, as well as GEO, with inclinations ranging from 0° to 97° , and served as the launch site for key missions, including modules of the Chinese Space Station (*Tiangong*).
- Sea Launch Platforms. China also operates two mobile platforms in the Yellow Sea Launch Area (YSLA) and the South China Sea Launch Area (SCSLA), enabling highly flexible launch azimuths for vehicles such as the CZ-11 and private solid rockets, primarily into LEO or SSO.

Table 2.6 details the coordinates of the mentioned launch sites, and Figure 2.10 presents the location of the principal orbital launch complexes in China.

Table 2.5: List of China launchers. (Source: [26])

Launcher Name	Version	Stages	Lift-off mass (t)	Payload (kg)
Long March 2	2C	2	191	LEO: 3850 SSO: 1400
	2D	2	232	LEO: 3500 SSO: 1300
	2F	2 + 4 boosters	462	LEO: 8800
Long March 3	3B/E	3 + 4 boosters	425	LEO: 11500 SSO: 7000
	3C/E	3 + 2 boosters	329	LEO: 9100 SSO: 6450
Long March 4	4B	3	249	LEO: 4200 SSO: 2295
	4C	3	250	LEO: 4200 SSO: 2947
Long March 5	5	1 + 4 boosters	867	LEO: 25000 GTO: 14000
	5B	2 + 4 boosters	840	LEO: 11500 SSO: 7000
Long March 6	6	2	103	LEO: 1500 SSO: 500
Long March 7	7	2 + 4 boosters	597	LEO: 1400 SSO: 5500
	7A	2 + 4 boosters	573	GTO: 7800
Long March 8	Standard	2 + 2 boosters	356	LEO: 8100 SSO: 5000
	Core Only	2	198	SSO: 3000
Long March 11	11	4	58	LEO: 700 SSO: 350
Long March 12	12	2	433	LEO: 12000 SSO: 6000
Kuaizhou	1A	4	30	LEO: 300 SSO: 250
Hyperbola-1	-	4	42	LEO: 300 SSO: 260
Jielong 3	-	4	145	LEO: 3850 SSO: 1500
Kinetica 1	-	4	135	LEO: 2000 SSO: 1500

Table 2.6: List of launch sites in China. (Source: [26])

Launch Site Name	Latitude	Longitude
Jiuquan Satellite Launch Centre (JSC)	40° 57' 29.00" N	100° 17' 28.00" E
Taiyuan Satellite Launch Centre (TSC)	38° 50' 54.53" N	111° 36' 31.32" E
Xichang Satellite Launch Centre (XSC)	28° 14' 45.66" N	102° 1' 35.60" E
Wenchang Satellite Launch Centre (WSC)	19° 36' 52.17" N	110° 57' 4.08" E
Yellow Sea Launch Area (YSLA)	34° 53' 60.00" N	121° 11' 24.00" E
South China Sea Launch Area (SCSLA)	21° 17' 0.00" N	112° 0' 0.00" E

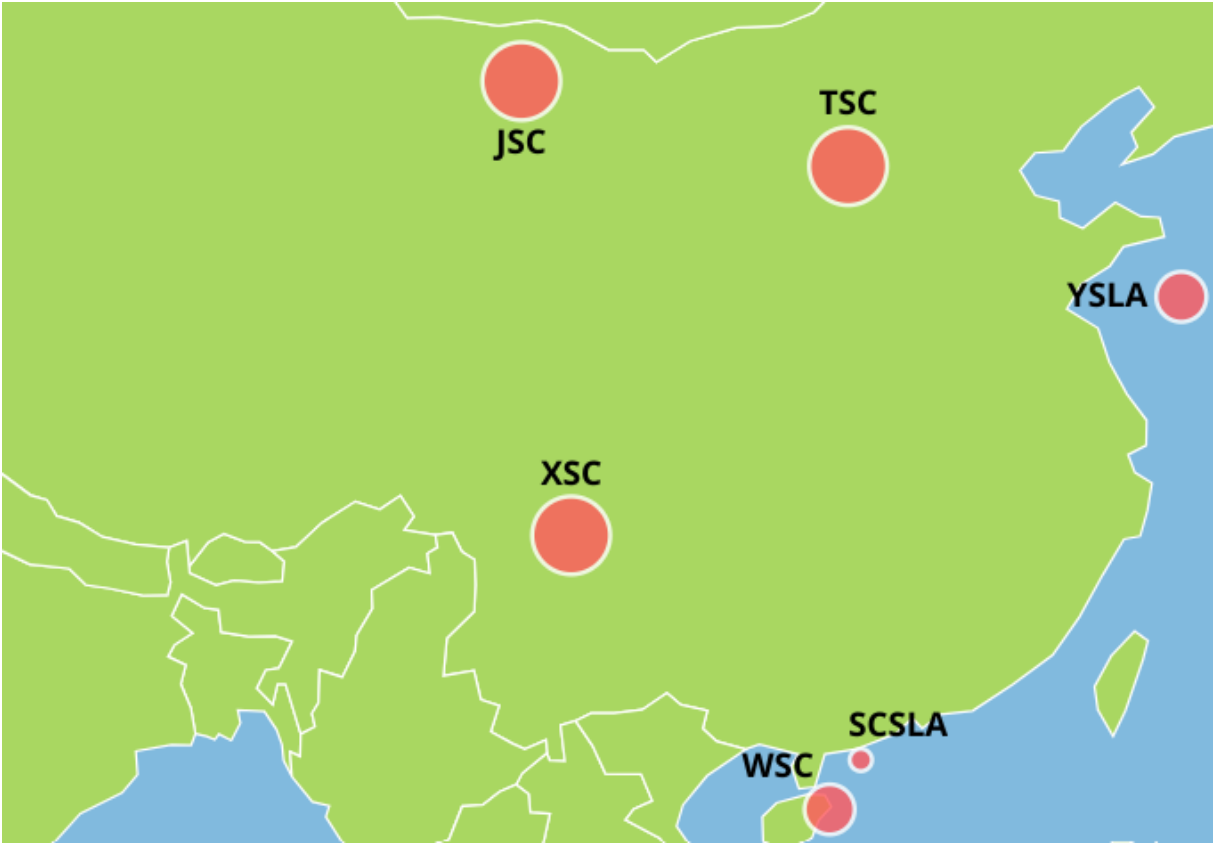


Figure 2.10: Map of the launch sites in China.

Russia

The Russian Federation, through Roscosmos, continues to operate and modernize a launch infrastructure inherited from the Soviet Union. This legacy provides Russia with high-cadence launch capabilities, complemented by ongoing modernization efforts aimed at reducing reliance on older, toxic propellants.

The Russian launch capability is historically defined by two main launcher families:

- The *Soyuz* family. This well-known launcher traces its heritage directly to the *R-7 Semyorka*, the world's first Intercontinental Ballistic Missile (ICBM) developed in the 1950s. Current versions (e.g., *Soyuz 2.1a* and *2.1b*) are two-stage vehicles, often equipped with four liquid-propellant side boosters, all using Kerosene (RP-1) and Liquid Oxygen (LOx). The *Soyuz* is globally recognised for its high reliability and versatility, capable of delivering payloads to all orbital regimes, often employing an additional upper stage, such as *Fregat* or *Volga*.
- The *Proton* family. This heavy-lift system, known for its reliance on highly toxic hypergolic fuels, is gradually being retired. The production on the *Proton* rocket is ceasing as the new *Angara* launch vehicle comes on line and becomes operational [29]. *Angara* represents Russia's primary modernization effort, being a modular, two-stage vehicle capable of replacing both the *Soyuz* and *Proton* families. It uses cleaner propellants and is designed to reduce environmental and operational hazards associated with traditional vehicles.

The launchers discussed are detailed in Table 2.7.

Table 2.7: List of Russia launchers. (Source: [26])

Launcher Name	Version	Stages	Lift-off mass (t)	Payload (kg)
Soyuz	2.1a	2 + 4 boosters	312	LEO: 7020 SSO: 1100
	2.1b	2 + 4 boosters	312	LEO: 8200 SSO: 4230
Proton	M	3	705	LEO: 23000 GTO: 6300
Angara	A1.2	2	172	LEO: 3800 GTO: 4000
	A5	2 + 4 boosters	790	LEO: 24500 SSO: 7500

Roscosmos operates launch facilities located both within and outside the Russian Federation, enabling access to broad range of orbital inclinations:

- Baikonur Cosmodrome. Located in Kazakhstan and historically known as the Tyuratam Missile and Space Complex (TTMTR) [30], Baikonur is the most iconic spaceport in the world, serving as the launch point for *Sputnik 1* and *Vostok 1*. It remains a key operational site for the *Soyuz* launch system and hosted the final missions of the *Proton-M*. From Baikonur, a wide range of orbital inclinations is achievable (from equatorial (0°) up to 99°).

- Plesetsk Cosmodrome. Also referred to as the Plesetsk Missile And Space Complex (PKMTR). Located at nearly 63° latitude in the north, Plesetsk is the northernmost orbital launch site in the world. It is the primary base for the *Angara* family and *Soyuz 2* operations. Its location makes it ideally suited for high-inclination orbits, including (SSO) and Molniya orbits (63.4°).
- Vostochny Cosmodrome (VOSTO). Situated in far-eastern Russia, Vostochny is the newest operational launch site, active since 2016. It currently supports *Soyuz 2* missions and is intended to become the future primary launch site for the *Angara* family. Its geography favours LEO and SSO missions, typically for inclinations between 87° and 98° .

Table 2.8 details the coordinates of the mentioned launch sites, and Figure 2.11 presents the location of the principal Russian orbital launch complexes.

Table 2.8: List of launch sites in Russia. (Source: [26])

Launch Site Name	Latitude	Longitude
Baikonur Cosmodrome (TTMTR)	$45^{\circ} 57' 54.00''$ N	$63^{\circ} 18' 18.00''$ E
Plesetsk Cosmodrome (PKMTR)	$62^{\circ} 55' 40.28''$ N	$40^{\circ} 34' 29.43''$ E
Vostochny Cosmodrome (VOSTO)	$51^{\circ} 50' 52.22''$ N	$128^{\circ} 21' 8.98''$ E



Figure 2.11: Map of the launch sites in Russia.

Other countries

Although the U.S., China, and Russia currently lead the world in annual launch cadence, as illustrated in Figure 2.12, several other nations and multinational organisations maintain independent and continuous access to orbit. The most distinguished among them are Europe, Japan, and India.

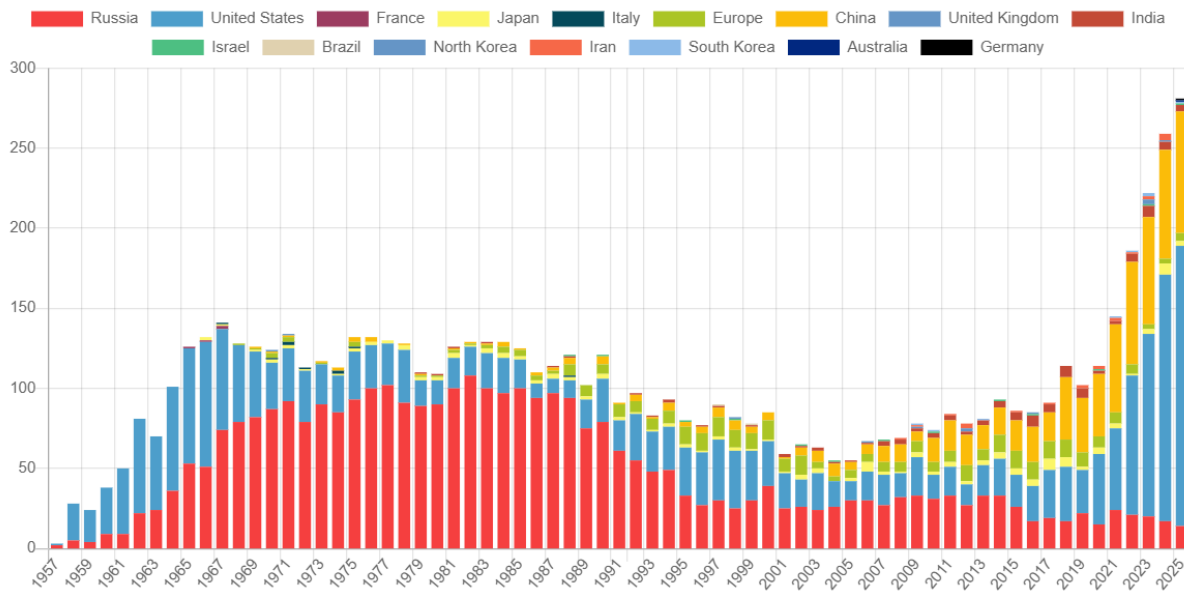


Figure 2.12: Orbital launches by year and country. (Source: *SpaceStatsOnline.com*)

The primary European launch provider is Arianespace, operating under the European Space Agency (ESA) framework. European access to orbit relies on two main launcher families:

- *Ariane* family. This has traditionally been the heavy-lift launcher (*Ariane 5*), now transitioning to its successor, the *Ariane 6*.
- *Vega* family. Developed by the Italian company Avio, the *Vega* and its upgraded version, the *Vega-C*, cover the small-to-medium payload market.

European launches are conducted exclusively from the Guiana Space Centre (GSC) in French Guiana. Located at approximately 5° latitude, this site provides a geographical advantage, benefiting from the Earth's rotational velocity and making it highly efficient for low-inclination and geostationary missions.

The Indian Space Research Organisation (ISRO) maintains an independent launch portfolio characterised by versatile, multi-stage vehicles using a combination of solid, hypergolic and cryogenic propulsion. Its two most relevant launchers are the *Polar Satellite Launch*

Vehicle (PSLV), primarily used for launching satellites into polar orbits or SSO, and the *Launch Vehicle Mark Three (LVM3)*, designed for heavy missions and the future crewed programme.

India uses a single launch site, the Satish Dhawan Space Centre (SDSC), located at low latitude on the southeastern coast. This position allows ISRO to achieve orbits ranging from equatorial GEO to high-inclination polar orbits.

Japan, through the Japan Aerospace Exploration Agency (JAXA), operates medium-lift and smaller solid-propellant systems. The country is transitioning from the hydrogen-fuelled *H-II* family to its successor, the *H3*, while maintaining solid-rocket launch capabilities with systems such as:

- *Epsilon*, capable of launching payloads of up to 1400 kg into LEO.
- *S-Series* launchers, including the *SS-520-5*, which became the smallest orbital rocket to date, successfully placing a 4 kg satellite into orbit in February 2018 [31].

Japan operates two primary launch sites, the Tanegashima Space Center (TNSC) and the Uchinoura Space Centre. Their southern locations allow launches eastward for low-inclination orbits and southward for SSO access.

Table 2.9 details the coordinates of the mentioned launch sites, while the launchers discussed are detailed in Table 2.10

Table 2.9: List of main launch sites in other nations. (Source: [26])

Launch Site Name	Latitude	Longitude
Guiana Space Centre (GSC)	5° 14' 22.66" N	52° 46' 6.80" O
Satish Dhawan Space Centre (SDSC)	13° 43' 12.00" N	80° 13' 48.00" E
Tanegashima Space Cente (TNSC)	30° 24' 3.45" N	130° 58' 39.27" E
Uchinoura Space Center	31° 15' 3.45" N	131° 4' 56.15" E

Table 2.10: List of main launchers from other nations. (Source: [26])

Country/Region	Launcher Name	Version	Stages	Lift-off mass (t)	Payload (kg)
Europe	Ariane 5	ECA	2 + 2 boosters	777	LEO: 21000 GTO: 10865
		A62	2 + 2 boosters	530	LEO: 10350 GTO: 4500
	Ariane 6	A64	2 + 4 boosters	860	LEO: 21650 GTO: 11500
		-	4	137	LEO: 2500 SSO: 1450
	Vega	C	4	210	SSO: 2300
India	PSLV	CA	4	230	LEO: 2100 SSO: 1100
		DL	4 + 2 boosters	260	LEO: <3800 SSO: 1260
		XL	4 + 6 boosters	320	LEO: 3800 GTO: 1300
	LVM3	-	3	137	LEO: 10000 GTO: 4000
Japan	HII	B	2 + 4 boosters	531	GTO: 8000
	H3	24	2 + 4 boosters	574	SSO: 6000 GTO: 7900
	Epsilon	-	3	95.4	LEO: 700 SSO: 590
	S-Series	520-5	3	2.5	LEO: 4

2.3.2. Launch phases and events

The objective of achieving a stable orbit requires the launch vehicle to execute a precise and energy-intensive trajectory capable of overcoming both gravity and aerodynamic drag. During ascent, the vehicle must escape the dense lower atmosphere and reach the target orbital altitude with a high enough horizontal velocity to remain in constant free-fall. To achieve the required orbital velocity (≈ 7500 m/s), the ascent is typically divided into distinct phases governed by different control strategies and marked by key mission events.

Although launcher profiles may vary between vehicles due to differences in propulsion architecture, staging configuration, and mission objectives, most launchers follow a comparable sequence, generally transitioning between an open-loop and a closed-loop system. The initial stages are often guided by a pre-programmed open-loop trajectory or a simple initial pitch programme, while subsequent stages rely on a closed-loop control law

that actively adjusts guidance based on real-time flight conditions and target parameters [32, 33]. An essential element of this transition is the gravity turn, a manoeuvre that takes advantage of the gravity to rotate the velocity vector towards the horizontal [34]. This reduces steering losses and propellant consumption, as the vehicle maintains a near-zero angle of attack and requires minimal control input. Figure 2.13 illustrates the characteristic curved trajectory associated with this manoeuvre.



Figure 2.13: Trajectory during gravity turn. (Source: SpaceX)

The nominal ascent of an orbital launch vehicle is commonly described by the following sequence of phases [35]: vertical ascent, pitch-over, gravity turn, and optimal guidance.

Within these phases, a typical multi-stage launch includes a series of characteristic mission events:

1. Engine ignition. Prior to lift-off, the main engines or solid motors are ignited and ramped up to operational thrust levels. Depending on the vehicle and propellant type, this may occur slightly before T0 to allow thrust verification through telemetry.
2. Lift-off and vertical climb. Once thrust exceeds the vehicle weight and ground checks confirm nominal performance, the launch tower releases the vehicle. The initial trajectory is typically close to vertical, although slight tower-clearance offsets may be introduced to minimise collision risk.
3. Pitch-over and roll alignment. Shortly after clearing the tower, the rocket performs a roll to align with the mission azimuth, followed by a deliberate pitch-over manoeuvre that reduces the flight-path angle. This marks the onset of the gravity turn, gradually prioritising horizontal acceleration.
4. Maximum dynamic pressure (Max-q). As the vehicle accelerates through the denser

layers of the atmosphere, it reaches a point of peak aerodynamic loading. This event often coincides with or occurs shortly after the vehicle reaches the speed of sound. Throttle reductions may be applied to limit structural stress.

5. Stage separation and upper-stage ignition. Once the first stage depletes its propellant, engines shut down and the stage is jettisoned. The upper stage then ignites to continue acceleration, usually already above the bulk of the atmosphere. Additional boosters or stages follow analogous separation sequences when applicable.
6. Fairing or abort-system jettison. When aerodynamic and thermal loads become negligible and the payload is no longer at risk from dense atmospheric interactions, the protective shroud (and, if present, the launch abort tower) is jettisoned to reduce mass.
7. Terminal guidance. Near the end of powered ascent, guidance algorithms refine the trajectory to precisely reach the desired injection conditions, adjusting thrust vectors to minimise targeting error.
8. Orbital insertion and payload deployment. Following main engine cut-off, the upper stage may perform additional burns, such as circularisation or escape manoeuvres, before releasing the payload. At this point, the launch sequence is considered complete.

Most of these events are illustrated in Figure 2.14.

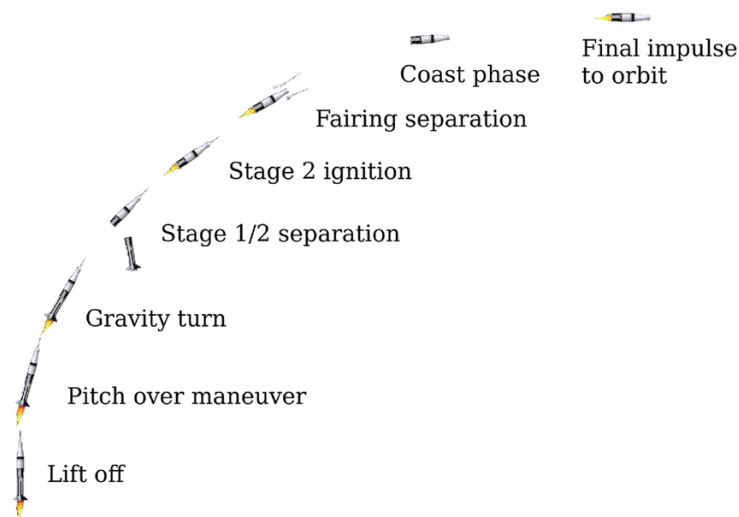


Figure 2.14: Typical flight sequence of a space launcher. (Source: [36])

3 | Trajectory modelling and simulation

In this chapter, a modelling framework is developed to simulate the boost phase of the three representative classes of launch systems previously discussed: ballistic missiles (BM), anti-satellite (ASAT) systems, and orbital launch (OL) vehicles. Although all three share common physical principles, their operational profiles differ in terms of guidance strategy, propulsion, vehicle architecture, and mission objectives, requiring specific adaptations in the dynamic formulation of the simulation model.

The chapter begins with a formal definition of the problem, including a set of assumptions and simplifications adopted to obtain a sufficiently accurate model. The equations of motion governing the three systems are then presented, distinguishing between the BM/ASAT formulation, both operating in similar flight regimes and with comparable thrust profiles, and the orbital launchers case, where higher system knowledge enables a more detailed parametrisation. Subsequently, the associated guidance and control laws for each system are defined.

The second part of the chapter focuses on the implementation of these models within a custom software framework developed in Java. The simulation workflow used to generate representative trajectories across a wide range of mission conditions and vehicle configurations is described.

Finally, the chapter concludes with model validation through four case studies followed by the generation of a large trajectory database that will later be used to support the system identification development presented in Chapter 4.

3.1. Problem formulation

This section details the specific assumptions, equations of motion, and guidance and control laws applied to each vehicle class. These elements constitute the basis for simulating trajectories and generating a representative database.

3.1.1. Assumptions and physical hypotheses

The following assumptions and simplifications are adopted for the modelling of the boost phase dynamics:

- Ballistic Missiles (BM):
 - Only ground-launched systems are considered.
 - Short-, Medium-, and Intermediate-Range Ballistic Missiles are modelled as single-stage vehicles, while ICBMs are represented with three stages.
 - A single re-entry vehicle (RV) is considered per missile; MIRVs are not modelled.
 - Guidance is only active during the boost phase. After stage burnout, no thrust or control action is applied during the ballistic phase.
 - Solid propellant is assumed for all stages.
 - Within each stage, the exhaust velocity U is considered constant.
 - The normalised mass burn rate q (mass flow per initial stage mass) is constant.
 - A constant ballistic coefficient c is assumed per stage.
- Anti-Satellite (ASAT) systems:
 - Only Direct-Ascent ASATs (DA-ASATs) are considered, hereafter referred to simply as ASATs.
 - Multi-stage configurations are modelled, according to typical operational designs.
 - Solid propellant is assumed for all stages.
 - The same stage-level simplifications as for BM are applied: constant exhaust velocity U , normalised burn rate q , and ballistic coefficient c per stage.

- Orbital Launch (OL) vehicles:
 - Multi-stage vehicles are considered, with propulsion parameters specific to each stage.
 - Orbital injection is assumed to occur at periapsis or apoapsis.

Together with these specific assumptions for each vehicle class, all vehicles are considered axially symmetric and cylindrical, and aerodynamic effects are accounted for only during the atmospheric portion of the boost phase (below approximately 100 km of altitude). Beyond this altitude, aerodynamic effects are negligible.

3.1.2. Equations of motion

During the boost phase, the vehicle is affected by thrust, aerodynamic drag, and gravity. When adopting the commonly used Earth-Centred Inertial (ECI) Coordinate System (CS) to describe the vehicle's motion, no more forces are considered. Therefore, according to Newton's second law, the translational dynamics governing the motion in the boost phase, together with the mass variation due to propellant consumption, are:

$$\begin{cases} \dot{\mathbf{r}}(t) = \mathbf{v}(t) \\ \dot{\mathbf{v}}(t) = \mathbf{a}_{thrust}(t) + \mathbf{a}_{drag}(t) + \mathbf{a}_{gravity}(t) \\ \dot{m}(t) = -\dot{m}_p(t) \end{cases} \quad (3.1)$$

where $\mathbf{r}(t) = [x(t), y(t), z(t)]^T$ and $\mathbf{v}(t) = [v_x(t), v_y(t), v_z(t)]^T$ represent the position and velocity vectors in ECI coordinates at time t , respectively, while $\mathbf{a}_{thrust}(t)$, $\mathbf{a}_{drag}(t)$ and $\mathbf{a}_{gravity}(t)$ denote the corresponding thrust, aerodynamic drag, and gravity accelerations. The term $\dot{m}_p(t)$ represents the propellant mass flow rate.

These expressions are particularised for each vehicle class according to the modelling assumptions defined in Section 3.1.1.

Dynamic model of BMs/ASATs

For ballistic missiles and ASAT systems, the assumptions introduced in Section 3.1.1 lead to the following acceleration components:

$$\begin{cases} \mathbf{a}_{thrust}(t) = \frac{Uq}{1-qt} \hat{\mathbf{u}}_{thrust}(t) \\ \mathbf{a}_{drag}(t) = -\frac{1}{2} \frac{\rho(h(t))}{c} \|\mathbf{v}_{rel}(t)\| \mathbf{v}_{rel}(t) \\ \mathbf{a}_{gravity}(t) = -\frac{\mu}{\|\mathbf{r}(t)\|^3} \mathbf{r}(t) \end{cases} \quad (3.2)$$

where $\hat{\mathbf{u}}_{thrust}(t)$ is the unit vector in the direction of thrust, $\rho(h(t))$ is the air density at altitude $h(t)$ following the U.S. Standard Atmosphere (1976) [37], $\mathbf{v}_{rel}(t)$ is the vehicle velocity relative to the atmosphere, and μ is Earth's gravitational parameter. The variable t represents the time elapsed since the ignition of the current stage, such that the instantaneous mass evolves according to $m(t) = m_0(1 - qt)$, with m_0 being the initial mass of the stage.

A maximum allowable thrust acceleration is imposed. Modern ballistic missiles can theoretically reach peak accelerations in the range of 25-50 g [38], while the final accelerations typically observed in operational systems are closer to 15 g [39]. Therefore, a limit of 20 g is selected as a reasonable compromise, consistent with realistic missile performance and theoretical studies.

Substituting the above acceleration expressions into Eq. (3.1) yields the evolution of missile's position, velocity and mass as:

$$\begin{bmatrix} \dot{x}(t) \\ \dot{y}(t) \\ \dot{z}(t) \\ \dot{v}_x(t) \\ \dot{v}_y(t) \\ \dot{v}_z(t) \\ \dot{m}(t) \end{bmatrix} = \begin{bmatrix} v_x(t) \\ v_y(t) \\ v_z(t) \\ \frac{Uq}{1-qt} \hat{u}_{thrust,x}(t) - \frac{1}{2} \frac{\rho(h(t))}{c} \|\mathbf{v}_{rel}(t)\| v_{rel,x}(t) - \frac{\mu}{\|\mathbf{r}(t)\|^3} x(t) \\ \frac{Uq}{1-qt} \hat{u}_{thrust,y}(t) - \frac{1}{2} \frac{\rho(h(t))}{c} \|\mathbf{v}_{rel}(t)\| v_{rel,y}(t) - \frac{\mu}{\|\mathbf{r}(t)\|^3} y(t) \\ \frac{Uq}{1-qt} \hat{u}_{thrust,z}(t) - \frac{1}{2} \frac{\rho(h(t))}{c} \|\mathbf{v}_{rel}(t)\| v_{rel,z}(t) - \frac{\mu}{\|\mathbf{r}(t)\|^3} z(t) \\ -qm_0 \end{bmatrix} \quad (3.3)$$

During the coast phases (i.e., ballistic and re-entry phases), the same translational equations of motion apply, with the omission of the thrust acceleration term, as no propulsion is active beyond burnout. This formulation is applied only to ballistic missiles, since

the other vehicle categories operate solely within the boost phase, and no meaningful post-boost trajectory propagation is required for these classes.

Dynamic model of orbital launchers

For orbital launch vehicles, the thrust acceleration at each time step is computed using the current vehicle mass and the relevant propulsion parameters of the active stage. For each stage, the available data typically include the thrust (T) at sea level, the thrust in vacuum, the specific impulse (I_{sp}) at sea level, and the specific impulse in vacuum.

The specific impulse varies with atmospheric pressure, an effect particularly significant for first-stage engines. It reaches its minimum at sea level, where atmospheric pressure is highest, and its maximum in vacuum. Then, by evaluating the atmospheric pressure at the current altitude, the specific impulse can be estimated using:

$$I_{sp}(h) = I_{sp,sea} \frac{p(h)}{p(0)} + I_{sp,vac} \left(1 - \frac{p(h)}{p(0)} \right) \quad [s] \quad (3.4)$$

where $p(0)$ is the atmospheric pressure at sea level

This expression allows the computation of the specific impulse for engines and boosters at any altitude, ensuring the appropriate value is obtained for the current flight conditions and stage configuration.

The thrust produced by each stage engine is also evaluated as a function of the atmospheric pressure. Given the thrust in vacuum, T_{vac} , the instantaneous thrust at altitude h can be estimated as:

$$T(h) = T_{vac} - p(h)A_e \quad [N] \quad (3.5)$$

where A_e is the nozzle exit area. This area can be approximated by evaluating the same expression at sea level, where:

$$T_{sea} = T_{vac} - p(0)A_e \quad (3.6)$$

Solving for A_e yields:

$$A_e = \frac{T_{vac} - T_{sea}}{p(0)} \quad (3.7)$$

With this expression, the thrust decreases from its vacuum value to its sea-level value as atmospheric pressure increases.

Once the thrust is evaluated, the corresponding thrust acceleration is obtained as:

$$\mathbf{a}_{thrust}(t) = n_e \frac{T(h(t))}{m(t)} \hat{\mathbf{u}}_{thrust}(t) \quad (3.8)$$

where n_e is the number of main engines, $T(h(t))$ is the thrust produced by each engine at altitude $h(t)$, $m(t)$ is the current mass of the vehicle, and $\hat{\mathbf{u}}_{thrust}(t)$ is the unit thrust direction.

If boosters are present, their contribution to the total acceleration should be added to that of the main engines. The resulting thrust acceleration can be written as:

$$\begin{aligned} \mathbf{a}_{thrust}(t) &= n_e \frac{T(h(t))}{m(t)} \hat{\mathbf{u}}_{thrust}(t) + \mathbf{a}_{thrust,boosters}(t) \\ &= \left(n_e \frac{T(h(t))}{m(t)} + n_b \frac{T_b(h(t))}{m(t)} \right) \hat{\mathbf{u}}_{thrust}(t) \end{aligned} \quad (3.9)$$

where n_b is the number of boosters and $T_b(h(t))$ is the thrust provided by a booster at altitude $h(t)$, calculated using Eq. 3.5 with the specific parameters of the boosters.

The throttling capabilities of some liquid-propellant engines, as well as the modulation provided by the fuel grain geometry in solid rocket boosters, are modelled in a simplified way. Specifically, the acceleration produced by the launcher is capped at approximately 4 g for configurations without boosters and 4.5 g when boosters are present. These maximum accelerations are typical of many existing launchers, since higher values can pose risks to both the payload (including crewed missions) and the vehicle's structural integrity.

The propellant mass flow rate of an engine, $\dot{m}_{engine}(t)$, at time t is evaluated by inverting the thrust definition formula:

$$\dot{m}_{engine}(t) = \frac{T(h(t))}{I_{sp}(h(t)) \cdot g_0} \quad (3.10)$$

where $g_0 = 9.81 \text{ m/s}^2$ is Earth's acceleration of gravity at sea level, $T(h(t))$ is the engine thrust at altitude $h(t)$, and $I_{sp}(h(t))$ is the specific impulse at the same altitude.

Therefore, the total mass flow rate, at time t , is:

$$\dot{m}(t) = -\dot{m}_{main}(t) - \dot{m}_{boosters}(t) = -n_e \dot{m}_{engine,main}(t) - n_b \dot{m}_{engine,boosters}(t) \quad (3.11)$$

where the negative sign indicates a reduction in vehicle mass over time.

Atmospheric drag is computed based on vehicle's altitude, relative velocity to the atmosphere, current cross-sectional area, and drag coefficient. The dynamic pressure $q(t)$ is given by:

$$q(t) = \frac{1}{2} \cdot \rho(h(t)) \cdot v_{rel}^2(t) \quad (3.12)$$

and the speed of sound at a given altitude is:

$$a = \sqrt{R_{air} \cdot \gamma \cdot T(h(t))} \quad (3.13)$$

where $R_{air} = 287.5 \text{ J/kgK}$ is the specific gas constant for air, $\gamma = 1.4$ is the adiabatic index, and $T(h(t))$ is the atmospheric temperature at altitude $h(t)$.

The Mach number M is then computed as the ratio of the relative velocity to the local speed of sound. This Mach number is used to interpolate the drag coefficient C_d from the curve provided in [40], which is based on the *Saturn V* and serves as a representative approximation for a generic launch vehicle. The curve is shown in Figure 3.1.

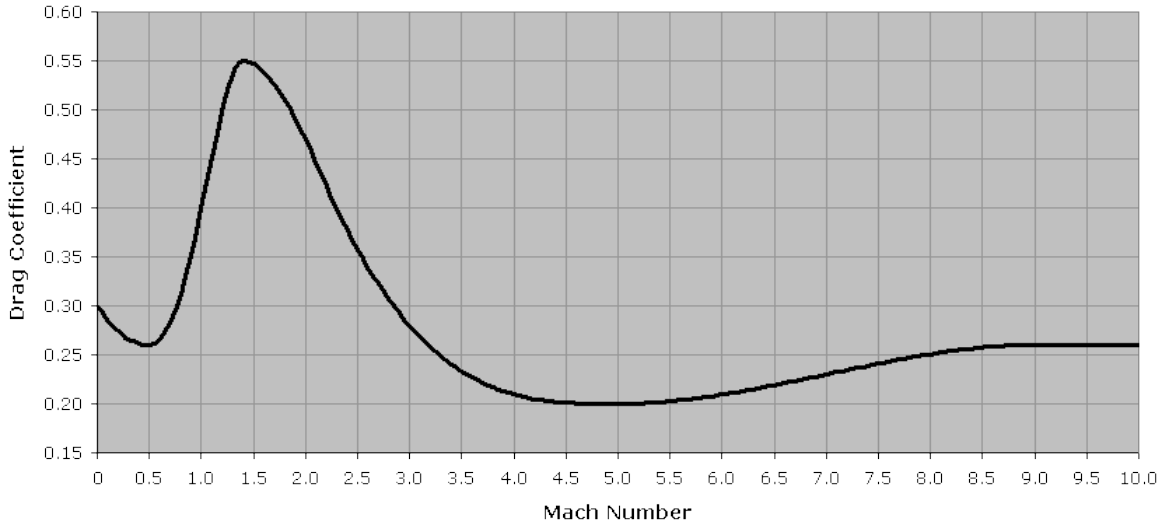


Figure 3.1: Variation of drag coefficient with Mach number. (Source: [40])

Finally, the drag acceleration is expressed as:

$$\mathbf{a}_{drag}(t) = -\frac{1}{2} \frac{\rho(h(t)) C_d A}{m(t)} \|\mathbf{v}_{rel}(t)\| \mathbf{v}_{rel}(t) \quad (3.14)$$

where A is the cross-sectional area of the stage, calculated from the known diameter.

Gravity acceleration follows the same expression as in Eq. (3.2):

$$\mathbf{a}_{gravity}(t) = -\frac{\mu}{\|\mathbf{r}(t)\|^3}\mathbf{r}(t) \quad (3.15)$$

Therefore, substituting all accelerations into Eq. (3.1) yields the evolution of launcher's position, velocity and mass as:

$$\begin{bmatrix} \dot{x}(t) \\ \dot{y}(t) \\ \dot{z}(t) \\ \dot{v}_x(t) \\ \dot{v}_y(t) \\ \dot{v}_z(t) \\ \dot{m}(t) \end{bmatrix} = \begin{bmatrix} v_x(t) \\ v_y(t) \\ v_z(t) \\ \left(n_e \frac{T(h(t))}{m(t)} + n_b \frac{T_b(h(t))}{m(t)} \right) \hat{u}_{thrust,x}(t) - \frac{1}{2} \frac{\rho(h(t))C_d A}{m(t)} \|\mathbf{v}_{rel}(t)\| v_{rel,x}(t) - \frac{\mu}{\|\mathbf{r}(t)\|^3} x(t) \\ \left(n_e \frac{T(h(t))}{m(t)} + n_b \frac{T_b(h(t))}{m(t)} \right) \hat{u}_{thrust,y}(t) - \frac{1}{2} \frac{\rho(h(t))C_d A}{m(t)} \|\mathbf{v}_{rel}(t)\| v_{rel,y}(t) - \frac{\mu}{\|\mathbf{r}(t)\|^3} y(t) \\ \left(n_e \frac{T(h(t))}{m(t)} + n_b \frac{T_b(h(t))}{m(t)} \right) \hat{u}_{thrust,z}(t) - \frac{1}{2} \frac{\rho(h(t))C_d A}{m(t)} \|\mathbf{v}_{rel}(t)\| v_{rel,z}(t) - \frac{\mu}{\|\mathbf{r}(t)\|^3} z(t) \\ -n_e \dot{m}_{engine,main}(t) - n_b \dot{m}_{engine,boosters}(t) \end{bmatrix} \quad (3.16)$$

3.1.3. Guidance profiles and control laws

For each system, the initial pitch profile, characterised by the duration of vertical ascent and the parameters defining the pitch-over and gravity turn manoeuvres, is specified together with the control laws used to get the final target. These thrust profiles determine how the vehicle orients its acceleration over time to achieve the desired trajectory while satisfying physical constraints and mission requirements.

It is important to note that the guidance profiles presented in this section are designed to provide a generic and broadly applicable framework suitable for a wide range of missions. If the objective were to optimise the trajectory for a specific missile configuration or mission scenario, a dedicated guidance law would be required. However, for the purpose of a generalised trajectory study, the guidance strategy cannot be specific for each individual configuration and must instead rely on a generalised profile such as the one adopted in this work.

Pitch profile of ballistic missiles

Before describing the guidance profile applied to ballistic missiles, it is important to clarify that one of the mission parameters is the flight-path angle (γ) with respect to the local horizontal at the end of the boost phase, illustrated in Figure 3.2. This final flight-path

angle is imposed by mission requirements, as it directly determines the subsequent ballistic trajectory and, consequently, the achieved range. In other words, the same missile can reach different impact points simply by adjusting the final flight-path angle of the boost phase. Therefore, the objective of the guidance and control system during the boost phase is to ensure that the missile reaches the required end-of-boost conditions (altitude, velocity magnitude, and pitch angle) necessary to follow the desired ballistic path. For this reason, the final pitch angle (θ_{end}) is treated as a tunable parameter in the trajectory design process, as it implies a final flight-path angle that is typically near it.

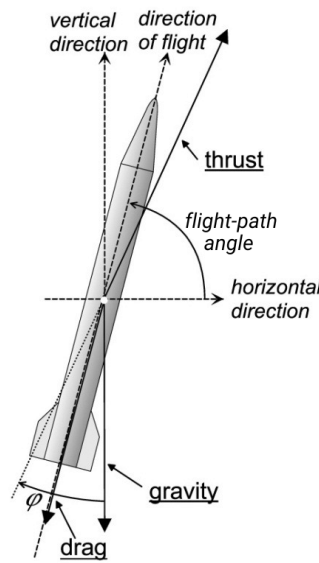


Figure 3.2: Forces acting during missile's boost phase. (Source: [41])

The adopted guidance strategy differs between Short-, Medium-, and Intermediate-Range Ballistic Missiles (SRBMs/MRBMs/IRBMs), modelled as single-stage vehicles, and Intercontinental Ballistic Missiles (ICBMs), which typically employ three stages.

For single-stage missiles, a vertical ascent duration of approximately 10 seconds is assumed, whereas for ICBMs this is reduced to 5 seconds, reflecting their higher thrust-to-weight ratios (TWR) and the reduced need for extended vertical ascent before initiating the gravity turn manoeuvre.

During this phase, the missile follows a straight trajectory, maintaining a thrust direction aligned with the launch elevation. After this initial ascent, an initial kick manoeuvre is applied: over an interval of 2 seconds, the thrust elevation transitions smoothly from its initial value to a direction 2.5° below the relative velocity elevation. From this point forward, the guidance profile diverges depending on whether the missile is single-stage or multi-stage.

The guidance sequence for single-stage missiles is implemented as follows:

1. The thrust direction is maintained 2.5° below the relative velocity elevation for 5 seconds.
2. A gravity turn segment of 30 seconds is executed, aligning the thrust with the relative velocity vector.
3. Once the gravity turn is completed and the relative velocity is sufficiently high, a continuous pitch-down command is applied to reach the final pitch angle (θ_{end}). This final descent of the thrust elevation follows a quadratic profile described by:

$$\theta(t) = \theta_{start} + (\theta_{end} - \theta_{start}) \cdot \left(1 - \left(1 - \frac{t - t_{start}}{t_{end} - t_{start}} \right)^2 \right) \quad (3.17)$$

Figure 3.3 illustrates this generic commanded pitch profile for a single-stage missile trajectory with a final pitch angle of 30° .

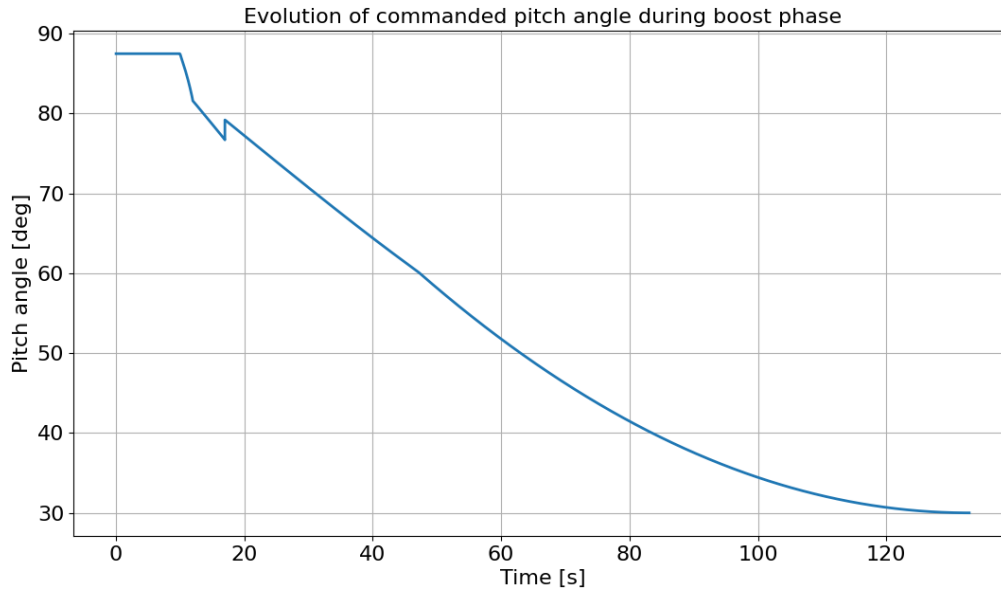


Figure 3.3: Generic commanded pitch profile for single-stage ballistic missiles.

The strategy for ICBMs differs significantly:

1. After the kick manoeuvre, the thrust direction is pointed 2.5° below the relative velocity elevation for approximately 15 seconds. This extended phase is specifically intended to reduce the velocity elevation sufficiently, promoting an effective gravity

turn that leads to a lower final pitch angle.

2. The missile then enters a gravity turn phase until atmospheric exit.
3. Once above the atmosphere, a proportional control law is applied to steer the missile toward the required final pitch angle with respect to the horizontal. This law serves an analogous purpose to the quadratic descent profile used for single-stage systems.

Figure 3.4 illustrates the generic commanded pitch profile for a multi-stage ICBM trajectory with a final pitch angle of 15° .

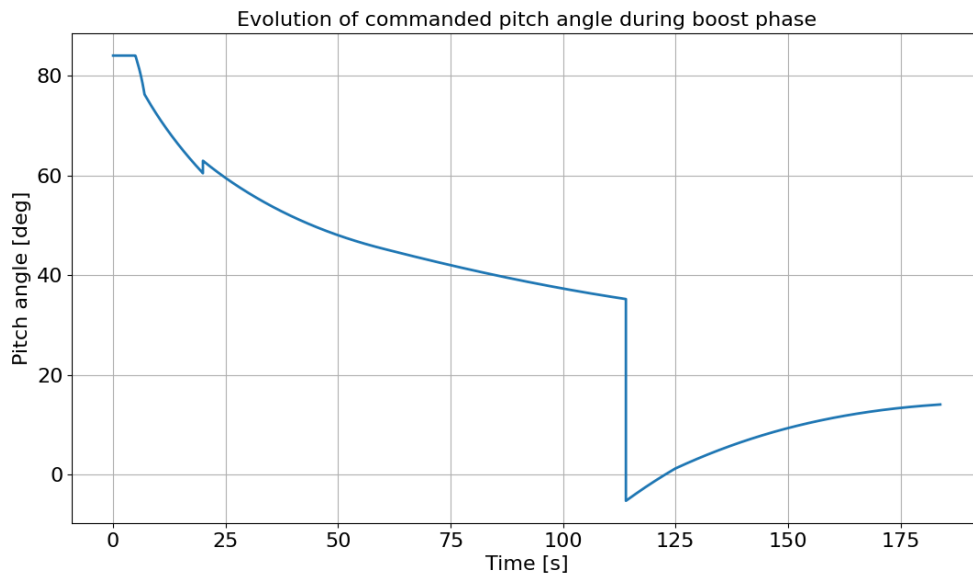


Figure 3.4: Generic commanded pitch profile for multi-stage ICBMs.

The angles, durations, and manoeuvre profiles described above have been selected to be consistent with typical boost phase lengths reported for these missile classes. This ensures that the imposed attitude changes are feasible and that the resulting angles of attack remain within acceptable operational limits for this type of vehicle.

The proportional controller implemented for the ICBM case represents a compromise between simplicity and accuracy. Although the commanded change in thrust elevation appears instantaneous in the model, it is implicitly assumed that the vehicle, once outside the atmosphere and during the final stages of powered flight, is capable of reorienting its thrust vector toward the required pitch angle within a short period of time (e.g., using auxiliary attitude control thrusters). This reorientation has been simplified as an instantaneous adjustment.

Pitch profile of ASATs

For ASAT systems, a guidance approach similar to that of ballistic missiles is adopted. The strategy is designed to deliver the interceptor to a prescribed altitude with a specific final velocity vector. Although the absolute velocity achieved by the vehicle is relatively modest (consistent with its suborbital nature [15]), the relative velocity with respect to the target can be extremely high, depending on the impact geometry.

Furthermore, ASAT engagements must carefully account for post-impact orbital debris dispersion. To minimise the injection of fragments into higher orbits, the elevation of the relative velocity at impact becomes a critical parameter. For this reason, the pitch angle with respect to the local horizontal at the end of the boost phase is also treated as a mission parameter.

With this, a guidance structure analogous to that of single-stage ballistic missiles is employed for ASAT systems:

1. A vertical ascent lasting approximately 5 seconds is assumed. During this phase, the interceptor follows a straight trajectory, maintaining the thrust direction aligned with the launch elevation.
2. After this initial ascent, the initial kick manoeuvre is performed: over a 2-second interval, the thrust elevation transitions smoothly from its initial value to a direction 2.5° below the relative velocity elevation.
3. The thrust direction is maintained for an additional 5 seconds.
4. A gravity turn segment of 15 seconds is executed, aligning the thrust vector with the relative velocity vector.
5. Once this gravity turn phase is completed and the relative velocity is sufficiently high, the commanded pitch angle decreases toward the required final value, following the quadratic profile introduced in Eq. (3.17).

Figure 3.5 illustrates the generic commanded pitch profile for an ASAT trajectory with a final pitch angle of 48° .

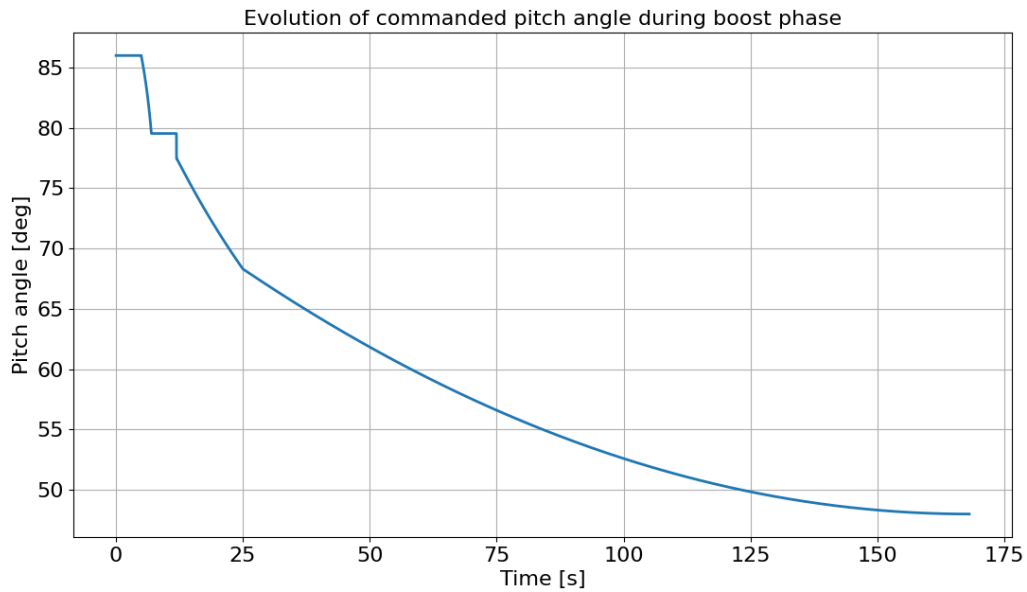


Figure 3.5: Generic commanded pitch profile for ASATs.

Pitch profile of orbital launchers

For orbital launch vehicles, the implemented control law is more complex than in the ballistic missile or ASAT cases, as the primary objective is not to achieve a specific flight-path angle but to reach a specific injection state. The target conditions are defined by a required injection altitude and velocity magnitude, together with a zero radial velocity component at burnout (implying injection at either periapsis or apoapsis). Consequently, the pitch profile is adjusted throughout the ascent to ensure that the vehicle reaches the desired orbital state at the end of the powered phase.

The initial pitch profile begins with a vertical ascent whose duration depends on the specific launcher, with typical operational values ranging from 10 to 20 seconds. Following this vertical rise, a smooth transition is commanded to reach the kick angle over an interval of 4 seconds. This kick angle is a key parameter that strongly influences the subsequent trajectory. It governs the aggressiveness of the gravity turn: a larger (more horizontal) kick produces a faster pitch-over, enabling insertion into lower orbits, while a more conservative kick supports injection into higher orbits.

Following this initial manoeuvre, the thrust direction is held at the kick angle for approximately 10 seconds before initiating the gravity turn. This segment aligns the thrust vector with the velocity vector, ensuring efficient acceleration and shaping of the ascent trajectory toward the required injection conditions. The duration of the gravity turn

segment depends on the specific launcher configuration and is treated as an adjustable parameter, typically constrained by the burn time of the first stage.

A minimum allowable pitch angle (θ_{min}) is established for each launcher. This constraint serves to prevent the vehicle's flight-path angle from becoming excessively shallow, especially at low altitudes where aerodynamic forces could become significant. Therefore, if the flight-path angle reaches this minimum value during the gravity turn, the control system commands a constant pitch angle equal to θ_{min} , maintaining the thrust vector at this fixed elevation.

Once the gravity turn is completed, the launcher transitions to an active closed-loop control phase.

Every operational launcher uses a specifically optimised guidance algorithm to achieve the intended orbital insertion conditions. The launchers of Arianespace, for instance, use the “bilinear tangent law” [32]. However, publicly available information on these algorithms is limited, and implementing multiple different real control schemes would introduce unnecessary complexity into the present work.

For these reasons, a generic and widely adopted guidance approach, the Powered Explicit Guidance (PEG) algorithm, is selected. PEG provides a closed-form approximation to the equations of motion during powered flight and generates a commanded pitch profile capable of steering the vehicle toward the required injection state with sufficient accuracy for a generic trajectory generator. The implemented version of the guidance equations is based on the formulation developed in [33].

As stated previously, the objective of this controller is to reach the orbital conditions given as input, specifically the norms of the orbital radius, r_{inj} , and orbital velocity, v_{inj} . PEG operates in discrete guidance cycles. At each cycle, the algorithm evaluates the current dynamical state and updates the commanded thrust direction so that, if maintained, it would lead the vehicle to the target orbital conditions at the estimated required burn time. The method relies on the linearised radial equation of motion, expressed in the radial–normal–circumferential (RNC) coordinate system.

The differential equation for the radius is:

$$\ddot{r} + \frac{\mu}{r^2} - \omega^2 r = a(t)(\hat{\mathbf{u}} \cdot \hat{\mathbf{r}}) \quad (3.18)$$

where $\hat{\mathbf{u}}$ represents the thrust direction unit vector. Considering $U = I_{sp}g_0$ as the exhaust

gases velocity, the characteristic time τ can be defined as:

$$\tau = \frac{U}{a(0)} \quad (3.19)$$

where $a(0)$ is the acceleration value at the PEG evaluation time. Then, during the time interval between two executions, t being the time elapsed from the evaluation time, the acceleration is expressed as:

$$a(t) = \frac{a(0)}{1 - \frac{t}{\tau}} \quad (3.20)$$

The proposed form for the guidance law of the radial component of the thrust is:

$$\hat{\mathbf{u}} \cdot \hat{\mathbf{r}} = A + Bt + \frac{\mu/r^2 - \omega^2 r}{a(t)} \quad (3.21)$$

Combining this with Eq. (3.18) gives:

$$\ddot{r} = a(t)(A + Bt) \quad (3.22)$$

Integrating Eq. (3.22) twice leads to an algebraic system solved for the values A and B , which allow for the vehicle to achieve the targeted values of $r(T)$ and $\dot{r}(T)$ at the estimated burnout time T , if within the vehicle's capabilities. The system takes the form:

$$\begin{bmatrix} b_0 & b_1 \\ c_0 & c_1 \end{bmatrix} \begin{bmatrix} A \\ B \end{bmatrix} = \begin{bmatrix} \dot{r}(T) - \dot{r}(0) \\ r(T) - r(0) - \dot{r}(0)T \end{bmatrix} \quad (3.23)$$

where the coefficients results from the integration of Eq. (3.20):

$$\begin{aligned} b_0 &= \int_0^T a(t)dt = -U \log(1 - T/\tau) \\ b_1 &= \int_0^T ta(t)dt = b_0\tau - UT \\ c_0 &= \int_0^T \left[\int_0^t a(s)ds \right] = b_0T - b_1 \\ c_1 &= \int_0^T \left[\int_0^t sa(s)ds \right] = c_0\tau - \frac{UT^2}{2} \end{aligned} \quad (3.24)$$

For the case of orbital insertion at the periapsis, the condition $\dot{r}(T) = 0$ simplifies the first equation. This also simplifies the evaluation of the norm of the angular momentum

at injection, which can be found as:

$$\|\mathbf{h}_{inj}\| = h = r_{inj}v_{inj} \quad (3.25)$$

The difference in angular momentum $\Delta h = h(T) - h(0)$ is required for the evaluation of the ideal Δv needed to reach the target conditions:

$$\Delta v = \frac{\left[\frac{\Delta h}{\bar{r}} + UT \left(\dot{u}_\theta + \ddot{u}_\theta \tau + \frac{\ddot{u}_\theta UT^2}{2} \right) \right]}{u_\theta + \dot{u}_\theta \tau + \ddot{u}_\theta \tau^2} \quad (3.26)$$

in which $\bar{r}$ is the mean value of the radius between the current and target states, and u_θ comes from the following approximate expression of the local horizontal component of the thrust direction unit vector:

$$\hat{\mathbf{u}} \cdot \hat{\boldsymbol{\theta}} = u_\theta + \dot{u}_\theta t + \ddot{u}_\theta t^2 \quad (3.27)$$

For the calculation of u_θ , the radial component of $\hat{\mathbf{u}}$ must first be determined:

$$u_r = A + C \quad (3.28)$$

where A is the value found by solving Eq. (3.23), and C is defined as:

$$C = \frac{\mu/r^2 - \omega^2 r}{a(0)} \quad (3.29)$$

For the derivatives of u_θ , the estimation of u_r at the expected burnout time T is also required:

$$u_r(T) = A + BT + \frac{\mu/r(T)^2 - \omega(T)^2 r(T)}{a(T)} \quad (3.30)$$

with $a(T) = \frac{a(0)}{1 - \frac{T}{\tau}}$, $r(T) = r_{inj}$, and $\omega(T)^2 r(T) = \frac{v_{inj}^2}{r_{inj}}$.

Therefore, once the coefficients A , B , and the estimated burnout time T are known, the commanded pitch angle is recovered from the radial thrust component. The commanded pitch (θ_{com}) is:

$$\sin \theta_{com} = A - (T_{prev} - T)B + C \quad (3.31)$$

where T_{prev} is the estimated burnout time from the previous guidance cycle.

Evaluating this expression at each guidance cycle produces a smooth pitch profile converging toward the required final conditions. A final check is executed before applying directly the obtained value. If the difference from the commanded pitch of the previous step exceeds 3° , the new commanded pitch angle becomes the previous one modified by 1.5° in the direction of the new command. This rate limit ensures the required angular velocity remains within acceptable limits.

The PEG algorithm thus provides a generic and consistent way of generating closed-loop pitch commands for a wide variety of launcher configurations. Its structure ensures sufficiently accurate orbital injection for the purposes of constructing a broad database of representative ascent trajectories, without the need to implement specific optimal controllers.

For further details over the derivation of the PEG algorithm and the extension to more stages using the calculus of variations, see [33].

Therefore, Figure 3.6 illustrates the generic commanded pitch profile for an orbital launcher (specifically, a *Soyuz 2.1a* injecting into a circular orbit of 330 km, with a kick angle of 2.49° measured from the local vertical). The appropriate selection of the kick angle results in a smooth transition from the gravity turn phase to PEG control at $t \approx 285$ s.

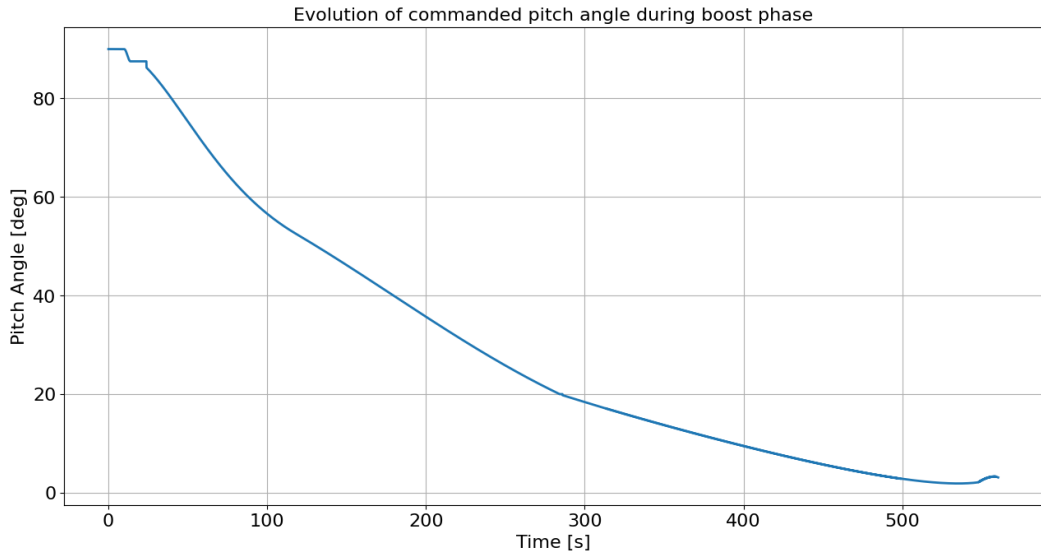


Figure 3.6: Generic commanded pitch profile for orbital launchers.

The information used to define the characteristics of several orbital launch vehicles and the parameters that determine their trajectories, as well as the different phases included in the guidance profile, the applied control law, and its implementation, have been derived and formulated based on the study presented in [26].

3.2. Software implementation and simulation framework

This section describes the software implementation developed to simulate the different trajectory types considered in this work. The entire framework has been implemented in Java.

The structure followed for the simulation of each trajectory family is presented, including the corresponding flowcharts, the parameters, and the methods employed for their numerical evaluation. In addition, the framework developed for data storage and post-processing is detailed.

3.2.1. Simulation inputs and initial conditions

Each trajectory class is defined by a set of input parameters that determine both the characteristics of the vehicle and the conditions of the mission. These inputs can be grouped into two main categories:

- Vehicle-dependent parameters, which specify the physical, aerodynamic and propulsive characteristics of the system (e.g., exhaust velocity or propellant mass).
- Mission-dependent parameters, which include the launch site coordinates, the initial date and time, the launch direction, and additional variables associated with the mission objective (e.g., target injection altitude, or final pitch angle).

The parameters common to ballistic missiles, ASAT systems, and orbital launch vehicles are presented in Table 3.1.

Table 3.1: General parameters common to all trajectory simulations.

Description	Symbol	Units
Date and time of launch [UTC]	-	-
Latitude of launch site	ϕ	rad
Longitude of launch site	λ	rad
Launch azimuth	Az	rad
Launch elevation	δ	rad

The parameters defining the trajectories of single-stage ballistic missiles, multi-stage ICBMs, ASATs, and orbital launch vehicles are presented in Tables 3.2, 3.3, 3.4, and 3.5, respectively.

Table 3.2: Parameters defining a single-stage ballistic missile trajectory.

Description	Symbol	Units
Exhaust velocity	U	m/s
Normalised burn rate	q	1/s
Ballistic coefficient	c	kg/m ²
Dry mass (structure + payload)	m_d	kg
Propellant mass	m_p	kg
Final pitch angle	θ_{end}	rad

Table 3.3: Parameters defining a multi-stage ballistic missile (ICBM) trajectory.

Description	Symbol	Units
Exhaust velocity of stage i	U_i	m/s
Normalised burn rate of stage i	q_i	1/s
Ballistic coefficient of stage i	c_i	kg/m ²
Structural mass of stage i	$m_{s,i}$	kg
Propellant mass of stage i	$m_{p,i}$	kg
Re-entry vehicle mass	m_{RV}	kg
Final pitch angle	θ_{end}	rad

Table 3.4: Parameters defining an ASAT trajectory.

Description	Symbol	Units
Exhaust velocity of stage i	U_i	m/s
Normalised burn rate of stage i	q_i	1/s
Ballistic coefficient of stage i	c_i	kg/m ²
Structural mass of stage i	$m_{s,i}$	kg
Propellant mass of stage i	$m_{p,i}$	kg
Final pitch angle	θ_{end}	rad

Remark: For ballistic missiles, the parameters include the mass of a payload or re-entry vehicle (RV), explicitly modelled as m_{RV} . In the case of ASAT vehicles, no separate payload is considered; the structural mass of the last stage is assumed to include the interceptor.

Table 3.5: Parameters defining an orbital launch vehicle trajectory.

Description	Symbol	Units
I_{sp} at sea level of stage i	$I_{sp,0,i}$	s
I_{sp} at sea level of boosters	$I_{sp,0,boosters}$	s
I_{sp} at vacuum of stage i	$I_{sp,vac,i}$	s
I_{sp} at vacuum of boosters	$I_{sp,vac,boosters}$	s
Thrust at sea level of stage i	$T_{0,i}$	N
Thrust at sea level of boosters	$T_{0,boosters}$	N
Thrust at vacuum of stage i	$T_{vac,i}$	N
Thrust at vacuum of boosters	$T_{vac,boosters}$	N
Structural mass of stage i	$m_{s,i}$	kg
Structural mass of boosters	$m_{s,boosters}$	kg
Wet mass of stage i	$m_{w,i}$	kg
Wet mass of boosters	$m_{w,boosters}$	kg
Number of engines of stage i	$n_{e,i}$	-
Number of boosters	$n_{boosters}$	-
Diameter of stage i	D_i	m
Diameter of boosters	$D_{boosters}$	m
Duration of stage i	t_i	s
Duration of boosters	$t_{boosters}$	s
Duration of vertical ascent	t_{ascent}	s
Duration of gravity turn phase	t_{gt}	s
Minimum allowable pitch angle	θ_{min}	rad
Kick angle	χ	rad
Injection altitude	h_{inj}	m
Injection velocity	v_{inj}	m/s
Payload mass	m_{PL}	kg

For the numerical integration of the full state, composed of the position and velocity of the vehicle in ECI coordinates, as well as its mass, the initial conditions must be correctly specified. The initial position corresponds to the launch site location, converted in the ECI frame, while the initial velocity is set to zero relative to the rotating Earth. This implies that the initial velocity of the vehicle in the ECI frame is not zero, but corresponds to the contribution of the Earth's rotational velocity at the launch latitude. Finally, the initial mass is specified according to the defined vehicle parameters.

3.2.2. Simulation methodology

The simulation of the trajectory is achieved by numerically integrating the equations of motion, propagating the state vector over time. For this purpose, the simulation employs advanced numerical propagators.

The trajectory during the boost phase (for all vehicle classes) and the re-entry phase (for ballistic missiles) is propagated using an adaptive step size 8(5,3) Dormand-Prince integrator. This choice ensures both high accuracy and computational efficiency when dealing with the non-linear dynamics of this model.

On the other hand, during the ballistic phase outside the atmosphere for ballistic missiles, the state is propagated using Orekit, an open-source space dynamics library [42].

Orekit is also used for several other complex computations necessary during the simulation, such as accurate conversions between different reference frames, precise time management, handling of celestial body properties, definition of atmosphere models (including space weather data), and the modelling of the different forces acting on the missile during its ballistic flight outside the atmosphere.

Therefore, a detailed explanation of the simulation workflow is provided for ballistic missiles, ASAT systems, and orbital launch vehicles, illustrating the sequence of computational steps, the guidance logic, and the phase transitions specific to each case.

Trajectory of ballistic missiles

The simulation of ballistic missile trajectories requires careful handling of transitions between the different phases or domains. The trajectory begins with the boost phase, propagating the missile's state by integrating the equations of motion (Eq. (3.3)). This integration stops when the first of two conditions is met: either the missile consumes all its propellant, or the missile impacts the ground (which is modelled as achieving an altitude below 5 m).

If the boost phase ends because no more fuel is available, propagation continues, and the next phase is determined by the altitude at burnout. If the burnout occurs below 100 km (i.e., within the atmosphere), the trajectory continues within an atmospheric coast phase. This is propagated by integrating the same equations of motion, excluding the thrust acceleration term.

On the other hand, if the burnout is above 100 km (outside the atmosphere), the trajectory enters the ballistic phase, which is propagated using Orekit. This transition to the Orekit propagator also occurs if the missile successfully exits the atmosphere after an initial atmospheric coast phase.

The ballistic phase may end in two possible ways. If the missile re-enters the atmosphere, the propagation continues again by integrating the equations of motion for the re-entry phase (Eq. (3.3), without thrust terms). If, on the other hand, the missile does not

re-enter the atmosphere during the simulation time, it implies the vehicle has sufficient velocity to escape from Earth gravity. This outcome is considered physically inconsistent with ballistic missile behaviour and is consequently discarded.

Figure 3.7 shows a conceptual scheme of the process followed during the simulation of ballistic missile trajectories.

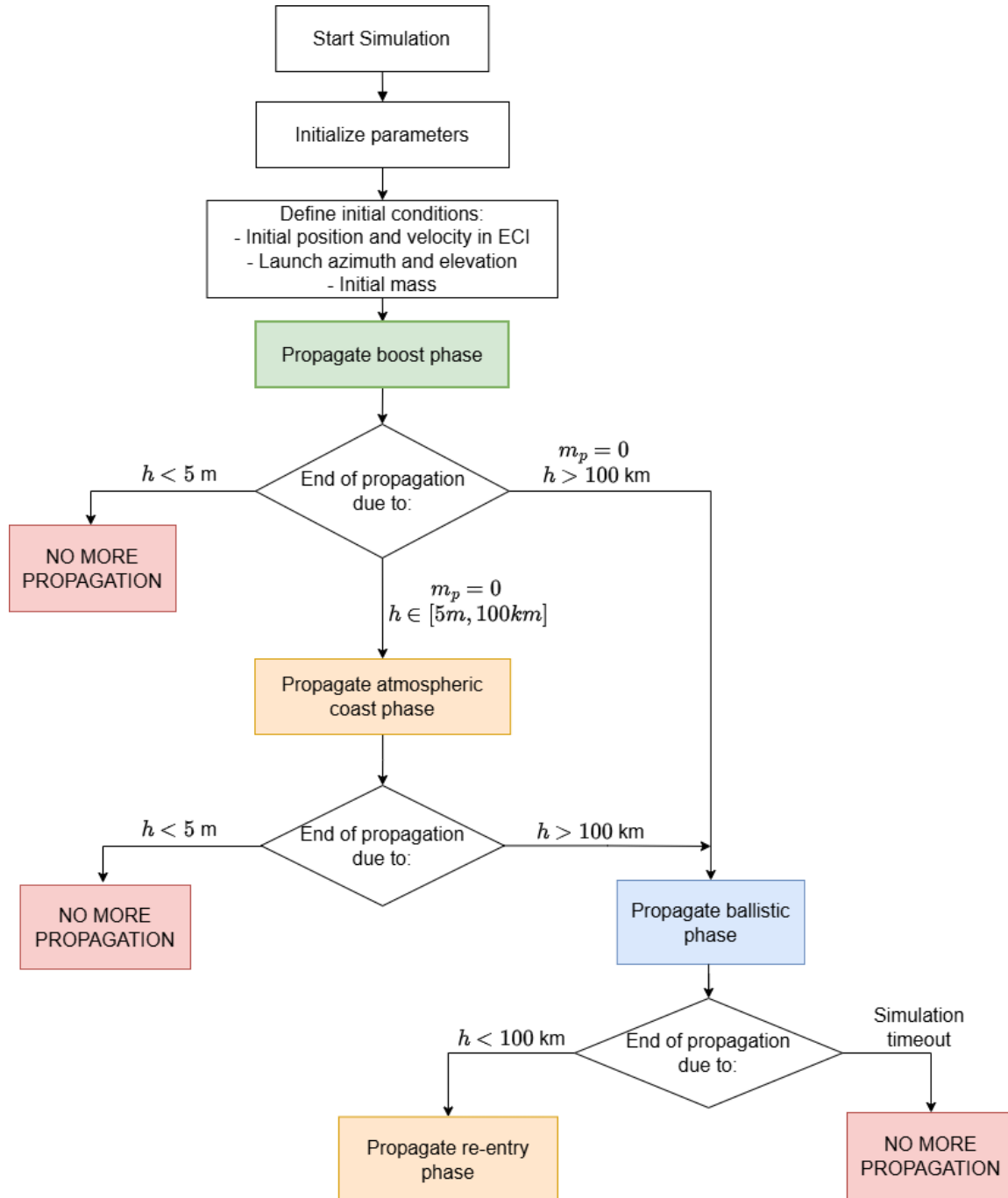


Figure 3.7: Conceptual scheme of the simulation workflow for ballistic missile trajectories.

During the propagation of the boost phase, if the missile is a multi-stage system, the parameters that define the equations of motion must be updated as the stages are depleted. For this reason, the simulation continuously refreshes these values following the logic illustrated in Figure 3.8. In this scheme, $m_{0,i}$ denotes the initial mass of stage i , computed as the sum of the structural and propellant masses of all remaining stages together with those of the current stage, $m_{d,i}$ denotes the dry mass of stage i , defined as the sum of the structural and propellant masses of the remaining stages plus the structural mass of the current one, and m_{RV} denotes the mass of the re-entry vehicle.

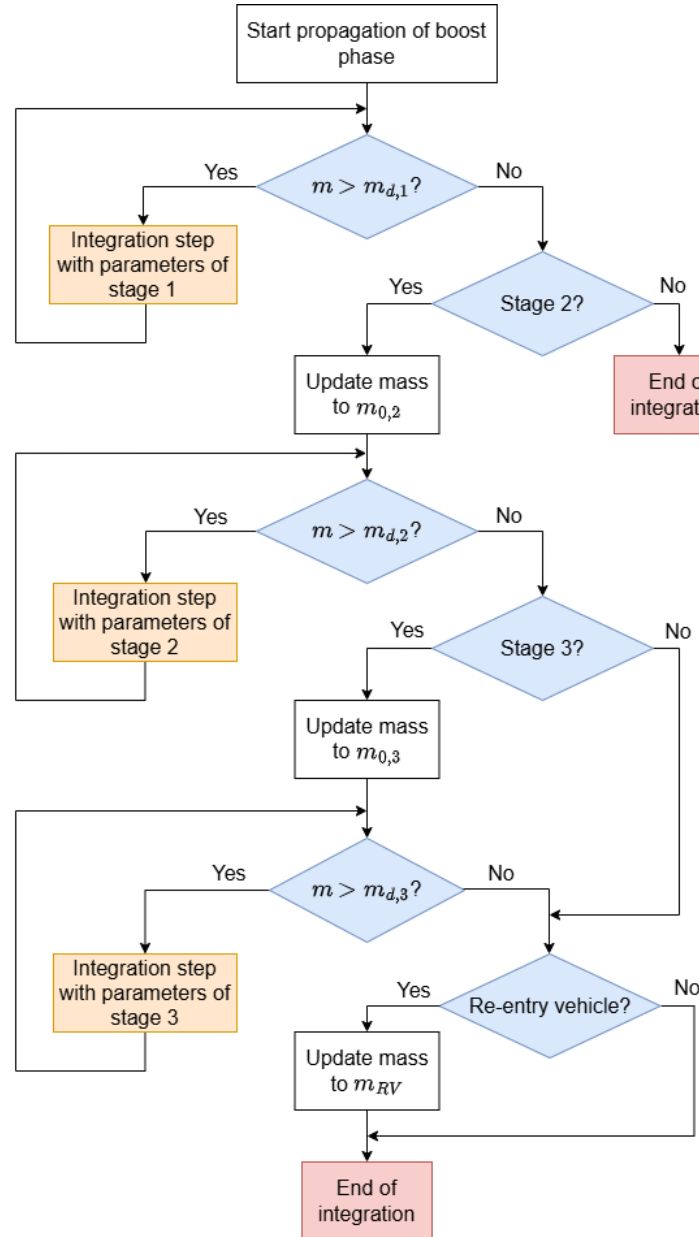


Figure 3.8: Conceptual scheme of the propagation during the boost phase of ballistic missile trajectories.

Furthermore, dedicated flow charts are provided to illustrate the implementation of the guidance profile during the boost phase for both single-stage ballistic missiles and multi-stage ICBMs in Figures 3.9 and 3.10, respectively.

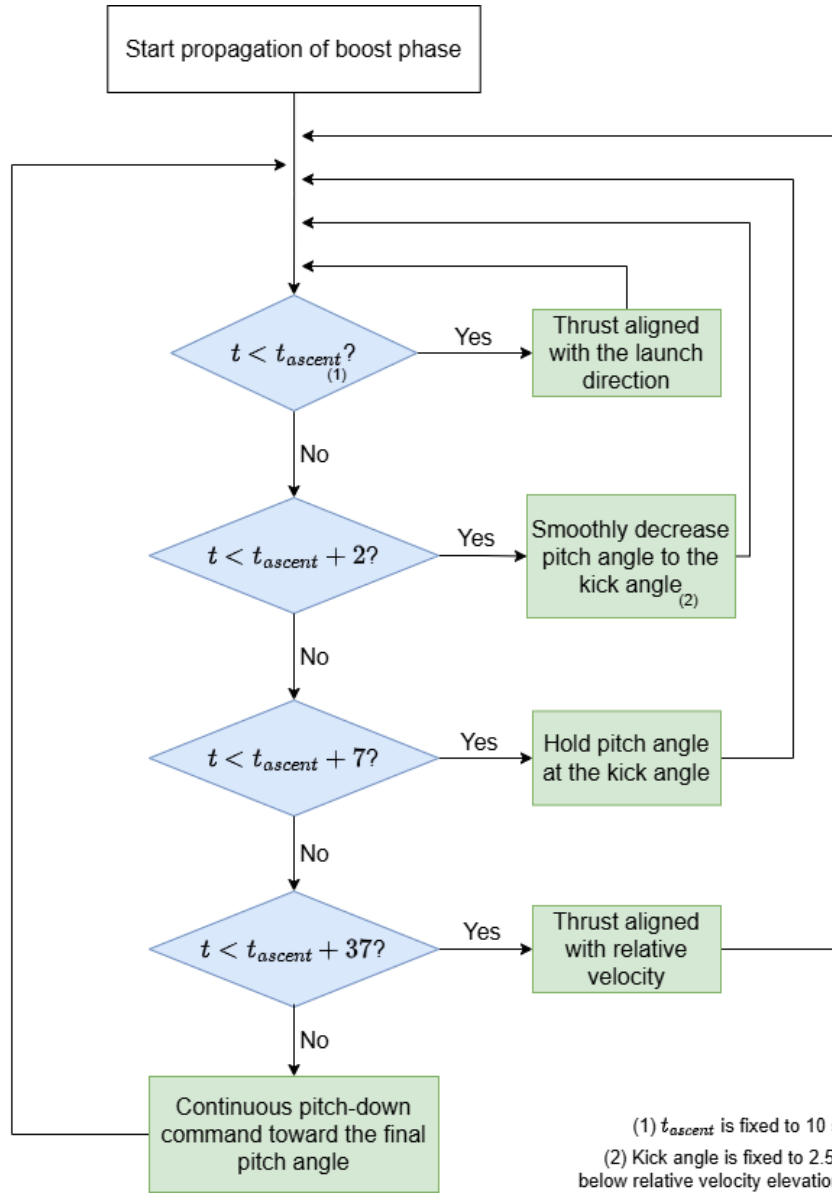


Figure 3.9: Conceptual scheme of the guidance logic implementation for single-stage ballistic missile trajectories.

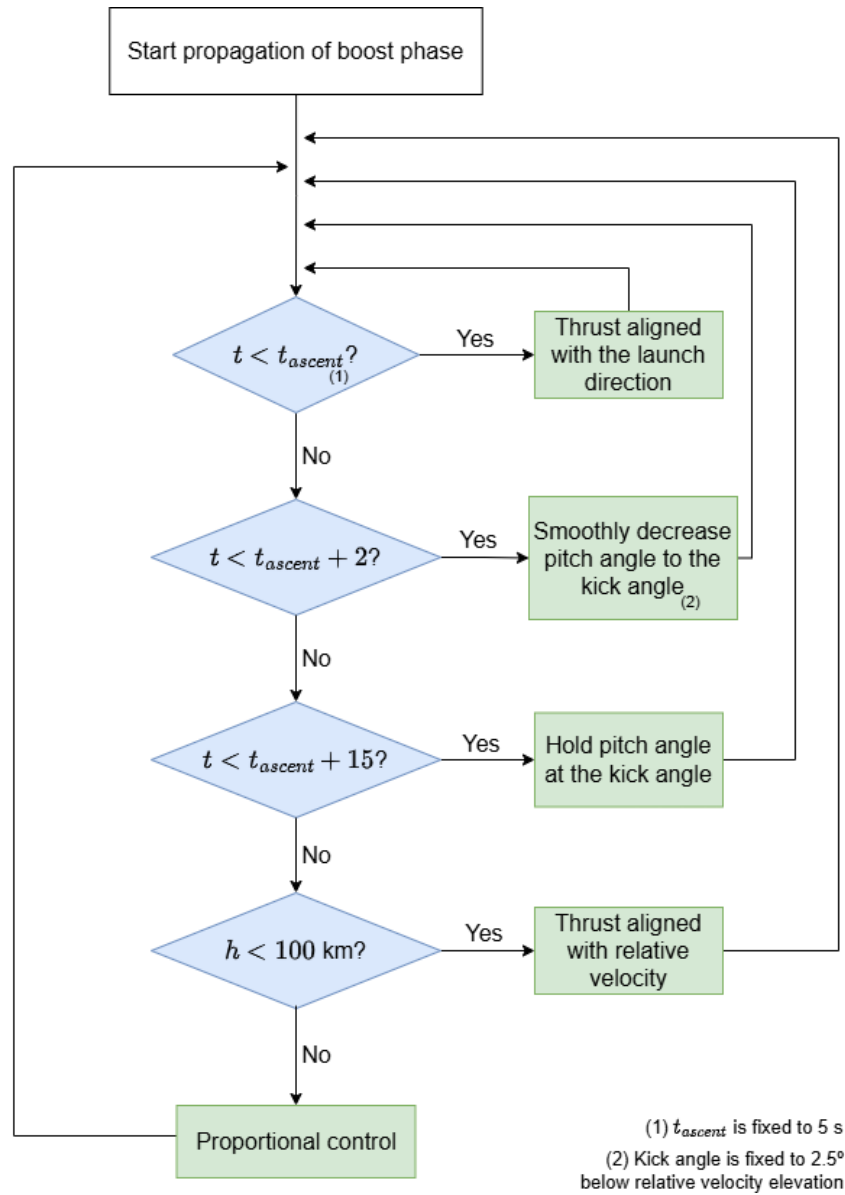


Figure 3.10: Conceptual scheme of the guidance logic implementation for multi-stage ICBM trajectories.

For the remaining phases, the simulation becomes simpler since no thrust is applied. Consequently, the parameters of the equations of motion remain constant, and no guidance law is required.

Trajectory of ASATs

In the case of ASAT systems, the situation is less complex, since only the boost phase is considered. Therefore, the overall scheme is shown in Figure 3.11.

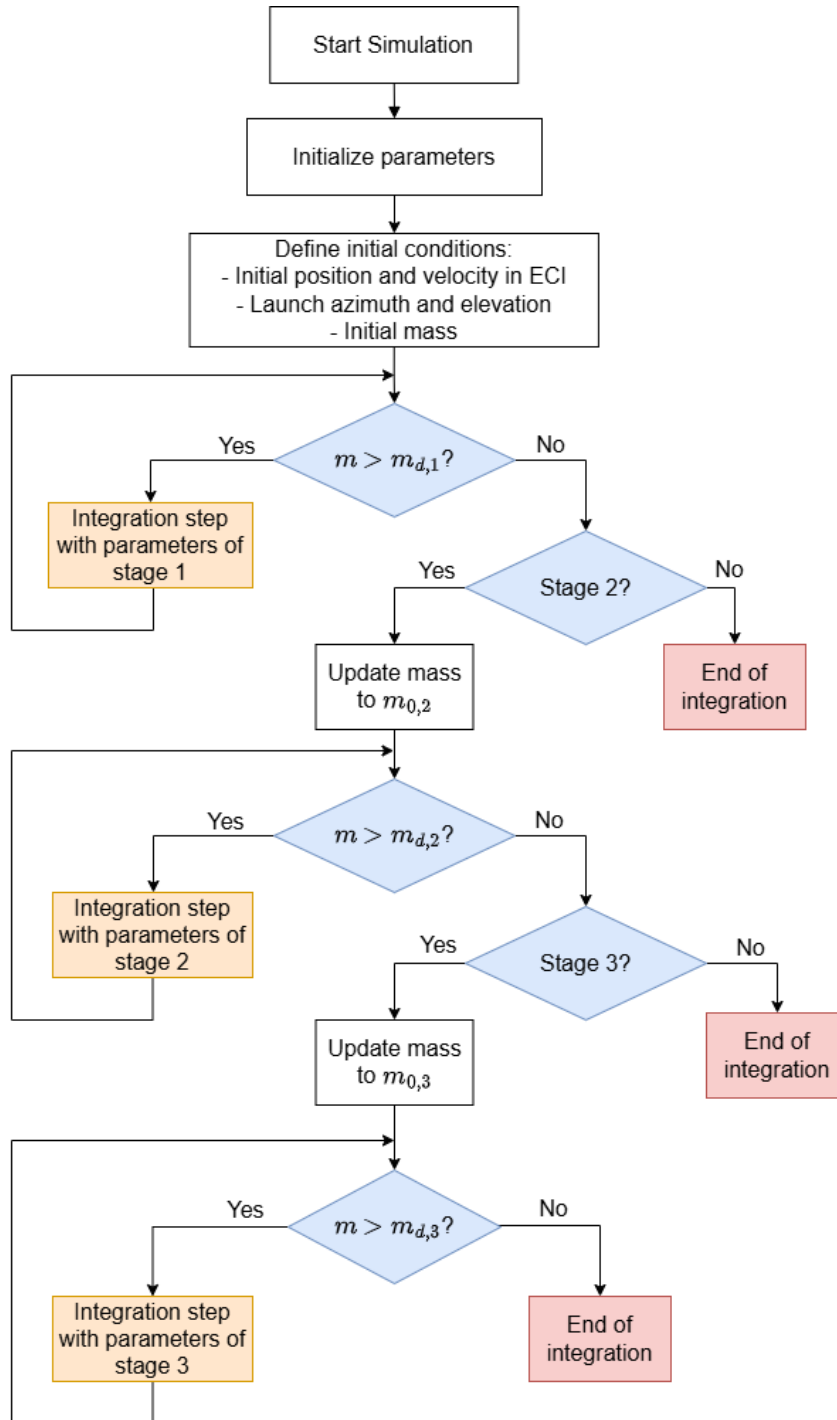


Figure 3.11: Conceptual scheme of the simulation workflow for ASAT trajectories.

Regarding the implementation of the guidance profile, this is illustrated by the flow chart presented in Figure 3.12.

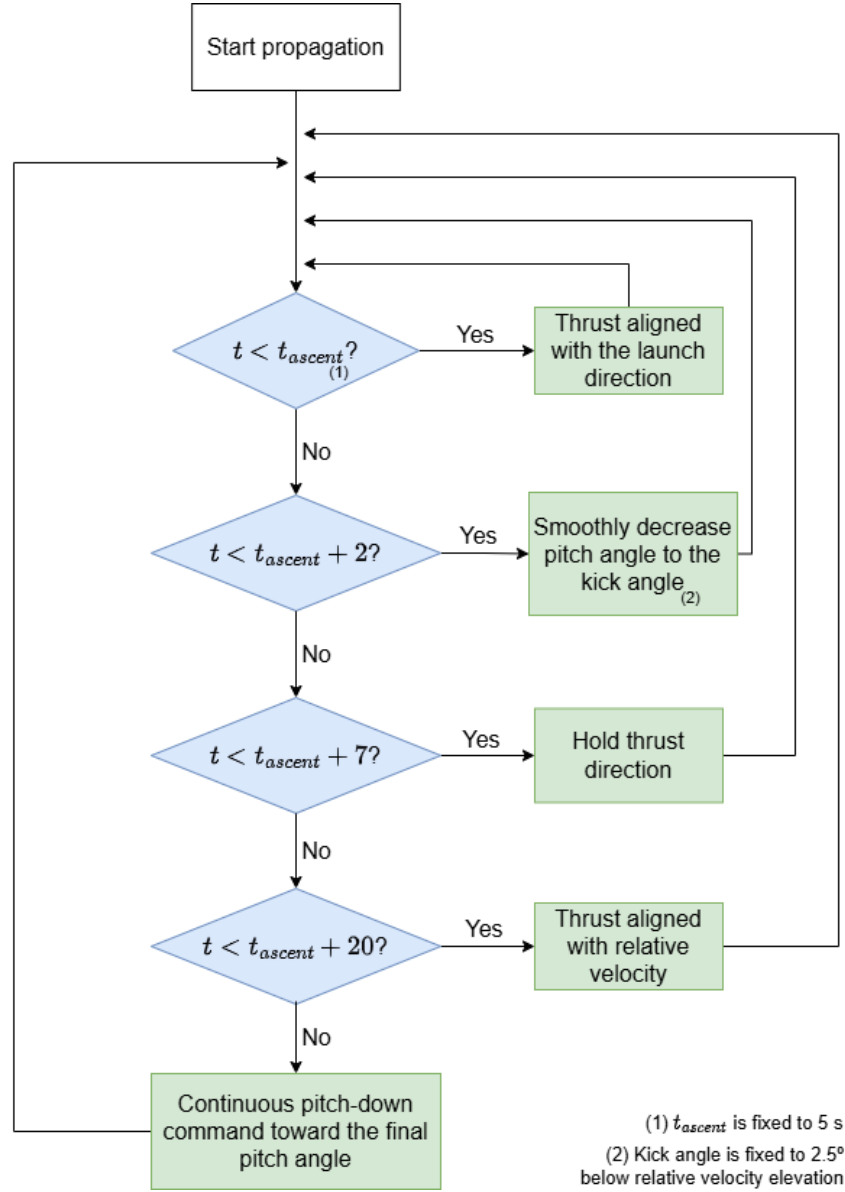


Figure 3.12: Conceptual scheme of the guidance logic implementation for ASAT trajectories.

Trajectory of orbital launchers

For orbital launchers, the workflow is analogous to that adopted for ASAT systems, including the possibility of incorporating boosters during the first stage. The overall simulation scheme is presented in Figure 3.13. In this case, for clarity, the update of the vehicle mass after the burnout of each stage or booster is not explicitly shown in the flow chart,

although this update must still be performed during the numerical propagation.

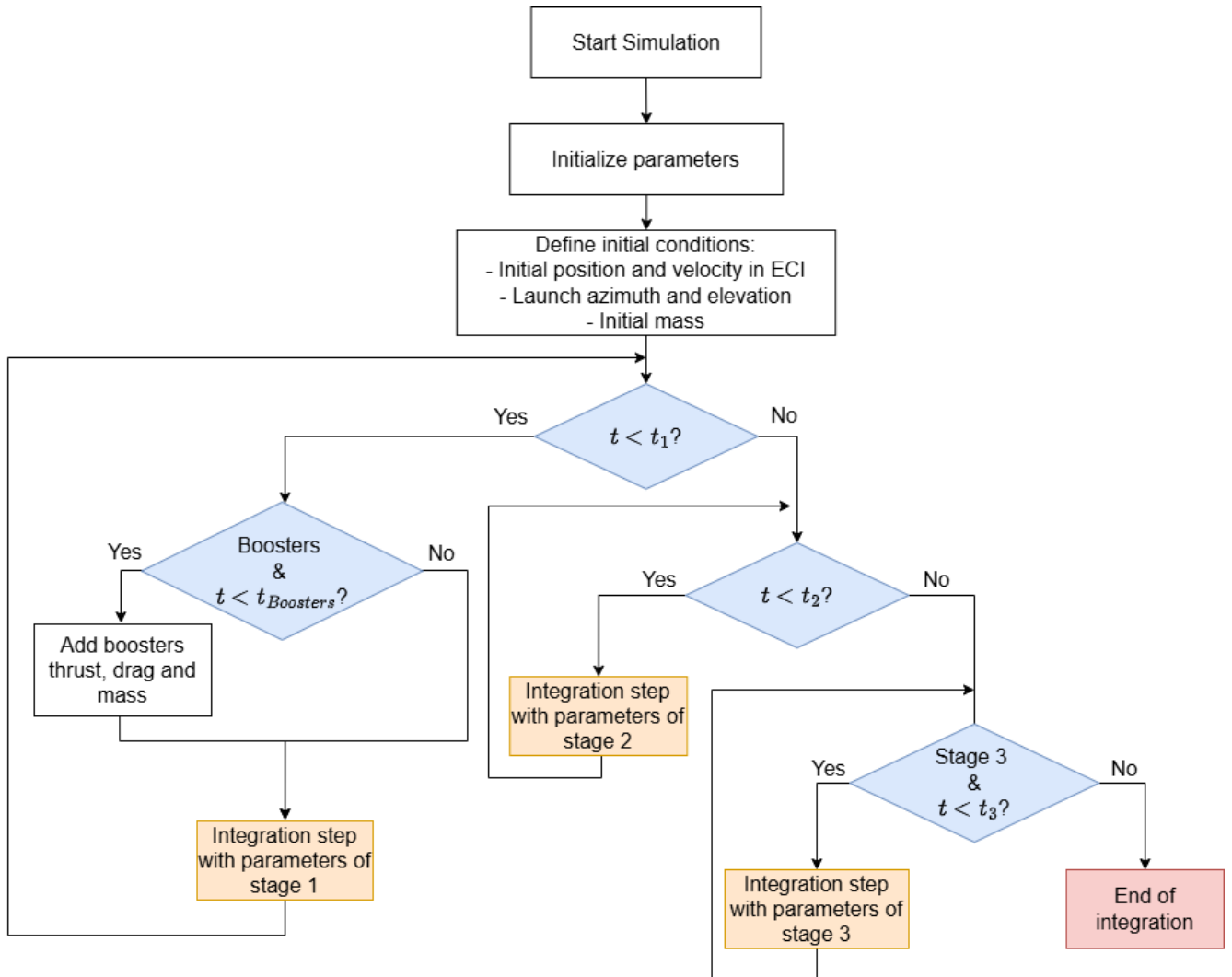


Figure 3.13: Conceptual scheme of the simulation workflow for orbital launcher trajectories.

Similarly, the implementation of the guidance profile for orbital launchers is illustrated in the flow chart shown in Figure 3.14.

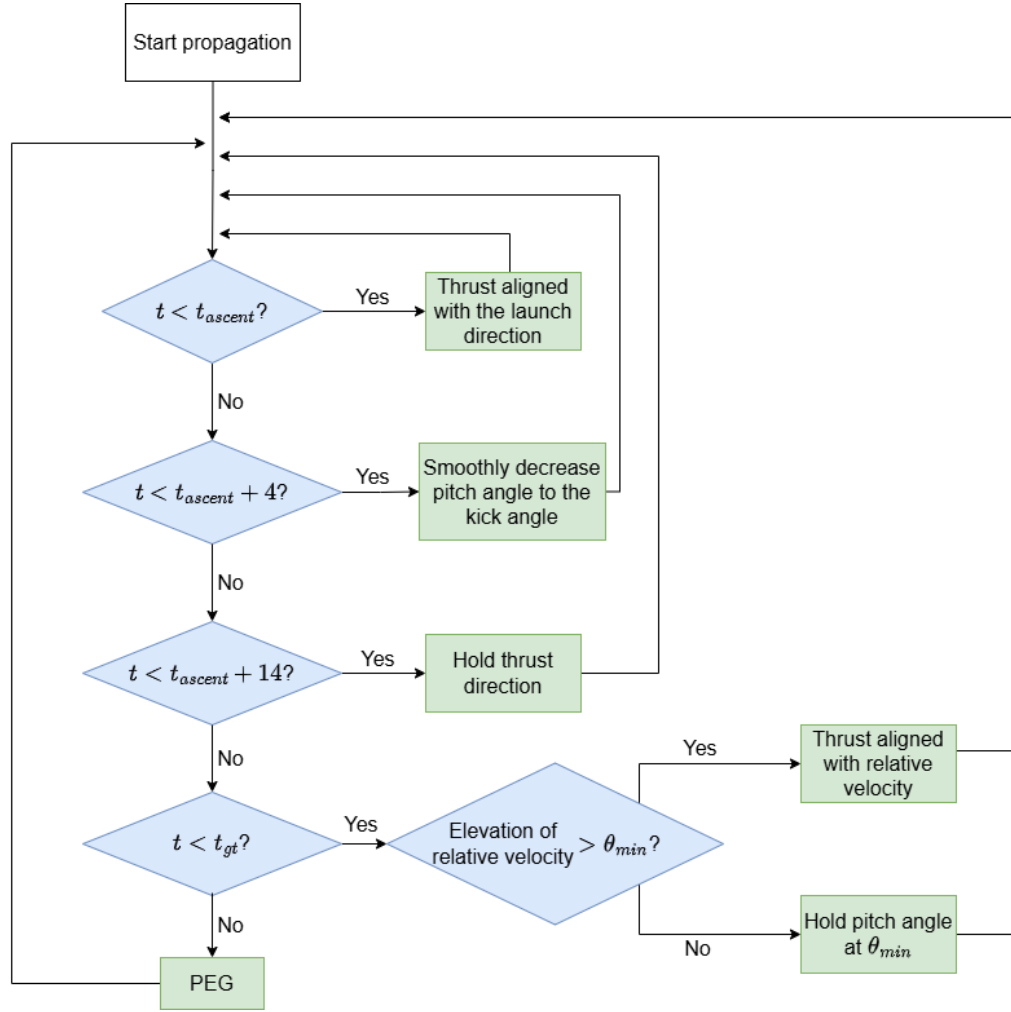


Figure 3.14: Conceptual scheme of the guidance logic implementation for orbital launcher trajectories.

3.2.3. Simulation outputs

For each simulated case, the software generates a dedicated *.csv* output file containing the complete time history of the trajectory. Each row of this file corresponds to a simulation time step and stores the following variables:

- Time step (s).
- The position (m) and velocity (m/s) vectors, expressed in the Earth-Centred Inertial (ECI), Earth-Centred Earth-Fixed (ECEF), and local East-North-Up (ENU) coordinate systems (a separate output file is produced for each reference frame).
- The mass of the vehicle (kg).
- The altitude above the Earth's surface (m).

- The range, measured as the horizontal ground distance from the launch site (m).

At the beginning of each file, a header is included specifying the type of vehicle, the trajectory parameters, and the launch conditions used in the simulation.

These outputs provide all the necessary information for post-processing and analysing the trajectory evolution, including the evolution of velocity, flight-path angle, or any other variable of interest.

Furthermore, several internal variables are recorded during the numerical integration and stored in a separate file. These include the direction of the relative velocity vector, the thrust direction, and the acceleration generated by thrust. These variables are used for assessing the feasibility of each trajectory, allowing the discarding of non-feasible or wrong trajectories, such as those failing to reach the required final flight-path angle, exceeding allowable angle-of-attack limits, or sustaining excessive accelerations for too long. Furthermore, the evaluation of these internal variables was used to correctly tune the general guidance profile for each vehicle class.

3.3. Validation of the model

To ensure the reliability of the developed simulation framework, each trajectory type is validated by comparing the obtained results with data available in the literature. This validation strategy guarantees that the implemented model is capable of accurately reproducing the physical behaviour expected from ballistic missiles, ASAT systems, and orbital launch vehicles.

In this section, specific trajectories for each vehicle class are simulated using dedicated input parameters. The results are then analysed and compared against references and expected results to formally validate the model.

3.3.1. Validation of ballistic missile trajectories

As discussed in the previous sections, simulations of ballistic missile trajectories are divided into two subcases: single-stage ballistic missiles and multi-stage ICBMs.

Single-stage trajectories

The validation of the single-stage ballistic missile model is performed by simulating a representative IRBM trajectory and comparing the results with the reference data presented in [43].

The simulated trajectory is defined by the parameters listed in Table 3.6. The exhaust velocity, normalised burn rate, and ballistic coefficient are directly extracted from [43], while the remaining parameters are selected to satisfy the required mission constraints, resulting in a trajectory that meets the expected burnout altitude and boost duration.

Table 3.6: Parameters of the validation trajectory for single-stage ballistic missiles.

Description	Symbol	Value	Units
Date and time of launch	-	2025-10-10, 23:30:00.000	UTC
Latitude of launch site	ϕ	36.418	deg
Longitude of launch site	λ	54.977	deg
Launch azimuth	Az	90	deg
Launch elevation	δ	80	deg
Exhaust velocity	U	2700	m/s
Normalised burn rate	q	0.01	1/s
Ballistic coefficient	c	4000	kg/m ²
Dry mass	m_d	300	kg
Propellant mass	m_p	3500	kg
Final pitch angle	θ_{end}	45	deg

The three-dimensional trajectory, propagated in the ECEF reference frame, is shown in Figure 3.15, while Figure 3.16 presents the evolution of altitude over ground range. The results indicate that the vehicle reaches a maximum altitude of approximately 1000 km and a total ground range of 3500 km.

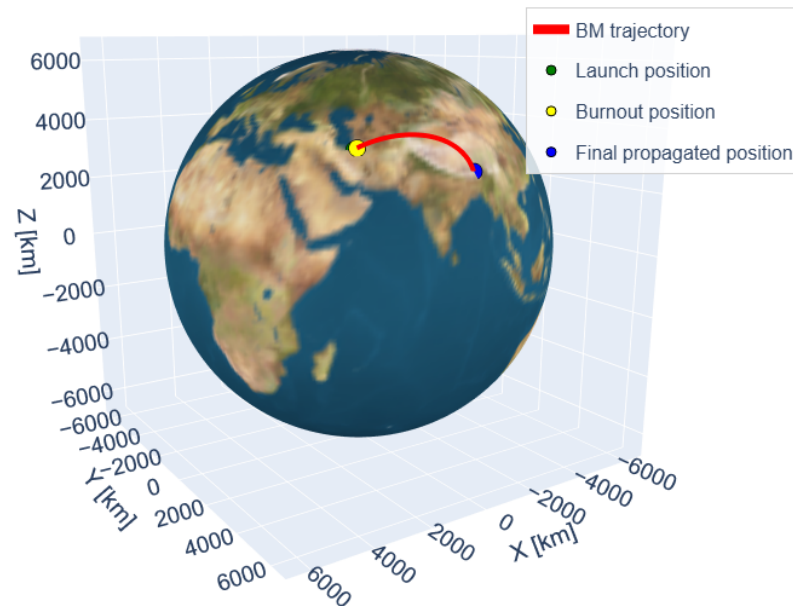


Figure 3.15: Simulated three-dimensional trajectory of a single-stage ballistic missile in ECEF coordinates.

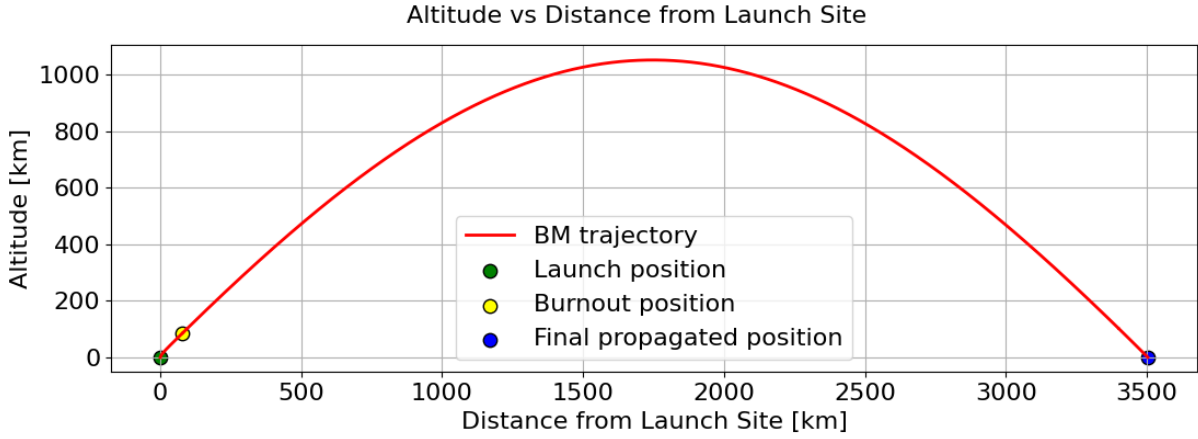


Figure 3.16: Altitude evolution with respect to ground range of a single-stage ballistic missile.

Focusing on the boost phase, Figure 3.17 illustrates the corresponding altitude profile, showing the vehicle transitioning from the initial vertical ascent to the selected final pitch angle.

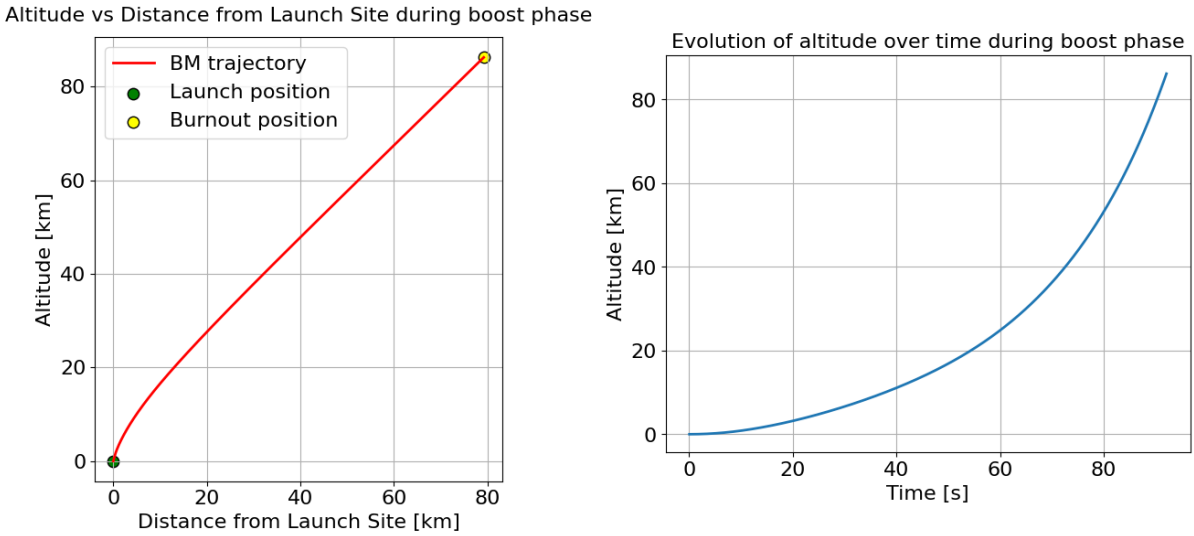


Figure 3.17: Altitude evolution during the boost phase of a single-stage ballistic missile.

The simulated trajectory results in a burnout altitude of approximately 88 km and a boost phase duration of 90 s. These values fall within the operational ranges reported in the literature (70 – 110 s boost phase flight time and 70 – 90 km burnout altitude [43]), confirming the validity of the model. The specific launch conditions and the selected mass parameters (m_d and m_p), together with the final pitch angle (θ_{end}), can be varied to generate a family of trajectories that also satisfy these typical mission constraints.

Further analysis of the dynamics validates the physical consistency of the model. The evolution of the velocity components during the boost phase (see Figure 3.18) shows a rapid, non-linear increase in the velocity magnitude, reaching approximately 4.3 km/s at burnout, which confirms the expected suborbital regime. Since the selected final pitch angle is 45° , the radial and tangential velocity components reach the same value at the end of boost phase, remaining close throughout most of the ascent.

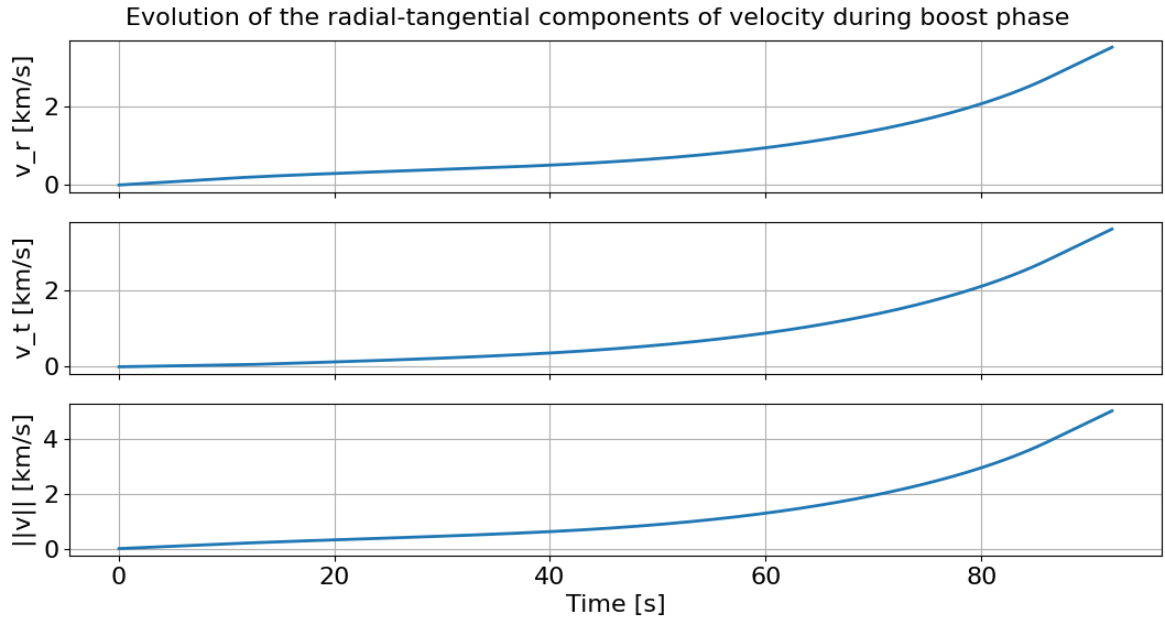


Figure 3.18: Evolution of velocity components during the boost phase of a single-stage ballistic missile.

Regarding the implementation of the guidance profile, Figure 3.19 compares the instantaneous flight-path angle with the commanded pitch angle. The difference between them confirms that the initial gravity turn successfully aligns the thrust vector with the velocity vector shortly after the vertical ascent. The final flight-path angle closely matches the commanded θ_{end} , validating the implementation of the quadratic guidance profile (Section 3.1.3).

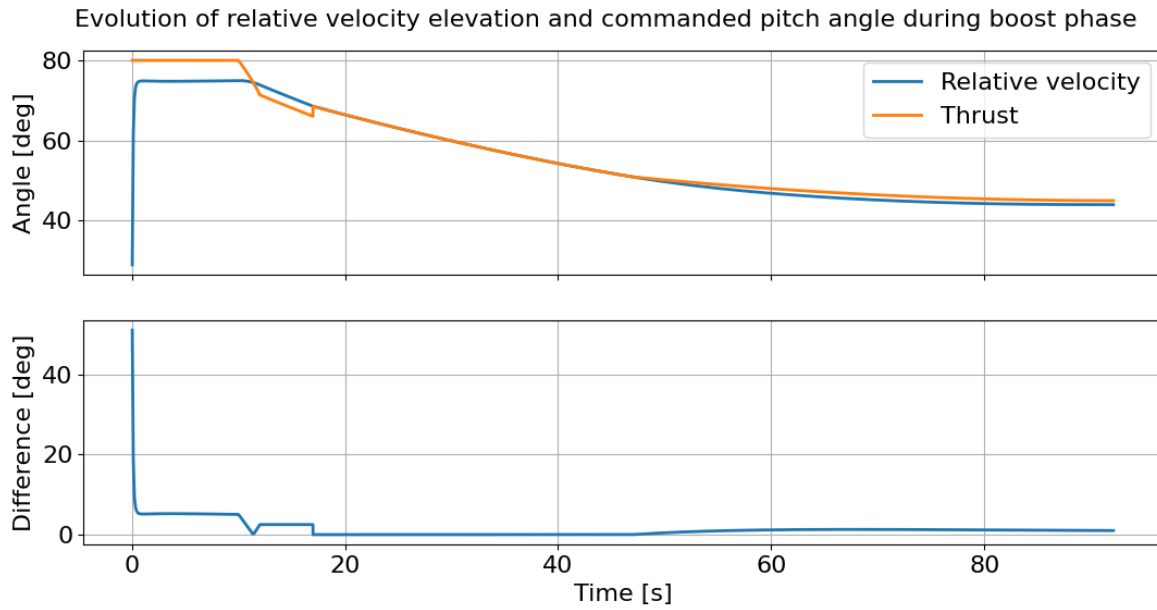


Figure 3.19: Comparison between commanded pitch angle and flight-path angle during the boost phase of a single-stage ballistic missile.

Finally, Figure 3.20 illustrates the evolution of the vehicle's thrust acceleration. The acceleration remains relatively low initially but increases exponentially as the mass decreases, reaching the maximum of 20 g shortly before burnout. Although large, this value is consistent with the performance expected from these systems [38].

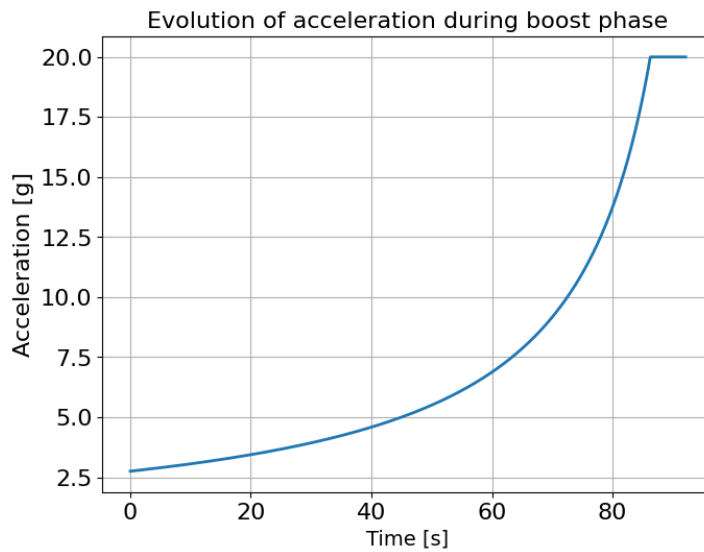


Figure 3.20: Thrust acceleration profile during the boost phase of a single-stage ballistic missile.

Multi-stage trajectories

The validation of the multi-stage ICBM model is carried out by simulating the trajectory of the LGM-30G Minuteman III and comparing the obtained results with the available reference data.

The parameters that define the performance of the missile are extracted from [3], which provides detailed information on stage masses, exhaust velocities, burn rates and burning times, as well as stage dimensions. The latter enables the computation of conservative ballistic coefficients (defined as $c = \frac{m}{C_d A}$). The remaining mission and guidance parameters are selected to obtain a representative nominal trajectory that achieves the expected ceiling and ground range. All parameters used for the validation case are listed in Table 3.7.

Table 3.7: Parameters of the validation trajectory for multi-stage ballistic missiles.

Description	Symbol	Value	Units
Date and time of launch	-	2025-10-10, 23:30:00.000	UTC
Latitude of launch site	ϕ	36.418	deg
Longitude of launch site	λ	54.977	deg
Launch azimuth	Az	90	deg
Launch elevation	δ	84	deg
Exhaust velocity of stage 1	U_1	2435	m/s
Exhaust velocity of stage 2	U_2	2435	m/s
Exhaust velocity of stage 3	U_3	2435	m/s
Normalised burn rate of stage 1	q_1	0.01007	1/s
Normalised burn rate of stage 2	q_2	0.00845	1/s
Normalised burn rate of stage 3	q_3	0.01383	1/s
Ballistic coefficient of stage 1	c_1	20000	kg/m ²
Ballistic coefficient of stage 2	c_2	10000	kg/m ²
Ballistic coefficient of stage 3	c_3	8000	kg/m ²
Structural mass of stage 1	$m_{s,1}$	2431	kg
Structural mass of stage 2	$m_{s,2}$	1000	kg
Structural mass of stage 3	$m_{s,3}$	292	kg
Propellant mass of stage 1	$m_{p,1}$	20716	kg
Propellant mass of stage 1	$m_{p,2}$	6205	kg
Propellant mass of stage 1	$m_{p,3}$	3308	kg
Re-entry vehicle mass	m_{RV}	463	kg
Final pitch angle	θ_{end}	15	deg

The resulting three-dimensional trajectory, propagated in the ECEF reference frame, is shown in Figure 3.21, while Figure 3.22 represents the altitude evolution with respect to ground range. As expected, the shape of the trajectory is similar to the single-stage case, since the selected launch site and launch azimuth are the same, but it covers larger distances. The simulation results indicate a maximum apogee of approximately 1100 km, and a total ground range of around 11000 km, which are consistent with the operational range and flight ceiling [44].

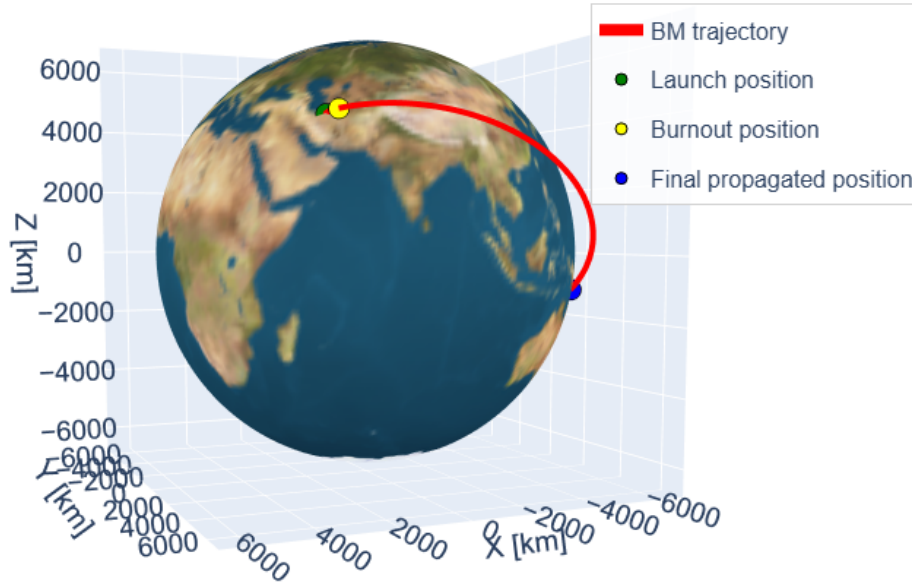


Figure 3.21: Simulated three-dimensional trajectory of a multi-stage ICBM in ECEF coordinates.

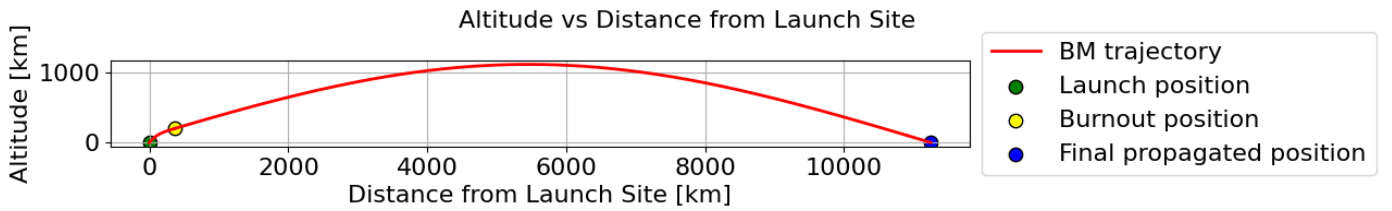


Figure 3.22: Altitude evolution with respect to ground range of a multi-stage ICBM.

Focusing on the boost phase, Figure 3.23 shows the corresponding altitude profile, where the vehicle transitions from the initial vertical ascent to the selected final pitch angle.

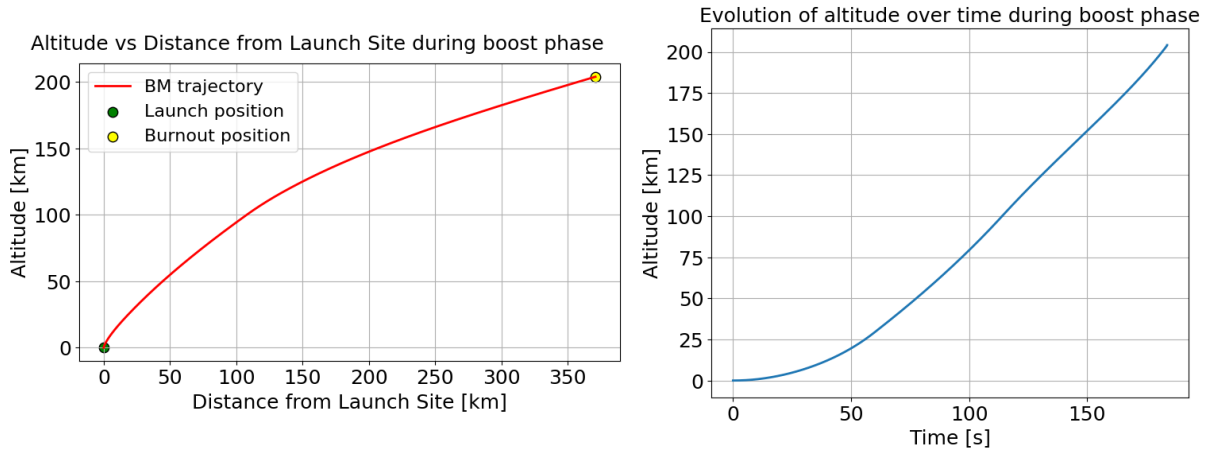


Figure 3.23: Altitude evolution during the boost phase of a multi-stage ICBM.

The simulated trajectory results in a burnout altitude of approximately 200 km and a boost phase duration of 184 s, both within the operational ranges reported in the literature. As in the previous case, the specific launch conditions, together with the final pitch angle (θ_{end}), can be varied to generate a family of trajectories with different ground ranges.

Further examination of the dynamic variables supports the physical consistency of the model. The evolution of the velocity components during the boost phase (see Figure 3.24) exhibits the expected trends, reaching approximately 6.8 km/s at burnout (again, this velocity is below the orbital velocity at that altitude). Since the selected final pitch angle is 15° , the tangential component dominates over the radial at the end of the boost phase.

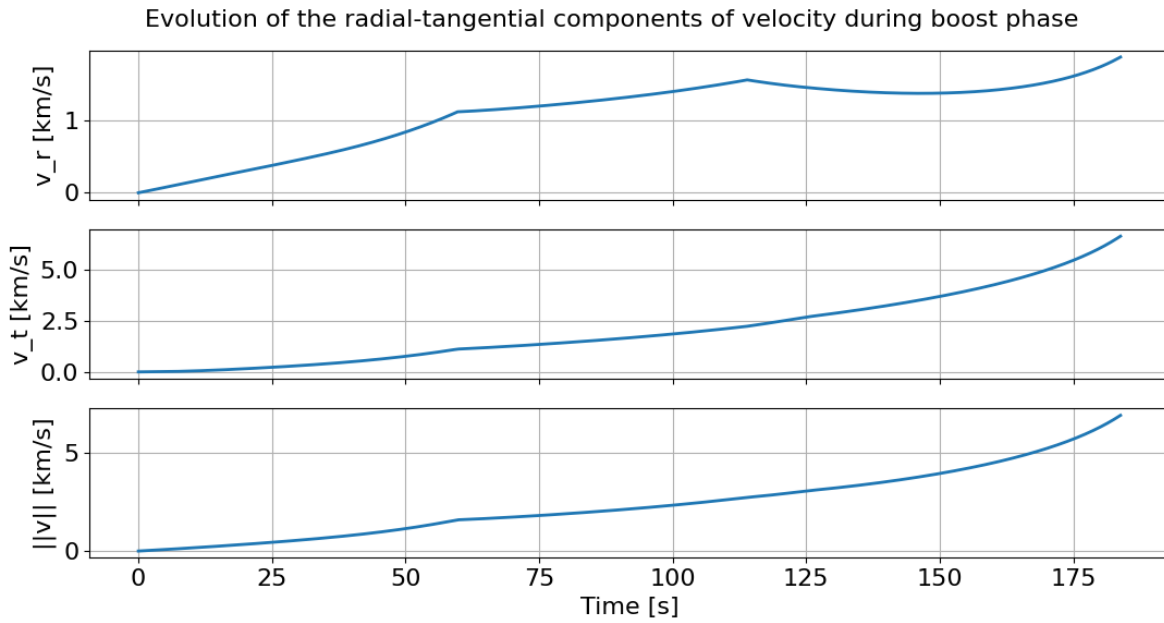


Figure 3.24: Evolution of velocity components during the boost phase of a multi-stage ICBM.

Regarding the implementation of the guidance profile, Figure 3.25 compares the instantaneous flight-path angle with the commanded pitch angle.

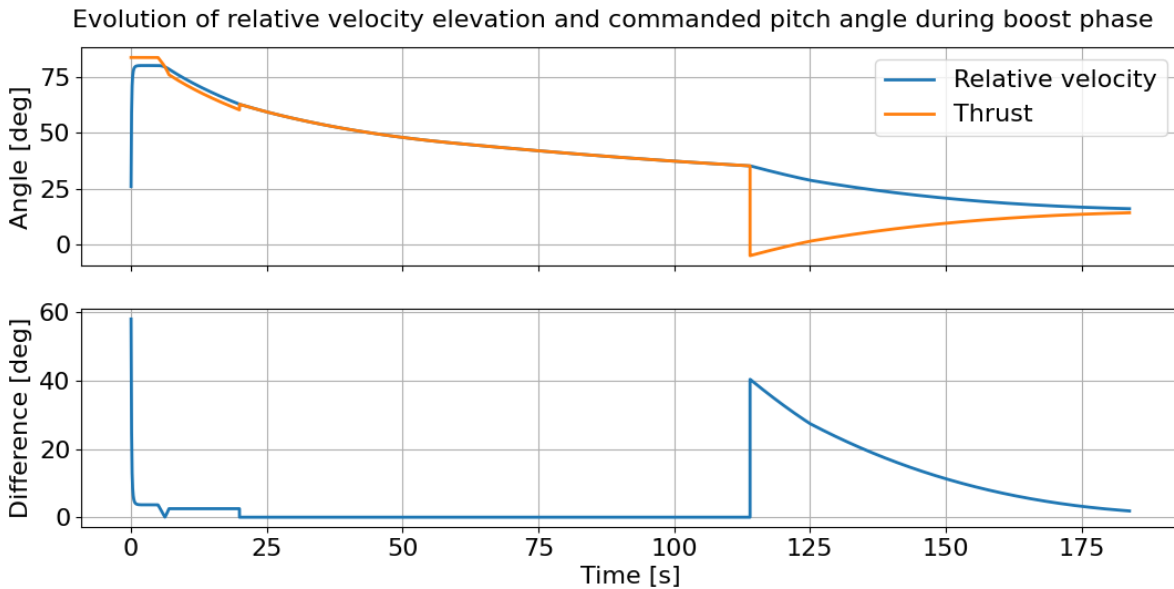


Figure 3.25: Comparison between commanded pitch angle and flight-path angle during the boost phase of a multi-stage ICBM.

The difference between them confirms that the initial gravity turn successfully aligns the thrust vector with the velocity vector. Then, the proportional control is applied once the missile exits the atmosphere, allowing the system to tolerate large angles of attack, since the atmospheric drag is negligible. The final flight-path angle closely matches the commanded θ_{end} , validating the implementation of the control law.

Finally, Figure 3.26 illustrates the evolution of the thrust acceleration. The different staging events and burn times match those in [3]. The acceleration remains relatively low initially but increases exponentially as the mass decreases, reaching the maximum of 20 g shortly before burnout, in agreement with these systems.

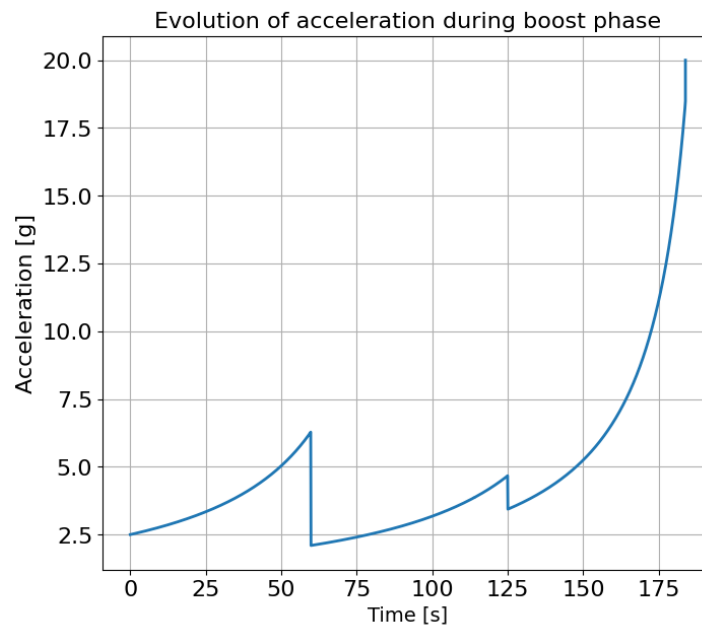


Figure 3.26: Thrust acceleration profile during the boost phase of a multi-stage ICBM.

3.3.2. Validation of ASAT trajectories

The validation of the ASAT trajectory model is based on the highly relevant study of the Indian Mission Shakti [21], from which significant results can be extracted for comparison:

- The total mission time, from launch to target interception, was 168 seconds.
- Interception occurred at an altitude of 284 km with a relative velocity of nearly 10 km/s.
- The missile's final velocity vector formed an angle of 135° with respect to the satellite's motion and 48° with the local horizontal.

From this information, combined with the known orbital velocity of the target satellite, it is possible to calculate the missile's absolute velocity at impact, which is approximately 3.5 km/s.

The staging sequence is also known. The initial stage was separated at 45 km after 75 seconds of flight, while the second stage burned for about 30 seconds until the missile reached an altitude of roughly 110 km. The third stage then continued until the impact [45]. This sequence defines the duration and altitude transitions for each of the three boost phases.

Furthermore, the mass properties are documented. The first two stages weigh around 17.2 tonnes, carrying 16.7 t of fuel, while the third stage, which includes the Kinetic Kill Vehicle (KKV), weighs approximately 1.8 t [21]. Based on this data, it is possible to make a robust estimation of the normalised burn rate for each stage. The ballistic coefficients are estimated using a conservative approach based on the dimensions and masses of each stage.

The exhaust velocities are then adjusted to satisfy the constraints on altitude, time, and final velocity. Additionally, by analysing the missile's trajectory profile (Figure 2.7) and the coordinates of the impact point [21], the total ground distance covered can be estimated at approximately 220 km, which provides an additional validation metric for the simulation.

Therefore, the simulated trajectory is defined by the parameters listed in Table 3.8.

Table 3.8: Parameters of the validation trajectory for ASATs.

Description	Symbol	Value	Units
Date and time of launch	-	2025-10-10, 23:30:00.000	UTC
Latitude of launch site	ϕ	20.7578	deg
Longitude of launch site	λ	87.0839	deg
Launch azimuth	Az	170	deg
Launch elevation	δ	86	deg
Exhaust velocity of stage 1	U_1	2100	m/s
Exhaust velocity of stage 2	U_2	2000	m/s
Exhaust velocity of stage 3	U_3	1500	m/s
Normalised burn rate of stage 1	q_1	0.00975	1/s
Normalised burn rate of stage 2	q_2	0.01945	1/s
Normalised burn rate of stage 3	q_3	0.00529	1/s
Ballistic coefficient of stage 1	c_1	41000	kg/m ²
Ballistic coefficient of stage 2	c_2	10000	kg/m ²
Ballistic coefficient of stage 3	c_3	6000	kg/m ²
Structural mass of stage 1	$m_{s,1}$	300	kg
Structural mass of stage 2	$m_{s,2}$	200	kg
Structural mass of stage 3	$m_{s,3}$	1200	kg
Propellant mass of stage 1	$m_{p,1}$	13900	kg
Propellant mass of stage 1	$m_{p,2}$	2800	kg
Propellant mass of stage 1	$m_{p,3}$	600	kg
Final pitch angle	θ_{end}	48	deg

The resulting three-dimensional trajectory, propagated in the ECEF reference frame, is shown in Figure 3.27

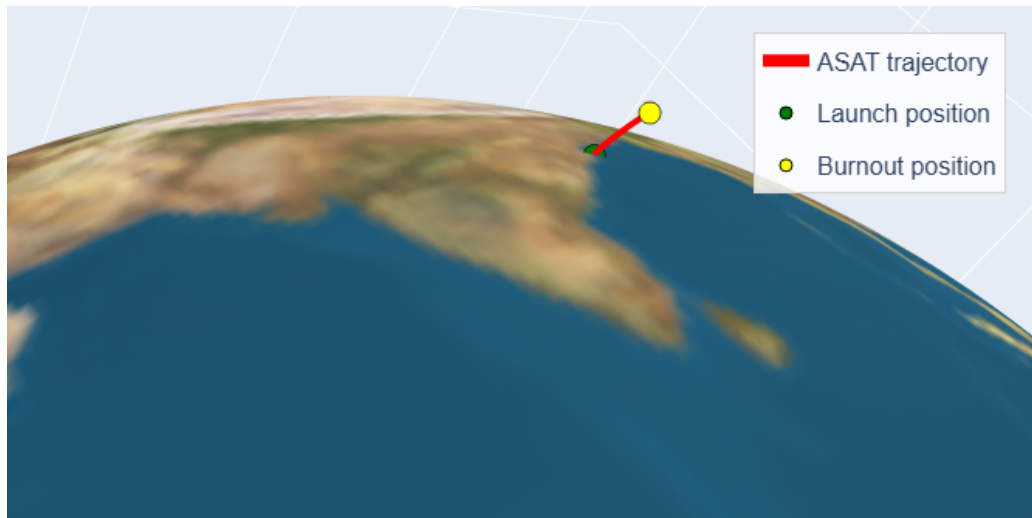


Figure 3.27: Simulated three-dimensional trajectory of an ASAT in ECEF coordinates.

Figure 3.28 presents the altitude evolution with respect to ground range. As expected, the trajectory is steeper than in the ballistic missile case, since the system is designed to reach higher altitudes by the end of the boost phase. The simulation results in a burnout altitude of 283 km, and a ground range of about 215 km, values that are consistent with those reported for the real system test.

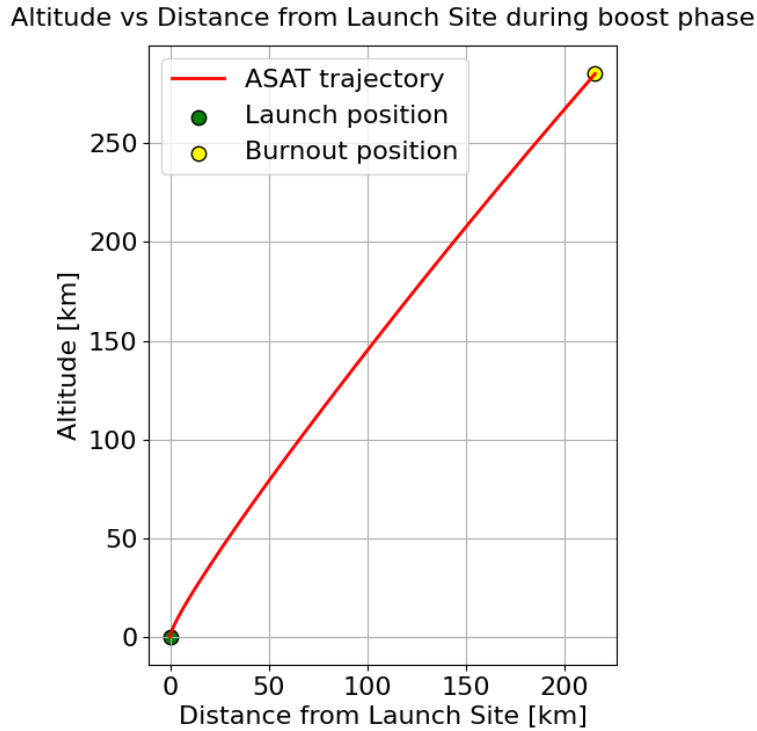


Figure 3.28: Altitude evolution with respect to ground range of an ASAT.

The temporal evolution of altitude and thrust acceleration is depicted in Figure 3.29. The simulated boost phase duration matches the real system, that is, 168 s, and the staging sequence also reproduces the altitude and timing transitions of the Mission Shakti, reaching 45 km and 110 km at the end of the first two stages. Initially, the acceleration remains relatively low, but it increases rapidly as the vehicle mass decreases, reaching peaks of approximately 8 g.

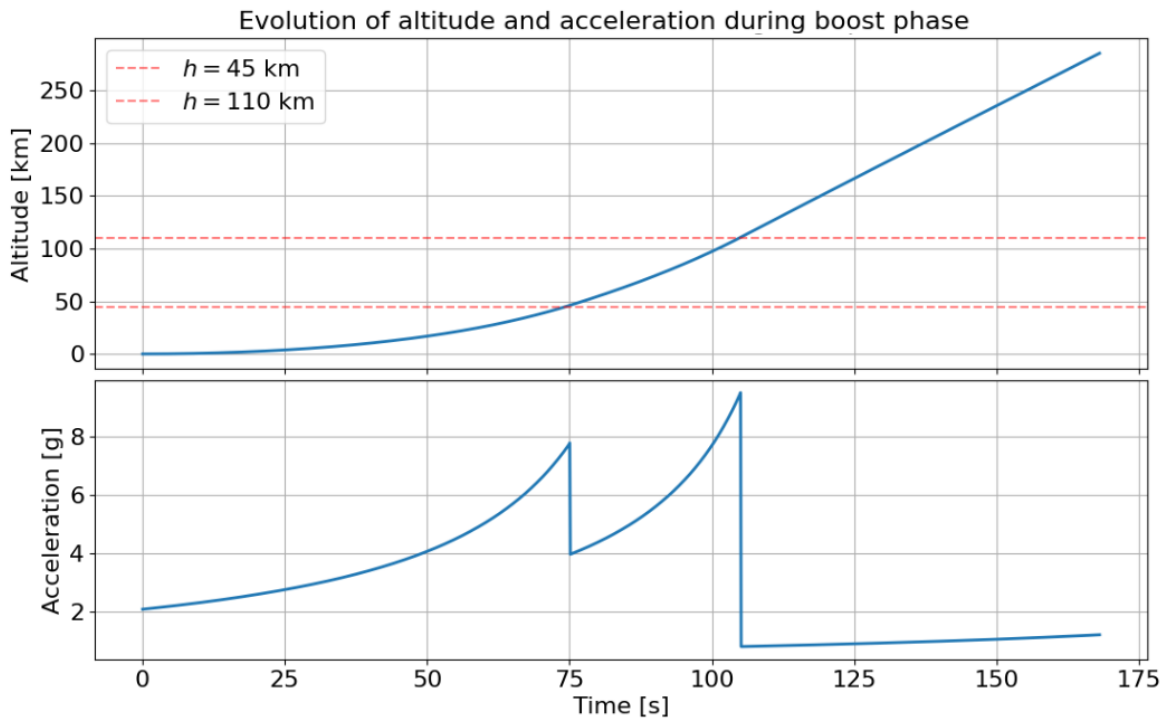


Figure 3.29: Evolution of altitude and thrust acceleration over time for an ASAT.

Further analysis of the dynamic variables confirms the physical consistency and validity of the model. Figure 3.30 shows the evolution of the velocity components during the boost phase, reaching approximately 3.7 km/s at burnout, which is close to what is expected from the real test case. Furthermore, the suborbital nature of these systems is also confirmed.

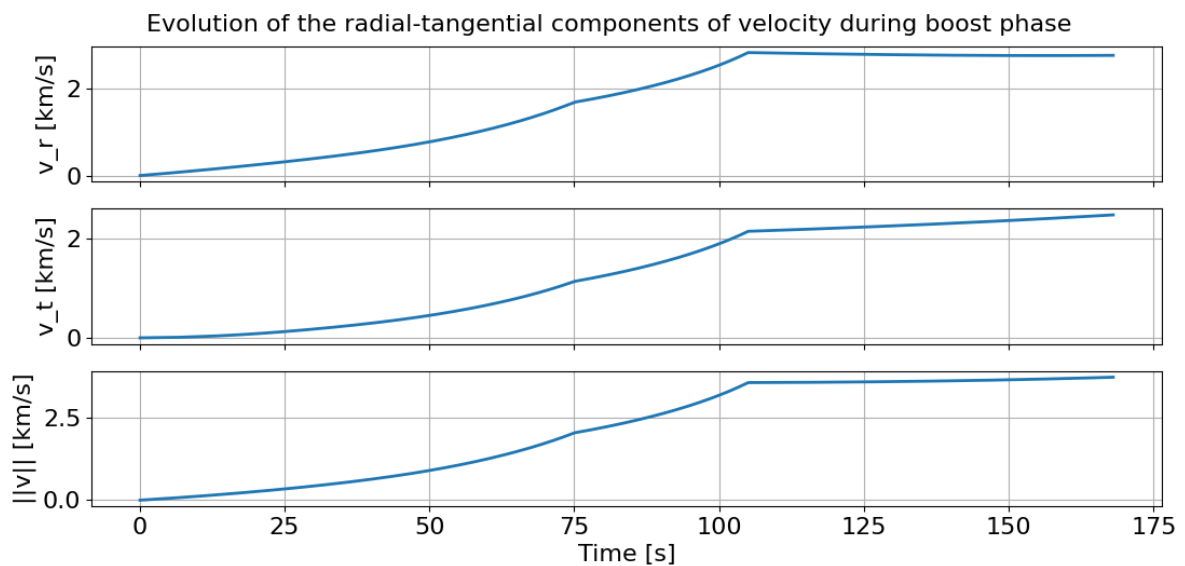


Figure 3.30: Evolution of velocity components during the boost phase of an ASAT.

Finally, Figure 3.31 compares the instantaneous flight-path angle with the commanded pitch angle.

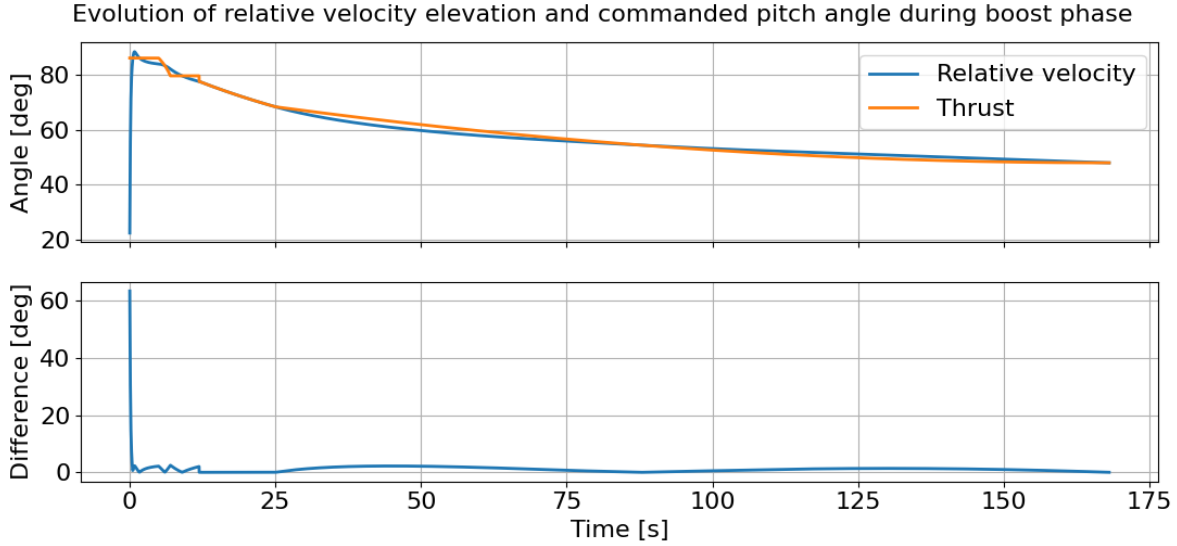


Figure 3.31: Comparison between commanded pitch angle and flight-path angle during the boost phase of an ASAT.

The difference between them confirms that the initial gravity turn successfully aligns the thrust vector with the velocity vector. Throughout the trajectory, the flight-path angle remains close to the thrust direction, maintaining an angle of attack close to zero. Therefore, the final flight-path angle converges to the commanded θ_{end} , resulting in an impact angle consistent with the real test and validating the implementation of the control law.

3.3.3. Validation of orbital launch vehicle trajectories

The validation of the orbital launcher model is critical, as it confirms the correct implementation of multi-stage dynamics and the PEG algorithm. The model is qualitatively validated by comparing the simulated injection against a reference case from the literature [26], together with different ascent profiles extracted from “*flightclub.io*”. This approach ensures that the developed simulator produces consistent and feasible results, with orbital injections aligned with the behaviour reported in the reference study.

The selected case for the simulation is an approximation of the COSMOS 2573 mission, launched by a Soyuz 2.1a from the Plesetsk Cosmodrome (PKMTR). The target orbit is defined by an injection altitude of 332 km and an injection velocity of 7.76037 km/s.

Once the launcher, launch site, payload and orbital target are defined, and the appropriate kick angle is selected to ensure a smooth transition from the gravity turn phase to the PEG control, the parameters defining the simulated trajectory are listed in Table 3.9.

Table 3.9: Parameters of the validation trajectory for orbital launchers.

Description	Symbol	Value	Units
Date and time of launch	-	2025-10-10, 23:30:00.000	UTC
Launch site	-	PKMTR	-
Launch azimuth	Az	90	deg
Launch elevation	δ	90	deg
I_{sp} at sea level of stage 1	$I_{sp,0,1}$	257.7	s
I_{sp} at sea level of boosters	$I_{sp,0,boosters}$	263.3	s
I_{sp} at vacuum of stage 1	$I_{sp,vac,1}$	320.6	s
I_{sp} at vacuum of boosters	$I_{sp,vac,boosters}$	320.2	s
I_{sp} at sea level/vacuum of stage 2	$I_{sp,0,2}/I_{sp,vac,2}$	326	s
Thrust at sea level of stage 1	$T_{0,1}$	792410	N
Thrust at sea level of boosters	$T_{0,boosters}$	839480	N
Thrust at vacuum of stage 1	$T_{vac,1}$	921860	N
Thrust at vacuum of boosters	$T_{vac,boosters}$	1019930	N
Thrust at sea level/vacuum of stage 2	$T_{0,2}/T_{vac,2}$	298000	N
Structural mass of stage 1	$m_{s,1}$	6545	kg
Structural mass of stage 2	$m_{s,2}$	2355	kg
Structural mass of boosters	$m_{s,boosters}$	3784	kg
Wet mass of stage 1	$m_{w,1}$	99765	kg
Wet mass of stage 2	$m_{w,2}$	27755	kg
Wet mass of boosters	$m_{w,boosters}$	44413	kg
Number of engines of stage 1	$n_{e,1}$	1	-
Number of engines of stage 2	$n_{e,2}$	1	-
Number of boosters	$n_{boosters}$	4	-
Diameter of stage 1	D_1	2.95	m
Diameter of stage 2	D_2	2.66	m
Diameter of boosters	$D_{boosters}$	2.68	m
Duration of stage 1	t_1	286	s
Duration of stage 2	t_2	270	s
Duration of boosters	$t_{boosters}$	118	s
Duration of vertical ascent	t_{ascent}	10	s
Duration of gravity turn phase	t_{gt}	286	s
Minimum allowable pitch angle	θ_{min}	20	deg
Kick angle ¹	χ	87.495	deg
Injection altitude	h_{inj}	332	km
Injection velocity	v_{inj}	7.76037	km/s
Payload mass	m_{PL}	4230	kg

¹ Measured from the local horizontal.

The resulting injection conditions obtained from the simulation are compared with the imposed values in Table 3.10.

Table 3.10: Comparison between imposed and simulated injection conditions for the validation case.

Parameter	Imposed value	Obtained value	Error
Injection altitude	332 km	333.6237 km	1.6237 km
Injection velocity	7.76037 km/s	7.76037 km/s	0 km/s
Injection radial velocity	0 km/s	$-3.3435 \cdot 10^{-5}$ km/s	$-3.3435 \cdot 10^{-5}$ km/s

The obtained velocity is extremely accurate, while the altitude error remains acceptable for the purpose of these simulations. The radial velocity is practically zero, confirming that the injection occurs tangentially, as required. Given these results, the simulated trajectory is considered valid.

The three-dimensional trajectory, propagated in the ECEF reference frame, is shown in Figure 3.32, while Figure 3.33 presents the evolution of altitude with respect to ground range, showing the expected transition from the initial vertical ascent to a progressively horizontal flight path.

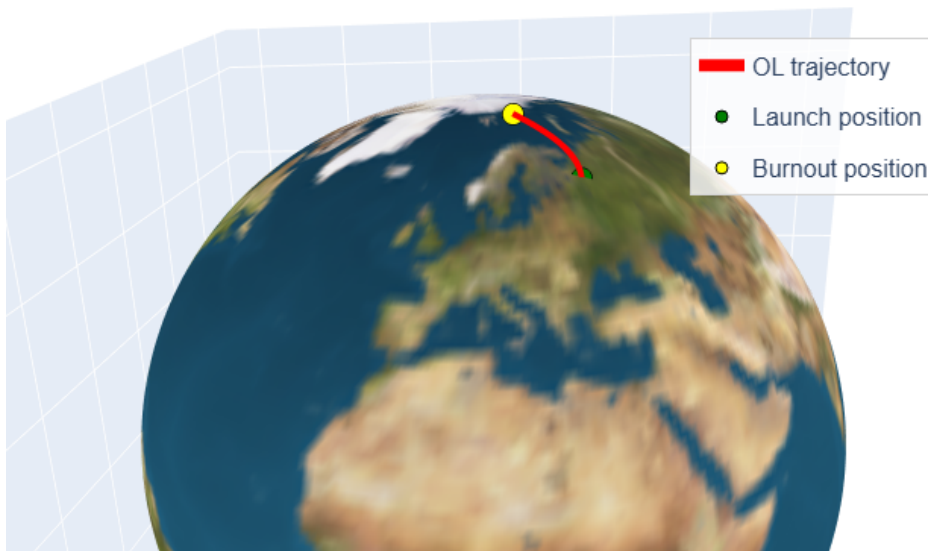


Figure 3.32: Simulated three-dimensional trajectory of an orbital launch vehicle in ECEF coordinates.

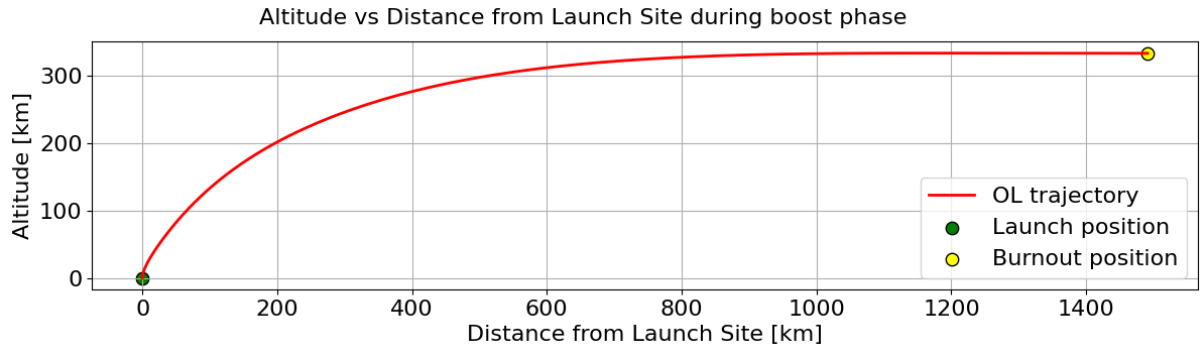


Figure 3.33: Altitude evolution with respect to ground range of an orbital launch vehicle.

Regarding velocity evolution, Figure 3.34 shows that the velocity magnitude reaches the target injection value, with the radial component converging to zero, so that the final velocity is completely tangential.

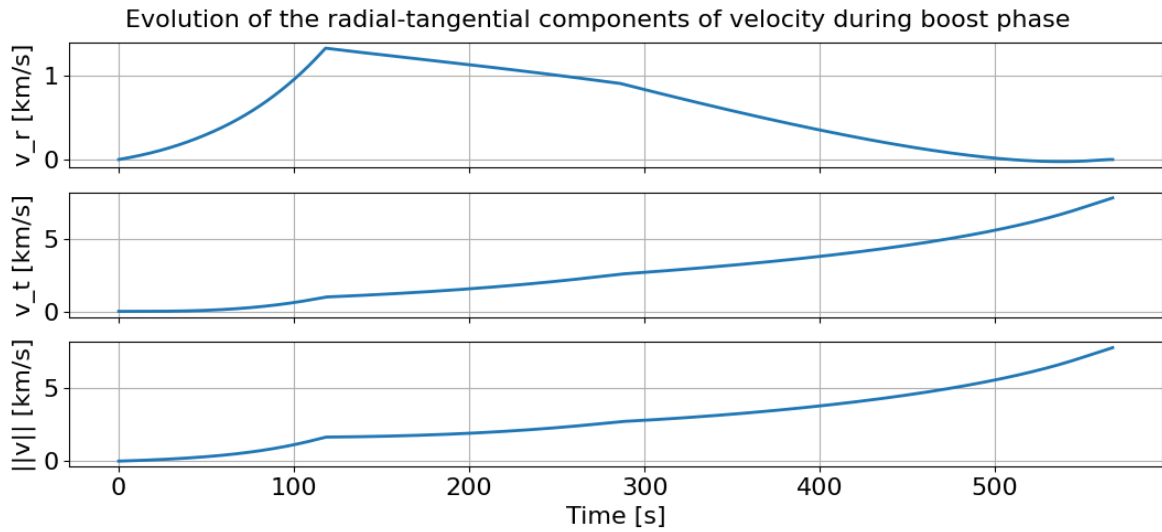


Figure 3.34: Evolution of velocity components during the ascent of an orbital launch vehicle.

The pitch profile obtained is practically a continuous curve, with small variations near the transition between the gravity turn phase and the PEG control, as Figure 3.35 shows, indicating a correct selection for the kick angle. The maximum angle of attack, reached during the terminal control phase, is approximately 6° , which is within the typical operational range for launch vehicles.

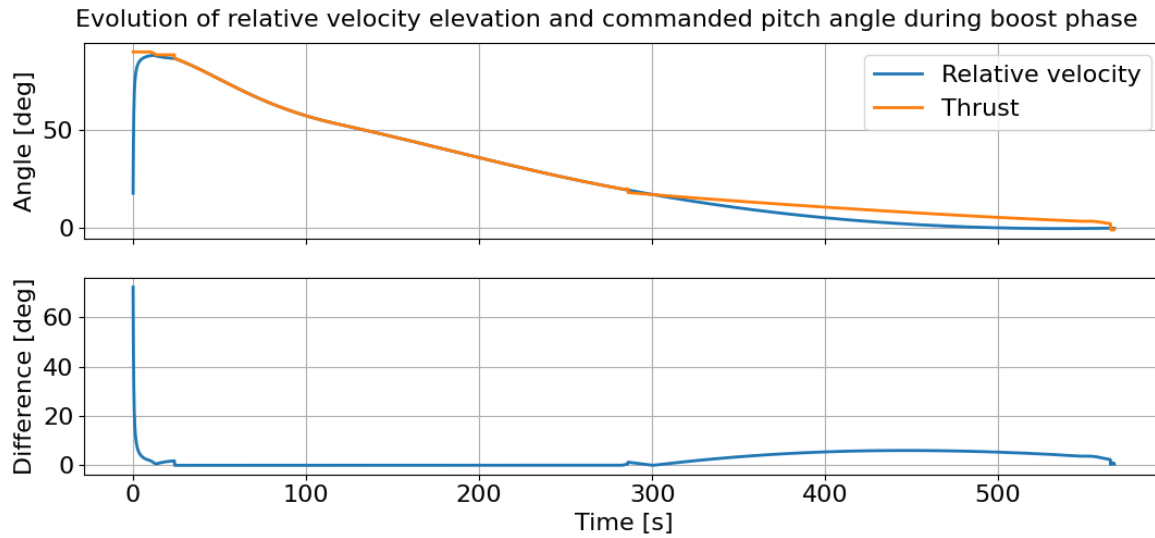


Figure 3.35: Comparison between commanded pitch angle and flight-path angle during the ascent of an orbital launch vehicle.

Finally, Figures 3.36 and 3.37 illustrate the evolution of thrust acceleration and altitude over time, respectively. The three distinct phases of boosters, core stage, and upper stage are identifiable in the acceleration profile, while the altitude evolution shows the flattening of the trajectory as the vehicle approaches the injection altitude. These profiles are consistent with the behaviour reported in the reference study [26].

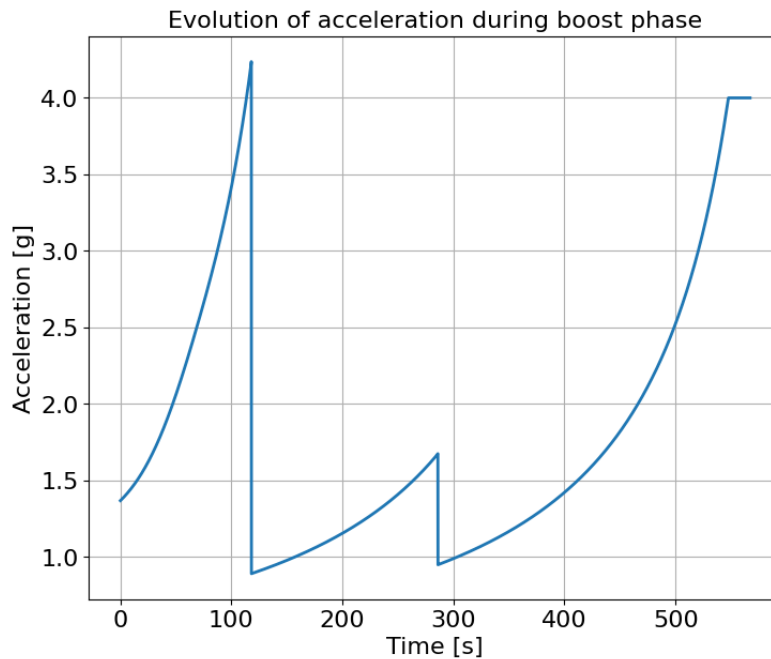


Figure 3.36: Evolution of thrust acceleration over time for an orbital launch vehicle.

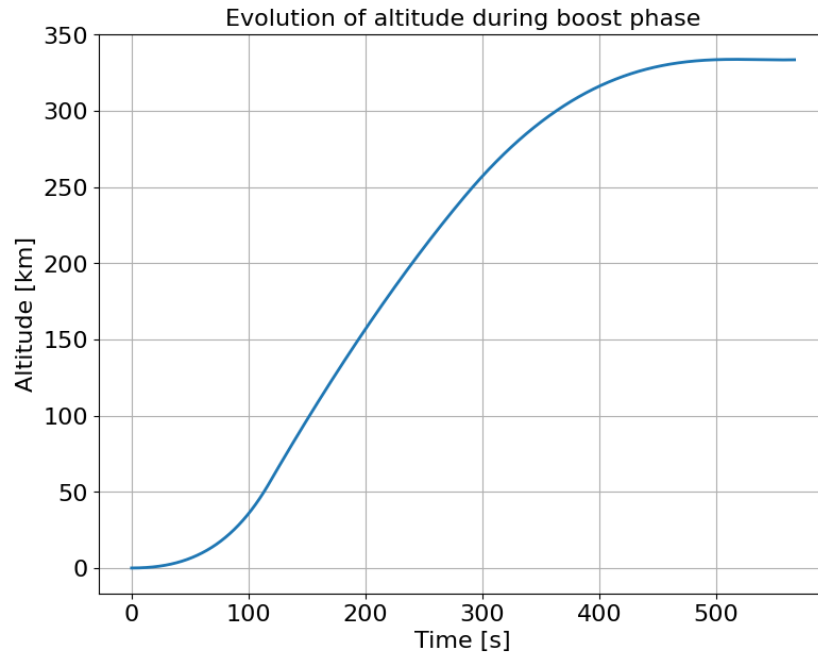


Figure 3.37: Evolution of altitude over time for an orbital launch vehicle.

3.4. Generation of trajectory database

As stated previously, one of the main objectives of the simulation framework is the generation of a trajectory database that covers a wide and representative range of flight profiles for each vehicle class. This section details the structure of the generated database and specifies the methodology applied.

3.4.1. Ballistic missile trajectory database

The database of ballistic missile trajectories is designed to cover the full operational range, from SRBMs to ICBMs. Before defining the generation strategy, an initial sensitivity analysis is performed to determine the influence of each parameter on the final achieved ground range. This preliminary study is focused on single-stage ballistic missiles, since the number of parameters is lower than in the multi-stage ICBM case. The analysis allows to limit the feasible parameter space to the most relevant region, ensuring that later trajectory searches converge efficiently.

Figures 3.38-3.44 illustrate this analysis, showing the absolute and relative variations in the achieved ground range caused by variations in each parameter.

Among all the variables considered, the exhaust velocity exhibits one of the strongest impacts. Variations in this parameter lead to significant percentage changes in range, as shown in Figure 3.38, which indicates that a broad interval of exhaust velocity values must be considered during the database generation process.

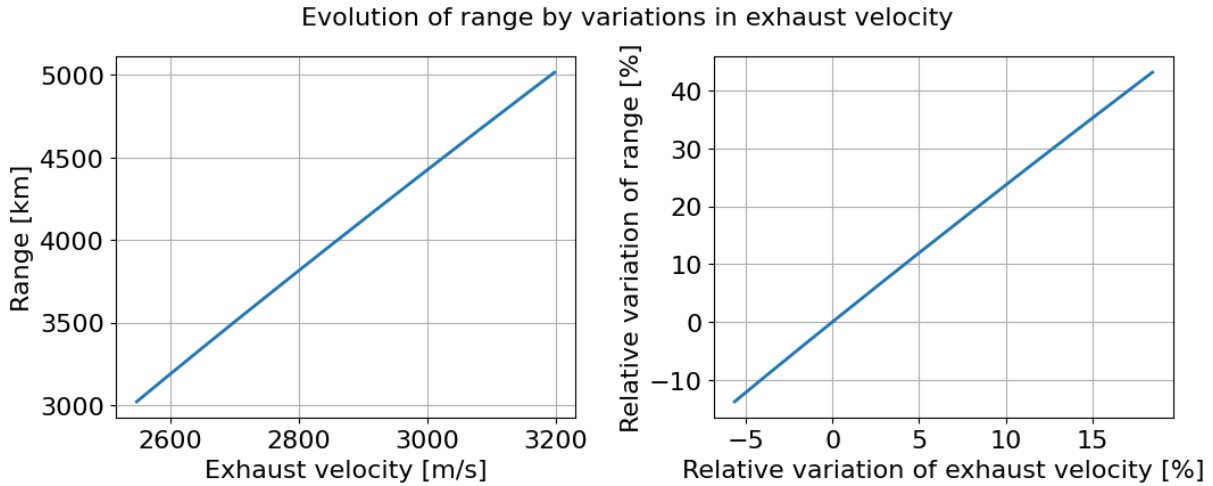


Figure 3.38: Evolution of ground range by variations in exhaust velocity.

The normalised burn rate (see Figure 3.39) also affects the outcome, but the magnitude of its influence is in lower orders. The results show a linear decrease in range as the burn rate increases across the observed interval. This indicates that slower combustion rates result in longer ranges, likely due to better trajectory shaping and lower gravity losses during the initial ascent phase. Theoretically, however, if this burn rate is too low, the range again starts decreasing, since the existence of an optimal burn rate is expected.

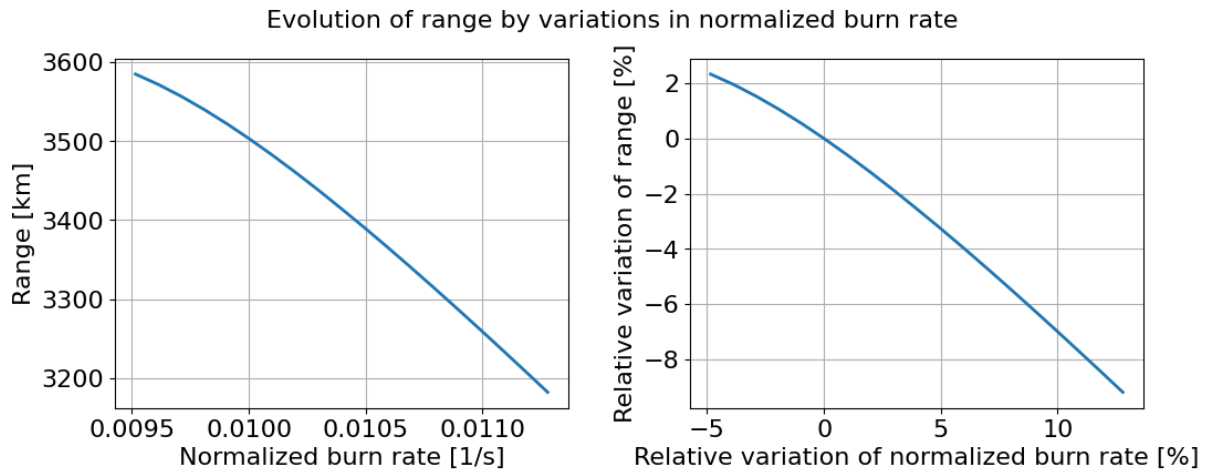


Figure 3.39: Evolution of ground range by variations in normalised burn rate.

For the ballistic coefficient, the analysis indicates that beyond a certain threshold, changes in this parameter no longer produce noticeable variations in range, as Figure 3.40 illustrates. At sufficiently high ballistic coefficients, atmospheric effects become negligible, which explains the asymptotic behaviour observed in the plots.

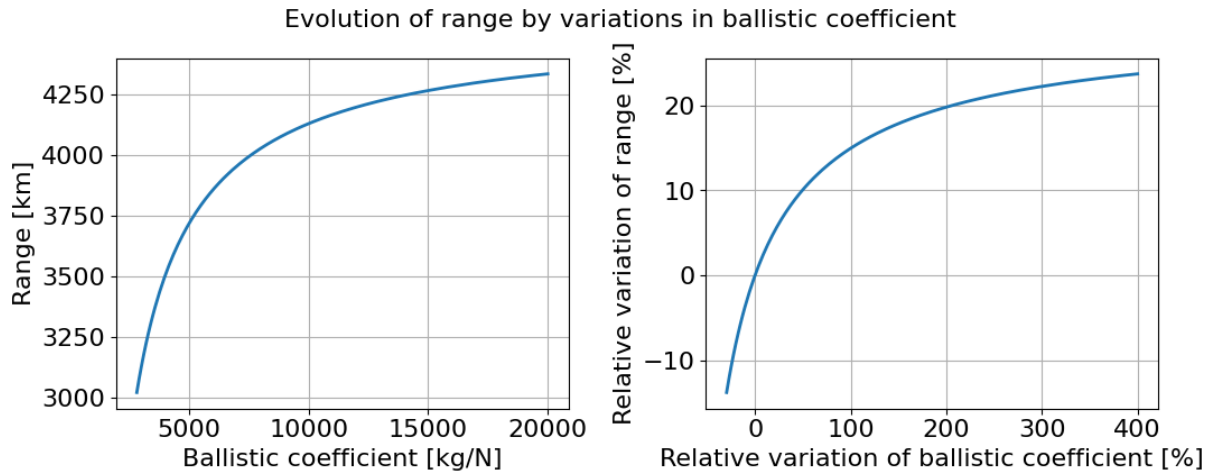


Figure 3.40: Evolution of ground range by variations in ballistic coefficient.

Changes in dry mass and propellant mass have a strong and intuitive effect. Decreasing the dry mass or, equivalently, increasing the propellant fraction results in an increase in the range (see Figures 3.41 and 3.42). This confirms that mass distribution must be treated as a key variable in any optimisation process.

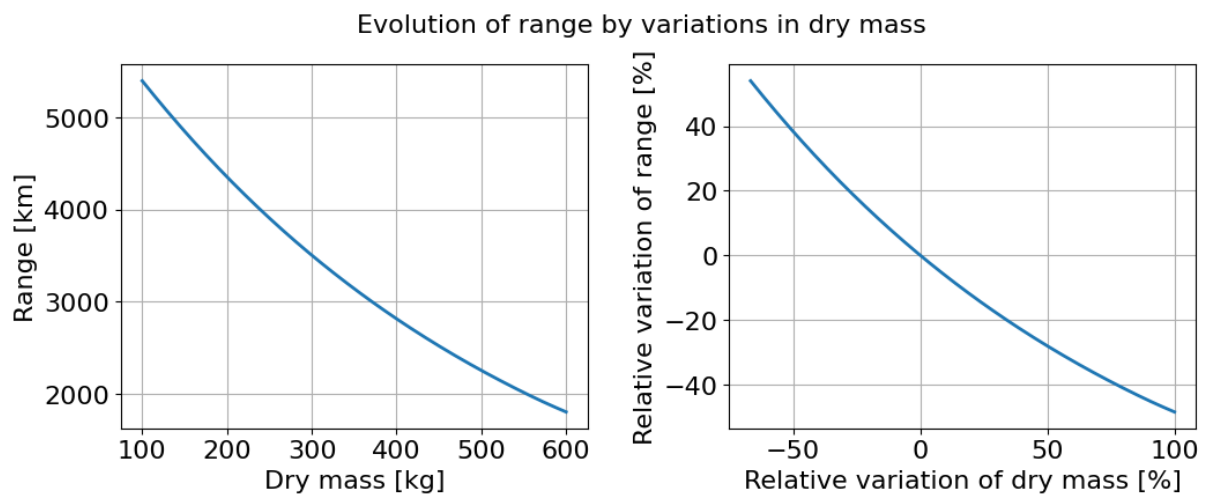


Figure 3.41: Evolution of ground range by variations in dry mass.

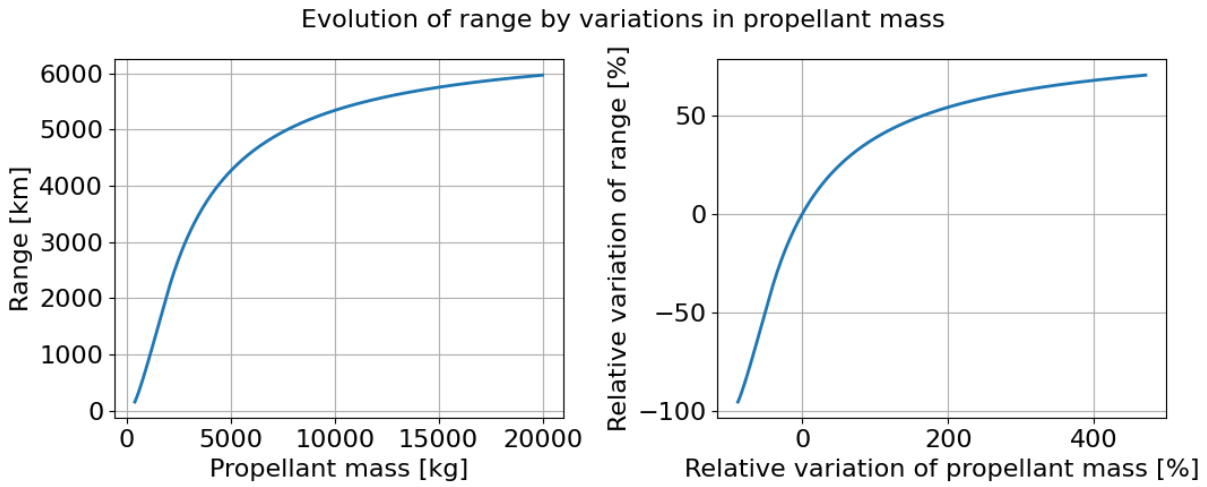


Figure 3.42: Evolution of ground range by variations in propellant mass.

The final pitch angle shows a more constrained but still relevant influence. For a selected final pitch angle, the launch elevation must be adjusted so that the guidance system allows the vehicle to reach a final flight-path angle close to the desired value. For this reason, only the selected final pitch angle is represented. The results highlight the existence of an optimal angle that maximises the range, as shown in Figure 3.43. Increasing the final pitch angle beyond this optimal point does not improve the horizontal distance achieved. Instead, it leads to similar range values but with a significantly longer flight time, as the missile reaches higher altitudes. In operational scenarios, particularly for this type of mission, longer flight times are undesirable due to the increased risk of interception. Therefore, final pitch angles are typically limited to values below approximately 50° for ballistic missiles.

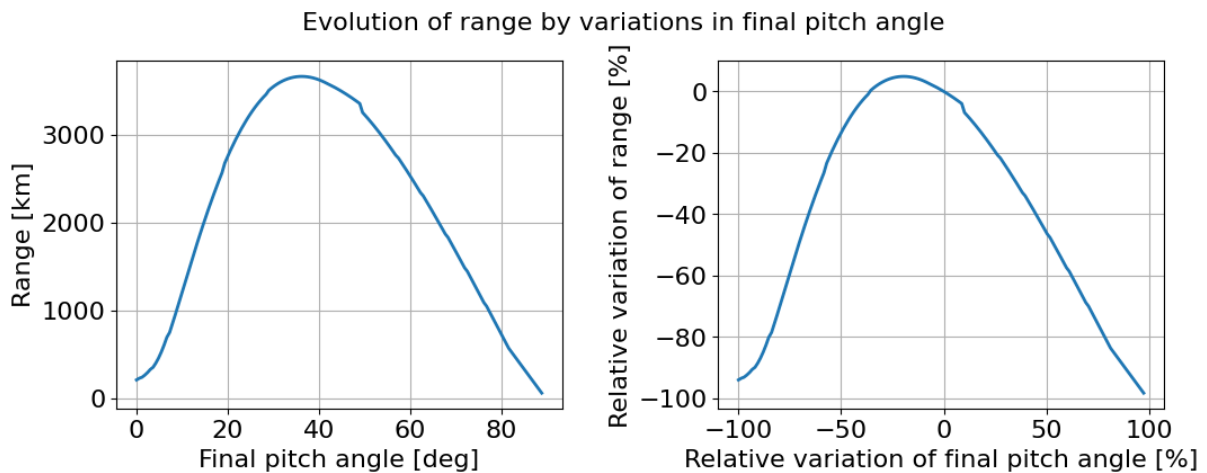


Figure 3.43: Evolution of ground range by variations in final pitch angle.

Finally, it is worth noting that, as discussed in Section 3.1.3, the vertical ascent time is fixed to a constant value. This is because its influence on the overall range is comparatively minor (see Figure 3.44), and treating it as a fixed parameter simplifies the guidance modelling without significantly affecting the results.

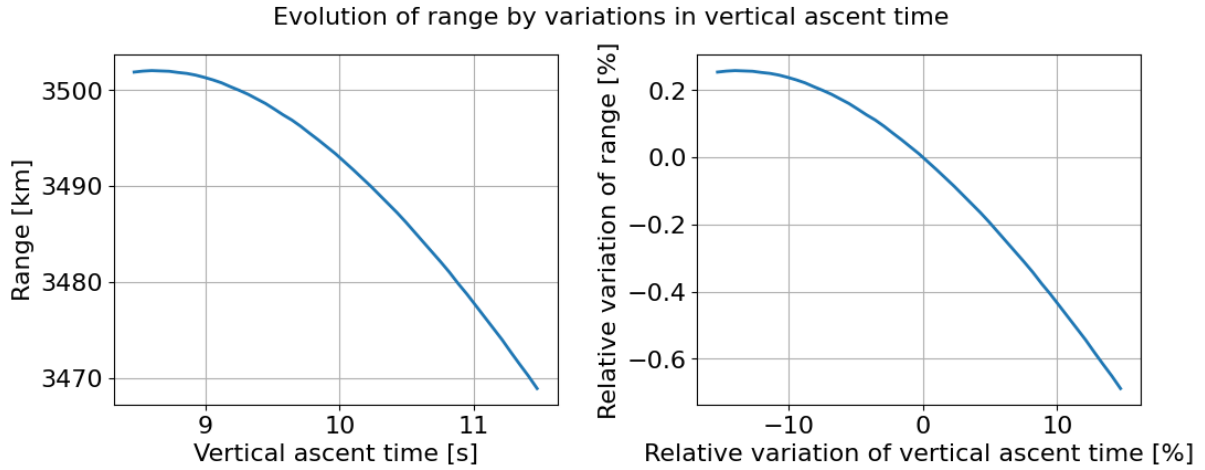


Figure 3.44: Evolution of ground range by variations in vertical ascent time.

Thus, the trajectory generation process for single-stage ballistic missiles consists of the following steps:

1. A set of target ground ranges is defined between 1000 and 5000 km, with increments of 500 km.
2. For each range, the optimal set of parameters achieving that exact range is computed. This nominal trajectory serves as a baseline for further variations.
3. Parameter sweeps are performed by varying the value of each parameter while keeping the rest fixed, ensuring that the database captures the dynamical impact of individual parameter uncertainties.
4. Finally, for each nominal trajectory, a full azimuth sweep from 0° to 360° is performed.

This process is repeated for two representative launch sites, one located in each hemisphere: Emamshahr Space Center, Iran ($\lambda = 54.976^\circ$, $\phi = 36.418^\circ$) and Cape York, Australia ($\lambda = 143.1^\circ$, $\phi = -12.25^\circ$). This methodology produces a dense and diverse dataset covering a wide range of possible flight directions and system behaviours.

The generation process for multi-stage ICBMs is similar, but with important differences due to the higher quantity of parameters of these systems. The nominal configuration

is based on the Minuteman III, already validated in Section 3.3.1. Consequently, several parameters remain fixed throughout the database generation process, including the structural masses and ballistic coefficients of each stage, and the re-entry vehicle mass.

Other parameters are allowed to vary slightly, such as exhaust velocities and normalised burn rate, while propulsion masses, launch elevation, and final pitch angle are treated with greater freedom due to their high influence on trajectory shaping.

The trajectory generation process is:

1. Target ground ranges between 5500 and 13000 km are defined.
2. For each range, the parameters achieving the desired impact distance are computed.
3. Since the number of parameters is significantly higher than in the single-stage case, no additional parameter sweeps are performed. Instead, a full azimuth sweep from 0° to 360° is generated for each nominal trajectory.

Similarly, this process is repeated for two representative launch sites: Emamshahr Space Center, Iran ($\lambda = 54.976^\circ$, $\phi = 36.418^\circ$) and Cape York, Australia ($\lambda = 143.1^\circ$, $\phi = -12.25^\circ$). In total, approximately 7000 ballistic missile trajectories are generated, covering ground ranges between 1000 and 13000 km.

3.4.2. ASAT trajectory database

The ASAT trajectory database is generated to represent the typical operational envelope of these systems, focusing on LEO, the region with highest satellite density and, thus, the most relevant target zone.

The generation process is based on the sweep of two mission parameters: the final burnout altitude and the final pitch angle. A sweep in burnout altitude is performed between 400 and 1000 km. Simultaneously, for each altitude, a sweep in the final pitch angle is conducted to model trajectories that impact at the same altitude but with different relative geometries and closure velocities.

For each combination of burnout altitude and final pitch angle, the optimal set of parameters leading to the desired final conditions is computed. Similar to the multi-stage ICBM case, the nominal vehicle is defined from Mission Shakti, validated in Section 3.3.2. Thus, the structural masses and ballistic coefficients are fixed, while exhaust velocities, normalised burn rates, propellant masses, and launch elevation are adjusted to achieve the desired impact conditions.

Finally, a full azimuth sweep from 0° to 360° is generated for each nominal trajectory.

This process is repeated for two representative launch sites: Abdul Kalam Island, India ($\lambda = 87.084^\circ$, $\phi = 20.758^\circ$) and Xichang Satellite Launch Centre, China ($\lambda = 102.03^\circ$, $\phi = 28.246^\circ$). The resulting database captures a wide range of ASAT interception geometries and impact conditions. The total number of ASAT trajectories is approximately 8500.

3.4.3. Orbital launch vehicle trajectory database

The database of orbital launcher trajectories is designed to represent a broad range of operational scenarios. The generation process involves three characteristic launchers (corresponding to the top three space powers) and a variety of launch sites, flight directions, and target orbits.

The selected orbital launch vehicles are the Falcon 9 (U.S.), Soyuz 2.1a (Russia), and Long March (CZ) 7 (China). Each launcher is simulated from a typical domestic launch site, complemented by an alternative site. For each launch site, six representative azimuths are considered, covering the typical operational azimuth range that enables injection into different orbital inclinations, as detailed in Table 3.11.

The Guiana Space Centre (GSC) is selected as the alternative launch site due to its equatorial location, which makes it a representative case. The selected azimuths for all sites take into account the geographical and operational constraints of each location [46, 47].

Table 3.11: Launch vehicles, sites, and selected azimuths for orbital launch vehicle database generation.

Launcher	Launch Site	Selected Azimuths (deg)
Falcon 9	AFETR ¹	35, 50, 75, 90, 105, 120
	GSC ²	35, 50, 75, 90, 105, 120
Soyuz 2.1a	PKMTR ³	4.8, 13.7, 15.2, 41.6, 76.2, 341.5
	GSC	35, 50, 75, 90, 105, 120
Long March (CZ) 7	WSC ⁴	90, 105, 130, 180, 200, 225
	GSC	35, 50, 75, 90, 105, 120

¹ Air Force Eastern Test Range (AFETR).

² Guiana Space Centre (GSC).

³ Plesetsk Cosmodrome (PKMTR).

⁴ Wenchang Satellite Launch Centre (WSC).

For each combination of launcher, launch site, and azimuth, the vehicle is simulated injecting into different target orbits. The injection altitude is swept from 300 to 700 km, and for each altitude, the injection velocity is varied to simulate injections into orbits with different eccentricities, representing different mission profiles.

This methodology produces a broad set of realistic ascent trajectories for orbital launch vehicles, resulting in a database of approximately 7000 trajectories.

To provide a visual illustration of the trajectory database, a representative subset of the generated trajectories is shown in Figures 3.45 and 3.46. These figures do not represent the full dataset, but rather aim to qualitatively illustrate its density and diversity.

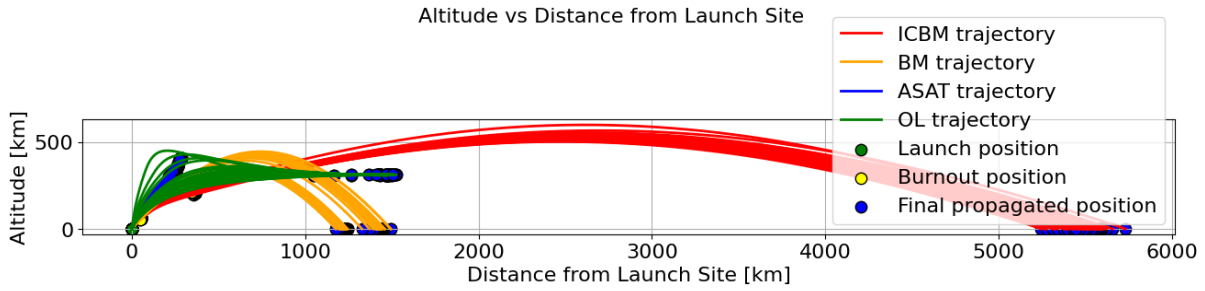


Figure 3.45: Representative subset of the generated trajectories.

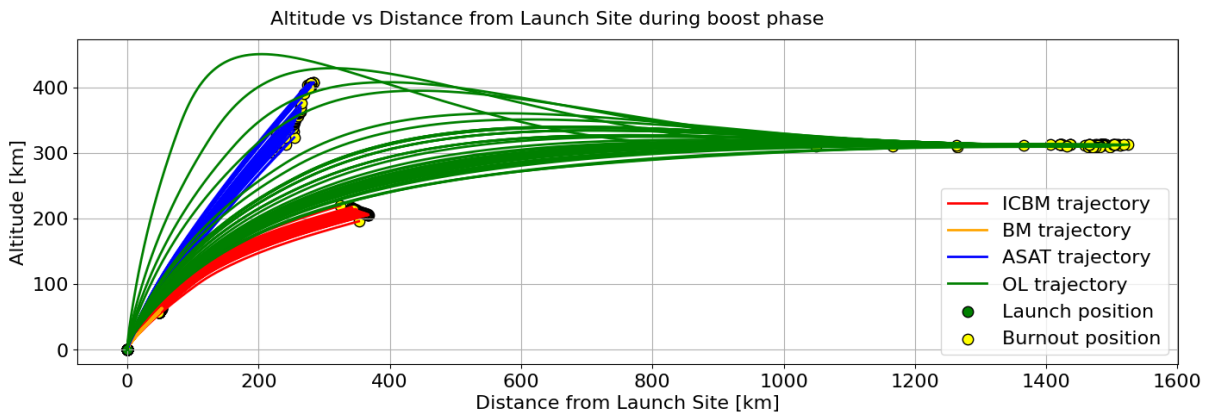


Figure 3.46: Representative subset of the generated trajectories (during boost phase).

4 | Trajectory identification and classification

In this chapter, a methodology for the identification and classification of direct ascent trajectories is developed, with the objective of differentiating between the different classes of vehicles considered in this work based on measured state variables.

The chapter begins with a post-processing and analysis of the generated trajectory database. The evolution of several kinematic and dynamic variables along the trajectory is examined, focusing on the main differences in behaviour among the considered vehicle classes. Based on this analysis, the most significant variables for trajectory differentiation are identified, providing the basis for the subsequent application of a classification algorithm.

The second part of the chapter focuses on the implementation of this trajectory classification algorithm. The adopted classification strategy, based on the K-Nearest Neighbours (KNN) algorithm, is presented together with its validation and optimisation.

Finally, the chapter is extended to a more realistic operational scenario by modelling the data acquisition process. A simplified sensor network is defined to represent the measurement of trajectory variables, and the impact of measurement noise and uncertainty is analysed. The performance and robustness of the classification algorithm are then assessed under noisy measurement conditions.

4.1. Post-processing and analysis of trajectory database

In this section, the trajectory database generated in Chapter 3 is post-processed to analyse the evolution of the different variables across the flight-path. The aim of this analysis is, first, to characterise the typical behaviour of ballistic missiles, anti-satellite (ASAT) systems, and orbital launch vehicles and, second, to identify those variables that exhibit the most significant differences between classes.

The analysis focuses on altitude, velocity magnitude, flight-path angle, radial and tangen-

tial velocity components, rate of climb, and acceleration magnitude. By examining the evolution of these variables during the trajectory evolution, trends that are distinctive for each trajectory type can be identified.

4.1.1. Evolution of trajectory variables

Firstly, the temporal evolution of these variables is analysed for all simulated trajectories. To facilitate qualitative comparison between the different classes, all trajectories are represented together in the same plot. Thus, Figures 4.1-4.7 represent the evolution of altitude, velocity magnitude, flight-path angle, radial and tangential velocity components, rate of climb and acceleration magnitude, respectively.

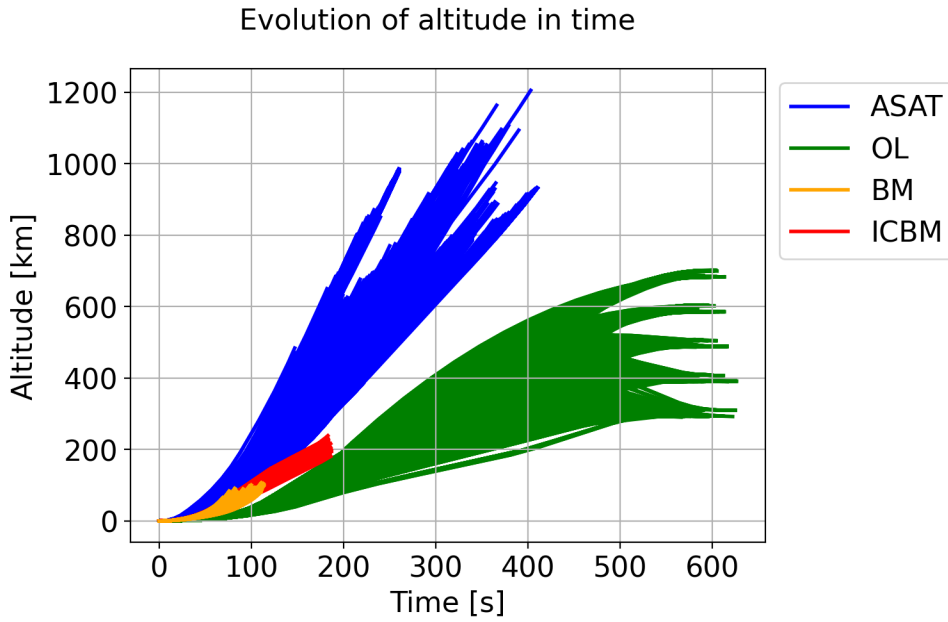


Figure 4.1: Temporal evolution of the altitude for all simulated trajectories.

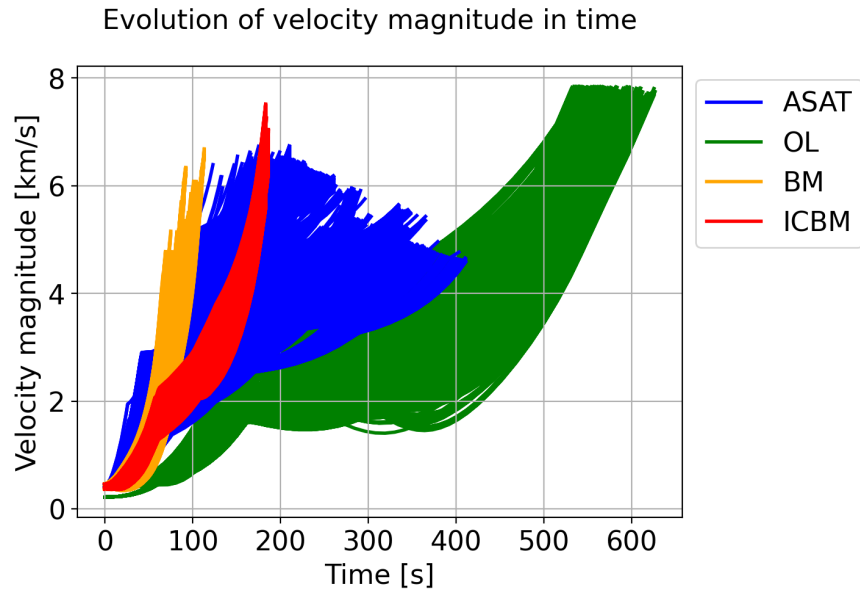


Figure 4.2: Temporal evolution of the velocity magnitude for all simulated trajectories.

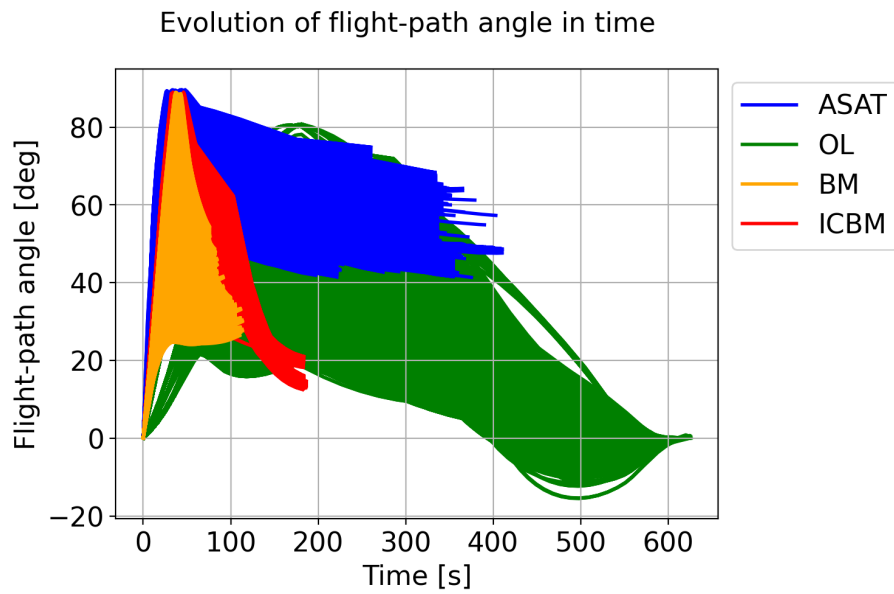


Figure 4.3: Temporal evolution of the flight-path angle for all simulated trajectories.

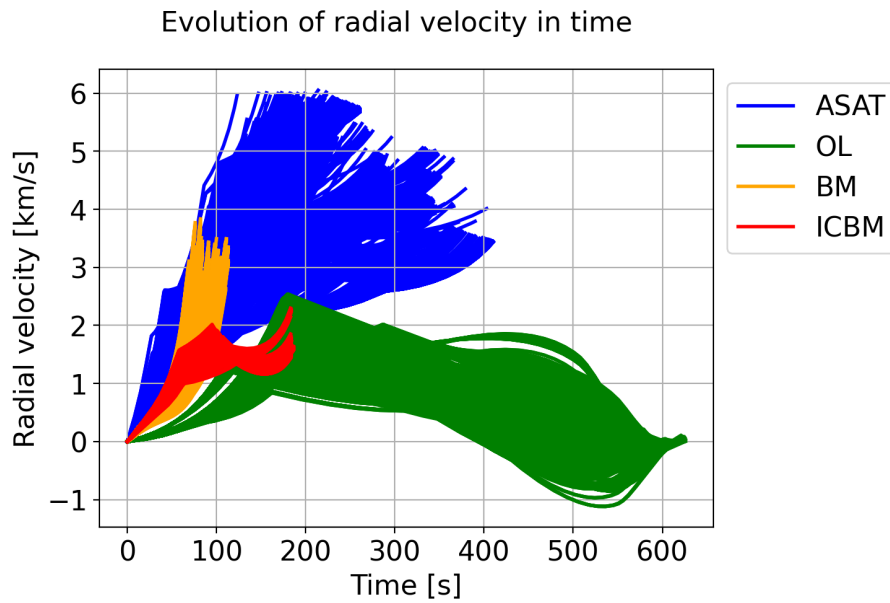


Figure 4.4: Temporal evolution of the radial velocity component for all simulated trajectories.

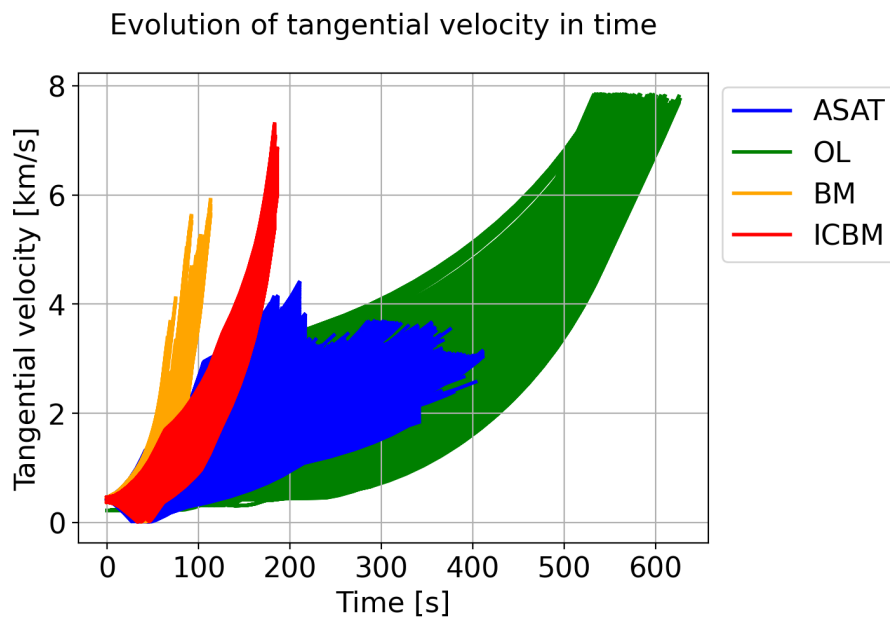


Figure 4.5: Temporal evolution of the tangential velocity component for all simulated trajectories.

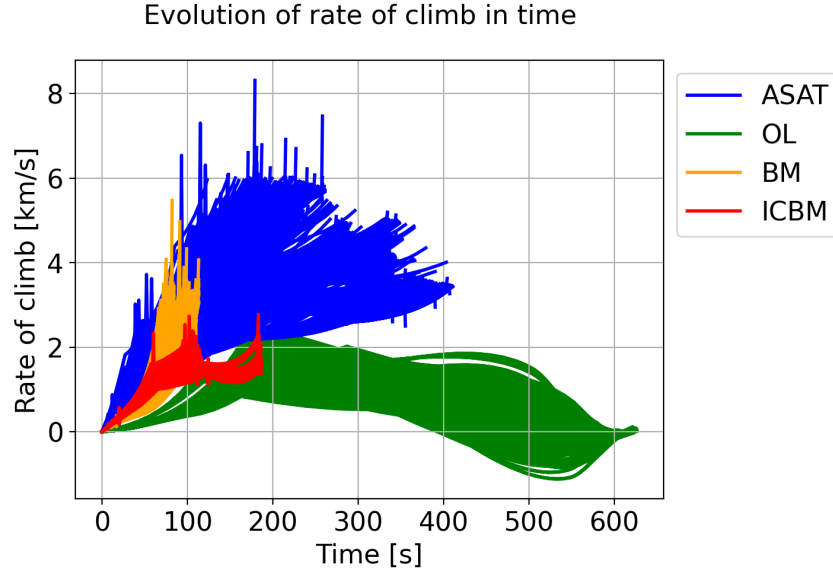


Figure 4.6: Temporal evolution of the rate of climb for all simulated trajectories.

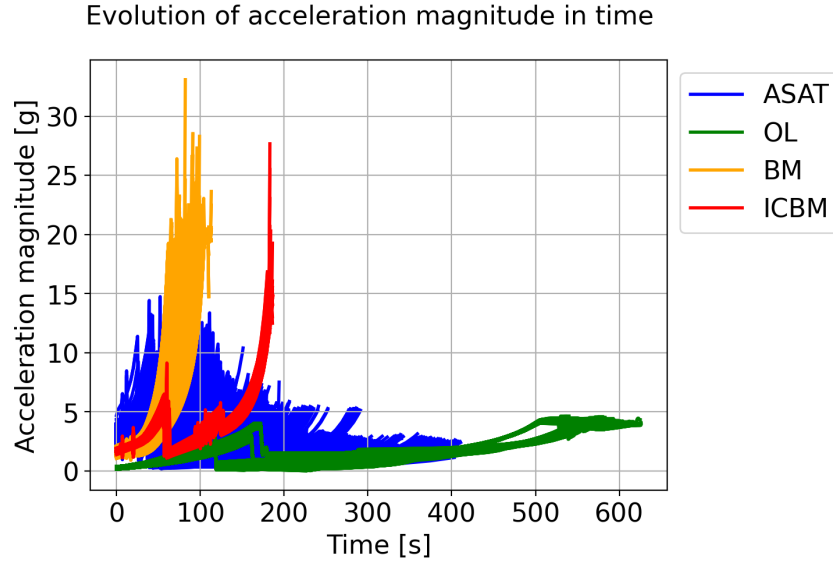


Figure 4.7: Temporal evolution of the acceleration magnitude for all simulated trajectories.

From this preliminary analysis, several conclusions can be drawn. First, for any given time instant, each vehicle class exhibits a distinguishable range of values for each trajectory variable. As a result, four distinct groups can be identified in all analysed variables, each corresponding to one of the vehicle classes considered. Although partial overlap may occur at certain time instants, the overall separation between classes becomes clearer as time grows.

Second, the presence of redundancy among some of the investigated variables. In particular, the radial and tangential components of the velocity vector can be directly obtained from the velocity magnitude and the flight-path angle. Consequently, for any given pair of velocity magnitude and flight-path angle values, a unique pair of radial and tangential velocity components exists, implying that these variables do not provide additional information. A similar observation applies to the rate of climb, which exhibits behaviour closely related to the radial velocity component.

Based on this consideration, the set of variables is reduced to those that are both physically meaningful and independent. From this point on, the analysis focuses on altitude, velocity magnitude, flight-path angle, and acceleration magnitude as the main variables of interest.

Furthermore, from an operational point of view, the time of flight is not always a suitable independent variable. When an object is first detected, the elapsed time since launch is generally unknown, while state variables such as altitude can be derived from sensor measurements. For this reason, altitude is selected as the independent variable for the subsequent analysis.

Consequently, the evolution of the velocity magnitude, flight-path angle, and acceleration magnitude is represented as a function of altitude, as shown in Figures 4.8, 4.9, and 4.10, respectively.

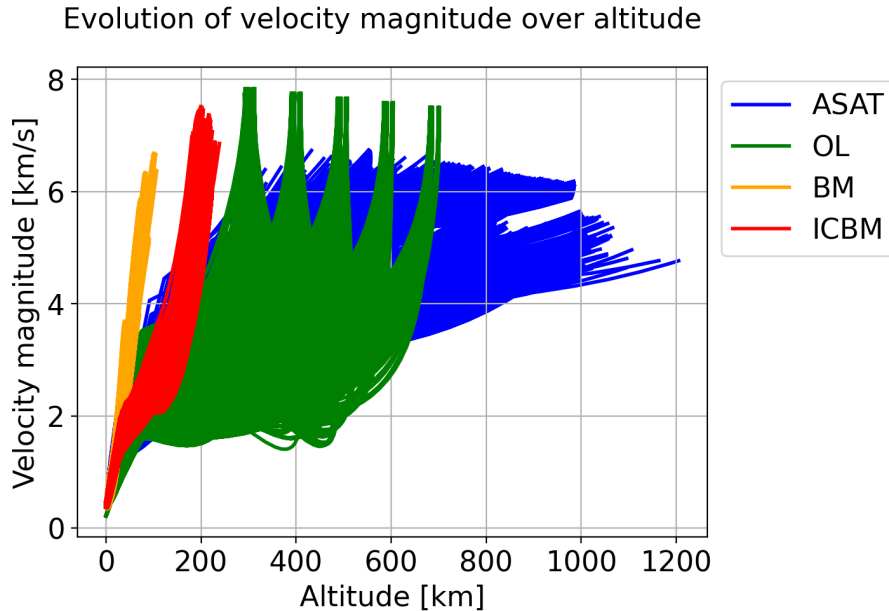


Figure 4.8: Evolution of the velocity magnitude over altitude for all simulated trajectories.

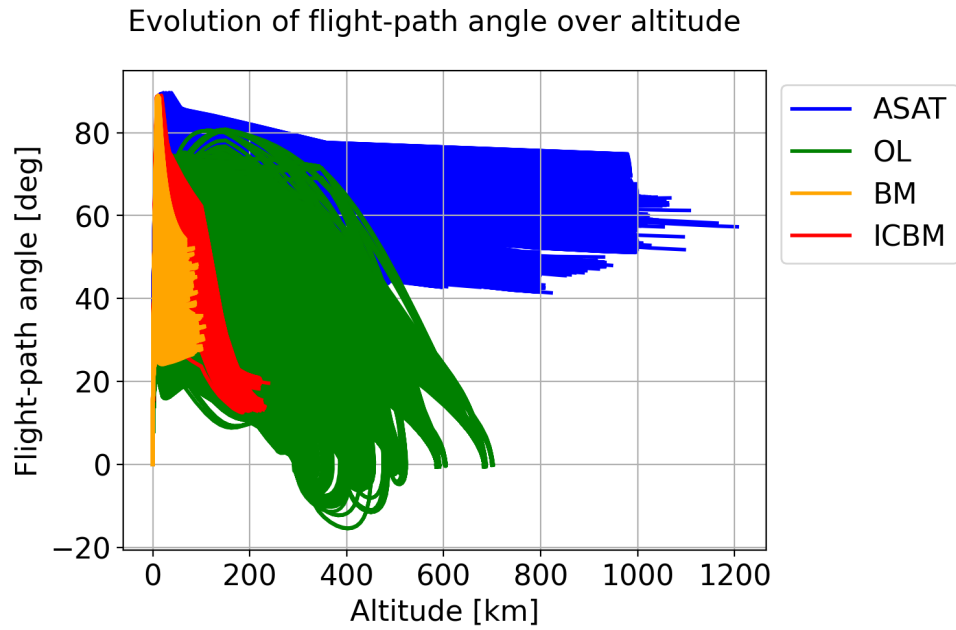


Figure 4.9: Evolution of the flight-path angle over altitude for all simulated trajectories.

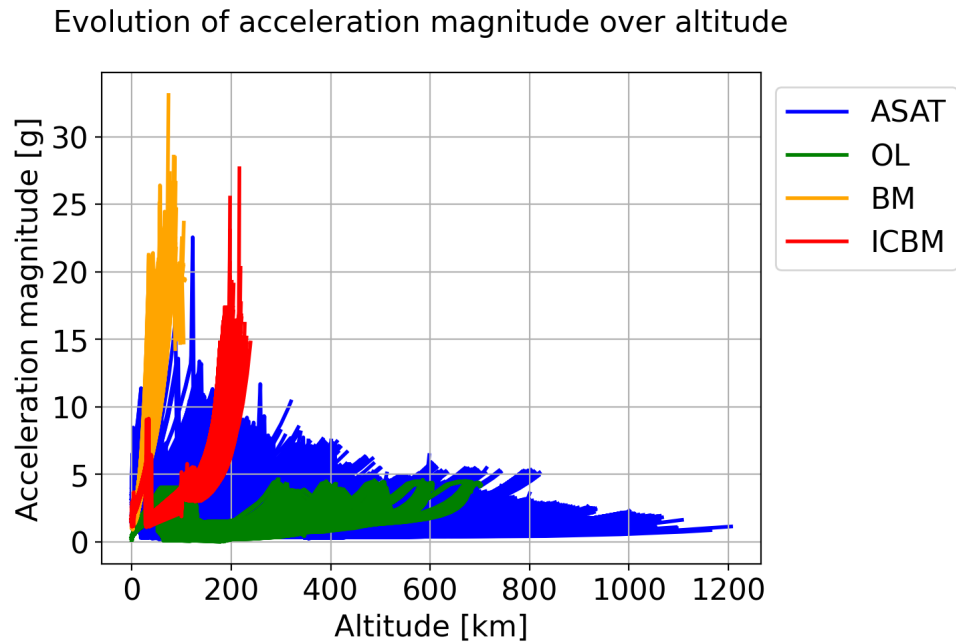


Figure 4.10: Evolution of the acceleration magnitude over altitude for all simulated trajectories.

If, for each of the analysed variables, the final value reached by the trajectory is represented together with the corresponding final altitude, as shown in Figures 4.11, 4.12, and 4.13, distinct and clearly separated point clouds are obtained. These point clouds indicate that the trajectory type could be classified with a high level of confidence once the final state of the boost phase is known, by comparing the observed values with those of the plot.

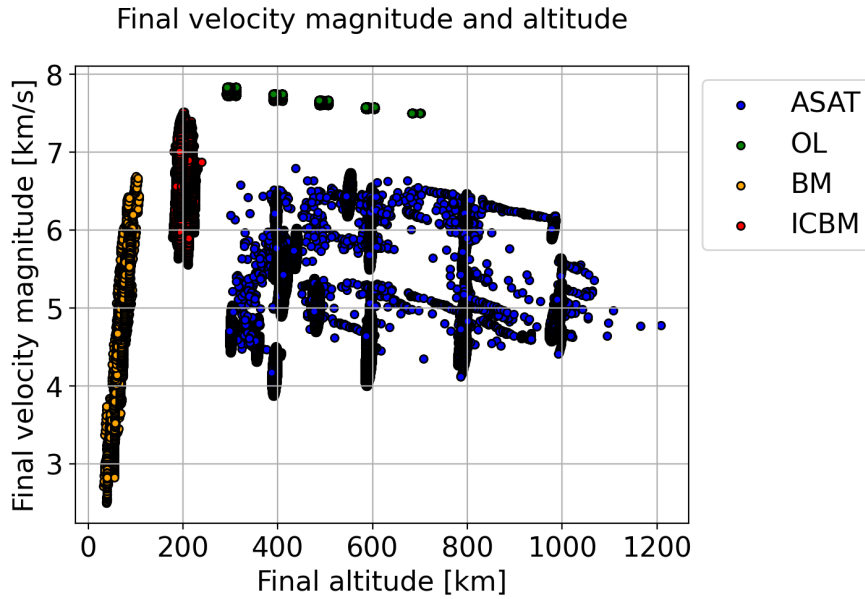


Figure 4.11: Final velocity magnitude and altitude reached for all simulated trajectories.

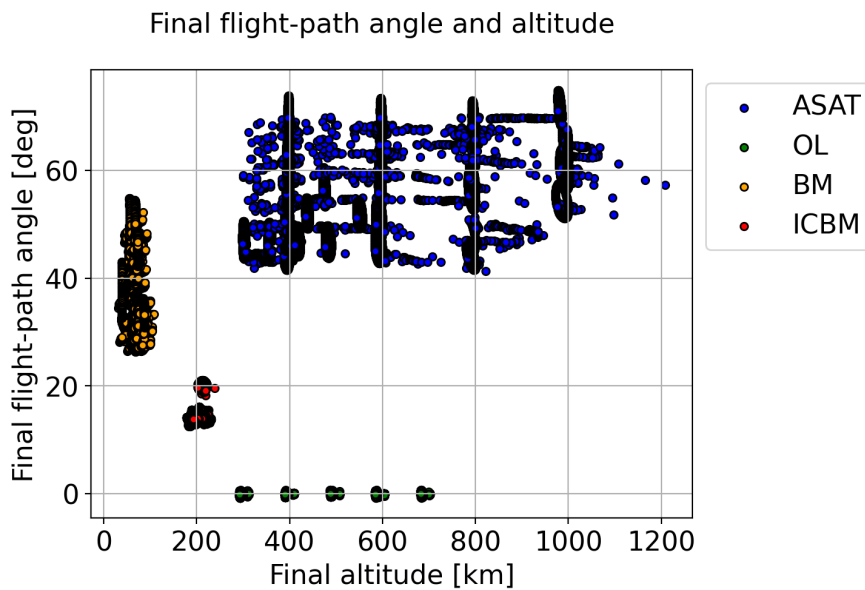


Figure 4.12: Final flight-path angle and altitude reached for all simulated trajectories.

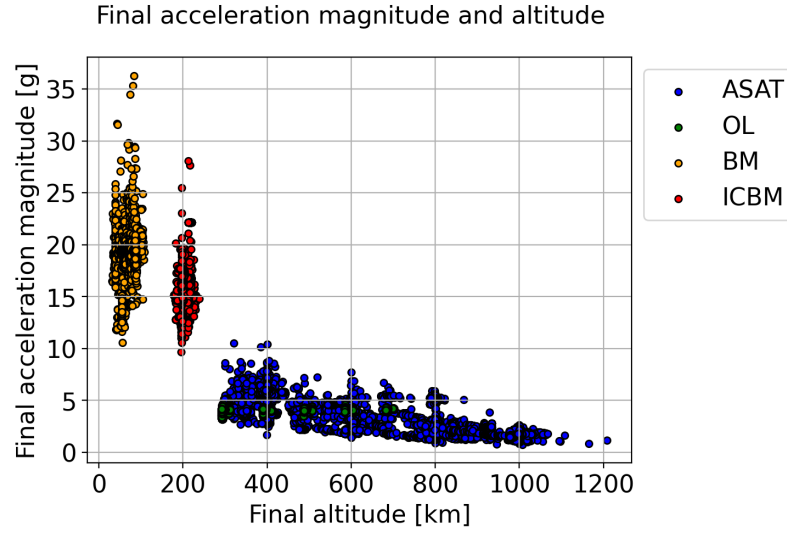


Figure 4.13: Final acceleration magnitude and altitude reached for all simulated trajectories.

However, with this methodology, trajectory classification would require waiting until the end of the boost phase, which significantly limits it. In realistic scenarios, early identification is essential, as it enables the preparation of appropriate defensive measures against a potential threat in the case of a hostile vehicle. Consequently, an alternative strategy is adopted.

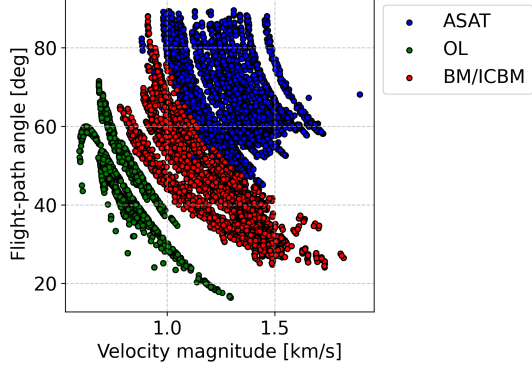
4.2. Trajectory classification methodology

Instead of waiting for the burnout state, this methodology analyses the selected variables at intermediate altitude levels. By comparing the instantaneous state of a vehicle against the corresponding point clouds obtained from the simulated trajectory database at the same altitudes, the trajectory can be classified during the boost phase.

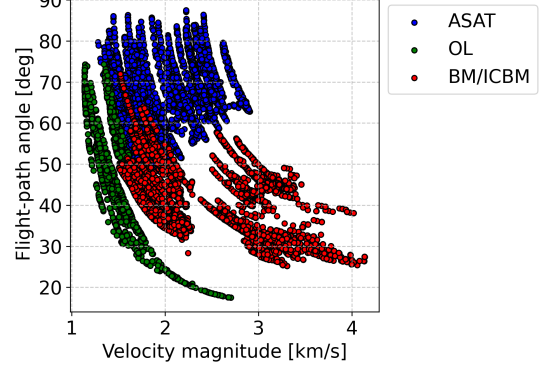
4.2.1. Altitude-based classification strategy

In order to evaluate the discriminative capability of the selected variables, their values are extracted and visualised at discrete altitude levels. This process allows for the identification of clustering patterns associated with each vehicle class as the trajectory evolves.

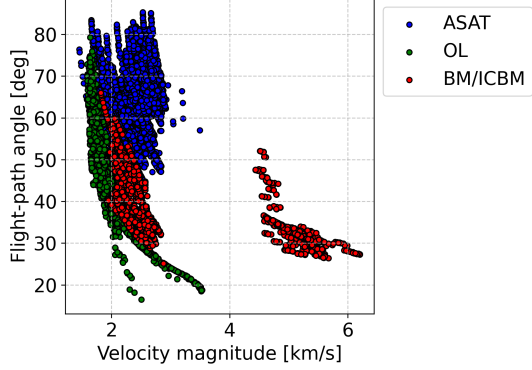
The analysis is initially performed using two-dimensional feature space, considering the velocity magnitude and the flight-path angle as classification variables. The evolution of the corresponding points with altitude is shown in Figure 4.14. For clarity, only a limited number of representative altitude levels are shown.

Flight-path angle vs. velocity magnitude at $h = 20$ km

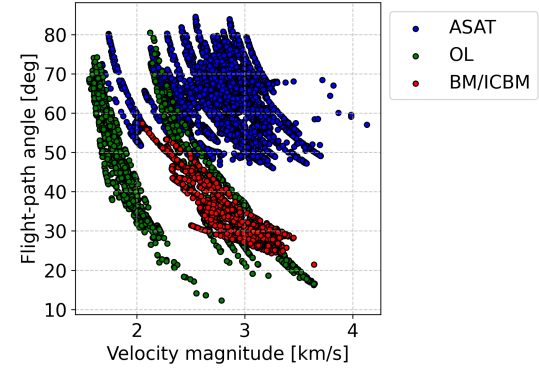
(a) 20 km

Flight-path angle vs. velocity magnitude at $h = 50$ km

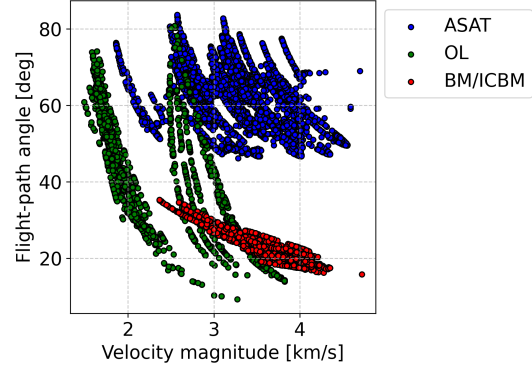
(b) 50 km

Flight-path angle vs. velocity magnitude at $h = 80$ km

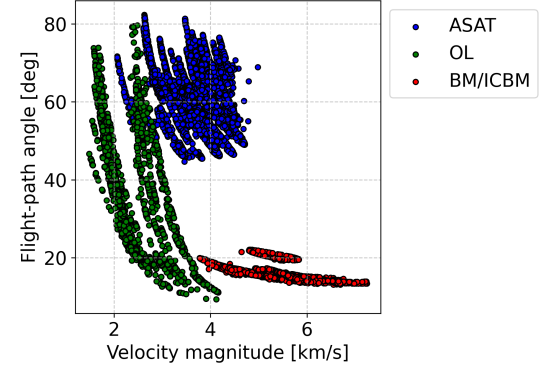
(c) 80 km

Flight-path angle vs. velocity magnitude at $h = 110$ km

(d) 110 km

Flight-path angle vs. velocity magnitude at $h = 145$ km

(e) 145 km

Flight-path angle vs. velocity magnitude at $h = 190$ km

(f) 190 km

Figure 4.14: Evolution of the two-dimensional feature space (velocity magnitude and flight-path angle) at increasing altitude levels.

The point clouds associated with each trajectory class are initially well separated. At intermediate altitude levels, they begin to converge and overlap, resulting in a less distinct

classification. However, at higher altitudes, they separate again, leading to a much clearer distinction between trajectory classes.

To further improve this separation, the acceleration magnitude is included to construct a three-dimensional feature space (an illustrative example is shown in Figure 4.15), whose projections evolution with altitude is represented in Figure 4.16. In a similar manner, only a limited number of representative altitude levels are shown. For completeness, the full altitude evolution of the three-dimensional feature space is provided in Appendix B.

Three-dimensional feature space at $h = 15$ km

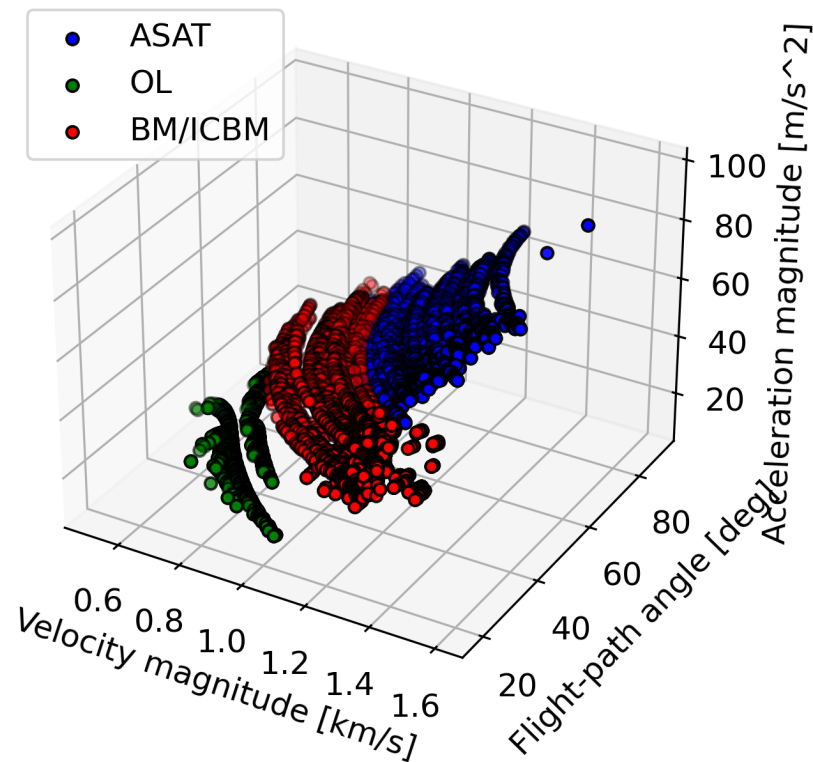
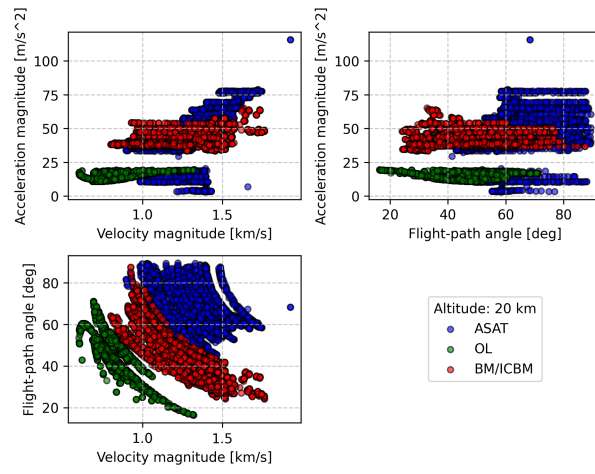
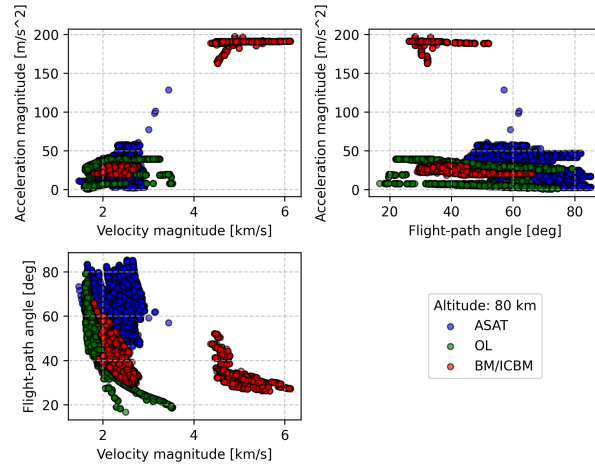


Figure 4.15: Three-dimensional feature space at a classification altitude of 15 km.

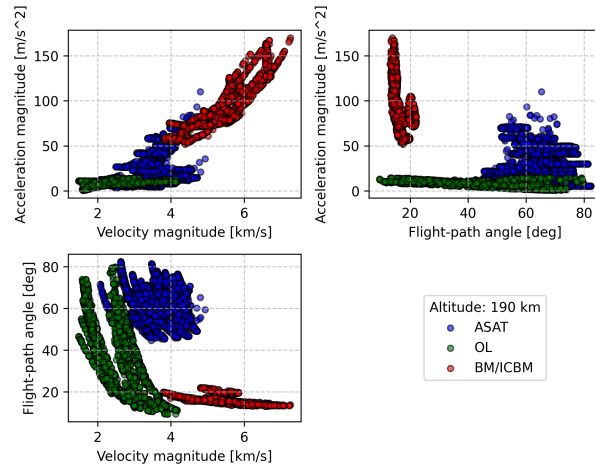
In this way, points that appeared to overlap in the two-dimensional representations at certain altitude levels can be distinguished when an additional dimension is introduced. In particular, differences in acceleration magnitude reveal that these points do not occupy the same region of the three-dimensional feature space, thereby enhancing the discriminative capability.



(a) 20 km



(b) 80 km



(c) 190 km

Figure 4.16: Evolution of the three-dimensional feature space (velocity magnitude, flight-path angle and acceleration magnitude) projections at increasing altitude levels.

Therefore, the clustering observed in the altitude-dependent feature space justifies the use of a distance-based classification approach, which is detailed in the following subsection.

4.2.2. Application of the K-Nearest Neighbours (KNN) algorithm

The K-Nearest Neighbours (KNN) algorithm is selected as the distance-based classification method, whose theoretical basis is briefly described in Appendix C. This algorithm is particularly suitable for this application, as it directly uses the clustering behaviour observed in the point clouds to classify new data points based on their proximity in the feature space.

For any given altitude, the trajectory to be classified is defined by its corresponding set of variables and compared against the labelled reference points of the database. The class of the unknown trajectory is assigned based on the majority class among its K nearest neighbours. Additionally, the algorithm provides a classification probability for each class, determined by the relative frequency of the classes among the neighbours and weighted by their respective distances to the query point.

The classification is initially implemented in a two-dimensional feature space using the velocity magnitude and the flight-path angle as classification variables. Figure 4.17 presents a representative classification case at an altitude of 30 km, where the yellow point corresponds to the unknown trajectory to be classified. In this specific example, the proximity to the ballistic missile cluster (red) enables a clear identification.

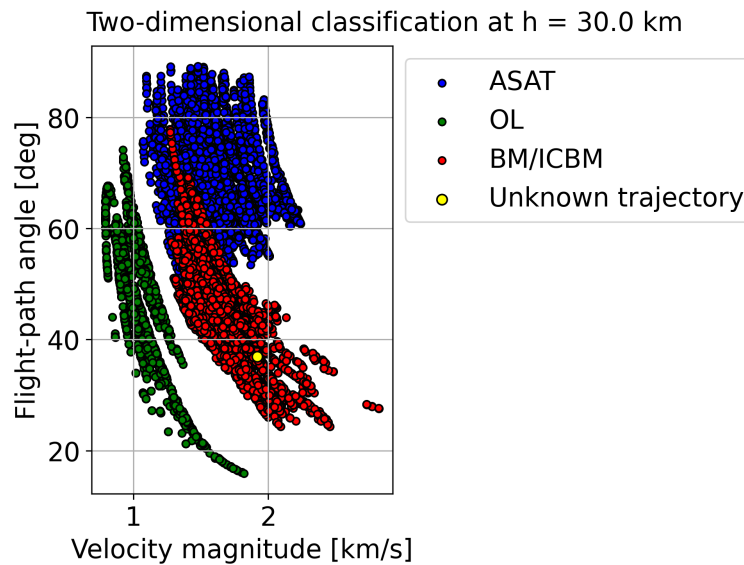


Figure 4.17: Example of two-dimensional classification at an altitude of 30 km.

In all of the following analysis, the reference point clouds are constructed using a subset of the simulated trajectory database, which is used as the training set. Then, 1000 random trajectories are extracted from the remaining subset, forming an independent validation set used for testing. For each of them, classification is performed across different altitude levels, and the predicted class is compared with the ground truth, evaluating the rate of success of the algorithm. With this process, the classification accuracy is evaluated as a function of altitude, allowing the assessment of both the overall performance of the algorithm and the potential for early identification.

Based on this framework, the following sections analyse the influence of key parameters and modelling choices on the effectiveness of the classification algorithm.

Influence of the number of neighbours K

The first parameter to be selected in the KNN algorithm is the number of neighbours, K , used in the classification decision. The choice of this parameter directly affects the balance between sensitivity to local structures and robustness against potential noise. Since trajectory classes form relatively compact and well-separated clusters in the feature space, excessively large values of K may lead to an undesirable smooth effect, incorporating samples from distant regions.

To evaluate this influence, the success rate of the algorithm is evaluated as a function of the altitude at which the classification is performed for $K = 5, 25$, and 55 , using 30 % of the database as the training set to build the feature space, as shown in Figure 4.18. Only odd values of K are considered in order to avoid potential ties in the counting.

The results indicate that the overall success rate exhibits low sensitivity to the value of K within the tested range. However, a slight decrease in performance is observed at high altitudes as K increases. Consequently, $K = 25$ is selected for the classification algorithm, as it provides a suitable compromise between local sensitivity and robustness. All results presented in the following sections correspond to this selected value.

Figure 4.19 presents the classification success rate as a function of the classification altitude for the selected value of $K = 25$. The accuracy decreases as the vehicle ascends towards an altitude of approximately 100 km, subsequently improving at higher levels. This behaviour can be explained by the evolution of the feature space previously discussed (see Figure 4.14), since, as altitude increases, the clusters converge and overlap before becoming more distinct again beyond approximately 100 km.

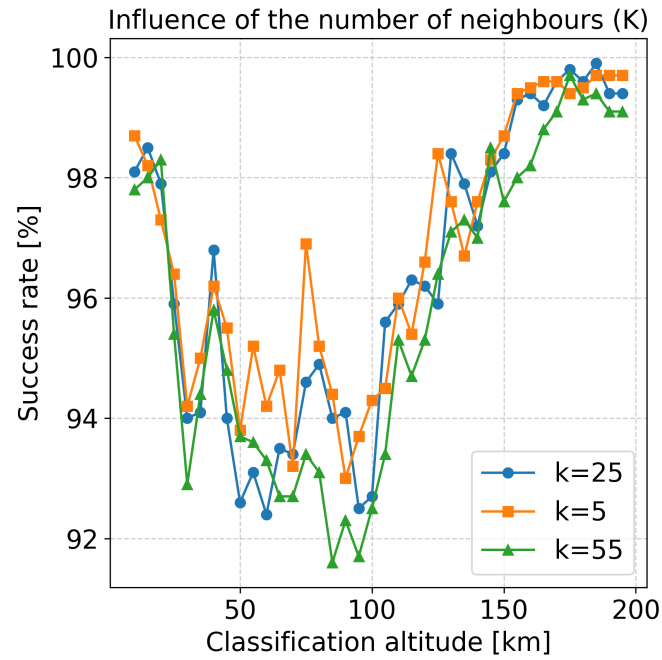


Figure 4.18: Influence of the number of neighbours K on classification performance.

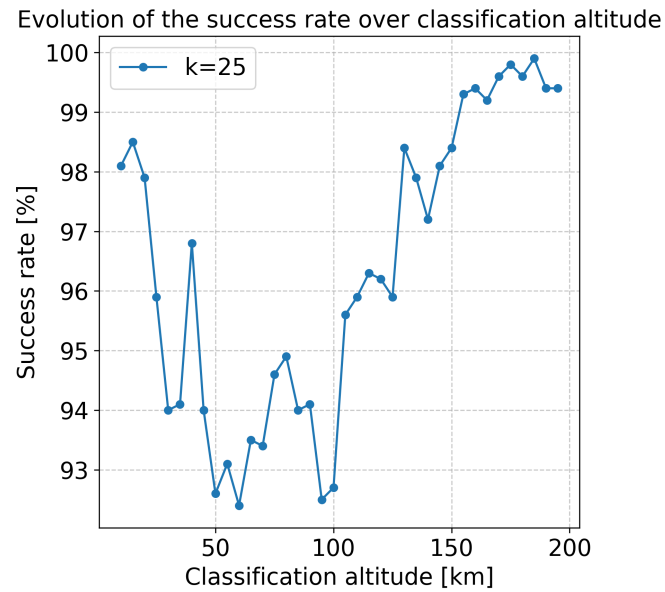


Figure 4.19: Classification success rate as a function of classification altitude for $K = 25$.

Effect of the training dataset size

The reliability of the classification also depends on the volume of reference data used to generate the feature space. Figure 4.20 compares the performance obtained using 30 % and 60 % of the database for training. A slight improvement in accuracy is observed at several altitude levels when the larger dataset is used, suggesting that further increases in this percentage could be beneficial. However, the training dataset size is limited to 60 % to avoid over-fitting. All results presented in the following sections correspond to this selected value, whereas for the final applicability of the implemented methodology, the entire database is used for training.

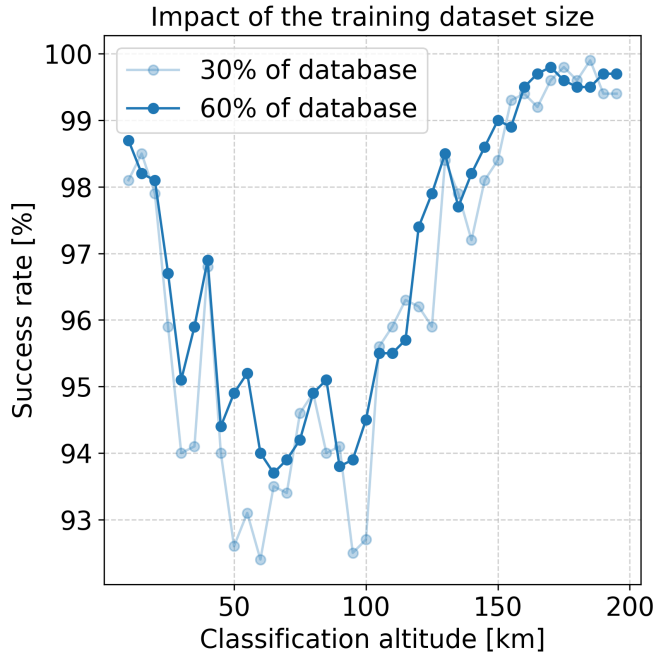


Figure 4.20: Influence of the training dataset size on classification performance.

Impact of adding acceleration magnitude to the classification variables

In order to improve the success rate at altitudes where accuracy is lower, the failure cases are analysed, as shown in Figure 4.21. The highest number of failures corresponds to cases in which the algorithm predicts an orbital launch vehicle when the true class is a ballistic missile (red curve), suggesting the inclusion of an additional representative variable that clearly differentiates these two vehicle classes, namely the acceleration magnitude.

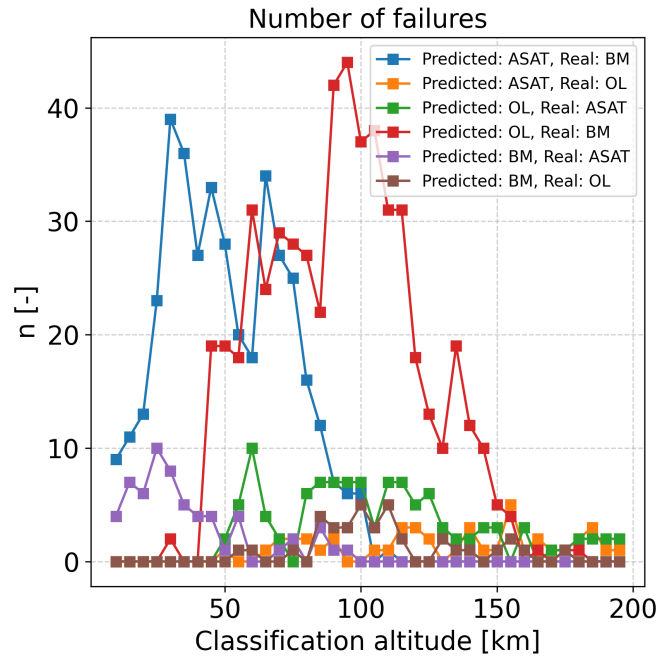


Figure 4.21: Classification failures using a two-dimensional feature space.

The transition to a three-dimensional feature space results in a significant improvement in the success rate across all altitudes, as shown in Figure 4.22, where the classification performance obtained with two-dimensional and three-dimensional feature spaces is compared. This confirms that the acceleration profile contains critical information for differentiating vehicle types with similar velocity and angular characteristics.

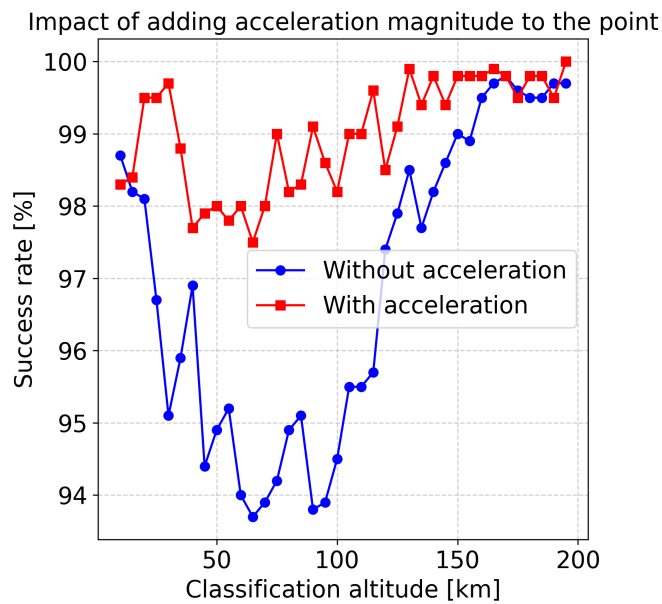


Figure 4.22: Comparison of classification performance between 2D and 3D feature spaces.

Single-point versus history-based classification

The classification methodology described so far operates on a single-point basis, where the vehicle is identified using only the instantaneous measurements at a specific altitude. However, the reliability of the classification can be further enhanced by incorporating the temporal evolution of the trajectory. The motivation for this history-based approach is to take advantage of the well separated clusters observed at early stages to maintain a stable and correct classification when the vehicle passes through regions where the feature space clusters tend to overlap.

To implement this, a cumulative classification strategy is developed. Instead of considering the output of the KNN algorithm as an isolated event, the algorithm accounts for the entire measurement history. This process is summarized in the flow chart illustrated in Figure 4.23.

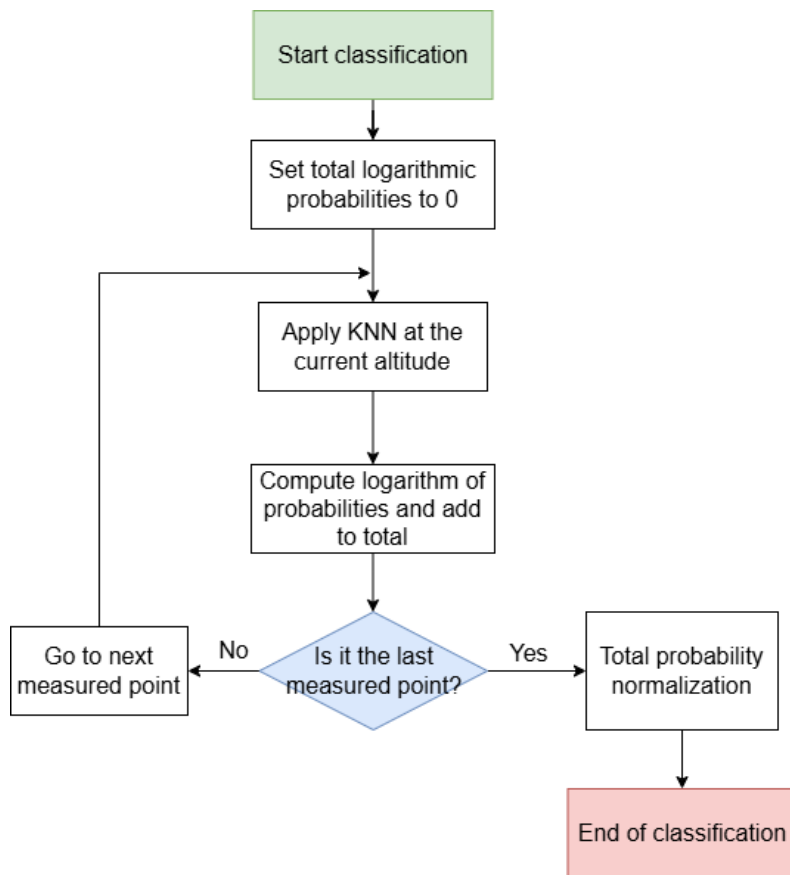


Figure 4.23: Flow chart of the history-based classification logic.

The logic consists of computing a cumulative probability for each class. At each new altitude level, the KNN algorithm provides a set of instantaneous probabilities based on the K nearest neighbours. These are then combined with the previous results to determine

a total probability. Logarithmic probabilities are used to improve numerical stability, since the product of multiple small probabilities can quickly lead to underflow in floating-point arithmetic. By operating in log-space, probability products are transformed into sums, enabling robust and reliable accumulation over long sequences.

Figure 4.24 compares the success rate between the single-point and history-based approaches. In this case, the classification altitude refers to the altitude at which measurements stop being received, i.e., the last measured altitude. As expected, incorporating the trajectory history significantly mitigates the loss of accuracy at intermediate altitudes, even achieving a 100 % success rate at several classification altitudes, thus providing a more robust and consistent identification throughout the boost phase.

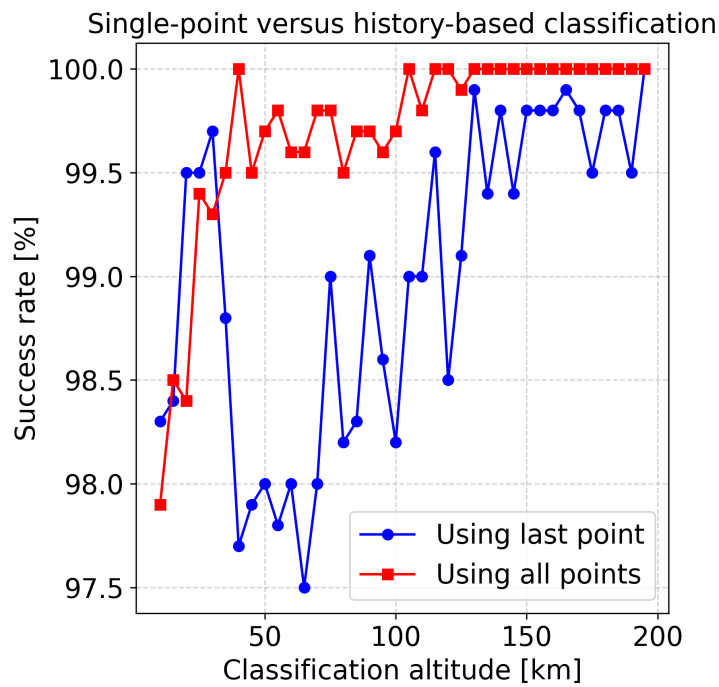


Figure 4.24: Comparison of classification success rate using single-point and history-based approaches.

To further analyse the performance of the history-based classification approach, the failure cases are examined and illustrated in Figure 4.25. The results indicate a clear distinction between orbital launch vehicles and the other classes, with the failures occurring exclusively in the differentiation between ballistic missiles (BM) and anti-satellite (ASAT) systems. This behaviour is consistent with the physical nature of these trajectories, as both vehicle types may follow similar suborbital profiles during certain stages of the boost phase.

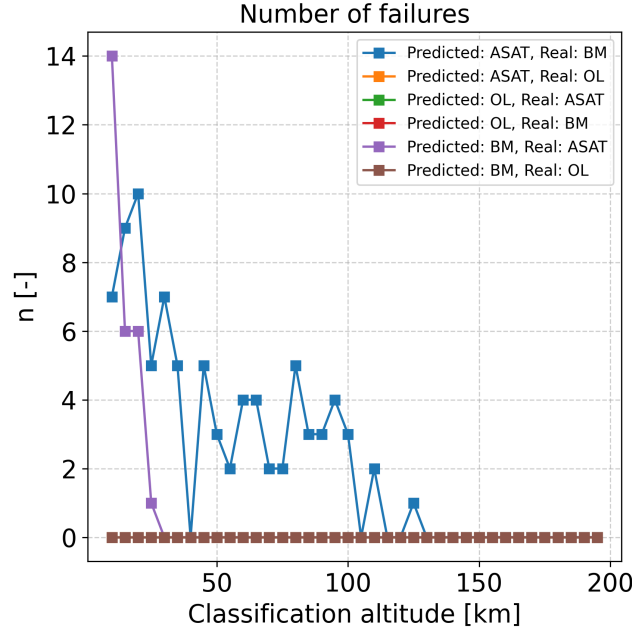


Figure 4.25: Classification failures using a three-dimensional feature space with history-based approach.

In summary, the systematic analysis and optimisation performed in this section lead to the definition of the final classification configuration. The implemented methodology is based on a three-dimensional feature space, integrating velocity magnitude, flight-path angle, and acceleration magnitude as classification variables. To ensure maximum robustness, the complete simulated database should be used as the training reference for the identification of external trajectories. Finally, the single-point classification is replaced by a history-based approach.

This optimised configuration provides the baseline for the subsequent analysis in a more realistic operational environment.

4.3. Realistic data acquisition and sensor modelling

The analyses presented in the previous sections assume ideal conditions with direct access to the true states of the trajectories. However, such an assumption does not reflect an operational environment, where classification must rely on noisy and potentially incomplete observations provided by remote sensing systems. This section introduces a simplified sensor network and measurement model to simulate the data acquisition process. Furthermore, it describes the state reconstruction methodology required to derive the classification variables and evaluates the impact of measurement noise on the performance of the algorithm.

4.3.1. Sensor characterisation and network configuration

The hot exhaust plumes generated during the launch of space and ballistic vehicles are powerful sources of infrared (IR) radiation. This emission can be observed from space, making space-based IR sensors a well-established and effective solution for the early detection and tracking of launch events during the boost phase. When deployed in high-altitude orbits, such as geostationary orbit (GEO) or Highly Elliptical Orbit (HEO), IR sensors are able to provide wide-area coverage of the Earth, which is particularly advantageous for early warning and surveillance applications [48].

Early confirmation of a launch and estimation of the initial flight direction are of critical importance, as discussed in previous sections. In practice, the capability of IR-based detection during the boost phase is primarily limited by atmospheric effects and observation constraints, including cloud coverage, absorption of infrared radiation by water vapour in specific spectral bands, and the temporal resolution associated with the scanning of the Earth's disk. Among these factors, cloud cover is generally considered the dominant limitation. Dense cloud layers are sufficiently rare above approximately 7 km of altitude, which allows for reliable detection of large rocket exhaust plumes once the vehicle has ascended above this region [48].

The infrared intensity of large launch vehicles is known to vary significantly with altitude due to changes in plume-atmosphere interaction. At low altitudes, strong after-burning processes lead to high IR intensities, whereas at intermediate altitudes a reduction in signal intensity may occur as mixing with atmospheric oxygen becomes less effective. At higher altitudes, the decreasing oxygen concentration and plume expansion further reduce the observed signal, eventually leaving only the intrinsic infrared emission of the exhaust products expanding into near-vacuum conditions. Additional variations in intensity are introduced by staging events and changes in engine thrust [39]. Although these phenomena play a key role in realistic infrared signature modelling, a detailed treatment of plume radiation is beyond the scope of this work.

Instead, the objective here is to incorporate only the primary limitations relevant to realistic data acquisition within the context of trajectory classification. Consequently, it is assumed that the space-based IR sensors modelled in this study are capable of detecting and continuously tracking a launching vehicle once it has reached an altitude above the main cloud layer, taken here to be approximately 10 km, until the end of the boost phase.

These IR sensors are assumed to be deployed in GEO, which provides continuous visibility over larger portions of the Earth's surface compared to LEO, as illustrated in Figure 4.26.

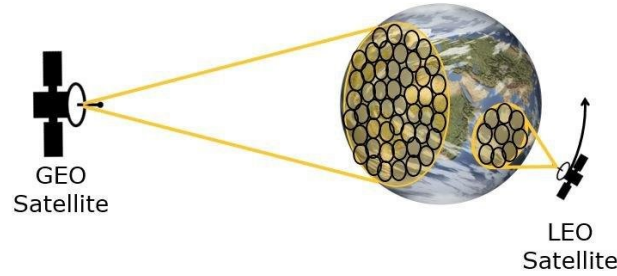


Figure 4.26: Coverage of Geostationary Earth Orbit (GEO) and Low Earth Orbit (LEO) satellites. (Source: [49])

Each satellite is nadir-pointing, with its primary IR sensor aligned along the vertical axis (z-axis) in the satellite body frame, as shown in Figure 4.27.

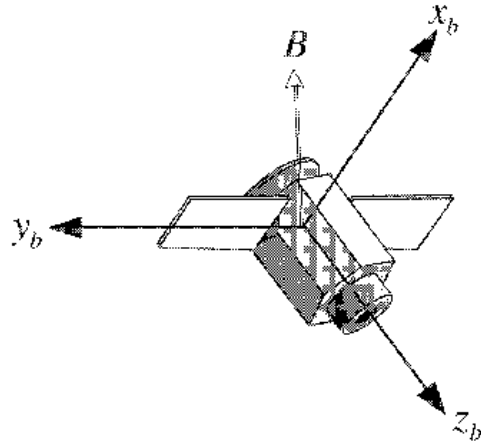


Figure 4.27: Satellite body coordinate axes. (Source: [50])

The sensing region of each instrument is modelled as a conical field of view (FOV), defined by a half-cone angle α of 30° , which characterises the maximum angular deviation from the bore-sight within which targets can be detected (see Figure 4.28).

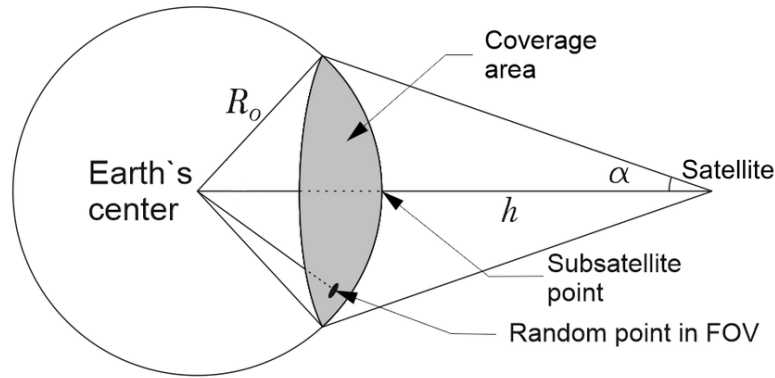


Figure 4.28: Geometric representation of the satellite field of view (FOV). (Source: [51])

Since these sensors provide only line-of-sight (LOS) measurements, which are unit vectors represented by two angles, azimuth and elevation of the target, at least two independent observations are required to estimate the target position. Consequently, in order to ensure that any point overflying the Earth's surface is observed by at least two sensors at all times, a constellation of six GEO satellites is considered, uniformly distributed in longitude. Figure 4.29 illustrates the footprint of a GEO satellite, highlighting the achieved coverage in latitude and longitude, and demonstrating that the selected constellation provides sufficient overlap to guarantee redundant observations.

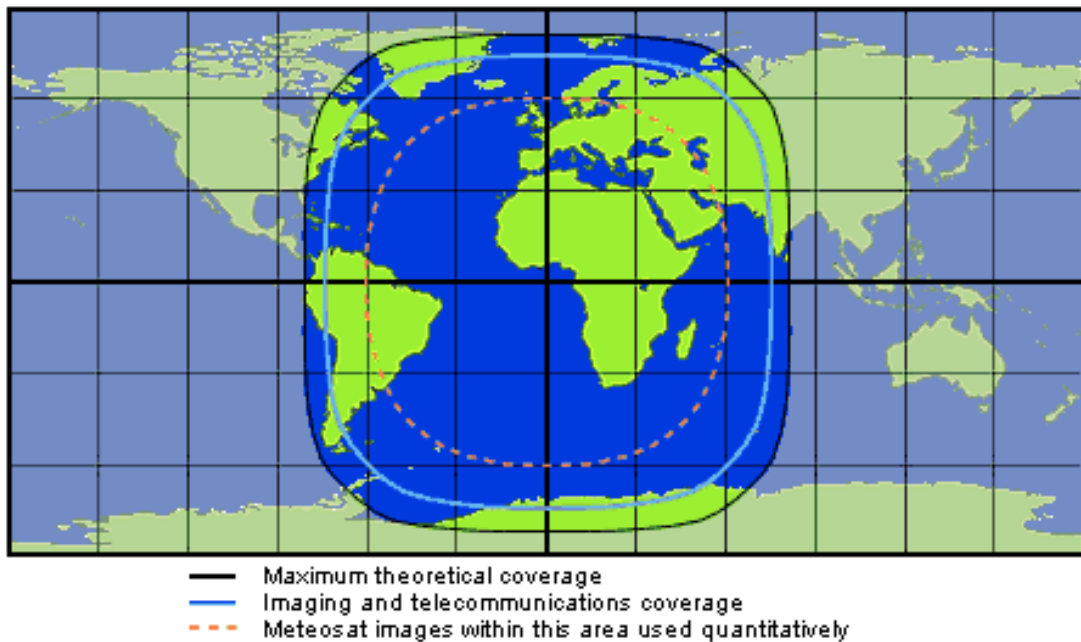


Figure 4.29: Footprint of a geostationary satellite over the Earth. (Source: [52])

4.3.2. Measurement noise modelling and state estimation

Once the sensor type and observation geometry have been defined, this section focuses on modelling the measurement process and introducing realistic noise effects.

Under operational conditions, IR sensors provide noisy line-of-sight (LOS) observations that must be processed in order to estimate the vehicle position. The procedure implemented to generate a noisy measured state from a simulated trajectory is summarised as follows:

1. From the true trajectory position, line-of-sight angles are computed for each observing satellite.
2. Measurement noise is then added to these angular observations.
3. A least-squares (LS) estimation process is applied to obtain a noisy estimate of the target position.

Given the relative position vector between the satellite and the target in the satellite body frame, $\mathbf{r}_{rel}$, the corresponding azimuth Az and elevation δ angles are computed as:

$$Az = \text{atan2}(r_{rel,y}, r_{rel,x}) \quad (4.1)$$

$$\delta = \text{atan2}\left(r_{rel,z}, \sqrt{r_{rel,x}^2 + r_{rel,y}^2}\right) \quad (4.2)$$

A visibility check is performed to ensure that target lies within the sensor's FOV and is not occluded by the Earth. If either condition is not met, the target is considered not visible from the satellite.

Measurement noise is modelled as zero-mean Gaussian noise applied independently to each angular component. For each sensor, the measured azimuth and elevation angles are obtained by adding a normally distributed random perturbation with standard deviation σ , representative of the angular accuracy of the sensor. Typical values for σ range from 15 to 60 μrad [53, 54].

$$\begin{aligned} \hat{Az} &= Az + N(0, \sigma^2) \\ \hat{\delta} &= \delta + N(0, \sigma^2) \end{aligned} \quad (4.3)$$

This process is summarised in the flow chart presented in Figure 4.30, illustrating the simulation of measurements for an on-board IR sensor.

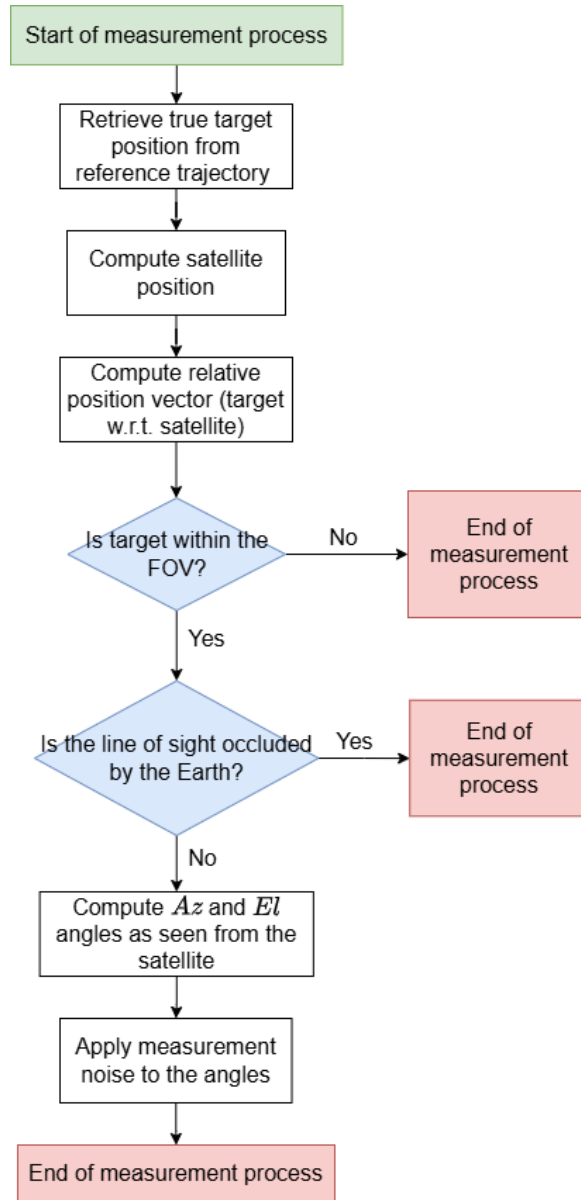


Figure 4.30: Conceptual scheme of the simulation of measurements for an on-board IR sensor.

Once noisy LOS measurements from multiple satellites are available, the corresponding observation lines are constructed. Each noisy angular measurement from satellite i defines a line of sight originating at the satellite position $\mathbf{s}_i$ and orientated along the unit direction vector $\mathbf{u}_{i,b}$, obtained from the measured azimuth and elevation angles in the satellite body reference frame.

$$\mathbf{u}_{i,b} = \begin{bmatrix} \cos \hat{E}l_i \cos \hat{A}z_i \\ \cos \hat{E}l_i \sin \hat{A}z_i \\ \sin \hat{E}l_i \end{bmatrix} \quad (4.4)$$

This direction vector is then transformed to the Earth-centred inertial frame, yielding $\mathbf{u}_i$. Due to measurement noise, the lines of sight do not intersect at a single point. Therefore, the target position is estimated by solving a least-squares problem that minimises the sum of the squared perpendicular distances between the estimated position and all observation lines.

$$\mathbf{r}^* = \arg \min_{\mathbf{r}} \sum_{i=1}^N \left\| (\mathbf{I} - \mathbf{u}_i \mathbf{u}_i^\top) (\mathbf{r} - \mathbf{s}_i) \right\|^2 \quad (4.5)$$

where $(\mathbf{I} - \mathbf{u}_i \mathbf{u}_i^\top)$ denotes the projection matrix onto the plane orthogonal to the i -th line of sight.

This least-squares problem admits a closed-form solution, as commonly derived in multiple view triangulation problems [55]:

$$\left(\sum_{i=1}^N (\mathbf{I} - \mathbf{u}_i \mathbf{u}_i^\top) \right) \mathbf{r}^* = \sum_{i=1}^N (\mathbf{I} - \mathbf{u}_i \mathbf{u}_i^\top) \mathbf{s}_i \quad (4.6)$$

which is solved at each epoch to obtain the reconstructed target position.

By repeating this process over time, a noisy reconstructed trajectory is obtained. From the estimated position history, velocity and acceleration are derived using numerical differentiation, and the flight-path angle is computed from the velocity components. It should be noted that numerical differentiation inherently amplifies measurement noise, particularly in the acceleration estimates. This noise amplification has a direct impact on the performance of the classification algorithm.

4.3.3. Impact of measurement noise on classification performance

The influence of sensor measurement noise on the performance of the trajectory classification algorithm is evaluated by simulating noisy measurements according to the procedure described in the previous section.

In all cases, 60 % of the generated trajectory database is used as the training dataset, while 1000 randomly selected trajectories from the remaining set are classified using sim-

ulated noisy measurements. The classifier configuration considered is the history-based approach in a three-dimensional feature space, using velocity magnitude, flight-path angle, and acceleration magnitude as classification variables. Regarding the measurements, the sensors provide an estimated position every 0.1 seconds, while the classification algorithm applies the KNN classification every second (i.e., every ten measurements, KNN is applied computing probabilities based on the feature space distribution, which are then added to the cumulative logarithmic probabilities).

Figure 4.31 presents the classification success rates as a function of the classification altitude for different levels of angular measurement noise, σ . As expected, higher noise levels generally reduces classification accuracy, particularly at lower altitudes, where fewer measurements are available and numerical differentiation amplifies the noise. Nevertheless, the algorithm achieves good performance once sufficiently high altitude is reached, demonstrating robustness to moderate levels of measurement noise.

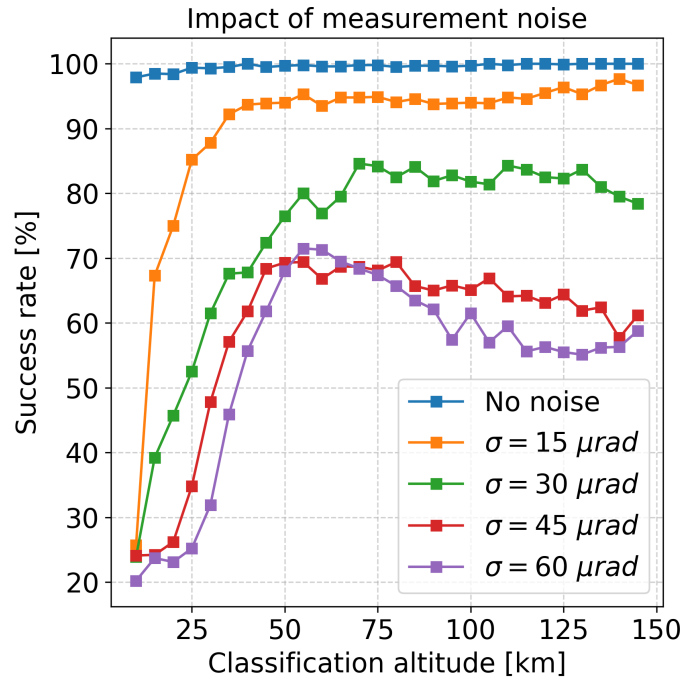
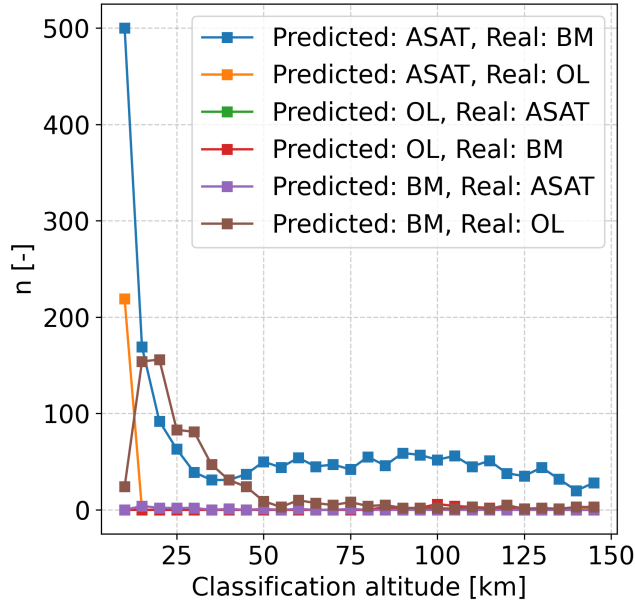


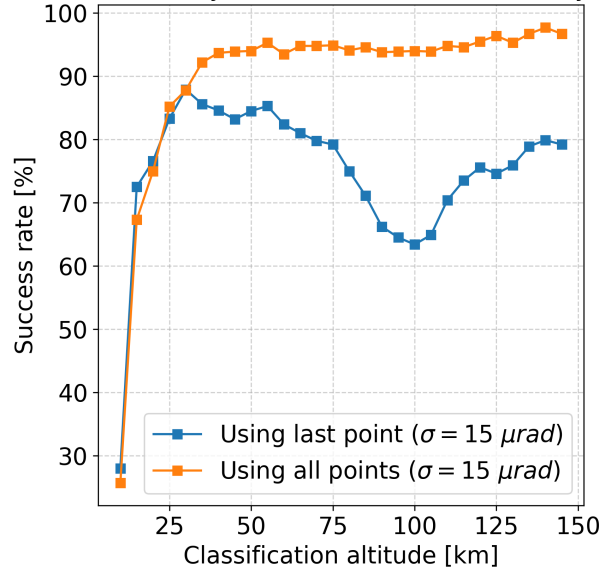
Figure 4.31: Evolution of classification success rate over altitude for different angular measurement noise levels (σ).

To further investigate the origin of the classification errors, Figure 4.32 shows the failure cases for $\sigma = 15 \mu\text{rad}$. Consistent with previous observations, most failures occur between ballistic missiles (BM) and anti-satellite (ASAT) systems, reflecting the inherent similarity of their suborbital profiles during certain stages of the boost phase.

Number of failures considering noisy measurements ($\sigma = 15 \mu\text{rad}$)Figure 4.32: Classification failures for $\sigma = 15 \mu\text{rad}$.

Finally, Figure 4.33 compares the performance of the history-based approach with the single-point classification method at a measurement noise level of $\sigma = 15 \mu\text{rad}$. Despite the presence of noise, the history-based classifier clearly outperforms the single-point method, demonstrating the robustness and improved reliability provided by considering the trajectory evolution.

Single-point versus history-based classification with noisy measurements

Figure 4.33: Comparison of classification success rate between single-point and history-based approaches for $\sigma = 15 \mu\text{rad}$.

5 | Mission analysis

This chapter presents a mission analysis focused on the evaluation of Earth coverage performance achieved by different satellite constellation configurations dedicated to early warning and launch detection. The study is motivated by the need to ensure the timely detection and redundant observation of direct-ascent launches, as introduced in the previous chapters. The results obtained for each satellite constellation configuration are evaluated using **Overflight**, a software tool developed by GMV.

5.1. Existing space-based early-warning systems

Space-based early-warning and missile-tracking architectures are a central element of national defence and space security, providing persistent surveillance of launch events and ballistic missile activity. A prominent example is the United States Space Based Infrared System (SBIRS), which comprises a constellation of satellites that includes six geostationary orbit (GEO) platforms and four Highly Elliptical Orbit (HEO) payloads, designed to detect and track missile launches and other infrared signatures around the world, including over high-latitude regions [56].

The Next Generation Overhead Persistent Infrared (Next Gen OPIR) system, which is expected to be a key part of the Golden Dome weapons system, is scheduled to begin launches in 2026. It will consist of three geosynchronous satellites and two polar orbiting missile warning systems, serving as the replacement for its predecessor, the SBIRS [57].

Open-source orbital data also indicate that China operates a combination of GEO and HEO satellites providing persistent northern high-latitude coverage. In particular, four satellites are placed in highly elliptical orbits synchronised to achieve continuous visibility over the northern hemisphere, alongside at least three GEO missile-warning satellites. This configuration enables near-continuous monitoring of northern high latitudes, consistent with regional strategic prioritisation [58].

Although detailed constellation parameters and sensor characteristics remain classified, these examples illustrate the practical use of mixed GEO/HEO architectures for infrared-

based launch detection and tracking, and serve as reference points for the constellation analyses presented in the subsequent sections.

5.2. Mission analysis methodology

This section details the methodology adopted to evaluate the coverage performance of a specific satellite constellation configuration, whose results are assessed using the **Overflight** software tool. The objective is to assess the capability of each constellation to provide continuous and redundant observation of potential launch locations over the Earth's surface.

5.2.1. GMV's **Overflight** software overview

Overflight is designed to support mission analysis, surveillance studies, and space domain awareness applications by enabling the evaluation of satellite observation events over predefined regions of the Earth.

The core functionality of **Overflight** is based on the prediction of satellite visibility events over ground locations or extended areas, combining orbital propagation with sensor geometry and line-of-sight constraints. Given a satellite orbit and a target region on Earth, the software determines whether the satellite sensor maintains an unobstructed line of sight to the target, accounting for Earth occlusion and sensor field of view (FOV) limitations. The software allows for the definition of both observation targets and satellite configurations.

Observation targets can be specified either as discrete ground points or as extended polygonal regions, defined by their geodetic latitude, longitude, and altitude. These inputs establish the point position in Earth-Centred Earth-Fixed (ECEF) coordinates for subsequent geometric computations.

Satellites are defined through user-provided orbital elements in the form of Two-Line Elements (TLEs). Orbits are propagated over a selected time window using an SGP4 propagator, which provides sufficient accuracy for coverage and visibility analyses while accounting for the dominant perturbation effects. For each satellite–target pair, **Overflight** evaluates visibility conditions over time, allowing the identification of observation windows and simultaneous coverage by multiple satellites.

Each satellite is assumed to carry a user-defined sensor, characterised primarily by its field of view and orientation relative to the satellite body. In this study, sensors are modelled using a conical FOV, defined by a half-angle that determines the observable region on the

Earth's surface. Sensor orientation is defined through the satellite attitude mode, with nadir-pointing configurations being of particular relevance for early-warning and Earth observation missions.

Although **Overflight** supports different sensor categories (optical, infrared, and radar) and additional filtering options related to illumination and weather conditions, the present analysis focuses on infrared capabilities. Furthermore, visibility assessments are driven primarily by geometric constraints, without explicitly modelling spectral or radiometric sensor performance.

The determination of overflight events is performed by identifying time intervals during which the target remains within the sensor FOV. This is achieved by evaluating a visibility function whose sign indicates whether the observation conditions are satisfied. The resulting event start and end times define the satellite overflight windows. The software outputs include time-tagged visibility events, sensor pointing angles, and coverage statistics, which are subsequently post-processed to assess spatial and temporal coverage performance.

By enabling systematic evaluation of satellite visibility over large regions and extended time intervals, **Overflight** provides an effective framework for the analysis and comparison of early-warning constellation architectures.

5.2.2. Evaluation procedure and coverage metrics

A systematic procedure is applied to each satellite constellation configuration analysed in this chapter to assess its Earth surface coverage and observation redundancy. This methodology is designed to quantify the capability of a given constellation to provide simultaneous observations of potential launch locations, which is a key requirement for reliable detection and trajectory identification.

The process begins by discretising the Earth's surface into a grid of 800 points, uniformly distributed in longitude and latitude. Specifically, 40 points are defined along the longitude, spanning the range from -180° to 180° , and 20 points along the latitude, covering the interval from -90° to 90° . Each point represents a potential launch location to be monitored by the satellite constellation.

Once the observation grid is established, the specific satellite constellation under analysis is defined. For each configuration, the number of satellites, their orbital parameters, and their relative distribution are specified according to the scenario being evaluated. All satellites are assumed to be equipped with identical infrared sensors, modelled with a

nadir-pointing orientation and a conical field of view characterised by a half-cone angle of 30° .

The constellation and surface grid are then processed using the **Overflight** software. For a selected time window, the software computes visibility conditions between each satellite and each surface point, determining whether a given point lies within the sensor field of view and is not occluded by the Earth. The resulting outputs provide time-tagged visibility events for all satellite–point pairs.

An illustrative example of an overflight event, showing the coordinates of the monitored point, the ID of the satellite overflying it, and the corresponding visibility time window, is presented in Figure 5.1.

```
<record index="75">
  <OVF_ZONE>z221</OVF_ZONE>
  <OVF_COORDS lenght="1">
    <point index="1">
      <COORD_LON> 0.000000</COORD_LON>
      <COORD_LAT> -0.703368</COORD_LAT>
    </point>
  </OVF_COORDS>
  <OVF_SAT>GEO1</OVF_SAT>
  <OVF_COUNTRY>SPAIN</OVF_COUNTRY>
  <OVF_TYPE>IR</OVF_TYPE>
  <OVF_START>2026/01/05-00:00:00.000</OVF_START>
  <OVF_END>2026/01/10-00:00:00.000</OVF_END>
  <OVF_DS stat="NOTFOUND">Not found in ananalysis</OVF_DS>
</record>
```

Figure 5.1: Illustrative example of event output provided by **Overflight** software.

Finally, the coverage performance of each constellation is evaluated by post-processing these visibility events. At each time step, the number of surface points simultaneously observed by at least two satellites is determined. The primary performance metric is the percentage of the total grid points covered by two or more satellites as a function of time. This metric directly reflects the capacity of the constellation to provide redundant observations, as at least two independent measurements are required to estimate the target position through a least-squares process, while additional simultaneous observations further improve estimation accuracy and robustness.

5.3. Constellation configurations and results

Using the methodology described in Section 5.2.2, several satellite constellation configurations are analysed. This section presents the architectures considered and discusses the results obtained.

5.3.1. Four Molniya in different orbital planes

The first analysed configuration consists of four satellites in Molniya orbits distributed across different orbital planes. The right ascension of the ascending node (RAAN) and the argument of perigee (ω) of the satellites are offset by approximately 180° in pairs, ensuring complementary coverage between opposing orbital planes in both hemispheres. The satellites are synchronised in phase, reaching their apogees at nearly the same time. A representation of this configuration is illustrated in Figure 5.2.

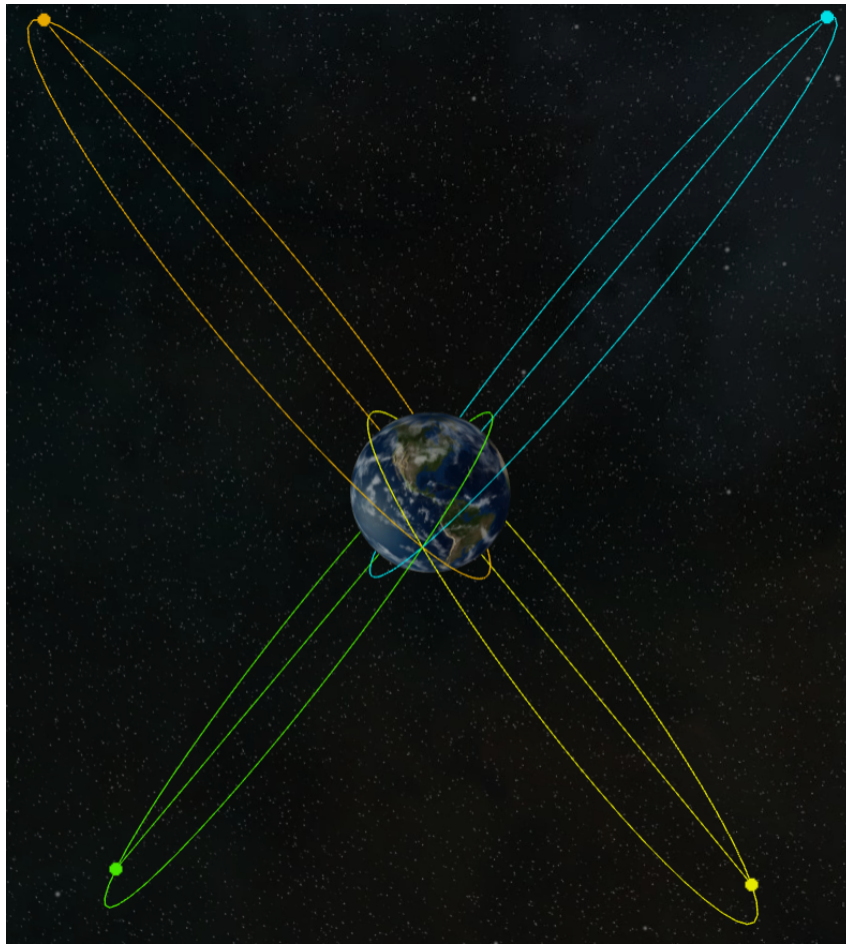
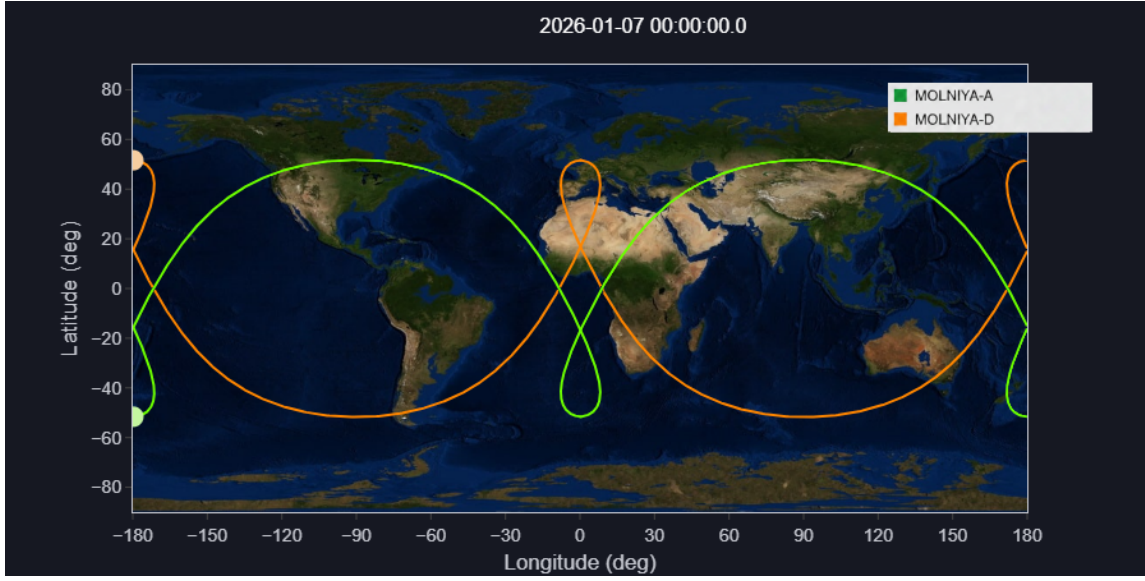
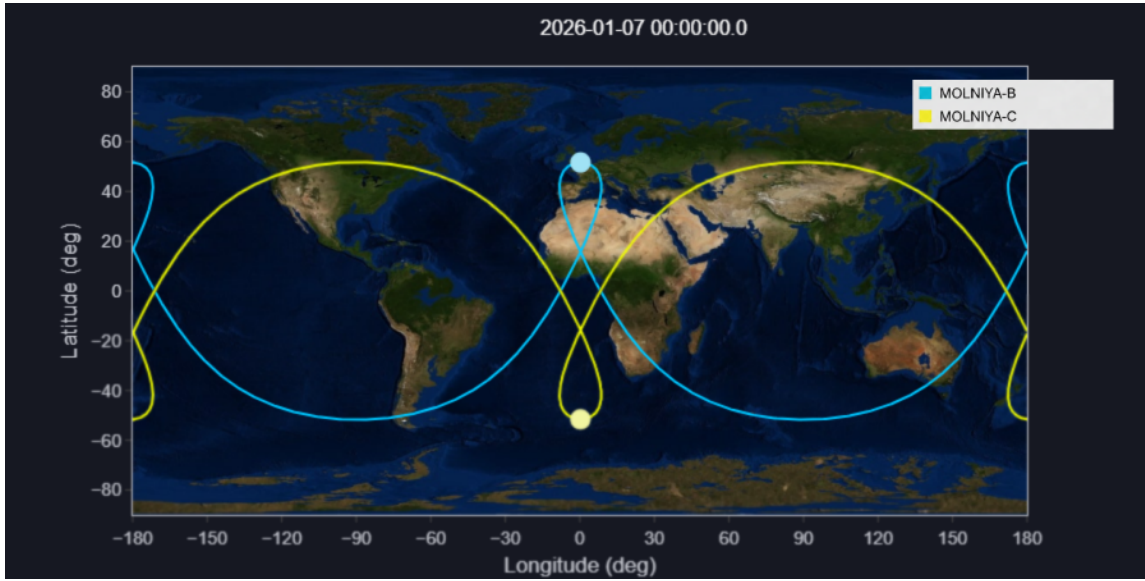


Figure 5.2: Orbital configuration of the four-Molniya constellation.

Figure 5.3 presents the satellite ground tracks, which help to visualise the spatial distribution and complementarity of the orbits within the four-Molniya constellation.



(a) *Molniya-A (MOLA)* and *Molniya-D (MOLD)*.



(b) *Molniya-B (MOLB)* and *Molniya-C (MOLC)*.

Figure 5.3: Satellite ground tracks for the four-Molniya constellation.

To evaluate the observation results of this configuration, Figure 5.4 presents the visibility windows of the four satellites over a representative surface point located at longitude $\lambda = 0^\circ$ and latitude $\phi = 40.3^\circ$. The individual observation intervals for each satellite are displayed together with a red line indicator that identifies the time instants at which the point is simultaneously observed by more than one satellite.

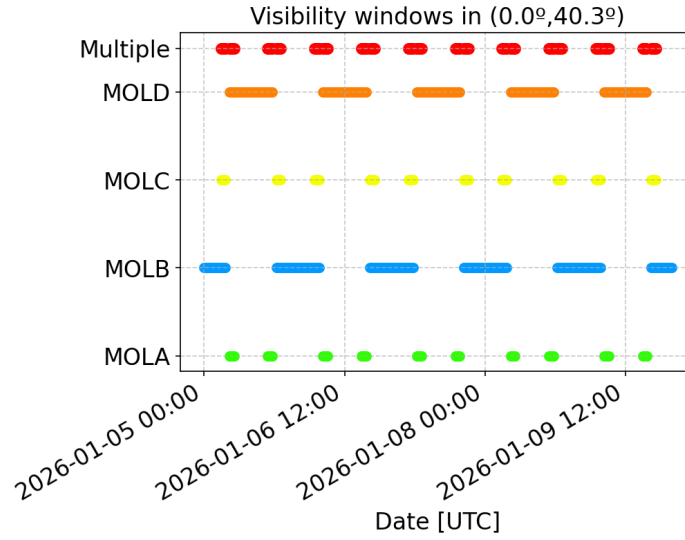


Figure 5.4: Visibility windows over the surface point at $\lambda = 0^\circ$ and $\phi = 40.3^\circ$ for the four-Molniya constellation.

The visibility windows exhibit a clear periodic pattern that repeats daily. This behaviour is a direct consequence of the Molniya orbital characteristics and the fixed ground-track repetition. As a result, the coverage performance over a 24-hour time window is representative of the behaviour observed on subsequent days (see Figure 5.5).

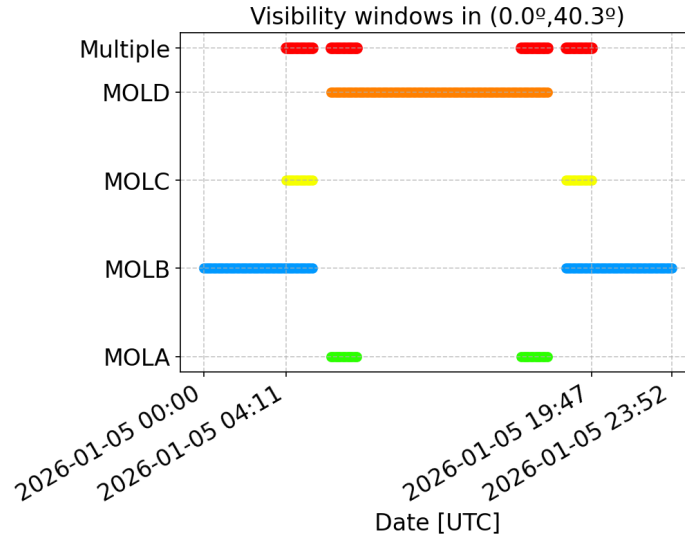


Figure 5.5: Visibility windows over the surface point at $\lambda = 0^\circ$ and $\phi = 40.3^\circ$ during one day for the four-Molniya constellation.

The global effectiveness of the constellation is assessed by evaluating the percentage of discretised Earth surface points simultaneously observed by at least two satellites as a function of time, which provides an estimate of the fraction of the Earth's surface covered

by multiple sensors at any given time. Figure 5.6 shows the resulting time history of the multi-satellite coverage percentage. The coverage profile exhibits a repeating pattern with a period of 12 hours, consistent with the orbital period of Molniya orbits.

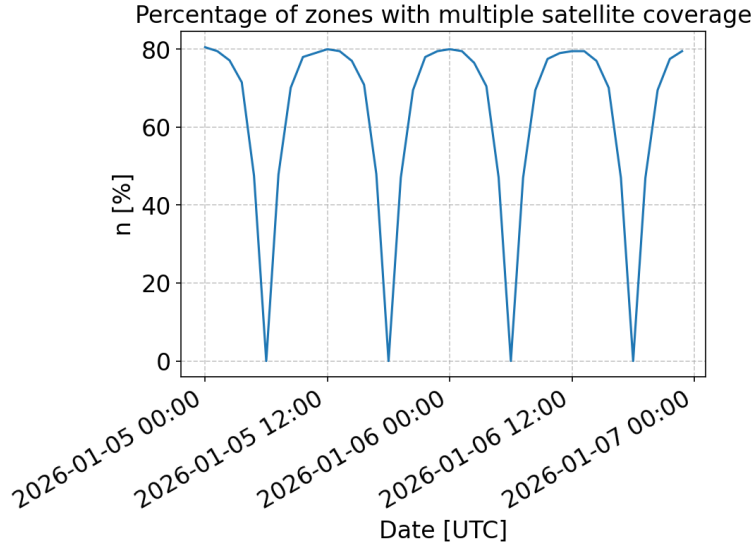


Figure 5.6: Time evolution of the percentage of Earth's surface simultaneously covered by multiple satellites for the four-Molniya constellation.

For this configuration, peak coverage values of up to approximately 80% are achieved during phases in which the satellites are near the apogee. Conversely, the coverage drops to absolute zero approximately six hours later, i.e., when all satellites are near the perigee. At this stage, the reduced altitude prevents any surface point from being simultaneously within the field of view of two or more sensors.

The coverage performance is further examined in Figure 5.7, which illustrates the geographical distribution of surface points simultaneously observed by at least two satellites in representative epochs that include maximum, intermediate, and minimum coverage conditions. During the first four hours of the cycle, the extent of the redundant covered areas (red points) remains nearly constant. This is due to the low orbital velocity near the apogee, which allows the satellites to maintain a persistent footprint over their respective target regions. In addition, the high altitude at this stage enables a wide coverage area. As the satellites approach perigee, the coverage degrades rapidly. At approximately the six-hour mark, the constellation reaches a null-redundancy condition, in which no point on the Earth's surface is simultaneously observed by more than one sensor. However, due to the high eccentricity and the rapid increase in velocity at perigee, the satellites pass through this region quickly and soon begin their ascent, restoring multi-satellite visibility, in agreement with Figure 5.6.

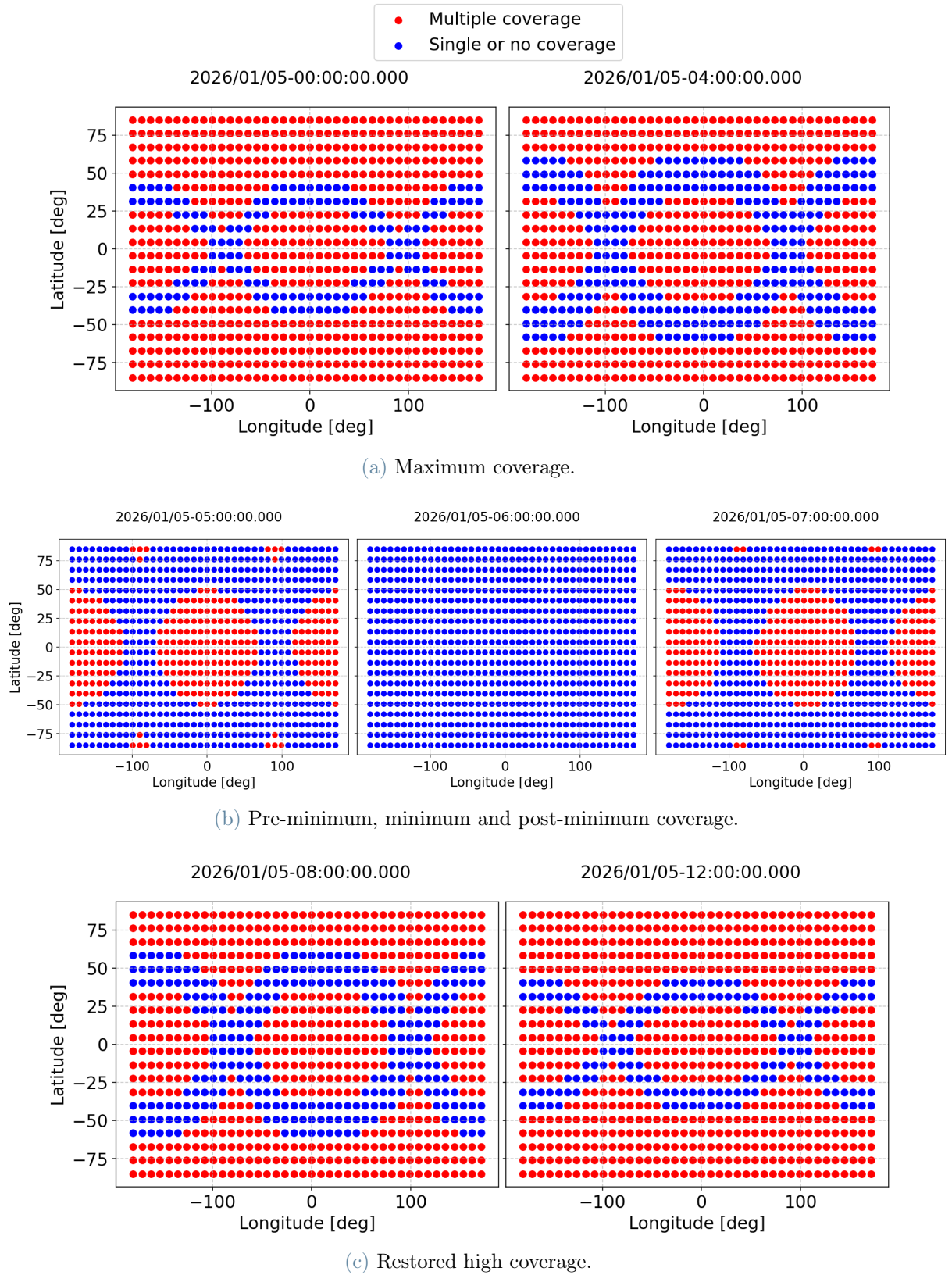


Figure 5.7: Geographical distribution of surface points covered by the four-Molniya constellation.

5.3.2. Four-Molniya and two-GEO constellation

The next analysed constellation consists of four satellites in Molniya orbits and two satellites in GEO uniformly distributed. A representation of this configuration is illustrated in Figure 5.8.

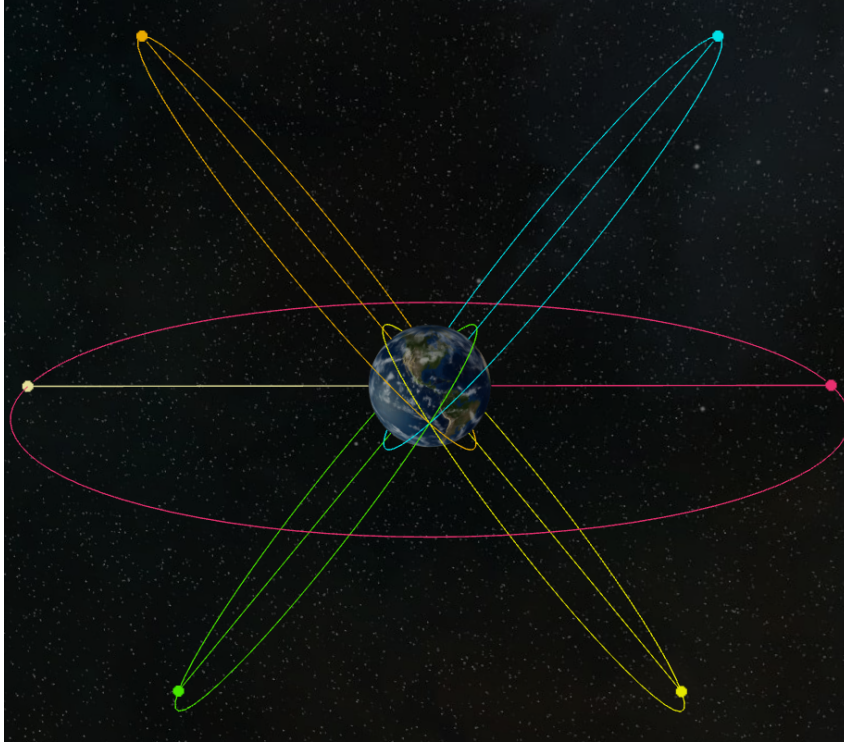


Figure 5.8: Orbital configuration of the four-Molniya and two-GEO constellation.

This modification is introduced to mitigate the coverage gaps observed at low latitudes. The addition of GEO satellites is expected to increase the coverage in these regions, thus improving the overall effectiveness of the constellation.

Figure 5.9 presents the satellite ground tracks, which help to visualise the spatial distribution and complementarity of the orbits within the four-Molniya and two-GEO constellation.

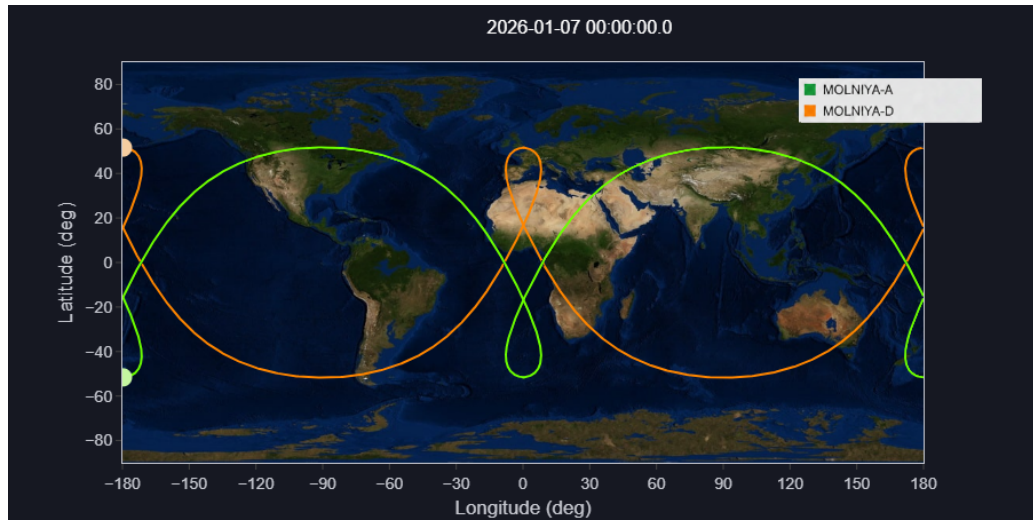
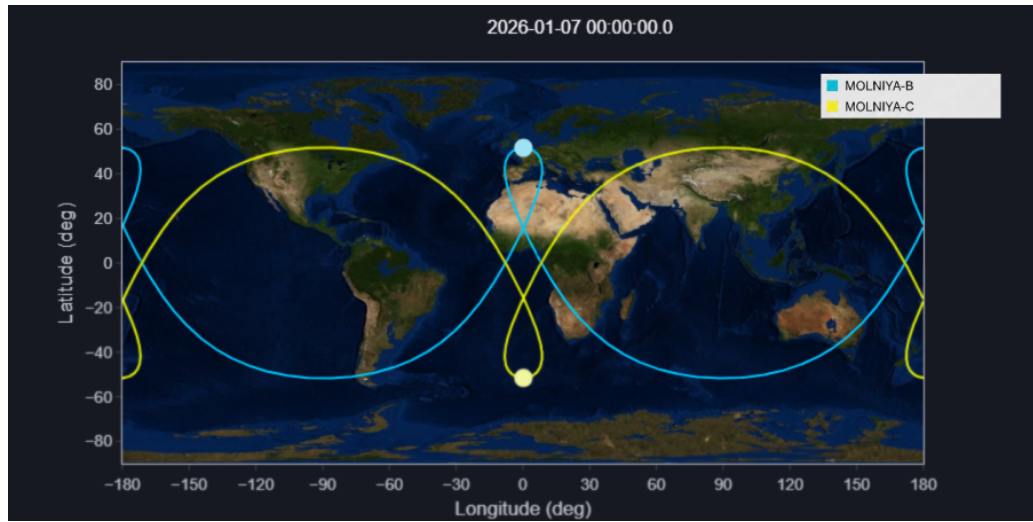
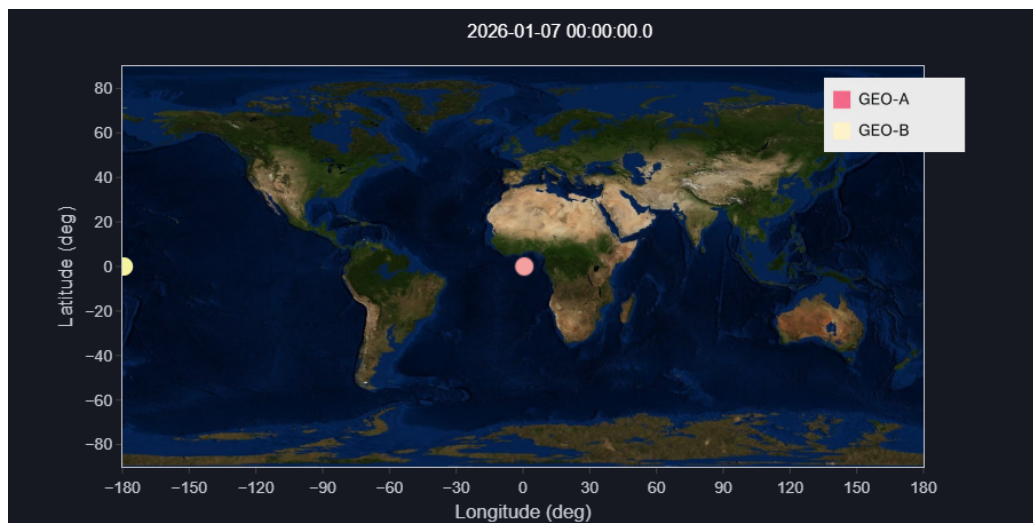
(a) *Molniya-A (MOLA)* and *Molniya-D (MOLD)*.(b) *Molniya-B (MOLB)* and *Molniya-C (MOLC)*.(c) *GEO-A (GEOA)* and *GEO-B (GEOB)*.

Figure 5.9: Satellite ground tracks for the four-Molniya and two-GEO constellation.

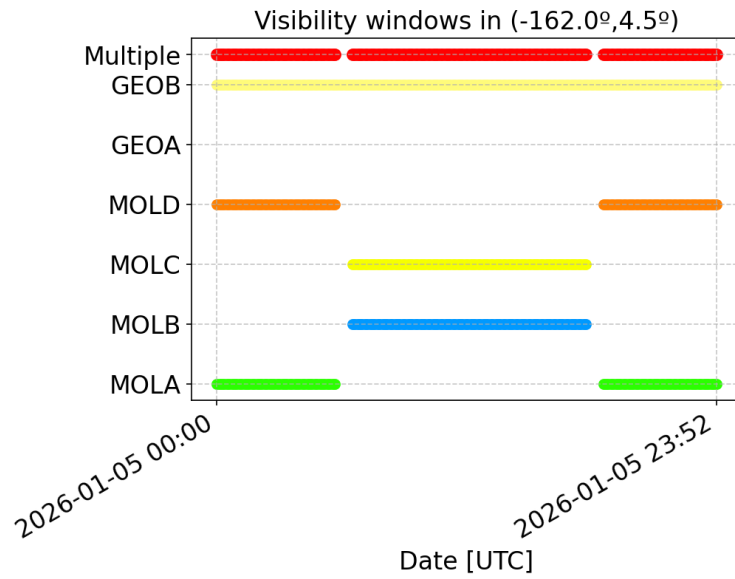


Figure 5.11: Visibility windows over the surface point at $\lambda = -162^\circ$ and $\phi = 4.5^\circ$ during one day for the four-Molniya and two-GEO constellation.

The global performance of the constellation is assessed by evaluating the percentage of discretised Earth surface points simultaneously observed by at least two satellites as a function of time. Figure 5.12 shows the resulting time history of the multi-satellite coverage percentage. The coverage profile again exhibits a repeating pattern with a period of 12 hours, consistent with the orbital period of Molniya orbits.

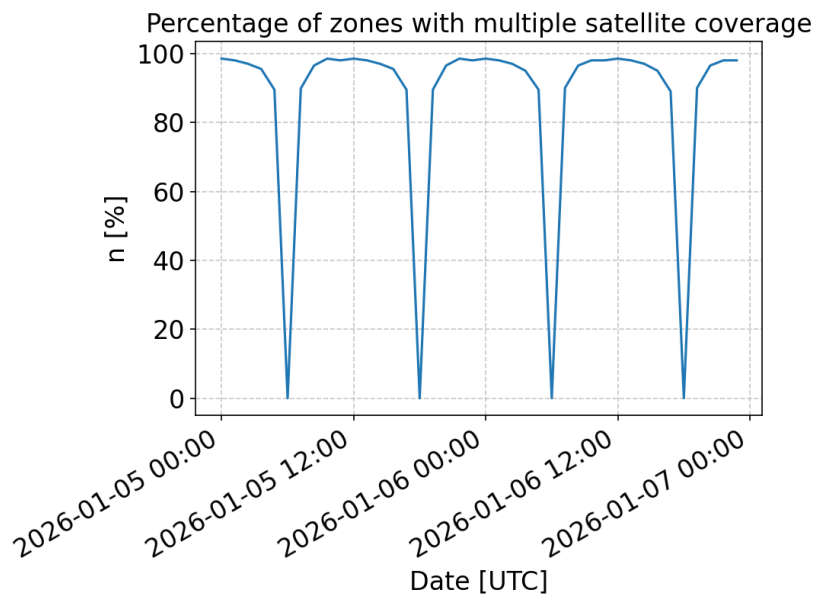


Figure 5.12: Time evolution of the percentage of Earth's surface simultaneously covered by multiple satellites for the four-Molniya and two-GEO constellation.

In this case, peak coverage values of up to approximately 97% are achieved when the Molniya satellites are near the apogee, as the GEO satellites provide persistent coverage at low latitudes and increase redundancy. Conversely, the multi-satellite coverage drops to zero approximately six hours later, i.e., when the Molniya satellites pass through the perigee. As in the previous configuration, there are time instants at which null redundancy is observed, since the regions observed by each Molniya satellite during their passage through the perigee are not simultaneously covered by the GEO satellites, and, therefore, multi-satellite coverage is not achieved during these intervals.

The coverage performance is further examined in Figure 5.13, which illustrates the geographical distribution of surface points simultaneously observed by at least two satellites in representative epochs that include maximum, intermediate, and minimum coverage conditions. Again, during the first four hours of the cycle, the extent of the redundant covered areas (red points), which is larger than in the Molniya-only configuration due to the presence of the GEO satellites, remains nearly constant. Similarly, as the Molniya satellites approach the perigee, the coverage degrades rapidly until the minimum coverage condition is reached, after which it starts to increase again.

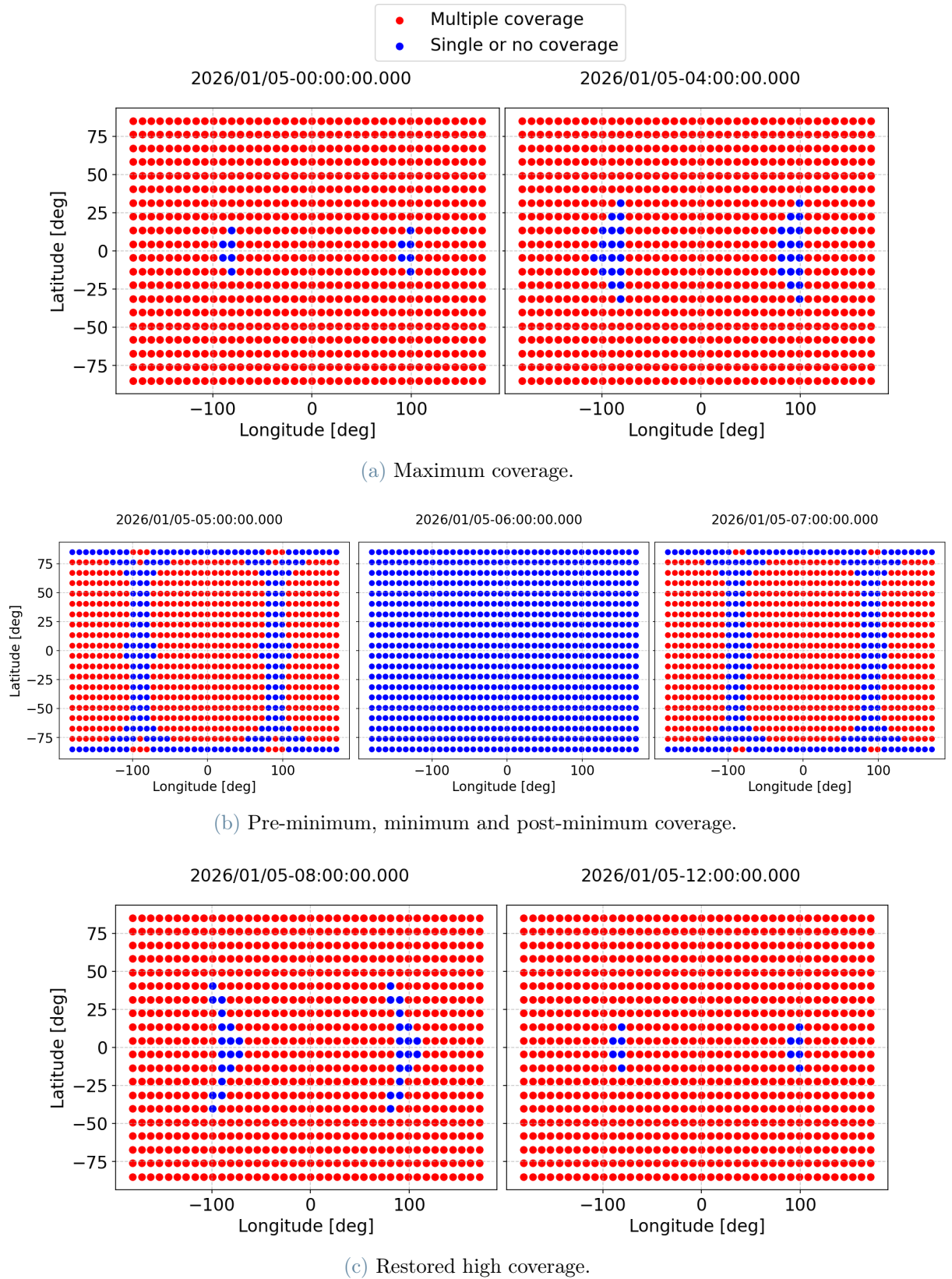


Figure 5.13: Geographical distribution of surface points covered by the four-Molniya and two-GEO constellation.

5.3.3. Two-Molniya and three-GEO constellation

Since improvement in coverage during low-redundancy intervals remains insufficient, it becomes necessary to increase the number of satellites in GEO. The next configuration therefore consists of three uniformly distributed GEO satellites and two Molniya satellites, one covering each hemisphere. This arrangement increases redundancy during minimal coverage periods while maintaining the advantages of Molniya satellites at high latitudes. A representation of this configuration is illustrated in Figure 5.14, while Figure 5.15 presents the satellite ground tracks.

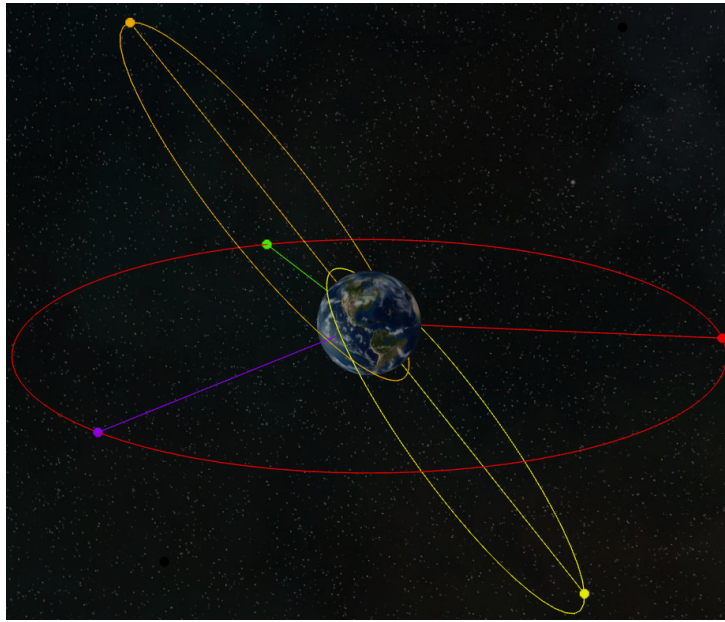


Figure 5.14: Orbital configuration of the two-Molniya and three-GEO constellation.

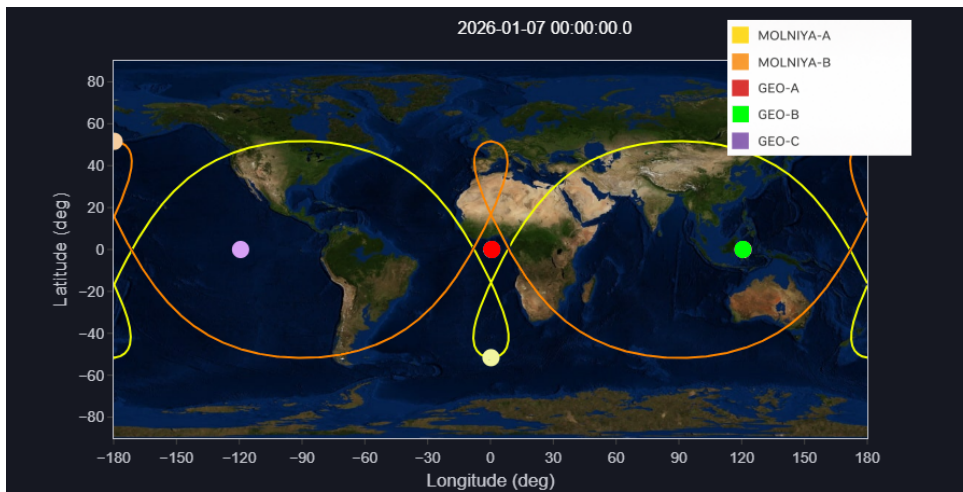


Figure 5.15: Satellite ground tracks for the two-Molniya and three-GEO constellation.

Figure 5.16 presents the visibility windows of each satellite over a representative surface point located at longitude $\lambda = -162^\circ$ and latitude $\phi = 4.5^\circ$, along with the time intervals during which the point is simultaneously observed by multiple satellites.

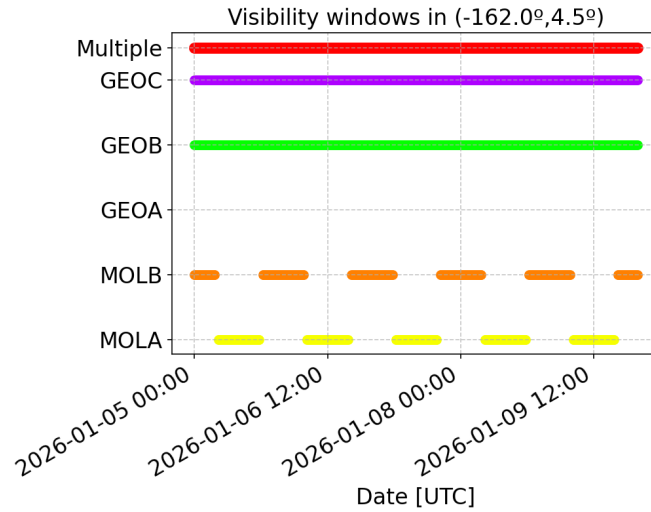


Figure 5.16: Visibility windows over the surface point at $\lambda = -162^\circ$ and $\phi = 4.5^\circ$ for the two-Molniya and three-GEO constellation.

The visibility windows exhibit a periodic pattern that repeats daily. As a result, the coverage performance over a 24-hour time window is representative of the behaviour observed on subsequent days (see Figure 5.17). Similarly to the previous configuration, regions within the field of view of a GEO satellite remain continuously covered by it due to the fixed ground-track of these orbits.

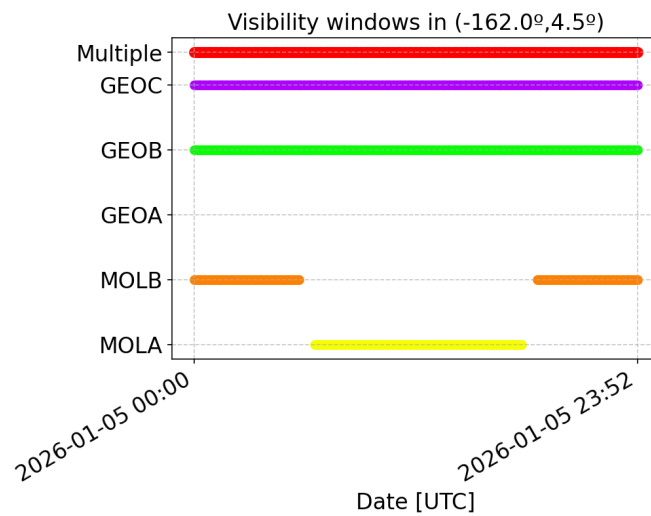


Figure 5.17: Visibility windows over the surface point at $\lambda = -162^\circ$ and $\phi = 4.5^\circ$ during one day for the two-Molniya and three-GEO constellation.

The global performance of this constellation is presented in Figure 5.18, which shows the resulting time history of the multi-satellite coverage percentage. This profile again exhibits a repeating pattern with a period of 12 hours. In this case, the maximum coverage reaches values of approximately 80%, comparable to those obtained for the Molniya-only configuration. However, the minimum coverage condition improves significantly, increasing to approximately 25%.

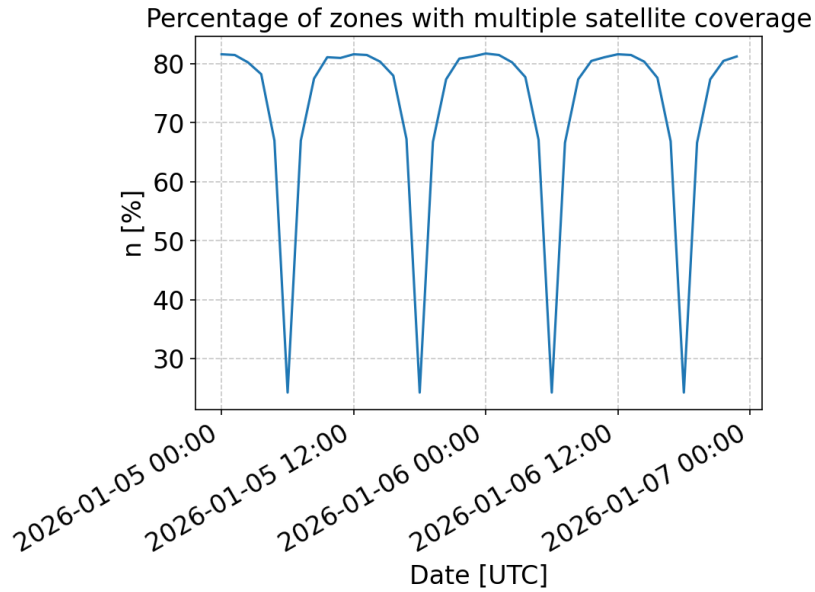
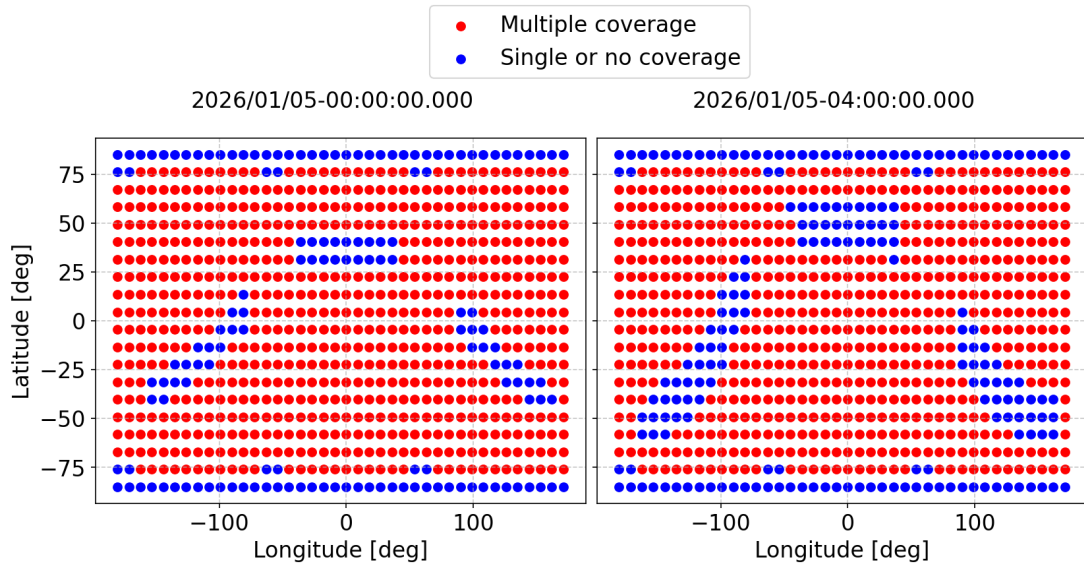


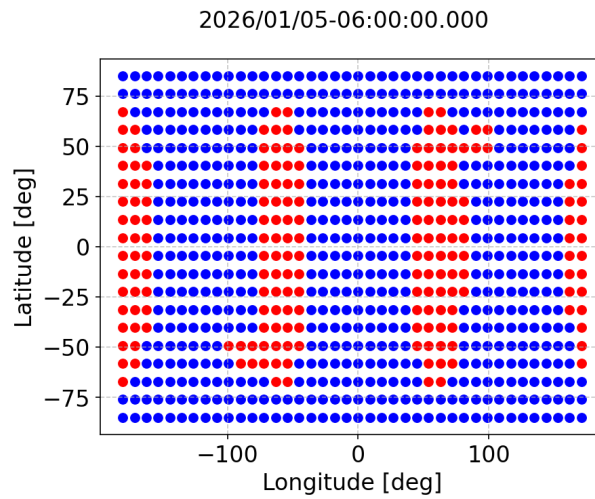
Figure 5.18: Time evolution of the percentage of Earth's surface simultaneously covered by multiple satellites for the two-Molniya and three-GEO constellation.

The coverage performance is further examined in Figure 5.19, which illustrates the geographical distribution of surface points simultaneously observed by at least two satellites in representative epochs that include maximum, intermediate, and minimum coverage conditions. In this configuration, the presence of only one Molniya satellite per hemisphere prevents multi-satellite coverage at high latitudes.

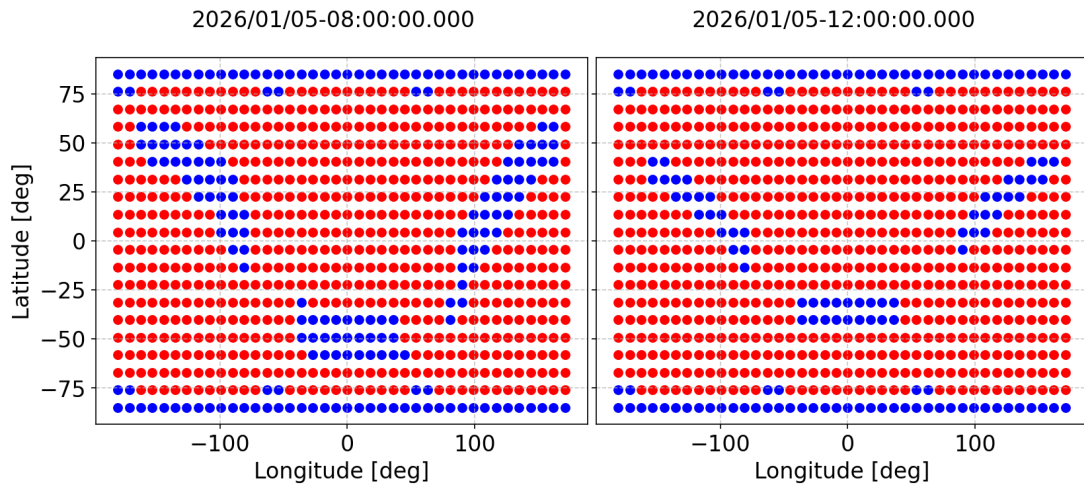
The remaining observations are similar to those identified in the previous cases. During the first four hours of the cycle, the extent of the redundant covered areas (red points) remains nearly constant, and, as the Molniya satellites approach perigee, the coverage degrades rapidly until the minimum coverage condition. At this stage, the principal regions that exhibit multi-satellite coverage are those resulting from the overlapping fields of view of the GEO satellites, resulting in a limited but non-zero redundant coverage.



(a) Maximum coverage.



(b) Minimum coverage.



(c) Restored high coverage.

Figure 5.19: Geographical distribution of surface points covered by the two-Molniya and three-GEO constellation.

5.3.4. Four-Molniya and three-GEO constellation

To address the limited multi-satellite coverage at high latitudes observed in the previous configuration, it becomes necessary to increase redundancy in these regions. This is achieved by reintroducing four Molniya satellites, two per hemisphere, while maintaining three uniformly distributed GEO satellites. A representation of this configuration is illustrated in Figure 5.20.

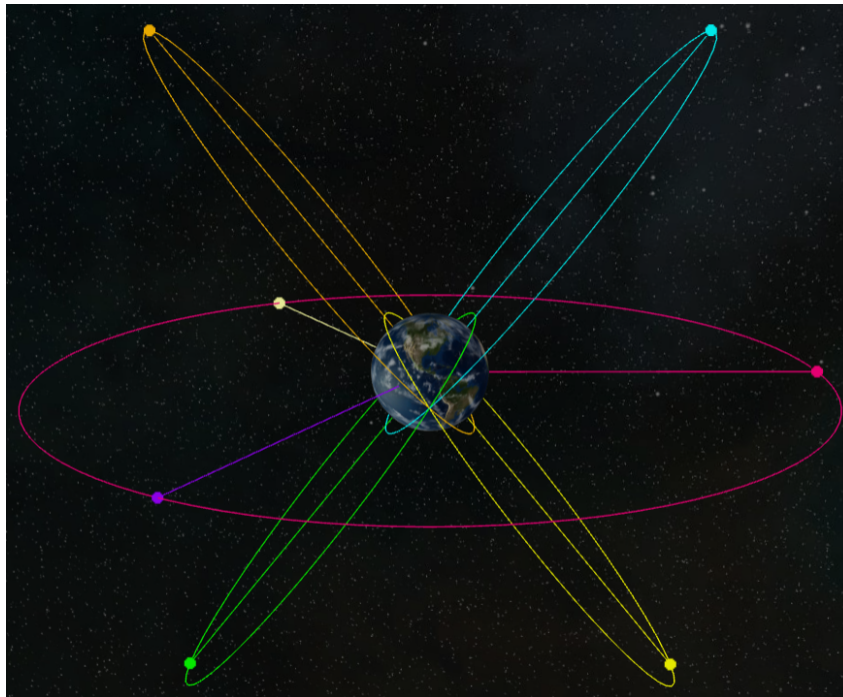


Figure 5.20: Orbital configuration of the four-Molniya and three-GEO constellation.

This configuration combines the persistent coverage provided by GEO satellites at low latitudes with enhanced redundancy from the Molniya orbits at high latitudes.

Figure 5.21 presents the satellite ground tracks, which help to visualise the spatial distribution and complementarity of the orbits within the four-Molniya and three-GEO constellation.

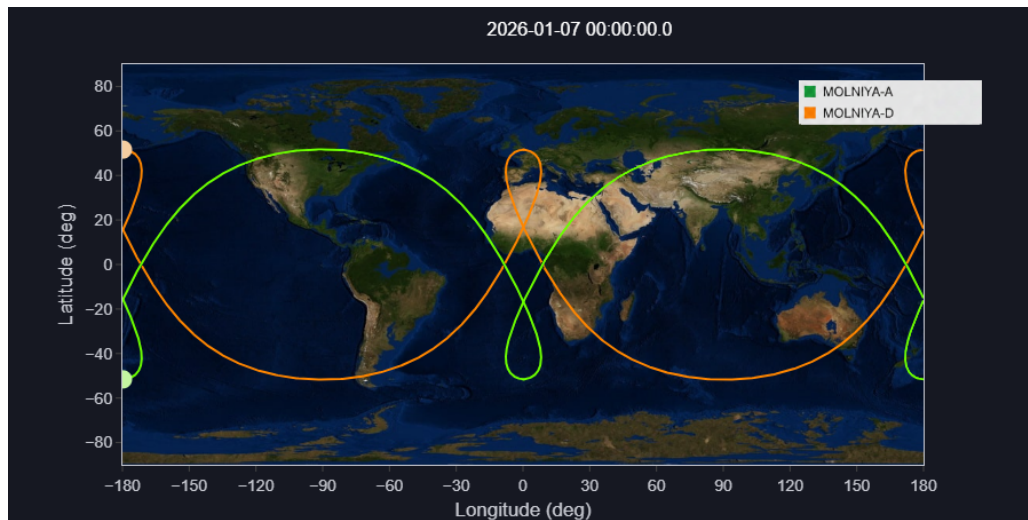
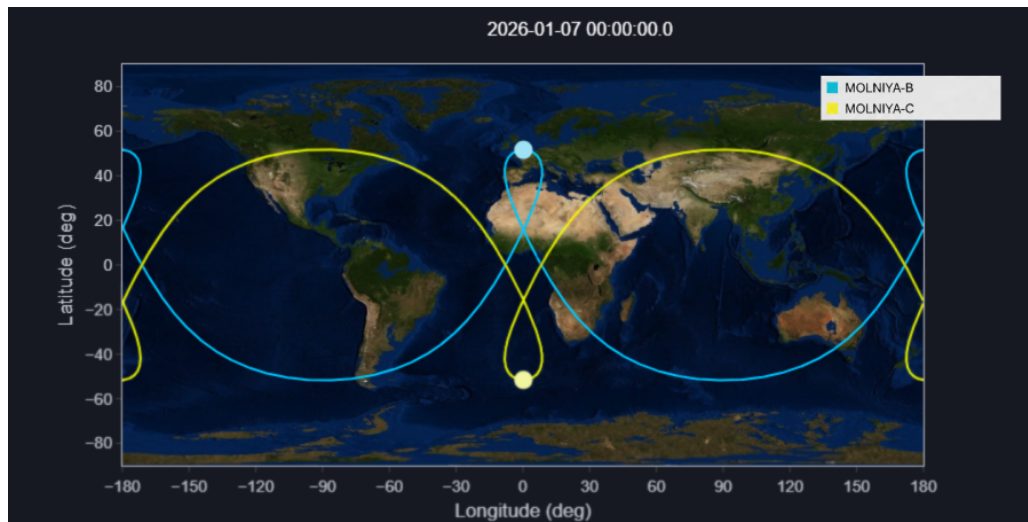
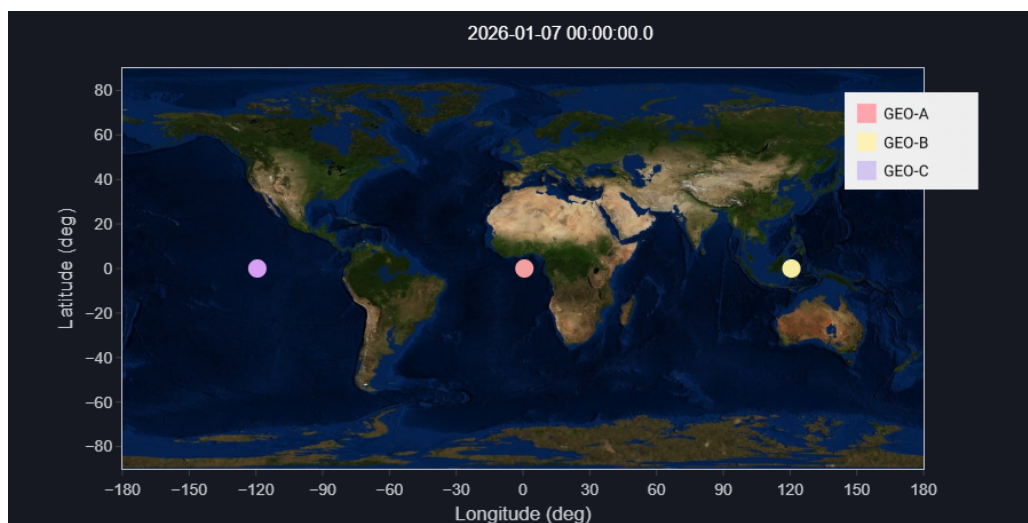
(a) *Molniya-A (MOLA)* and *Molniya-D (MOLD)*.(b) *Molniya-B (MOLB)* and *Molniya-C (MOLC)*.(c) *GEO-A (GEOA)*, *GEO-B (GEOB)* and *GEO-C (GEOC)*.

Figure 5.21: Satellite ground tracks for the four-Molniya and three-GEO constellation.

Figure 5.22 presents the visibility windows of each satellite over a representative surface point located at longitude $\lambda = -162^\circ$ and latitude $\phi = 4.5^\circ$, along with the time intervals during which the point is simultaneously observed by multiple satellites.

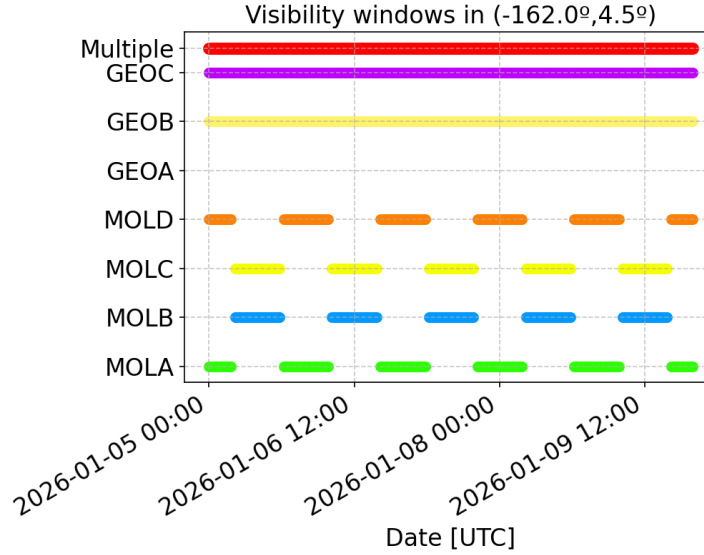


Figure 5.22: Visibility windows over the surface point at $\lambda = -162^\circ$ and $\phi = 4.5^\circ$ for the four-Molniya and three-GEO constellation.

The visibility windows exhibit a periodic pattern that repeats daily. As a result, the coverage performance over a 24-hour time window is representative of the behaviour observed on subsequent days (see Figure 5.23).

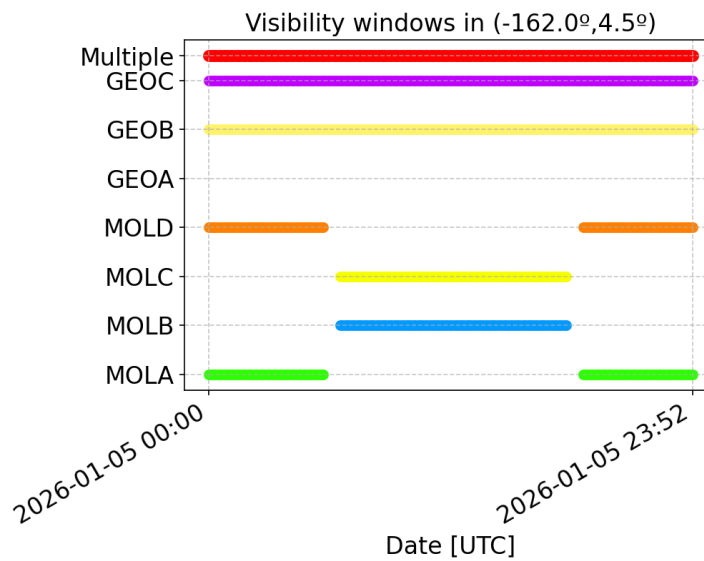


Figure 5.23: Visibility windows over the surface point at $\lambda = -162^\circ$ and $\phi = 4.5^\circ$ during one day for the four-Molniya and three-GEO constellation.

The percentage of discretised Earth surface points simultaneously observed by at least two satellites as a function of time is shown in Figure 5.24, which again exhibits a repeating pattern with a period of 12 hours. The maximum coverage reaches approximately 100%, indicating full coverage of the Earth’s surface, and the minimum coverage remains similar to the previous configuration, with approximately 25% of the total surface, since the number of GEO satellites is unchanged and the regions with multi-satellite coverage during these intervals are therefore the same.

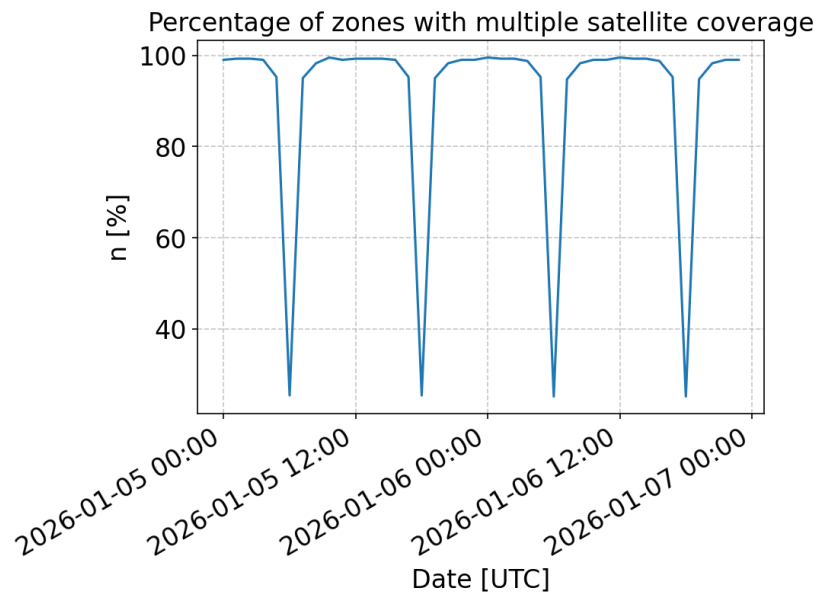


Figure 5.24: Time evolution of the percentage of Earth’s surface simultaneously covered by multiple satellites for the four-Molniya and three-GEO constellation.

The coverage performance is further examined in Figure 5.25, which illustrates the geographical distribution of surface points simultaneously observed by at least two satellites in representative epochs that include maximum, intermediate, and minimum coverage conditions.

For a 12-hour cycle, nearly full coverage is achieved during the first and last four hours, when the Molniya satellites are near their apogees, resulting in eight hours of almost complete multi-satellite coverage. The remaining four hours, corresponding to the passage of the Molniya satellites through the perigee, represent the minimum coverage condition. During these intervals, regions with redundant coverage are limited to the areas where the fields of view of the GEO satellites overlap. Consequently, each day provides roughly 16 hours of full coverage and 8 hours at around 25% redundancy.

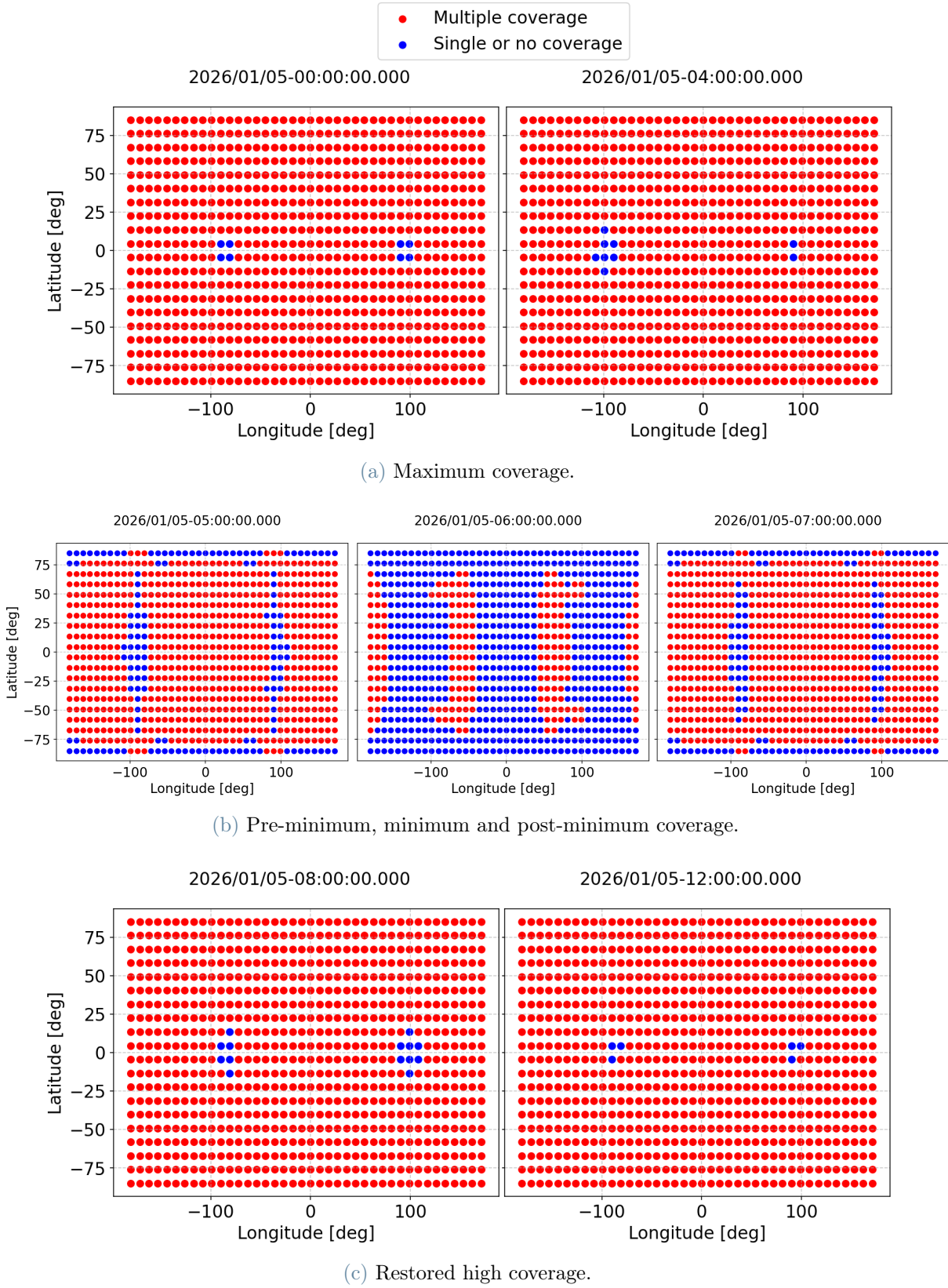


Figure 5.25: Geographical distribution of surface points covered by the four-Molniya and three-GEO constellation.

5.3.5. Four-Tundra and three-GEO constellation

Another alternative to the use of Molniya orbits for the HEO satellites is the adoption of Tundra orbits. The main difference with respect to the previous configurations lies in the orbital period, which is 24 hours for Tundra orbits, resulting in a ground track that remains fixed over a given region of the Earth. Accordingly, this constellation consists of four Tundra satellites and three GEO satellites.

The constellation phasing is selected so that the apogees of the Tundra ground tracks are located between the longitudes of the GEO satellites. This arrangement enhances longitudinal complementarity and improves simultaneous coverage by avoiding the concentration of multiple satellites over the same regions. In addition, it enables persistent coverage at high-latitude areas. A representation of this configuration is illustrated in Figure 5.26, while Figure 5.27 presents the corresponding satellite ground tracks.

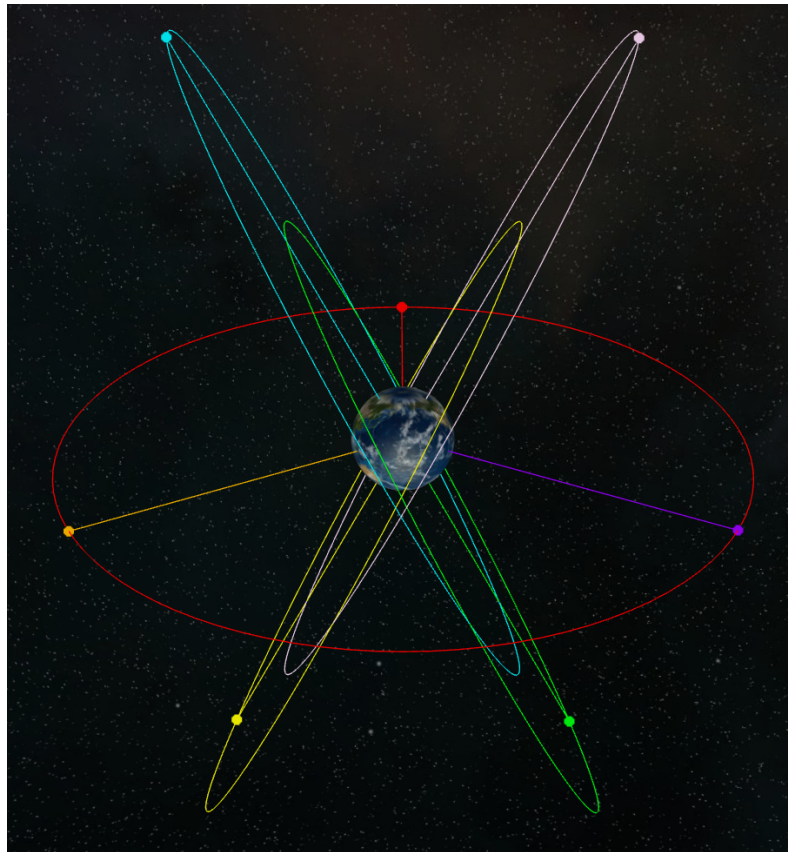


Figure 5.26: Orbital configuration of the four-Tundra and three-GEO constellation.

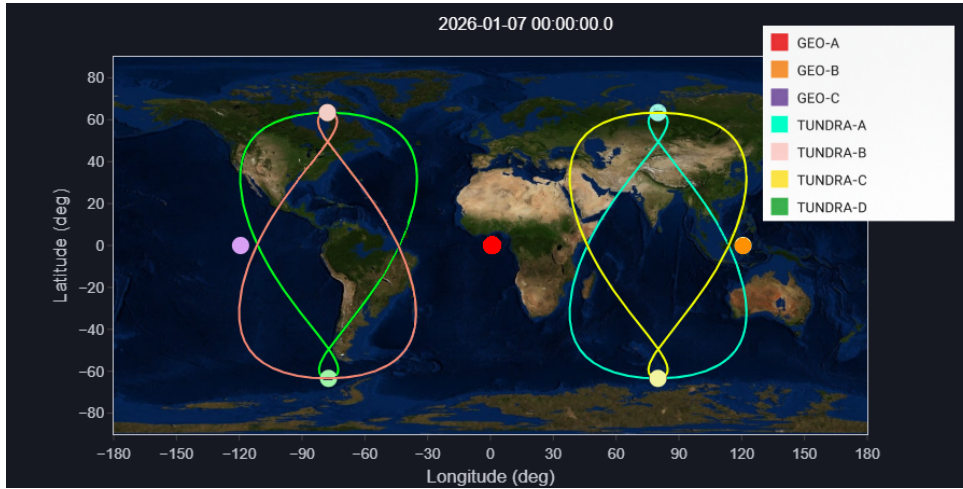


Figure 5.27: Satellite ground tracks for the four-Tundra and three-GEO constellation.

Figure 5.28 presents the visibility windows of each satellite over a representative surface point located at longitude $\lambda = -162^\circ$ and latitude $\phi = 4.5^\circ$, along with the time intervals during which the point is simultaneously observed by multiple satellites.

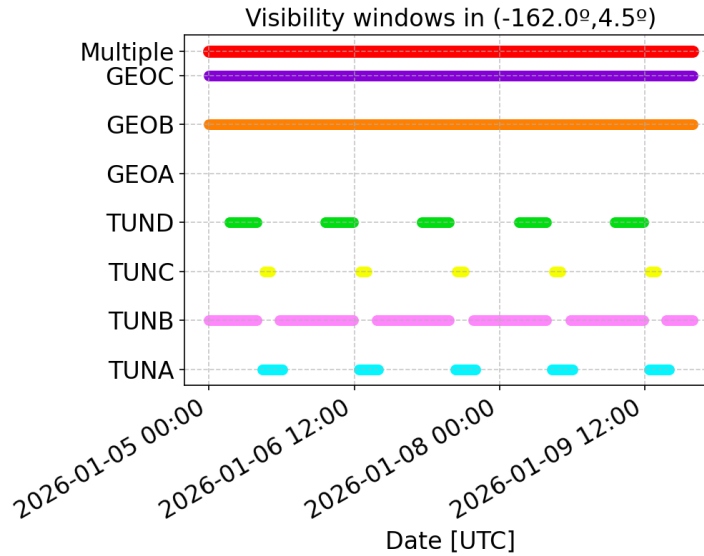


Figure 5.28: Visibility windows over the surface point at $\lambda = -162^\circ$ and $\phi = 4.5^\circ$ for the four-Tundra and three-GEO constellation.

The visibility windows exhibit a periodic pattern that repeats daily. As a result, the coverage performance over a 24-hour time window is representative of the behaviour observed on subsequent days (see Figure 5.29). Similarly to the previous configuration, regions within the field of view of a GEO satellite remain continuously covered by it due to the fixed ground-track of these orbits.

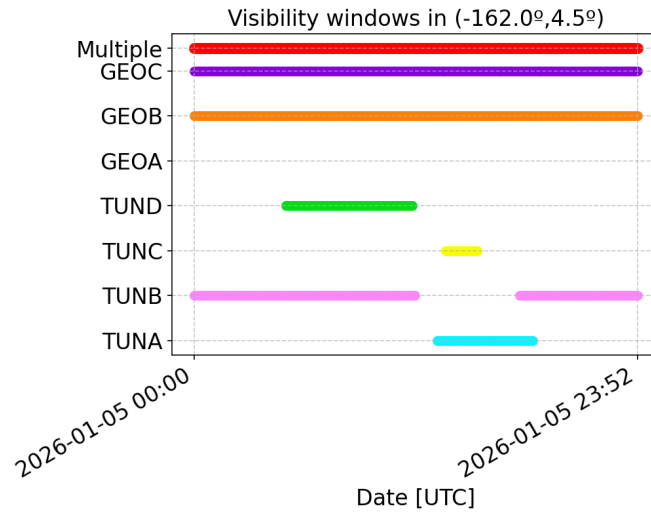


Figure 5.29: Visibility windows over the surface point at $\lambda = -162^\circ$ and $\phi = 4.5^\circ$ during one day for the four-Tundra and three-GEO constellation.

Figure 5.30 presents the global performance of the constellation through the resulting time history of the multi-satellite coverage percentage, which now exhibits a repeating pattern with a period of one day. With this configuration, the maximum coverage reaches values of approximately 100%, while the minimum coverage condition improves significantly, increasing to around 80%. This improvement is attributed to the less aggressive passage through the perigee of the Tundra satellites, which occurs at higher altitudes and lower velocity compared to Molniya orbits, allowing them to overfly the Earth covering larger surface areas.

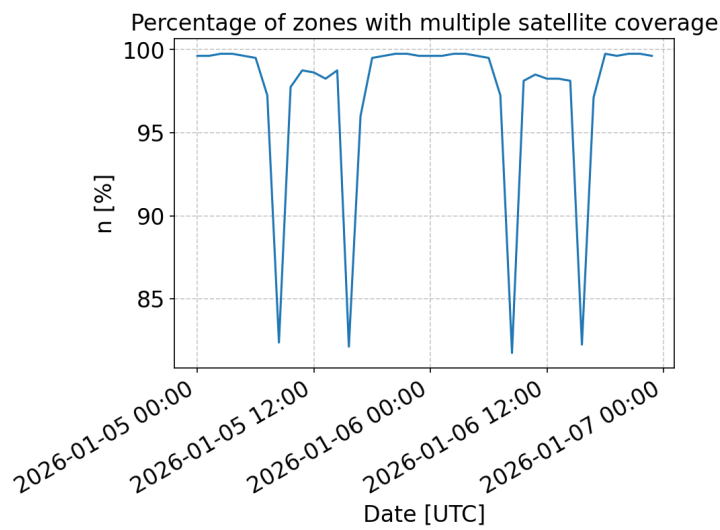
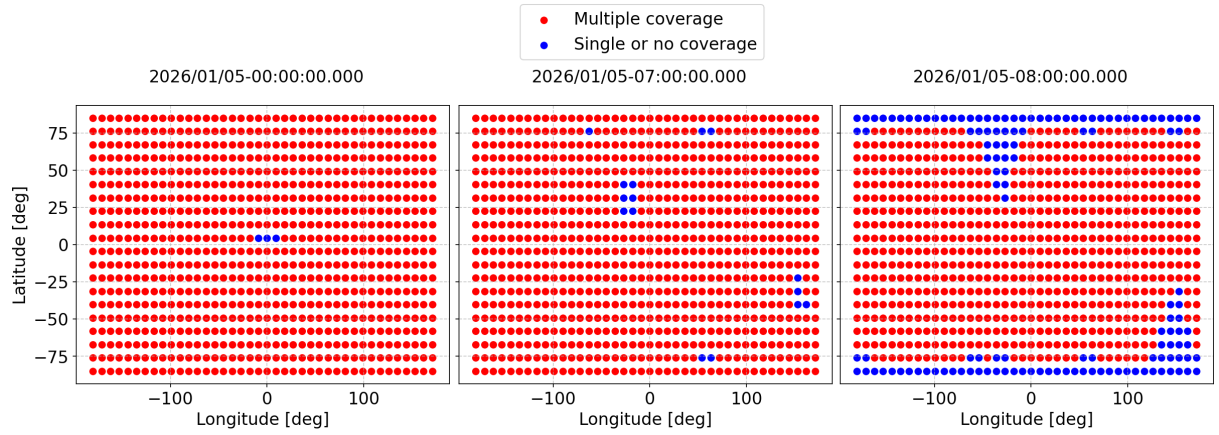


Figure 5.30: Time evolution of the percentage of Earth's surface simultaneously covered by multiple satellites for the four-Tundra and three-GEO constellation.

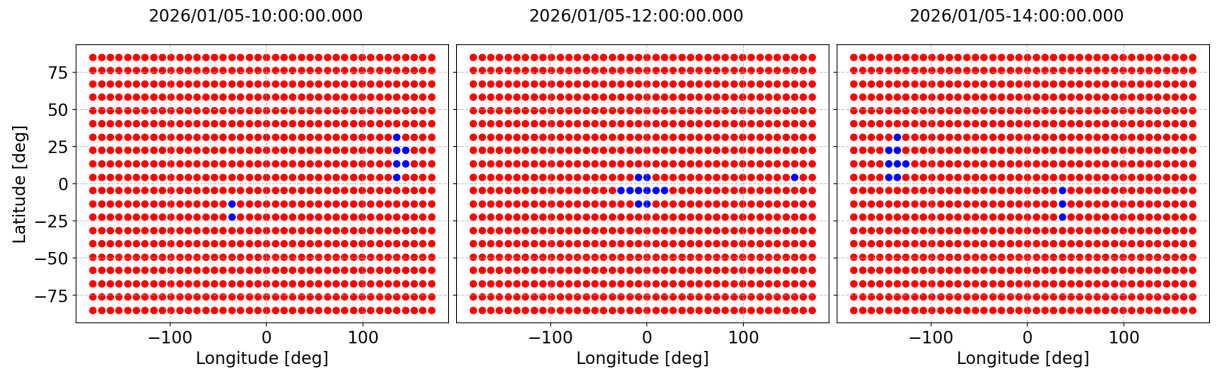
Two distinct minima can be observed within each daily cycle, corresponding to the instants at which the Tundra satellites cross the equatorial plane. During these phases, the multi-satellite coverage at high latitudes is temporarily lost.

The coverage performance is further examined in Figure 5.31, which illustrates the geographical distribution of surface points simultaneously observed by at least two satellites in representative epochs that include maximum, intermediate, and minimum coverage conditions.

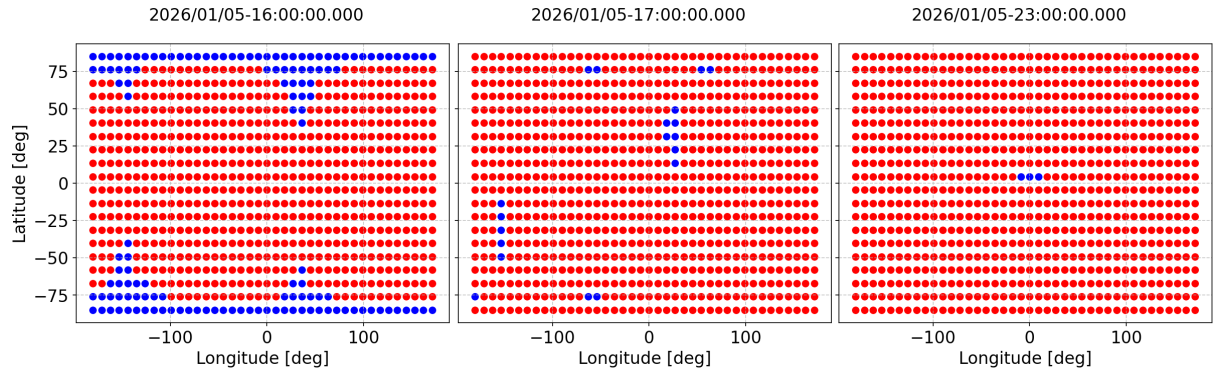
For a daily cycle, nearly full coverage is achieved during the first and last seven hours, when the Tundra satellites remain near their apogees, as well as during the four-hour interval between the two minima. This results in approximately 18 hours of almost complete multi-satellite coverage in one day. The remaining four hours correspond to the two crossings of the equatorial plane by the Tundra satellites, during both the ascending and descending nodes, and represent the minimum coverage condition. During these intervals, the areas with redundant coverage are significantly wider than in the previously analysed configurations, although high-latitude regions are not covered.



(a) Maximum coverage and first minimum coverage zone.



(b) Intermediate coverage.



(c) Second minimum coverage zone and restored high coverage.

Figure 5.31: Geographical distribution of surface points covered by the four-Tundra and three-GEO constellation.

5.3.6. Chinese space-based early-warning system configuration

The analysis is extended to a real operational constellation in order to validate and compare the results obtained from the previously proposed configurations. In particular, the Chinese constellation is considered, as the list of satellites composing the system is publicly available, allowing the corresponding TLEs to be retrieved.

Table 5.1 summarises the satellites included in the Chinese space-based early-warning system, while Figure 5.32 illustrates a representation of the orbital configuration.

Table 5.1: Satellites of the Chinese space-based early-warning system.

Satellite	NORAD ID	Orbit type
TJS-21	66586	HEO
Shiyan-10 02	54878	HEO
TJS-13	62188	HEO
Shiyan-10 01	49258	HEO
TJS-2	41911	GEO </td
TJS-5	44978	GEO
TJS-6	47613	GEO

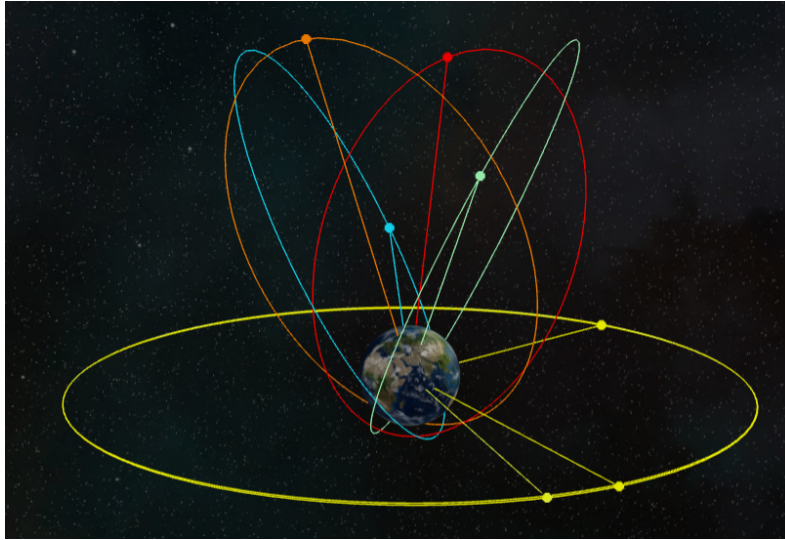


Figure 5.32: Orbital configuration of the Chinese space-based early-warning system.

Two main observations can be drawn from the orbital configuration. First, the GEO satellites are not uniformly distributed along the orbit, but are clustered in specific regions, primarily in the eastern longitudes. Second, each of the HEO satellites is placed on a different orbital plane, with their apogees uniformly spaced in longitude and their phasing approximately synchronised in pairs. In addition, all the apogees are located in the northern hemisphere. This configuration ensures persistent coverage at high latitudes.

The coverage performance of this constellation is analysed using the same metrics described in the previous configurations.

Figure 5.33 presents the visibility windows of each satellite over a representative surface point located at longitude $\lambda = 0^\circ$ and latitude $\phi = 40.3^\circ$, along with the time intervals during which the point is simultaneously observed by multiple satellites.

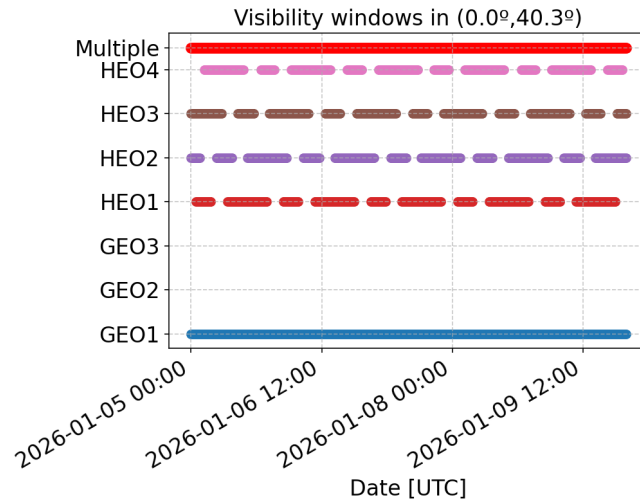


Figure 5.33: Visibility windows over the surface point at $\lambda = 0^\circ$ and $\phi = 40.3^\circ$ for the Chinese space-based early-warning system.

The visibility windows exhibit a periodic pattern that repeats daily. As a result, the coverage performance over a 24-hour time window is representative of the behaviour observed on subsequent days (see Figure 5.34).

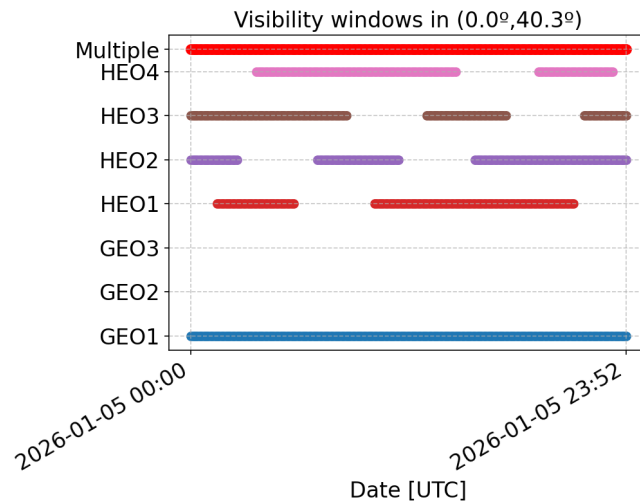


Figure 5.34: Visibility windows over the surface point at $\lambda = 0^\circ$ and $\phi = 40.3^\circ$ during one day for the Chinese space-based early-warning system.

The global performance of the constellation is presented in Figure 5.35, showing the time history of the multi-satellite coverage. In this configuration, the coverage exhibits a repeating pattern with a period of one day, with values ranging between 70% and 80% of the total Earth's surface.

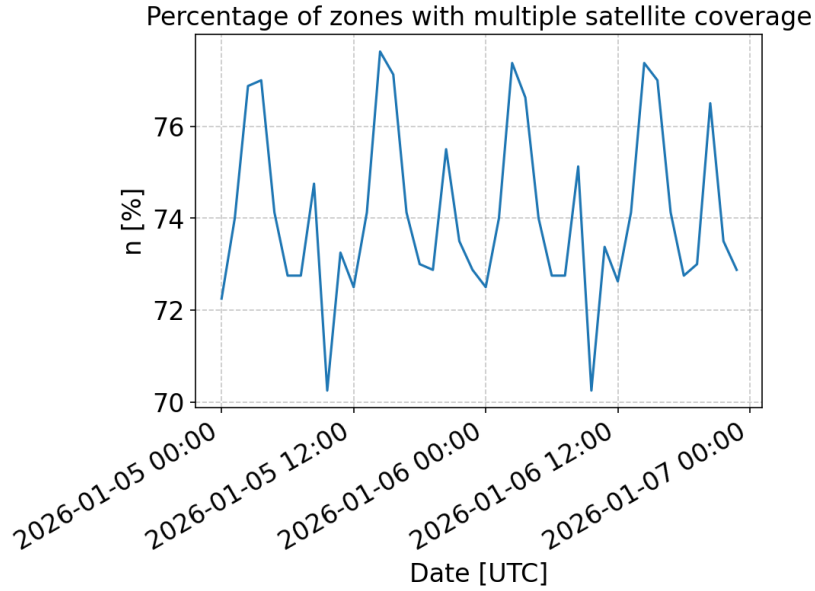


Figure 5.35: Time evolution of the percentage of Earth's surface simultaneously covered by multiple satellites for the Chinese space-based early-warning system.

This behaviour is consistent with the orbital configuration, as the system is designed to focus on a specific region rather than providing global coverage. Figure 5.36 provides further details, illustrating the geographical distribution of surface points simultaneously observed by at least two satellites in representative epochs. It can be observed that the covered region remains approximately constant, with the south-west outside the intended area of focus.

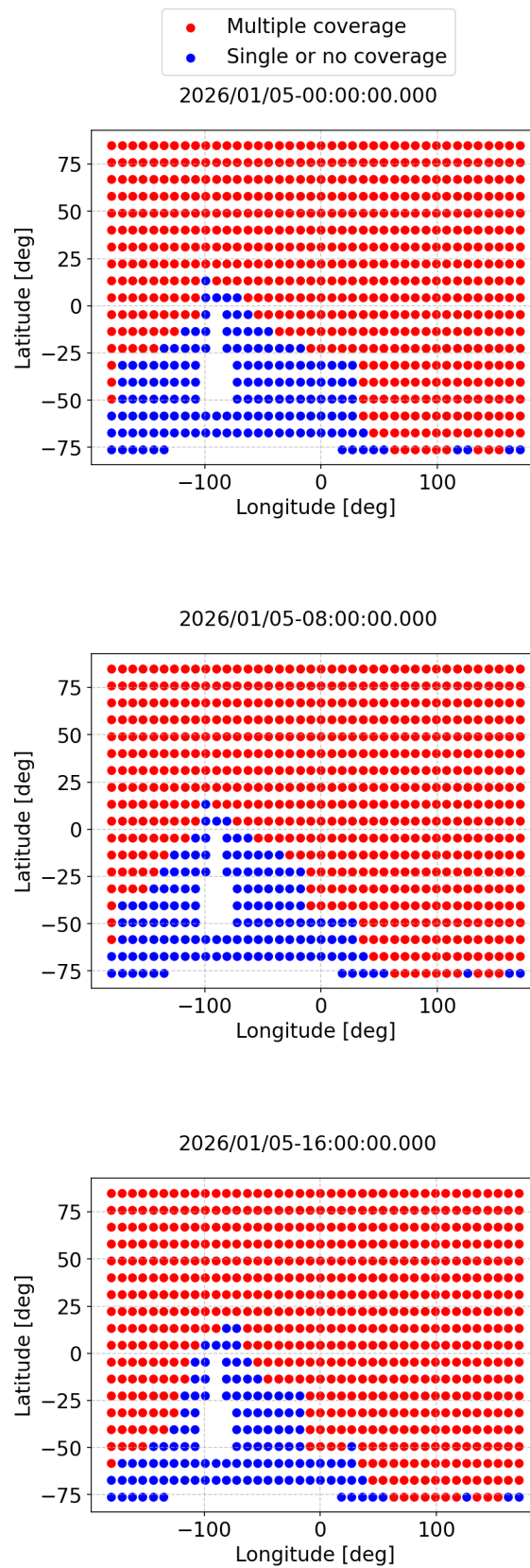


Figure 5.36: Geographical distribution of surface points covered by the Chinese space-based early-warning system.

5.3.7. Eight-HEO and three-GEO constellation

The final configuration analysed consists of eight HEO satellites and three GEO satellites. This configuration is designed to demonstrate that it is possible to maintain multi-satellite coverage of the whole Earth's surface, using the same type of orbits employed in the real Chinese constellation.

If no constraints on the number of satellites are imposed, this can be achieved simply by distributing the three GEO satellites uniformly along their orbit to cover all longitudes, and duplicating the HEO orbits by adding 180° to the argument of perigee of each satellite. In this way, continuous coverage is ensured not only in the northern hemisphere (as in the Chinese system), but also in the southern hemisphere.

To simplify and avoid overcrowding in the results due to the large number of orbits in this constellation, the orbital configuration and the satellite ground tracks are omitted.

Figure 5.37 presents the visibility windows of each satellite over a representative surface point located at longitude $\lambda = 0^\circ$ and latitude $\phi = 40.3^\circ$, along with the time intervals during which the point is simultaneously observed by multiple satellites.

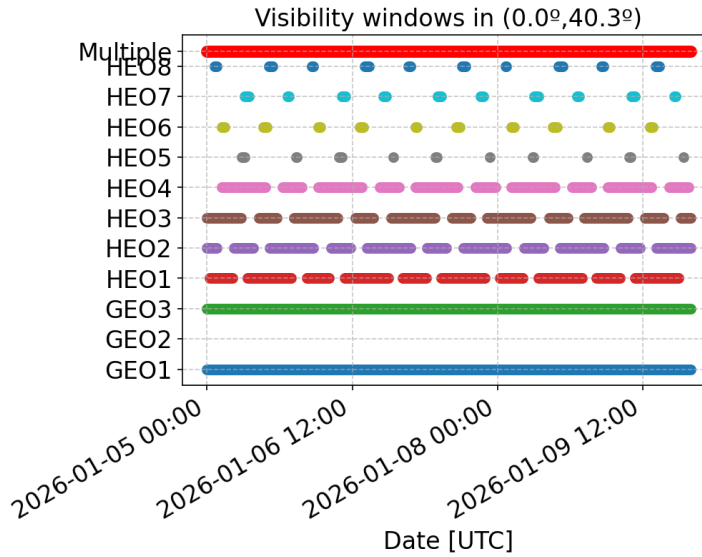


Figure 5.37: Visibility windows over the surface point at $\lambda = 0^\circ$ and $\phi = 40.3^\circ$ for the eight-HEO and three-GEO constellation.

These visibility windows serve to illustrate that the visibility patterns repeat daily, and therefore, the analysis over a single day is representative of the constellation's behaviour on subsequent days (see Figure 5.38).

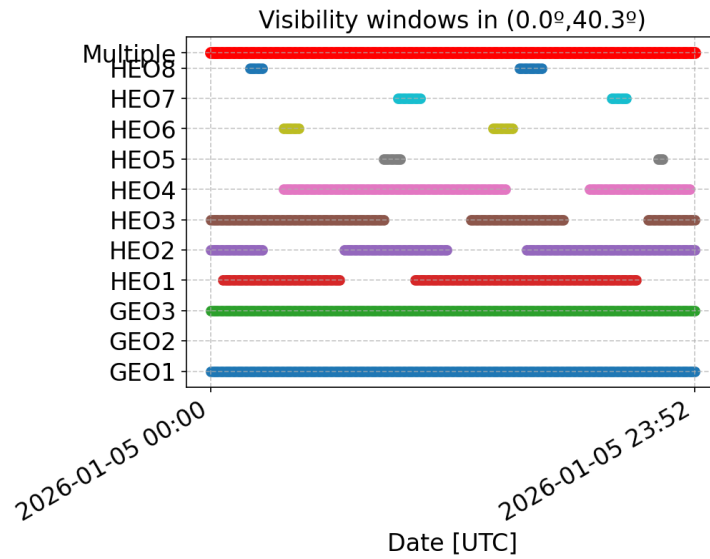


Figure 5.38: Visibility windows over the surface point at $\lambda = 0^\circ$ and $\phi = 40.3^\circ$ during one day for the eight-HEO and three-GEO constellation.

The global performance of the constellation is presented in Figure 5.39, showing coverage values that exceeds 99% at all time instants. Therefore, near-complete multi-satellite coverage is achieved permanently.

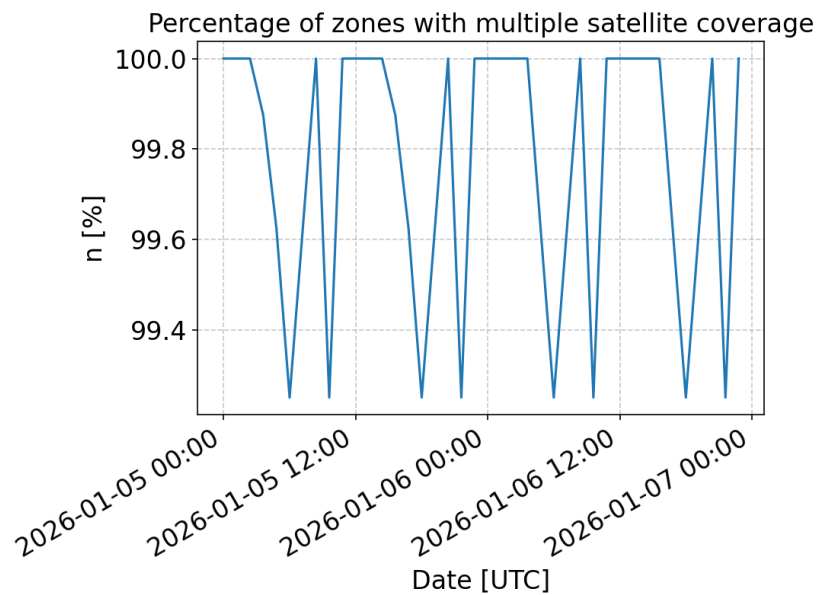


Figure 5.39: Time evolution of the percentage of Earth's surface simultaneously covered by multiple satellites for the eight-HEO and three-GEO constellation.

5.3.8. Performance comparison and conclusions

A summary of the main characteristics and coverage performance of all the satellite constellations analysed in this chapter is presented in Table 5.2. The values reported are approximate and extracted from the time histories of the multi-satellite coverage percentage.

Table 5.2: Summary of analysed constellations and their coverage performance.

Constellation	HEO sats.	GEO sats.	Hours/day with >95% coverage	Minimum coverage
Four Molniya	4	0	0	0%
Four Molniya and Two GEO	4	2	≈ 16	0%
Two Molniya and Three GEO	2	3	0	$\approx 25\%$
Four Molniya and Three GEO	4	3	≈ 16	$\approx 25\%$
Four Tundra and Three GEO	4	3	≈ 18	$\approx 80\%$
Chinese system	4	3	0 (region-specific)	$\approx 70\%$
Eight HEO and Three GEO	8	3	24	$\approx 99.2\%$

The following conclusions can be drawn:

- At least four HEO satellites and three GEO satellites are required to ensure robust multi-satellite coverage performance at a global scale.
- When the objective is global coverage, Tundra orbits offer clear advantages over Molniya orbits, providing longer intervals of nearly full coverage and significantly higher coverage levels under minimum coverage conditions (if the constellation geometry and phasing are properly designed).
- When the objective is coverage focused on a specific region, either Molniya or Tundra orbits are sufficient to guarantee continuous coverage.
- In the absence of constraints on the number of satellites, increasing the number of HEO satellites allows permanent near-complete multi-satellite coverage, without the need to add further GEO satellites.

6 | Conclusions and future work

After the development of the project, we need to reflect on the applied procedure, on the algorithms developed, and on the validity of the results obtained.

6.1. Conclusions

The research carried out for this thesis led to the development of a simulation and analysis framework for space-based early warning and surveillance applications oriented towards launch operations. The main contributions and conclusions of the work can be summarised as follows.

First, a trajectory simulation tool has been developed for the modelling of ballistic missiles (BM), including short-range, medium-range, intermediate-range, and intercontinental systems, in single-stage and multi-stage configurations, as well as direct-ascent anti-satellite (ASAT) weapons and orbital launch vehicles, for which multi-stage configurations have been considered. The simulator is capable of generating physically consistent trajectories for the different vehicle classes, accounting for the distinct characteristics of each case.

Second, the generated trajectories have been validated against real cases through comparative analyses, ensuring that the simulated motion, flight phases, and characteristic parameters are representative of realistic scenarios. This validation step provides confidence in the reliability of the simulator as a basis for subsequent processing and analysis.

Third, an identification and classification algorithm has been implemented to discriminate between the different types of trajectories based on their observable characteristics. The algorithm has been subsequently optimised, improving its performance in terms of classification accuracy and robustness. The results demonstrate the capability of the proposed approach to effectively distinguish between different types of launch events. It was shown that the best performance is achieved by considering a three-dimensional feature space, with velocity magnitude, flight-path angle, and acceleration magnitude as classification variables, and a history-based approach, in which all measurements up to the classification instant are considered, rather than only the most recent one. Furthermore, the inclusion

of measurement noise highlights the existence of a minimum classification altitude below which the results of the algorithm are not reliable.

Finally, a mission analysis has been carried out to evaluate the coverage performance of different satellite constellations. Several HEO-GEO configurations have been analysed and compared, including Molniya, Tundra, and a real operational system. The study highlights the impact of orbital selection, constellation phasing, and satellite distribution on multi-satellite coverage performance, and demonstrates that near-complete global redundant coverage can be achieved with appropriately designed configurations.

6.2. Future work

There are several aspects that could be further developed and considered as future work:

- The current trajectory simulator could be extended by incorporating rotational dynamics in addition to translational motion, allowing the vehicle attitude evolution to be modelled explicitly.
- A more detailed modelling of ballistic missile re-entry dynamics could be implemented.
- The ballistic missile model could be extended to include Multiple Independently Targetable Reentry Vehicles (MIRVs).
- More realistic guidance profiles and control laws could be adopted for the different vehicle classes.
- Some of the current modelling assumptions could be relaxed, such as allowing vehicle-dependent parameters to vary with time or replacing the use of a constant ballistic coefficient with a more accurate aerodynamic drag model.
- The performance of the classification algorithm could be further improved by expanding the database of simulated trajectories with representative real-world orbital launcher trajectories obtained from publicly available information.
- Machine learning techniques, such as neural networks, could be used to complement or replace the current classification methodology.
- The mission analysis could be refined by considering additional factors that affect infrared sensor performance.
- The incorporation of ground-based and airborne early-warning assets could be analysed to improve low-altitude detection and tracking performance, complementing

the space-based systems considered in this work.

- The development of a methodology for trajectory determination and estimation based on sensor measurements.

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A | Appendix A: Catalogue of ballistic missiles and U.S. ASAT tests

Table A.1: List of SRBMs (1).

Country	Missile	Range (km)	Propellant	Launcher
U.S.	MGM-140 ATACMS	300	Solid	Road-mobile
U.S.	MGM-164 Lancer (PrSM)	500	Solid	Road-mobile
U.S.	MGM-52 Lance	130	Solid	Road-mobile
Russia	SCUD B (SS-1c Mod 1)	300	Liquid	Road-mobile
Russia	SS-1c Mod 2	240+	Liquid	Road-mobile
Russia	SS-21 Mod 3	120	Solid	Road-mobile
Russia	SS-26	300	Solid	Road-mobile
Russia	Iskander-E	280	Solid	Road-mobile
China	CSS-11 Mod 1	800+	Solid	Road-mobile
China	CSS-6 Mod 1	600	Solid	Road-mobile
China	CSS-6 Mod 2	850+	Solid	Road-mobile
China	CSS-6 Mod 3	725+	Solid	Road-mobile
China	CSS-7 Mod 1	300	Solid	Road-mobile
China	CSS-7 Mod 2	600	Solid	Road-mobile
China	CSS-9 Mod 1	150	Solid	Road-mobile
China	CSS-9 Mod-X-2	260	Solid	Road-mobile
China	CSS-14 Mod-X-2	280	Solid	Road-mobile
China	CSS-X-15	280	Solid	Road-mobile
North Korea	SCUD B	300	Liquid	Road-mobile
North Korea	SCUD C	500	Liquid	Road-mobile
North Korea	Toksa	150	Solid	Road-mobile
North Korea	ER SCUD	700-995	Liquid	Road-mobile

Table A.2: List of SRBMs (2).

Country	Missile	Range (km)	Propellant	Launcher
India	Prithvi I	150	Liquid	Road-mobile
India	Prithvi II	250	Liquid	Road-mobile
India	Dhanush	400	Liquid	Ship-based
India	Agni I	700	Solid	Road-mobile
Pakistan	Hatf-9	60	Solid	Road-mobile
Pakistan	Hatf-1	50	Solid	Road-mobile
Pakistan	Shaheen I	750	Solid	Road-mobile
Pakistan	Ghaznavi	250	Solid	Road-mobile
Iran	Fateh-110	200-300	Solid	Road-mobile
Iran	Shahab 1	300	Liquid	Road-mobile
Iran	Shahab 2	500	Liquid	Road-mobile
Iran	CSS-8 (M-7)	150	Solid/Liquid	Road-mobile
Iran	Qiam-1	unknown	Liquid	Road-mobile
Syria	SCUD D	700	Liquid	Road-mobile
Israel	Jericho I	500	Liquid	Road-mobile

Table A.3: List of MR/IRBMs.

Country	Missile	Range (km)	Stages	Propellant	Launcher
U.S.	PGM-17 Thor	1900	1	Liquid	Silo
U.S.	PGM-19 Jupiter	2400	1	Liquid	Silo
U.S.	LRHW	2875	2	Solid	Road-mobile
China	CSS-2	3000	1	Liquid	Transportable
China	CSS-5 Mod 1	1750+	2	Solid	Road-mobile
China	CSS-5 Mod 2	1750+	2	Solid	Road-mobile
China	CSS-5 Conventional	1750+	2	Solid	Mobile
China	CSS-5 ASBM	1500+	2	Solid	Mobile
North Korea	No Dong	1250	1	Liquid	Road-mobile
North Korea	IRBM	3000+	1	Liquid	Road-mobile
India	Agni II	2000+	2	Solid	Rail-mobile
India	Agni III	3200+	2	Solid	Rail-mobile
India	Agni IV	3500+	2	Solid	Rail-mobile
Pakistan	Ghauri	1250	1	Liquid	Road-mobile
Pakistan	Shaheen 2	2000	2	Solid	Road-mobile
Iran	Shahab 3	2000	1	Liquid	Silo & Road-mobile
Iran	Sejjil	2000	2	Solid	Road-mobile
France	S-3	3500	2	Solid	Silo

Table A.4: List of ICBMs.

Country	Missile	Range (km)	Stages	Propellant	Launcher
U.S.	Minuteman III (LGM-30G)	14000	3	Solid	Silo
U.S. & U.K.	Trident II	11300	3	Solid	Submarine
Russia	SS-18 Mod 5	10000+	2 + PBV	Liquid	Silo
Russia	SS-19 Mod 3	9000+	2 + PBV	Liquid	Silo
Russia	SS-25	11000	3 + PBV	Solid	Road-mobile
Russia	SS-27 Mod 1	11000	3 + PBV	Solid	Silo & Road-mobile
Russia	SS-27 Mod 2	11000	3 + PBV	Solid	Silo & Road-mobile
China	CSS-3	5500+	2	Liquid	Transportable
China	CSS-4 Mod 1	12000+	2	Liquid	Silo
China	CSS-10 Mod 1	7000+	3	Solid	Road-mobile
China	CSS-10 Mod 2	11000+	3	Solid	Road-mobile
North Korea	Taepo Dong-2	5500+	2/3	Liquid	Fixed
North Korea	Hwasong-17	15000	2	Liquid	Road-mobile
India	Agni V	5000+	3	Solid	Undetermined
Israel	Jericho III	11500	3	Solid	Road-mobile

Table A.5: Summary of known or suspected U.S. DA-ASAT Tests. (Source: [15])

Date	Interceptor	Target	Altitude (km)	Deb. ¹	Result
Sep. 1959	High Virgo	None	12	0	Failed test
Oct. 13, 1959	Bold Orion	Explorer VI	200	0	Successful test
Oct. 1, 1961	SIP	None	Unknown	0	Successful rocket test
Oct. 5, 1961	HiHo	None	Unknown	0	Rocket failure
Mar. 26, 1962	HiHo	None	Unknown	0	Rocket failure
May. 5, 1962	SIP	None	Unknown	0	Successful rocket test
July 26, 1962	HiHo	None	1600	0	Successful rocket test
Dec. 17, 1962	Nike Zeus	None	160	0	Successful intercept point
Feb. 15, 1963	Nike Zeus	None	241	0	Successful intercept point
Mar. 21, 1963	Nike Zeus	None	-	0	Failed simulated intercept
Apr. 19, 1963	Nike Zeus	None	-	0	Failed simulated intercept
May. 24, 1963	Nike Zeus	Agenda D	Unknown	0	Successful close intercept
Jan. 4, 1964	Nike Zeus	None	146	0	Successful sim. intercept
Feb. 14, 1964	Program 437	Transit 2A	1000	0	Success
Mar. 1, 1964	Program 437	Unknown	674	0	Success
Apr. 21, 1964	Program 437	Unknown	778	0	Success
May 28, 1964	Program 437	Unknown	932	0	Failed
Nov. 16, 1964	Program 437	Unknown	1148	0	Success
Apr. 1, 1965	Nike Zeus	None	Unknown	0	-
Apr. 5, 1965	Program 437	Transit 2A	826	0	Success
June-July 1965	Nike Zeus	None	Unknown	0	3/4 successful tests
Jan. 1966	Nike Zeus	None	Unknown	0	Successful sim. intercept
Mar. 31, 1967	Program 437	Unknown deb.	484	0	Success
May 15, 1968	Program 437	Unknown	823	0	Success
Nov. 21, 1968	Program 437	Unknown	1158	0	Success
Mar. 28, 1970	Program 437	Unknown sat.	1074	0	Success
Jan. 21, 1984	ASM-135	None	1000	0	ASM-135 fired from F-15
Nov. 13, 1984	ASM-135	Star	1000	0	Failed test
Sep. 13, 1984	ASM-135	Solwind	555	285	Successful test
Aug. 22, 1986	ASM-135	Star	1000	0	Successful test in tracking
Sep. 29, 1986	ASM-135	Star	1000	0	Successful test in tracking
Feb. 20, 2008	SM-3	USA 193	270	174	Successful test

¹ Number of debris created.

B | Appendix B: Evolution of the three-dimensional feature space

This appendix presents the complete altitude evolution of the three-dimensional feature space used for trajectory classification. Figures are provided at 10 km altitude increments for illustrative purposes.

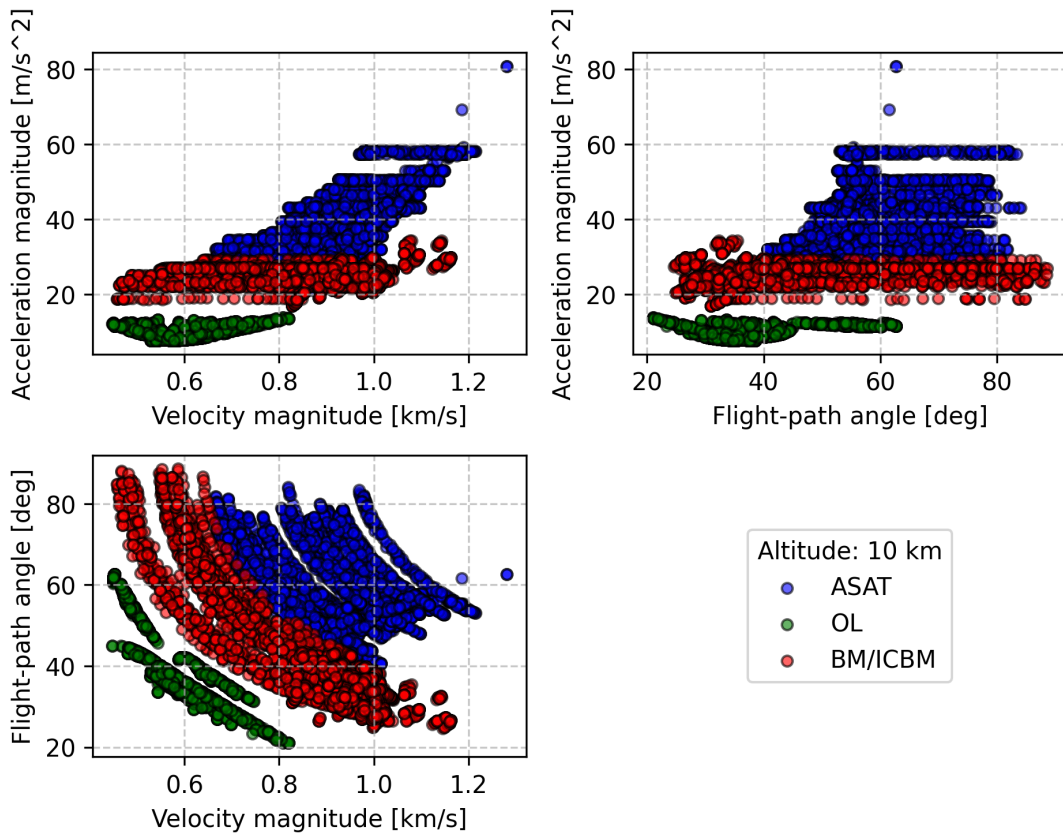


Figure B.1: Three-dimensional feature space projections at a classification altitude of 10 km.

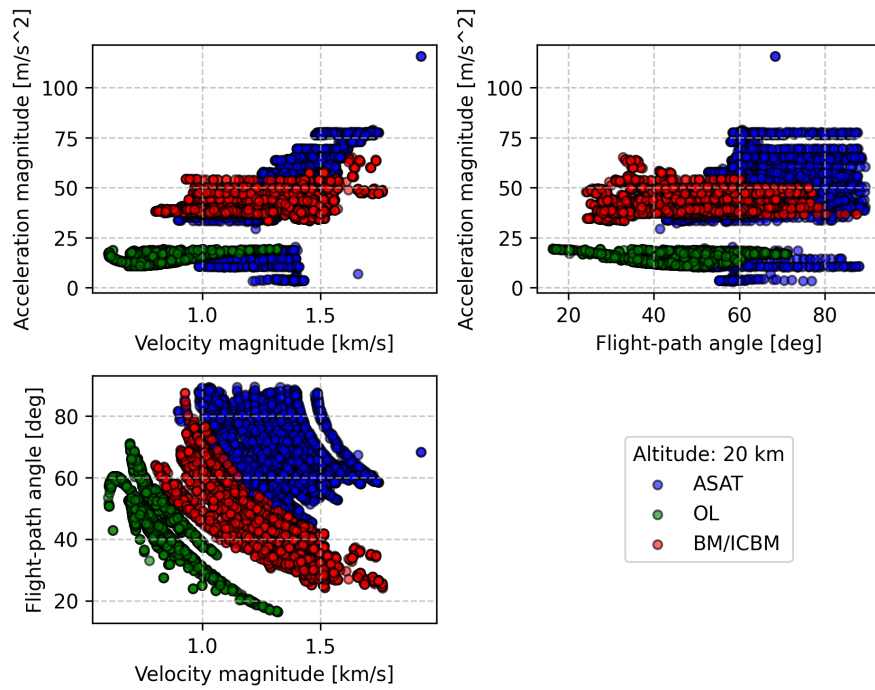


Figure B.2: Three-dimensional feature space projections at a classification altitude of 20 km.

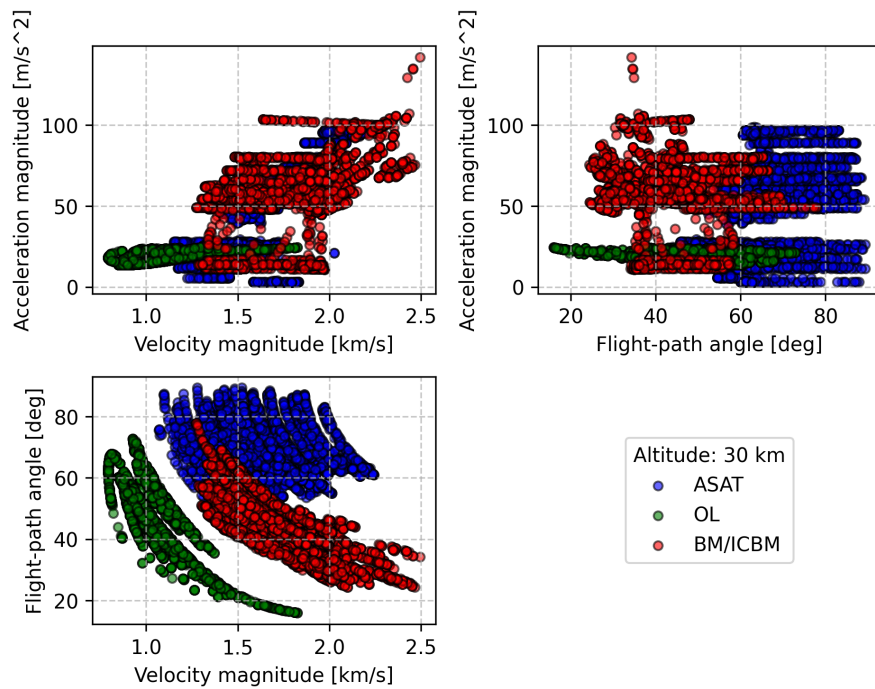


Figure B.3: Three-dimensional feature space projections at a classification altitude of 30 km.

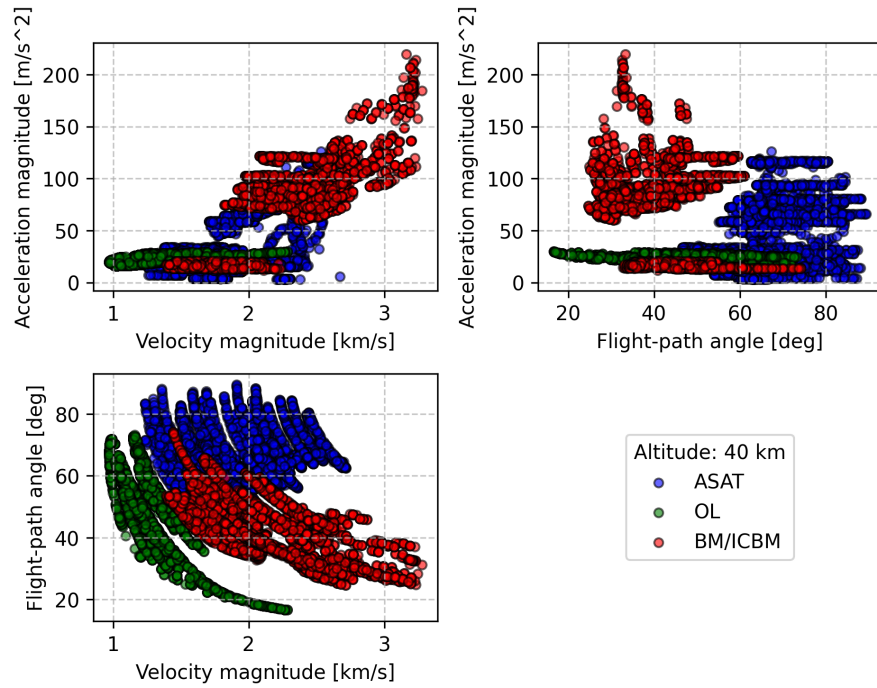


Figure B.4: Three-dimensional feature space projections at a classification altitude of 40 km.

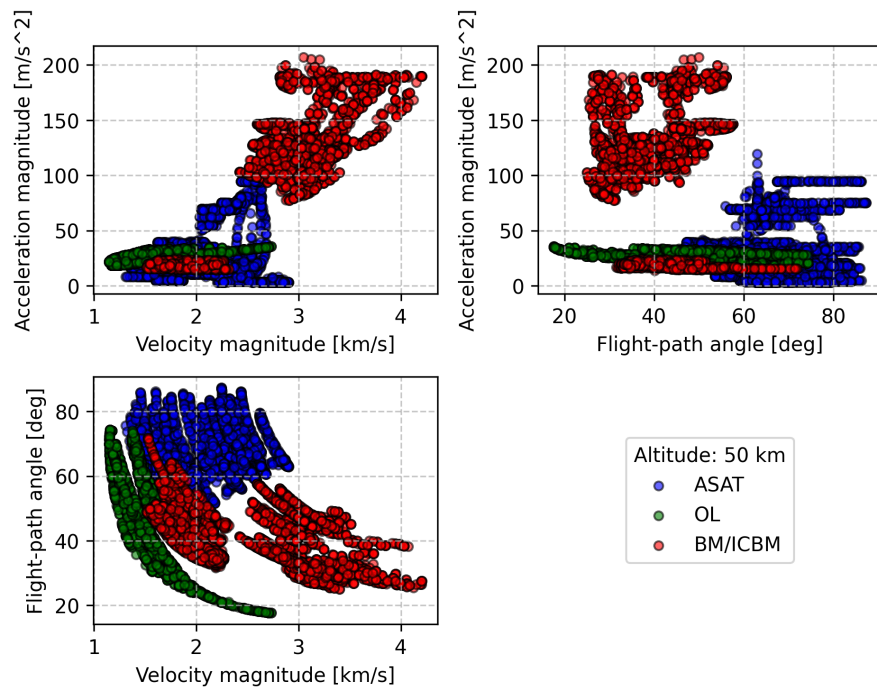


Figure B.5: Three-dimensional feature space projections at a classification altitude of 50 km.

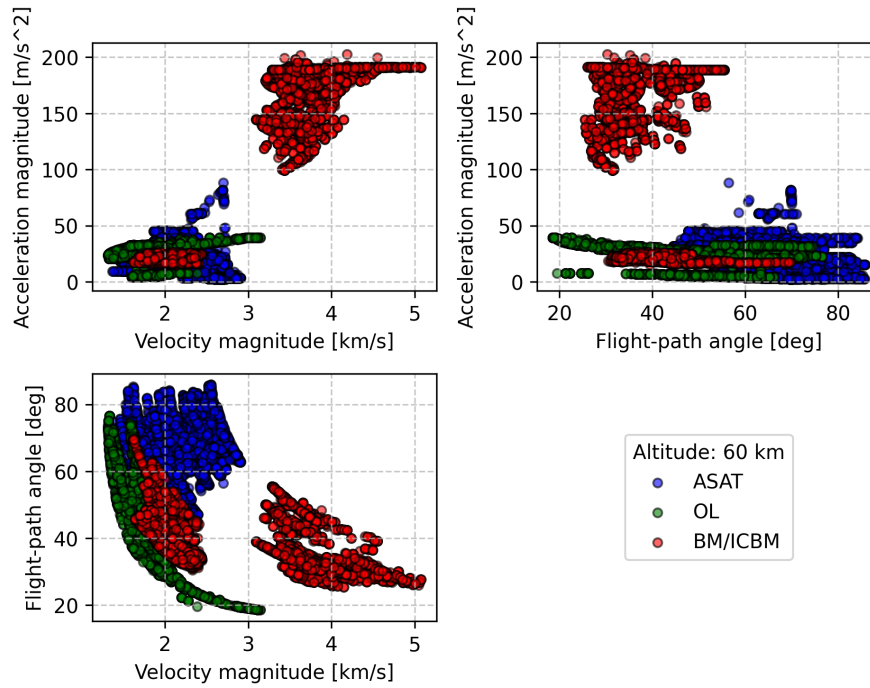


Figure B.6: Three-dimensional feature space projections at a classification altitude of 60 km.

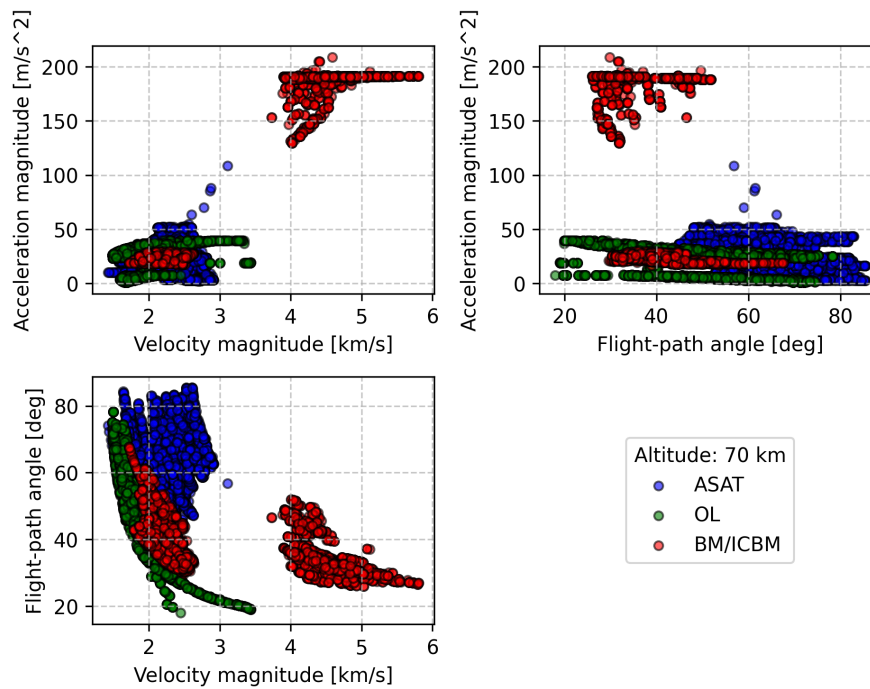


Figure B.7: Three-dimensional feature space projections at a classification altitude of 70 km.

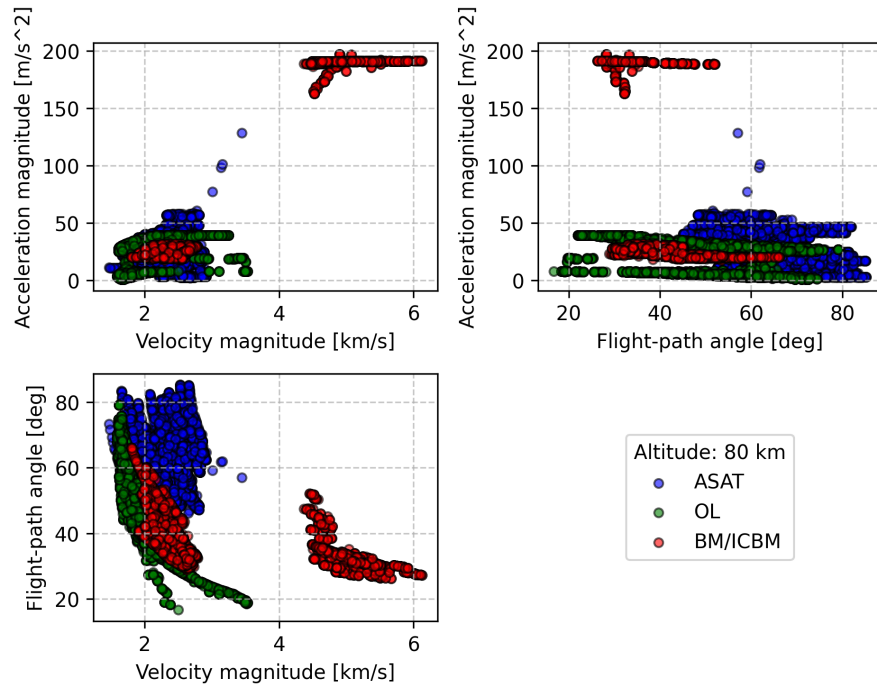


Figure B.8: Three-dimensional feature space projections at a classification altitude of 80 km.

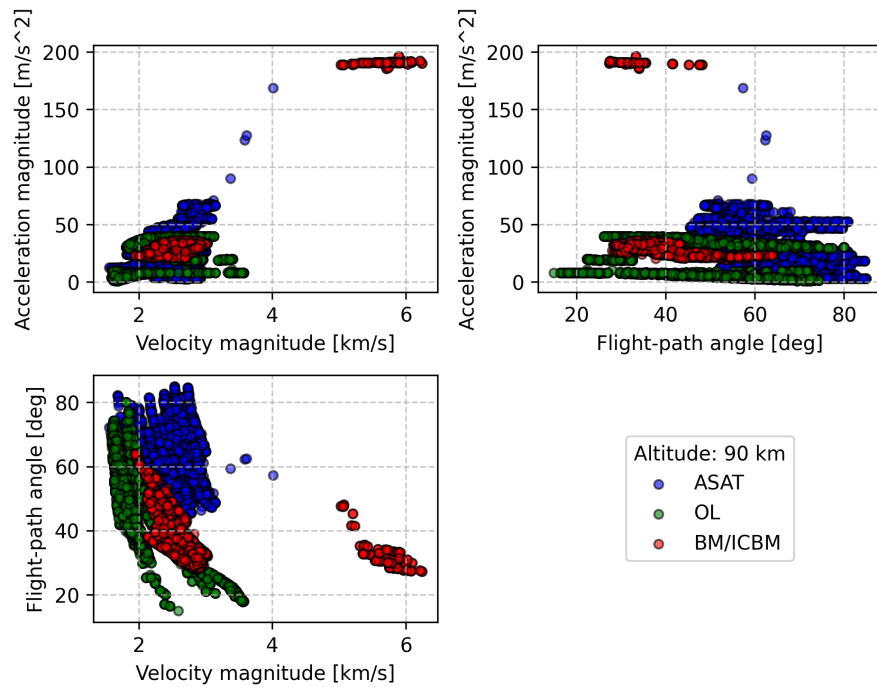


Figure B.9: Three-dimensional feature space projections at a classification altitude of 90 km.

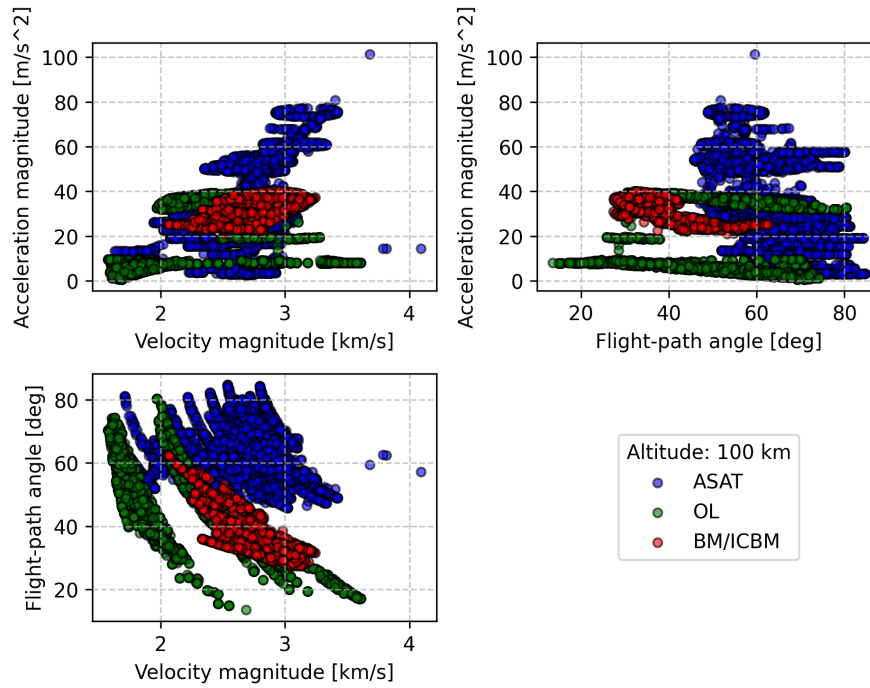


Figure B.10: Three-dimensional feature space projections at a classification altitude of 100 km.

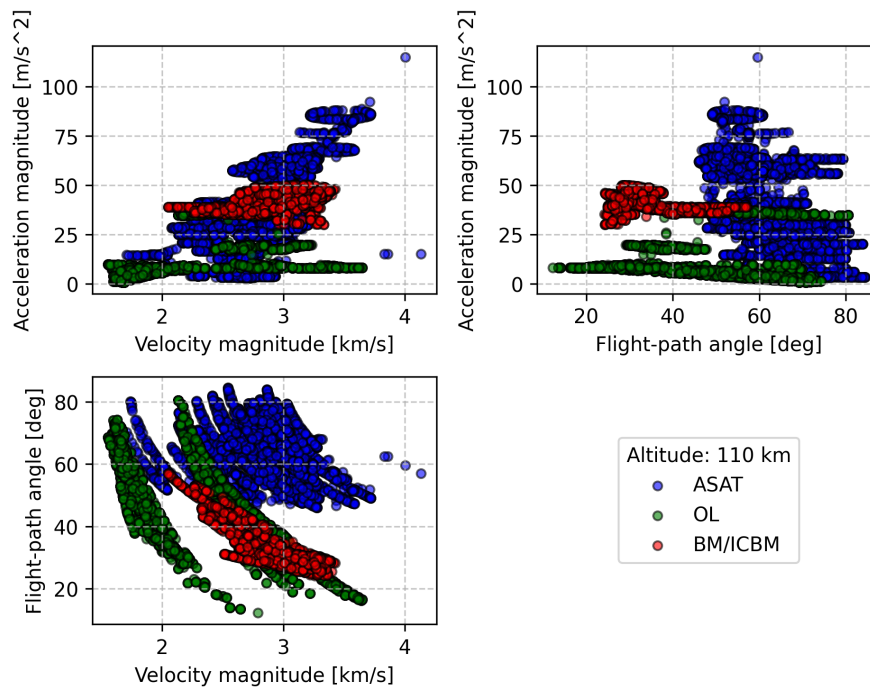


Figure B.11: Three-dimensional feature space projections at a classification altitude of 110 km.

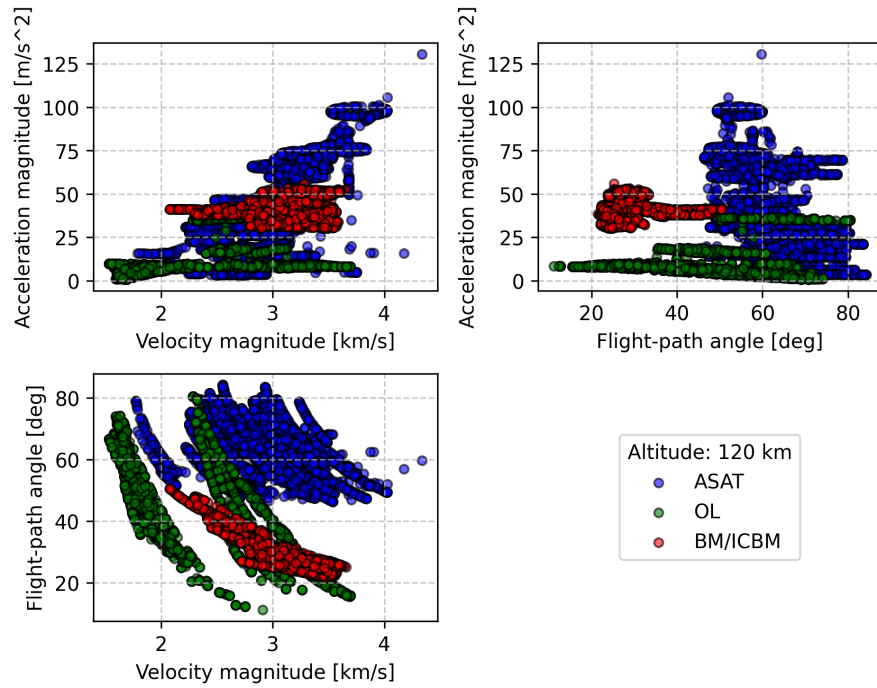


Figure B.12: Three-dimensional feature space projections at a classification altitude of 120 km.

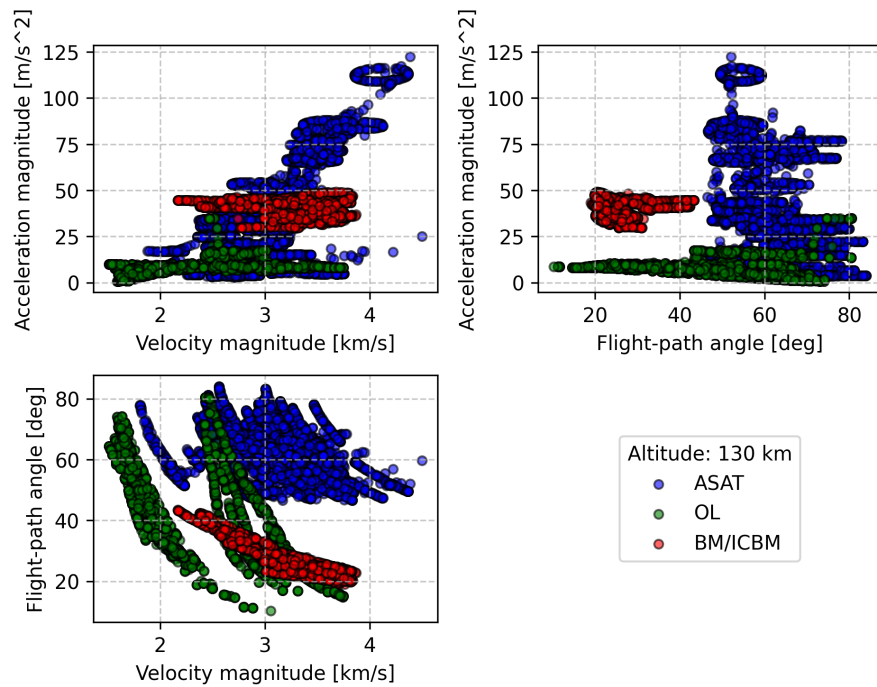


Figure B.13: Three-dimensional feature space projections at a classification altitude of 130 km.

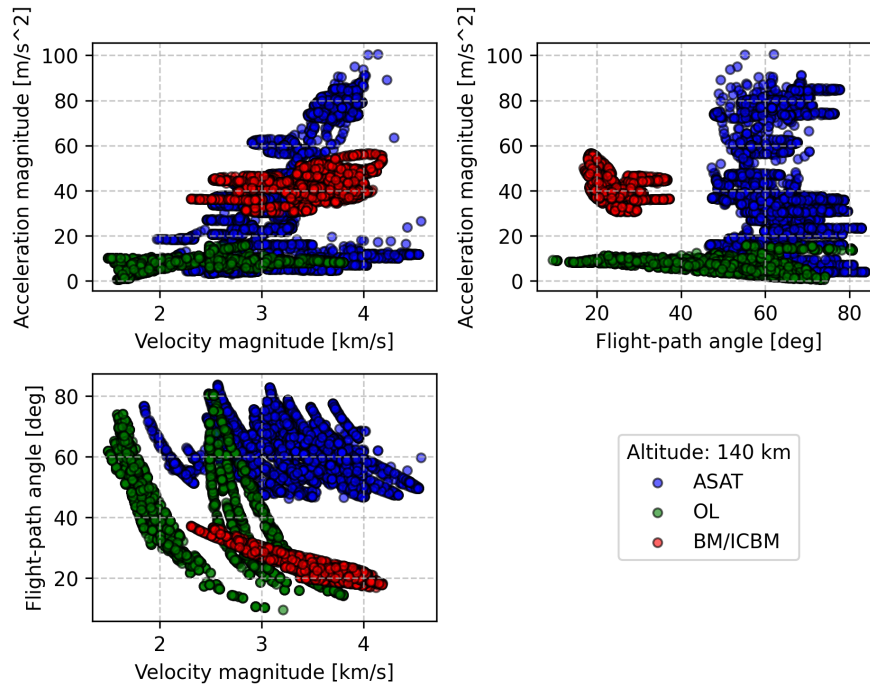


Figure B.14: Three-dimensional feature space projections at a classification altitude of 140 km.

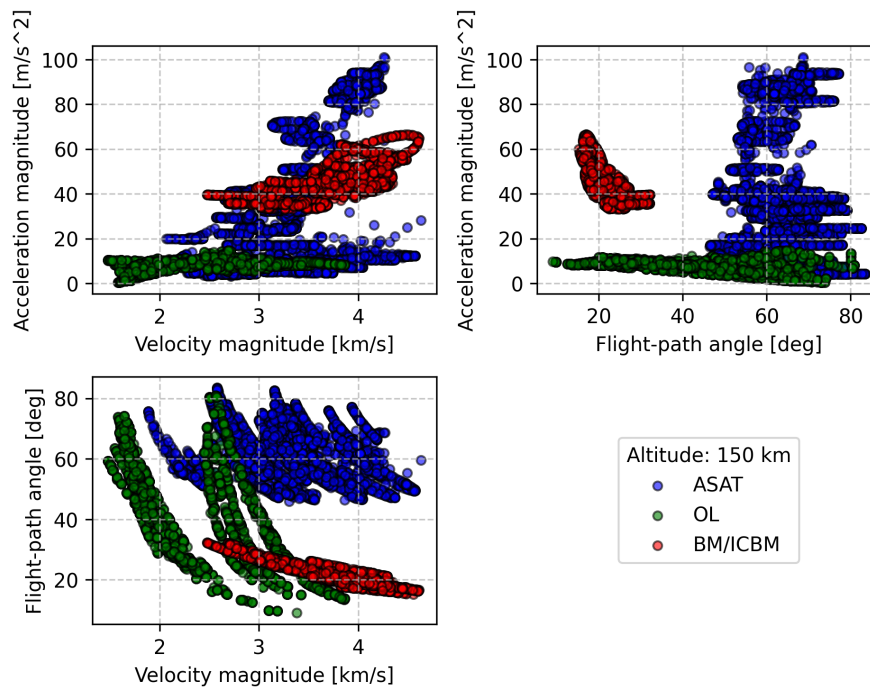


Figure B.15: Three-dimensional feature space projections at a classification altitude of 150 km.

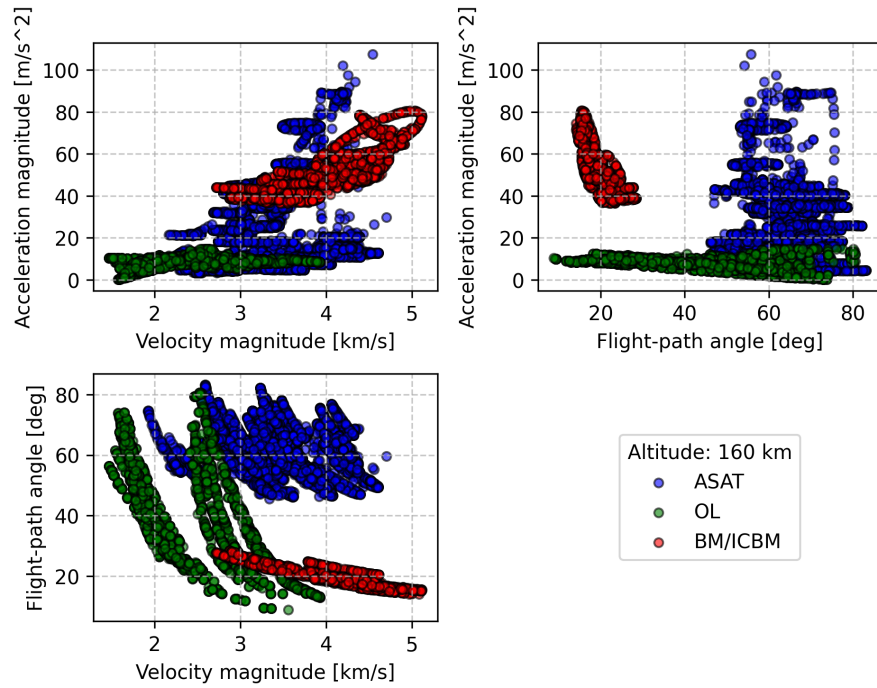


Figure B.16: Three-dimensional feature space projections at a classification altitude of 160 km.

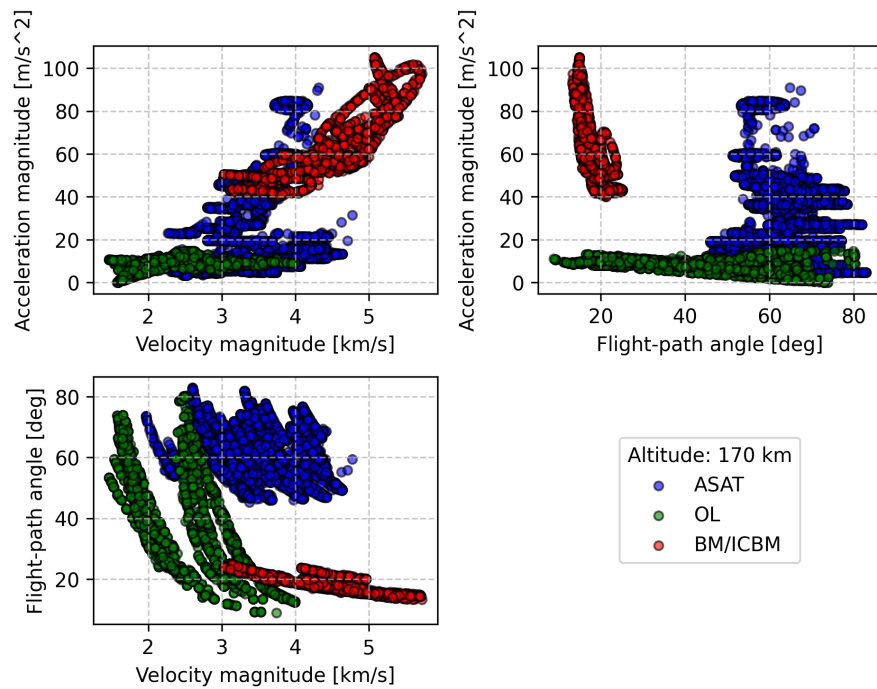


Figure B.17: Three-dimensional feature space projections at a classification altitude of 170 km.

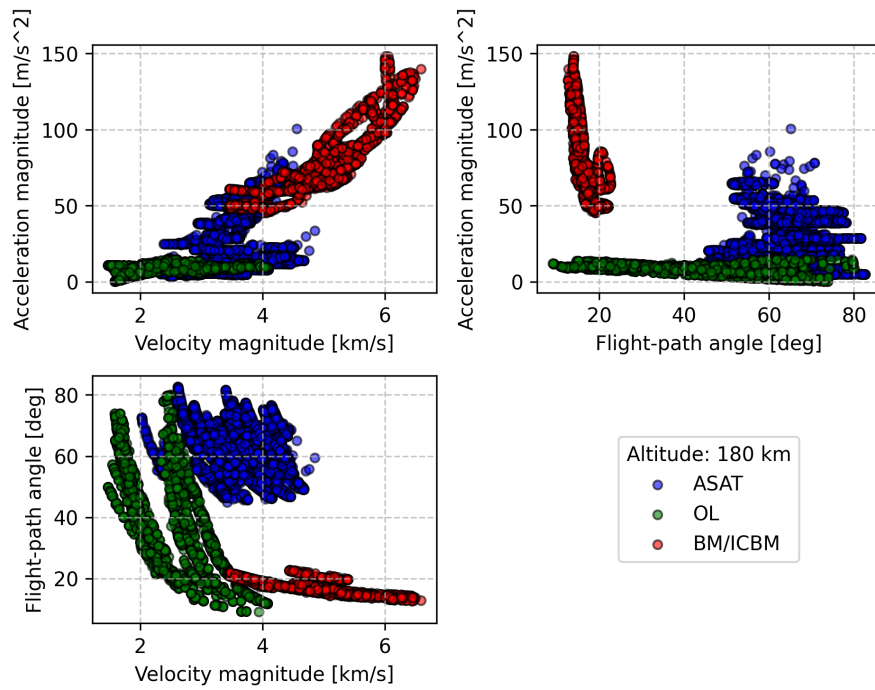


Figure B.18: Three-dimensional feature space projections at a classification altitude of 180 km.

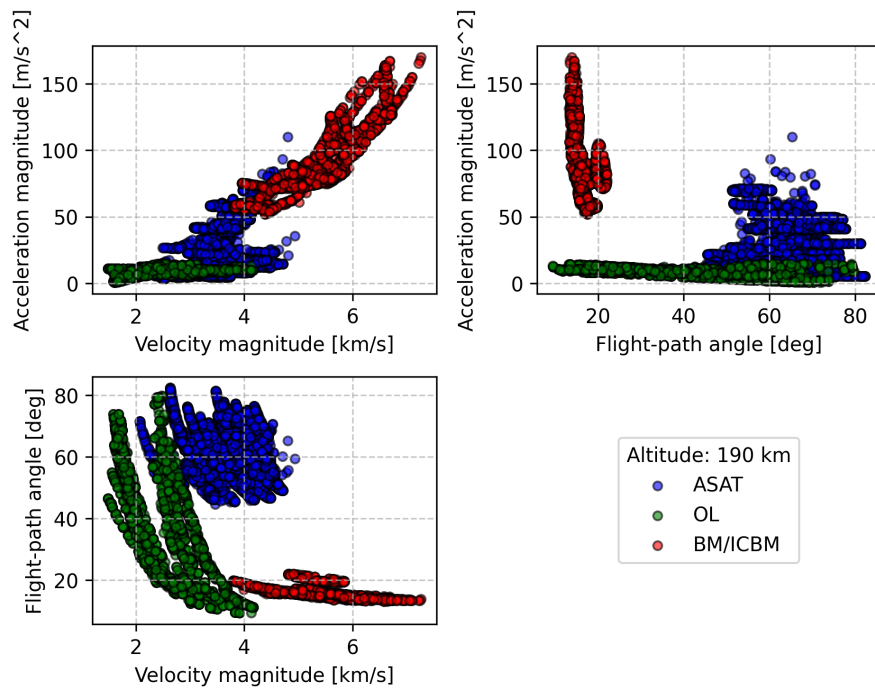


Figure B.19: Three-dimensional feature space projections at a classification altitude of 190 km.

C | Appendix C: K-Nearest Neighbours (KNN) algorithm

This appendix briefly introduces the fundamental principles of the methodology used to classify the trajectory types, namely the K-Nearest Neighbours (KNN) algorithm [59].

The K-Nearest Neighbour (KNN) algorithm is a non-parametric, supervised machine learning method widely used for both classification and regression tasks. The algorithm does not construct a generalized model during a training phase. Instead, it stores the entire labelled training dataset and performs all computations only when a new query instance requires classification. This characteristic makes KNN particularly suitable for applications in which the underlying data distribution is unknown or difficult to describe parametrically.

The core principle of KNN is based on similarity. It assumes that data points belonging to the same class tend to be located in close proximity within the feature space. When a new unlabelled sample is introduced, the algorithm performs the following operational steps:

1. Storage: The algorithm retains the complete set of labelled training data.
2. Distance calculation: For the new input, the distance to every point in the training set is computed to quantify similarity.
3. Neighbour identification: The algorithm identifies the K training samples that are closest to the new input (the nearest neighbours).
4. Classification decision: The new input is assigned to the class most frequently represented among its K nearest neighbours (majority voting).

To determine the nearest neighbours, KNN relies on a distance metric. For continuous data, the Euclidean distance is the standard metric used to calculate the spatial separation between the query instance and the training samples. Given a training dataset $D = \{(x_1^{(1)}, x_2^{(1)}, x_3^{(1)}, \dots, x_n^{(1)}), \dots, (x_1^{(m)}, x_2^{(m)}, x_3^{(m)}, \dots, x_n^{(m)})\}$

and a new unlabelled query point $\mathbf{y}$, the Euclidean distance is computed as

$$d(\mathbf{x}^{(j)}, \mathbf{y}) = \sqrt{\sum_{i=1}^n (x_i^{(j)} - y_i)^2}$$

where $\mathbf{x}^{(j)}$ denotes a generic sample from the dataset and $\mathbf{y}$ represents the unlabelled point. To ensure that features with large magnitudes do not dominate the distance calculation, it is essential to perform normalisation on the parameters before applying the algorithm.

The performance of the algorithm is highly sensitive to the selection of the parameter K , which defines the number of neighbours considered in the classification process. For $K = 1$, the input is assigned to the class of its single nearest neighbour. While this yields a zero error rate on training data, it results in over-fitting and makes the classifier susceptible to noise and outliers. Increasing the value of K reduces the effect of noise and results in smoother decision boundaries. However, excessively large values may introduce bias by including neighbours from distinct classes that are structurally far from the query point.

The optimal value of K is typically determined empirically through cross-validation techniques, in order to balance precision and generalization.

Figure C.1 illustrates a simple example of the KNN classification process. The green circle represents the sample to be classified, while the blue squares and red triangles correspond to samples from two different classes in the training set. For $K = 3$, the three nearest neighbours (illustrated by the black circle with the solid line) will decide which class the analysed sample will be assigned. For $K = 5$, the five nearest neighbours will instead be considered (illustrated by the black circle with the dotted line).

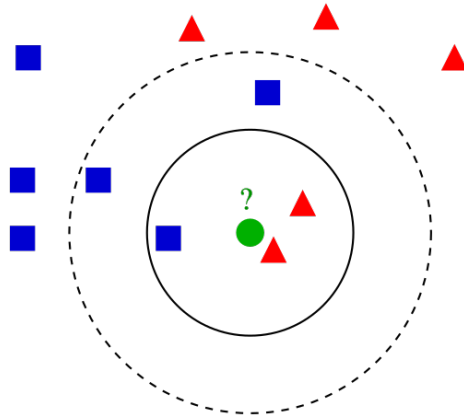


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List of Symbols

Variable	Description	SI unit
$\mathbf{a}$	acceleration vector	m/s^2
A	cross-sectional area	m^2
A_e	nozzle exit area	m^2
ASAT	Anti-Satellite	—
Az	azimuth	rad
BM	Ballistic Missile	—
C_d	drag coefficient	—
c	ballistic coefficient	kg/m^2
ECI	Earth-Centred Inertial	—
ECEF	Earth-Centred Earth-Fixed	—
ENU	East-North-Up	—
FOV	Field Of View	—
g	Earth's acceleration of gravity	m/s^2
GEO	Geostationary Earth Orbit	—
GTO	Geosynchronous Transfer Orbit	—
h	altitude	m
HEO	Highly Elliptical Orbit	—
ICBM	Intercontinental Ballistic Missile	—
IRBM	Intermediate-Range Ballistic Missile	—
I_{sp}	specific impulse	s
K	number of neighbours	—
KKV	Kinetic Kill Vehicle	—
KNN	K-Nearest Neighbours	—
LEO	Low Earth Orbit	—

Variable	Description	SI unit
LOS	Line-of-Sight	—
LTI	Trans-Lunar Injection	—
m	mass	kg
M	Mach number	—
MIRV	Multiple Independently Targetable Reentry Vehicle	—
m_d	dry mass	kg
m_p	propellant mass	kg
m_s	structural mass	kg
MRBM	Medium-Range Ballistic Missile	—
$\dot{m}$	mass flow rate	kg/s
n_b	number of boosters	—
n_e	number of engines	—
OL	Orbital Launch	—
p	atmospheric pressure	Pa
PEG	Power Explicit Guidance	—
q	normalised burn rate	s^{-1}
RNC	Radial-Normal-Circumferential	—
RV	Re-entry Vehicle	—
$\mathbf{r}$	position vector	m
$\mathbf{s}$	satellite position vector	m
SDA	Space Domain Awareness	—
SLBM	Submarine-Launched Ballistic Missile	—
SRBM	Short-Range Ballistic Missile	—
SSO	Sun-Synchronous Orbit	—
T	thrust	N
t	time	s
TEL	Transporter Erector Launcher	—
U	exhaust velocity	m/s
$\hat{\mathbf{u}}$	thrust direction unit vector	—
$\mathbf{v}$	velocity vector	m/s

Variable	Description	SI unit
α	half-cone angle	<i>rad</i>
δ	elevation	<i>rad</i>
γ	flight-path angle	<i>rad</i>
λ	longitude	<i>rad</i>
μ	Earth's gravitational parameter	m^3/s^2
ω	angular velocity	<i>rad/s</i>
ϕ	latitude	<i>rad</i>
σ	angular measurement noise	μrad
ρ	air density	kg/m^3
τ	characteristic time	<i>s</i>
θ	pitch angle	<i>rad</i>
χ	kick angle	<i>rad</i>

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